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On January 14th, the President announced a Vision for Space Exploration. In his address, the President presented a vision that is bold and forward-thinking, yet practical and responsible – one that explores answers to longstanding questions of importance to science and society and will develop revolutionary technologies and capabilities for the future, while maintaining good stewardship of taxpayer dollars.

The material contained herein is intended as one step in developing a broader understanding as to what is required for human space exploration beyond low-Earth orbit. A multi-center NASA team led by the Lyndon B. Johnson Center during the spring and summer of 2004 conducted analyses contained in this report. Included are analyses of requirements definition, exploration architectures, system development, technology roadmaps, and risk assessments for advancing the Vision for Space Exploration. The analysis contained herein is intended to provide an understanding as to what is required for human space exploration beyond low-Earth orbit. In addition, these analyses help identify system "drivers", or significant sources of cost, performance, risk, and schedule variation along with areas needing technology development. These analyses were conducted as part of an integrated analysis plan and are merely a snapshot of analysis to date. In such, any recommendations, results, and conclusions gained from this report are not conclusive in themselves and must be coordinated with other analyses being performed.

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ACRONYMS AND ABBREVIATIONS

A

ATCS	Active Thermal Control System

C

C&T	Communications and Tracking
CAD	Computer-Aided Design
CES	Crew Escape System
CEV	Crew Exploration Vehicle
CM	Crew/Command Module
CONUS	Continental United States
CSM	Command and Service Module

D

DSN	Deep Space Network

E

ECLSS	Environmental Control and Life Support System
EDL	Entry, Descent, and Landing
EDS	Earth Departure Stage
EELV	Evolved Expendable Launch Vehicle
EOD	Earth Orbit Departure
EOR	Earth Orbit Rendezvous
ERO	Earth Rendezvous Orbit
ESMD	Exploration Systems Mission Directorate
EVA	Extra-Vehicular Activity

F

FLO	First Lunar Outpost
FOM	Figure of Merit

G

GN&C	Guidance, Navigation, and Control

H

HEO	High Earth Orbit
HRLV	Human-Rated Launch Vehicle
HQ	Headquarters

I

IMLEO	Initial Mass in Low Earth Orbit
IOC	Initial Operational Capability
ISRU	In-Situ Resource Utilization
ISS	Internation Space Station
IVHM	Integrated Vehicle Health Management

J

JSC	Johnson Space Center

L

L1	Cislunar Libration Point L1
LDRM	Lunar Design Reference Mission
LEO	Low Earth Orbit
LIDS	Low Impact Docking System
LLO	Low Lunar Orbit
LLV	Lunar Landing Vehicle
LM	Lunar Module
LOA	Lunar Orbit Arrival
LOC	Loss of Crew
LOD	Lunar Orbit Departure
LOI	Lunar Orbit Insertion
LOM	Loss of Mission
LOR	Lunar Orbit Rendezvous
LPA	Libration Point Arrival
LPO	Lunar Parking Orbit
LPR	Libration Point Rendezvous
LSR	Lunar Surface Rendezvous

M

MCC	Mid-Course Correction
MET	Mission Elapsed Time
MLI	Multi-Layer Insulation
MMOD	Micrometeoroid and Orbital Debris
mt	Metric Ton (1,000 kg)
MTV	Mars Transit Vehicle

N

NASA	National Aeronautics and Space Administration

O

OMS	Orbital Maneuvering System
OSP	Orbital Space Plane

P

PDI	Powered Descent Initiation
PEM	Proton Exchange Membrane
PTCS	Passive Thermal Control System
PVA	Photovoltaic Array

R	
RCS	Reaction Control System
RFT	Requirements Formulation Task
S	
S&MA	Safety and Mission Assurance
SM	Service Module
SPE	Solar Particle Event
SR&QA	Safety, Reliability, and Quality Assurance
T	
t	Metric Ton (1,000 kg)
TBD	To Be Determined
TDRSS	Tracking and Data Relay Satellite System
TPS	Thermal Protection System
TRL	Technology Readiness Level
TRM	Trade Reference Mission

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1.0 Introduction

The Lunar Design Reference Mission 2 (LDRM-2) study was initiated by NASA Code T, later renamed the Exploration Systems Mission Directorate, in April of 2004 to perform a series of focused lunar mission trade studies intended to provide a better understanding of the relative benefits of differing mission approaches, as well as to determine mission sensitivities to key system design parameters.

The intent of the planned lunar missions is to support a wide range of scientific investigations, technology and systems development, and integrated testing to reduce the risks of future human exploration of Mars. Three LDRM studies were originally outlined to bracket a range of potential lunar mission scenarios and associated flight element functionality. LDRM-1 consists of a seven-day surface stay in the equatorial region of the moon. LDRM-2 is also based on a seven-day lunar surface stay, but includes global lunar access with the capability to initiate an Earth return at any time. LDRM-3 provides the capability for a long-duration lunar surface stay in the range of thirty to ninety days with multiple missions to a single polar landing site outfitted with additional surface elements.

The LDRM-2 exploration objectives and requirements were selected as the starting point for the lunar architecture study. The Phase 1 deliverables from the LDRM-2 study document the results of the architecture analyses associated with short duration lunar missions with global access capability. The LDRM-2 study was subsequently expanded with a Phase 2 effort focused on the LDRM-3 exploration objectives. The LDRM-2 Phase 2 deliverables document the results of the architecture analyses associated with long duration lunar missions with a restricted range of surface access.

The results of the LDRM studies will support the development of Level 1 requirements for the Crew Exploration Vehicle (CEV) by quantifying the sensitivities of flight elements to key system design parameters and subsystem technologies in the context of an end-to-end lunar mission. The definition and sizing of a complete lunar mission also provides valuable insight into the launch vehicle characteristics and infrastructure that are needed to support the delivery of flight elements to low Earth orbit (LEO).

2.0 Study Scope

2.1 LDRM-2 Background

There are three basic architectures for executing a human lunar exploration mission - direct return (also referred to as lunar surface rendezvous or LSR), libration point rendezvous (LPR) and lunar orbit rendezvous (LOR) - defined by the method that is employed to return the crew to the Earth after the conclusion of lunar surface operations. Due to orbital mechanics considerations, each of these architectures offers distinct advantages and disadvantages with respect to a given set of mission objectives and requirements. The purpose of the LDRM-2 task is to provide focused "down-and-in" assessments of specific lunar exploration mission designs. The LDRM-2 study results are intended to complement the data from parallel NASA exploration studies. Some of these studies are focused on the launch and lunar surface system segments required for a lunar exploration mission. Others are targeted to a broad, higher-level assessment of lunar architecture alternatives including advanced subsystem technologies.

The LDRM-2 study evolved into two separate phases, each focused on a specific lunar exploration architecture. The Phase 1 mission leverages the cislunar Earth-moon libration point known as "L1" as an orbital staging point to enable global lunar access with anytime return capability to Earth. The Phase 2 mission is based on a variant of the lunar orbit rendezvous approach employed successfully during the Apollo Program. Unlike Apollo, however, the Phase 2 mission is targeted to long duration, near-polar lunar surface missions. The Phase 1 and Phase 2 results are documented in separate volumes of the LDRM-2 Final Report.

Element mass is a primary driver for the Earth-to-orbit launch vehicles and is often also used as the basis for cost estimation. Therefore, the flight element mass estimates developed using the Envision parametric sizing tool are key products of the LDRM-2 study. Element sizing is based upon a top-level nominal mission timeline, functional requirements, critical spacecraft dimensions, mission environments and subsystem component and propellant selections. A common set of mission environments and subsystem technology options, which were identified early in the execution of the LDRM-2 Phase 1 task, were applied to the sizing for both the Phase 1 and Phase 2 missions. The associated technology and environment reports provided in Section 20.0 were also submitted to the NASA Headquarters Exploration Systems Mission Directorate to support the development of an overall exploration technology development and testing plan.

Independent studies of launch vehicle capabilities and lunar surface infrastructure are being pursued in parallel to the LDRM-2 task. Initially the flight elements designed in the LDRM-2 study may not fit within the payload mass and volume envelopes deemed feasible for the next generation of launch vehicles. Similarly, the cargo delivery capability of the LDRM-2 flight elements has not yet been linked with the infrastructure requirements currently being defined for long duration lunar surface missions. In subsequent design cycles, however, the requirements and constraints associated with the ground, flight and lunar surface segments will be blended to establish comprehensive and integrated lunar mission architectures.

2.2 Phase 1 Mission Definition

The LDRM-2 Phase 1 mission leverages the cislunar Earth-moon libration point known as "L1" as an orbital staging point to enable global lunar access with anytime return capability to Earth. As shown in Figure 2.2-1, L1 is one of five points of balance between the gravitational fields of the Earth and Moon. The L1 rendezvous approach takes advantage of the fixed orbital relationships of the Earth, Moon and the L1 libration point to provide global lunar access with the ability to initiate a return to Earth from the lunar surface via the L1 point at any time. The same orbital mechanics considerations facilitate element phasing and rendezvous at the libration points, making it feasible and practical to pre-deploy a lander or other assets to the L1 libration point. Pre-deployment of flight assets in a multi-launch Earth-to-orbit strategy minimizes the required number of unique element mating interfaces and also reduces the duration of flight element exposure to low Earth orbit debris.

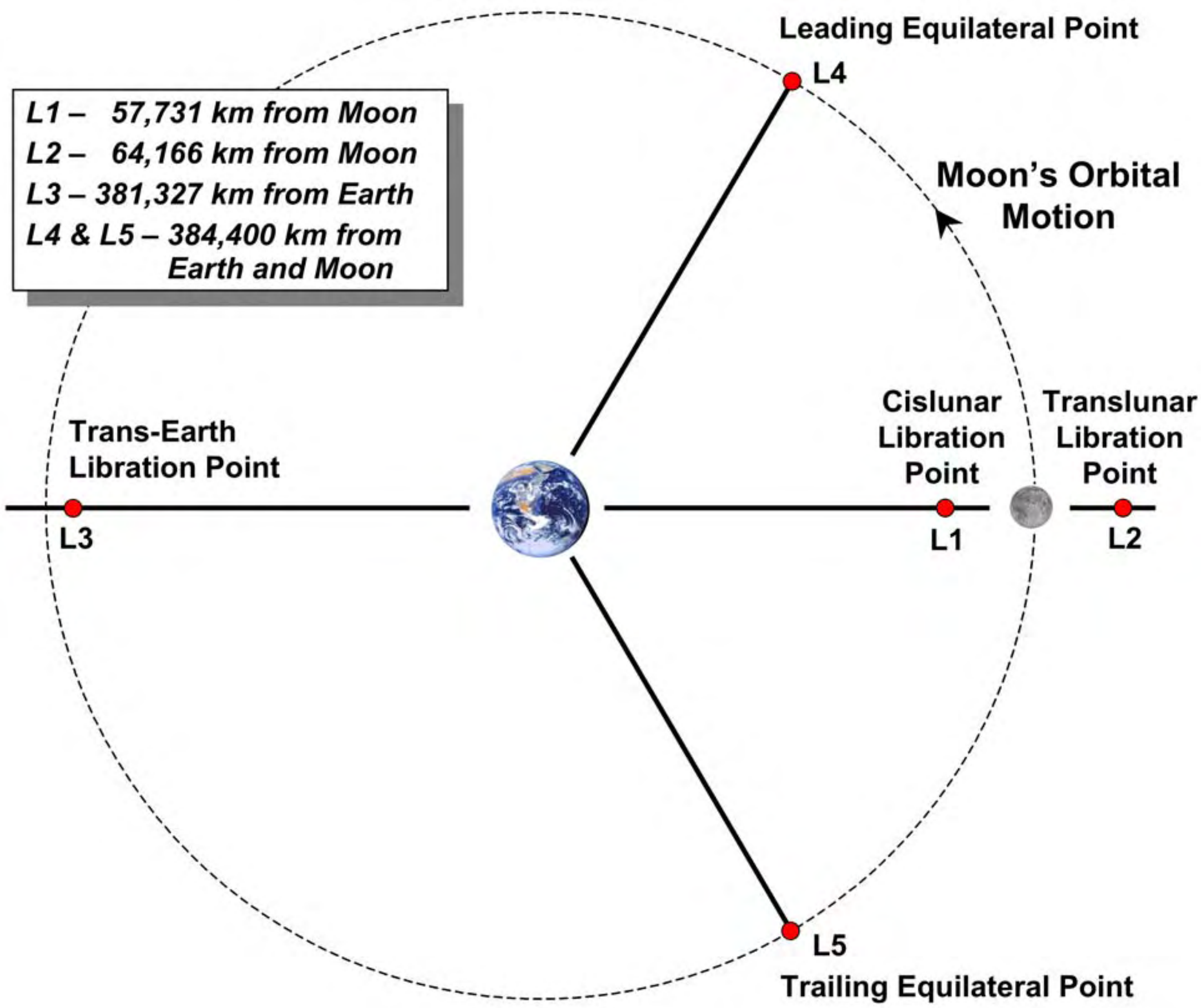

Figure 2.2-1: Earth-Moon Libration Points

Libration points other than L1 have been considered for human lunar exploration missions. But L1 offers the best overall combination of performance (ΔV) and transit time. A spacecraft parked at an Earth-Moon libration point is subject to small, destabilizing gravitational influences from other bodies, most notably solar perturbations. Elements loitering at L1 must perform occasional, small station-keeping maneuvers to maintain orbital position.

2.3 Phase 1 Study Approach

The LDRM-2 Phase 1 study was defined in the context of a "trade reference mission" or TRM which later became known as the "L1 TRM" to distinguish it from its Phase 2 polar LOR counterpart. As shown in Figure 2.3-1, the primary inputs to the Phase 1 study are the customer requirements (task statement assumptions), design environments and selected subsystem technologies. In order to provide a more complete understanding of the L1 TRM sensitivities, the reference mission is supplemented with a number of incremental architectural and parametric variations. With the exception of the Phase 1 LOR case, only one primary design variable is modified for each of the mission variants.

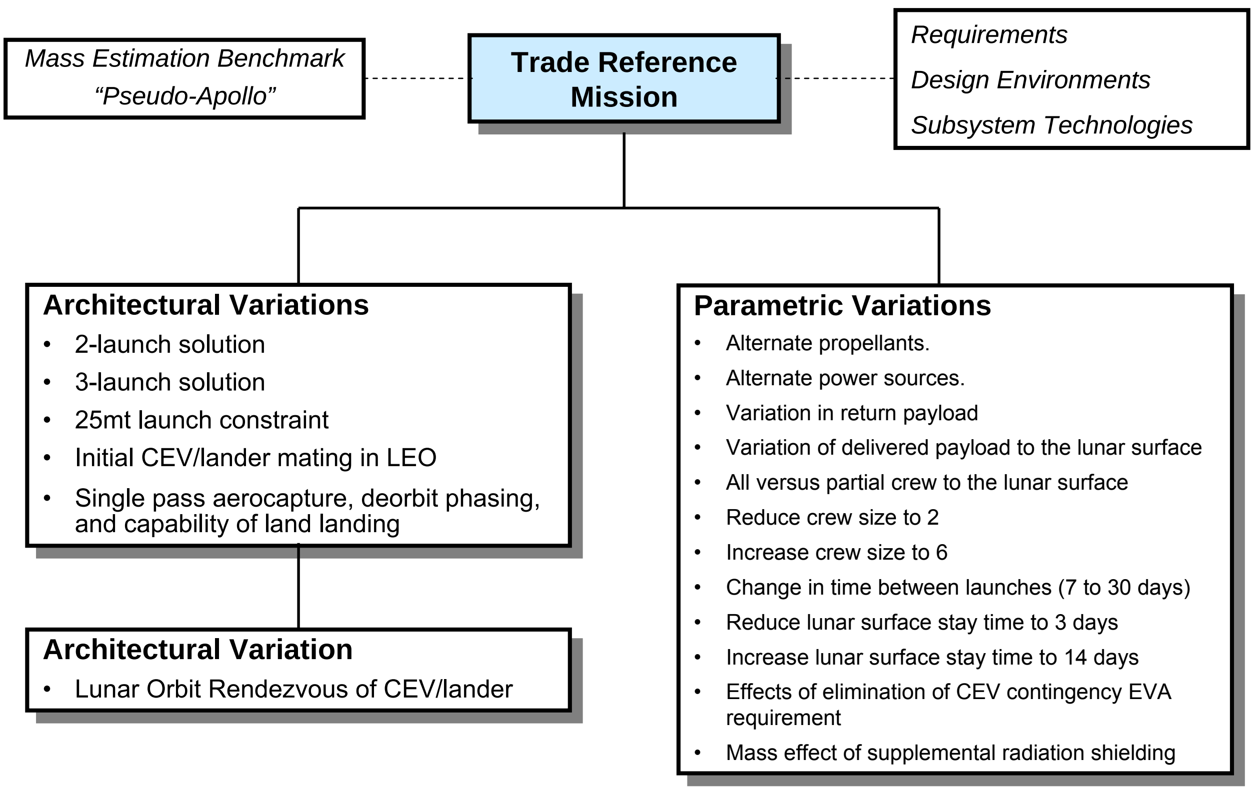

Figure 2.3-1: LDRM-2 Phase 1 Study Approach

The Phase 1 LOR architectural variant represents a significant departure from the L1 TRM in terms of orbital mechanics. While it is relatively straightforward to depart from LEO, rendezvous with a pre-deployed element at L1 and access a desired landing site on the lunar surface, the same operation is potentially much more complicated for a lunar orbit rendezvous architecture. The Phase 1 LOR approach uses an optimized parking orbit that is aligned with the nominal descent and ascent points on the lunar surface to minimize plane change requirements. If the LOR mission is not executed on schedule then the orientation of the parking orbit of the pre-deployed element may not be optimal for either the Earth orbit departure and lunar orbit rendezvous, or the descent to the lunar landing site. Therefore, the Phase 1 LOR variant employs a tandem Earth orbit departure in which the CEV, lander and other propulsive stages rendezvous in LEO and depart for the Moon as an integrated stack. Except for the use of multiple launches to deliver the flight elements to LEO, this is very similar to the Earth orbit departure approach that was employed during the Apollo Program.

The primary objective of the Envision mass estimation software is to provide relative sizing results to assist in the identification of promising design alternatives. In order to provide confidence in the general magnitude of the mass results, a “Pseudo-Apollo” sizing benchmark was developed during the Phase 1 study. Additional information on the Envision sizing tool and the Pseudo-Apollo benchmark is provided in Section 6.0.

2.4 Figures of Merit

Figure of Merit (FOM) categories and subcategories were defined by the NASA Exploration Systems Mission Directorate to support a methodical comparison of lunar architecture options. Where possible, the LDRM-2 lunar architectures were quantitatively or qualitatively evaluated using the FOM categories defined in the LDRM-2 task statement (see section 4.3 for additional information):

- Safety and Reliability
- Effectiveness and Evolvability
- Development Risk and Schedule
- Affordability

The LDRM-2 study provides several types of numerical data to support relative comparisons of lunar mission architectures. Element and total architecture masses are traditional figures of merit used to estimate program development and recurring costs. Elements masses are also used to define launch vehicle requirements in terms of payload capacity and the required number of launches, both of which are likely to be significant cost drivers. In addition, the number and type of flight elements in combination with the launch vehicle size and launch frequency are key factors in determining ground infrastructure requirements.

Operational considerations, particularly those that affect crew safety, are also of critical importance in assessing the relative merits of lunar architectures. Critical events lists were developed for the L1 TRM and each of the architectural variants to provide a basis for evaluating safety and mission risks. Operational concepts and mission timelines were also developed to quantify the number of element interfaces and propulsive maneuvers as well as the mission durations for crew and flight elements, both total and by mission phase. Consideration was also given to functional redundancy and Earth return capability for off-nominal situations involving major systems failures. However, additional time and resources will be required to perform detailed abort assessments.

2.5 Apollo Program Background

The total ΔV required for a lunar mission from Earth departure through Earth return, including the lunar descent and ascent maneuvers, is roughly 9 to 10 km/s (29,500 to 32,800 ft/s). That total is comparable to the ΔV required for launching payloads from the surface of the Earth to low Earth orbit. In contrast, typical crewed LEO missions require only about 0.35 km/s (1,150 ft/s) of on-orbit ΔV for rendezvous and deorbit – less than 4% of the lunar mission value. As a result, the combined mass of the flight elements delivered to low Earth orbit for a chemical propulsion lunar mission is typically well in excess of 100t (220 Klb), and can easily exceed 200t (441 Klb) for a robust set of mission requirements. The Apollo lunar stack, which is often used as a basis of comparison for lunar exploration studies, had an IMLEO equivalent of roughly 138t (304 Klb). Several different lunar mission architectures were debated at length in the early years of the Apollo Program. The ability to package a complete lunar mission in a single launch of a Saturn V-class vehicle was a critical factor in favor of the lunar orbit rendezvous approach.

The Apollo Program employed single-launch LOR missions to provide access to near-equatorial lunar landing sites for periods up to 3 days. The Lunar Module (LM) transported two astronauts between low lunar orbit (LLO) and the lunar surface and supported crew activities during the surface mission. The Command Service Module (CSM) covered a range of functions throughout the mission from launch through re-entry and landing. The Service Module (SM) provided main propulsion for the lunar orbit insertion of the mated CSM/LM configuration and the lunar orbit departure of the CSM, and also provided power and life support consumables to the Command Module (CM). The primary responsibilities of the Command Module included crew accommodations and flight avionics functions including guidance, navigation and control, data management and communications. The Command Module also included the thermal protection and parachutes required to safely recover the three astronauts at Earth. The same basic functionality must be provided for any lunar exploration mission.

In the context of the LDRM-2 study, the flight element most similar to the Apollo LM is referred to simply as the "lander" while the closest analog to the Apollo CSM is referred to as the Crew Exploration Vehicle (CEV). The relative similarity of the LDRM-2 elements to the Apollo spacecraft in terms of size and function is dependent upon the selected mission approach and the allocation of functions among the flight elements. For instance, an L1 rendezvous mission places greater functional responsibility on the lander element for both propulsion and crew accommodations than does an LOR mission. A direct return mission, in contrast, essentially blends the functions of the LM and CSM (or lander and CEV) into a single spacecraft.

3.0 Executive Summary

Although the LDRM-2 Phase 1 study focuses on L1 rendezvous, the Phase 1 architectural variants provide valuable insight into the lunar orbit rendezvous and direct return alternatives for human lunar exploration. The L1 rendezvous approach offers a mixture of the characteristics of the lunar orbit rendezvous and direct return architectures. In terms of basic orbital mechanics, L1 rendezvous can be viewed as a special case of lunar orbit rendezvous in which the L1 libration point represents a very high lunar orbit with unique properties. In terms of inherent mission flexibility for lunar landing site access and Earth return opportunities, L1 rendezvous resembles direct return, but with an interim L1 return target rather than Earth. In terms of initial mass in low Earth orbit, L1 rendezvous falls between lunar orbit rendezvous at the low end and direct return at the high end. Each of the three lunar architectures is capable of supporting a robust lunar exploration program, but each also involves a unique set of strengths and weaknesses that must be considered in the context of the mission objectives and requirements.

The fundamental characteristics of the three lunar architectures are dictated by the orbital mechanics of the Earth-Moon system. The key considerations in the architecture selection process are the exploration mission objectives and the Earth-to-orbit launch strategy. Exploration objectives involving global lunar access and extended mission durations with anytime Earth return mesh well with the L1 rendezvous and direct return architectures. Constraints on launch mass will tend to favor an optimized lunar orbit rendezvous approach, as will constraints on the required range of access and/or duration of lunar surface missions. A long-term mission plan that involves rendezvous with space-based assets works well with an L1 rendezvous approach. Architecture selection criteria emphasizing mission flexibility and the ability to deliver greater cargo mass to the lunar surface during a cargo mission over lower initial mass in low Earth orbit will tend to favor direct return.

3.1 Lunar Surface Access and Earth Return Capability

The L1 rendezvous and direct return architectures inherently provide a wide range of mission flexibility in terms of landing site access and surface stay times for a long-term lunar exploration program. Both of these architectures support short duration expeditions targeting sites of scientific interest as well as longer duration missions utilizing emplaced surface assets anywhere on the lunar surface. Both architectures also inherently offer the capability to initiate a return to Earth at any time during the lunar surface stay. The direct return architecture provides anytime Earth return by eliminating the inbound rendezvous in favor of optimized lunar ascent and departure maneuvers. As a result, however, the direct return architecture incurs the mass penalty of transporting the propellant required for Earth return (ΔV ~850 m/s) to the lunar surface and back to lunar orbit. The L1 rendezvous approach takes advantage of the fixed orbital relationships of the Earth, Moon and L1 to provide the capability for anytime Earth return from the lunar surface. However, the lander ascent stage must provide the additional propulsion needed for the lunar orbit departure (ΔV~600 m/s) and L1 arrival (ΔV~250 m/s) maneuvers. In addition, the L1 rendezvous approach adds approximately 2.5 days to the one-way transit time between the Earth and the Moon.

By including substantial plane change capability on the CEV and the lander elements, the lunar orbit rendezvous architecture can provide the same mission flexibility in terms of global lunar access and variable mission duration with anytime Earth return. However, the Phase 1 mass trend lines indicate that the total mass of the lunar orbit rendezvous architecture will be roughly comparable to the mass of the L1 rendezvous approach when sized for maximum mission flexibility. The primary advantage of the lunar orbit rendezvous architecture is that it can be tailored to envelope a desired subset of mission capabilities by optimizing the characteristics of the lunar parking orbit and the plane change capabilities of the CEV and lander. Significant flight element size and mass reductions can be realized by constraining the range of landing sites, surface mission durations or off-nominal Earth return opportunities, and optimizing the flight elements for the performance variations over the range of planned missions. Similarly, the capability to initiate a return to Earth at any time during the lunar surface stay is an optional capability for the lunar orbit rendezvous architecture. In general, nominal and off-nominal Earth return opportunities for lunar orbit rendezvous missions are obtained via plane change capability, loiter capability, or a combination of the two. Anytime return to Earth from the lunar surface requires sufficient plane change capability in the CEV and lander to handle a worst-case alignment of the landing site with the CEV parking orbit as well as a worst-case alignment of the CEV parking orbit with the Earth return departure vector.

3.2 Initial Mass in Low Earth Orbit

The primary disadvantage of the L1 rendezvous and direct return architectures is a high initial mass in low Earth orbit (IMLEO) relative to a constrained and optimized lunar orbit rendezvous approach. Although the direct return architecture requires the lowest total mission ΔV, the burden of transporting the Earth return propellant to the lunar surface results in the highest architecture IMLEO. The L1 rendezvous architecture is more mass efficient than the direct return approach because its Earth return propellant is not carried to the lunar surface. However, the additional ΔV associated with L1 arrival and departure maneuvers still results in a relatively high architecture IMLEO (Phase 1 estimate of 230t). The direct return and the L1 rendezvous architectures are functional “package deals” providing maximum mission flexibility – global lunar access with extended mission durations and anytime Earth return capability – at the expense of higher architecture IMLEO and massive lander elements.

Despite the Phase 1 mission requirements for global lunar access with anytime Earth return capability, it is possible to provide a lower mass solution using the lunar orbit rendezvous architecture by taking advantage of the limited duration of the lunar surface mission. Based on the propulsive efficiencies of the Phase 1 flight elements, the lowest mass solution uses an optimized lunar parking orbit that minimizes the lunar descent and ascent plane change required for anytime Earth return over the seven-day surface stay. Using this approach, a lunar orbit rendezvous mission satisfying the Phase 1 mission requirements resulted in an IMLEO estimate of 199t in comparison to the L1 rendezvous value of 230t – a reduction of 31t (13.5%).

The LDRM-2 Phase 1 study results demonstrate that the goal of optimizing architecture IMLEO is not necessarily achieved by minimizing total mission ΔV. The distribution of ΔV among the flight elements is also of critical importance because of the flow down of mass from flight element to flight element via the rocket equation. In general, the lowest mass solution will be achieved by biasing the ΔV distribution from the lander ascent and descent stages towards the CEV and Earth Departure Stages. This is especially true when the EDS employs high-performance cryogenic propellants.

3.3 Definition of Flight Elements

The propellant selection, number of flight elements and allocation of major maneuvers among flight elements are key design variables that can have major mass and complexity impacts within a given lunar architecture.

Early in the Phase 1 study the decision was made to baseline the use of pressure-fed liquid methane and oxygen for the CEV and lander and a higher performance combination of liquid hydrogen and oxygen for the Earth Departure Stages. The choice of pressure-fed liquid methane and oxygen for the CEV and lander is supported by its simplicity, reliability, moderately high Isp and energy-density, storability in the space environment and compatibility with potential in-situ resource development, particularly at Mars. In addition, lander operational considerations such as landing stability, ascent stage height for crew egress/ingress and launch packaging constraints, favor the more compact lander design offered by an oxygen/methane propulsion system. It is possible to reduce the mass of the lander descent stage through the use of liquid hydrogen and oxygen propellants, although the magnitude of the benefit depends on the ascent stage mass (payload) and the magnitude of the descent stage ΔV. Reductions in IMLEO of 11% or more are possible for the direct return architecture, for example, when the lunar orbit arrival maneuver is allocated to the lander descent stage. However, the lander mass benefit must be considered in light of the greater complexity of the pump-fed engines and the large increase in tank volume associated with the storage of liquid hydrogen. In the case of the Earth Departure Stage and the Kick Stage used in the L1 trade reference mission (TRM), the higher performance of the pump-fed liquid hydrogen and oxygen propulsion system was the primary consideration.

Variations in the allocation of major propulsive maneuvers (ΔV) and the propellant selection (Isp) among flight elements will significantly affect total architecture mass and mass distribution. In addition, the maneuver allocation affects element functional requirements including operational environments, design lifetime and number of planned engine restarts. The optimal number of propulsive stages is driven by a range of considerations including total mission ΔV, launch vehicle payload constraints, number of launches, number of dynamic element interfaces and complexity of on-orbit assembly. A lunar mission naturally splits into three to five propulsive stages depending upon the selected mission architecture and associated launch constraints – a CEV, one or two lander stages and one or two Earth Departure Stages. Given the total lunar mission ΔV of roughly 9,000 to 10,000 m/s, the mass benefits of additional propulsive stages are generally modest because the staging efficiency is partially offset by an increase in dry mass. In the case of the L1 TRM, however, a sixth "Kick Stage" using liquid hydrogen and oxygen propellants was added to the lander and assigned the outbound L1 arrival, L1 departure and lunar orbit arrival maneuvers totaling approximately 1,800 m/s. The transfer of approximately one-fifth of the total mission ΔV to the Kick Stage reduced the mass of the lander descent stage by 41%. The increase in the number of propulsive stages in combination with the higher Isp of the Kick Stage relative to the lander descent stage resulted in an overall IMLEO reduction of nearly 7% for the L1 TRM.

3.4 Launch Vehicle Considerations

The size and mass of the flight elements directly impact the design and cost of the launch system in terms of maximum payload mass, maximum payload volume/dimensions and number of launches required per lunar mission. Depending upon the selected mission architecture and associated mission requirements, the flight elements may not divide cleanly into an arbitrary number of launches, resulting in a more expensive launch vehicle with both a large payload fairing and a high payload mass capability. A requirement to divide flight element launches between dedicated cargo and human-rated launch vehicles adds another constraint to the design of the flight elements.

A multi-launch mission strategy using low Earth orbit assembly of flight elements can be employed to reduce the maximum required payload capacity of the launch vehicle. In contrast to the daily mission opportunities afforded by a single-launch approach, however, an Earth orbit departure opportunity for L1 or the Moon only occurs roughly every eight to twelve days from an established low Earth orbit. Therefore, a lunar mission architecture using low Earth orbit assembly of flight elements may be more sensitive to launch delays or other schedule impacts.

The simplest launch vehicle packaging approach is to split the elements functionally, and then subdivide the larger elements, as required, to meet the launch vehicle payload constraints. Using this approach, the element breakdown for the four-launch L1 trade reference mission includes a CEV, lander and two Earth Departure Stages. The Earth Departure Stages are unequally split in terms of mass to support pre-deployment of the lander element to L1. The lander EDS (94t) is the payload mass driver for the cargo launch vehicle while the lander (70t including the Kick Stage) is the likely driver for the payload fairing dimensions. At approximately 27t, the CEV is towards the upper range of payload capability for current generation expendable launch vehicles and is the design driver for the human-rated launch vehicle.

Different groupings of the L1 TRM flight elements were examined to support two-launch, three-launch and payload-limited (25t) launch scenarios. From a crew standpoint, the two-launch and three-launch approaches result in the elimination of CEV rendezvous operations with the EDS in LEO and a reduction in on-orbit mission duration by roughly four days. The grouping of the CEV and its EDS into a single human-rated vehicle launch of 57t plus a launch escape system also offers the capability for a daily CEV Earth orbit departure opportunity for a split-mission Earth orbit departure. The three-launch scenario has no material impact on the required payload capacity for the cargo launch vehicle relative to the four-launch case. In contrast, the two-launch scenario requires a cargo launch capacity of 159t, but provides only modest operational advantages relative to the three-launch case. The use of a launcher limited to a payload of 25t results in ten launches for the L1 TRM, seven for the lander mission and three for the CEV mission. The lander EDS is divided into four separate launches in order to fit on the 25t launcher, and the lander descent, ascent and kick stages must be launched separately. The three launches for the CEV mission consist of two EDS elements and the CEV. The operational issues associated with designing, launching and assembling a large number of elements for a single lunar mission are significant. In addition, the 25t launch vehicle approach increases the complexity of the mission planning associated with the disposal of spent propulsive stages. Because the majority of the Earth-to-orbit launch mass is in the form of propellant, the option of using on-orbit fueling should be considered for payload-limited launch vehicle strategies.

The four-launch strategy for the lunar orbit rendezvous approach is essentially the same as described for the L1 trade reference mission. However, the tandem Earth orbit departure employed for the LOR variant enables an equal-mass split of the Earth Departure Stages (71t each), thus significantly reducing the maximum launch vehicle payload requirement. Because the lander is operating from a low lunar orbit, its mass (33t) and dimensions are also substantially less than estimated for an L1 rendezvous mission. In contrast, the similarity in CEV propulsive requirements between the L1 TRM and the LOR variant resulted in similar CEV gross masses (27t versus 23t for the LOR variant). It is important to keep in mind that the gross masses of the flight elements can vary significantly in response to changes in key sizing inputs such as ΔV distribution, propulsive efficiency or Earth orbit departure strategy.

A four-launch strategy for the direct return approach includes a CEV, lander descent stage and two Earth Departure Stages. The most straightforward CEV design implementation for the direct return architecture combines the functionality of the lander ascent stage with the CEV propulsion and crew accommodations needed for the Earth-Moon transits. Detailed sizing estimates for the direct return architecture were deferred until after the completion of the Phase 1 and Phase 2 studies, although preliminary results indicate a gross mass in the range of 260t using the Phase 1 requirements with low Earth orbit assembly. Substantial reductions in IMLEO are likely with the use of cryogenic propellants on the lander descent stage.

3.5 Earth Orbit Departure Strategy

The Earth orbit departure (EOD) strategy can have significant impacts on the mass distribution, functionality and complexity of the flight elements within a given lunar architecture. One approach is the tandem EOD used in the Apollo missions in which all of the flight elements depart Earth orbit as an integrated stack. The split mission EOD is an alternate approach in which the flight elements depart Earth orbit as separate stacks. The split mission EOD can be further divided into convoy and pre-deployment approaches that are defined by the relative timing of the split mission EOD maneuvers. In a convoy approach the separate stacks depart Earth orbit in close succession, thus simulating a tandem EOD without the need to mate all of the elements in low Earth orbit. In a pre-deployment approach one or more flight elements are dispatched to a forward location prior to the launch of the crew.

3.5.1 Tandem Earth Orbit Departure

A tandem Earth orbit departure approach works well with any of the three basic lunar architectures, but requires low Earth orbit assembly of launch elements for a multi-launch Earth-to-orbit strategy. The tandem EOD approach enables an equal-mass split between the Earth Departure Stages which typically results in a lower maximum payload requirement for the launch vehicle, and also eliminates the need for an outbound lunar rendezvous for the LOR and direct return architectures. The drawback of the tandem EOD approach for a multi-launch strategy is that two dynamic mating interfaces will be needed on some flight elements to enable the assembly of a complete lunar stack.

3.5.2 Split Mission Earth Orbit Departure

For the four-launch case specified in the L1 TRM, a split mission EOD limits the number of dynamic mating interfaces to one per flight element. Because the CEV and lander must mate to enable the transfer of crew, the same dynamic mating interface can be replicated on each of the Earth Departure Stages in the four-launch case.

Operationally, the convoy version of the split mission EOD is effectively a tandem EOD with a deferred flight element rendezvous in the lunar vicinity. Assuming that the convoy EOD maneuvers are executed nominally, this approach should work well with any of the three basic lunar architectures. Preliminary investigations indicate that an outbound phasing and rendezvous of convoyed flight elements in lunar orbit is feasible. In contrast to the more familiar low Earth orbit case, however, the CEV chaser will phase down in altitude to rendezvous with the lander target.

In the pre-deployment version of the split mission EOD, a mission asset can be delivered to the lunar surface or parked in the lunar vicinity. Due to the unique nature of the libration points, an Earth orbit departure opportunity for L1 naturally provides a straightforward rendezvous opportunity with a mission asset parked at L1. As a result, flight elements or other assets can be pre-deployed to L1 without imposing unreasonable schedule or rendezvous constraints on a lunar mission. The same rationale explains why L1 is often used as a “waypoint” for more ambitious near-Earth exploration approaches in which a space station or refueling depot is located at L1, sometimes in combination with lunar propellant production and a reusable lander.

In contrast to the L1 rendezvous situation, however, a pre-deployed flight element in a general lunar orbit may not be easily accessible from a defined low Earth orbit except on infrequent occasions, or with the use of additional propulsion. This outbound rendezvous problem is simplified if the asset is located in an equatorial or polar orbit about the Moon rather than in an arbitrary orbit. In either case, however, the lunar parking orbit of the pre-deployed asset will define the landing site opportunities for a given mission within the performance constraints of the flight elements.

3.6 Cargo delivery

The ability to deliver a lunar habitat, power generation equipment and cargo to the lunar surface is an important consideration for lunar exploration missions involving lengthy surface stays. Depending on the surface mission requirements, the sizing of the lander and in-space propulsion stages may be driven by cargo delivery requirements rather than by the ascent stage and crew module associated with a crewed lander.

If the lander is optimized for the human exploration mission, the maximum potential cargo capability of the lander descent stage is roughly the mass of the lander ascent stage and crew module. As a result, the direct return architecture offers the highest single-mission cargo delivery capacity of the three architecture alternatives. Conversely, an optimized lunar orbit rendezvous approach offers the lowest cargo delivery capability.

3.7 Mission Environments

The CEV must be designed to operate in a wide range of environments beginning with the complex, but well-understood thermal, radiation and MMOD environments of low Earth orbit. The duration of CEV exposure to the LEO environment could vary from hours to days depending on the selected launch strategy and Earth orbit departure approach.

After the CEV departs LEO, it will pass through the Van Allen radiation belts and enter a "free-space" thermal, radiation and micrometeoroid environment during the transit to the vicinity of the Moon. During the transit phase the Earth and the Moon will have limited influence on the CEV operating environment. A CEV parked at the L1 libration point will experience this free-space environment throughout the lunar surface phase of the mission. A slow roll of the CEV, if compatible with the design of the spacecraft subsystems, can be employed to more evenly distribute the solar flux.

In the lunar orbit rendezvous architecture the CEV remains parked in orbit around the Moon during the lunar surface phase, and is subject to a more complex range of thermal inputs. Because the Moon has no atmosphere to moderate surface temperatures, lunar surface infrared emissions vary widely depending on lighting conditions, and can represent a significant thermal input to a CEV in a low lunar orbit. The orbiting CEV will also be subject to frequent light/dark cycles that are a function of the relative positions of the CEV, Sun and Moon as well as the altitude and inclination of the CEV lunar orbit. A slow roll of the CEV, if compatible with the design of the spacecraft subsystems, can be employed to more evenly distribute the thermal inputs.

In the direct return architecture the CEV is integrated with the lander and carried to the lunar surface. During the lunar surface mission the CEV will experience the range of thermal and lighting conditions associated with the latitude of the selected landing site. In general, the lunar surface temperature varies from approximately +250 ^{O}F to –300 ^{O}F near the equator with more moderate temperature peaks at higher latitudes. In practice the lunar terrain can have a significant influence on the thermal environment of the lander on the lunar surface. Barring an unusual lander design, the CEV will be in a fixed orientation relative to the Moon during the lunar surface mission.

3.8 General Mission Constraints

Any nominal mission design constraint that is coupled with the orbital mechanics of the Earth-Moon system has the potential to greatly constrain the frequency of lunar mission opportunities. These constraints are typically associated with orbital departure opportunities from the Earth or Moon, mandatory outbound or inbound rendezvous events, or landing site locations/characteristics at the Earth or Moon. Some nominal mission constraints may be generally incompatible with off-nominal mission events, such as an in-transit abort or an early return from the lunar surface.

One example of a landing constraint at the Moon is the Apollo Program specification of a near-dawn lighting condition to facilitate visual identification of landing hazards. Lighting conditions on the lunar surface repeat on approximately a monthly basis (lunar synodic period of 29.531 days). If a surface lighting condition is specified as a constraint for the lunar exploration program, then it must be satisfied in conjunction with an Earth departure opportunity. The use of a multi-launch strategy with LEO assembly restricts the Earth departure frequency to once every 8 to 12 days. Therefore, any slip from the nominal mission schedule will result in a lengthy delay until the next intersection of an EOD opportunity with the desired landing site and lighting conditions. A backup landing site approach is not viable in the context of long duration missions to a single landing and relying on a surface habitat.

A lighting constraint can also be imposed for the Earth landing site, which is located near the lunar antipode for a direct entry mission. The lighting condition at the lunar antipode is dictated by the position of the Moon at lunar departure. To ensure a daytime Earth landing for a direct entry return, the CEV must depart the Moon within the two-week period centered on the full moon. If the Earth lighting constraint is also applied to off-nominal mission events, such as an in-transit abort or anytime Earth return from the lunar surface, then the CEV must be capable of a skip re-entry or aerocapture to enable a daylight touchdown at Earth.

Another potential mission constraint is a requirement to rendezvous with an asset in a fixed low Earth orbit on the return (inbound) leg of a lunar mission. Even with nominal mission execution, the necessary orbital alignment for rendezvous and phasing with a fixed low Earth orbit on the inbound leg of a lunar mission is infrequent and will significantly reduce mission opportunities. Rendezvous with a fixed LEO asset becomes even more limiting when considered in combination with other mission constraints and may not be feasible for off-nominal mission events that force an early return to the Earth.

3.9 Earth Return and Recovery

The Apollo Program utilized a direct entry with water landing for the recovery of the astronauts at the conclusion of the lunar missions. The Apollo capsule and similar blunt body shapes typically offer low hypersonic lift-to-drag ratios (L/D~0.3) at modest angles of attack, resulting in very limited cross range steering in the range of 60 to 80 nautical miles. As a result, the landing latitude for a direct entry approach is defined by the position of the lunar antipode at the initiation of the lunar orbit departure burn. The landing longitude can be varied along the ground track of the antipode by modifying the flight time of the inbound transit to take advantage of the Earth's rotation. For a general lunar mission, the opportunities for the land recovery of a spacecraft using direct entry are relatively limited, particularly if the mission design involves the disposal of a propulsive stage (e.g., Apollo Service Module) at the Earth.

In an aerocapture approach, a spacecraft dissipates much of its relative velocity in the Earth's atmosphere before exiting to a temporary phasing orbit. The spacecraft subsequently performs a de-orbit maneuver to a landing site on or near its orbital ground track. The ability of the spacecraft to target a specific land or water site is driven by its hypersonic L/D and the period of time it is capable of loitering on-orbit. The number of landing areas, size of landing areas, orbital inclination, orbital period and loiter time all factor into the orbital phasing calculations. The ability of a spacecraft to perform an Earth aerocapture with LEO phasing greatly increases the frequency of opportunities for land landing.

A variant of the L1 TRM was developed to assess the design impacts of adding aerocapture and loiter functionality to the CEV capsule. The increase in the required propulsive capability from 10 m/s to 112 m/s for the de-orbit maneuver resulted in a switch from the simple Tridyne RCS to a higher performance liquid oxygen/ethanol bipropellant system. The capsule design was also modified to incorporate the additional stand-alone resources required for life support, thermal control and power generation for a twelve-hour LEO loiter period and to address the impact accelerations associated with a land landing. These enhancements resulted in a total architecture mass increase of approximately 10t or 4.5% over the L1 TRM IMLEO of 230t. The size of the launch escape rocket, which was not addressed as part of the LDRM-2 study, would also increase as a result of the changes to the CEV capsule design.

3.10 Crew-Days on the Lunar Surface

The LDRM-2 Phase 1 study included parametric variations on the number of crew and the length of the surface mission. The L1 TRM is sized for four crew and a seven-day surface mission, or 28 crew-days on the lunar surface. Changes of plus or minus two crew to the L1 TRM approach resulted in corresponding IMLEO changes of approximately +/- 10%, or roughly 10t per crew-person. In the context of the L1 TRM, each additional crewperson provides seven additional crew-days on the lunar surface. Variations in the duration of the surface stay from three to fourteen days were also examined for the L1 TRM, and resulted in an IMLEO delta of roughly 2.4t per day for a crew size of four. For a fourteen day surface stay the IMLEO was 246t versus the 230t for the L1 TRM. Since each additional day results in four additional crew-days on the lunar surface, the additional 16t required for a fourteen day surface mission results in twenty-eight additional crew-days on the lunar surface.

On the basis of crew-days on the lunar surface, it is much more efficient to increase the length of a surface mission than to increase the number of crew. An increase in the length of the surface mission primarily involves additional consumables for power, thermal and life support functions. An increase in the number of crew involves additional consumables plus the mass impacts associated with increases in the habitable volumes of the CEV and lander and additional EVA suits, emergency supplies and similar equipment.

It should be noted that the applicability of this data is limited to lighted surface missions of limited duration. Longer duration surface missions will involve changes in subsystem technologies that exceed the scope of these parametric analyses.

3.11 Internal Cabin Design Pressure

The selection of the cabin atmosphere pressure and composition for a lunar mission is influenced by a number of considerations including human physiology, EVA, materials flammability and structural mass. Human physiological needs can be met over a fairly wide range of pressures as long as the oxygen level is conducive to crew health and effectiveness. EVA factors such as pre-breathe duration and risk of decompression sickness favor a lower cabin pressure with an enriched oxygen concentration. A low internal pressure also reduces the structural mass of the cabin pressure vessel. Flammability considerations favor a higher cabin pressure with a reduced concentration of oxygen. An operational nitrogen/oxygen atmosphere of 9.5 +/- 0.5 psia with oxygen concentrations in the range of 27 – 30 % is believed to represent a reasonable compromise for lunar exploration missions. As a basis of comparison, the cabin pressure of the Space Shuttle Orbiter is reduced to 10.2 psia with an oxygen concentration of 30% prior to EVAs.

Although the CEV crew cabin may operate at reduced pressures during a lunar mission, there is merit to the idea of designing it to a full atmosphere structural requirement plus relief valve margin. Based on preliminary sizing estimates, the internal design pressure has a relatively small impact on the mass of the cabin pressure vessel – roughly 200kg for a delta design pressure of 6 psi and a relatively large pressurized volume of 22 m^3 (780 ft^3). Using the CEV gear ratios developed during the LDRM-2 study, 200kg of additional CEV dry mass translates to relatively small IMLEO increases in the range of 1.0 to 1.3t for the L1 rendezvous and lunar orbit rendezvous architectures. The CEV cabin design and operational environments also differ considerably from that of the lander. Unlike the lander, the CEV carries crew during the launch and re-entry phases of a lunar mission. By eliminating the need to vent the CEV cabin during ascent, a full atmosphere design would simplify operations in the event of an ascent abort, and would also provide additional structural robustness in an ascent blast overpressure environment. A CEV cabin designed to a full atmosphere specification could also be operated at sea level conditions during the transit phases of a lunar mission, providing increased crew safety by reducing flammability concerns. While EVA is a significant driver for the lander crew module, the CEV only supports contingency EVA functionality and is not driven as strongly by operating pressure concerns such as prebreathe duration or risk of decompression sickness. Although not necessarily a major consideration, a full atmosphere CEV cabin design would also improve potential compatibility with the International Space Station or future space-based assets.

4.0 Task Description

4.1 Background

Human missions to the moon will be conducted in preparation for future human missions to Mars. Three lunar design reference missions (LDRMs) have been developed to bracket a range of lunar mission scenarios to determine required functionality of the system elements. These missions serve as a point of departure for subsequent architecture analysis. The trade reference mission scenario for LDRM-1 is a 7-day surface stay in the equatorial region of the moon. LDRM-2 is a 7 day surface stay, with global access of the lunar surface enabled via multiple missions. LDRM-3 is a 30-90 surface stay, multiple missions to a single polar site with additional surface elements.

The LDRM-2 study, which was selected as the starting point for the lunar architecture studies, employs a trade reference mission approach supplemented with a number of incremental mission variations to establish design parameter sensitivities. These variants were grouped into architecture and parametric categories.

4.2 Phase 1 Task Description

The purpose of the Phase 1 mission is to enable global access for human exploration of the lunar surface via seven-day missions to multiple landing sites. These missions will support a wide range of scientific investigations, technology and operations development and systems testing to reduce the risks of future human exploration of Mars.

4.2.1 Ground Rules

- Subsystem technology freeze at (TRL 6) six years before IOC (use TRL 6 by 2009 as your reference for design). Freeze time increases to 9 years for “major architectural” drivers (e.g., in-flight refueling).
- First lunar mission 2015-2020
- Exploration missions are expected to be mass and volume limited, thus placing a premium on design efficiency.
- The primary focus of the study is to provide Code T with the information needed to develop effective CEV Level 1 requirements.

4.2.2 Trade Reference Mission Assumptions

1. One human lunar mission per year
2. Return mass from the moon is 100 kg. Return samples may require conditioning (consider biological and planetary materials samples, TBD)
3. Payload to lunar surface (science and enhanced EVA mobility) is 500kg
4. All mission elements placed in LEO (28.5 deg 407km circular)

5. DRM analysis should determine and baseline minimum launch capability required for a 4-launch solution.
6. Consider the lunar mission elements to be “cargo” in terms of delivery to the LEO parking orbit. The propulsive capabilities of the lunar mission elements will not be employed for orbit insertion, but may be required for orbit maintenance.
7. Automated rendezvous and docking shall be used to assemble the elements (identify required interfaces, resources across the interfaces, and contingency operations)
8. Assume 2 weeks between launches (identify any sensitivities/major architectural implications).
9. Crew must be launched on a human rated launch system
10. A dedicated lunar lander element with a separate crew module will be used to transfer the crew from the lunar vicinity to the lunar surface and back to lunar vicinity.
11. Surface stay 7 days
12. 4 crew with all crew going to the lunar surface
13. Daily EVAs will be conducted on the surface of the Moon from the lunar lander.
14. The CEV and lunar lander are not required to be reusable and will not be explicitly designed for reusability.
15. The lunar lander will not be designed to provide functionality beyond that required for the planned lunar surface stay time.
16. The reference lunar surface environment for landing operations and the surface stay is a relatively benign, Apollo-type thermal and lighting condition.
17. A Crew Exploration Vehicle (CEV) element will provide the crew habitation function from the earth’s surface to lunar vicinity and back to the earth’s surface.
18. The nominal Earth return for the CEV is a direct entry with a water landing.
19. The CEV design will incorporate functionality for land landing as a contingency for an ascent abort.
20. CEV shall include the capability for contingency EVA’s
21. Radiation shielding shall be incorporated into the design of the CEV and lunar lander crew modules to provide a core level of biological protection for the crew during transit and on the lunar surface (Code T to give guidance).
22. Libration point L1 is used as the lunar vicinity rendezvous point to enable global lunar surface access.
23. Communications and tracking systems will be emplaced to support global human and robotic surface operations.
24. The lunar lander will be pre-deployed to lunar vicinity prior to initiation of the CEV mission.

25. Assume LH2/LO2 propellants for the L1 transfer stage(s).
26. Assume CH4/LO2 propellants for all other propulsive stages.

4.2.3 Specific Trades Studies

After completion of the L1 trade reference mission, a series of major variations will be conducted to show the effect of the architectural approaches. The variations are listed in priority order. Unless otherwise stated, the trades involve changing only the single, listed parameter from the L1 TRM. In addition to the architectural variations, a series of smaller-impact parametric variations will be performed on select systems to gain an appreciation of the sensitivities for more subtle changes against the L1 TRM.

4.2.3.1 Architectural Variations

1. L1 TRM but with a 2-launch solution. The crew launch will be included in 1 of the 2 launches.
2. L1 TRM with a 3-launch solution. In this case, the crew launches separately, e.g., the 3rd of the launches.
3. L1 TRM but assuming a constraint of 25 metric tons maximum per launch. Determine number of launches required.
4. L1 TRM but with lunar orbit rendezvous instead of L1.
5. L1 TRM but with initial CEV/lander mating in LEO instead of L1.
6. L1 TRM but with single pass aerocapture, de-orbit phasing, and capability of land landing instead of direct entry and water landing.
7. L1 TRM but with direct lunar landing instead of separate lander and L1 rendezvous (not an original task requirement – conduct only as time allows).

4.2.3.2 Parametric sensitivity variations (conducted against the L1 TRM)

1. Effect of alternate propellants.
2. Effect of different power source options.
3. Effect of variation in return payload from the 100 kg baseline.
4. Effect of variation of payload to lunar surface from the 500 kg baseline.
5. Effect of all vs. partial crew to the lunar surface.
6. Change in crew size to 2.
7. Change in crew size to 6.
8. Change in time between launches from 1 week to 30 days.
9. Effect of changing lunar surface stay time to 3 days.
10. Effect of changing lunar surface stay time to 14 days.
11. Effects of elimination of CEV contingency EVA requirement
12. Define sensitivity to total mass as a function of radiation shielding (e.g., curve of total mass vs. probability of medical issue). In parallel, continue developing automated tools/processes for determination of radiation protection as function of spacecraft configuration.

4.3 Figures of Merit

Figures of Merit (FOMs) are provided for guidance in helping the analysis team develop the baseline DRM within the constraints listed above. Data from trades and analysis should support an independent FOM assessment. Note some FOM data has been identified as not required for this study.

4.3.1 Safety/Reliability

To what degree does an architecture ensure safety and productivity for all mission phases?

- Reliability estimates (Not required in this assessment)
- Design redundancy (For this study, only an assessment of functional redundancy between elements is required)
- Abort options for all mission phases
- Time required to return the crew to Earth at various key points in the mission in the event of a contingency.
- Identification of mission risks and system hazards
- Launch risks (Not required in this assessment)

4.3.2 Effectiveness and Evolvability

To what degree does an architecture provide flexibility to meet current mission and future mission needs?

- Applicability and evolvability of technologies, systems (life support, in-space propulsion, power), elements (CEV, landers, habitat, EVA suit, surface power, etc.), and operations of a lunar architecture to future Mars missions, and Mars mission risks that are retired.
- Assessment of degree to which the architecture allows for simple interfaces between elements.
- Assessment of architecture mission complexity (e.g. number of elements, docking and assembly requirements, total mission duration, launch and return opportunities, etc.).
- Assessment of capability to satisfy science objectives (not required for this assessment).

4.3.3 Development Risk and Schedule

To what degree does an architecture reduce development and schedule risks?

- New technologies used
- Benefits of the new technologies (either to lunar missions or as a development step to support Mars missions)
- Current TRL of new technologies, and assessment of effort required to bring technology to TRL 6 by 5 years prior to initial ops capability date

- Assessment of technologies used versus IOC date
- Assessment of ability to develop required architecture elements within integrated schedule (not required for this study)

4.3.4 Affordability

To what degree is an architecture expected to provide lower initial and total life cycle costs?

- New technologies identified
- Program flight elements, mass
- Program facility needs
- Identification of Program elements that will have fixed operating costs (e.g. sustaining engineering hardware production, ground and mission ops, etc.).
- Identification of Program elements that will have recurring cost for each mission (e.g. sustaining engineering hardware production, ground and mission ops, etc.)
- Identification of investments in Lunar missions that directly support future Mars missions (technologies, systems, elements)
- Total mass required to be delivered to LEO to support initial mission (includes pre-deployed/infrastructure, if any) and for each subsequent mission.

5.0 LDRM-2 Study Participants

5.1 Roles and Responsibilities by Organization

The LDRM-2 study benefited from the contributions of a wide range of personnel currently supporting space flight activities at the Johnson Space Center. The principal LDRM-2 team members are listed below by organizational code. Credit is also due to the NASA Headquarters personnel managing and integrating the human and robotic exploration studies across the agency.

The LDRM-2 study leverages decades of spaceflight experience and historical design data in combination with modern analysis tools and techniques. As a result, special acknowledgement must be given to the contributions of the entire NASA and contractor team, both past and present.

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EX	Deputy Study Lead / Architecture Sizing	Jim Geffre
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DM	Mission Operations	Don Pearson
EC	ECLS/ATCS	Kathy Daues
EC	ECLS/ATCS	David Westheimer
EC	EVA	Michael Rouen
EC	EVA	Robert Trevino
EG	Mission Analysis Team Lead	Jerry Condon
EG	Mission Design and Orbital Mechanics	Sam Wilson (retired consultant)
EG	Mission Analysis, Rendezvous	Robert Merriam
EG	Mission Analysis, Earth Return	Tim Dawn
EG	Mission Analysis, Earth Entry	Mike Tigges

EG	Mission Analysis, Landing and Recovery	Chris Madsen
EG	Trajectory Analysis and Visualization	Carlos Westhelle
EG	Trajectory Visualization	Dick Ramsell
EG – UT	Trajectory Design and Rendezvous	Dr. Juan Senent
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EG	Guidance, Navigation and Control	David Strack
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EP	Propulsion	Mike Baine
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ES	Structures	Greg Edeen
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ES	TPS/PTCS	Steve Rickman
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EV	Software	Sid Novosad
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EX	Operations and Systems Integration	Jon Lenius
EX	Mass Properties	Wayne Peterson
EX	Design Integration	Ann Bufkin
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EX	Crew Survival	Leo Langston
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SK	Radiation	Frank Cucinotta
SL	Space and Life Sciences	John Charles
SL	Space and Life Sciences	Tom Sullivan
SX	Micro-Meteoroid and Orbital Debris	Eric Christiansen

5.2 Final Report Documentation

The following individuals authored or provided material contributions to sections of the LDRM-2 Final Report.

Section	*Description*	*Authors*
Section 1	Introduction	Ed Robertson
Section 2	Study Scope	Ed Robertson
Section 3	Executive Summary	Ed Robertson
Section 4	Task Description	Bret Drake
Section 5	LDRM-2 Study Participants	Ed Robertson
Section 6	Study Methods, Tools and Validations	Jim Geffre
Section 7	Introduction to Major Architectural Considerations L1/LOR Hybrid Architecture	Ed Robertson Tom Sullivan
Section 8	Lunar Mission Design Considerations	Jerry Condon Sam Wilson (retired) Robert Merriam Michael Tigges Tim Dawn Carlos Westhelle Dr. Juan Senent (UT) Dave Hammen (Odyssey)

Section 9	Element Overview	Liana Rodriggs Ann Bufkin
Section 10	L1 Trade Reference Mission (TRM) Safety & Mission Success Mission Abort Options Element Overview System Technology and TRL Summary	Jim Geffre Jan Railsback Randy Rust Bryan Fuqua (GHG) Clint Thornton (GHG) Leo Langston Liana Rodriggs Ann Bufkin Jim Geffre Keith Williams
Section 11	TRM with Two-Launch Solution	Jim Geffre
Section 12	TRM with Three-Launch Solution	Jim Geffre
Section 13	TRM with 25t Launch Limit	Jim Geffre
Section 14	TRM with Lunar Orbit Rendezvous	Jim Geffre
Section 15	TRM with CEV/Lander Mating in LEO	Jon Lenius
Section 16	TRM with Aerocapture, Phasing and Land Touchdown	Jon Lenius
Section 17	TRM with Direct Earth Return	Ed Robertson
Section 18	Architecture Comparison	Ed Robertson
Section 19.1	Alternate Propellants	Eric Hurlbert Mike Baine
Section 19.2	Alternate Power Sources	Jim Geffre
Section 19.3	Variation in Return Payload Mass	Jim Geffre
Section 19.4	Variation in Delivered Payload Mass	Jim Geffre
Section 19.5	Effect of All vs. Partial Crew to the Lunar Surface	Jon Lenius
Section 19.6	Reduction in Crew Size– 2 Crew	Jon Lenius
Section 19.7	Increase in Crew Size – 6 Crew	Jon Lenius
Section 19.8	Variation in Launch Spacing from 7 to 30 Days	Jon Lenius
Section 19.9	Reduction in Lunar Surface Stay – 3 Days	Jon Lenius
Section 19.10	Increase in Lunar Surface Stay – 7 Days	Jon Lenius

Section 19.11	Assessment of CEV Contingency EVA Capability	Stan Love
Section 19.12	Recommended Cabin Design Pressure	David Westheimer
Section 20.1	System Technology – Propulsion	Eric Hurlbert Mike Baine
Section 20.2	System Technology – Power	Karla Bradley
Section 20.3	System Technology – ECLSS	Kathy Daues
Section 20.4	System Technology – ATCS	David Westheimer
Section 20.5	System Technology – Habitation Systems	Susan Baggerman
Section 20.6	System Technology – EVA	Robert Trevino
Section 20.7	System Technology – Avionics	Coy Kouba David Jih Helen Neighbors
Section 20.8	System Technology – GN&C Contributing Engineers:	Thomas Moody Brian Rishikof David Strack Tim Crain Howard Hu
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Section 20.10	System Technology – Structures	Gregg Edeen
Section 20.11	System Technology – PTCS	Steve Rickman
Section 20.12	System Technology – TPS	Chris Madden
Section 20.13	System Technology – Mechanisms	James Lewis
Section 20.14	Mission Environment – Thermal	Steve Rickman
Section 20.15	Mission Environment – Radiation	Frank Cucinotta
Section 20.16	Mission Environment – MMOD	Eric Christiansen
Section 20.17	Risks and Hazards Assessment	Jan Railsback Randy Rust Bryan Fuqua (GHG) Clint Thornton (GHG)

6.0 Study Methods, Tools, and Validations

This section describes the design processes and tools used to conduct the LDRM-2 study. The LDRM-2 task was primarily an architecture trade study to examine the impacts of various lunar transportation strategies, as opposed to the more familiar detailed vehicle design studies using a distinct pre-selected mission architecture. Therefore, the study modified the methods applied to reflect the nature of the given task better. Measuring relative differences between architectures, instead of absolute highly optimized vehicle mass estimates, received greater emphasis. To generate the necessary vehicle properties quickly for the many architecture and parametric variations requested, the study team employed a spreadsheet-based parametric mass/power/volume estimating tool instead of more time-consuming manual sizing methods. The need to produce high confidence results with the tool was recognized, however, so measures were taken to validate it against historical human spaceflight examples. The team chose the as-built vehicles used in the Apollo 17 lunar mission for comparison, and this section describes the results of those efforts.

6.1 Study Methods

Due to the large number of architecture and parametric trades requested in the LDRM-2 task request statement, a traditional design process where subsystem experts generate initial subsystem mass, power, and volume estimates for a vehicle and iterate on their estimates until the design converges was considered too impractical for the limited time allotted for the study. Instead, a single integrated sizing tool representing the major subsystems in a typical exploration vehicle was used to reduce greatly the length of time required to analyze a given architecture. This tool, called Envision and described in detail in Section 6.2, contains embedded mass, power, and volume parametric estimating relationships to evaluate the components of a vehicle and performs any necessary iteration internally.

The process for conducting the LDRM-2 study began by first establishing a trade reference mission (TRM) from the architecture assumptions enumerated in the task request statement. Seven unique candidates for the TRM were identified using different delta-V allocations and propulsion system types for the assumed suite of architecture elements (CEV, Lunar Lander, etc.). From these seven candidates, one option was downselected as the phase I baseline. The LDRM-2 team developed mission and abort timelines, critical events lists, vehicle mass properties, and technology needs for this baseline. Next, architecture variants performed in the study using different rendezvous strategies, launch packaging, and CEV landing modes then used the TRM as a reference to determine the relative merits of those changes. Parametric variants used the TRM, without affecting the architecture, to modify key vehicle design parameters such as crew size, propulsion system type, and others to measure their impacts. Sections 10.0 – 19.0 of the report describe the TRM and architecture/parametric variations in depth.

As mentioned, mass properties for the architecture elements were generated using the Envision parametric sizing tool. This process involved first developing technology lists for TRM vehicles by the study leads and presenting these options to subsystem experts on the team. The LDRM-2 team included specialists representing all of the major subsystems included in a typical human spacecraft. Subsystem experts verified the lists of technologies and added, removed, or changed the selections where appropriate. One example of a subsystem technology changed was the vehicle mating system. The initial technology list used an Apollo-like probe & drogue system to provide pressurized docking and crew transfer between the CEV and Lunar Lander. Instead, the LDRM-2 mating system lead recommended a fully androgynous, low impact docking system. After making any necessary changes, the team finalized the technology list. The Envision sizing tool then generated preliminary mass properties for the CEV, Lunar Lander, Earth Departure Stages, and Kick Stage included in the TRM. Mass properties were presented to the subsystem experts for validation, and inputs to the sizing tool or Envision sizing algorithms were altered to fix any outstanding sizing discrepancies. The team approved resulting mass properties for the TRM, and finally, the study leads generated mass properties for the requested architecture and parametric variations.

6.2 Envision Tool Description

The Envision Exploration Vehicle System Estimation tool is a Microsoft Excel-based integrated parametric systems engineering tool developed to assess rapidly system mass, volume, and power requirements for future human exploration concepts such as interplanetary transportation or habitation vehicles. It has been in development since 2001 to assist exploration architecture and vehicle designers in providing quick-turnaround responses to questions of mission or vehicle concept feasibility. This tool consists of a series of linked spreadsheets representing each of the major subsystems in a typical exploration vehicle, with each spreadsheet in the tool having either been developed by JSC vehicle subsystem experts or by the tool developer using mass, power, and volume estimating relationships supplied by the subsystem experts. Such relationships might be physics-based or empirically-derived from past human exploration concepts. Given user inputs, the tool sizes each of the systems and then presents vehicle mass, volume, and power properties on a summary sheet. Efforts are currently underway to independently verify and validate Envision sizing relationships and tool outputs for completeness and correctness.

Three major layers comprise the Envision tool. These layers are (1) the *main input* layer, (2) the *system sizer* layer, and (3) the *vehicle summary* layer.

1) The *main input* layer is a single worksheet within the tool providing a centralized location for user inputs regarding high-level vehicle design parameters. These parameters include such data as crew size, mission timeline, pressurized volume, delta-V, cabin atmosphere pressure, payload size, and others. The tool distributes inputs provided on the main input layer to the system sizers for mass, volume, and power calculations.

2) The *system sizer* layer consists of a series of linked worksheets embedded in the Microsoft Excel workbook that compute mass, volume, and power requirements for exploration vehicle concepts. These systems include avionics, crew accommodations, descent & landing, environmental control and life support, EVA and suits, power, propulsion, structures and thermal protection, and thermal control. Each system worksheet divides further into four sections: (1) a reserved input/output section, (2) a user interface section, (3) an analysis section, and (4) an output section. A worksheet's reserved input/output section provides data connectivity to other sizers within the tool. All external variable values required by the sizer or variable values produced by the sizer that are required in another sizer are in this section. The user interface section allows the user to specify subsystem component types, technologies, and quantities relevant to the vehicle analyzed. The

analysis section within a worksheet is the heart of the system sizer. It performs the mass, volume, and power estimates for a system using pre-programmed estimating relationships based on historical data or physics. A typical mass relationship might take the form:

$$\text{Fuel Turbopump Mass} = 1.95 * \left(\frac{\text{Pump Power Required}}{\text{Rotation Speed}} \right)^{0.6}$$

This relationship uses a power law equation to scale engine turbopump mass using pump power, pump rotation speed, and two empirically derived scaling coefficients. The analysis section retrieves user inputs from the reserved input/output section and user interface section, computes mass, volume, and power estimates using relationships similar to the turbopump equation, and provides its results to the final section, the output section. The output section summarizes the results of a system sizing into a few quantities used in the vehicle summary layer.

3) Finally, the *vehicle summary* layer is a single worksheet within the tool used to summarize concisely results of the sizer calculations. This worksheet includes a mass, volume, power summary of all major system components, vehicle dry, inert, total mass estimates, and charts detailing allocations between systems.

Figure 6.2-1 depicts a notional construct of the current Envision application. The diagram shows the three tool layer – the main input layer, the system sizer layer, and the vehicle summary layer. Lines show connections between the layers and the individual software tools. In most cases, these lines represent two-way communication between components.

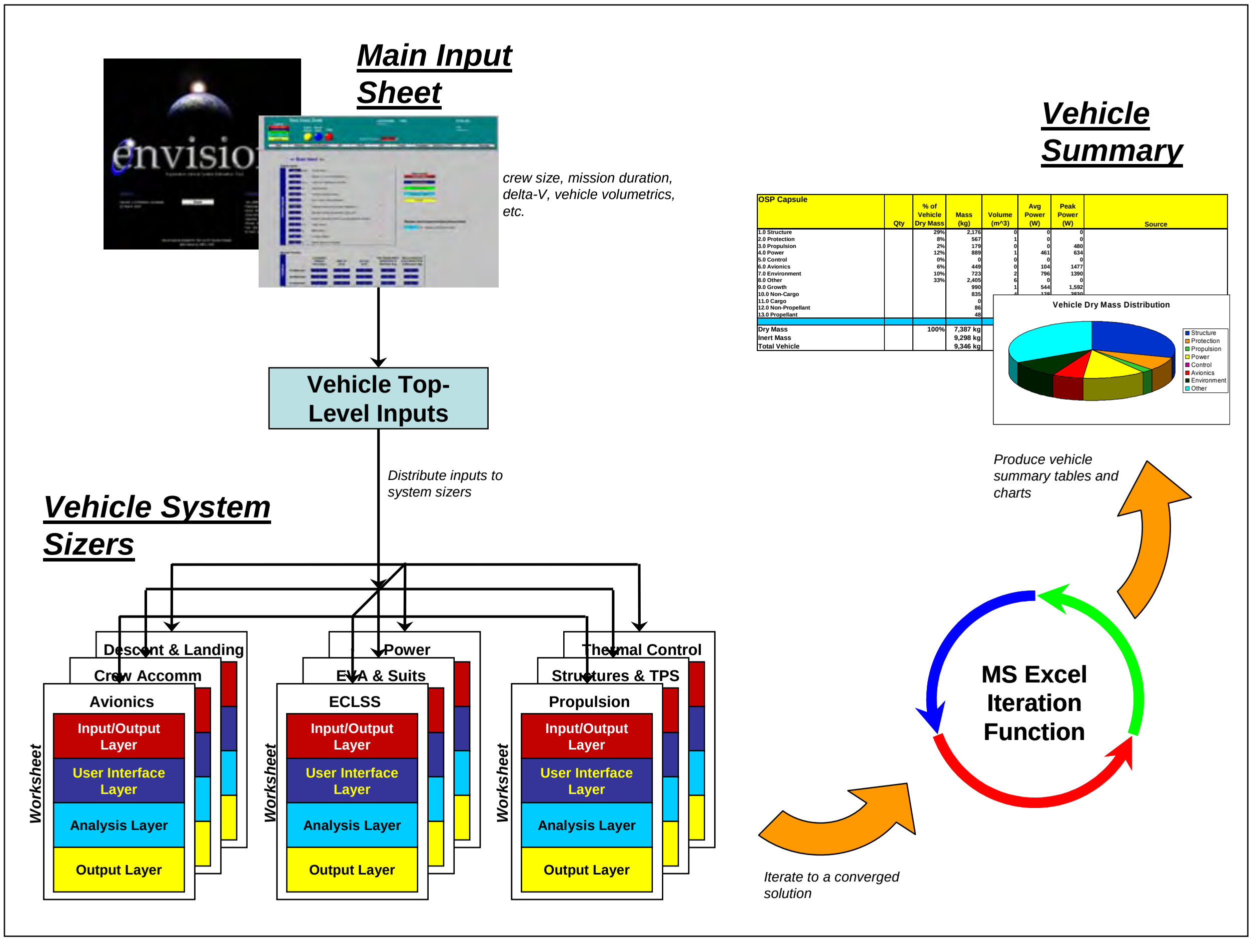


Figure 6.2-1: Envision Configuration Diagram

6.3 Mass Properties Validation

After producing mass properties for the trade reference mission and subsequent architecture variants, the Envision tool generated an additional test case for validation against a historical human spaceflight example. The example selected for validation was the last and most ambitious Apollo mission, Apollo 17. The validation process was not intended to replicate the exact design of the Apollo vehicles, rather to replicate the capabilities of that mission using modern technologies and vehicle design practices while retaining as much commonality between the two as possible. Thus, some key architecture parameters such as crew size, surface duration, and rendezvous strategy were identical to the Apollo 17 mission, while others, such as number of launches, amount of radiation protection, level of fault tolerance, and propulsion system types, were common with the TRM. Table 6.3-1 outlines the resulting validation test case, called "Pseudo-Apollo", and compares its features to Apollo 17.

	Pseudo-Apollo	*Apollo 17*
Number of Launches	4 per mission 2 weeks between launches Element assembly in LEO	1 per mission
Dedicated Radiation Protection?	~1,600 kg polyethylene for CEV and Lunar Lander	None
Service Module Propulsion	Oxygen/Methane	NTO/Aero50-50
Lunar Lander Propulsion	Oxygen/Methane	NTO/Aero50-50
Crew Size	3 crew total 2 crew to lunar surface	3 crew total 2 crew to lunar surface
Architecture Type	Near equatorial-limited lunar orbit rendezvous	Near equatorial-limited lunar orbit rendezvous
Surface Duration	3 days	3 days
Mission Payload	500 kg down / 100 kg return	558 kg down / 112 kg return
Airlock?	No	No

Table 6.3-1: Pseudo-Apollo Benchmark Characteristics

Next, the sizing tool generated mass properties for Pseudo-Apollo vehicles and mass properties for the as-flown Apollo 17 vehicles were researched. The resulting architecture initial mass in low Earth orbit (IMLEO) was 142 metric tons for the Pseudo-Apollo case versus 138 metric tons for Apollo 17. The close proximity of these values gave some measure of confidence in the Envision tool's outputs while recognizing that some important differences still exist between the two test cases. For architecture trade studies such as LDRM-2, the tool is likely to be sufficiently precise.

7.0 Introduction to Lunar Mission Design Considerations

The total ΔV required for a lunar mission from Earth departure through Earth return, including the lunar descent and ascent maneuvers, is roughly 9 to 10 km/s (29,500 to 32,800 ft/s). That total is comparable to the ΔV required for launching payloads from the surface of the Earth to low Earth orbit. In contrast, typical crewed LEO missions require only about 0.35 km/s (1,150 ft/s) of on-orbit ΔV – less than 4% of the lunar mission value. As a result, the combined mass of the flight elements delivered to low Earth orbit for a chemical propulsion lunar mission is typically well in excess of 100t (220 Klb), and can easily exceed 200t (441 Klb) for a robust set of mission requirements. The Apollo lunar stack, which is often used as a basis of comparison for lunar exploration studies, had an IMLEO equivalent of roughly 138t (304 Klb). Because the lunar mission architecture has a strong influence on the magnitude and distribution of propulsive ΔV among the flight elements, it also plays a major role in how the mission ΔV translates to launch mass.

There are three basic architectures for executing a human lunar exploration mission – libration point rendezvous, lunar orbit rendezvous and direct return. Each of these lunar architectures offers a different orbital mechanics approach to the task of returning of the crew to the Earth after the conclusion of lunar surface operations. Two of the approaches, libration point rendezvous and lunar orbit rendezvous, stage Earth return assets in orbit to avoid the mass penalty of transporting them to the lunar surface. The third approach, direct return, eliminates the CEV rendezvous on the return leg of the mission at the expense of carrying the propellant and systems required for Earth return and recovery to the lunar surface. Each of the lunar architectures offers a range of mission design options associated with the launch strategy, assembly strategy, Earth orbit departure approach, surface exploration objectives and the definition of flight elements.

7.1 General Mission Design Parameters

The mission design for a lunar exploration program involves a wide range of parameters and constraints, many of which are coupled either directly or indirectly. A clear understanding of the mission objectives is essential to the development of an effective mission design. The lunar architecture is also a key part of the mission design process because it imposes an orbital mechanics and flight environment framework on the mission, and plays a significant role in its overall operational characteristics. In addition, if several types of missions are planned over the duration of the lunar exploration program, then the flight elements must either envelope the full range of required mission functionality from the outset, or support an evolutionary path for the development of additional mission functionality, as required.

The process of defining and sizing a lunar mission to satisfy a nominal mission plan is reasonably involved. The task becomes even more complex when off-nominal considerations are factored into the design process. This is especially true when a series of successful and timely launches are required to meet the nominal mission schedule. The potential for launch delays, rendezvous and mating problems, subsystem failures and software issues will necessitate a wide range of contingency planning with impacts flowing down to the flight elements and mission timeline.

The following list of topics and associated parameters are representative of the fundamental issues that must be addressed during the development of a lunar mission design. This list is primarily targeted to the definition and sizing of the lunar flight elements and their general compatibility with the Earth-to-orbit launch system, and should not be considered to represent a comprehensive list of lunar mission design inputs or options.

Mission Definition

- Mission objectives
- Location of landing site(s)
- Landing site constraints (e.g., lighting conditions at touchdown or during the surface mission)
- Duration of surface missions
- Cargo delivery requirements (e.g., surface infrastructure, habitat & resupply)
- Mission rate

Mission Architecture

Note: The mission architecture specifies the location at which the crew transfers to the spacecraft that provides the Earth return functionality. An outbound rendezvous between flight elements or with surface assets is a separate consideration.

- L1 libration point rendezvous
- Lunar orbit rendezvous
- Direct Return – no rendezvous
- Hybrid concepts

Abort Opportunities

- Earth ascent aborts
- Aborts during transit phases
- Aborts during lunar descent (terminal descent phase requires particular attention)
- Aborts from the lunar surface (anytime return versus loiter capability)
- Safe haven options for Earth rescue operations

Earth-to-Orbit Launch System

- Payload capacity (mass and dimensions)
- Number of launches
- Types of launches, if segregated (crewed versus cargo)
- Launch rate (spacing)
- Launch contingency planning (weather delays, etc.)

<u>Flight Element Assembly Strategy</u>

- Pre-launch integration (no on-orbit assembly)
- Low Earth orbit assembly
- On-orbit propellant loading

<u>Earth Orbit Departure Strategy</u>

- Tandem departure
- Split-mission with convoy departure
- Split-mission with pre-deployment of a flight element

<u>Lunar Surface Assets</u>

Note: Fixed surface assets will likely result in a precision landing requirement at the Moon. Surface asset mobility would provide some useful design flexibility.

- Habitat and resupply for extended duration missions
- Resources to support a dormant lander
- In-situ resource development and utilization (oxygen, water, construction materials, etc.)

<u>Earth Return and Recovery</u>

- Direct atmospheric entry, skip trajectory or aerocapture and de-orbit
- Water versus land landing
- Aerocapture and rendezvous with LEO asset – either in a defined orbit (e.g., transportation node or spacecraft parked in LEO), or launched to an orbit compatible with the CEV Earth return trajectory

<u>Flight Element Definition</u>

- Delta-V allocation
- Propellant selection
- Expendable versus reusable
- Safe disposal of expended elements

<u>Missions Environments</u>

Note: Natural environments and the duration of exposure must be defined for each mission phase.

- Thermal
- Radiation
- Micrometeoroid and orbital debris

<u>Programmatic Issues</u>

- Schedule

- Human rating requirements
- Technology readiness level
- Development and recurring cost targets

7.2 Primary Factors in Architecture Selection

The preferred architecture for human lunar exploration will depend on the combination of mission constraints, mission objectives and crew safety requirements identified for the Constellation Program. The primary mission design constraints are the total number of launches and the payload mass and volume capacities of the cargo and human-rated launch vehicles. The driving mission objectives include the complete set of lunar landing sites, surface mission durations and EVA/IVA activities deemed necessary to prepare for the future human exploration of Mars. Tying these factors together is the overriding desire for the safe return of the crew to Earth in the event of a major systems failure in any phase of the lunar mission.

7.2.1 Launch Vehicle Payload Mass and Volume

From a launch vehicle standpoint, the distribution of mass and volume among flight elements is of greater importance than the total architecture mass. Any element that exceeds the payload capacities or payload fairing dimensional limits of the available launch vehicles must either be divided into multiple elements or modified in coordination with the other flight elements. The mission architecture provides the basic framework that couples the orbital mechanics of the Earth-Moon system with the flight element design and mission objectives. A mission design that efficiently utilizes launch resources while emphasizing simple flight element interfaces and operations is likely to be preferred over a more complex alternative that results in a lower IMLEO.

For a lunar mission based on chemical propulsion, it is worth noting that roughly three-quarters of the flight element IMLEO is propellant. As a result, on-orbit fueling is an effective technique for reducing the maximum required launch vehicle payload capacity for the larger flight elements. Commercial launch resources may be a viable option for the delivery of bulk propellant to a low Earth orbit depot.

7.2.2 Mission Objectives

Mars missions typically involve months of in-space transit time and up to several years of total mission duration with no capability to rapidly return to Earth. Long missions require large habitable volumes, robust radiation shielding, more numerous spares and large quantities of crew consumables. The primary technical issues are hardware reliability, systems automation, closure of life support systems and shielding from ionizing radiation. Human psychological issues associated with long-duration spaceflight are also of major importance. The International Space Station is a valuable resource for assessing the physiological and psychological impacts of long-duration missions in a micro-gravity environment. Lunar exploration missions will supplement the ISS data by providing a more thorough understanding of the technological, physiological and psychological challenges associated with operations in a hostile, partial-gravity surface environment.

The key to the selection of a lunar mission architecture is a thorough understanding of the mission objectives in terms of the landing site environments, mission durations and surface activities deemed necessary to reduce the risks of future human exploration of Mars. Although the L1 rendezvous and direct return architectures generally result in a high IMLEO, they also inherently provide a wide range of mission flexibility in terms of landing site access and surface stay times, plus the capability to initiate a return to Earth at any time during the lunar surface stay. The lunar orbit rendezvous architecture offers the same anytime return capability with the potential for significant reductions in IMLEO relative to the L1 rendezvous and direct return architectures, if it is possible to constrain the range of landing site locations or surface mission durations.

7.2.3 Crew Safety

Crew safety is a primary concern for human exploration missions. Not only must the nominal mission provide for the safe return of the crew to Earth, but all credible, safety-related failure modes must also be addressed through element and system redundancy and contingency planning. Analyses are in progress to provide a more thorough understanding of the orbital mechanics drivers of the abort modes for each of the three basic lunar architectures.

Each of the architectures provides a different set of contingency Earth return capabilities during the outbound transit, lunar surface and inbound transit phases of a lunar mission. One of the key architectural discriminators is the frequency of Earth return opportunities from the lunar surface. Both the L1 rendezvous and direct return architectures inherently provide the option of initiating an Earth return at any time during the lunar surface mission. The same Earth return functionality is possible for the lunar orbit rendezvous architecture given sufficient plane change capability on the CEV and lander to handle worst-case orbital alignments. The ΔV and IMLEO cost for anytime return from the lunar surface for the lunar orbit rendezvous architecture can be greatly reduced if the range of landing sites or surface mission durations is constrained, or if loiter time on the lunar surface and/or in lunar orbit is considered to be a viable alternative to anytime return. In the case of the lunar orbit rendezvous architecture, the lunar exploration objectives must be addressed in combination with the desired abort functionality from the lunar surface.

The architectural discriminators during the outbound and inbound transit phases are primarily associated with the time required to return to Earth and the need for flight element rendezvous and crew transfer. The Earth orbit departure strategy, number of flight elements and distribution of ΔV will also influence outbound transit abort options through the availability of functional redundancy and margin at the element and subsystem levels. Because flight elements are typically expended during a mission, contingency options are generally much more limited during the inbound transit phase.

The direct return architecture provides the shortest inbound and outbound transit times of the three architectures under discussion – typically 3.5 to 4 days for a one way Earth-Moon transit – and is also the only architecture option that does not require a rendezvous to return the crew to Earth. The single crew module, however, eliminates the functional redundancy at the element level that is possible in a dual crew module approach with a tandem Earth orbit departure.

The lunar orbit rendezvous architecture is very similar to the direct return architecture in terms of outbound transit aborts. After the separation of the lander and CEV in lunar orbit, however, a flight element rendezvous in lunar orbit is required to transfer the crew back to the CEV for Earth return. In addition, the multi-burn lunar orbit departure that may be needed to optimize the

Earth return plane change ΔV adds an extra day to the inbound transit time relative to the direct return architecture.

The L1 rendezvous architecture also requires an Earth return rendezvous following the separation of the CEV and lander on the outbound leg of a lunar mission. However, the additional 2.5 days required for a one-way L1-to-Moon transfer increases the time required to return to Earth in the event of a mission abort relative to the other architectures.

7.3 Alternate Mission Concepts

Although LDRM-2 study resources were focused on the definition and analysis of the three basic mission architectures, the team was encouraged to seek innovative solutions to the human lunar exploration mission defined in the task statement. Additional study will be required to assess the flight element design and operational and safety implications of the alternate mission concepts relative to the basic lunar architectures.

7.3.1 Reference L1/LOR Hybrid Architecture

The L1/LOR hybrid architecture was developed in an attempt to blend the benefits of the L1 and Lunar Orbit Rendezvous mission architectures. Although there was insufficient time in the LDRM-2 study to evaluate this approach, a description of a reference L1/LOR hybrid concept is presented for possible use in future studies. As illustrated in Figure 7.3.1-1, the major operations in this architecture are as follows:

1) The mission begins with the LEO rendezvous and mating of the CEV, lander and Earth Departure Stage(s).
2) Following a tandem Earth orbit departure for the Moon, the lander and CEV coast in a mated configuration for functional redundancy
3) The crew checks out the lander en route and transfers to the lander prior to lunar orbit arrival. The CEV undocks from the lander and maneuvers towards the L1 libration point via lunar swing-by.
4) The lander inserts into the appropriate low lunar orbit and descends to the desired landing site.
5) The unoccupied CEV transits to L1 and loiters at L1 during the lunar surface mission. From its location at L1 the CEV can target a rendezvous orbit over any lunar landing site when departing for the Moon, thus supporting a wide range of surface mission durations.
6) If an abort is declared, then the CEV performs an early departure from L1 for a rendezvous orbit over the lunar landing site. The L1-to-Moon transit time is ~2.5 days.
7) For a nominal mission, the CEV departs L1 for a lunar rendezvous orbit approximately 2.5 days prior to the end of the surface mission.
8) The lander ascends from the lunar surface and rendezvous with the CEV in low lunar orbit. The crew transfers to the CEV and then undocks from the lander.
9) The CEV departs lunar orbit for an Earth return.

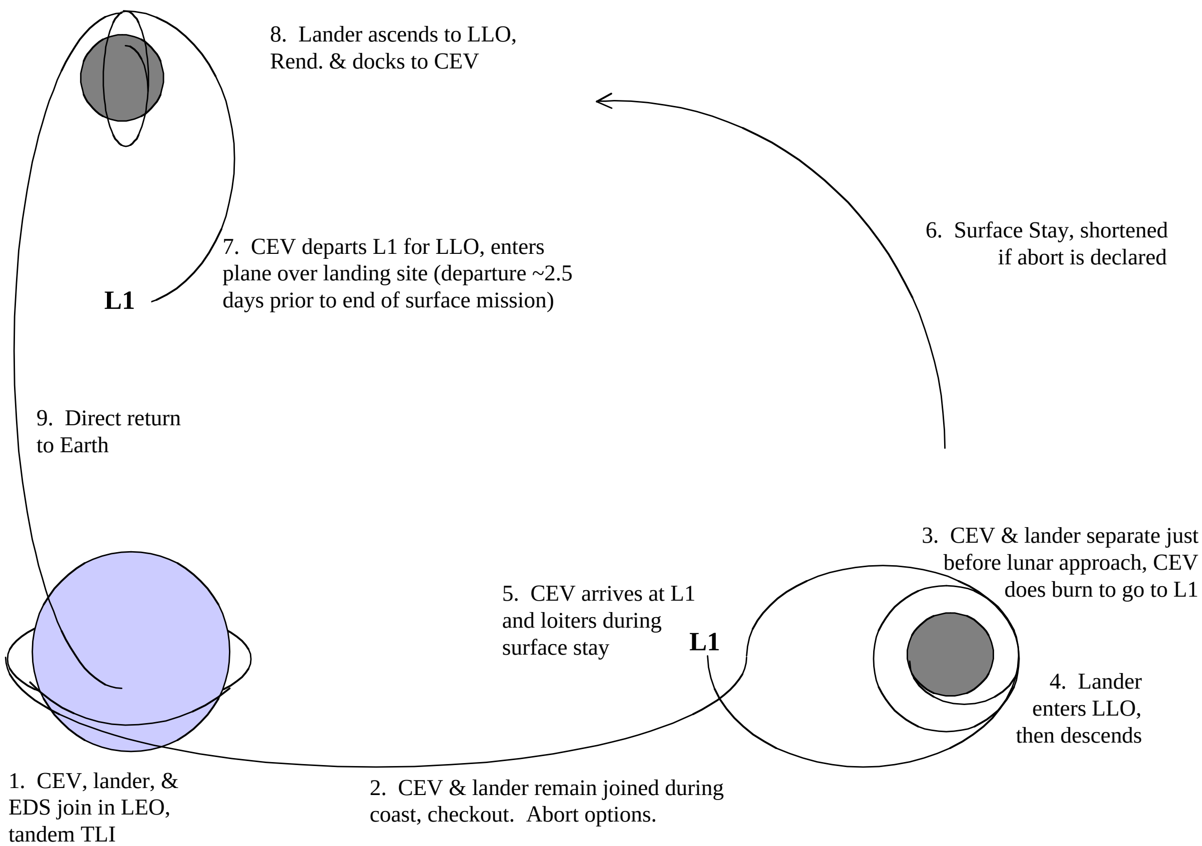


Figure 7.3.1-1: Operations Concept for the Reference L1/LOR Hybrid Architecture

The main thrust of the L1/LOR hybrid approach is the use of lunar orbit rendezvous to minimize the size and mass of the lander element and to shorten the trip time. During the outbound transit the lander bypasses L1 and proceeds directly to a low lunar orbit, thus reducing the transit time for the crew as well as eliminating the L1 arrival and departure ΔV for the lander. The L1 rendezvous characteristics of global lunar access and extended duration surface missions with anytime Earth return are retained in the hybrid approach by parking the CEV at L1 during the lunar surface mission. In the L1/LOR hybrid architecture, however, the CEV departs L1 and enters low lunar orbit in order to pick up the crew from the lander and return them safely to Earth.

An end-to-end assessment in terms of orbital mechanics, critical events and abort modes will be needed to fully assess the merits of the L1/LOR hybrid architecture. In its favor, however, the hybrid L1/LOR architecture reduces the ΔV allocated to the lander by shifting the responsibility for several major maneuvers to the CEV and EDS elements. Under the LDRM-2 study assumptions, this general approach was shown to minimize IMLEO for the basic L1 rendezvous and lunar orbit rendezvous architectures. The ΔV associated with the L1-Moon transit and lunar orbit departure, roughly 850 m/s, is transferred from the lander ascent stage to the CEV by performing the CEV/lander rendezvous in low lunar orbit. Furthermore, in contrast to the lunar orbit rendezvous approach for long duration lunar surface missions, the plane change ΔV required for the lander descent and ascent maneuvers can be minimized by the use of coplanar CEV lunar parking orbits. Finally, the ΔV required in the L1 rendezvous architecture to stop the lander at L1 during the outbound leg of the mission is eliminated in the hybrid approach at the expense of a somewhat larger lunar orbit arrival maneuver.

Architecture mass is not the only important figure of merit for a lunar mission. Crew safety is impacted by the duration of the Earth-Moon transits, availability of aborts throughout the mission, and the time required to return the crew to the Earth in the event of an emergency. It is important to note that the mission duration for the crew is typically shorter in the hybrid L1/LOR architecture than in the L1 rendezvous architecture. Because the CEV is unoccupied during the transits between L1 and low lunar orbit, the nominal crewed mission duration is actually quite similar to that of the basic LOR approach, and roughly five days less than for the basic L1 rendezvous architecture. In the event of an unplanned early termination (abort) of the lunar surface mission, however, the return to Earth time for the hybrid L1/LOR architecture would be about six days due to the delay in retrieving the CEV from L1. That is similar to the return to Earth time for the basic L1 rendezvous architecture, and approximately 1.5 days longer than the return to Earth time for the basic LOR architecture.

The L1/LOR hybrid architecture also offers some interesting variants in which the CEV does not actually loiter at the L1 libration point. While the lander is performing its lunar orbit insertion and descent maneuvers, the CEV will be receding from the moon en route to L1. Should an event require the return of the crew to the CEV, the lander can depart lunar orbit or stop its descent and proceed to L1 to join the CEV. At some point, however, enough propellant will have been used that the lander will be unable to make it to L1. At that point, the lander can remain in lunar orbit until the CEV returns and rendezvous. A complete analysis of this case has not been performed, but it is fair to say that the amount of time these operations may take is longer than that required in the Apollo-style LOR abort scenarios.

An interesting possibility exists for early reconnaissance missions to the moon. The duration of these missions has been suggested to be in the range of four to fourteen days on the surface, during which time the crew would live out of the lander. Since the time for the CEV to coast to L1 after swinging by the moon is on the order of 2.5 days, a round trip back to the moon will take roughly five days. This is about the right amount of time to support a four-day surface mission. For such a mission, the CEV would not insert itself at L1, and thus would not need to perform a burn to depart it, either. It is highly likely, however, that a modest plane change burn at L1 would be required to establish the proper orbit at the Moon for the lander rendezvous. Once the CEV arrives back at the Moon, it would perform a burn to circularize its orbit and prepare for rendezvous with the lander ascent stage. Later missions may choose to have the CEV do another 'lap' or two out to L1 before circularizing at a low lunar orbit, extending the surface duration by another five or ten days. Intermediate stay times are possible by lowering the apogee by the appropriate amount, thus decreasing the orbital period. Consequently, support for exploration missions with surface stay times of 4 to 14 days are possible. This stay time matches what is considered as reasonable for living out of the lander, without support from a separate habitat. This approach is very similar to the high lunar orbit option (24 hour period) discussed for the LOR variant within this study. There are differences, however, both positive and negative. The longer orbital period, five days versus one day, means that at certain times the CEV is less available to support emergency aborts from the surface. It is likely that a burn can be performed to more rapidly return the CEV to a low lunar orbit to support a surface abort, but there will be a price to pay in both propellant and time. The L1/LOR hybrid architecture is also capable of defaulting to a lunar orbit rendezvous approach for near-equatorial or near-polar landing sites to minimize the Earth return time from the lunar surface in the event of an emergency.

If the orbital mechanics of the L1/LOR hybrid architecture prove to be viable and practical, then it may be an interesting alternative to the basic architecture approaches assessed during the LDRM-2 study.

7.3.2 Hybrid Mission Options

A new mission concept naturally leads to its own set of variants. The following two variants to the reference L1/LOR hybrid architecture appear worthy of consideration:

1) Crew transfer to the lander in LEO followed by a split mission Earth orbit departure
2) Basic LOR architecture supplemented by a backup CEV pre-deployed to L1

The reference L1/LOR hybrid mission employs a tandem Earth orbit departure with the CEV departing the lander for L1 as the pair approaches the Moon. In the hybrid split mission variant the crew transfers from the CEV to the lander in LEO. Afterwards, the lander and CEV depart on separate, optimized trajectories to the Moon and L1, respectively. The functional redundancy of the lander and CEV during the outbound leg of the mission is traded for a simplified orbital mechanics approach for delivering the CEV to L1.

The reference L1/LOR hybrid concept also led to the idea of using the LOR architecture as the primary mission approach with a CEV parked at L1 as a supplemental means of Earth return. The backup CEV would require about 2000 m/s of ΔV capability to perform its mission, and would be employed as described in the reference L1/LOR hybrid architecture.

7.3.3 On-Orbit Fueling of Flight Elements

For a lunar mission based on chemical propulsion, roughly three-quarters of the flight element IMLEO is propellant. On-orbit fueling can therefore be an extremely effective technique for reducing the maximum required launch vehicle payload capacity for “high value” mission assets.

In the L1 TRM roughly 76% of the total launch mass is usable propellant, and that percentage climbs to nearly 79% if you discount the CEV CM from the calculations. The cryogenic Earth Departure Stages are the most likely candidates for on-orbit fueling because of their high mass fractions, with approximately 84% to 87% of their masses composed of the liquid oxygen and liquid hydrogen propellants. In addition, the cryogenic Earth Departure Stages have an oxidizer-to-fuel ratio of 6.0, meaning that nearly 86% of the EDS propellant is liquid oxygen. It is possible, therefore, to obtain significant launch mass reductions by offloading only the liquid oxygen from a cryogenic propulsive stage. In the case of the lander EDS the offloading of the liquid oxygen decreases its launch mass by roughly 75%, from 94t to 24t. The same technique is applicable to a propulsive stage using liquid oxygen and methane propellants, but the associated oxidizer-to-fuel ratio of 3.8 renders the approach somewhat less effective.

The use of propellant offloading appears viable for any of the proposed lunar architectures. An aggressive application of the on-orbit fueling approach can result in the delivery of the flight elements to LEO in only two or three launches using a launch vehicle with roughly 25% to 40% of the lift capacity of the Saturn V. The remaining launch mass for a lunar mission, perhaps 100t or more, is delivered to LEO in the form of bulk liquid oxygen. These additional launches could be contracted to the private sector to maximize cost effectiveness. The use of a propellant depot to aggregate the liquid oxygen deliveries in LEO would provide wide flexibility in the number and spacing of launches needed to support a lunar mission. The oxygen depot might also serve as the assembly location for the lunar flight elements prior to Earth orbit departure.

8.0 Lunar Mission Design Characteristics

8.1 Introduction

The prime focus of this study is a libration point rendezvous (LPR) mission employing the cis-lunar Earth-Moon libration point (L1) as a stopover and staging point for round-trip missions to the lunar surface. The study uses libration point rendezvous as a point of departure for subsequent mission analysis. A latter part of this section will compare characteristics and associated performance (delta-V) of a LPR with that of a several lunar orbit rendezvous (LOR) trajectory profiles and a lunar surface rendezvous (LSR) profile. In particular, it contrasts the relative consistency of performance cost for the latter types with the varying costs associated with lunar orbit rendezvous profiles subjected to similar mission constraints and requirements.

Recent interest in the Moon as a stepping-stone for future robotic and human mission targets (e.g., Mars, asteroids) has revitalized the evaluation of concepts for establishing a sustained human presence on the Moon. An underlying assumed constraint for this mission profile is summed up in the phrase “anytime-abort from the lunar surface.” Specifically it is taken to mean that a flight crew faced with a life-support system failure or a medical emergency at the landing site should not have to wait for orbital planar alignment to initiate a lunar orbit departure (LOD) maneuver that will return them to Earth atmospheric entry and landing.

Mission modes to be compared include LOR, LPR, and LSR. Past work[1-5] indicates that LPR missions subjected to differing constraints possess a more consistent cost, in terms of required propulsive velocity increments (ΔVs), than Apollo-style LOR missions. In the case of LPR missions, this paper focuses on L1 rather than L2 (the translunar libration point), for a number of reasons.

8.1.1 The Earth-Moon Libration Points

There exist five equilibrium (libration) points in any two mass-body system (e.g., Earth-Moon, Sun-Earth, Sun-Mars). Of these five libration points, three collinear points (L1, L2, and L3) lie on the line between the mass bodies. The final two libration points (L4 and L5) create equilateral triangles with the two mass bodies. The distance of these points from the mass bodies (as shown in Figure 8.1.1-1 for the Earth-Moon system) is determined by the relative mass of those two bodies.

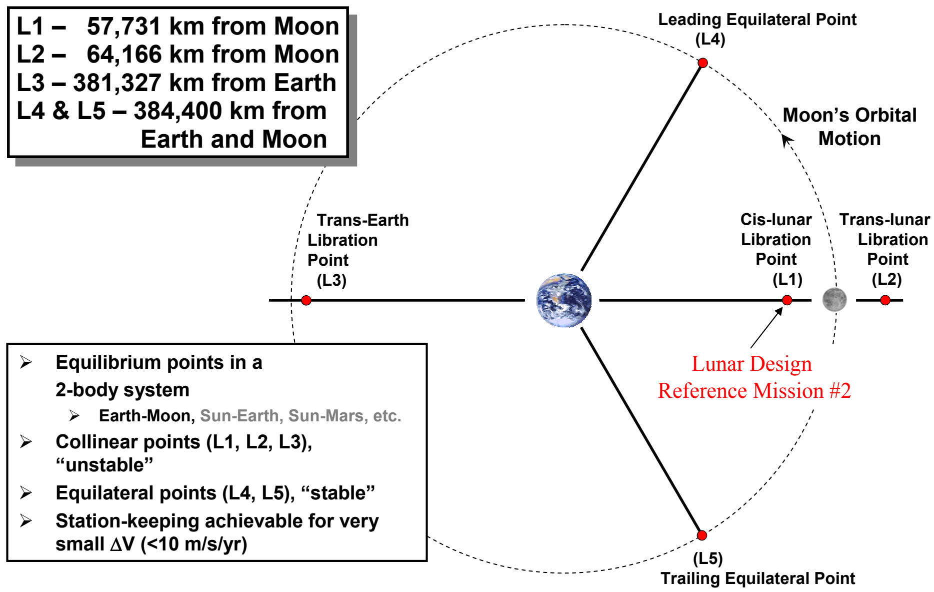


Figure 8.1.1-1: Earth-Moon Libration Points

The collinear libration points (or Lagrange Points) are "unstable", however past work[6,7] indicates that the delta-V (ΔV) cost to maintain orbit at these locations is low. This relative geometry is maintained as the Moon rotates about the Earth-Moon barycenter. The ratios of the distances of the collinear libration points from the Earth and Moon remain constant though the actual distance varies cyclically with the Moon's position in its slightly elliptical orbit.

8.1.2 Selection of the Libration Point for LPR (L1 vs. L2)

In general, the overall ΔV cost for a direct (Apollo-style) mission to lunar orbit is lower than for missions possessing stopovers at either of the two libration points (L1, L2) in lunar proximity (see Figure 8.1.2-1). While the L2 stopover mission has a slightly lower ΔV cost than that of an L1 stopover mission, its trip time is nearly double that of the L1 stopover mission.* The L2 stopover mission also requires an additional major maneuver during lunar swingby, without which the direct low Earth orbit (LEO) to L2 flight would possess a greater ΔV cost and a higher trip time requirement than the comparable flight to L1. For the direct libration point transfer case, an L1 target provides a lower ΔV cost than the L2 target. For the lunar swingby case, the L1 target results in a somewhat higher ΔV cost, but the trip time is nearly halved (see Figure 8.1.2-2). A spacecraft at L1 also has direct and continuous communication access to Earth. A halo orbit about L2 of sufficient size could provide continuous communication for a permanent facility parked in it[2], but it would introduce a 14-day cyclical variation in the ΔV required for injection into and departure from the halo orbit. This applies to transfers bound for and arriving from both the Earth and the Moon. The magnitude of the ΔV variation is on the order of 250 m/s

* Data from this chart was generated in past work[5] with different ground rules than will be used in this paper. However, the overall comparison is still useful and appropriate.

for a one-way transfer between Earth and the lunar surface by way of the halo orbit. The trade reference mission (TRM) for this study is a direct-to-L1 stopover mission.

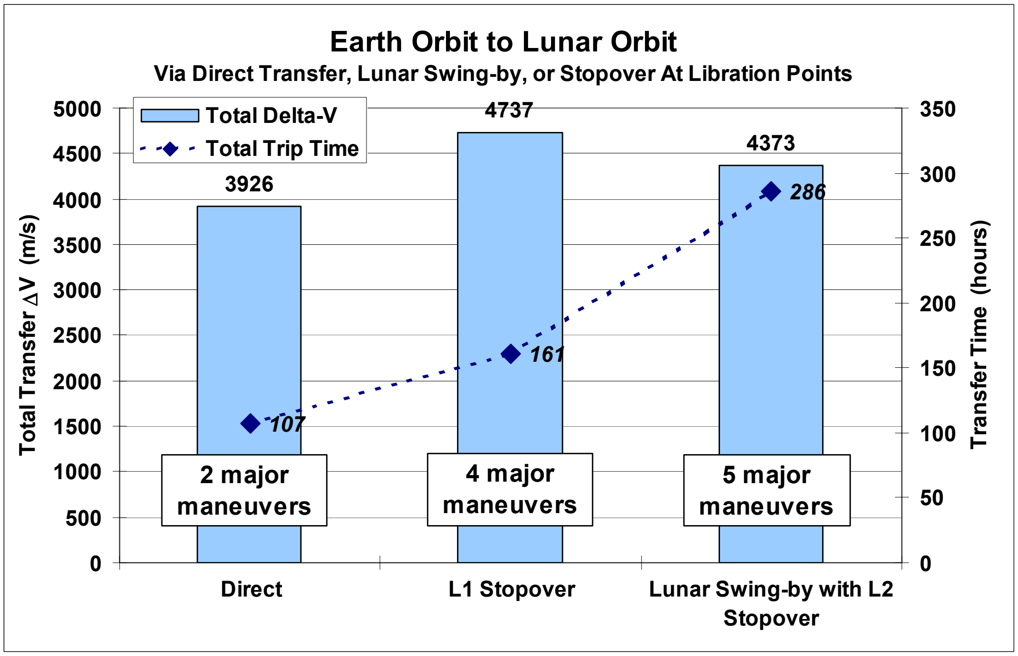


Figure 8.1.2-1: ΔV, Trip Time Comparison Of Earth-Moon Transfers Employing Either A Direct Transfer From Earth To The Moon Or One Including A Stopover At One Of The Collinear Earth-Moon Libration Points (L1, L2)

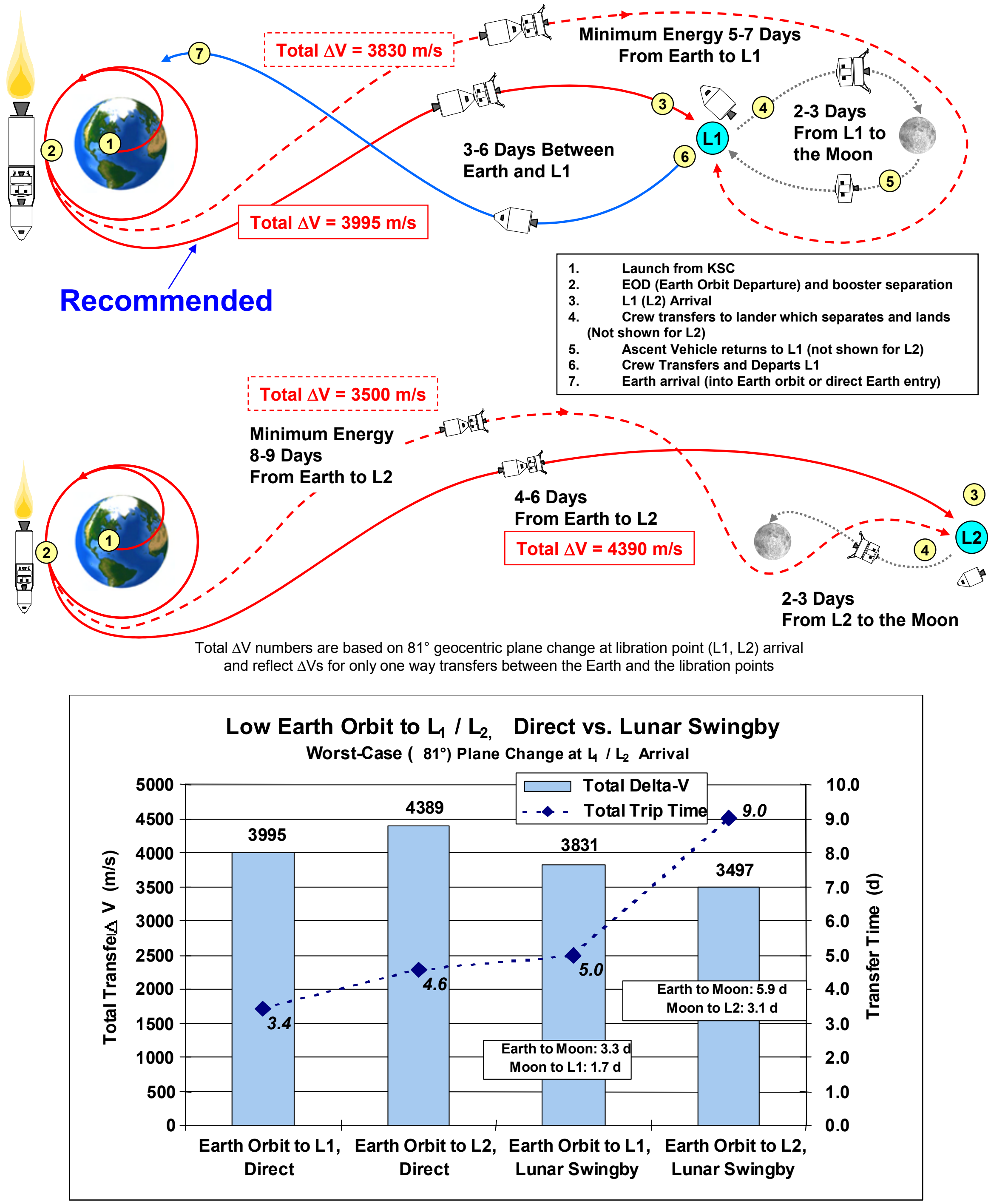


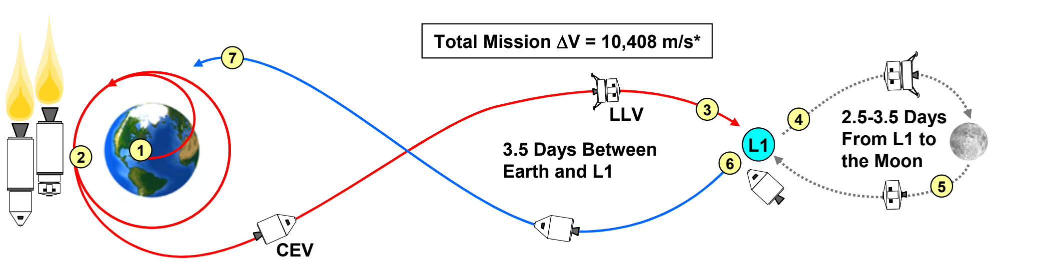
Figure 8.1.2-2: ΔV And Trip Time Comparison Of Transfers From Earth Orbit To L1 And L2 Via Either Direct Transfers Or Lunar Swing-By

8.2 Trade Reference Mission – Trajectory Profile Design Considerations

The TRM mission design objectives include assessment of the flight time, performance (ΔV), issues, mission options, mission parameter relationships, and additional required information for the nominal LPR mission profile. Another objective is to compare and contrast the nominal TRM with other mission profiles including the Apollo-style LOR and the LSR. The analysis of the TRM and comparison to other mission design approaches provides a basis for profile design recommendation.

The nominal trade reference mission consists of a 7-day surface stay with global lunar landing site access and an anytime abort from the lunar surface followed by an anytime return to Earth. The mission is comprised of two primary vehicles, a crew exploration vehicle (CEV) and lunar landing vehicle (LLV), both deployed to L1 from a 407 km circular, 28.5° Earth rendezvous orbit (ERO). The mission is accomplished via four total launches in sets of two with the first set consisting of an Earth departure stage (EDS) and the LLV and the second set consisting of another EDS and the CEV. Note that the mission profile design allows for a selectable right ascension of the ascending node (RAAN). This approach allows for a daily launch opportunities and can accommodate lighting constraints at lunar arrival.

The TRM consists of a LPR to a cislunar (L1) staging location (Figure 8.2-1). After an initial launch sequence places the LLV and its EDS in orbit for rendezvous, the mated configuration departs LEO to emplace the LLV at L1 where it awaits arrival of the crew on the crew exploration vehicle (CEV).



Mission Features

- **7-day surface stay** *(mission design allows for unlimited stay time)*
- **Global lunar landing site access**
- **Anytime return from lunar surface to L1**
- **Anytime return from L1 to Earth**
- **Daily Earth launch opportunities** *(coordinated multiple launches via ERO)*

Mission Phases

	Mission Phases
Outbound	1. Launch from KSC (1st set: EDS + LLV; 2nd set: EDS + CEV)
	2. EOD (Earth Orbit Departure) and booster separation
	3. L1 Arrival
Lunar Vicinity	4. Crew transfers to LLV which separates and lands on Moon
	5. Ascent Vehicle returns to L1
Inbound	6. Crew Transfers and Departs L1
	7. Earth arrival (into Earth orbit or direct Earth entry)

* Note: Mission ΔV is only a preliminary indicator of mission performance – Vehicle sizing based on mission ΔVs provides a more comprehensive metric

Figure 8.2-1: TRM consisting of a libration point (L1) rendezvous and staging to the lunar surface

Following emplacement of the LLV at L1, another Earth launch sequence delivers a 2nd EDS and the CEV to LEO for rendezvous. The CEV/EDS departs LEO bound for L1 and rendezvous with the awaiting LLV. After arrival at L1, the crew transfers from the CEV to the LLV and departs L1 in the LLV for the lunar surface. The LLV employs a 100 km altitude lunar phasing orbit to effect landing at a desired surface site. After a seven day stay, a lunar ascent vehicle departs the surface bound for L1, via a 100 km phasing orbit, and rendezvous with the CEV. At L1, the crew transfers from the LLV back to the CEV for return to Earth via a direct entry to a water landing.

The coordinated launch and far-field rendezvous sequence provides preferred Earth orbit orientation for departure. This approach provides for a missed Earth orbit departure injection opportunity to pre-emplace the LLV at L1. With a 2-week period between successive planned launches, a missed LLV departure could be recycled to launch (on the average) approximately 9-10 days later. A missed CEV departure opportunity however would mean a failed mission.

The variable length rendezvous profile provides for 360° of phase window for the LLV. For a 48-hour mission elapsed time, a constant length rendezvous profile provides 360° of phase window for the CEV. This 360° of phase window affords a daily launch opportunity for both the EDS/LLV and the EDS/CEV rendezvous profiles. The details of the rendezvous will be discussed in section 8.2.2.

While this mission design accommodates a 7-day stay on the lunar surface with anytime abort, longer surface stays are available with this approach, for the same ΔV cost. The fundamental mission profile remains consistent for all surface stay times, allowing for standardized mission design. A drawback to this approach is that, in the event of a required immediate abort from the lunar surface, the crew is still 2-3 days away from their backup habitat (the CEV at L1).

The ΔV costs of the individual flight phases for the TRM flight profile are shown in Figure 8.2-2. The total nominal mission ΔV cost of 10,408 m/s includes all maneuvers between the Earth and L1, L1 and the Moon, and powered lunar descent and ascent. The flight times between the Earth and L1 (both outbound and inbound) lie in the vicinity of 3.5 days. The flight times for transfers between L1 and lunar phasing orbit range from 2-3.5 days, depending on the landing site. These lunar transfer flight times are optimized to produce the minimum ΔV cost for landing at

Nominal LPR Mission

ΔV Cost By Flight Phase

Mission Features / Flight Phase	Maneuver ΔV (m/s): 28.5 Deg ERO Lch Expendable Lander Global Access Surface Stay = 7 days
LPD2	800
LP Rendezvous - Dep	100
LPA2	241
LOD	631
LO Rendezvous - Dep	0
Ascent to Lunar Orbit	1834
Descent to Landing Site	1881
LO Rendezvous - Arr	0
LOI	631
LPD1	244
LP Rendezvous - Arr	100
LPA1	889
EOD	3057
TOTAL	**10408**

| Beginning of Mission → End of Mission |

Figure 8.2-2 Trade Reference Mission (TRM) flight phase ΔV costs.

any surface site. Note that these performance data represent worst-case or conservative performance allowing the mission to support lunar landing at any location (latitude or longitude). It also supports all lunar orbit inclinations (with respect to the Earth equator) over the 18.6 year lunar inclination cycle in addition to worst-case lunar arrival/departure distances (perigee) and angular distances of the Moon from the Earth equator (e.g., ranging from nodal crossing to orbit apex). The performance data do not include Earth orbit departure (EOD) injection window, gravity losses, and performance reserves.

8.2.1 Earth Departure Window

<u>LEO to L1</u>

The Earth departure injection window is dependent upon the initial altitude of the LEO parking orbit, the initial orbit inclination with respect to the earth equator, the inclination with respect to the Earth-Moon plane, and the flight time from LEO to L1. For this study, a 24-hour injection window was selected. The total ΔV cost for a two maneuver sequence (Earth departure and L1 arrival) is shown in Figure 8.2.1-1a with the associated Earth departure cost shown in Figure 8.2.1-1b. The flight time for this 24-hour injection window is shown in Figure 8.2.1-1c. This injection window assumes a 407 km circular parking orbit with a 28.7° inclination. A nominal (82-hour) transfer time reflects a near minimum ΔV cost for a bounded case of spacecraft arrival at L1 coincident with lunar perigee. The ΔV cost of a 1-day (24-hour) injection window depends upon the constraints on outbound flight and L1 arrival times. For example, demanding a consistent 82-hour flight time from LEO to L1 throughout the 24-hour injection window results in a total ΔV cost of 1485 m/s. Requiring a constant L1 arrival time reduces this cost to 339 m/s. A full release of the flight time constraint results in a 52 m/s total ΔV cost for a 24-hour injection window. For this recommended case, the flight time for a departure at the opening of the window is about 97 hours and about 67 hours for a departure at the close of the window. This represents about a 6-hour variation in L1 arrival time over the 24-hour injection window.

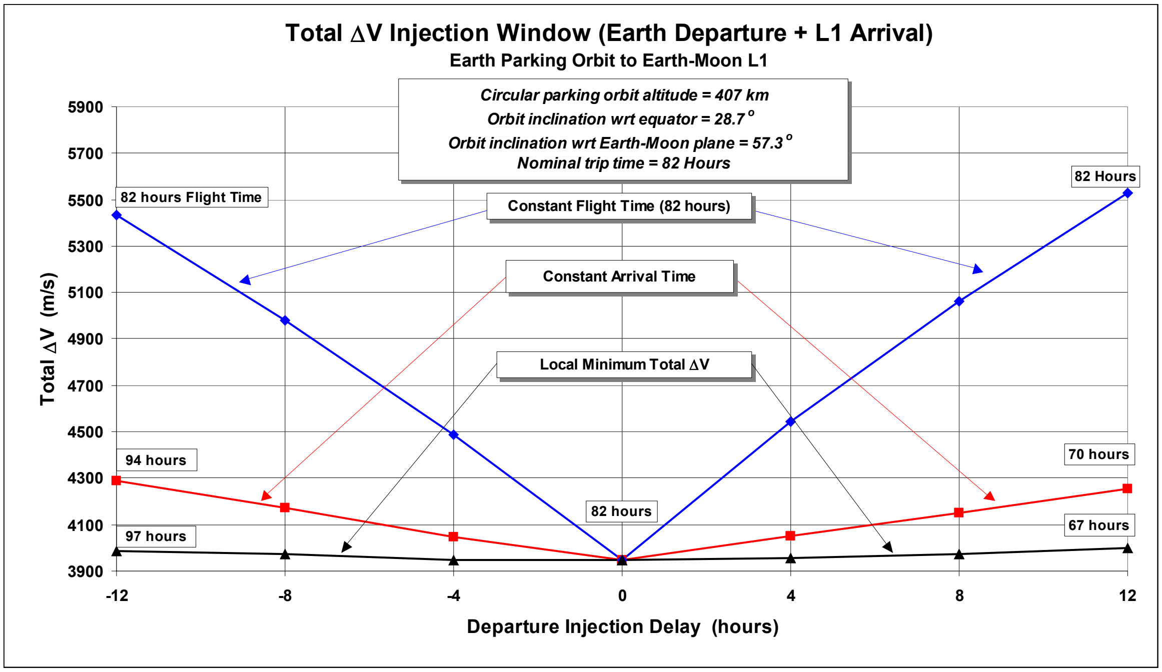


Figure 8.2.1-1a: Total ΔV Cost Versus Earth Departure Time For A Twenty-Four (24) Hour Earth Departure Injection Window

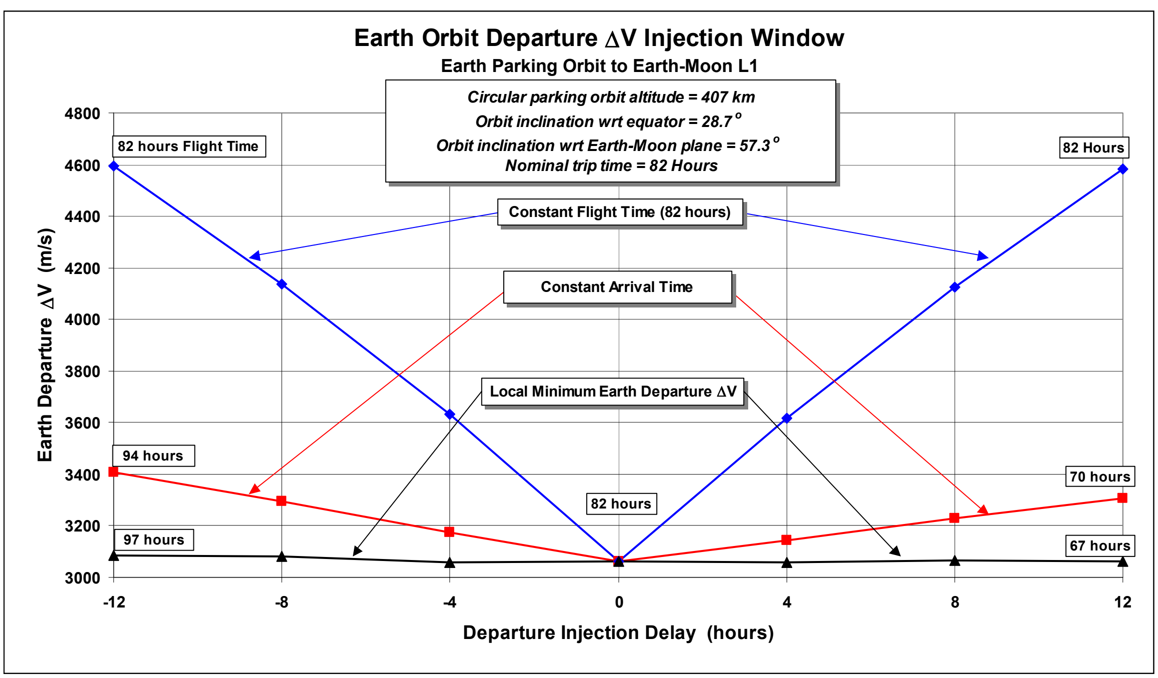


Figure 8.2.1-1b: Earth Departure ΔV Cost Versus Earth Departure Time For A Twenty-Four (24) Hour Earth Departure Injection Window

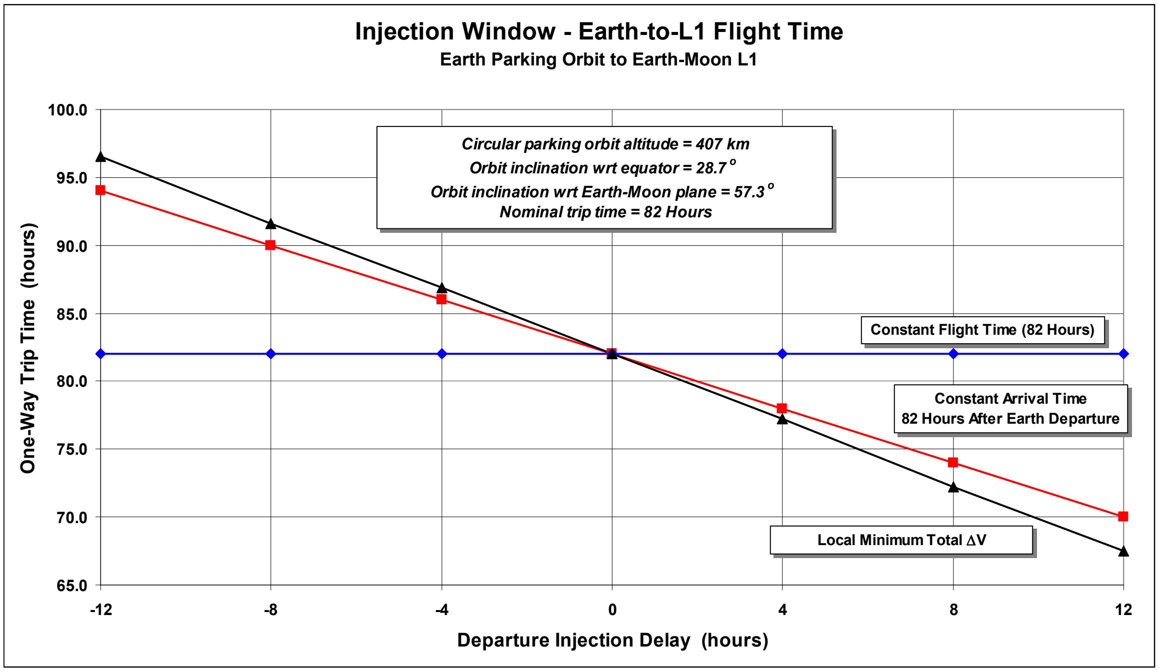


Figure 8.2.1-1c: Flight Time Versus Earth Departure Time For A Twenty-Four (24) Hour Earth Departure Injection Window

8.2.2 Earth Orbit Rendezvous

Introduction

The LDRM-2 trade reference mission plan assumes four launches from Cape Canaveral and two LEO rendezvous profiles. For the first rendezvous case, an EDS will be launched, followed by a Lunar Lander. The Lander will perform the rendezvous with the EDS. In the second rendezvous case, an EDS will be launched, and the Crew Exploration Vehicle (CEV) will rendezvous with it. In general, the orbital mechanics problems associated with effecting a Lander rendezvous with the EDS are similar to the problems associated with effecting a CEV rendezvous with the EDS. In this document, these problems will be discussed collectively. Then, when it is necessary to differentiate between the rendezvous profiles, data that are unique to each will be included in the discussion.

Assumptions and Limitations

In order to design the two rendezvous profiles, the following assumptions and limitations have been made.

1. Expendable Launch Vehicle (ELV)

   a. The on-orbit engines have multiple restart capability.

b. Since the required Longitude of the Ascending Node (LAN) varies with phase angle through the launch window, the Guidance, Navigation, and Control (GN&C) accommodates a variable LAN target parameter.

c. The ELV will perform two on-orbit translational maneuvers to place the Lander into a low circular Earth orbit.

d. The ELV will perform two on-orbit translational maneuvers to place the CEV into a low circular Earth orbit.

2. The EDS is the rendezvous target vehicle.

3. Lunar Lander

   a. The Lander will perform a double coelliptic rendezvous maneuver profile.

   b. The Lander GN&C will accommodate a variable length rendezvous profile.

4. Crew Exploration Vehicle

   a. The CEV will perform a stable orbit rendezvous maneuver profile.

   b. Crew operational constraints will preclude a variable length rendezvous profile for the CEV

5. Translational maneuver dispersions are not considered.

6. Lighting constraints are not considered.

7. Communications constraints are not considered.

Initial Conditions

By definition, the initial orbital conditions are as follows.

– EDS Target Orbit	: Apogee	=	407 km
	: Perigee	=	407 km
	: Inclination	=	28.7 deg
– ELV Insertion Orbit	: Apogee	=	176 km
	: Perigee	=	93 km
	: Inclination	=	28.7 deg

Launch Window Considerations

A rendezvous launch window is defined by the overlay of the planar window and the phase window, with the additional overlay of the launch operational constraints. In general, the launch site will be in the target vehicle plane twice a day. These two times are defined to be the inplane lift-off times: one on a northerly azimuth, one on a southerly azimuth. The launch vehicle performance will be maximized at the inplane time. In order for the launch vehicle to get into the plane of the target vehicle by the time of insertion, liftoff at any other time will require the launch vehicle to perform a plane change during ascent. Thus, the launch vehicle performance will degrade. The length of the plane window is determined by the ascent vehicle performance capability (i.e., by how much additional propellant is available for the plane change). The length of the plane window is also dependent upon the inclination of the target orbit. In general, the higher the inclination, the smaller the plane window becomes. The maximum performance (i.e., the longest plane window) is achieved when the launch site latitude is slightly less than the orbital inclination of the target vehicle orbit.

A phase window is defined to be the range of phase angles that, given particular mission constraints, will allow a rendezvous to occur with minimum on-orbit performance costs. The length of the phase window is determined by the rendezvous maneuver profile and by the minimum and maximum time allowed for the completion of the rendezvous. The opening of the phase window is defined by the smallest phase angle that will effect a rendezvous. The closing of the phase window is defined by the largest phase angle that will effect a rendezvous. If the rendezvous maneuver sequence and the maximum rendezvous time account for 360 degrees of phasing, then the phase window is no longer a launch constraint. In this instance, the planar window will determine the length of the launch window, and launch may occur every day.

Lander Rendezvous Mission Profile

Fourteen days after the EDS1 is placed into a 407 km circular orbit, with an inclination of 28.7 degrees, the ELV/Lander is launched into LEO. This particular launch spacing and rendezvous orbit altitude reflect assumptions made for the trade reference mission. Since the Lander is unmanned, it will execute a double-coelliptic rendezvous sequence. This sequence will allow the Lander, during the near-field phase of the rendezvous, to slowly approach the EDS1 while evaluating the relative motion data. In addition, assuming that the Lander remains within its 3-σ performance error ellipse during this phase, the maneuver sequence will preclude the unintentional intersection of the EDS1 and Lander trajectories.

Two rendezvous mission profiles are defined here: one for the opening of the phase window and one for the closing of the phase window. In the opening of the phase window, the phase angle at insertion is approximately 64 degrees. Following insertion, the ELV will perform two translational maneuvers that will place the Lander into a 375 km circular orbit. Once the Lander has separated from the ELV, the Lander will initiate an eight maneuver rendezvous sequence, with braking occurring at approximately 48 hours mission elapsed time (MET). The two ELV translational maneuvers will cost approximately 141 meters per second (mps). The eight Lander translational maneuvers will cost approximately 19 mps.

In the closing of the phase window, the phase angle at insertion is approximately 424 degrees. Following insertion, the ELV will perform two translational maneuvers that will place the Lander into a 200 km circular orbit. Once the Lander has separated from the ELV, the Lander will initiate an eight maneuver rendezvous sequence, with braking occurring at approximately 51 hours MET. The two ELV translational maneuvers will cost approximately 40 mps. The eight Lander translational maneuvers will cost approximately 122 mps. Note that the total delta velocity magnitude for both sequences is approximately 160 mps. Also note that the time of rendezvous in the closing case is approximately 3 hours later than in the opening case. Given the assumed initial conditions, this increase in the time of rendezvous is necessary if 360 degrees of phase angle are to be covered.

CEV Rendezvous Mission Profile

Fourteen days after the EDS2 is placed into a 407 km circular orbit, with an inclination of 28.7 degrees, the ELV/CEV is launched into LEO (launch spacing and rendezvous orbit parameters are TRM assumptions). It is assumed that the CEV will execute a stable orbit rendezvous sequence. This sequence will allow the CEV to stop at two stable orbit points in the sequence, if the trajectory conditions warrant it.

Two rendezvous mission profiles are defined here: one for the opening of the phase window and one for the closing of the phase window. In both cases, following insertion, the ELV will perform two translational maneuvers that will place the CEV into a 200 km circular orbit. In the opening of the phase window, the phase angle at insertion is approximately 48 degrees. Once the CEV has separated from the ELV, the CEV will initiate an eight maneuver rendezvous sequence, with braking occurring at approximately 47 hours MET. The two ELV translational maneuvers will cost approximately 40 mps. The eight CEV translational maneuvers will cost approximately 126 mps.

In the closing of the phase window, the phase angle at insertion is approximately 447 degrees. Following insertion, the ELV will perform two translational maneuvers that will place the CEV into a 200 km circular orbit. Once the CEV has separated from the ELV, the CEV will initiate an eight maneuver rendezvous sequence, with braking occurring at approximately 47 hours MET. The two ELV translational maneuvers will cost approximately 40 mps. The eight CEV translational maneuvers will cost approximately 122 mps. Note that the total delta velocity magnitude for both sequences is approximately 160 mps.

Trajectory Event	Delta Velocity Magnitude of Translational Maneuvers (meters per second)	Mission Elapsed Time from ELV Launch (Days: hours: minutes)
Opening of Lander Phase Window		
ELV: Insertion through Circularization	141	0:01:39
Lander: Phasing through Theoretical Braking	19	1:17:43
Closing of Lander Phase Window		
ELV: Insertion through Circularization	40	0:01:39
Lander: Phasing through Theoretical Braking	122	2:02:29
Opening of CEV Phase Window		
ELV: Insertion through Circularization	40	0:01:39
Lander: Phasing through Theoretical Braking	126	1:22:59
Closing of CEV Phase Window		
ELV: Insertion through Circularization	40	0:01:39
Lander: Phasing through Theoretical Braking	125	1:22:47

Note: At the time of rendezvous initiation, the EDS is in a 407 km circular LEO.

Table 8.2.2-1: Rendezvous ΔV Cost and Mission Elapsed Time

Conclusions

The following conclusions have been made:

1. A variable length rendezvous profile will provide a 360 degree phase window for the Lander.
2. Assuming that the time of theoretical braking occurs near 47 hours mission elapsed time, a constant length rendezvous profile will provide a 360 degree phase window for the CEV.
3. Since the rendezvous profiles provide for a 360 degree phase window, they provide a daily launch opportunity.

8.2.3 Post-EOD Stage Disposal

This section addresses the question of where and how to dispose of Earth Departure Stages (EDS) used to transfer the CEV and LLV from LEO to the Earth-Moon L1 (cislunar) libration point. Performance results are based on past work supporting a lunar gateway study conducted at JSC[9]. In particular, this section assesses the delta-V (ΔV) cost to retarget an Earth-Moon L1-bound spent EDS to a selected disposal destination.

LPA Time Frame for Evaluation of Performance Requirements

A two-week period in October of 2006 was chosen for an evaluation timeframe. As indicated in Table 8.2.3-1, libration point arrivals in that period exhibit a near-maximum variation in LPA plane change requirement (Xfr Orbit iEMP). This period begins with the Moon simultaneously very near perigee and its ascending node on Earth's equator, and ends with the Moon very near apogee and its descending node. It can be seen that an aggregate of 22 launch opportunities were examined. Libration point arrivals at perigee were combined with minimum and maximum LPA plane change angles, and likewise for arrivals at apogee, with a variety of combinations between these extremes.

SWW:dmd

Earth-to-LL1 Transfer and Upper Stage Disposal Data

11-May-02

All transfers involve coplanar departure from circular earth parking orbit having an altitude of 407 km and an inclination of 51.6 deg

GO,DROA, LVI, HO, and SROA maneuver times selected to minimize delta-v for stage disposal

Lunar L1				Earth	Northerly LL1 Arrival Azimuth										Southerly LL1 Arrival Azimuth									
Arr Time	RA	Decl.	Dist.	Park Orbit	Park	Xfr	Maneuver Delta-V, m/s								Park	Xfr	Maneuver Delta-V, m/s							
(Nominal)			1000	RAN Epoch	Orbit	Orbit	EOD	LPA	GO	DROA	LVI		HO	SROA	Orbit	Orbit	EOD	LPA	GO	DROA	LVI		HO	SROA
2006 Oct	deg	deg	km	2006 Oct	RANo	iEMP	MC	MC	MC	MC	MC	OC	OC	OC	RANo	iEMP	MC	MC	MC	MC	MC	OC	OC	OC
10/6/06 4:00	-1.0	-0.1	304	10/2/06 16:00	-1.0	23.7	3061	782	52	50	87	88	66	106	178.9	81.0	3060	984	91	55	104	106	87	124
10/7/06 4:00	12.4	7.2	304	10/3/06 16:00	6.7	24.0	3059	784	59	45	87	88	66	106	198.1	79.8	3061	980	91	55	105	106	88	126
10/8/06 4:00	26.2	14.0	305	10/4/06 16:00	14.7	24.3	3060	781	61	42	87	88	65	111	217.7	75.9	3059	960	90	55	106	106	87	128
10/9/06 8:00	42.9	20.7	309	10/5/06 20:00	25.0	28.7	3060	781	65	43	93	94	71	117	240.9	68.2	3059	916	87	55	107	108	87	131
10/11/06 0:00	68.1	27.1	317	10/7/06 12:00	43.0	35.0	3063	776	63	53	101	101	78	126	273.2	54.4	3064	838	77	58	109	109	86	132
10/12/06 8:00	88.5	28.7	324	10/8/06 20:00	61.2	44.0	3063	787	62	59	110	109	86	132	295.8	44.3	3063	786	61	62	110	109	87	132
10/13/06 18:00	109.2	27.2	332	10/10/06 6:00	84.0	55.2	3066	810	59	61	115	115	92	135	314.5	35.9	3066	748	33	69	109	109	83	129
10/15/06 18:00	135.4	20.7	339	10/12/06 6:00	117.6	69.2	3071	851	61	58	117	118	96	134	333.3	28.0	3070	726	7	83	107	107	83	124
10/17/06 4:00	151.8	14.0	343	10/13/06 16:00	140.3	75.4	3072	875	63	53	116	117	95	132	343.4	24.9	3072	724	5	89	105	106	82	120
10/18/06 12:00	166.2	6.9	344	10/15/06 0:00	160.7	78.7	3074	890	65	51	115	117	95	132	351.8	23.3	3073	727	10	92	104	105	82	120
10/19/06 18:00	179.2	-0.1	345	10/16/06 6:00	179.3	80.1	3074	900	66	49	114	117	94	131	359.2	23.3	3073	733	11	93	104	106	81	121

RA Right Ascension
RAN Right Ascension of Ascending Node
RANo Right Ascension of Ascending Node at RAN Epoch
iEMP Inclination of Xfr Orbit wrt Earth-Moon Plane
EOD Earth Orbit Departure to L1 Lunar Libration Point
LPA Libration Point Arrival (3.5 days after EOD)
GO Upper Stage Disposal in "Safe" Geocentric Orbit (6600 km Perigee Alt, 300000 - 370000 km Apogee Alt)
DROA Upper Stage Disposal in Remote Ocean Area (Direct,20 deg Atmospheric Entry Angle, 240 deg Longitude Spread)
LVI Upper Stage Disposal on Lunar Surface (Vertical Impact)
HO Upper Stage Disposal in Heliocentric Orbit (via Lunar Swingby)
SROA Upper Stage Disposal in Remote Ocean Area (via Lunar Swingby)
OC Overlapped Conic Trajectory
MC Multi-Conic Trajectory

Table 8.2.3-1: Earth Departure Stage Disposal Cost

EDS Disposal Options

After execution of the EOD maneuver, the Earth Departure Stage and CEV share a trajectory having a perigee altitude near that of the pre-departure orbit, an apogee altitude equal to that of the libration point and, significantly, an orbit orientation and energy comparable to that required for reaching the near vicinity of the Moon. These circumstances immediately bring to mind the trajectory design problems solved in the Apollo program, wherein the trans-lunar injection (TLI) or EOD maneuver routinely put the Command & Service Module/Lunar Module on or very near a trajectory that provided a free return to the Earth, and the spent Saturn S-IVB stage was variously diverted (after TLI) onto trajectories that ended in heliocentric orbit or with lunar impact. Accordingly, the following options for upper stage disposal were selected for evaluation:

1. Lunar Swing-by to Heliocentric Orbit (HO)

2. Lunar Vertical Impact (LVI)
3. Direct Return to Remote Ocean Area (DROA)
4. Lunar Swingby to Remote Ocean Area (SROA)
5. Transfer to Long Lifetime Geocentric Orbit (GO)

As illustrated in Figure 8.2.3-1, in a real-world situation where the ascending node of the Earth parking orbit is regressing under the natural influence of Earth oblateness, at most a couple of launch opportunities can occur during any two-week interval. In this study, for the purpose of sampling all combinations of lunar distance and LPA plane change angle, the ascending node location for the LEO orbit (the columns labeled "Park Orbit RANo" in Table 8.2.3-1) was treated as an arbitrary parameter that could be changed at will.

In terms of propulsive ΔV, it was found that sending the upper stage to any Earth atmosphere entry point via close encounter with the Moon (SROA) is the most expensive of all the disposal modes studied (Figure 8.2.3-1). On the other hand, guaranteed direct return to a mid-ocean line (DROA) is cheaper than lunar impact (LVI) or heliocentric orbit (HO) disposal, and probably cheaper than disposal in any geocentric orbit (GO) having an adequate lifetime to satisfy public safety concerns.

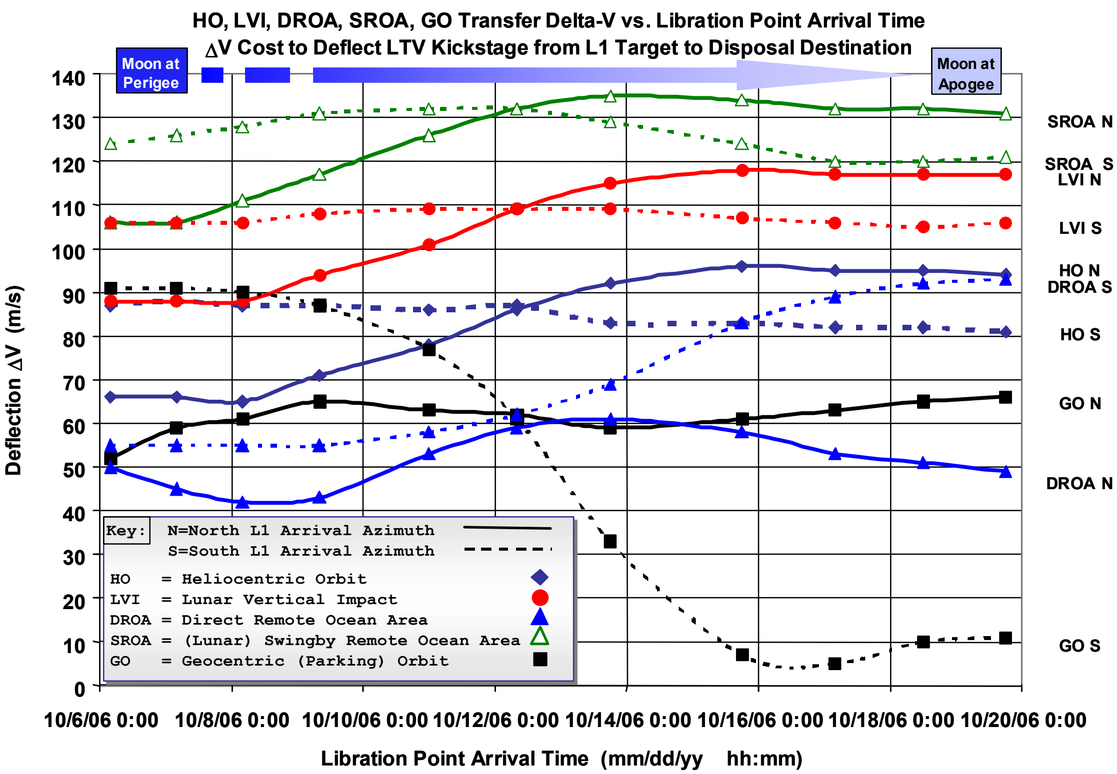


Figure 8.2.3-1: Summary of Disposal Maneuver Deflection ΔV's

The DROA disposal option appears to offer the best suite of desirable features. It provides for controlled Earth contact using a relatively small disposal ΔV. It also avoids a close encounter with the Moon, which would result in not only a greater ΔV cost, but also a near certainty of requiring midcourse correction maneuvers before and after perisel passage. In addition, DROA avoids littering the lunar surface and geocentric space with debris. However, this approach would not serve well for cases where the EDS kickstage contains hazardous (e.g., radioactive) materials. In that case the "next best" disposal option (HO) avoids Earth or lunar disposal issues (e.g., impact location, debris footprint, litter) by taking the EDS around the Moon and into heliocentric space. This approach also carries a relatively low ΔV cost. Further study would be required for this disposal option to determine the probability of subsequent re-contact with the Earth, and the cost of precluding such an event. The 100 m/s stage disposal ΔV budget, used for the LDRM-2 study, provides for either a DROA or a HO stage disposal. If a lunar vertical impact (LVI) disposal approach was desired, a slightly higher stage disposal budget (120 m/s) would be required.

8.2.4 Earth to L1 Transfer - Nominal And Aborted Mission

8.2.4.1 Nominal Mission

The nominal mission design is an Earth departure from a 407 km circular orbit inclined 28.7° and an arrival at L1. The two major ΔV maneuvers are at Earth departure and L1 insertion. The flight time from Earth to L1 depends on the ΔV capability of the injection stage(s) and on mission operation constraint(s). For minimum energy transfers, the flight time ranges from about 3.5 to 5 days depending on if a lunar flyby is or is not performed. Compared to a direct transfer to L1, a trajectory that includes a lunar flyby will lower the ΔV requirement while increasing the flight time.

Assumptions

The ΔV requirement is based on a worst case Earth-Moon geometry and for a direct transfer (no lunar flyby). This geometry occurs when the Moon is at perigee during L1 arrival and the Moon's orbit inclination is at its maximum of 28.6°. The maximum wedge angle at L1 insertion is 57.3°, which is the summation of the inclination of the Earth parking orbit and the Moon's orbit inclination. This occurs when the right ascension of the ascending nodes of these orbits are 180° apart. The Earth departure is near coplanar (with the Earth parking orbit) in order to minimize the departure ΔV.

Results

The nominal Earth departure ΔV is about 3058 m/s with an L1 insertion ΔV of about 887 m/s. These numbers are derived for an Earth departure date of 2006 October 2 18:00 and a flight time to L1 of 3 days 10 hours. The following figures show how the Earth departure, L1 insertion, and total ΔVs change with flight time for the two extreme cases of the Moon at apogee or perigee. In Figure 8.2.4.1-1 for flight times of 80 hours or more, the difference between Earth departure ΔVs is less than 20 m/s. The difference between the L1 insertion ΔVs is between 50 to 190 m/s. In Figure 8.2.4.1-2 for flight times less than 77 hours, the total ΔV is greater when the moon is at apogee than at perigee.

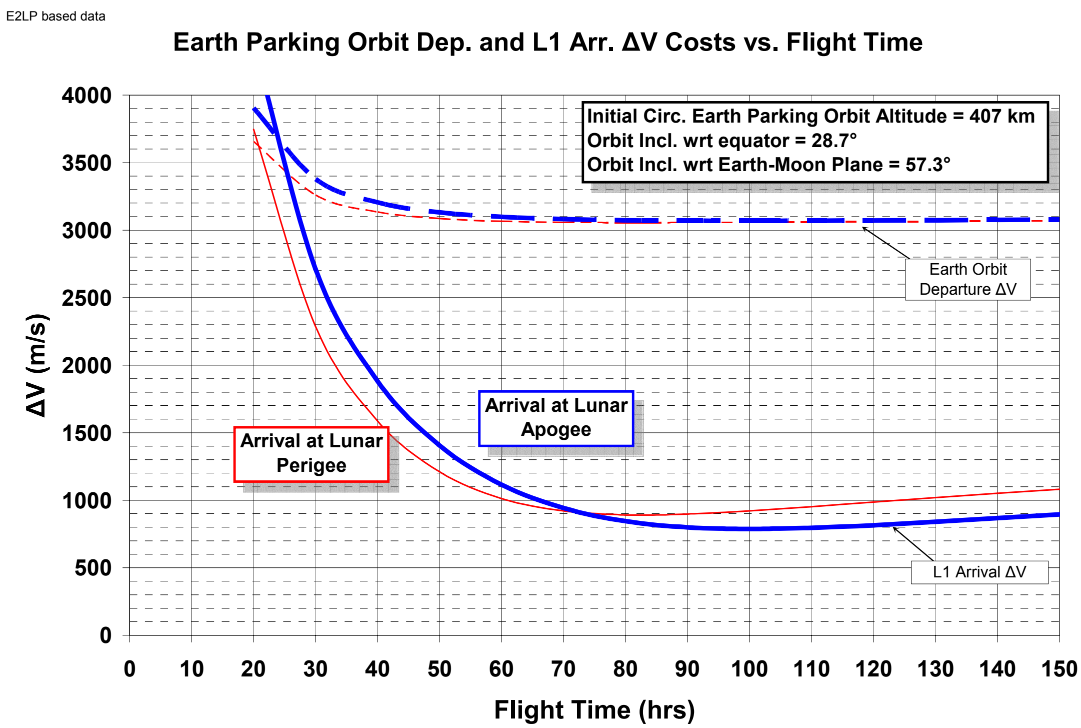


Figure 8.2.4.1-1: Earth Departure and L1 Insertion ΔV

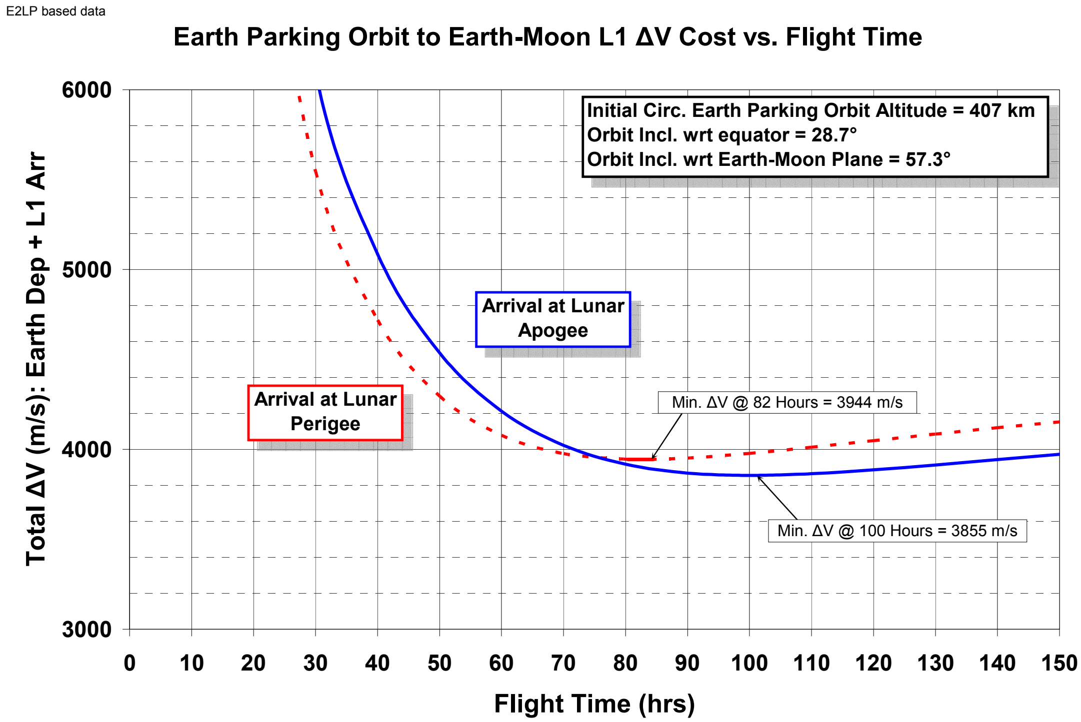


Figure 8.2.4.1-2: Total Outbound ΔV

8.2.4.2 Aborted Mission

During the transfer to L1, a mission abort may be necessary depending on the situation (e.g., a propellant tank that becomes unusable) and when it occurs. For flights with a crew, it may be necessary to target for shorter return flight times.

Assumptions

The abort ΔV maneuver is computed based on a nominal Earth departure and unexecuted L1 insertion. The nominal target is a direct entry at Earth with no particular landing site. For a worst case geometry, the moon is at perigee and has an orbit inclination of 28.6º. Also, the nominal L1 insertion requires the worst case plane change of 57.3°.

Results

A scan space was created for an abort execution time of ±24 hours from the nominal L1 arrival time and a return flight time ranging from 2 to 7 days. The minimum abort ΔV ranges from 20 to 30 m/s for a favorable Earth return flight time associated with an abort time. The abort ΔV is sensitive to the return flight time, especially when flight times are shortened. The longer return flight times are presented for completeness. If the mission is aborted about 2 days or more into the flight and the nominal L1 insertion ΔV is available, a 2.5 day return flight time is feasible. For missions with a crew, the available ΔV budgeted for the return to Earth will enable shorter return flight times.

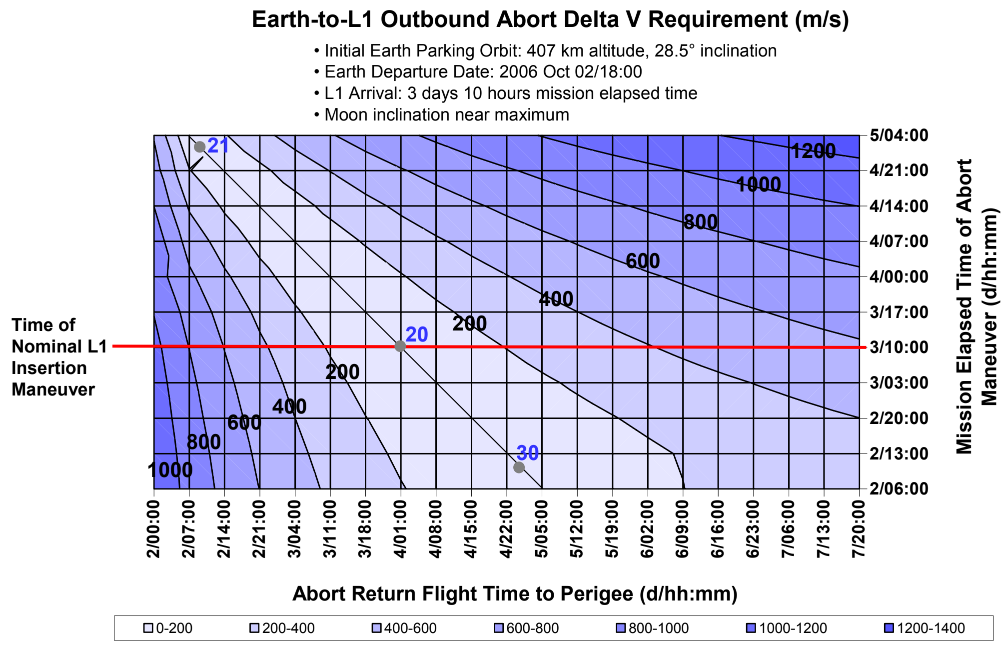


Figure 8.2.4.2-1: Abort ΔV

8.2.5 Rendezvous at L1

In this section the sequence of a L1-rendezvous trajectory is described. This section is a proof of concept of what the actual trajectory might be. The transfer consists of sequence of several pair of burns; after each one the distance to the target is fixed and the target-chaser relative velocity is zero. The distance to the target and the number of burn pairs, are parameters that will be determined by the details of mission. These details include error dispersion analysis data, availability of navigation information, and safety constraints among others.

The idea behind this transfer is to get close enough to the target after the Earth departure burn (1st burn) so the relative navigation can be used for the 2nd and subsequent burns. In Figure 8.2.5-1 an overview of a possible configuration for the L1-rendezvous is shown. After burns 2, 4 and 6 the distance to the target is reduced until the final distance for proximity operations is reached. In the present configuration the chaser is always traveling behind the target. Although it not shown in this report, a similar solution to the one the presented here where the chaser travels ahead of the target can be obtained. The performance of this other solution is very similar to the one presented in this report.

In the next sections the numerical results of the study together with the assumptions and operational constraints considered are described. Finally, a short section describing deviations from the nominal case has been included; this section takes into account two possible failures in burns 2 to 6.

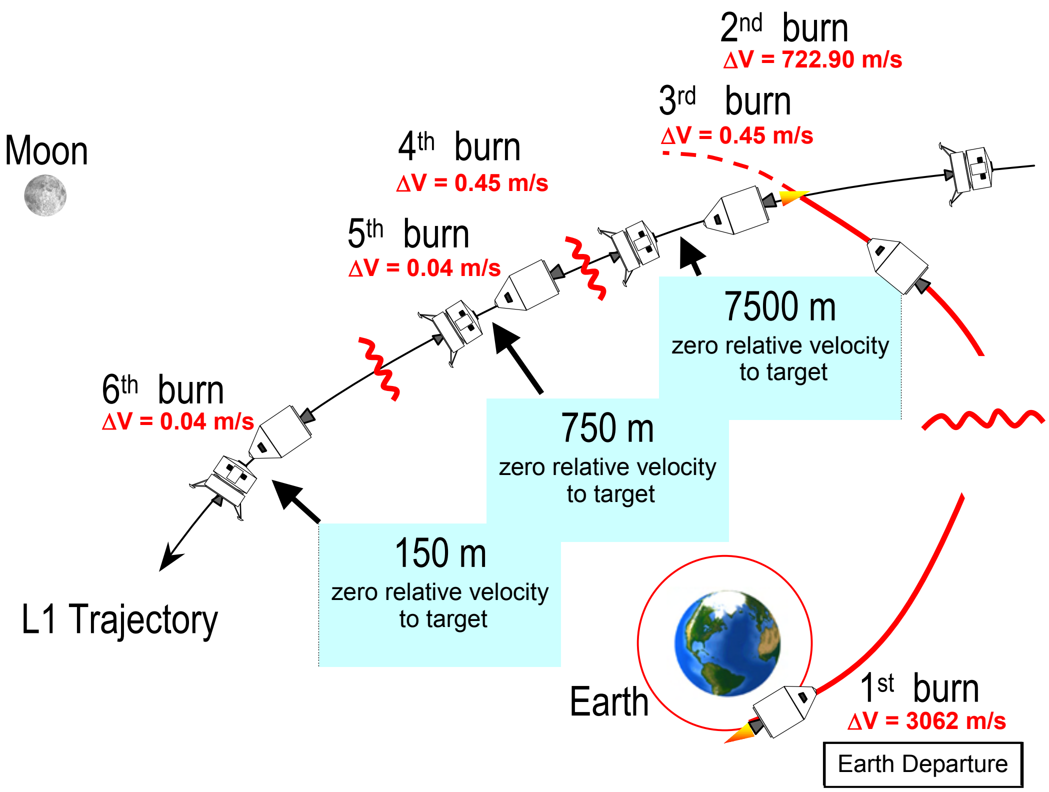


Figure 8.2.5-1: Overview of the L1 Rendezvous

In order to perform this study the following assumptions were made:

– Dynamic model: Elliptic Restricted Three Body Problem (ERTBP)[11] with lunar orbital elements:

a	= 0.386478491D+06	: semi-major axis (km)
e	= 0.458577647D-01	: eccentricity
I	= 0.215675467D+02	: inclination (deg)
Ω	= 0.372907832D+03	: ascending node (deg)
ω	= 0.115539088D+03	: argument of periapsis (deg)
f	= 235.00	: true anomaly at epoch (deg)

o No four body effects have been considered for this case (i.e., Sun).

    - No perturbations have been considered (i.e., solar radiation or plasma environment).
- Operational constraints
    - In order to account for the navigation information to be ready the minimum time between burns has to be more than 15min. (this applies to burns 2, 3, 4, 5 and 6).
    - The time of flight <=4 days. A typical TOF for a mission like this, where the total ΔV is minimized and only two burns, is about 3.5 days[10] therefore 12h are used for the final rendezvous maneuvers (burns 3, 4, 5 and 6).
    - The number of burns is set to 6. This is only for this report, the final number of burns will be determined by the specific details of the mission.
    - The chaser will coast behind the target if no burns are performed (this is only valid if the 2nd burn has been performed correctly, that is for burns 3, 4, 5 and 6). This situation is only temporary, due to possible perturbations the chaser will eventually drift away (if no station-keeping maneuvers are performed).
    - To avoid collisions during the rendezvous maneuvers the distances to target after each pair of burns are (zero relative velocity between target and chaser is also considered):
        - 7.5km (in this way relative navigation can be used from the second burn to the end of the rendezvous)
        - 0.75km
        - 150m (in this point proximity operations will start).
    - To avoid a possible collision before capture and docking the final burn<10cm/s.
    - Parking orbit: 407x407 (28.5 deg. Inclination).
- The final goal is to minimize the total ΔV.
- The study spans for 28 days. Since no disturbances and real ephemeris are considered the solutions will repeat after 28 days (approx.).

Results

For the nominal mission, a 28-day simulation with the assumptions above has been performed. In Figure 8.2.5-2 a nominal transfer obtained with Copernicus is shown. The transfer is composed of six burns, a first major burn departing the Earth and then a sequence of five burns than completes the rendezvous. For the example in Figure 8.2.5-2 the results are summarized in the next table:

ΔV1 (m/s)	3.0621E+3	ΔT1 (1^{st} -2^{nd}) (day)	3.60		
ΔV2 (m/s)	7.229 E+2	ΔT2 (2^{nd} -3^{rd}) (day)	0.01	Distance to Target after 2^{nd} burn (m)	7552.64
ΔV3 (m/s)	4.5 E-1	ΔT3 (3^{rd} -4^{th}) (day)	0.17		
ΔV4 (m/s)	4.5 E-1	ΔT4 (4^{th} -5^{th}) (day)	0.01	Distance to target after 4^{th} burn (m)	757.01
ΔV5 (m/s)	3.5 E-2	ΔT5 (5^{th} -6^{th}) (day)	0.20		
ΔV6 (m/s)	3.5 E-2			Distance to Target after 6^{th} burn (m)	152.11

Table 8.2.5-1: Results for a L1-Rendezvous example

In Figure 8.2.5-3, the total ΔV for the 28-day simulation is shown (bottom figure) together with the distance and velocity of L1 with respect to the Earth at arrival (top figures).

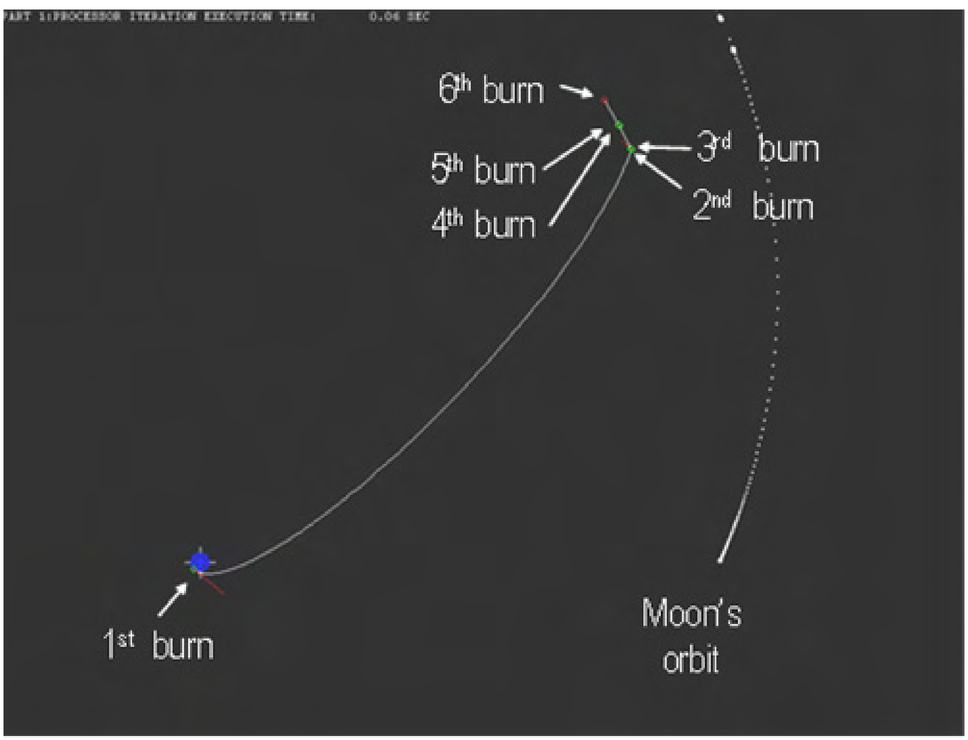


Figure 8.2.5-2: Nominal L1-rendezvous obtained with Copernicus

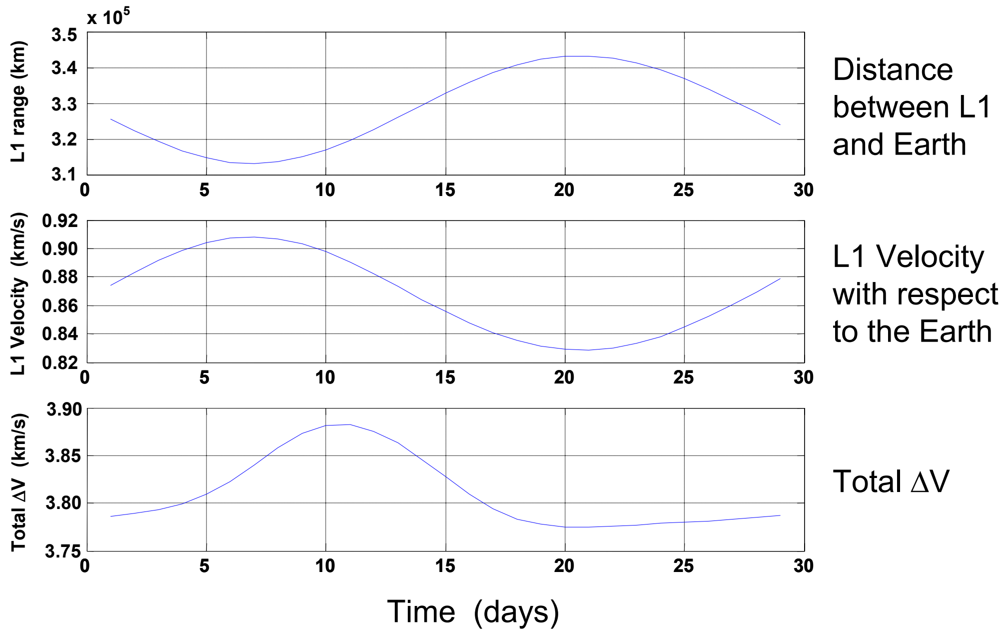


Figure 8.2.5-3: Total ΔV after 28 days (bottom figure) and L1 distance and velocity with respect to the Earth at arrival (top figure)

Deviations From The Nominal Case

In this section an example of a failed 2nd burn in the L1-rendezvous sequence is described (see Figure 8.2.5-4). The idea is to show that if the 2nd burn is not executed then the spacecraft will go back to Earth without colliding with the target in five days (approx.). The spacecraft will not return to the same parking orbit because of the Moon's effect on the trajectory. Since it might be interesting to return to a specific orbit, the first pair of burns and the distance to the target can be chosen such that in case of a failed 2nd burn the spacecraft return to a specific orbit around the Earth.

If the 2nd burn is performed correctly then the subsequent burns will have the following feature: if one of the burns is not performed then the spacecraft will coast after the target. It necessary to point out that this situation is only temporary; the spacecraft will drift away without any maneuver for station-keeping.

In general, if the odd burns (3 and 5 burns in the example above) are not carried out then the spacecraft will coast for a while behind the chaser and eventually it will drift away due to perturbations. If the even burns (2, 4 and 6 burns in the example above) are not carried out then the spacecraft will drift away from the target but no collision will occur as we are not aiming for the target itself but a position behind the target (actually we are aiming for the position the target was at some time before).

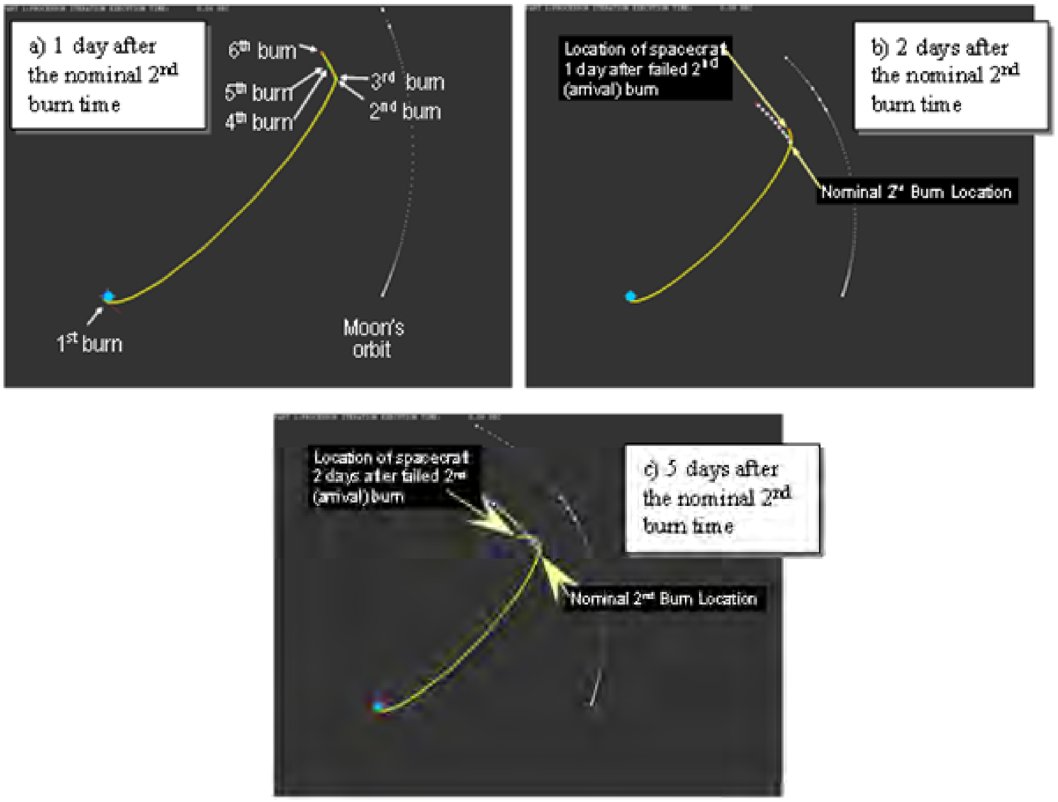


Figure 8.2.5-4: Chaser and target trajectories after a failed 2nd burn

Finally, this study should be completed with an error dispersion analysis of each burn. This analysis may result in a different sequence of burns (i.e. a correction burn few hours later the 1st burn, greater distances to the target after each pair of burns in order to avoid collisions, etc.)

<u>Conclusions</u>

This is only a proof of concept of a L1-rendezvous transfer. A further study that incorporates real ephemeris data and four-body disturbances should be carried out. It is interesting to note that the final part of the transfer (burns 2 to 6) lasts 12h so it seems that this time is not long enough to take into account for solar pressure and other environment disturbances in the dynamic model.

In the section: deviations from the nominal case, only the case when the burns are not performed has been studied. Therefore, it is also necessary to perform error dispersion analysis of each possible burn. From this analysis we will obtain:

- Distance between target and chaser. Errors in the steering angles or misburns might lead to collision. The error dispersion analysis provides information about the minimum safe distance that can be used after each pair or burns and therefore the number of burns needed to complete the rendezvous.
- Correction burns. Between the first and second burns it might be necessary to include a correction burn. The time between the first and the second burns is 3.5 days (approx.);

this long coasting arc increases the sensitivity of the final position and velocity due to steering errors or misburns in the first burn. Preliminary data[11] suggest that this high sensitivity can be reduced by including an intermediate burn. This new burn would change the way the rendezvous is carried out, and this information will determined the distances to the target in the subsequent burns (as mentioned before).

8.2.5.1 L1 Station-Keeping

Introduction

Although L1 is an equilibrium point in the Elliptic or Circular Restricted Three Body Problem a spacecraft placed at this point will not remain on it; due to the unstable dynamics of L1 a minor perturbation will provoke the drift of the spacecraft. This perturbation may come from the Sun or other major planets, or from the natural environment at L1, e.g. solar pressure. Therefore station-keeping will be required along the operation of the spacecraft (for the stage-disposal case this might not be the case). The station-keeping phase will consist of maneuvers performed at different times in order to compensate for the drift due to perturbations. These maneuvers can be carried out at regular time intervals or at different time intervals obtained through the optimization of a cost function (see Ref Scheeres) e.g. propellant, total DV, etc. Although the later approach is optimal, for this high-level study the regular time interval case will be considered.

Regular interval approach

The idea is to study how different time intervals require different DV costs. Due to the unstable nature of L1, it is known that the longer this interval is, the higher the control effort will be; at the same time it is not convenient to perform maneuvers very frequently, so a trade-off study trying to maximize the time interval but at the same time minimizing the total DV would be interesting. For this study, we will consider the following assumptions:

- Dynamic model: real ephemeris data (JPL's DE405) with Earth, Moon and Sun as main bodies.
- Operational constraints:
    - Only regular time interval maneuvers are allowed. For this study the time intervals will be 1, 2, 4, 6, and 8 days.
    - The time span will be only one month.

Table 8.2.5.1-1 Station-Keeping Total ΔV after one month

Time Interval (days)	Total ΔV (m/s)
1	0.110
2	0.075
3	0.124
6	14.356
8	997.358

Results

In general, an increase in the interval time will increase the total DV associated after one month (see Table 8.2.5.1-1). Although in the 4, 6 and 8-day cases this trend is clear, we cannot claim the same about the 1 and 2-day cases. The effect of the perturbation (the Sun) may mask this trend, for example by making the 2-day case more efficient.

Conclusions

Even for a high level study like this one, the use of real ephemeris data and the inclusion of the Sun as a perturbation in the system are key factors in order to obtain data that can be useful for future studies. Although frequent maneuvers reduce the total cost associated with the station-keeping they also require the use of navigation data more frequently; this might be a limitation to the approach. This study suggests that maneuvers in time intervals between 2 and 4 days are good starting points for a more detailed analysis of the station-keeping phase of this problem.

8.2.5.2 Earth to L1 Dispersion Analysis

Introduction

This section of the report describes preliminary analyses regarding the transfer of a chaser vehicle from low-Earth orbit (LEO) to the Earth-Moon L1 area and the subsequent rendezvous with a target vehicle previously transferred to the L1 area.

Orbit Transfer and Rendezvous Concerns

The goal of a series of orbit transfers that precede a rendezvous is to bring the chaser vehicle into proximity with the target vehicle and to do so within time and fuel consumption constraints. After completing the orbit transfers, the chaser vehicle performs a series of maneuvers that conclude with docking or berthing with the target vehicle.

Caution must be exercised in planning and performing the transfers and maneuvers. The goal of moving the chaser vehicle ever closer to the target vehicle must be tempered with the need to avoid collisions between the vehicles. Factors that lead to potential collisions include the orbital environment, imperfect state information, imperfect sensors, imperfect effectors, and equipment failures. The trajectory must be collision-free up to some point of no return that occurs just before the docking.

Comparison of LEO and L1 Environments

Several factors distinguish the LEO and L1 environments.

- Error accumulation. It takes over 3.5 days to travel to L1 (see Reference 1). Even very small velocity errors in the transfer burn will build-up to form a very large error at arrival at L1. This large error build-up does not occur with the short (~45 minute) transfer from one LEO orbit to another.
- Gravity gradient. Even slight changes in LEO orbit altitude result in different orbital rates. LEO rendezvous plans use the gravity gradient to their advantage (e.g., phasing orbits). In contrast, there is no gravity gradient at L1 from the short-duration perspective of

rendezvous. The L1 environment is a virtually flat space gravitationally for the distance, time, and relative velocities of concern for a rendezvous. Some consequences:

  - The transfer to L1 must be direct (no phasing orbits) for crewed vehicles. Anything else would take too much time.
  - The area around L1 is a point-and-shoot environment. To go to a target point, aim and fire. Fire again when the target is reached (though midcourse corrections are expected).
  - Collisions due to a dead chaser vehicle can be addressed. For example, aim at a point somewhat askew from the target vehicle. Suppose the chaser fails completely during the burn. The chaser will either drift through the target point and miss the target or will drift so slowly (in which case gravity gradient may become a concern) that the failed vehicle does not present a short-term hazard.

- Atmosphere. In LEO, atmospheric drag can complicate the design and be used to an advantage. Chaser and target vehicles with different ballistic coefficients present a challenge in LEO. L1 is a high-vacuum environment. Atmospheric drag is not a concern at L1.
- Navigation and communication infrastructure. LEO has the advantage of proven, reliable, highly accurate navigation infrastructure (GPS, ground tracking, TDRSS tracking, etc.) and communication infrastructure (TDRSS, numerous ground sites, etc.). L1 currently only has Deep Space Network (DSN), which is a highly subscribed resource.

<u>L1-rendezvous. The V-bar/R-bar approach - Transfer from LEO to L1</u>

The transfer from LEO to the L1 environment must

- Be direct (no phasing orbits)
- Accommodate the LEO orbital plane differing from the Earth-Moon orbital plane
- End with the chaser no closer to the target than the RSS of the target and chaser position errors (plus margin)
- End with the chaser no further from the target than the range of the relative navigation sensors (plus margin for drift)

A very small (0.1 meters/second/axis 3 sigma) error in the LEO-to-L1 transfer burn results in a very large (400 km 3 sigma) position error at L1. This large error is beyond the state-of-the-art for relative navigation sensors. One or more correction burns are needed. In addition, a large time and/or delta-v penalty will be incurred in correcting a 400 km error at L1. A correction burn will incur a much smaller penalty.

<u>*Case study 1: Common orbital plane*</u> The large position error in the L1 arrival position (closest approach to L1) from an uncorrected LEO transfer burn dictates that the LEO transfer burn target a point well below or above L1 to avoid the potential for a drifting collision with the target vehicle. This burn targets a point 400 kilometers below or above L1. Near-optimal fuel consumption occurs when this target point is the orbit's apofocus, in which case an impulsive burn of about 3100 meters/second is needed.

A correction burn compensates for errors in the initial burn and raises or lowers the orbit to within relative navigation sensor range. The correction burn is quite small: about 1.1 meters/second per 100 kilometers of change in the orbit's apofocus. The case study suggests that targeting a point 15 kilometers above or below L1 will be safe for this correction maneuver. (Caveat: some error sources were not taken into account in this initial study.)

An insertion burn at L1 arrival stops the vehicle with respect to L1. This burn is about 700 meters/second in magnitude. The insertion burn must be performed very close to the planned time, as it will take about 30 seconds to drift out of relative navigation sensor range at 700 meters/second relative velocity given a worst-case targeting error and a 40 kilometer relative navigation sensor range. The burn might thus take place before acquiring relative navigation; the burn must be performed with the aid of absolute navigation equipment only.

A final insertion correction burn compensates for errors in the primary insertion burn. This final burn must be performed after acquiring relative navigation but before drifting out of relative navigation sensor range. Assuming a 40 kilometer relative navigation sensor range and a 1% (7 meters/second) error in the performance of the insertion burn gives 35 minutes to acquire relative navigation and to perform the correction burn.

<u>*Case study 2: Different LEO and Earth-Moon orbital planes*</u> The LEO orbital plane is 28.5° inclination for a due-east launch from the Kennedy Space Center. A second study investigated the case where the LEO and Earth-Moon orbital inclinations differ by ten degrees.

A five-burn solution uses nearly the same delta-V as the four-burn common orbital plane solution and provides very good accuracy for the L1 insertion point. The additional burn is an orbital plane change maneuver that occurs between the correction burn and the insertion burn. Figure 8.2.5.2-1 portrays the trajectory from LEO to the L1 vicinity; locations at which burns occur are marked.

Several advantages result from adding the plane change maneuver:

- The initial burn can target a point much closer to L1. The chaser will miss the target with a large (2345 +/- 22 kilometers) out-of-plane component if the chaser fails after performing the initial burn but before performing the plane change maneuver.
- The second burn becomes a pure correction burn; it is not performed in the nominal (error-free) case.
- The plane change maneuver can be used as a second correction burn, reducing the error at L1 insertion.

– The cost of the plane change is greatly reduced by performing some of the plane change at LEO and some at the planned plane change maneuver point.

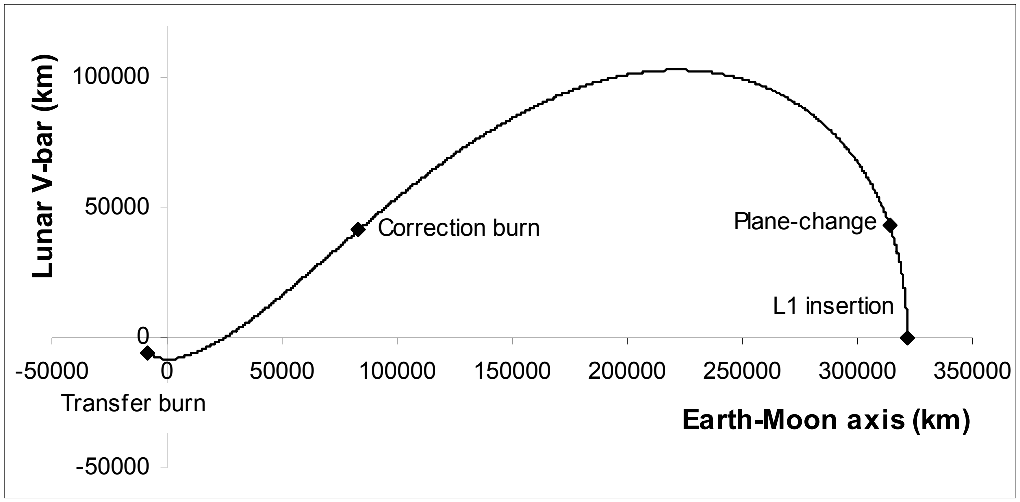


Figure 8.2.5.2-1: LEO to L1 Transfer Trajectory

The plane change could also be performed in LEO as a part of the initial transfer burn or at L1 insertion. The first alternative is cost prohibitive. The second alternative is slightly more costly than adding a separate plane change maneuver. Moreover, both alternatives require a suboptimal target point for the initial LEO-to-L1 transfer burn to ensure a collision-free trajectory.

<u>Methodology and Assumptions</u>

A Monte-Carlo simulation was used to study the effects of state estimation and propulsion errors on the trajectory. Table 8.2.5.2-1 presents the error sources used in performing this study. The errors are estimates and are not quantified against any particular navigation technology or sensor suite.

Burn	Navigated position error	Navigated velocity error	Delta-v error
LEO-to-L1 transfer	Not studied (position and velocity errors are highly correlated)	0.1 m/s per axis	0.1 m/s per axis
Correction	100 m (along velocity vector)	0.1 m/s per axis	0.5% axial, pointing error of 1.5 degrees max, decreases as burn time increases
Plane change	1 km (along velocity vector)	0.1 m/s per axis	Same as correction burn
L1 insertion	Not simulated	Not simulated	Not simulated
Correction	Not simulated	Not simulated	Not simulated

Table 8.2.5.2-1: 10 Degree Plane Separation Burns Error Sources

Several assumptions were made to simplify the development of this preliminary case study:

- The Moon is in a circular orbit.
- Earth and the Moon are point masses.
- No perturbing forces exist.
- Burns are impulsive.

Results

Table 8.2.5.2-2 presents the results of the Monte-Carlo study. Figures 8.2.5.2-2 and 8.2.5.2-3 portray the dispersions in the closest approach to L1 given failures after performing the initial transfer burn, the correction burn, and the plane insertion burn.

Burn	Burn magnitude (mean ± dispersion, in m/s)	L1 closest approach for dead vehicle after burn (mean ± dispersion, in km; x = distance above L1, y = distance ahead of L1, z = out-of-plane position)
LEO-to-L1 transfer	3063.7 ± 0.1	[7, 118, -2353] ± [315, 1, 19]
Correction	0.1 ± 0.3	[-10, 118, -2355] ± [70, 1, 17]
Plane change	38.2 ± 0.3	[-15, 0, 0] ± [6, 0, 15]
L1 insertion	716.9 ± 3.5 (error due to inertial guidance)	
Total	3819.0 ± 3.6	

Table 8.2.5.2-2: 10 Degree Plane Separation Monte-Carlo Study Results

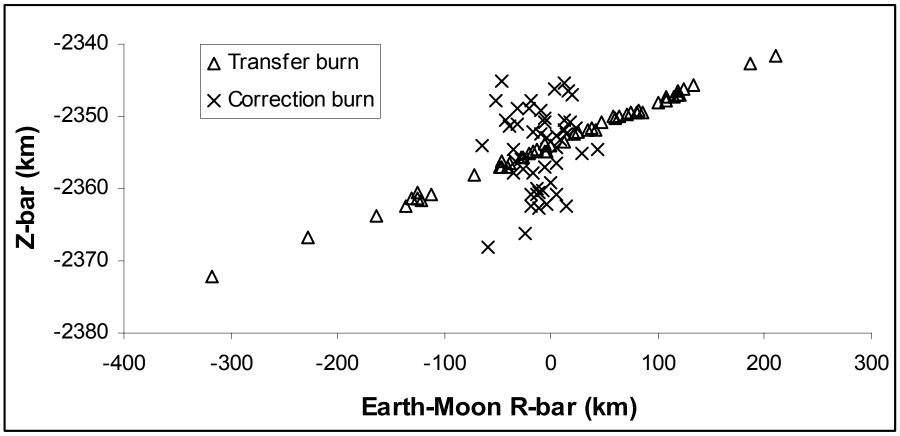


Figure 8.2.5.2-2: Dispersions with Failures after Initial Transfer Burn and Correction Burn

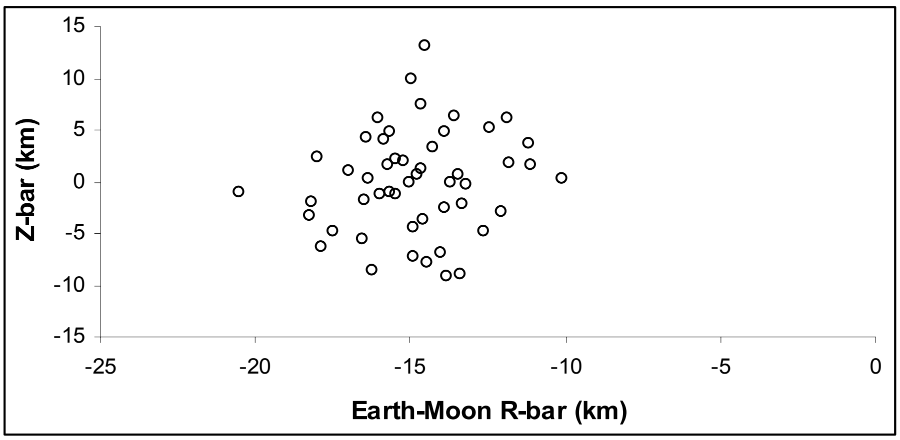


Figure 8.2.5.2-3: Dispersions with Failure after Plane Insertion Burn

L1 Proximity Operations

As noted above, L1 is a point-and-shoot environment. In this section, a zigzag path from the L1 insertion point to the target vehicle via a series of way points will be constructed that ensures that the trajectory is safe. The solution presented below is a feasible solution. Other solutions that take less time and consume less fuel most likely exist.

Summary

The path comprises a set of way points and straight-line legs between the way points. The first way point is a point 25 kilometers below the target vehicle on the R-bar axis, the nominal L1 insertion point. The final way point is 25 meters from the target vehicle docking port on the V-bar axis. The chaser proceeds to dock from this final way point. Intermediate way points are at 2.5

kilometers on the V-bar axis and 250 meters on the R-bar axis. Figure 8.2.5.2-4 depicts this trajectory.

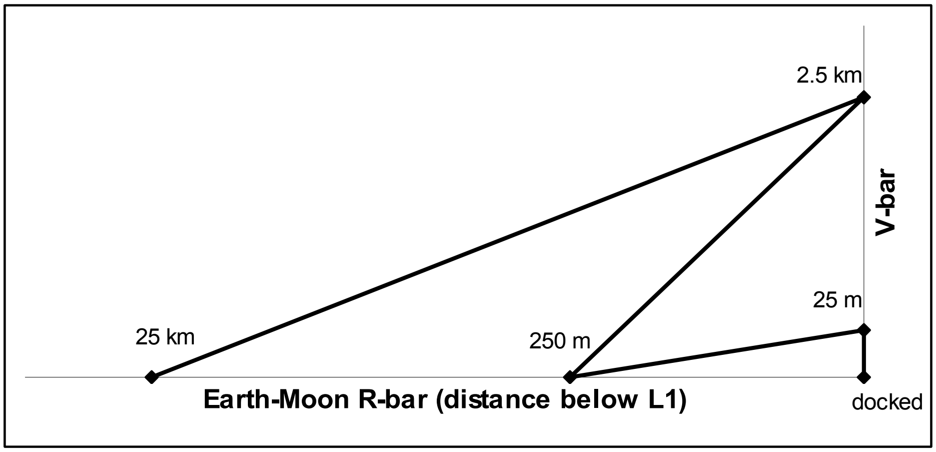


Figure 8.2.5.2-4: Proximity Operations

The 10-to-1 scale factor reduction in the distances for each leg pertains to the leg velocities as well. The vehicle drifts at 10 meters per second for the first leg (25 kilometers to 2.5 kilometers), 1 meter/second for the second leg, 0.1 for the third leg, and 0.01 for the docking. It thus takes about 42 minutes to perform a transit from one way point to the next. Adding a rather arbitrary 18 minutes of stationkeeping per way point results in a 4 hour rendezvous after transiting to the first way point. Given the errors in section 8.2.5.2, a zeroth leg is needed that involves transferring to the first way point. This transfer to the initial way point would target a point on the R-bar directly below the docking port. Since the three-sigma dispersion of the insertion point from the R-bar axis is 15 kilometers, a 10 meters/second velocity for this initial transfer would take less than 42 minutes to accomplish. The rendezvous could thus be completed in five hours with this scenario and would require about 42 m/s delta-v.

Keep-out zones

The Space Station adopted a set of keep-out zones to ensure safe operations by visiting vehicles. The visiting vehicles must stay out of the relevant keep-out zone until permission has been granted to enter that zone. A similar concept will help ensure safe operations for L1 rendezvous operations.

The first keep-out zone is a 1 kilometer radius cylinder with semispherical end caps that surrounds the target vehicle. The cylinder axis is on the Earth-Moon y-axis. The docking port is at one end of the cylinder. The goal of the leg between the first and second way points is to pass to the outside of the docking port end cap around the docking port and hence miss the entire keep-out zone. The chaser vehicle is granted permission to enter this keep-out zone after successfully stopping at the second way point.

The second keep-out zone is a 10-to-1 scaled down version of the first keep out zone. The vehicle is granted permission to enter this keep-out zone after successfully stopping at the third (250 meter) way point. The final keep-out zone includes a 10 meter radius end cap around the docking port. The cylinder, however, is not reduced by a factor of 10, as the target vehicle may have solar arrays that extend beyond 10 meters from the V-bar. This keep-out zone applies to the transfer from 250 meters on the R-bar to 25 meters on the V-bar. The end-cap will remain the region to avoid if the chaser vehicle is indeed on the R-bar at 250 meters distance.

Collision scenarios

Deviations from the nominal trajectory will result from errors and limitations in the vehicle's sensors, effectors, and control algorithms. A safe trajectory design must accommodate these expected errors that occur in all real vehicles. An even more important consideration is to make the trajectory safe in spite of unexpected errors. Experience with LEO rendezvous has shown that failure scenarios are often the driving factor in the trajectory design. Failure scenarios that the L1 rendezvous design must take into account include

a) The chaser begins the transfer from one way point to the next by imparting some delta-V to itself and then fails completely. The delta-V is sufficiently small that the vehicle has not corrected any errors that resulted from the delta-V but is large enough that the free drift trajectory poses a collision threat.

b) The chaser detects a deviation from the nominal path and imparts a delta-V to correct the deviation. The vehicle fails during or after performing the maneuver. The resultant free drift trajectory impinges upon the relevant keep-out zone, posing a collision threat.

c) The chaser vehicle must halt relative to the target vehicle at each way point before proceeding to the next. The chaser kills its tangential relative velocity but still has some residual radial velocity directed away from the target. The vehicle over-burns while attempting to kill this radial velocity and then fails completely. The over-burn results in the chaser vehicle aimed directly at the target vehicle, again posing a collision threat.

Analysis

The latter two collision scenarios can be addressed by building redundancy and safety margins into the vehicle's sensors, effectors, avionics, electronics, control algorithms, and flight procedures. The first scenario, however, does constrain the design of the trajectory path and the accuracy of the relative navigation sensors, the attitude control system, and the effector alignment. Figure 8.2.5.2-5 depicts these errors.

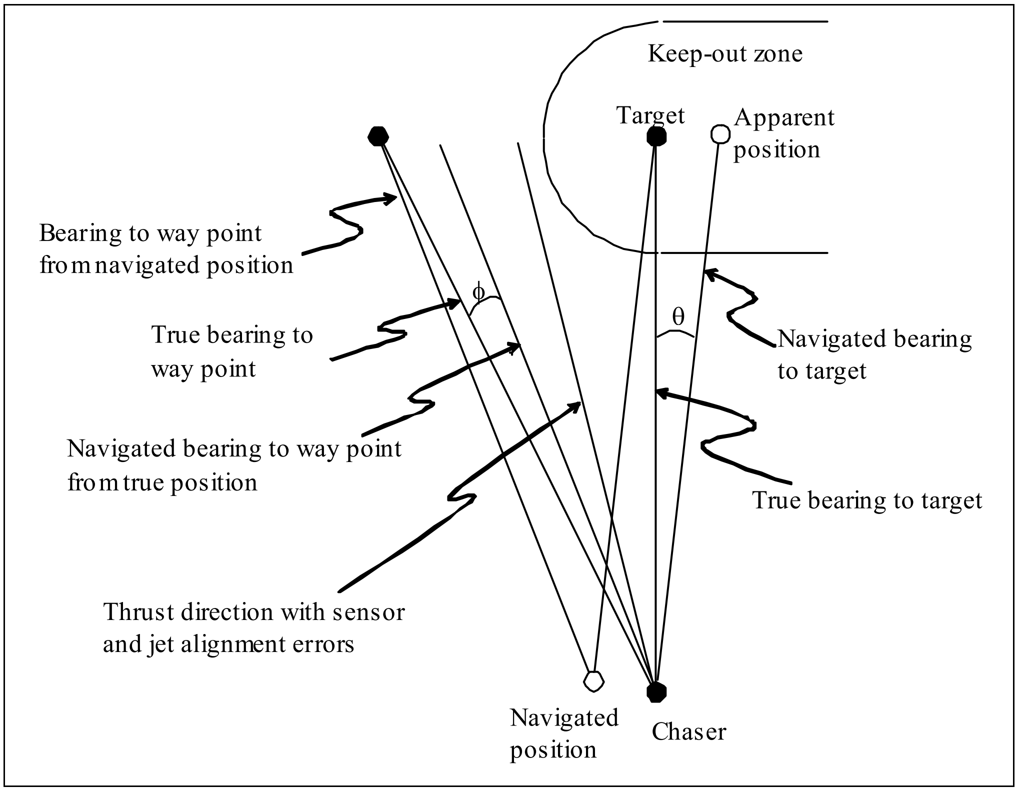


Figure 8.2.5.2-5: Errors impacting bearing to way point (not to scale)

The 10-to-1 reduction in distance for each step represents an angle of 5.71 degrees. For the initial step with the vehicle up to 25 kilometers below the docking port, a keep-out zone end-cap of 1.0 kilometers represents an angle of 2.29 degrees. The same angle pertains to the second leg given the 10-to-1 scaling in the size of the keep-out zones. The residual 3.4 degrees less some margin for safety represents the maximum allowable error arising from erroneous attitude (navigated and commanded) and from the propulsion effectors. The navigation, attitude control, and jet alignment errors are independent and thus should be RSSed rather than added to form a combined error. Assuming a 2-to-1 safety factor and 1 degree errors in jet alignment and attitude control error reduces the residual 4.3 degrees to 0.94 degrees, which is the maximum allowable error contributions from navigation sources. The 1 degree errors in jet alignment and attitude control are representative values for current vehicles. The 0.94 degree error (angle ϕ in Figure 8.2.5.2-5) represents a combination of the navigation sensor bearing error (angle θ), the navigation sensor range error, and the navigation filter error. The challenge is to provide accurate relative range and bearing up to 25 kilometers (or more) from the target.

Summary

This report developed safe and feasible rendezvous and proximity operations scenarios for L1. The preliminary studies outlined in this report made a number of simplifying assumptions. These simplifying assumptions need to be addressed. The study presented is but the first step from theory to practice.

Based on this study, three navigation technologies were identified as critical for L1 operations:

- Ability to control the LEO-to-L1 transfer burn to about 0.1 meters/second/axis accuracy. Very small velocity errors in the initial transfer burn, if uncorrected, result in large position errors upon arrival in the vicinity of L1.
- Relative navigation, including accurate bearing, to about 40 kilometers. The chaser vehicle cannot rely upon relative navigation in performing the L1 insertion burn. Errors in the burn will result in a drift. The chaser needs to acquire relative navigation before it goes out of sensor range.
- Absolute navigation near L1. The only source available presently is the DSN, and this has low position accuracy and is heavily subscribed.

The above should not be construed as a complete list of enabling navigations technologies required for L1 operations. The only two environments with which the human space flight community has any experience are LEO and around the moon. The environment near L1 differs significantly from both. Additional work is needed in identifying the technologies that must be developed to make L1 a practical option for lunar operations.

Alternative rendezvous and proximity operations architectures exist and need to be investigated. The studies themselves suggest better alternatives than those investigated for both the in-plane and out-of-plane rendezvous. The proximity operations architecture outlined in this study requires over 40 meters/second of delta-V. Lower cost solutions certainly exist.

8.2.6 L1 to Moon Transfer – Nominal and Abort

<u>Introduction</u>

As part of the Lunar Design Reference Mission an abort ΔV assessment was developed based on a nominal L1 to Moon trajectory. The study was divided into three portions:

a. L1 to Moon Abort using a 60 hr nominal transfer to the Moon, with abort executed 2 hours past periapse.

b. L1 to Moon Abort using nominal trajectories ranging from 50 to 70 hrs, with abort executed 2 hours past periapse.

c. L1 to Moon Abort using 60 hr nominal transfer to the Moon, with abort executed on time ranging from 5 to 100 hours past L1 departure.

In these studies all transfer involved trajectories from L1 to a 90° inclined orbit about the Moon. Under nominal circumstances a lunar orbit arrival (LOA) maneuver is performed at lunar periapse to place the vehicle into a 100 km altitude circular orbit about the Moon. In the event that a failure to perform the LOA maneuver is detected an abort maneuver can be used to redirect the vehicle back to the L1 safe haven. The nominal L1 departure ΔV for a 60 hr transfer is 248 m/s and the LOA circularization maneuver of 632 m/s (880 m/s total ΔV).

Assumptions

The time at which the abort maneuver is performed and the desired return trip time directly affect the final budget for the mission, this assessment outlines these costs. Satellite Tool Kit (STK) was used to simulate the trajectories and to compute the ΔV for this analysis. STK is a COTS software tool allowing 2-D and 3-D visualization of studies. The tool allows for derivation of detailed spacecraft technical requirements and specifications with cross-discipline analysis capabilities. Some of the main capabilities include high-fidelity orbit propagation, maneuver planning, detailed link analysis and MATLAB interface. A selenocentric with J2 effects propagator was used, including the Sun and Earth as additional point masses for gravitational effects. A simplified GLGM2 gravity model with 2 degrees and zero order and an eight order Runge-Kutta-Verner integrator with ninth order error control was used to propagate to vehicle.

Results

a. For the first study involving a nominal transfer of 60 hrs, with abort executed 2 hours past periapse, the abort return time was allowed to vary between 10 and 100 hrs. The constraint of aborting 2 hours past periapse was derived from the Apollo era mission design requirements. The total abort ΔV requirement ranges from 1488 to 2624 m/s, and a minimum total abort ΔV was found to occur using a return to L1 time of 50 hours. These results can be seen in Table 8.2.6-1, and in Figure 8.2.6-1.

b. The second study involved the investigation of the effect of modifying the nominal outbound trajectory time. The outbound time was allowed to vary from 50 to 70 hrs, while the abort return time to L1 crossed the span of 10 to 100 hrs. All aborts were still performed in the Apollo style time of 2 hrs past periapse. As can be seen in Figure 8.2.6-2, the abort ΔV approaches a minimum when a return time to L1 is approximately 50 hrs, and with outbound trajectories in the range of 50 to 54 hrs.

Abort Return Time (hr)	Abort ΔV Return to L1 (m/s)	L1 Stop ΔV (m/s)	Total Abort ΔV (m/s)
10	1295	1329	2624
15	991	844	1835
20	875	597	1472
25	824	448	1272
30	802	351	1153
35	796	285	1080
40	799	238	1037
45	809	207	1016
50	825	187	1012
55	845	176	1021
60	870	173	1043
65	899	177	1076
70	934	186	1119
80	1015	217	1232
90	1105	259	1363
100	1188	300	1488

Table 8.2.6-1: L1 to Moon Abort -- 2 hrs past periapse

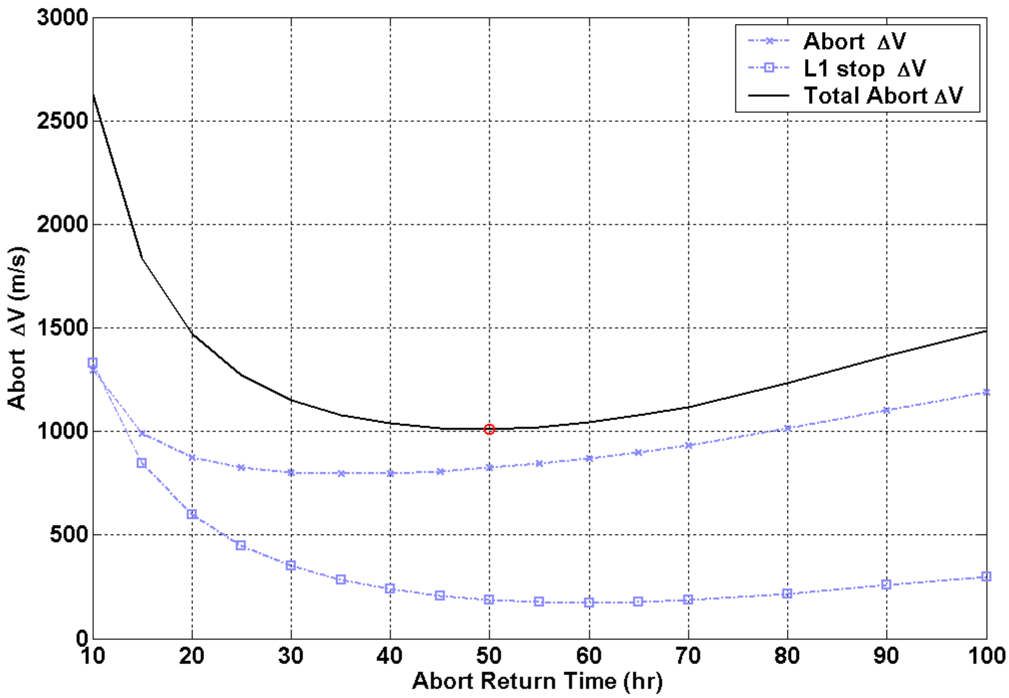


Figure 8.2.6-1: L1 to Moon Abort -- 2 hrs past periapse

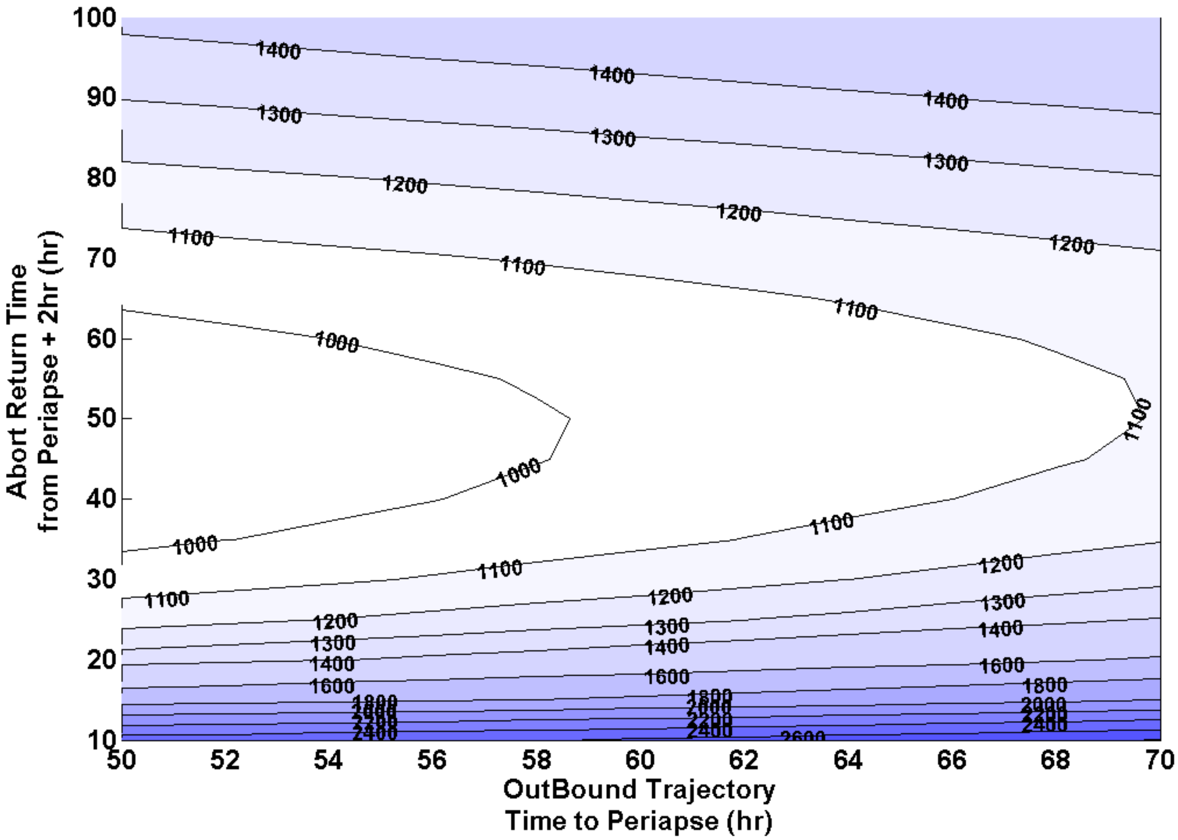


Figure 8.2.6-2: Abort ΔV w.r.t to OutBound Time and Return Time

The final study focused on the nominal 60 hr mission, but removed the Apollo style constraint of performing the abort maneuver 2 hours past periapse. In this case, the abort sequence was performed at a range of times spanning 5 to 100 hrs past L1 departure. The return time was also allowed to vary from 5 to 100 hrs. Two maneuvers are computed, an abort maneuver performed to return the vehicle to L1, and a stopping maneuver designed to stop the vehicle at L1. Results for the combined total cost for the abort can be seen in Table 8.2.6-2 and Figures 8.2.6-3 and 8.2.6-4.

As can be seen in Figure 8.2.6-3 and 8.2.6-4, the minimum abort ΔV is dependent on the abort return flight time and the outbound time. It is important to note that no data exists for the 60 hr outbound time with a 5 hour abort return time as that trajectory would intersect the Moon. The higher ΔV values for abort maneuvers performed 60 hours into the trajectory are coincident with the vehicle being located behind the Moon. Due to the polar inclination of the arrival orbit the targeter has a limited solution range to achieve the target.

Outbound Time (hr) \ Abort Time (hr)	5	10	15	20	25	30	35	40	45	50	55	60	65	70	75	80	85	90	95	100
5	248	124	83	63	51	43	37	33	30	28	26	25	24	24	24	24	25	26	28	31
10	494	248	166	125	101	85	74	66	59	55	51	49	47	46	46	47	49	51	55	60
15	735	368	247	186	150	126	109	96	87	80	75	72	69	68	68	69	71	75	80	89
20	969	486	325	244	196	164	142	125	113	104	97	92	89	87	87	88	92	97	104	114
25	1197	599	400	300	240	200	172	152	136	125	116	110	106	104	104	106	110	116	125	137
30	1418	708	471	353	281	234	200	175	157	143	133	126	121	119	119	121	125	132	143	157
35	1633	814	540	402	319	264	225	196	174	158	146	138	133	130	130	133	138	147	159	175
40	1846	918	607	450	355	291	246	213	188	170	156	147	141	139	139	143	150	160	174	192
45	2065	1024	673	495	387	315	264	226	198	178	163	153	147	145	147	152	161	174	191	211
50	2306	1138	742	541	418	336	278	236	206	183	168	158	153	153	157	166	178	195	215	238
55	2603	1273	821	590	450	357	292	246	214	192	178	172	171	176	185	199	217	237	259	281
60	NaN	1433	907	649	501	411	354	320	301	292	291	295	302	311	321	331	342	352	361	370
65	2513	1228	789	565	428	336	271	223	189	163	146	135	129	130	136	147	164	186	212	239
70	2363	1166	760	553	426	341	280	235	201	175	155	141	132	126	125	127	134	146	164	187
75	2375	1179	776	572	448	364	305	261	227	202	183	168	158	152	148	148	151	158	167	181
80	2473	1233	817	607	480	395	335	290	257	231	211	197	186	179	175	174	176	180	186	195
85	2611	1304	868	648	516	428	366	319	285	258	238	223	212	204	199	197	198	201	205	212
90	2760	1381	921	690	552	460	394	346	310	282	261	245	233	225	219	217	217	218	222	227
95	2906	1455	971	729	584	488	419	369	331	302	280	263	250	241	235	232	231	232	235	240
100	3036	1521	1016	763	612	511	440	388	348	318	295	277	264	254	248	244	242	243	245	249

Table 8.2.6-2: Total Abort Sequence ΔV

Outbound Time (hr) \ Abort Time (hr)	5	10	15	20	25	30	35	40	45	50	55	60	65	70	75	80	85	90	95	100
5	494	369	327	306	293	284	278	273	269	265	262	260	258	255	253	251	249	246	244	240
10	735	487	404	362	336	319	306	297	289	283	277	272	268	264	260	256	252	248	243	238
15	970	602	478	416	378	353	335	321	310	301	294	287	281	276	271	266	261	256	250	244
20	1199	713	550	469	420	388	364	347	334	323	314	306	300	294	288	283	278	273	268	262
25	1420	821	621	522	463	424	397	377	362	350	341	333	326	321	315	311	306	302	298	294
30	1636	928	694	578	511	467	437	416	400	388	378	371	365	360	355	352	348	346	343	341
35	1851	1038	771	641	567	520	488	466	451	439	431	425	420	416	412	410	408	406	406	405
40	2073	1158	861	719	639	591	559	538	524	514	507	501	498	495	493	491	490	490	490	490
45	2317	1300	976	825	743	694	664	645	633	624	619	615	612	610	609	608	607	607	607	606
50	2616	1495	1147	991	910	864	837	821	810	803	799	796	793	792	790	788	786	784	782	779
55	3065	1839	1480	1331	1258	1220	1198	1184	1175	1169	1163	1159	1154	1149	1143	1137	1129	1121	1112	1103
60	NaN	3198	3052	3020	3018	3028	3042	3057	3074	3091	3107	3124	3140	3155	3169	3183	3195	3206	3217	3226
65	2388	1244	915	773	697	653	626	608	598	592	590	591	596	604	616	632	653	680	712	746
70	2394	1271	923	760	666	606	564	533	510	492	477	466	456	449	444	441	440	444	454	471
75	2487	1331	959	777	670	598	547	508	477	452	431	413	397	383	371	360	351	344	340	341
80	2621	1401	999	800	679	598	539	494	458	428	402	380	361	343	327	313	300	288	279	272
85	2768	1470	1038	821	689	600	534	484	443	409	381	356	334	315	297	281	266	253	241	231
90	2912	1533	1071	838	696	599	528	473	430	393	363	336	313	292	274	257	242	229	217	207
95	3041	1586	1097	850	699	596	520	462	416	378	346	319	295	274	256	240	226	214	203	195
100	3149	1627	1115	855	696	589	510	450	402	363	330	303	280	260	243	229	217	207	199	193

Table 8.2.6-3: Abort Maneuver ΔV

Outbound Time (hr) \ Abort Time (hr)	5	10	15	20	25	30	35	40	45	50	55	60	65	70	75	80	85	90	95	100
5	248	124	83	63	51	43	37	33	30	28	26	25	24	24	24	24	25	26	28	31
10	494	248	166	125	101	85	74	66	59	55	51	49	47	46	46	47	49	51	55	60
15	735	368	247	186	150	126	109	96	87	80	75	72	69	68	68	69	71	75	80	89
20	969	486	325	244	196	164	142	125	113	104	97	92	89	87	87	88	92	97	104	114
25	1197	599	400	300	240	200	172	152	136	125	116	110	106	104	104	106	110	116	125	137
30	1418	708	471	353	281	234	200	175	157	143	133	126	121	119	119	121	125	132	143	157
35	1633	814	540	402	319	264	225	196	174	158	146	138	133	130	130	133	138	147	159	175
40	1846	918	607	450	355	291	246	213	188	170	156	147	141	139	139	143	150	160	174	192
45	2065	1024	673	495	387	315	264	226	198	178	163	153	147	145	147	152	161	174	191	211
50	2306	1138	742	541	418	336	278	236	206	183	168	158	153	153	157	166	178	195	215	238
55	2603	1273	821	590	450	357	292	246	214	192	178	172	171	176	185	199	217	237	259	281
60	NaN	1433	907	649	501	411	354	320	301	292	291	295	302	311	321	331	342	352	361	370
65	2513	1228	789	565	428	336	271	223	189	163	146	135	129	130	136	147	164	186	212	239
70	2363	1166	760	553	426	341	280	235	201	175	155	141	132	126	125	127	134	146	164	187
75	2375	1179	776	572	448	364	305	261	227	202	183	168	158	152	148	148	151	158	167	181
80	2473	1233	817	607	480	395	335	290	257	231	211	197	186	179	175	174	176	180	186	195
85	2611	1304	868	648	516	428	366	319	285	258	238	223	212	204	199	197	198	201	205	212
90	2760	1381	921	690	552	460	394	346	310	282	261	245	233	225	219	217	217	218	222	227
95	2906	1455	971	729	584	488	419	369	331	302	280	263	250	241	235	232	231	232	235	240
100	3036	1521	1016	763	612	511	440	388	348	318	295	277	264	254	248	244	242	243	245	249

Table 8.2.6-4: L1 Arrival ΔV

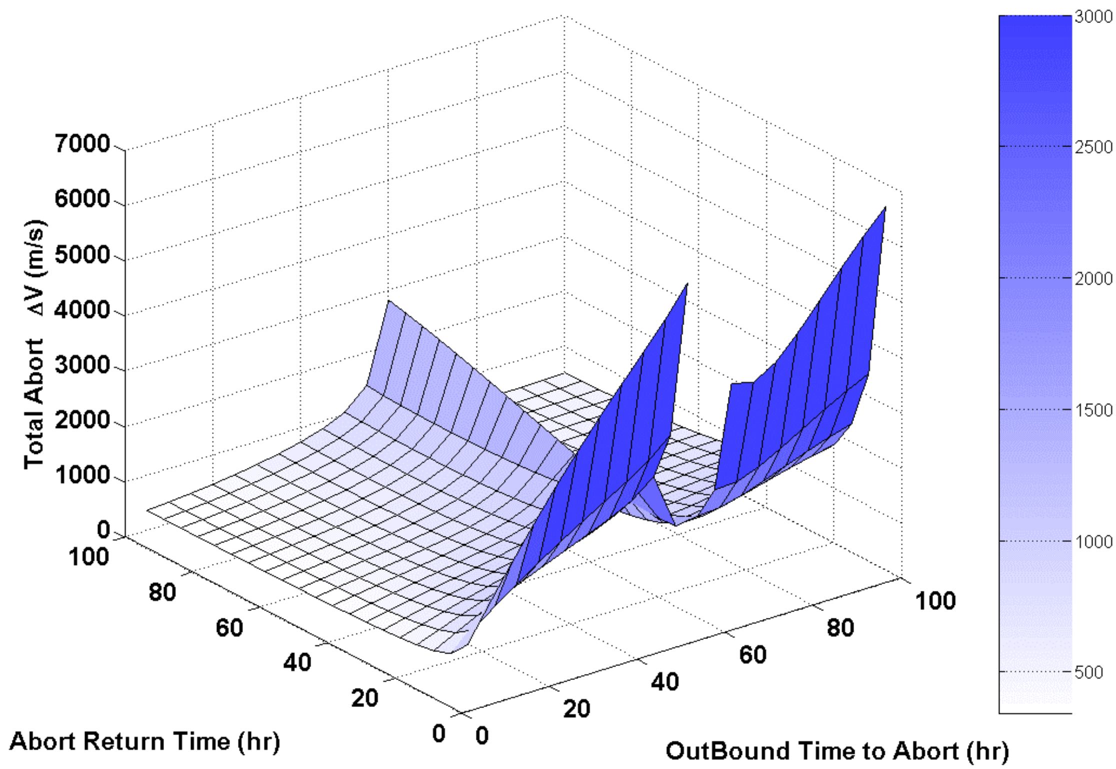


Figure 8.2.6-3: Mesh Plot of Total Abort ΔV

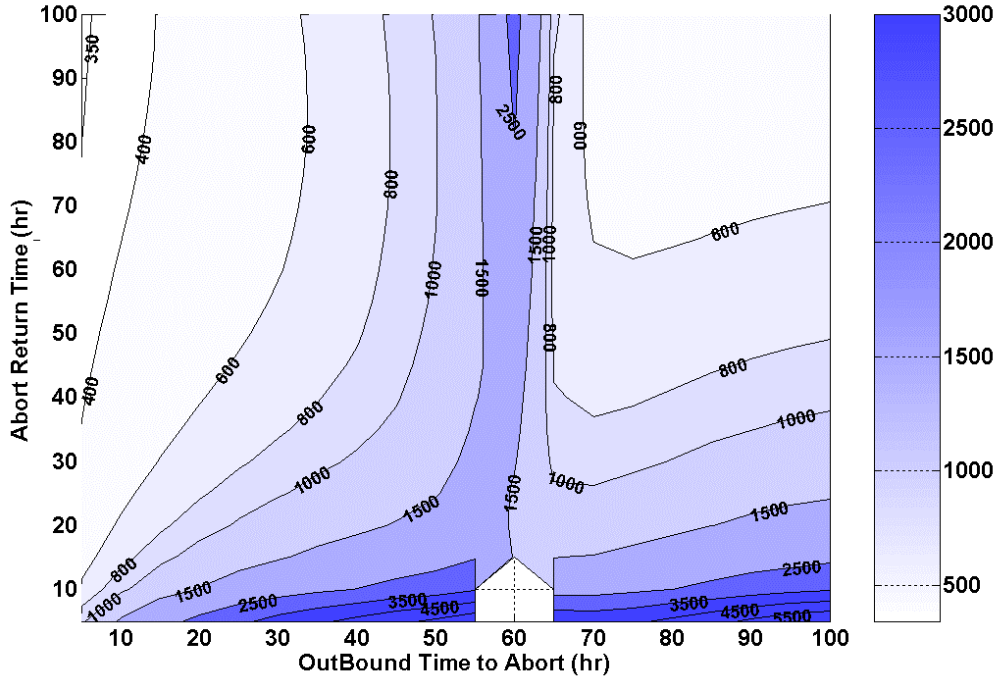


Figure 8.2.6-4: Contour Plot of Total Abort ΔV

8.2.7 Lunar Powered Descent and Ascent

Introduction

This section provides a detailed description of a possible lunar powered descent design along with performance analysis for a nominal descent and selected abort cases. This analysis is a product of the First Lunar Outpost (FLO) study performed for the JSC/Exploration Programs Office in 1992. Similar analyses could not be performed for the LDRM-2 study because of the lack of a detailed Lander vehicle design and specific mission approach. While the FLO mission focused on a lunar surface rendezvous (LSR) mission, the general vehicle performance trends can be used for powered descent in a lunar orbit rendezvous (LOR) or a libration point rendezvous (LPR) architecture. The following text is extracted from report JSC-25896, Lunar Lander Design for the First Lunar Outpost.

Where appropriate, rationale is given to support assumptions made in the analysis. The FLO lander trajectory was designed to provide a fuel efficient deorbit and powered descent to the lunar surface while meeting terrain clearance and final approach acceleration constraints. The descent design was then evaluated for compatibility with potential abort scenarios which include: in-plane abort to orbit following complete descent stage failure, in-plane abort to orbit with and assist from a partially failed descent stage (1 or 2 engines out in a 4 engine array), abort to the lunar surface following a partial descent stage failure, and target re-designation in the final vertical landing phase of the descent.

Ongoing work should address a non-coplanar ascent from an aborted descent that has a specific required return plane or a specific LOD target. The LOD target is based upon Earth landing location and ascent/LOD performance considerations. In addition, the nominal and abort trajectories will be updated to include the latest vehicle mass statement, transport lags, rate limited steering, and possibly navigation related dispersions.

In the case of an LSR approach such as was used in the FLO study, the same pressure vessel (crew module) was used for the lunar descent, ascent, and earth return. Program cost was a prime driver for the single pressure vessel approach. The absence of a lunar rendezvous made this approach simpler and thus more attractive, operationally. It also provided for anytime liftoff from the lunar surface, a highly desirable option to enhance crew safety.

The Apollo information, used for comparison against the FLO trajectory design, was obtained from Apollo 11 lunar trajectory notes and pre-flight operational mission profile documents[15, 16]. The Apollo Lunar Module (LM) powered descent was used as a point of reference to give perspective to the FLO design.

Lander Propulsion System

The thrust-to-weight (T/W) ratio has a significant effect on the ΔV cost for the lunar powered descent (and ascent). Figure 8.2.7-1 shows the powered descent ΔV as a function of the initial T/W. The selected FLO lander initial T/W was 0.4, where T/W was referenced using Earth's gravitational acceleration (i.e. g = 9.8 m/s^2). This T/W, which is designed to maximize the useful payload to the lunar surface given the use of existing engines, is based on a descent stage consisting of four cryogenic (liquid hydrogen and oxygen) throttle-able RL10-3-3A Centaur engines. These engines can be gimballed +4° and each engine has a maximum thrust of 73392 N (16500 lbf) and an Isp of 444.4 seconds. The numerical optimization tool, or simulation to optimize rocket trajectories (SORT), used in the descent performance analysis employed a model of engine Isp as a function of thrust level providing more realism to the lander's performance under throttling conditions. The ascent stage T/W at liftoff from the lunar surface (and from an aborted descent) was set to 0.43. This T/W was based on the use of three AJ10-118 (Delta second stage) non-throttleable storable bi-propellant (monomethyl hydrazine fuel with a nitrogen tetroxide oxidizer) engines that could be gimballed +3.5° and have a thrust rating of 43813 N (9850 lbf) each at an Isp of 320 seconds.

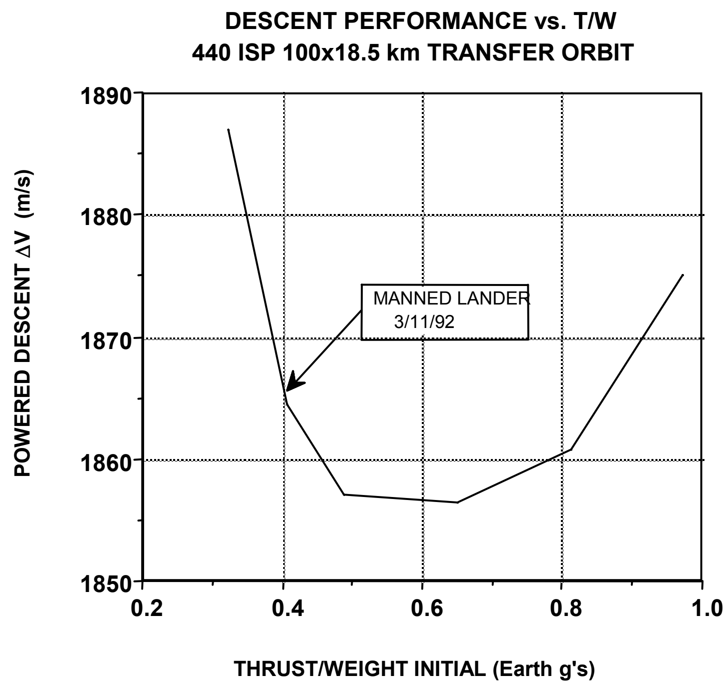


Figure 8.2.7-1: Powered descent ΔV as a function of the initial thrust-to-weight ratio with a legend indicating the vintage of the FLO lander design

It is also important that, in the case of an aborted descent, the engine startup transient for the ascent stage be minimized. Once the descent stage has failed, it must be jettisoned and the ascent engines brought to full throttle as quickly as possible. This can be both a performance and a surface impact issue especially during the final phases of the descent profile. The longer the lander falls without thrust to reduce its altitude rate, the greater the ascent ΔV cost to get back to lunar orbit becomes and the greater the chance of a surface impact.

Mission Scenario

The mission scenario for the lunar descent begins with the FLO lander in a 100 km temporary circular lunar parking orbit and ends with the lander touching down on the lunar surface. The parking orbit, established by the lunar orbit insertion (LOI) maneuver, is required to provide the capability to achieve any lunar landing site for small changes in the performance cost. Performance will be affected by the inclination of the powered descent trajectory. A retrograde equatorial powered descent costs about 9 m/s more than a posigrade equatorial descent. The small difference is attributed to the moon's slow rotation rate.

The mission scenario is a trajectory that was used for vehicle sizing. The trajectory is a retrograde (180°) descent to a zero longitude landing site. From the 100 km circular parking orbit, the lander executes an 18.6 m/s deorbit maneuver placing it on a 100 x 18.5 km transfer orbit. The lander then coasts (59 minutes) to the transfer orbit periapse region where it executes a powered descent initiation (PDI)* and performs a powered descent to the surface. The details of the powered descent are discussed in the next section.

A positive periapse altitude (18.5 km) was chosen to provide for contingency multiple passes before executing PDI. Early performance work showed that the lunar descent performance was fairly insensitive to the periapse altitude of the transfer orbit. A trade study comparing descent performance as a function of periapse altitude is currently being conducted to supplement this early performance work. The 18.5 km periapse altitude in combination with the flight profile provides for better terrain clearance than the Apollo lunar descent.

Nominal Powered Descent Profile

The powered descent trajectory was designed for minimum fuel use subject to terrain clearance and final approach acceleration constraints. Terrain clearance constraints dictate that the final part of the descent be suitably steep. There were no actual window viewing constraints for the FLO lander, as with Apollo Lunar Module (LM), since the visual information is intended to be provided by closed circuit television (CCTV) cameras on the lander. As a result, the only attitude constraint on the lander was that it be oriented vertically at landing.

The powered descent ignition occurs very near the periapse (true anomaly = 1.1º) of the 100 x 18.5 km transfer orbit. The T/W is 0.4 at powered descent initiate. The flight profile is comprised of three phases: a braking phase, a pitchup/throttledown phase, and a vertical landing phase (see Figure 8.2.7-2). The braking phase is a minimum fuel trajectory which reduces downrange velocity. The 60 second pitchup/throttledown phase sees the lander rotated from its attitude at the end of the braking phase to a vertical attitude. A slow pitchup rate (1.3º/sec) provides a terrain clearing approach that is steeper than that of the Apollo LM while providing CCTV viewing of the landing site. The time of visual acquisition of the landing site will depend on the lunar landscape surrounding the site. Also during this phase, the engines are brought from a full throttle condition to a 33% throttle level. The 33% throttle level provides a 1.2 lunar gravity acceleration (1.95 m/s^2) which is maintained during the final landing phase. At the conclusion of the pitchup/throttledown phase, the vehicle is oriented vertically at a 100 m altitude. The 1.2 lunar gravity acceleration level allows a reasonably slow vertical descent allowing the crew to adequately assess the intended landing site for hazards. The end of the 24 second vertical descent is marked by touchdown on the lunar surface.

The braking phase has been divided up into four pitch segments designed to minimize fuel use. The targeting for these pitch segments has been computed to incorporate the pitchup/throttledown and vertical descent phases. Table 8.2.7-1 (below) shows highlights of the powered lunar descent.

From the table, at the beginning of the coast, the apoapse and periapse altitudes (Ha and Hp) are 100 and 18.5 km, respectively. But at the beginning of the powered descent, Ha and Hp are 101.1 and 17.5 km, respectively. The descent simulation was set up using osculating orbital elements (which vary with time). The mean orbital elements (i.e., Ha and Hp), however, are 100 and 18.5 km, respectively, at the beginning of the powered descent.

* The acronym, PDI of powered descent initiation, has been borrowed from the Apollo program.

Preliminary studies have shown a need for engine throttleup to effect ranging following navigation state updates during the powered descent. The navigations studies are, as yet, incomplete. However, based on the preliminary results, it is suggested that the throttle level in the nominal descent profile be chosen to allow for throttleup in response to navigation state updates. Further study is required to assess the suitability of the powered descent flight profile to navigation state updates, abort to orbit performance, abort to landing performance, and auto system landing site redesignation. Though the descent profile design was not affected by these considerations, it should accommodate them.

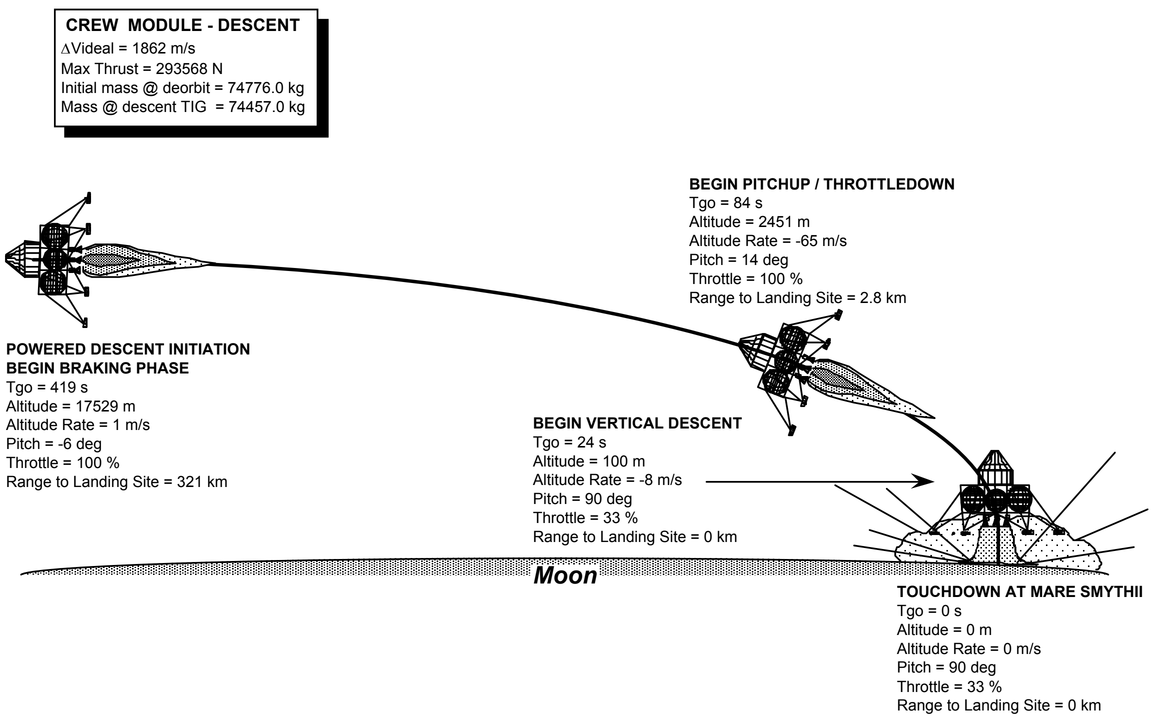


Figure 8.2.7-2: FLO Powered lunar descent flight profile

PET (sec)	Ha (km)	Hp (km)	ΔVideal* (m/s)	Mass (kg)	Comment
0	100	100	0	74776.0	Deorbit
4.7	100	18.5	18.6	74457.3	Begin Coast
3440.9	101.1	17.5	18.6	74457.3	Beg Powered Descent
3597.7	15.28	-1330	685.3	63893.7	End Pitch #1
3657.7	12.21	-1544	970.0	59851.6	End Pitch #2
3717.7	7.49	-1670	1274.7	55809.4	End Pitch #3
3777.7	3.70	-1731	1602.3	51767.3	End Pitch #4
3837.7	0.12	-1738	1834.1	49059.5	End Pitchup / Throttle-down Mnvr.
3861.7	0.0	-1738	1881.3	48516.6	Touchdown

Table 8.2.7-1: Lunar Descent Data

Comparison To Apollo Nominal Powered Descent Profile

The FLO lander descent profile was compared to that of the Apollo LM to give perspective to the FLO design as well as to reinforce assumptions and constraints used in the FLO design. The Apollo program, a successful manned lunar mission, provided an excellent point of comparison for the current design.

Design Drivers For FLO And Apollo

Three primary design drivers for the FLO lander powered descent were fuel optimality, terrain clearance, and a low acceleration final approach. The fuel optimality driver meant minimizing the fuel use during the powered descent, hence maximizing the usable payload to the surface of the moon. The terrain clearance constraint was driven by the program's requirement for total lunar landing access. No specific landing sites have yet been chosen, so the descent profile has to be capable of landing at any site. It was suggested that it would be prudent to make the FLO trajectory at least as steep as the Apollo lunar descent. The third design driver, a low acceleration on final approach, would afford the crew ample time to assess the targeted landing site and, if necessary, redesignate the landing to another site location. As it turns out, this low acceleration, along with the multiple (four) engine descent stage configuration, results in possible descent aborts to the lunar surface.

There are other important factors to be considered in the trajectory or systems design. They are the ability of the descent design to cope with navigation state updates, providing sufficient auto system target redesignation, and sufficient abort capability from the descent. Though these factors did not actually drive the design, the design should accommodate them.

The Apollo LM descent design was more driven by operational flexibility issues than was the FLO design*. The Apollo descent had three primary design drivers: fuel optimality, surface

* Ideal delta velocity refers to the entire velocity change due to thrusting. It includes all velocity losses (e.g., gravity losses, thrust pointing losses, etc.).

viewing, and a final approach that was compatible with large navigation uncertainties and surface irregularities. The fuel optimality driver was intended to provide a minimum fuel descent subject to mission constraints. The Apollo design was divided into 3 phases: the braking phase, the approach phase, and the landing phase. Like the FLO, the braking phase was designed to remove downrange velocity in a fuel efficient manner. The entire powered descent lasted 12 minutes and the 6.5 minute braking phase took from 12 to 3.5 minutes time to go until touchdown. The surface viewing design driver was intended to allow the crew to view the landing site area early in the powered descent (3.5 minutes before touchdown). The LM had to pitch up enough so that the astronauts could see out the LM windows to the landing site. This pitchup allowed the pilot to view the intended landing site at the beginning of the approach phase or high gate (altitude = ~7000 ft). In order to achieve this viewing constraint (vehicle attitude constraint), a false set of targets were used in the braking phase which removed downrange velocity, but allowed a large altitude rate at the end of the phase. In order to reduce this altitude rate, the LM pitched up significantly at the beginning of the approach phase. The third primary driver was a final approach that was compatible with potentially large navigation uncertainties and surface irregularities. At the time, navigation performance at the moon had not been verified, so the trajectory was tailored to allow for reasonably large navigation state updates.

Though not a primary design driver, an important consideration in the Apollo powered descent was an approach path that stayed close to the design reference trajectory. This allowed the crew to more easily evaluate the health of the descent guidance system. Another consideration was a sufficient capability to perform manual landing site redesignation. The descent trajectory was designed to accommodate both the auto landing system and the manual takeover during the final approach and landing phase. Another important consideration in the Apollo descent design was providing ascent capability from an aborted descent. This ascent abort capability did not cover the entire LM descent, because there existed a "dead man's curve" or region in the descent trajectory where an ascent from an aborted descent was not possible.

Performance And Profile Comparison

The FLO lander trajectory out performs the Apollo descent by 193 m/s while maintaining a steeper descent profile and a shorter phase elapsed time (see Figure 8.2.7-3). The FLO lander has a higher T/W (FLO = 0.4, Apollo = 0.33) allowing the FLO lander to complete its powered descent in a shorter time, thus incurring lower integrated gravity losses. The window viewing constraints in the Apollo descent trajectory required the LM to pitchup 3.5 minutes before touchdown compared to less that 1.5 minutes (84 sec) for the FLO lander (see Figure 8.2.7-4). The LM flew steeper pitch attitudes for a longer period of time, which again, resulted in a higher gravity loss penalty for the LM trajectory. The proposed CCTV viewing system for the FLO lander freed its trajectory from window viewing constraints. Figure 8.2.7-5 shows highlights of the LM descent.

* The FLO descent design is currently in a conceptual stage. As the FLO program matures, it may be more heavily driven by operations considerations.

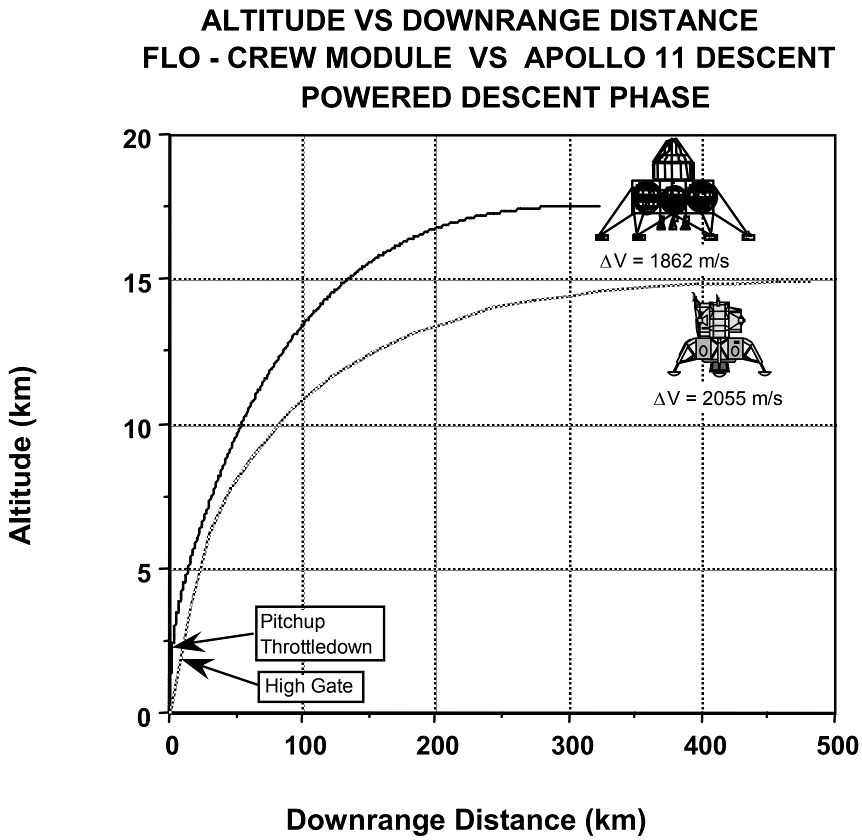


Figure 8.2.7-3: FLO lander and Apollo LM altitude versus downrange distance for the powered descent to the lunar surface

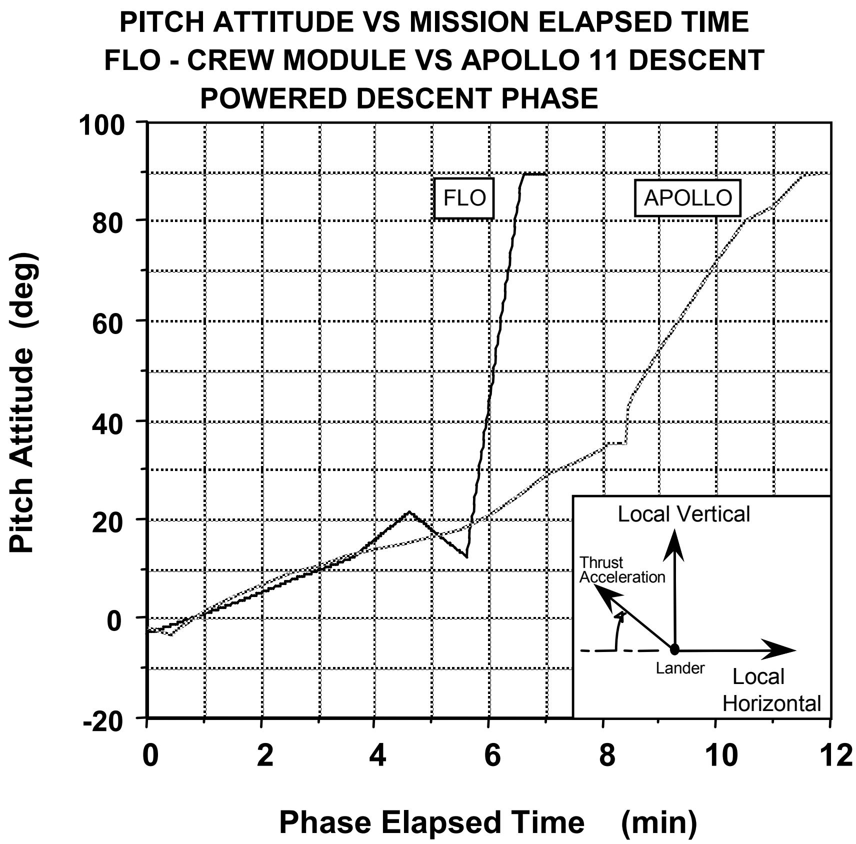


Figure 8.2.7-4: FLO lander and Apollo LM pitch attitude versus phase elapsed time for the powered descent to the lunar surface

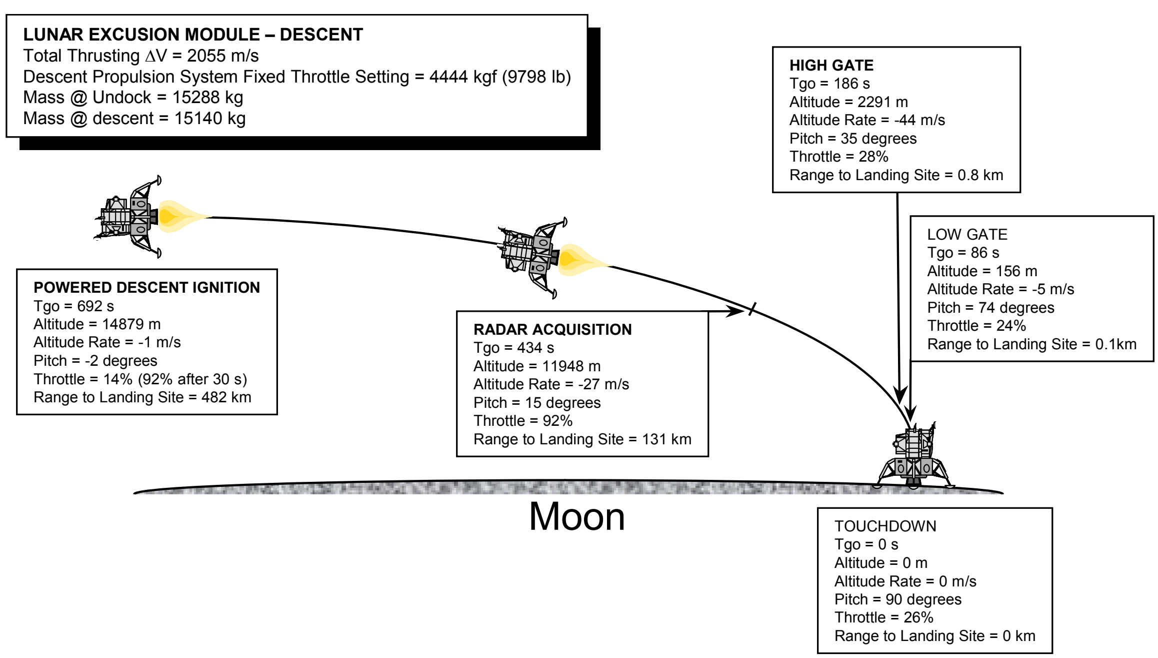


Figure 8.2.7-5: Highlights of the Apollo descent flight profile

The highlights of the FLO and Apollo powered descent profiles are shown below.

MISSION PROFILES AT A GLANCE

Apollo – Lunar Module	FLO - Crew/Habitat Lander
Initial T/W = 0.33	Initial T/W = 0.40
PET = 692 seconds	PET = 421 seconds
Videal = 2055 m/s	Videal = 1862 m/s
111.1 x 15.2 km Transfer Orbit (60 n.mi. X 50000 ft)	100 x 18.5 km Transfer Orbit
Isp 303.5 Fixed 293.6 - 303.7 Throttled	Isp 444 Fixed 433 - 444 Throttled
Braking Phase - False Targets	Braking Phase - Actual Targets
92% Throttle - Throttling Late in Braking Phase to 57%[*]*	Full Throttle
Approach Phase - Begins @ High Gate ~3.5 Minutes Before Touchdown - Targets Used	Pitchup/Throttledown 84 seconds Prior To Touchdown
Low Gate - 156 m	Vertical Descent - 100 m
Powered Descent Range = 482 km	Powered Descent Range = 321 km

Descent Aborts

Performance analysis to date shows that a coplanar abort to low lunar orbit (i.e., 100 x 18.5 km) is achievable throughout the powered descent trajectory given either a partial or complete descent stage failure. Further, there are regions late in the powered descent trajectory where it is possible to complete a surface landing with one or two failed descent stage engines. This performance analysis, which assumes instantaneous jettison of the descent stage and throttleup of the ascent stage, revealed no instances of surface impacts for any of the descent abort cases. Since these events are not necessarily instantaneous, ongoing work will incorporate the finite time it takes to perform them. An evaluation of maximum control rates is also planned for ongoing analysis. There are certain regions of the powered descent trajectory where an abort to orbit would involve large pitch rates (i.e. 10° to 11°/second). Planned future work will either verify that vehicle propulsion systems are capable of generating the required pitch rates or provide rate

[*] The LM descent propulsion system (DPS) did not allow a sustained throttle setting between 92% and 65%.

limited steering and/or updates to the nominal profile. This future work will also investigate the performance for a non-coplanar ascent, from an aborted descent, to a specific LOD target.

Coplanar Abort To Orbit

This case of the in-plane abort assumes a complete failure of the descent stage. The ascent stage (instantaneously) separates from the descent stage and begins an ascent back to a 100 x 18.5 km transfer orbit where it coasts to apoapse for a circularization into a 100 km orbit. The ascent ideal velocity cost (total thrusting velocity) for the inplane abort to orbit is shown as a function of the phase elapsed time during the descent in Figure 8.2.7-6. Since performance for a coplanar ascent becomes an issue late in the trajectory, the plot reflects the time during the powered descent from the beginning of the pitchup/throttledown maneuver to lander touchdown. The nominal ascent ideal velocity for an ascent from the lunar surface is 1815 m/s. The most costly (in terms of ideal velocity) time to perform a descent abort is 40 seconds into the pitchup/throttledown phase where the ascent ideal velocity is 1831 m/s (16 m/s greater than a nominal ascent from the surface). This is a relatively low additional cost to maintain complete descent abort coverage. The ascent was restricted to pitching 20° from the vertical in the first 10 seconds of ascent powered flight. This was done to avoid close encounters (impacts) with the lunar surface.

Figure 8.2.7-7 shows the altitude of the FLO lander at the time of descent abort and the minimum altitude that the subsequent ascent trajectory encounters. The altitude rate and downrange velocity plots in Figure 8.2.7-8 show that early in the pitchup phase of the descent, there is a significant downrange velocity. So, even though there is a negative altitude rate, the ascent performance cost is reduced overall due to the significant downrange velocity. The performance cost becomes a maximum about 40 seconds into the pitchup maneuver because, while the altitude rate has been reduced, the downrange velocity has been reduced to a much greater degree. The cost to overcome the negative altitude rate overshadows the benefit derived from the relatively small downrange velocity.

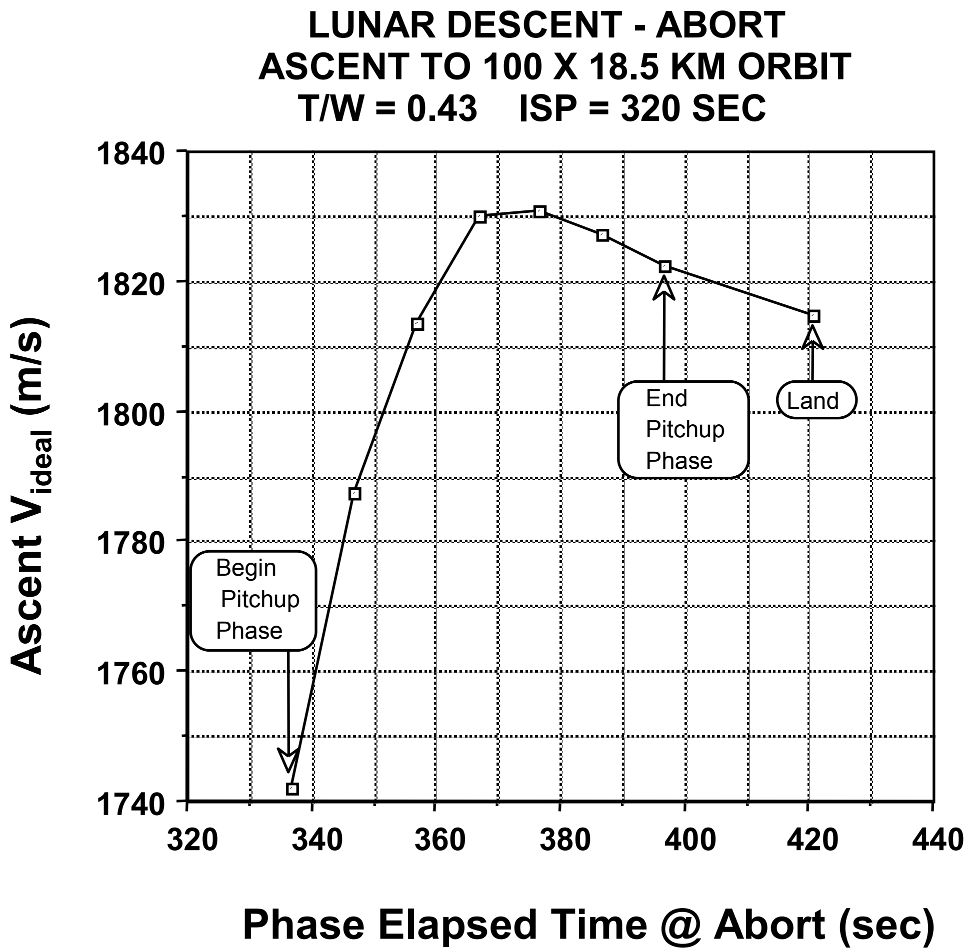


Figure 8.2.7-6: Ascent ideal velocity versus phase elapsed time from the beginning of the pitchup/throttledown phase to touchdown (Abort from touchdown is the same as a nominal ascent)

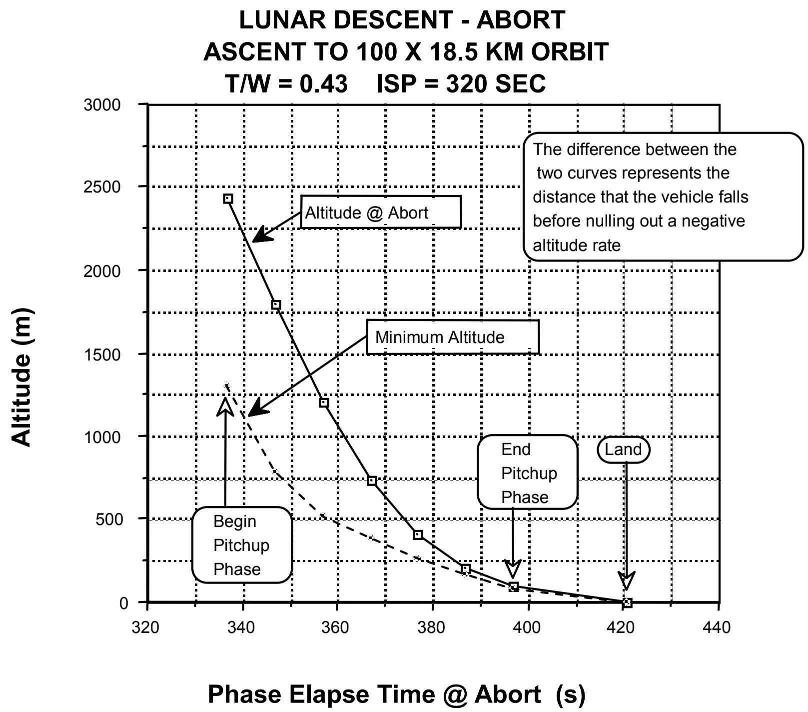


Figure 8.2.7-7: Descent abort altitude and minimum altitude during ascent to orbit versus phase elapsed time during the powered descent

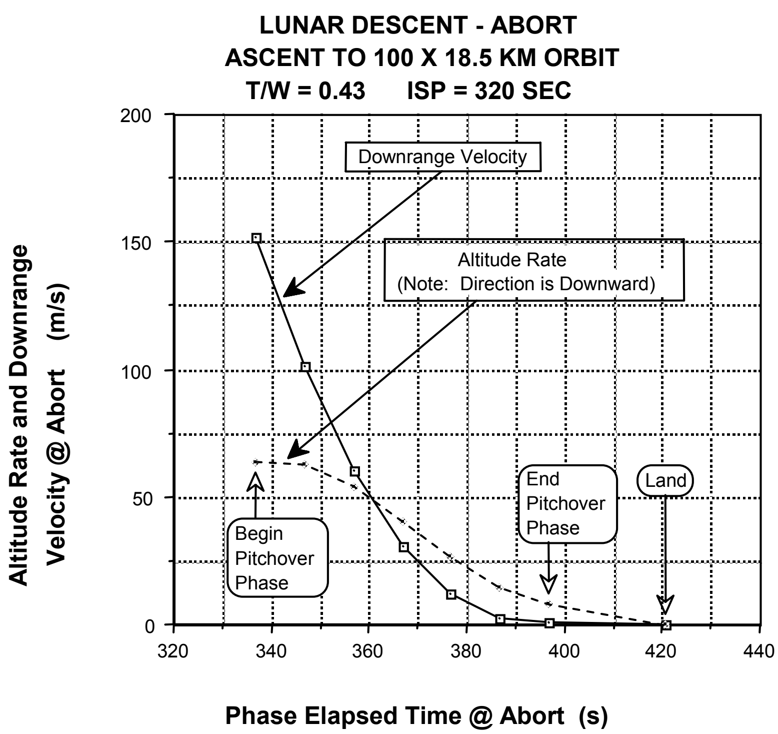


Figure 8.2.7-8: Altitude rate and downrange velocity versus phase elapsed time during the powered descent

Comparison To Apollo Abort Conditions

Figure 8.2.7-9 shows the altitude rate during the powered descent for the Apollo and the FLO landers as a function of the downrange velocity. The plot data was intended to provide an understanding of the flight conditions at an aborted descent for the FLO lander as compared to that of the LM. There existed a region on the lunar descent flight profile where an ascent abort to orbit was impossible. The existence of this region can be attributed to either ascent stage performance limits or a potential surface impact. To date, the ongoing Apollo mission review has not revealed the cause (or causes) for this region. A cursory comparison of Apollo/FLO descent flight profiles and ascent performance data seems to indicate that a potential surface impact plays a key role in the existence of this region.

The plot shows that the LM has as lower altitude rate throughout most of the descent for a given downrange velocity. For a coplanar ascent to orbit from an aborted descent, the greater the downrange velocity at abort, the better the ascent performance. The plot does not, however, reveal the driving cause for the region in the LM descent where ascent to orbit from the aborted descent stage was impossible.

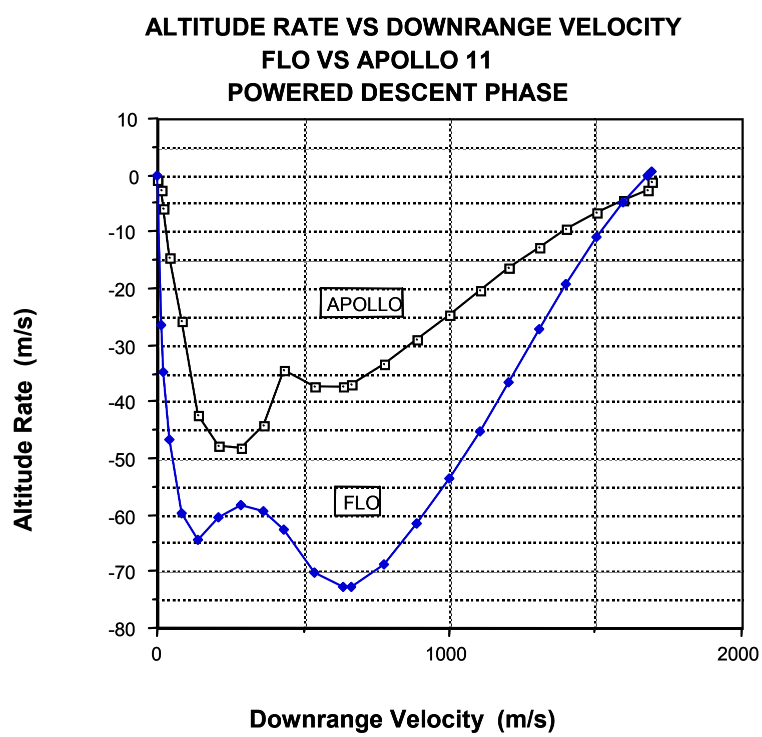


Figure 8.2.7-9: Altitude rate versus downrange for Apollo and FLO landers

The ascent performance of the LM ascent stage can be compared to that of the FLO ascent stage for an aborted descent which occurs at a time during the FLO descent profile such that the ascent performance cost is a maximum (i.e., 40 seconds into the pitchup/throttledown phase). The following table shows a comparison of the FLO ascent stage performance cost for both a nominal and a descent-abort case as compared to a similar cost for the LM. The descent-abort case, again, assumes a complete descent stage failure 40 seconds into the pitchup/throttledown maneuver. A FLO ascent target orbit (i.e., 100 x 18.5 km) is used for both cases.

	T/W	Isp	Nominal Ascent ΔV (m/s)	Descent-Abort ΔV (m/s)
FLO	0.43	320.0	1815	1831
APOLLO	0.33	303.5	1844	1861

The performance penalty associated with the descent-abort for the FLO vehicle is 16 m/s. For the LM under the same abort conditions, the descent-abort performance penalty is 17 mps. There is very little difference in the descent-abort performance penalty between the two vehicles. Further, the plot shows us that the LM was always in a preferred condition for doing an abort to orbit than the FLO vehicle in that the LM had a greater downrange velocity than did the FLO lander for a given altitude rate. This indicates that the actual Apollo descent-abort might have had even a smaller performance penalty (over the nominal) than the FLO vehicle. It would seem that since the performance impact of an aborted Apollo descent was not very large, that the threat of a surface impact was likely a key factor in determining the region in the LM descent where abort to orbit was impossible.

Partial Descent Stage Failure

A partial descent stage failure includes the loss of one or two engines. With a partial failure, several abort options are available depending on when, during the descent, the failure occurs. One option is an abort to orbit using the remaining descent stage propulsion system to assist in the ascent. The other option is to abort to a surface landing on the moon. Performance studies to date indicate that the best policy for three failed engines is to treat the situation as if the descent stage had failed completely.

Abort To Orbit With Descent Stage Assist

Given one or two descent engine failures, an abort to orbit can be accomplished using the remaining propellant in the descent stage to assist in the ascent. Figure 8.2.7-10 shows the ascent ideal velocity cost for ascents to orbit using the descent stage with one and two engines out. It also shows the ascent cost for a completely failed descent stage (recall Figure 8.2.7-6). A descent stage assisted ascent reduces ideal velocity required by the ascent stage to achieve the 100 x 18.5 km target orbit.

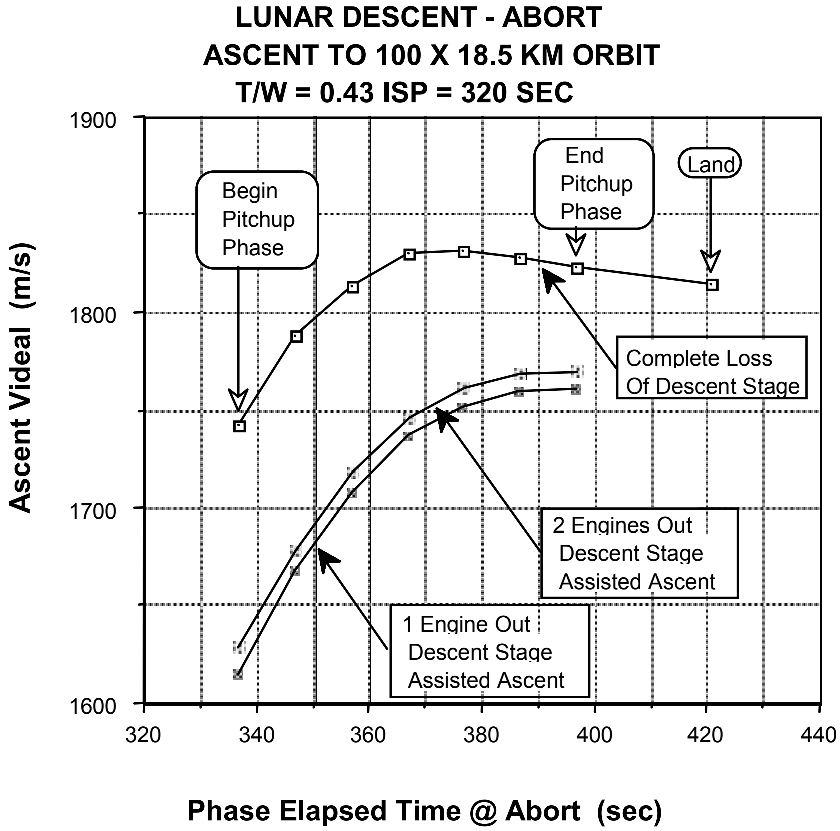


Figure 8.2.7-10: Altitude rate versus downrange for Apollo and FLO landers

The descent stage assisted ascent was conservatively designed. Upon abort (one or two engines out), the lander performs a 10 second rotation to a vertical attitude. It maintains this attitude until it has depleted the descent stage propellant. The lander then jettisons the descent stage to complete the ascent with the ascent stage.

Figure 8.2.7-11 shows a plot of the altitude following depletion of the descent stage propellant as a function of the time during the critical one minute pitchup/throttledown phase. These are the altitudes at which the spent descent stage would be jettisoned. A higher altitude is achieved for the single engine out case compared to the two engine out case. The higher altitudes result in lower ascent ideal velocity costs required by the ascent stage.

The plot in figure 8.2.7-12 shows the minimum altitude encountered for a one and two engine out descent abort. Twenty seconds into the pitchup maneuver, the minimum altitude for the two engine out case is 38 m. This is uncomfortably low. Surface impact constraints may preclude the FLO lander from flying a descent stage assisted ascent when the descent stage has lost two engines.

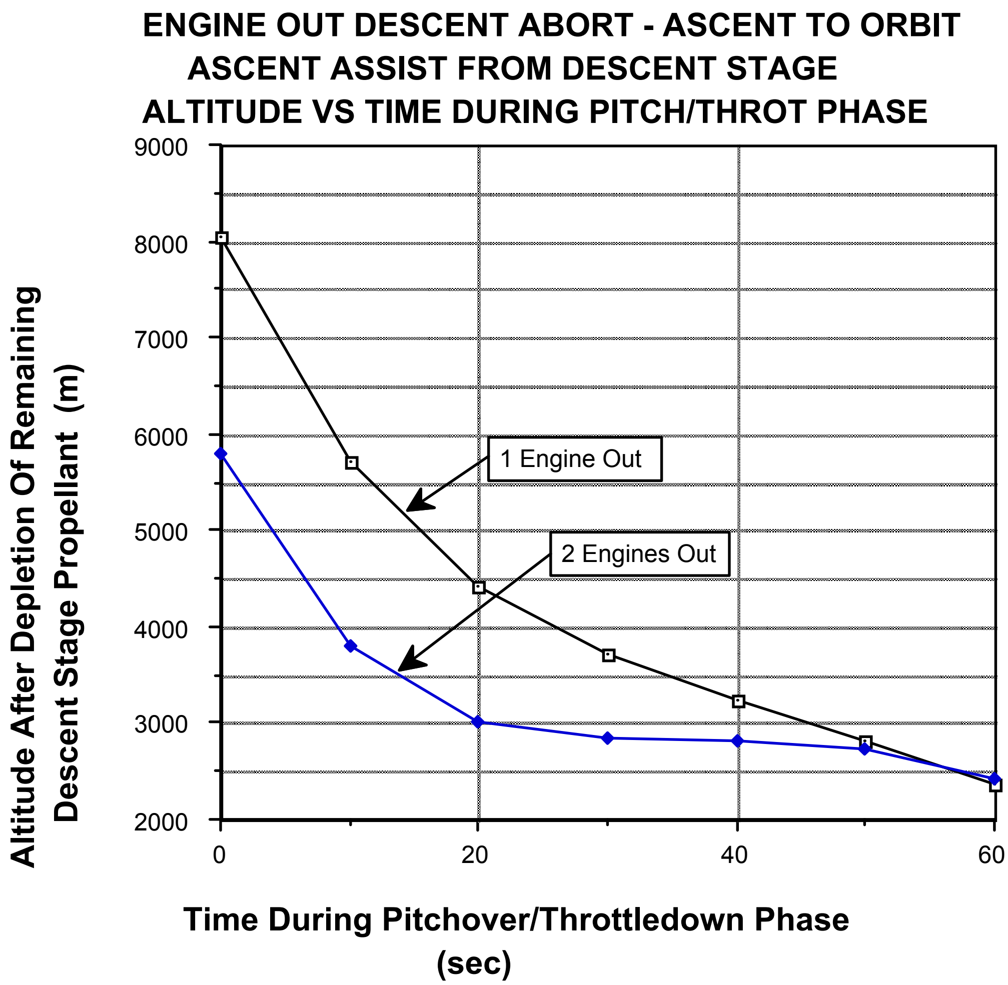


Figure 8.2.7-11: Altitude after depletion of descent stage propellant versus the time during the 60 second pitchup/throttledown maneuver, for a one and two engine out descent stage assisted ascent to orbit

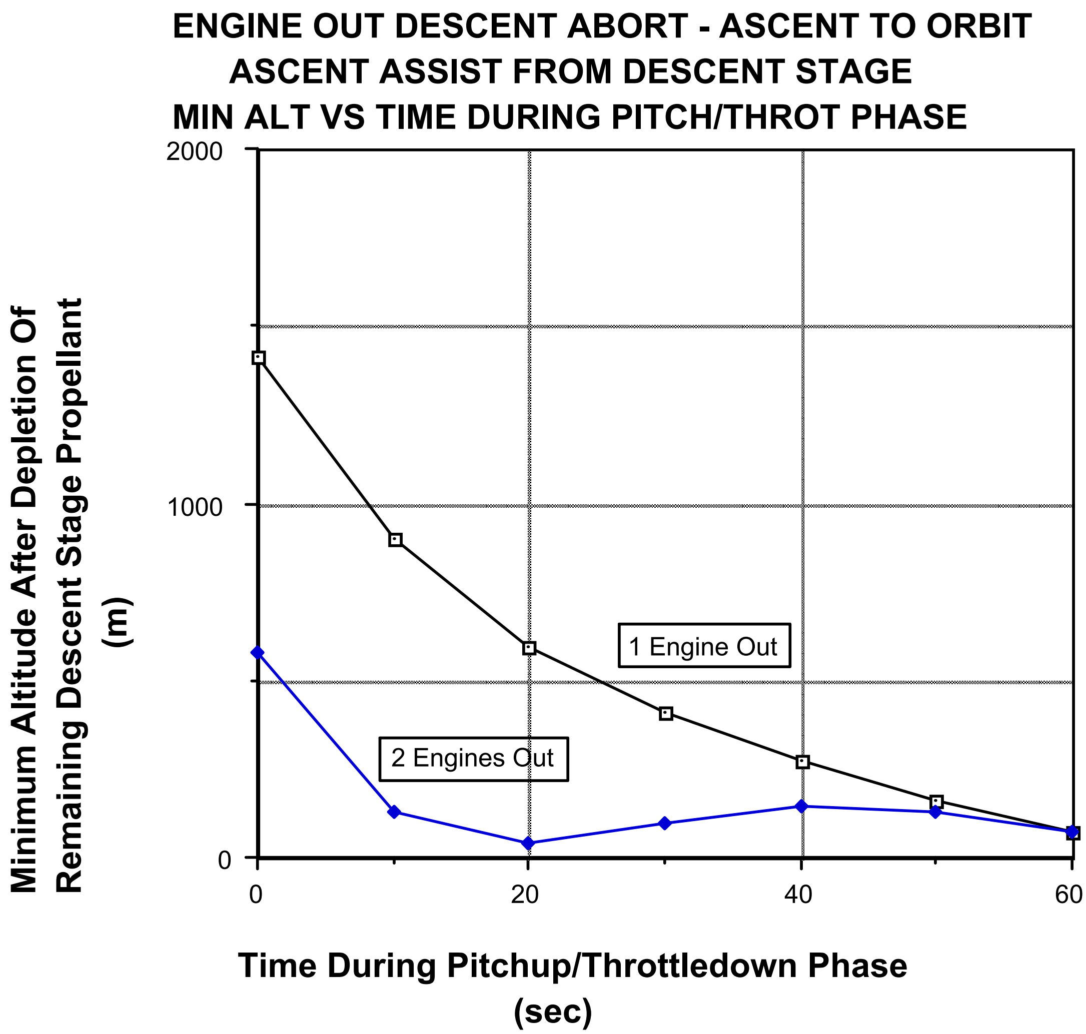


Figure 8.2.7-12: Minimum altitude encountered versus time during the 60 second pitchup/throttledown phase for a one and two engine out descent stage assisted ascent to orbit

Descent Abort To The Lunar Surface

Another abort option for a descent stage with one or two engines out is to continue to the lunar surface. Figure 8.2.7-13 shows the plot from Figure 8.2.7-6 with shaded regions indicating the time during the powered descent, when a nominal descent to the surface can be continued with either one or two engines out. Recall that during the pitchup/throttledown phase, the throttle is reduced from 100% to about 33%. Assuming no transport delays (i.e. instantaneous staging and ascent stage throttleup to full thrust), then after a phase elapsed time of 359 seconds into the descent or 22 seconds into the pitchup/throttledown phase, a nominal descent can be continued with a single engine out. After a phase elapsed time of 380 seconds or 43 seconds into the pitchup/throttledown phase a nominal descent to the lunar surface can be achieved with two engines out.

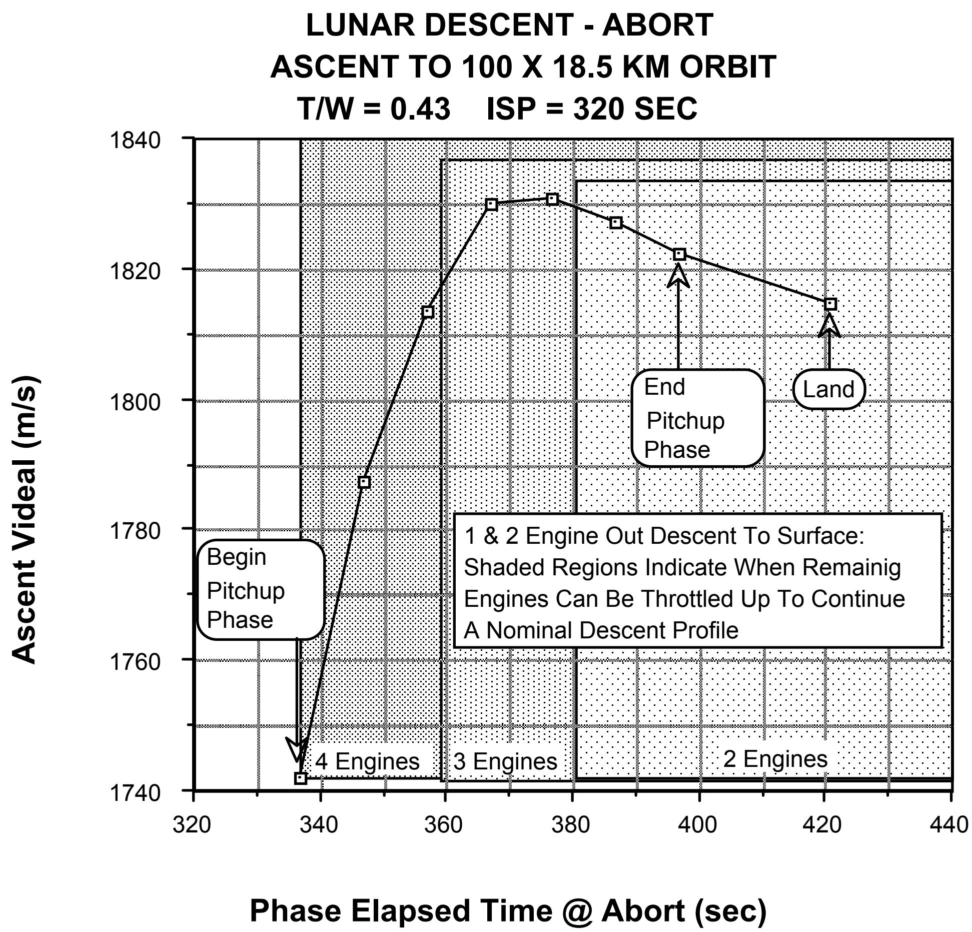


Figure 8.2.7-13: Minimum altitude encountered versus time during the 60 second pitchup/throttledown phase for a one and two engine out descent stage assisted ascent to orbit

Target Redesignation

The final landing phase of the powered descent, following the pitchup/throttledown phase, begins with the FLO lander at 100 meters altitude with an altitude rate of -8 m/s (downward). The vehicle attitude is oriented vertically, poised for final descent. The pilot would begin a target redesignation to avoid a hazard at the landing site by commencing a horizontal traverse through a prescribed distance during its descent to the surface. The end of the pitchup/throttledown phase provides a performance bound on this target redesignation. Any redesignation prior to the 100 meter altitude would incur less performance impact. The target redesignation can be accomplished in any direction for the same performance cost, since the lander has no downrange velocity at this point.

One of the several assumptions for the target redesignation is that the vehicle can actively throttle during the target redesignation phase. The throttle range for the target redesignation lies between 25% and 100%. This equates to a thrust range of 73,392 to 293,568 N (16,500 to 66,000 lbs.). There was some concern about engine stability with a large throttle throw range such as 10% -> 100% or 10-to-1 in addition to concerns about engine development time and the costs involved. Performance for a target redesignation with a 10-to-1 throttle throw is slightly better than that with a 4-to-1 range. However, the 25% minimum throttle on the 4-to-1 throttle range poses control problems for the redesignation. For example, suppose there were a problem with a redesignation and it became necessary to quickly build a negative altitude rate (Hdot). (Propellant budgets may dictate increasing negative Hdot and then increasing acceleration before touchdown). A 25% minimum throttle capability may not be low enough to allow the FLO lander, depending on its mass, to slowly reduce Hdot or even achieve a negative Hdot at all. It is possible that a modest increase in throttle throw range (e.g., 5-to-1, 6-to-1, etc.) may provide the capability to expeditiously achieve negative Hdots while still providing good performance as well as potentially easing engine development timelines and budgets. A 10% throttle rate limit and a 5º/second pitch rate limit have also been imposed on the target redesignation. The vehicle attitude was constrained to be no greater that 25º from the vertical during a redesignation.

Work to date has produced a fuel optimal set of target redesignation trajectories subject to the above mentioned constraints. Though the trajectory path itself was fairly smooth and benign looking, the throttle history was quite lively. Continuing work in this area will focus on calming the movement of vehicle controls (i.e., throttle) without sacrificing performance. It would be more settling, from the lander crew's viewpoint, to avoid large (even moderate) throttle excursions during the final landing phase.

The bull's eye chart on Figure 8.2.7-14 shows the change in required ideal delta velocity in addition to the nominal vertical descent delta velocity needed to achieve a range of target redesignation distances from 50 to 150 meters. The figure also shows the additional landing time and the additional propellant required.

It was decided to budget 12 m/s for a 50 m divert capability. The performance cost was reasonably low and it was felt that, for a 20 meter FLO lander base diameter (from landing leg to landing leg), that 50 meters was ample distance to avoid landing on an isolated obstacle. A large field of obstacles (such as the boulder field encountered by the Apollo 11 crew) or a very large obstacle was not considered in this budget as it was assumed that the landing site would be surveyed *a priori*. The picture in Figure 8.2.7-15 shows the FLO lander executing a target redesignation at 100 m altitude. The trajectory and vehicle orientations are fairly accurate representations of the actual simulated trajectory. The box shows highlighted events during the divert.

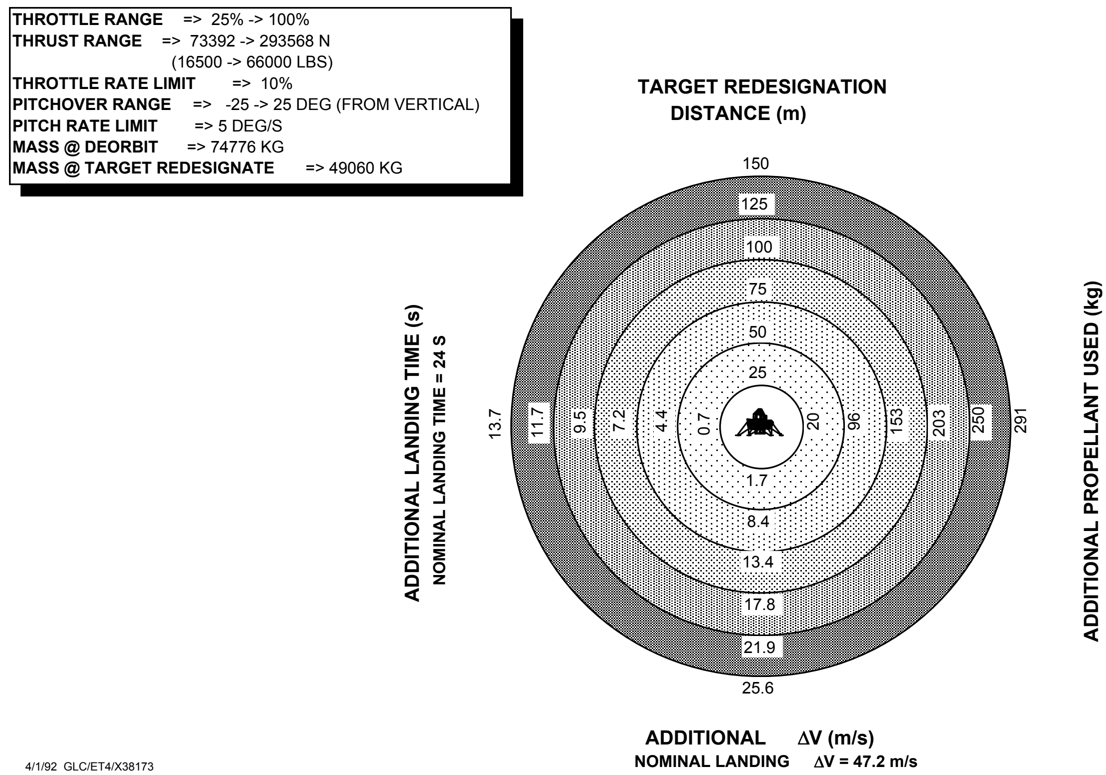
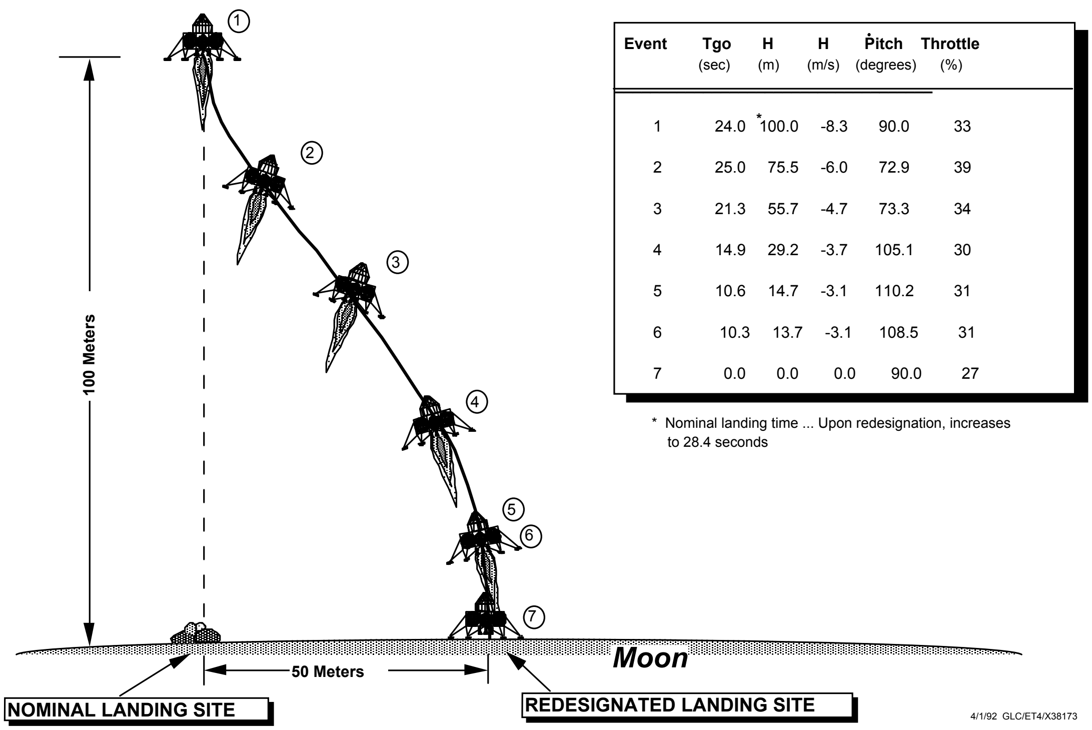


Figure 8.2.7-14: Lunar descent - target redesignation performance costs



Event	Tgo (sec)	H (m)	H (m/s)	Pitch (degrees)	Throttle (%)
1	24.0	*100.0	-8.3	90.0	33
2	25.0	75.5	-6.0	72.9	39
3	21.3	55.7	-4.7	73.3	34
4	14.9	29.2	-3.7	105.1	30
5	10.6	14.7	-3.1	110.2	31
6	10.3	13.7	-3.1	108.5	31
7	0.0	0.0	0.0	90.0	27

* Nominal landing time ... Upon redesignation, increases to 28.4 seconds

Figure 8.2.7-15: Lunar descent - target redesignation flight profile

8.2.8 Stage Disposal From L1

In this section, the disposal of a spent stage or spacecraft from L1 is described. There are three different scenarios to consider: Moon, Earth and the heliocentric space. While the first two cases are studied in this report, the disposal to heliocentric space will be included in a further study. The idea is to perform a unique burn to dispose of the stage while minimizing the propellant (or ΔV) cost. At the same time, the duration of the disposal should be such that the required monitoring lasts less than several days. It is known that for the restricted three body problem, low-cost transfers can be obtained using Capture Dynamics and Chaotic Motions[11] but these small burn transfers require normally more than one to several months to complete. In that case, monitoring the stage or spacecraft for more than several days would be required. Those cases will not been considered in this study. In Figure 8.2.8-1, an overview of the disposal problem is shown. In this scenario, the coasting spacecraft or spent executes a maneuver targeting the stage to a disposal trajectory back to the Earth, to the Moon, or to heliocentric space.

In order to perform this study the following assumptions were made:

- Dynamic model: real ephemeris data (JPL's DE405) with Earth, Moon and Sun as main bodies
- Operational constraints
  - Only one maneuver is allowed.
  - The maximum time of flight is limited to 5 days.
  - Daily disposal maneuvers are evaluated over one year period.
  - Disposal time of day is constant. No coasting is allowed before the maneuver.
- The final goal is to minimize the total ΔV

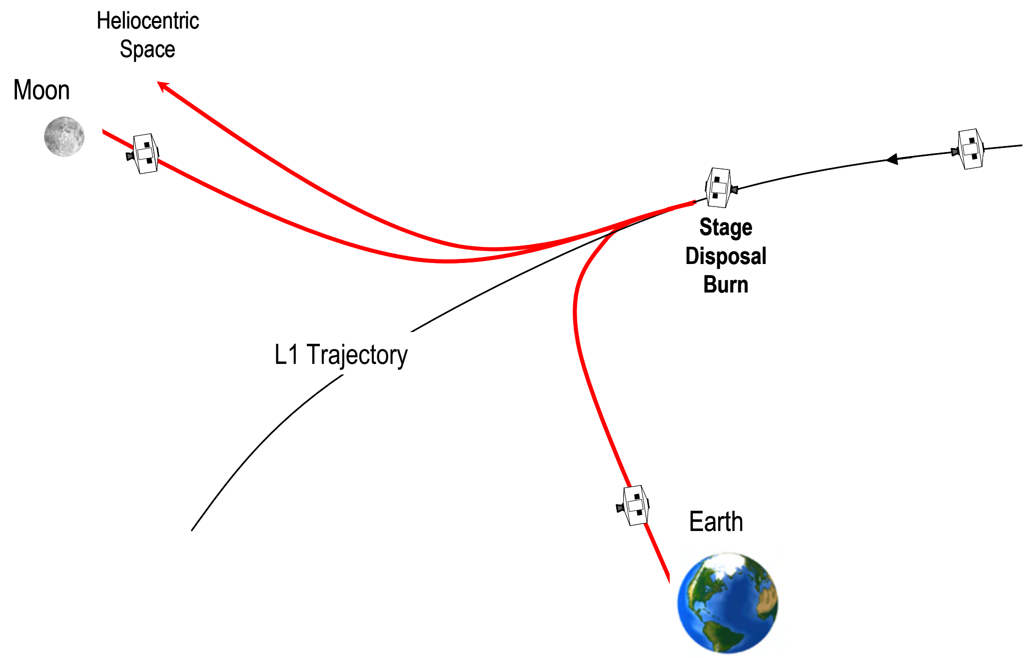


Figure 8.2.8-1: Overview of the stage disposal problem

<u>Results - Stage disposal to the Earth</u>

In Figure 8.2.8-2, the trajectory of the disposed stage to Earth is shown. In order to study the effect of the Sun's perturbation in the problem, stage disposal maneuvers are examined over a span of one year. In Figure 8.2.8-3, the time of flight and the cost of the disposal are shown. The results do not repeat every month as one can expect if an elliptical restricted three body problem model is used. The use of real ephemeris data and the effect of the Sun perturbation seem to affect the results. The transfer time range is from 3.4 to 4.1 days while the disposal maneuver ranges form 692 to 765 m/s.

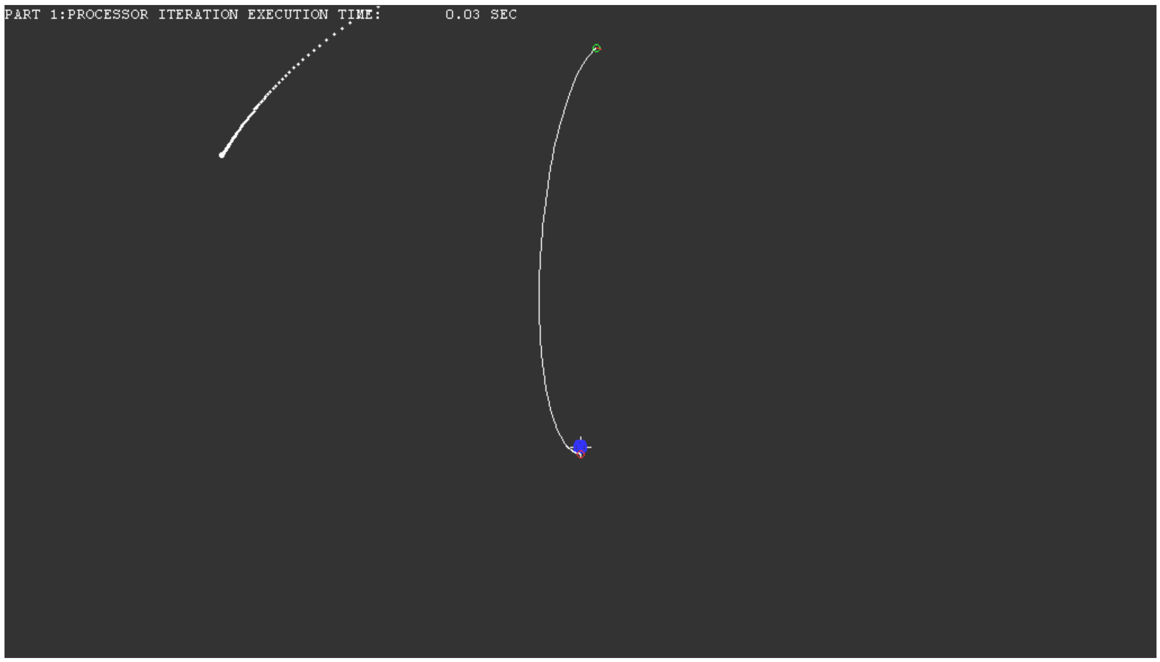


Figure 8.2.8-2: Stage disposal trajectory to the Earth obtained with Copernicus. Transfer time = 4 days. Disposal maneuver ΔV = 682 m/s

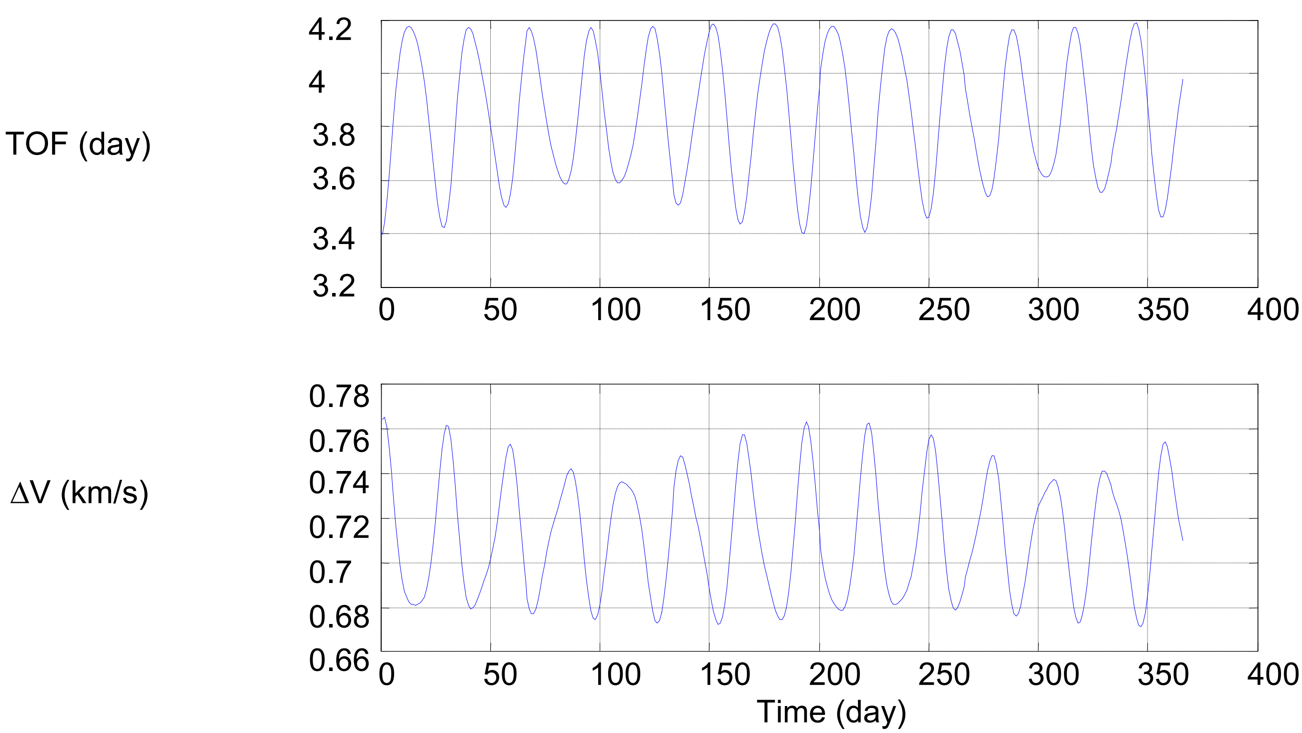


Figure 8.2.8-3: Time of flight and ΔV for the stage disposal to the Earth in the year 2018

Results - Stage disposal to the Moon

In Fig. 8.2.8-4 the trajectory of the disposed stage to Moon is shown. As in the previous case the time spans for one year. In Figure 8.2.8-5, the time of flight and the cost of the disposal are shown. The results do not repeat every month as one can expect if an elliptical restricted three-body problem model is used, the use of real ephemeris data and the effect of the Sun perturbation affect the results. The transfer time range is from 2.5 to 3.0 days while the disposal maneuver ranges form 124 to 136 m/s.

Conclusions

The disposal of the stage or spacecraft to the Moon is much cheaper and takes less time than the one to the Earth. The same conclusion cannot be achieved if the dynamic model does not include the Sun. With L1 being an equilibrium point makes perturbations play a key factor in this problem, particularly the perturbations provided by the Sun.

Although we are using real ephemeris data and we are considering the Earth, the Moon and the Sun in the dynamic model, these results can be improved if we allow the stage to coast before the burn. In this study a maneuver is implemented everyday at the same time of day; this is not optimal and by allowing the spacecraft to coast the DV cost can be reduced. The only limitation to this approach is that the coasting time must be constrained such that the total time of flight is less than five days.

Finally, due to the duration of the transfer arcs, an error dispersion analysis should be considered in order to complete the study. Only nominal trajectories have been obtained in this work and a further study containing errors in the steering angles and misburns should be implemented.

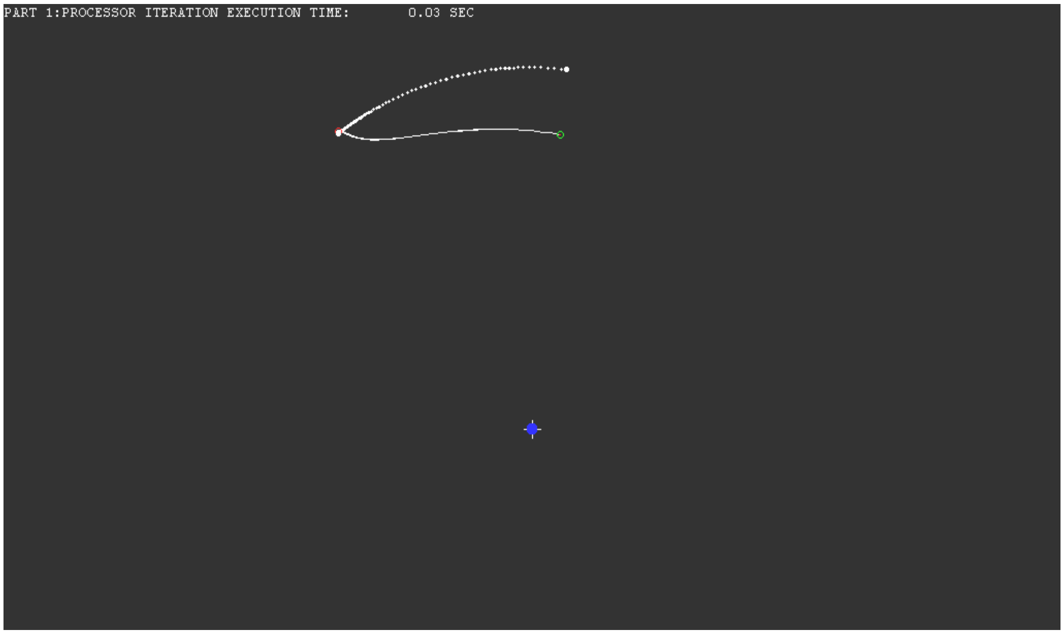


Figure 8.2.8-4: Stage disposal trajectory to the Moon obtained with Copernicus. Transfer time = 2.6 days. Disposal maneuver ΔV = 135 m/s

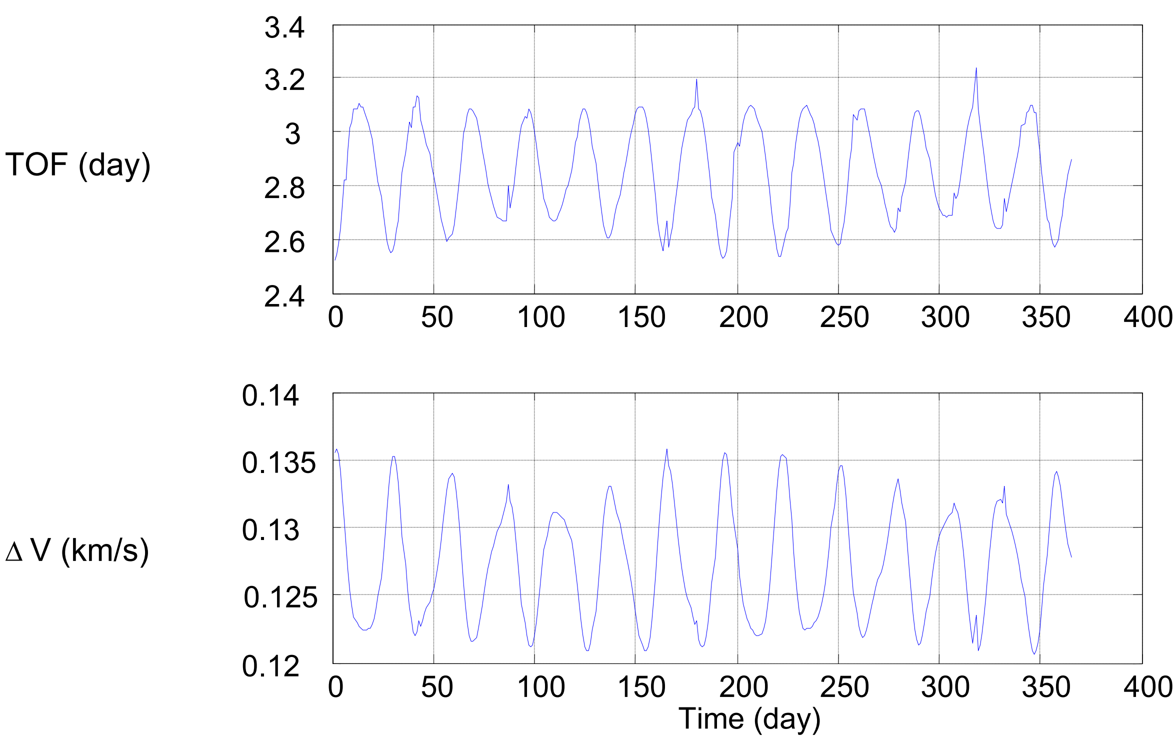


Figure 8.2.8-5: Time of flight (TOF) and ΔV for the stage disposal to the Moon in the year 2018

8.2.9 Earth Return Mission Design

Minimum ΔV Requirements

When the moon is at perigee, the minimum ΔV requirement for a coplanar return is about 750 m/s with a flight time close to 4 days. Flight times that are longer or shorter than 4 days require additional ΔV.

Lighting Conditions At Earth Return

The lighting condition at the Earth landing site depends on the Moon's phase (illumination of the Moon) at the time of lunar departure. In general, the vehicle lands in daylight when the Moon's phase travels from first quarter to last quarter. Night landings occur when the Moon's phase travels from last quarter to first quarter. The vehicle must be capable to return to Earth in either daylight or darkness, since the L1 departure time cannot be controlled due to delays or an abort during the mission.

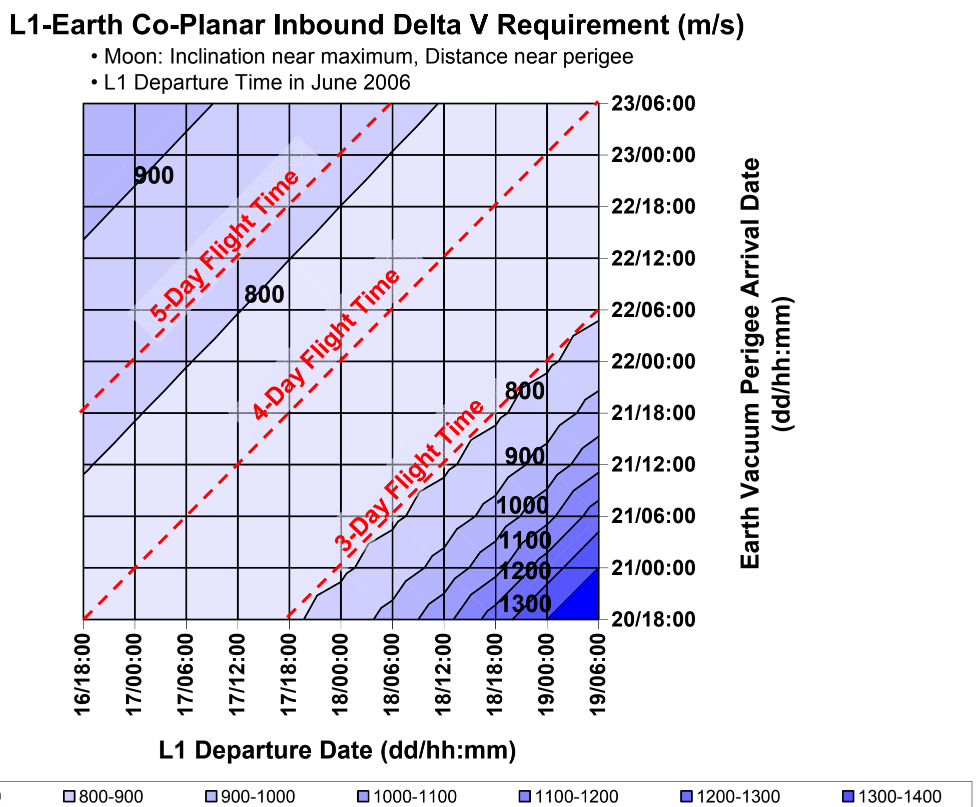


Figure 8.2.9-1: Earth Return ΔV

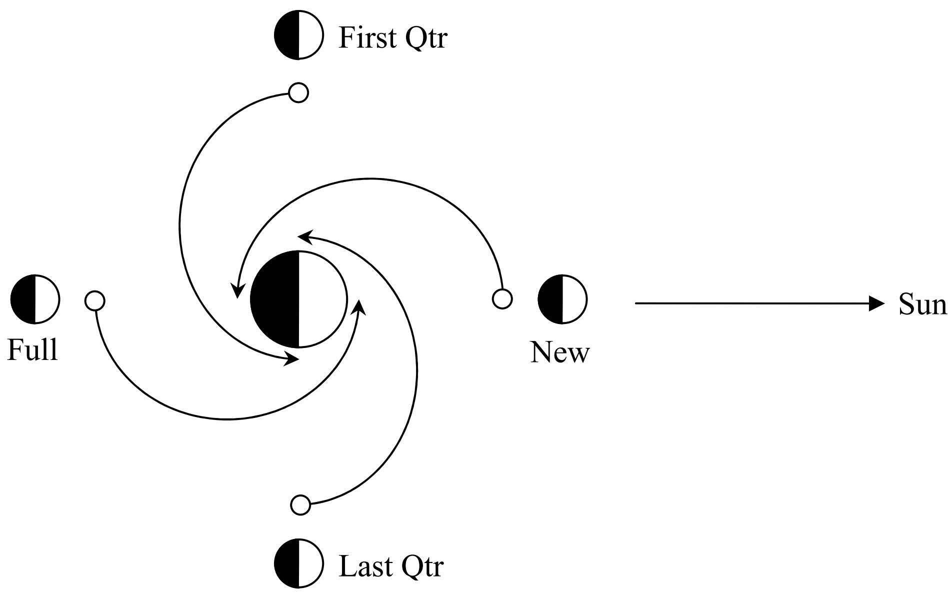


Figure 8.2.9-2: Phases of the Moon and Lighting at Earth Return

Control Variables

Three parameters can be varied for targeting a landing site on the Earth: longitude, latitude, and the azimuth angle.

Longitude Control

The vacuum perigee of the return trajectory is fixed at a point above the Earth's surface and corresponds to a geographic longitude depending on the return flight time. Since the Earth rotates 360° in a 24 hour period, full accessibility of Earth longitudes can be achieved by varying the flight time ±12 hours. This variation is used to align the longitudes of the Earth landing site and the vacuum perigee point. The 24 hour variation incurs a small difference in the trajectory's transfer angle.

Latitude Control

The Earth arrival latitude depends mainly on the Moon's declination (with respect to the Earth's equatorial plane) at L1 departure. In general, latitude control can be achieved by allowing the time of L1 departure to vary. In case of an abort or delays during the mission, the nominal range of latitudes may not be accessible. Latitude also varies with the return flight time. Shorter flight times result in a wider range of latitudes, while longer flight times result in a narrower range of latitudes. The range of latitudes also varies with the distance of L1 to the Earth. The range of latitudes narrows as L1 approaches perigee; the range widens as L1 approaches apogee.

Azimuth Control

The return azimuth is the vehicle's direction of motion and is controlled at L1 departure. This control is present when the vehicle has the ability to perform a plane change at L1. Azimuth control is required to maximize the range of accessible latitudes. The maximum latitude is achieved by a polar return trajectory. The worst case departure azimuth change is 118.6°. This number comes from adding 28.6° (the maximum inclination of the Moon's orbit) to 90° (inclination for a polar orbit) when the Moon is at 0° declination.

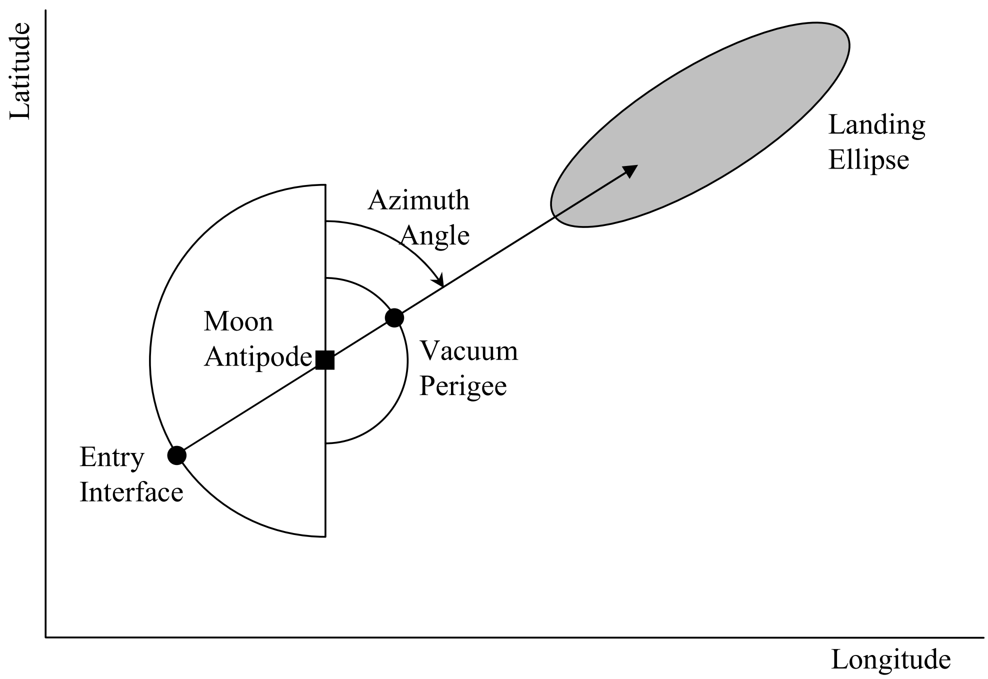


Figure 8.2.9-3: Entry to Landing Illustration

Landing Footprint

The landing footprint is an area of potential locations that the vehicle can touch down. The size of the footprint is dependent on the vehicle's lift-to-drag (L/D) ratio and atmospheric dispersions (e.g., the Apollo reentry capsule had an L/D of 0.3 with a footprint length of ~1389 km). Downrange capability is the distance along the trajectory's ground track measured from entry interface to landing (e.g., Apollo's downrange capability is ~2380 km as shown in Figure 8.2.9-4).

When longitude control is applied, an overlapped region is formed from the footprints covering flight times from 3.5 to 4.5 days. This flight time interval was derived by applying ±12 hours to the nominal return flight time of 4 days. The shortest overlapped region occurs when L1 is at apogee. Figure 8.2.9-5 shows that the location of the vacuum perigee from the antipode varies with flight time, thus the overlapped region is formed. Figure 8.2.9-6 shows how the overlapped region varies with L1 distance. When azimuth control is applied, the area of the 24 hour continuously accessible region stretches down to the antipode line.

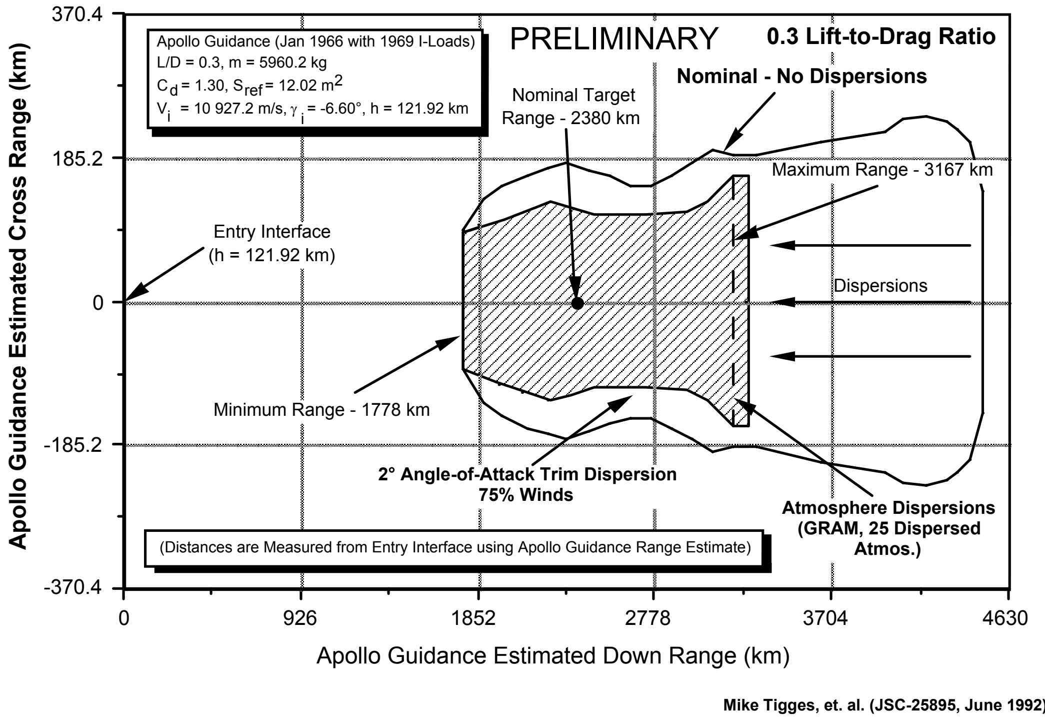


Figure 8.2.9-4: Normal and Dispersed Apollo Footprint

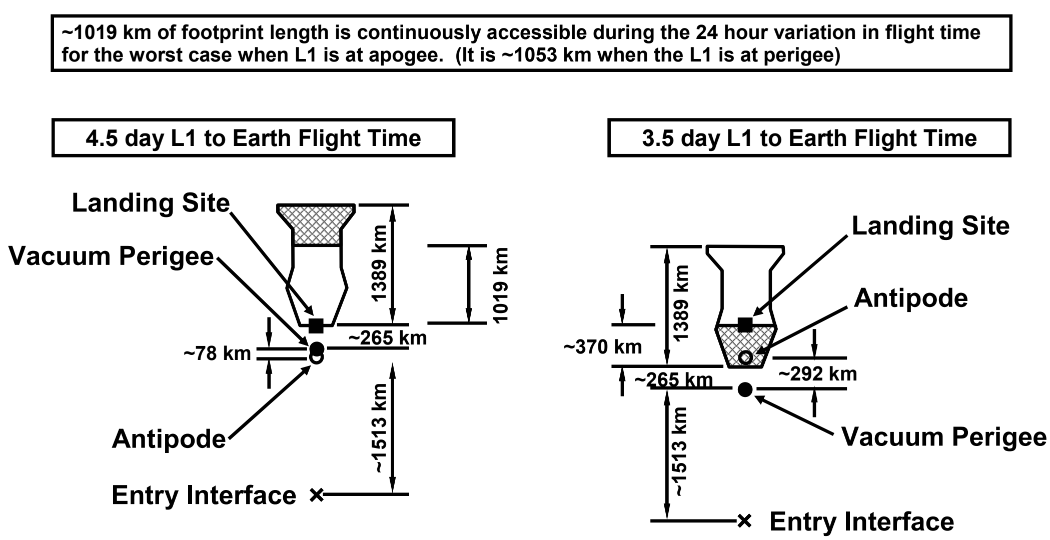


Figure 8.2.9-5: Effective Apollo Footprint for a 24 Hour Flight Time Variation

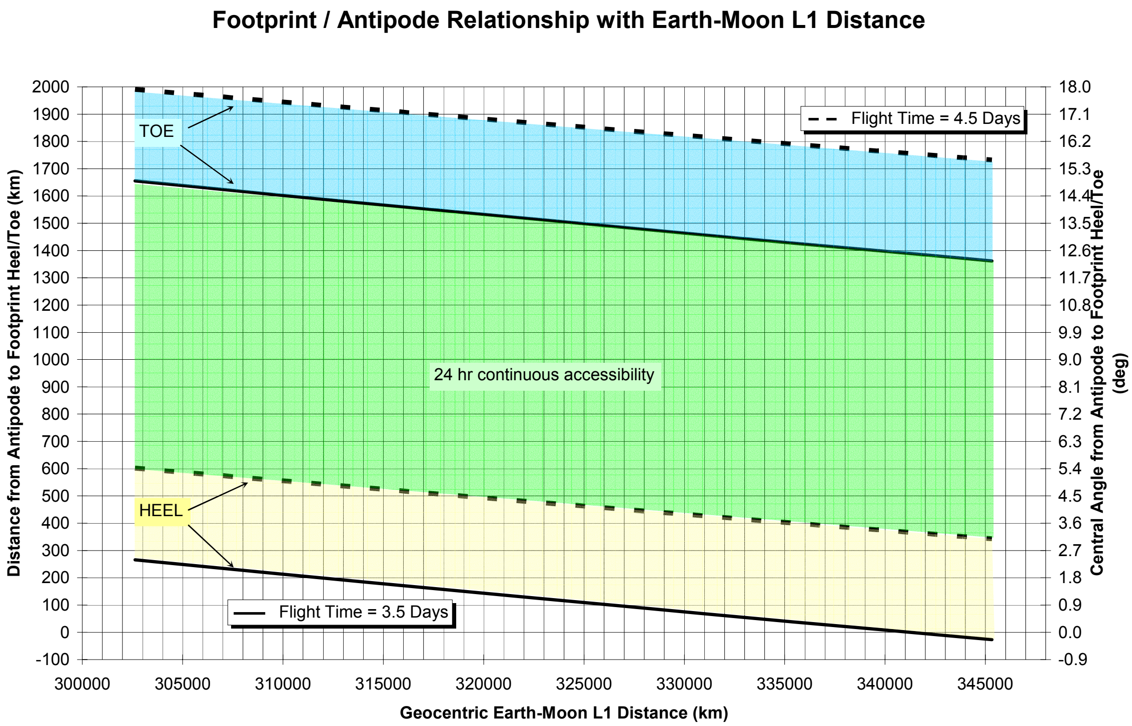


Figure 8.2.9-6: Effective Apollo Footprint with respect to L1 Distance

<u>Latitude Band</u>

Putting together the longitude, latitude, and azimuth controls and the landing footprint, a range of latitudes, or a latitude band, can be drawn around the Earth for a specific L1 departure date (the line of circles). The daily variation of the latitude band can also be plotted to show which latitudes will be available during the mission, in case of delays or an abort during the mission (the sinusoidal curve).

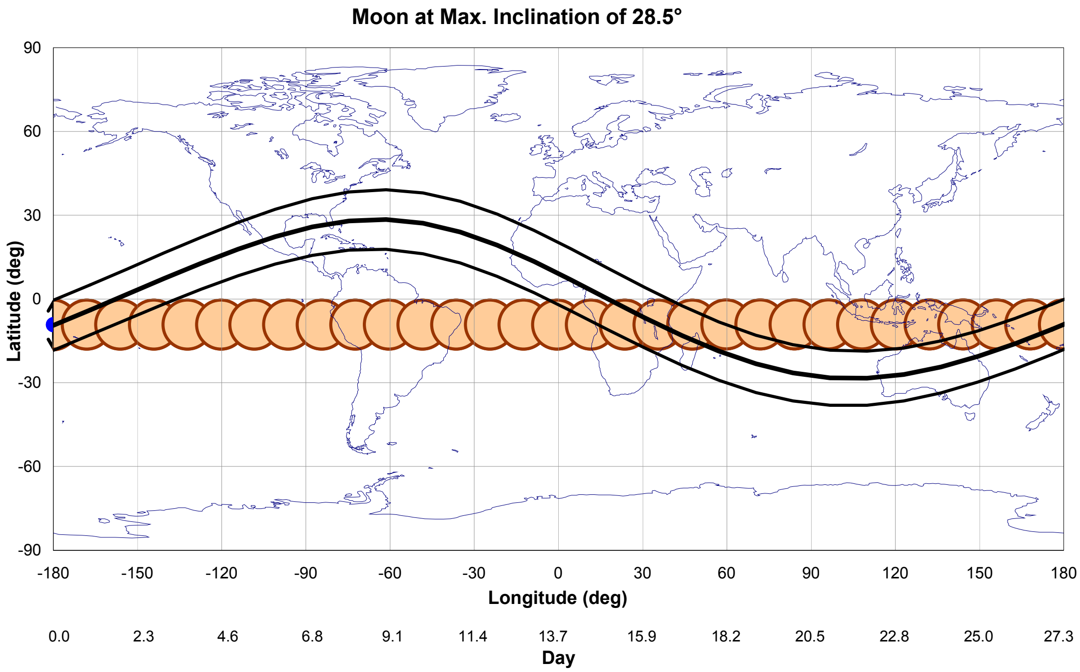


Figure 8.2.9-7: Daily Variation of the Landing Latitude Band

<u>Landing Site</u>

Considerations in choosing a landing site will need to include safety, cost, and site accessibility (opportunities). The following come from the First Lunar Outpost study on landing and recovery options.

Water Landing

Safety	The vehicle sinking is a risk, recoveries are more complex, specialized personnel are required, night recoveries should be avoided, emergency self-egress is risky (must deploy a raft and exit a moving vehicle), and motion sickness can occur.
Cost	Costs for refurbishment are slightly higher than land landing systems. Recovery costs for coastal ocean and inland water landings are similar to land landing. Open ocean recovery operations are more costly.
Landing sites	Two recovery fleets (one in the Pacific and the other in the Atlantic) are required for open ocean landings. The fleets will either be deployed to a designated area (nominal) or follow the latitude of the moon's antipode (for an anytime abort scenario). Coastal ocean and inland water landings will require several landing sites depending on the vehicle characteristics.

Land Landing

Safety	Vehicle sinking is not a risk, landing system must be robust to ensure crew safety, and emergency self-egress is less risky.
Cost	Recovery costs are slightly lower compared to water landings. A vehicle can be reused for more missions.
Landing sites	Depending on the vehicle landing footprint, a specific number of sites (primary and alternate) will need to be identified, outfitted, and manned during a mission. In a past study, the First Lunar Outpost showed that a minimum of 3 primary sites would be required. Also, at least 3 or more alternate sites will be needed.

8.2.10 Earth Return Aeroentry and Aerobrake Performance

Introduction

Two entry scenarios will be assessed for returning a crew capsule from the moon. A Direct Entry (DE) scenario will utilize the same technique exercised during the Apollo program where the entry vehicle intercepts the Earth's atmosphere from the Lunar return trajectory and directly enters to a desired landing site (nominally a water site). A second scenario will assess an Aerocapture Entry (AE) technique for aerodynamically skipping into a LEO where phasing for a predetermined time will enable a second entry to a desired landing site (nominally a ground site). This scenario requires a perigee raising burn and a deorbit burn.

Both entry scenarios require proper targeting at the moon to insure that the proper Earth Entry Interface conditions (EI) of velocity, flightpath angle, latitude, longitude, and altitude are achieved at the proper time. The proper EI conditions are determined from knowledge of the desired landing point, the Range/Crossrange capability of the vehicle, and the vehicle and crew constraints (thermal protection system rates and loads, total aerodynamic acceleration, and dynamic pressure).

For the DE scenario, landing will occur approximately at the antipode of the Lunar return trajectory. The antipode (which is always within about 0 to 6 degrees of lunar return orbit perigee), can vary over a wide range (+-18.3 to +-28.6 degs). The Earth longitude of the antipode point is targeted at LPD (LOD for LOR missions) nominally in proximity of a water recovery force. No extra propellant is required for this scenario other than a small Trans-Lunar Mid-Course-Correction (MCC) maneuver. If a land landing is desired, previous studies indicate a minimum of 11-20 landing sites over the globe with +-90 degree entry azimuth control (fuel cost at LOD).

A third approach (which will not be addressed in this report) should be mentioned which permits a vehicle, with an L/D in the 0.5 range, Continental United States (CONUS) landing capability. This approach does require "Up Control" guidance (an un-validated Apollo guidance capability), and possibly a small exo-atmospheric second entry correction targeting burn, to facilitate the second entry. Long range targets can theoretically be achieved; however, the 0.5 L/D "requirement" may preclude the use of an Apollo capsule shield configuration.

Assumptions

The same common vehicle will be used for both the DE and AE entry simulations. Tables 8.2.10-1 and 8.2.10-2 provide the vehicle initializations. A 62 standard atmosphere model was used for the flights.

Common Capsule Design Properties:	
Weight	7.551 t
Diameter	5 m
Aero	
C_L	0.39
C_D	1.29
L/D	0.3
W/C_DA	75 kg/m^2 (61 psf)

Table 8.2.10-1: Vehicle Definition

Entry Conditions (121.92 km, 400Kft):

Lunar

Inertial Velocity = 11 km/sec (36,200 ft/sec)

Inertial Flightpath Angle = -6.32 deg

LEO

Inertial Velocity = 7.9 km/sec (25,900 ft/sec)

Inertial Flightpath Angle = -1.63 deg

Table 8.2.10-2: Earth Entry Interface Initial Conditions

Results

The following charts (Figure 8.2.10-1 – 8.2.10-5) provide the data for the AE and DE return-to-Earth flight simulations. Each plot contains three flights. The Direct Entry plot is a simulated return from the moon, while the Aerocapture Entry plot simulates an aerocapture trajectory followed by a Leo entry once proper ground landing site phasing has been achieved. The aerocapture trajectory targeted for a 480 km exit orbit apoapsis altitude. The nominal exit perigee is 56 km, requiring a 125 mps circularization burn to achieve circular orbit.

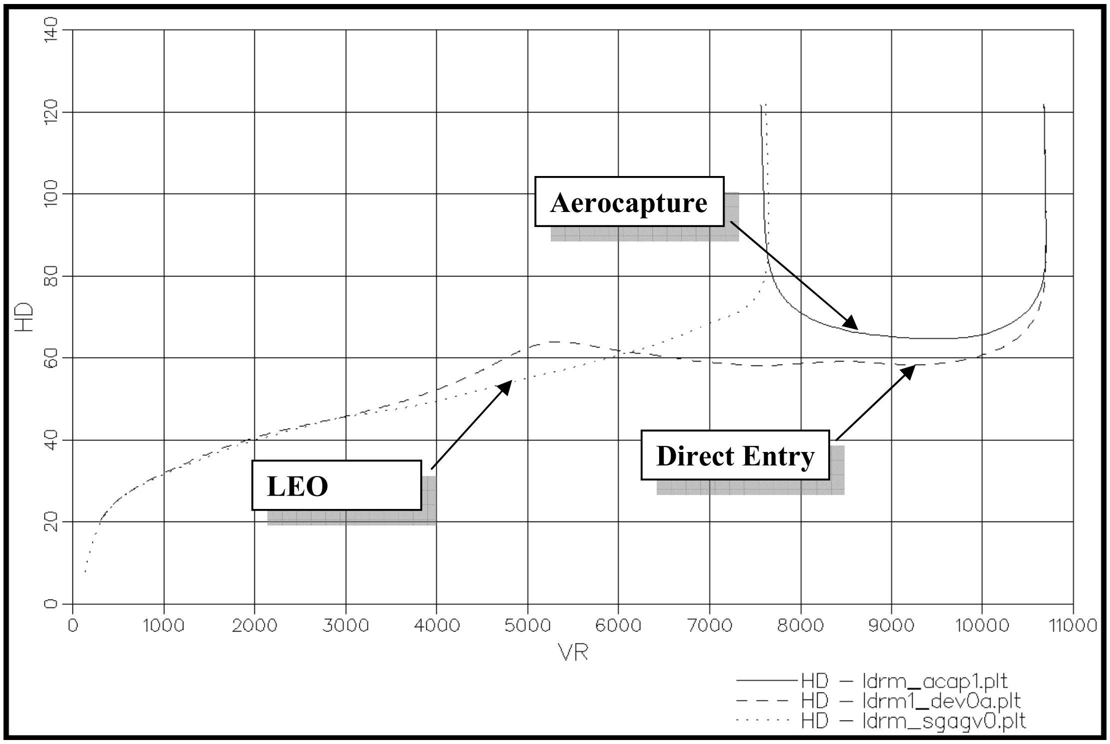


Figure 8.2.10-1: Altitude (km) versus Relative Velocity (m/s)

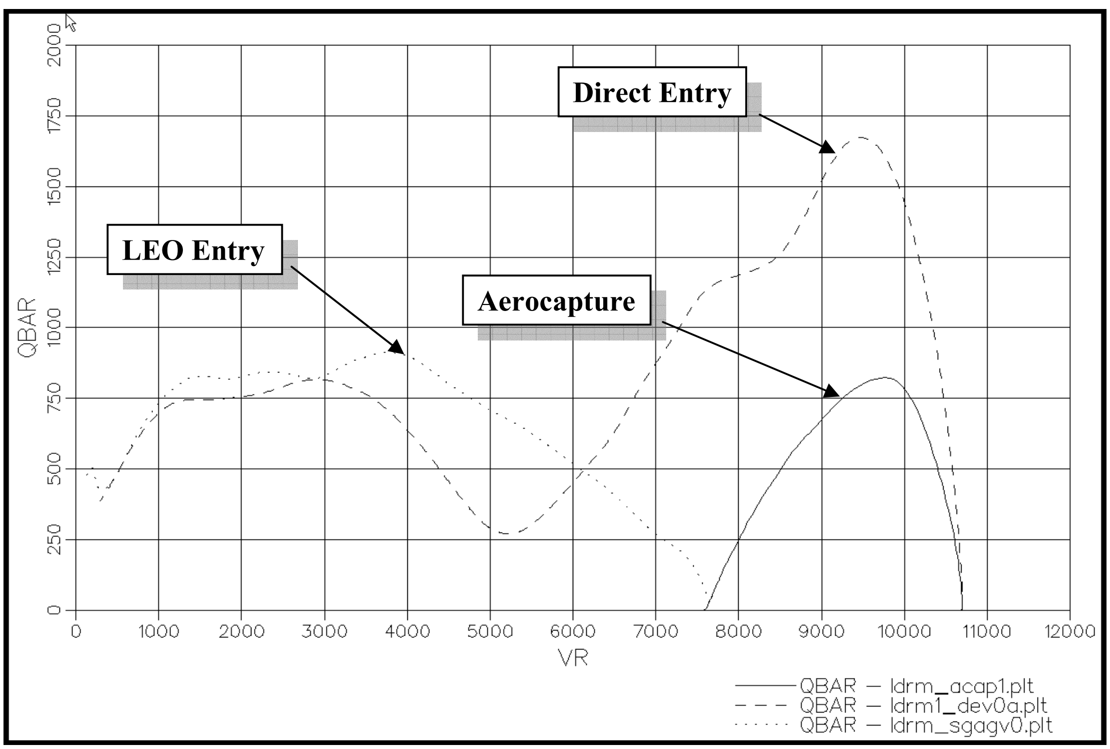


Figure 8.2.10-2: Dynamic Pressure (kg/m2) versus Relative Velocity (m/s)

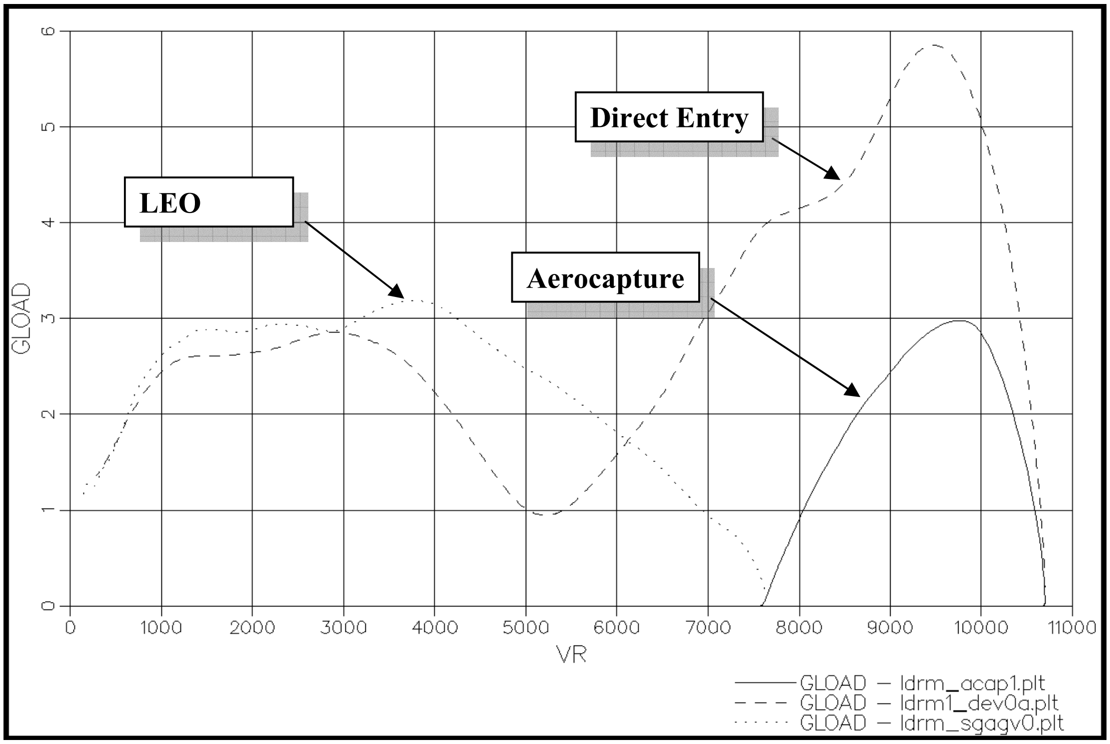


Figure 8.2.10-3: G-Load (Gs) versus Relative Velocity (m/s)

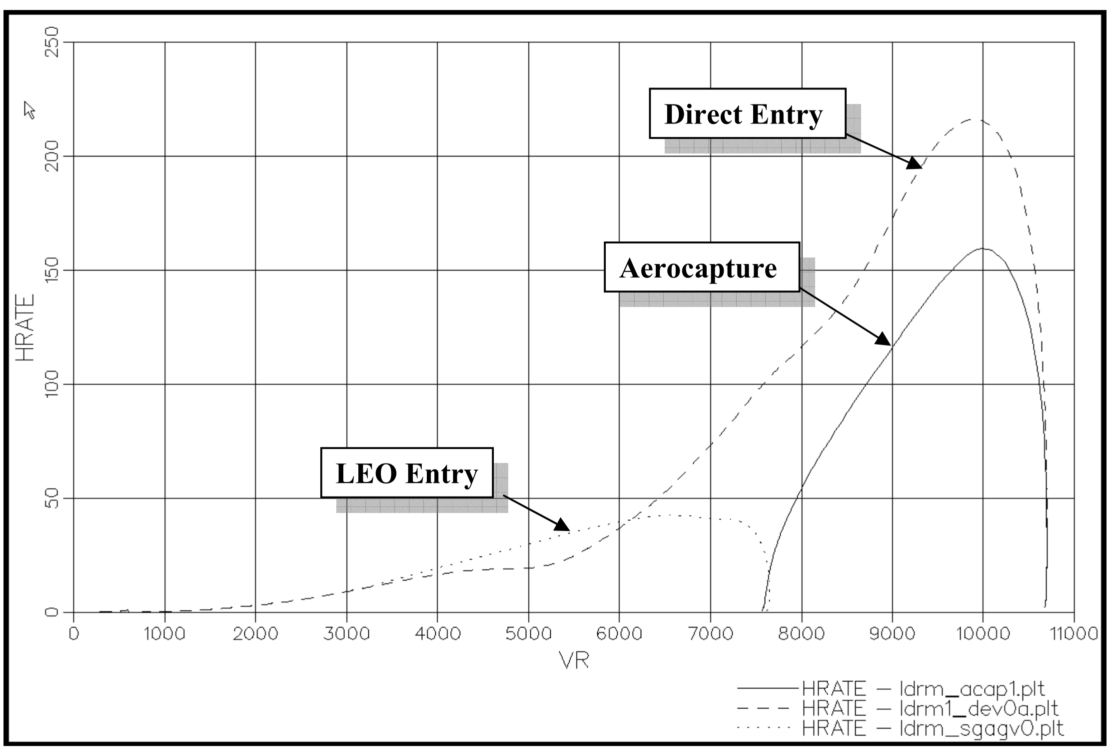


Figure 8.2.10-4: DKR Heat Rate (W/cm2) versus Relative Velocity (m/s)

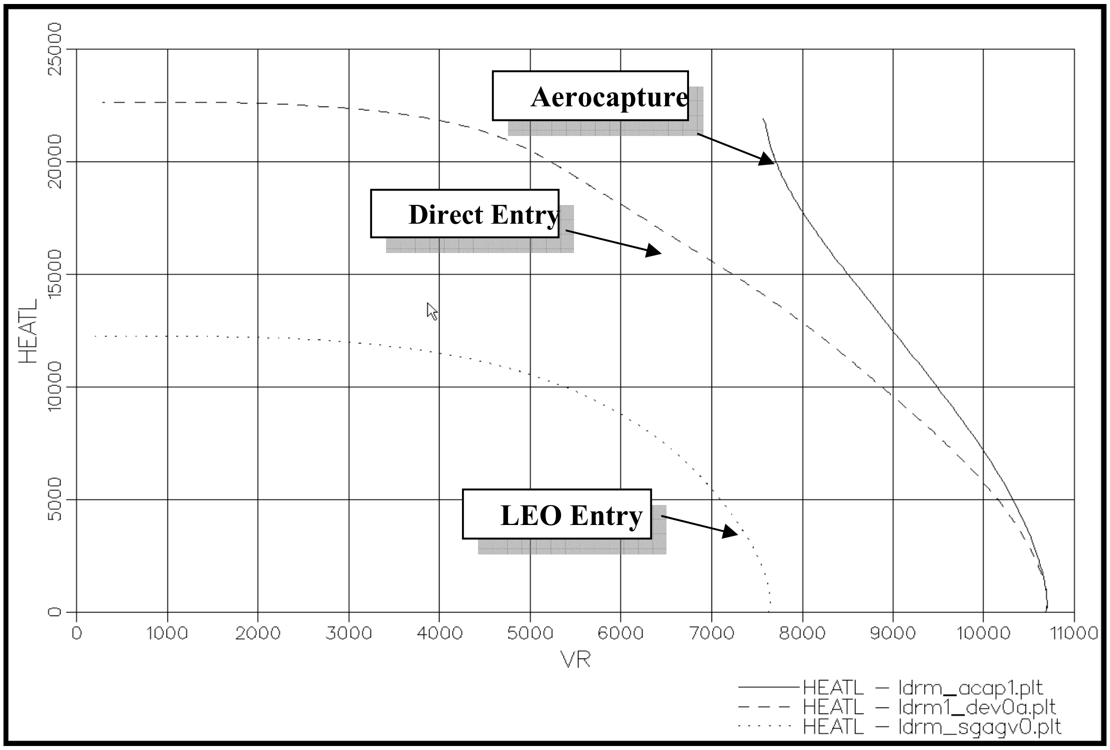


Figure 8.2.10-5: Heat Load (J/cm2) versus Relative Velocity (m/s)

Conclusions

An actual strategy for returning crew to Earth could combine the DE and AE strategies, depending on the actual ranging requirements for the given return scenario. For example, a mission with a short ranging requirement and favorable antipode positioning could permit a DE flight to a Continental United States (CONUS) land landing site. This is the simplest entry scenario, providing a minimum impact on vehicle consumables (power, ECLS, propulsion), however it also provides the most extreme thermal and load environment (dynamic pressure, heat rate, and aerodynamic acceleration). For longer ranging requirement mission scenarios, an AE strategy could be employed. Under this scenario a vehicle would enter and then exit the Earth's atmosphere.

As stated previously, a third alternate approach for Earth return entry was available but never flown during Apollo. In this scenario, long range targets are achieved by targeting exit conditions to achieve a second entry point that immediately follows the first entry, without requiring an orbit insertion maneuver. This approach was outside the navigation and control capabilities of the original Apollo vehicle; however, improvements in navigation, and incorporation of a second re-entry targeting maneuver successfully simulated for similar Mars entry missions, warrant future consideration of this scenario.

8.2.11 Mission Design & Performance Requirements for LPR, LOR, and LSR Missions

While the LDRM-2 study primarily focused on the L1 rendezvous or LPR mission and its TRM, it did also direct JSC to provide a comparison of LOR mission performance. This section compares and contrasts the ΔV costs and mission design considerations for ten different missions comprised of eight different LOR missions, the LPR TRM, and one LSR mission (added for the sake of completeness). An underlying constraint for all missions provides for anytime abort from the lunar surface. Namely, the crew retains the capability to immediately launch from the lunar surface to an alternate or backup habitat in the event of emergency. Specifically, it is taken to mean that a flight crew faced with a life-support system failure or a medical emergency at the landing site should not have to wait longer than three times the period of the lunar phasing or rendezvous orbit to initiate a LOD maneuver that will return them to Earth atmospheric entry and landing.

Notwithstanding the elliptical lunar orbit and its accompanying variations in apogee and perigee velocities, the fixed (Earth-L1-Moon) geometry employed by the LPR mission results in consistent nominal and abort performance requirements. At any time, and for essentially the same ΔV cost, the crew to could launch from the surface back to L1 by choosing any launch azimuth to establish a 100 km phasing orbit with a selectable inclination and associated right ascension of the ascending node that is properly aligned for a coplanar transfer to L1. The slow rotation rate of the moon (i.e., about 5 m/s at the lunar equator) provides minimal impact on the selected departure phasing orbit inclination.

Note that the various ΔV costs for these ten missions provide the performance requirements needed for vehicle mass sizing. They are designed to provide the minimum performance requirement subject to the overall anytime lunar surface abort constraint and a set of primary mission constraints listed in Table 8.2.11-1. The bold face constraints in this table indicate a constraint change from the previous mission. The LDRM study focus was a 7 day mission. This report also includes 3 and 11 day surface missions in order to envelope all lunar missions with daylight landing and launch. The 3-day mission mimics the longest surface stay Apollo mission. The 11-day mission reflects the longest surface stay that could accommodate a lunar landing using the Apollo landing lighting constraints (i.e., with the sun lies approximately 7°-22° off the lunar horizon, behind the lander) and a daylight launch.

Mission 1 is intended to provide an Apollo type mission as a reference. This mission encompasses close to the length of the longest Apollo surface stay and maximum landing site latitude (26.1° for Apollo 15). The ground launch (Canaveral) provides for daily launch opportunities and a minimum geocentric plane change at lunar arrival. All missions employ an expendable or Earth-based lander freeing the mission from a potentially more expensive lunar rendezvous with a pre-established parking orbit at the Moon. Missions 2, 3, and 4 are all ground-launched LOR missions with global lunar landing site access. The only differences in the mission constraints among mission 2, 3, and 4 are the limits on total surface stay time (i.e., 3, 7, and 11 days). These missions were designed to provide the cheapest (i.e., minimum ΔV) cost by sacrificing on-orbit time, with the CEV loitering in lunar orbit long enough to effect inexpensive coplanar maneuvers while maintaining the capability for anytime abort off the lunar surface. LOR missions 5, 6, and 7 also possess global lunar landing site access for 3, 7, and 11 day surface stays, respectively. However, these missions differ from missions 2, 3, and 4 in that they depart from a fixed 28.7°

LEO parking orbit. Mission 8 adds a long duration lunar surface stay (i.e., > 28 days). Note that the maximum mission ΔV performance is encompassed by a surface stay of approximately 28 days or more as a result of the lunar orbit period. For longer surface stays, the orbital geometry is repeated. Mission 9 represents the TRM and serves as a convenient reference. Finally, mission 10 is a LSR mission (much like that of the First Lunar Outpost mission design study performed at JSC[12]).

	MISSION	1	2	3	4	5	6	7	8	9	10
CONSTRAINTS	Mode	LOR	LOR	LOR	LOR	LOR	LOR	LOR	LOR	**LPR**	**LSR**
	Launch	Canaveral	Canaveral	Canaveral	Canaveral	**28.7° Orbit**	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	**Canaveral**
	Lander	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable
	Landing Site Latitude	30°	**Global Access**	Global Access	Global Access	Global Access	Global Access	Global Access	Global Access	Global Access	Global Access
	Surface Stay Time	3d	3d	**7d**	**11d**	**3d**	**7d**	**11d**	**≥ 28 d**	≥ 28 d	≥ 28 d
	Comment	Apollo Type	Minimum ΔV with Lunar Loiter			Non-Minimum ΔV - No Lunar Loiter				LDRM-2 BRM	Min. ΔV - No Lunar Loiter

Table 8.2.11-1: Mission Constraints for 10 Selected Missions

Mission Configuration Details – Original Study

The mission parameters were determined with the intention of providing the best possible mission design with the intent to minimize the delta-V of the required on-orbit vehicles. That being said, the mission designs also contain worst-case parameter values intended to provide the mission with a robustness (e.g., arrival at the Moon and allow for crew safety-related capability). The EOD maneuvers for all ten missions are coplanar. While a rigorously optimized solution may result in a small EOD plane change component, the comparative parametric analyses in this report are well served with a near optimal coplanar maneuver. Table 8.2.11-2 provides a quick look at the mission profile characteristics for missions 1 through 10.

Mission 1. Mission 1 constraints, as listed in Table 8.2.11-1, are included in a "bat chart" of the mission profile in Figure 8.2.11-1. This short stay (3 day) ground-launched mission is designed to achieve a maximum lunar landing site latitude of 30° and provides an Apollo-like mission for comparison purposes.

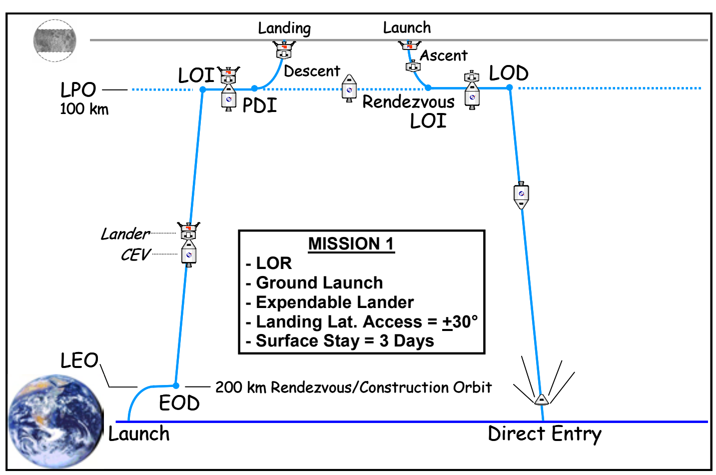


Figure 8.2.11-1 – LOR Mission 1

Variation in the launch azimuth, also reminiscent of Apollo, ranges from 72° to 108° for range safety considerations, depending on the time and date of launch. A 72° launch azimuth places the spacecraft onto a 200 km circular phasing orbit with a 32.54° inclination. The maximum geocentric Earth-Moon (EM) inclination of 27.4° arises from targeting lunar arrival when the Moon, in its minimum geocentric orbit inclination (18.3°), is at its apex or point of highest geocentric latitude. This results in a 3025 m/s EOD cost for a 4-day transfer to the Moon with arrival at lunar perigee. The 4-day flight time represents a near minimum ΔV cost subject to lunar arrival with the Moon at a distance resulting in the largest ΔV cost, namely perigee lunar arrival. Note that, at Earth departure, the outbound trajectory is elliptical relative to Earth, but at lunar arrival, the incoming trajectory is a hyperbola.

The combination of lunar arrival distance (i.e., perigee) and flight time results in a lunar arrival V_∞ vector magnitude of 903 m/s. The desire for a minimum lunar ascent wedge angle (after the surface stay) and a 30° maximum possible landing site latitude results in a 31° lunar parking orbit inclination. The worst case relative arrival declination of the V_∞ vector of 50° stems from the sum of the parking orbit inclination (31°) and the worst case declination of the incoming V_∞ vector (19°). Note that the 19° arrival V_∞ vector declination stems from the sum of approximately 10° due to a worst case angle (perpendicular) between the Moon's and the spacecraft's geocentric velocities at lunar arrival plus approximately 7° to account for lunar libration in addition to 2° to account for 2nd order variations.

Retrograde parking orbits are desirable for manned missions as they provide better (lower ΔV) abort options. For the case of mission 1, the maximum 30° landing site can be accommodated by a retrograde orbit inclination of 149° which is merely the supplement of the 31° parking orbit inclination that provides a minimum ascent wedge angle, hence minimum ascent ΔV.

The 50° relative arrival V_∞ vector declination along with a 100 km circular lunar parking orbit target results in a 1143 m/s LOA ΔV. The LOA maneuver consists of a 3-impulse sequence with a 24-hour transfer orbit period. For this case, the 3-impulse sequence possessed a lower ΔV cost than a single impulse. The 100 km circular LPO poises the Lander for a coplanar powered descent to the surface at a cost of 1881 m/s with the CEV remaining in the 149° parking orbit. Note that the powered flight descent and ascent ΔVs are based on past work[8]. After a 3-day surface stay, a 1850 m/s powered ascent sequence takes the crew from the surface to the awaiting CEV in LPO (100 km circular orbit altitude). In order to accommodate an anytime abort from the surface, the 149° parking orbit target from a maximum 30° landing site requires a maximum ascent plane change of about 1° at a cost of 29 m/s. The Lander and CEV perform rendezvous and docking in the 100 km LPO. After crew transfer from the Lander to the CEV, the Lander is jettisoned for CEV return to Earth.

The return phase targets a direct Earth entry to an Earth vacuum perigee (EVP) of 38 km (based on Apollo 17 mission design). A geocentric Moon-Earth (ME) transfer plane of 40° provides for coverage of favorable Earth landing latitudes. For the Moon's position, at lunar departure, at the apex of its orbit, the 40° geocentric transfer orbit inclination results in a 36.2° ME transfer orbit inclination with respect to the EM plane. This combined with a shorter 3.5-day return flight time from the Moon at perigee results in a slightly larger LOD V_∞ of 952 m/s. This V_∞ combined with a relative departure declination of 50° produces a 1152 m/s LOD ΔV maneuver which places the CEV on an Earth return path.

Missions 2 – 7. The architecture for these LOR missions include a CEV and Lander, each boosted to LEO to rendezvous with pre-launched EDS stages. In this mission scenario, four launches are required, with about 2 weeks between launches. While this approach results in a fixed construction orbit with multiple LEO rendezvous, the CEV departure parking orbit can be pre-determined on the ground to provide an optimal (e.g., minimum IMLEO) mission. This approach provides the performance benefit of a ground launched mission. If the Lander is to be pre-emplaced, this architecture may tolerate Lander EOD delays by recycling a missed departure to the next opportunity (about 9-10 days later on average, depending on the LEO parking orbit inclination and altitude). This flexibility stems from, among other things, the Lander being unmanned until CEV arrival, rendezvous, and subsequent crew transfer. However, mission success will heavily depend on the EOD for the CEV occurring within the design injection window. In this mission design case, the CEV and Lander are assumed to fly to the Moon in a mated configuration, much the same as the Apollo missions. The constraints for missions 2, 3, and 4, as listed in Table 8.2.11-1, are included in a "bat chart" of the mission profile in Figure 8.2.11-2.

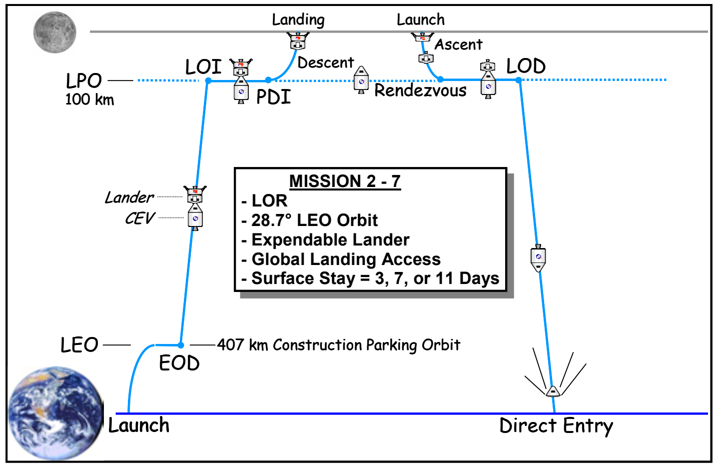


Figure 8.2.11-2 – LOR Missions 2-7

Missions 2, 3, and 4 have the same mission profile with the exception that the lunar surface stay time varies from 3 to 7 to 11 days, respectively. The focus of the original LOR mission was a 7-day surface stay. A 3-day surface stay was added as a comparison to an Apollo stay time. An 11-day stay was also added which addresses a full lunar day stay with Apollo-like landing conditions (i.e., with the Sun 7° to 20° off the horizon behind the spacecraft at landing) and an ascent during the lunar day (approaching lunar sunset).

For missions 2-4, the plane changes for LOA and LOD (at the Moon) are eliminated by performing a loiter in lunar orbit until coplanar maneuvers can be used. This is done to achieve the minimum possible mission ΔV at the expense of additional lunar on-orbit time. Missions 5, 6, and 7 also represent 3, 7, and 11 day surface stay missions, respectively. However, missions 5-7 differ from missions 2-4 in that they carry enough plane change performance (ΔV) to accomplish the maneuver sequences with no on-orbit lunar loiter. While the design for missions 5, 6, and 7 endeavors to minimize overall ΔV, it sacrifices some performance in favor of the ability to perform lunar on-orbit maneuvers with no required loiter time to remove LOA or LOD plane change components.

A near due-east launch places the CEV (with required EDS booster stages) into a 474 km, 28.7° inclination LEO parking orbit. Note that the original performance analysis for missions requiring construction orbits used a 407 km circular parking orbit altitude. Later, the altitude of this orbit was raised to 474 km to provide a phase repeating orbit for the purposes of maximizing the occurrences of a daily launch opportunity as well as minimizing required on-orbit phasing time via a consistent daily rendezvous phase angle. For purposes of this initial parametric mission design study, the difference between these two possible Earth rendezvous orbits (ERO) has a minimal effect on overall mission ΔV. The 28.7° target inclination is designed to extend the launch window. Rendezvous considerations are discussed in more detail in section 8.2.2.

For the entire mission set (i.e., 2-7), the maximum geocentric transfer orbit inclination of 57.3° results from a worst case orientation of the 28.7° LEO departure parking orbit combined with a lunar arrival at nodal crossing with the Moon at the maximum in its 18.6 year lunar inclination cycle (i.e., 28.6°). This worst case orientation is represented by the sum of the LEO departure orbit and the lunar inclination (i.e., 28.7°+28.6°). A 4-day flight time provides a minimum transfer ΔV for lunar arrival with the Moon at perigee. For this case, the 3074 m/s EOD takes the spacecraft to a lunar arrival V_∞ vector magnitude of 986 m/s. Depending on the lunar landing site latitude and longitude, the lunar parking orbit could range from 90° to 180° and would be tailored to the landing site latitude. Note that, again, a retrograde parking orbit is selected for its favorable abort performance costs.

For the case of missions 2-4, the LOA maneuver establishes a preferred orbit inclination for a given landing site latitude. A 0° landing site latitude results in a worst case 19° relative declination of the arrival V_∞. A less expensive 3-impulse arrival (compared to a 1-impulse arrival) with a 1-day intermediate transfer orbit period produces a LOA ΔV cost of 978 m/s. Note that both a single and 3-impulse case were considered for this mission with the 3-impulse LOA producing a lower ΔV cost at the expense of an extra day of flight time. After up to a 7-day lunar loiter, the Lander performs an 1881 m/s coplanar powered descent to the lunar surface.

For missions 5-7, a less expensive 3-impulse LOA sequence (compared to 1-impulse) produced a smaller LOA ΔV with a 1-day transfer time between the 1st and 3rd impulses. These missions require the post-LOA phasing orbit to have a specific inclination and longitude of the ascending node to accommodate a 1881 m/s coplanar powered descent to the surface. This phasing orbit inclination and node constraint requirement produce a (worst case) 90° relative declination of the arrival V_∞ for a 1416 m/s LOA ΔV cost.

Following a surface stay of 3, 7, or 11 days for missions 2, 3, and 4 and for missions 5, 6, and 7, respectively, a powered ascent takes the Lander ascent stage back to a 100 km LPO. Note that considerations for anytime abort from the lunar surface dictate maximum ascent plane changes (and associated plane change ΔVs) of 1.2° (34 m/s), 6.7° (191 m/s), and 17.9° (510 m/s) for surface stay times of 3, 7, and 11 days, respectively. Once back in LPO, the CEV and ascent stage perform rendezvous and crew transfers to the CEV. For missions 2-4, the CEV loiters in lunar orbit until its parking orbit most closely aligns with the departure V_∞ vector. In a best case, the relative declination of the departure V_∞ is 0°. However, in a worst case, the combination of geocentric coazimuth and lunar libration effects can result in a relative declination of about 19°. Missions 5-8 are designed with enough performance (ΔV) to intercept a departure V_∞ vector without any lunar loiter, for a shorter overall mission time.

As with mission 1, the direct Earth entry for missions 2-7 targets a 38 km EVP altitude, a geocentric ME transfer orbit inclination of 40° (with respect to the Earth equator) and an associated 36.2° ME transfer orbit inclination with respect to the EM plane. Combining these constraints with a 3.5-day flight time results in a lunar departure V_∞ vector magnitude of 952 m/s.

For missions 2-4, the availability of a lunar loiter allows the CEV time to minimize the relative declination of the departure V_∞ (i.e., 19° in the worst case). A 3-impulse departure possesses a lower LOD ΔV cost of 966 m/s (compared to the cost of a 1-impulse departure) at the expense of 1 day of additional flight time in the intermediate transfer orbit. For missions 5-7, the requirement to perform LOD immediately after ascent and rendezvous could, in a worst case, result in a 90° relative declination of the V_∞ vector. For this case the LOD ΔV cost is increased to 1410 m/s.

Mission 8. This LOR mission, shown in Figure 8.2.11-3 follows the same fundamental architecture as missions 2-7 except that the mission 8 design accommodates a long lunar surface stay (>28 days). Note that a surface stay longer than 28 days will not result in an increase in the overall ΔV cost for this mission as 28 days represents a full lunar rotational cycle containing all transfer orbit geometries. For surface stays greater than 28 days, these geometries merely repeat.

After rendezvous in a 28.7°, 407 km circular ERO, the CEV/Lander configuration performs a 3074 m/s EOD placing the tandem spacecraft on a worst case 57.3° geocentric EM transfer orbit inclination destined to arrive when the Moon is coincidentally at its perigee and orbit node. A minimum ΔV 4-day transfer produces a lunar arrival V_∞ vector magnitude of 986 m/s. A zero degree LPO inclination, designed to minimize LOA and LOD ΔV costs, produces a worst case 19° relative declination of the arrival V_∞ vector and a 879 m/s LOA ΔV. In order to reduce the impact of possible worst case 90° powered descent and ascent plane changes, the circular LPO altitude was raised to 3000 km. Following LOI, the spacecraft performs powered descent to the surface via a 100 km circular lunar phasing orbit. Descent departure from the 3000 km altitude circular LPO includes a ΔV of 1629 m/s for transfer to a 100 km circular phasing orbit altitude including a 90° plane change. An 1881 m/s powered descent to the surface brings the total ΔV cost of the descent and plane change to 3510 m/s.

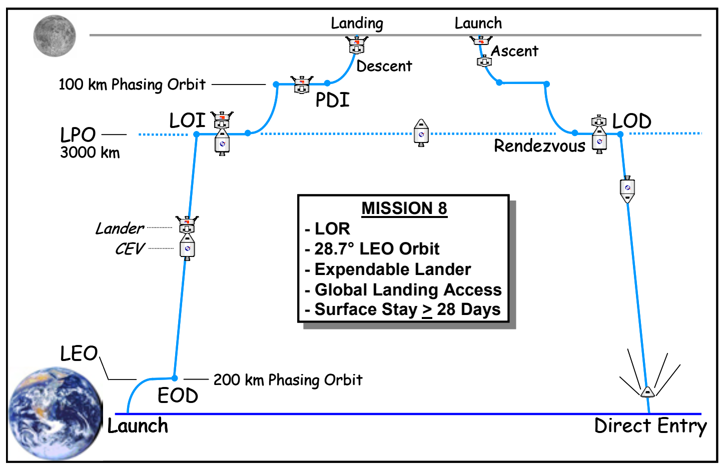


Figure 8.2.11-3 – LOR Mission 8

After a surface stay that is not limited by trajectory considerations, the ascent stage lifts off from the lunar surface and returns, via a 100 km phasing orbit, to the 3000 km circular LPO altitude. During circularization at 3000 km, the ascent stage performs a worst case 90° plane change. The total ascent consists of a coplanar ascent from the lunar surface to the 100 km circular phasing orbit at a ΔV cost of 1834 m/s. This is followed by an apoapse raise maneuver and circularization at 3000 km altitude accompanied by a 90° plane change at a ΔV cost of 1629 m/s for a total ascent ΔV cost of 3463 m/s.

The return phase targets a direct Earth entry to an EVP of 38 km. The geocentric ME transfer plane of 40° and 36.2° ME transfer orbit inclination with respect to the EM plane match that of missions 1-7. The 3.5-day Earth return flight time results in a LOD V_∞ of 952 m/s. For departure from a 0° lunar parking orbit inclination, this V_∞, when combined with a worst-case relative departure declination of 19° produces a 864 m/s LOD ΔV cost.

Mission 9. This architecture represents the TRM which employs a LPR. The CEV performs all maneuvers from LEO through LPA and rendezvous with a previously placed Lander at L1. This is followed by subsequent LPD back to a direct entry to the Earth's surface. The pre-emplaced Lander (at L1) performs all maneuvers from LPD to a powered descent to the lunar surface and subsequent powered ascent back to LPA at L1.

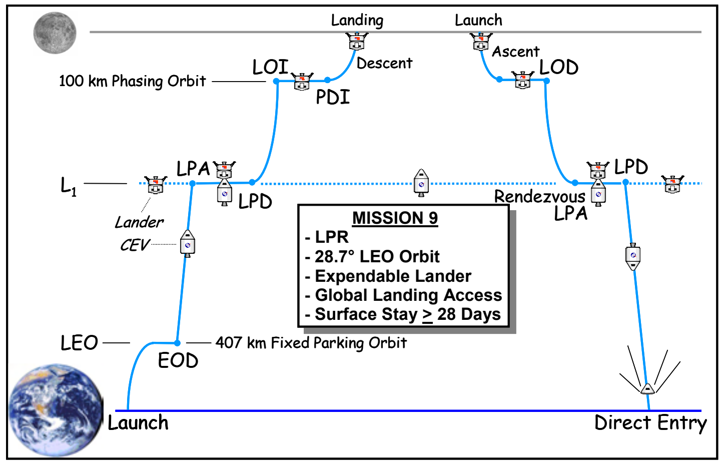


Figure 8.2.11-4 Mission 9

In this scenario, the crewed CEV departs a 407 km circular altitude, 28.7° ERO and takes a 3.5-day flight to the Earth-Moon libration point, L1, where it performs a rendezvous with a pre-emplaced Lander. A 3057 m/s EOD maneuver places the CEV on an Earth-L1 (EL) transfer with a 57.3° inclination to the EM plane. The CEV arrives at the Moon with the Moon coincidentally at nodal crossing and perigee. At L1 arrival, the CEV performs a 57.3° plane change for a total libration point arrival (LPA) ΔV cost of 889 m/s.

After transfer to the Lander from the CEV and subsequent vehicle checkout, the Lander performs a 244 m/s libration point departure (LPD) maneuver placing it on a 2.3-day journey to the Moon. They depart L1, in the Lander, and take a 2.3-day flight to the lunar surface via a 100 km circular altitude phasing orbit. This scenario employs a 631 m/s LOA maneuver to a selectable LPO inclination. In this case the LPD and LOA ΔV cost accommodate landing at a worst case polar surface site. The selectable inclination allows for a coplanar powered descent with a ΔV of 1881 m/s and, following a surface stay, a coplanar 1834 m/s powered ascent back to a 100 km circular altitude phasing orbit. From this phasing orbit, the ascent stage of the Lander performs a 631 m/s ΔV LOD to a 2.3-day transfer back to L1 where it then performs a 241 m/s LOA. After rendezvous with the CEV and crew transfer, the CEV performs a 3.5-day, 800 m/s minimum ΔV transfer (in the EM plane) to an EVP target of 38 km. As with the other missions, a 40° L1-Earth (LE) transfer orbit inclination (w.r.t. the Earth equator) and associated 36.2° LE transfer orbit (w.r.t. the EM plane) ensure that the returning CEV lands in the +40° latitude band back at Earth.

Mission 10. This LSR architecture uses a CEV/Lander combination to take the crew from LEO all the way to the lunar surface, landing near a previously emplaced surface habitat. The CEV performs all maneuvers from EOD through powered lunar descent to the lunar surface and the ascent stage portion of the CEV performs lunar ascent through LOD to a direct entry to the Earth's surface. All post-EOD rendezvous sequences are removed from this approach.

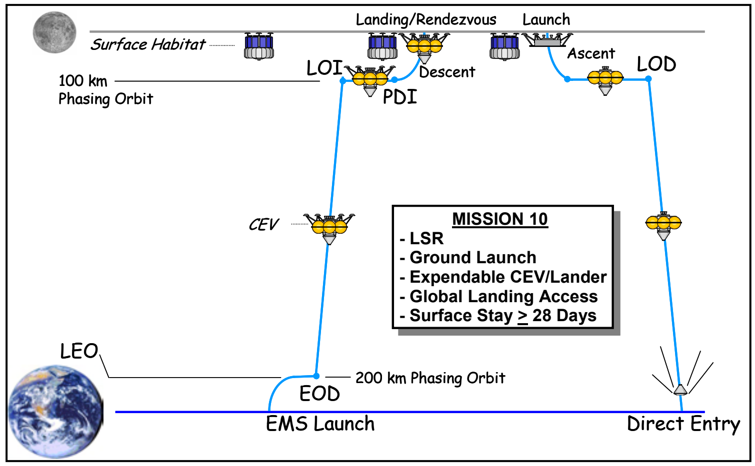


Figure 8.2.11-5 Mission 10

The mission begins after a ground launch to a 90° azimuth that delivers the CEV to a 28.5°, 200 km Earth phasing orbit. A 3125 m/s EOD places the CEV on a 4-day flight to the Moon with a 22° EM transfer orbit inclination (w.r.t. the Earth equator). This ground launch approach is based on the selection of the better of 2 launch opportunities per day. These opportunities are assessed for the worst case condition of arrival when the Moon is at its minimum inclination (w.r.t. the Earth equator) over its 18.6 year cycle (i.e., 18.3°) and when the Moon is at its apex (maximum latitude). For this case, a selectable phasing orbit inclination, tailored to the landing site latitude, allow for a minimum lunar arrival V_∞ vector magnitude of 893 m/s with a 0° arrival relative declination. With an LOA arrival ΔV of 843 m/s, the CEV temporarily inserts into a 100 km altitude circular arrival phasing orbit followed by an 1881 m/s coplanar powered descent to the lunar surface. Following a selectable long duration surface stay time, unlimited by trajectory constraints, the CEV performs a 1834 m/s coplanar ascent back to a temporary 100 km circular phasing orbit with a selectable inclination and longitude of the ascending node designed to provide an EOD with a minimum ΔV of 865 m/s. The EOD maneuver places the CEV on a 3.5-day Earth return with a direct entry to the lunar surface.

Mission	1	2	3	4	5	6	7	8	9	10
EPO Altitude	200 km	407 km			407 km			407 km	407 km	200 km
EM (EL) Transfer Orbit Inclination w.r.t. Earth Equator	32.54°	28.7°	28.7°	28.7°	28.7°	28.7°	28.7°	28.7°	28.7°	28.5°
	72° Launch Azimuth	Optimum ERO Rendezvous			Optimum ERO Rendezvous			Optimum ERO Rendezvous	***	72° Launch Azimuth
EM (EL) Transfer Orbit Inclination w.r.t. EM Plane	27.40°	57.28°			57.28°			57.28°	57.3°	22°
	Moon @ Orbit Apex With Minimum Inclination	Moon @ Orbit Node with Maximum Inclination								Moon @ Orbit Apex With Minimum Inclination
EOD ΔV	3125 m/s	3074 m/s			3074 m/s			3074 m/s	3057 m/s	3125 m/s
OLPA ΔV	***	***			***			***	889 m/s	***
OLPD ΔV	***	***			***			***	244 m/s	***
Lunar Orbit Arrival (LOA) V_{inf}	903 m/s	986 m/s			986 m/s			986 m/s	***	893 m/s
	4.0 Day Xfer Time, Moon @ Perigee	4.0 Day Xfer Time, Moon @ Perigee			4.0 Day Xfer Time, Moon @ Perigee			4.0 Day Xfer Time, Moon @ Perigee	3.5 Day Xfer - EPO to L1, 2.3 Day Xfer - L1 to LPO	4.0 Day Xfer Time, Moon @ Perigee
LPO Altitude	100 km	100 km			100 km			3000 km	100 km	100 km
LPO Inclination	149.0°	90°-180°			90°-180°			0°	Select	Select
	Optimium For 3-Day Surface Stay @ 30° Lat.	Depending On Lunar Landing Site Latitude			Depending On Lunar Landing Site Latitude			Designed to Minimize LOI and LOD ΔV Costs	Selectable LPO Incl. for Min. LOI and LOD ΔV	Tailored to Latitude of Lunar Landing Site
Relative Delcination of Arrival V_{inf}	50°	19°			90°			19°	'***	0°
	50° = 19° + 31° 31° = 180° - 149°	Anytime Arrival to Selected Selenographic Inclination With Loiter For Desired Longitude of the Ascending Node			Worst Case Arrival Plane Change			Worst Case	***	***
LOA ΔV (# Impulses)	1143 m/s (3-impulse)	978 m/s (3-impulse)			1416 m/s (3-Impulse)			879 m/s (1-impulse)	631 m/s	843 m/s (1-Impulse)
Descent Plane Change 1881 m/s (Coplanar)	0°	0°			0°			90°	0°	0°
	***	***			***			Descent to 100 km Phasing Orbit w/ 90° Plane Change	Descent to 100 km Phasing Orbit w/ 0° Plane Change	Xfer Direct to 100 km Phasing Orbit w/ 0° Plane Change
Ascent Plane Change 1834 m/s (Coplanar)	1°	1.2°	6.7°	17.9°	1.2°	6.7°	17.9°	90°	0°	0°
	Plane Change ΔV = 29 m/s	Plane Change ΔV = 34 m/s (Worst Case)	Plane Change ΔV = 191 m/s (Worst Case)	Plane Change ΔV = 510 m/s (Worst Case)	Plane Change ΔV = 34 m/s (Worst Case)	Plane Change ΔV = 191 m/s (Worst Case)	Plane Change ΔV = 510 m/s (Worst Case)	Plane Change ΔV = 1629 m/s (Worst Case)	Plane Change ΔV = 0 m/s	Plane Change ΔV = 0 m/s
EVP Altitude	38 km (Apollo 17)									
ME (LE) Transfer Orbit Inclination w.r.t. Earth Equator	40° - For Favorable Landing Latitude									
ME Transfer Orbit Inclination w.r.t. EM Plane	36.22° - Moon @ Orbit Apex w/ Minimum Inclination In Lunar Cycle (18.6 years)									
Lunar Orbit Departure (LOD) V_{inf}	952 m/s	952 m/s			952 m/s			952 m/s	***	952 m/s
	3.5 Day Xfer Time for Earth Landing Lon. Ctrl.	3.5 Day Xfer Time for Earth Landing Longitude Control			3.5 Day Xfer Time for Earth Landing Longitude Control			3.5 Day Xfer Time for Earth Landing Lon. Ctrl.	2.3 Day Xfer - LOD to L1, 3.5 Day Xfer - L1 to Earth Entry	3.5 Day Xfer Time for Earth Landing Lon. Ctrl.
LOD ΔV	***	***			***			***	631 m/s	***
ILPA ΔV	***	***			***			***	241 m/s	***
ILPD ΔV	***	***			***			***	800 m/s	***
Relative Declination of Departure V_{inf}	50°	19°			90°			19°	***	0°
	***	Worst Case Departure Plane Change; 2 Opportunities Per Month for LPO Inclination > 19°			Worst Case Departure Plane Change			Worst Case	***	***
LOD ΔV (# Impulses)	1152 m/s (3-impulse)	966 m/s (3-impulse)			1410 m/s (3-impulse)			864 m/s (1-impulse)	***	865 m/s (1-impulse)
Comment	Apollo Type	Minimum ΔV with Lunar Loiter			Non-Minimum ΔV - No Lunar Loiter				LDRM-2 BRM	Min. ΔV - No Lunar Loiter

Table 8.2.11-2: Mission Profile Characteristics for 10 Selected Missions

Mission Performance – Original Study

The missions in this report are designed to provide the best possible performance approach (i.e., minimum ΔV) that accommodates the worst possible orbital mechanics-related performance impacts (e.g., worst case plane changes for anytime surface abort) for a given mission. As in the Apollo program of the 1960s and 1970s, planning for aborts and off-nominal situations played a significant role in these mission designs. Any mission design process involving a flight crew must provide as many options for safe return as are reasonably possible. The anytime-abort requirement, adopted for all mission profiles in this report, serves to enhance crew safety and survivability in the event of system failure or medical emergency. The ΔV cost of an "abort-preferred" mission design can be considerably greater than that of a nominal mission design. One approach to reducing this cost is to trade time for ΔV as an alternate method of addressing off-nominal or emergency situations.

By loitering in orbit or on the lunar surface long enough for the (rotational) motion of the Moon to produce favorable alignment of the selenocentric orbit plane (with very small rotation), the ground-launched LOR ΔV performance cost can be reduced to something comparable with that of the LSR mission. However, the required delay in getting back from an arbitrary site to the CEV (and its backup habitat) in the rendezvous orbit plus possible additional time for alignment of the rendezvous orbit with the LOD V_∞ vector could take from 3-4 weeks, depending on the length of the surface stay. While an emphasis on reliability and a reserve supply of consumables might make delays due to some system failures possible, these safeguards would be of little help to a crew experiencing a severe medical emergency.

The ΔV performance costs for all primary phases of the ten missions in this (Phase 1) report are shown in Figure 8.2.11-5a along with the constraints for each mission and the availability of lunar departure. The corresponding numerical data can be seen in the following Figure 8.2.11-5b.

The Apollo-style mission (Mission 1) carries a total mission ΔV of 9264 m/s. This 3-day surface stay mission differs from all other missions in that it achieves only limited lunar landing site access (+30°) vs. global access for the other missions. The LOA and LOD for this mission (and all others in this set) employ a 3-impulse lunar arrival and departure maneuver sequence, respectively. In all cases, the additional two days of flight time (one day for arrival and one day for departure) are offset by desirable performance gains. For missions 5 through 7, a single impulse lunar arrival or departure would reduce the flight time by one day for each maneuver, but at the expense of an additional ΔV cost of about 1650 m/s for each maneuver. Like all missions, Mission 1 possesses the plane change capability to launch the crew from the surface at any time during their 3-day surface stay, in the event of emergency. Once back to the LPO, this mission also affords the crew the ability to immediately depart the LPO back to Earth. Thus the performance cost of this mission provides anytime surface abort and immediate Earth return. The total time spent in the lunar vicinity is about 5 days, due to the additional 2 days of lunar inbound and outbound flight time associated with the 3-impulse arrival and departure maneuvers, respectively.

Expanding to global access for 3, 7, and 11-day surface stays, combined with anytime surface abort and immediate post-LOA descent and post-ascent Earth return gives missions 5, 6, and 7 a total ΔV cost of 9749 m/s, 9906 m/s, and 10225 m/s, respectively. For the 3-day surface stay missions, the cost of increasing lunar landing site availability from +30° latitude (Mission 1) to global access (Mission 5) is about 500 m/s. For the global access case with anytime landing and ascent to Earth return (i.e., Missions 5-7), the ΔV cost to increase the surface stay time from 3 to 7 and from 7 to 11 days is 157 m/s and 319 m/s, respectively. Missions 5-7 provide for immediate descent to the surface after LOA and for immediate return to Earth after ascent to the LPO. Every day of the continuous lunar surface launch capability also provides Earth return capability. The number of days spent in the lunar vicinity will be 2 days greater than the total days spent on the lunar surface due to the additional day of flight time for the lunar inbound and outbound 3-impulse maneuver sequences.

Allowing more loiter time in lunar orbit (waiting for coplanar or near coplanar maneuver geometry), reduces the ΔV cost for 3, 7, and 11 day surface stay missions. The lunar loiter missions (2, 3, and 4) have total ΔV costs (8867 m/s, 9024 m/s, and 9343 m/s, respectively) about 900 m/s lower than that of missions 5, 6, and 7 for corresponding lunar surface stay times. This ΔV savings comes at a cost of time spent in lunar orbit. For the 3-day surface stay, global lunar landing site access of mission 2, all three days on the surface are available for ascent to the LPO. However, there is only one opportunity in a maximum of about 21 days for the crew to return to Earth from the LPO. The two days of added flight time due to 3-impulse maneuver sequences brings the total time spent in lunar orbit to a worst case 23 days. This maximum time in lunar orbit also applies to a 7-day surface stay mission. For an 11-day surface stay mission, a single lunar departure opportunity occurs (in a worst-case situation) about every 28 days. With the two days of time in lunar vicinity due to 3-impulse orbit transfer flight time, the total time spent in lunar orbit is about 30 days (worst case).

For long duration surface stays of 28 days or more (mission 8), the total ΔV cost of 11890 m/s is about 1670 m/s larger than that of a similar mission with a shorter 11-day surface stay (mission 7). Note that missions with surface stays greater than 28 days do not reflect any increase in total mission ΔV due to the Moon's approximately 28 day orbit cycle (27.3 days). The worst-case mission geometry of the 28-day surface stay mission does not increase for a longer surface stay. For this mission, the anytime surface abort capability of the 28-day stay is accompanied by an anytime Earth return capability. The 1-impulse lunar arrival and departure maneuvers limit the time in the lunar vicinity to be close to that of the surface time. This mission employs a zero degree inclination parking orbit providing a minimum LOA and LOD plane change requirement at the expense of a worst case (90°) LOA and LOD plane change.

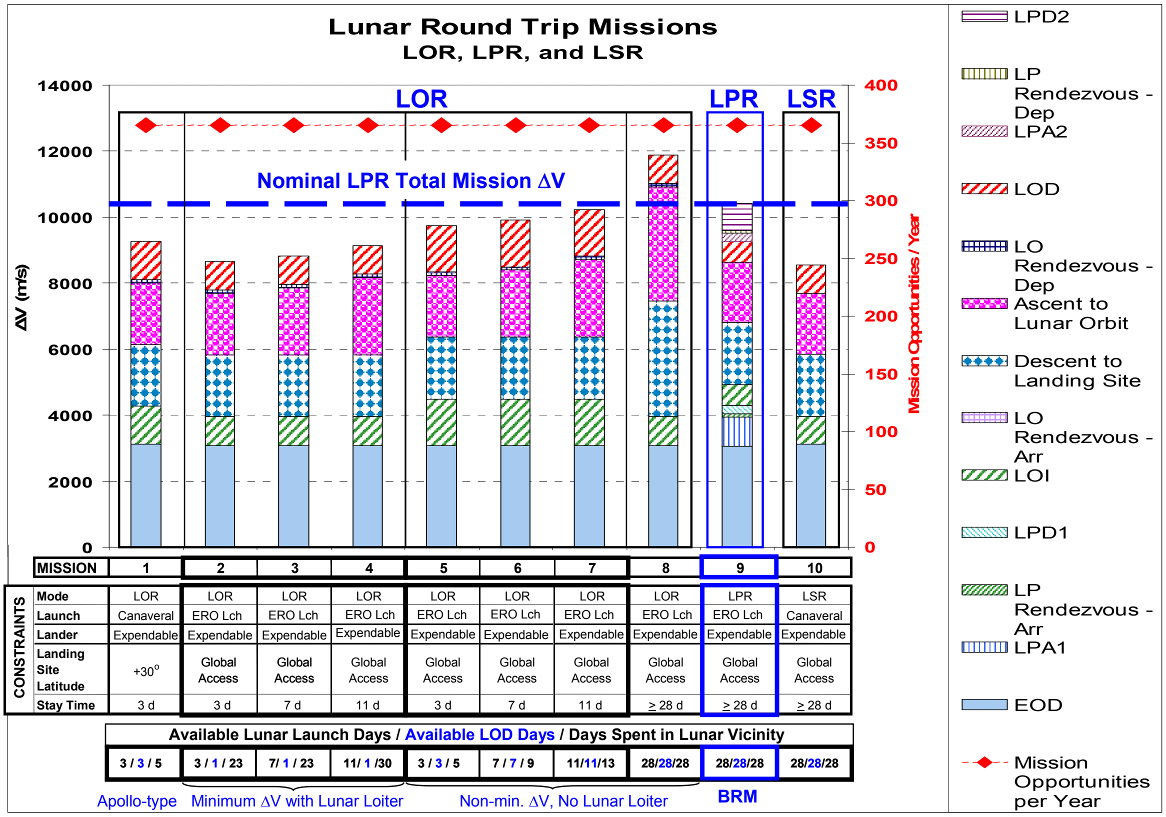


Figure 8.2.11-6a: ΔV cost, constraints, and availability for 10 selected round-trip lunar mission scenarios.

ΔV Requirement for Selected Lunar Missions (m/s)										
	Lunar Orbit Rendezvous								Libration Point Rendezvous	Lunar Surface Rendezvous
Mission #	**1**	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**10**
Mission Features / Flight Phase	Canaveral Launch Expendable Lander Landing Lat. = 30° Surface Stay = 3 d	**28.7 Deg ERO Lch** Expendable Lander **Global Access** Surface Stay = 3 d	28.7 Deg ERO Lch Expendable Lander Global Access **Surface Stay = 7 d**	28.7 Deg ERO Lch Expendable Lander Global Access **Surface Stay = 11 d**	**28.7 Deg ERO Lch** Expendable Lander Global Access **Surface Stay = 3 d**	28.7 Deg ERO Lch Expendable Lander Global Access **Surface Stay = 7 d**	28.7 Deg ERO Lch Expendable Lander Global Access **Surface Stay = 11 d**	28.7 Deg ERO Lch Expendable Lander Global Access **Surface Stay ≥ 28 d**	28.7 Deg ERO Lch Expendable Lndr Global Access Surface Stay ≥ 28 days	Canaveral Lch Expendable Lndr Global Access Surface Stay ≥ 28 days
LPD2	0	0	0	0	0	0	0	0	800	0
LP Rendezvous - Dep	0	0	0	0	0	0	0	0	100	0
LPA2	0	0	0	0	0	0	0	0	241	0
LOD	1152	865	865	865	1410	1410	1410	864	631	865
LO Rendezvous - Dep	100	100	100	100	100	100	100	100	0	0
Ascent to Lunar Orbit	1863	1868	2025	2344	1868	2025	2344	3463	1834	1834
Descent to Landing Site	1881	1881	1881	1881	1881	1881	1881	3510	1881	1881
LO Rendezvous - Arr	0	0	0	0	0	0	0	0	0	0
LOI	1143	978	978	978	1416	1416	1416	879	631	843
LPD1	0	0	0	0	0	0	0	0	244	0
LP Rendezvous - Arr	0	0	0	0	0	0	0	0	100	0
LPA1	0	0	0	0	0	0	0	0	889	0
EOD	3125	3074	3074	3074	3074	3074	3074	3074	3057	3125
TOTAL	**9264**	**8766**	**8923**	**9242**	**9749**	**9906**	**10225**	**11890**	**10408**	**8548**
	EARTH DEPARTURE WINDOWS / YEAR									
	365	365	365	365	365	365	365	365	365	365
	Available Lunar Launch Days / Available LOD Days / Days Spent in Lunar Vicinity									
	3 / 3 / 5	3 / 1 / 23	7 / 1 / 23	11 / 1 / 30	3 / 3 / 5	7 / 7 / 9	11 / 11 / 13	28 / 28 / 28	28 / 28 / 28	28 / 28 / 28

	MISSION	**1**	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**10**
CONSTRAINTS	**Type**	LOR	LOR	LOR	LOR	LOR	LOR	LOR	LOR	**LPR**	**LSR**
	Launch	Canaveral	Canaveral	Canaveral	Canaveral	**28.7° Orbit**	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	**Canaveral**
	Lander	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable
	Landing Site Latitude	30°	**Global Access**	Global Access	Global Access	Global Access	Global Access	Global Access	Global Access	Global Access	Global Access
	Surface Stay Time	3d	3d	**7d**	**11d**	**3d**	**7d**	**11d**	**≥ 28 d**	≥ 28 d	≥ 28 d

Figure 8.2.11-6b: ΔV cost, constraints, and availability for 10 selected round-trip lunar mission scenarios.

The LOR mission 8, with global lunar landing site access, has a ΔV cost about 1500 m/s larger than the LPR TRM for LDRM-2 (mission 9) with total ΔV cost of 10408 m/s). However, preliminary vehicle sizing shows a total 230 t IMLEO for the TRM in mission 9 compared to about 199 t for the LOR mission 8. When additional lunar orbit loiter is added to remove or minimize LOA and LOD plane changes, the total IMLEO for mission 8 is reduced to about 169 t (see Section 19). The LSR approach (i.e., mission 10) provides the minimum total ΔV cost along with global access, anytime surface abort and anytime Earth return. However, the staging characteristics (i.e., the CEV required to perform descent, ascent, and LOD) can result in a large IMLEO for this configuration (on the order of the TRM and possibly larger). The low ΔV cost and simplicity of the mission 10 profile, as well as the beneficial elimination of critical lunar orbit rendezvous maneuvers, are countered by the large IMLEO.

8.3 Final Thoughts on LOR and LSR vs. LPR

Given the objectives and operational constraints it was designed to meet, the Apollo mission profile would be hard to improve. However, major modifications are required before it can satisfy the objectives defined for this study. The Apollo landing sites were all situated within 30° of the lunar equator, and the lunar surface stay times were all shorter than a week. In the case at hand, the mission profile must provide the ability to land at any arbitrary site on the lunar surface, and to stay there a week or perhaps much longer while preserving the Apollo capability for anytime abort from the lunar surface (i.e., to a nearby backup habitat and thence to Earth).

Initial mass in low Earth orbit (IMLEO) is often used as a proxy for the monetary expense of a space mission in preliminary studies such as this one. IMLEO minimization usually is achieved by separating, sometime before final descent to the lunar surface, the assets needed by the flight crew while on the surface from those needed to transport them between Earth and the lunar vicinity. Before the mission can be accomplished and the flight crew returned to Earth, there must be a rendezvous – in a chosen locale near the Moon – between the separated assets. The rendezvous locales considered in this study are the cislunar libration point (LPR), selenocentric orbit (LOR), and the lunar surface itself (LSR).

No matter what locale is chosen for rendezvous, a selenocentric phasing orbit is required for economical access to an arbitrary landing site. The reason is essentially the same as that which applies to the launch of a lunar or interplanetary spacecraft from a site on the Earth surface: Although on-orbit plane-change penalties can be eliminated easily for such a launch (by choosing a launch azimuth and time of day such that the plane of the predeparture orbit will contain the required departure velocity vector), in the general case a coasting arc in a phasing orbit is required to avoid the penalty associated with non-optimal flight path angles during injection into the departure trajectory. Absent ΔV penalties associated with orbit plane or flight path angle, the total propulsive velocity increment is minimized by setting the altitude of the phasing orbit as low as possible – consistent with atmospheric drag effects in the case of the Earth, and terrain clearance in the case of the Moon. Because it is sometimes advantageous to let the phasing orbit serve also as the rendezvous orbit, estimation of terrain clearance must account for the long-term effect of large perturbations arising at low altitude from scattered concentrations of lunar mass.

For this study, the altitude of the selenocentric phasing orbit was chosen to be 100 km. The propulsive ΔV required for an in-plane round trip from that altitude to the lunar surface is a little more than 3700 m/s, which is greater than the EOD velocity increment by about 20%. Consequently, the IMLEO required for a lunar round trip can sometimes be reduced by leaving all assets not needed on the lunar surface – before descending to it – at the rendezvous locale, where they can be retrieved/reoccupied by the landing crew after ascending from the surface. The LOR and LPR trajectory profiles are designed to utilize such a scheme.

8.3.1 Lunar Orbit Rendezvous

8.3.1.1 Descent/Ascent Plane Changes Associated with LOR

For stays shorter than about 11 days, the sum of descent/ascent plane-change angles – required for descent to the chosen landing time and for ascent at the most inopportune time(s) during the nominal surface stay period – can be minimized by orienting the rendezvous orbit so that its plane contains the landing site at landing time, and its apex (maximum latitude) in the landing-site hemisphere is a few degrees closer to the nearest pole than the landing site itself.

For longer stays, the minimum sum of plane-change angles is realized by choosing either an equatorial or a polar plane for the rendezvous orbit, depending on whether the landing site lies closer to the equator or to one of the poles. For either choice, landing exactly at latitude 45°(north or south) requires a plane change of 45° during transfers in both directions between rendezvous orbit and landing site.

Descent/ascent plane changes of this magnitude benefit from establishment of two separate phasing orbits (one for descent and the other for ascent), each having an altitude of 100 km and oriented independently so as to contain the lunar site at landing and at liftoff time, respectively. The rendezvous orbit (equatorial or polar) is established at a considerably higher altitude so that the major part of the plane change can be made where the orbital speed is lower. This allows any given plane change angle to be achieved with a smaller velocity increment, and facilitates further economy by allowing a change of orbital energy and of orbital plane to be accomplished simultaneously with a single impulse. The down side of this stratagem is that it adds 4 major maneuvers to the round-trip profile, together with a moderate increase in the time required for the Earth-Moon-Earth round trip.

The altitude chosen for the elevated rendezvous orbit is 3000 km, where the circular orbit period is approximately 8 hours. A higher altitude would provide a further reduction in the required descent/ascent propulsive velocity increments, but would increase the associated flight times, decrease the frequency of opportunities for transfers to and from the surface, and increase the susceptibility of the rendezvous orbit to earth and solar perturbations.

8.3.1.2 LOD Plane Changes Associated with LOR

The plane of a selenocentric orbit is stationary with respect to inertial space if it is polar, and is nearly stationary for any other orientation. Since the inertial rotation rate of the moon itself is about 13° per day, the plane of the LRO regresses at that rate in the selenographic frame, which is fixed with respect to surface features rather than inertial space.

Conservation of geocentric angular momentum in the Moon-Earth transfer orbit dictates that the selenographic declination of the departure V_∞ vector for any return-to-earth trajectory lies within the range of ±19°, after taking lunar libration into account. For reasonable flight times (on the order of 2.5 to 5.0 days) – no matter when lunar departure occurs – the selenographic longitude of such a vector lies within the approximate range of 30°-95°.

Said another way, the gist of the two preceding paragraphs is this: At LOD time the V_∞ vector to be achieved will always be confined within a quasi-rectangle that is bounded by the 19th parallels of north and south latitude and by the 30th and 95th meridians of longitude on the surface of the selenographic reference sphere. The LOD plane-change penalty will be moderate if the track of the LRO at that time passes through aforesaid rectangle. Otherwise it will be more severe, depending on the minimum angular distance between the orbit track and the perimeter of the rectangle.

If the rendezvous orbit plane coincides with the equator, its track will always pass through the center of the rectangle, the relative declination of the LOD V_∞ vector (i.e., the angle it makes with the orbit plane) will never exceed 19°, and the LOD plane-change penalty will be minimal.

Otherwise, the relative declination of the LOD V_∞ vector at the most inopportune departure time (during a nominal surface stay longer than about 14 days) will be either 90° or the sum of 19° and the selenographic inclination of the rendezvous orbit (or the supplement of the inclination if the orbit is retrograde), whichever is smaller. (A retrograde rendezvous orbit is usually preferred when a choice is available, because it yields a smaller abort velocity increment for a nonstop return to Earth if, for some reason, the LOA maneuver cannot be executed at the planned time).

If the landing site lies more than 71° from the lunar equator, the relative declination of the LOD $V\infty$ vector that must be achieved for immediate departure will be equal to or very near 90° at one or more times during the longer stays required in this study. The attendant ΔV penalty is severe (with a 3-impulse maneuver sequence, on the order of 350 m/s for departure from a 3000 km rendezvous orbit, or 450 m/s for departure from a 100 km orbit), and appears to be unavoidable if LOR is used to satisfy the operational requirements previously described.

8.3.1.3 LOA Plane Changes Associated with LOR

In contrast to lunar orbit departure, the lunar orbit insertion plane-change penalty associated with the lunar orbit rendezvous trajectory profile is minimal if the rendezvous orbit plane is equatorial, or nil if it is polar.

The ΔV requirements determined in this study for transferring between landing sites and a polar rendezvous orbit are based on the assumption, in each case, that the ascending node of the rendezvous orbit on the lunar equator lies at the worst possible location it could have for landing at or launching from the site under consideration. This means the node location can always be chosen so that the V_∞ vector, at LOA time, will lie in the rendezvous orbit plane and therefore there will be no plane-change penalty at all associated with LOI.

No comparable node can be defined for an equatorial orbit, but (as pointed out earlier in the discussion of the LOD maneuver) the declination of the LOA V_∞ vector relative to an equatorial orbit can never exceed 19°. The associated ΔV penalty is typically on the order of 350 m/s for a 1-impulse LOA maneuver if the rendezvous orbit altitude is 100 km, but only about 150 m/s if the orbit altitude is 3000 km, or 100 m/s if a 3-impulse maneuver sequence with an intermediate 24-hour ellipse is used for insertion into the 100 km orbit.

8.3.2 Libration Point Rendezvous

As a rendezvous locale for round trips to the surface of the Moon, the primary advantages of the cislunar libration point are (1) it has a continuously unobstructed line of sight to Earth, (2) it is a point on the line connecting Earth and Moon, rather than an orbit about the Moon thus eliminating requirement for large descent or ascent plane change, and (3) its position is nearly fixed in the selenographic frame of reference thus providing a consistent mission profile to any lunar landing site. With regard to the latter, selenocentric distance variations amount to about ±6% of the mean value due to eccentricity of the lunar orbit, and selenographic latitude and longitude variations (which result from lunar librations) amount to approximately ±7° apiece.

Items (2) and (3) in the preceding paragraph make LPR spacecraft performance requirements less dependent on the location of the lunar landing site – and the time spent there – than is the case for their LOR counterparts. They also make it possible to eliminate plane-change penalties for all maneuvers except those executed during libration-point arrivals and departures. The velocity magnitudes and the plane changes associated with those events are such as to produce only moderate ΔV penalties, as compared to those which characterize the LOR profile when it is required to stay a long time at an inconvenient site. Nevertheless, the sum of LOR propulsive velocity increments required for a short stay at an easily-accessible site is significantly smaller than the LPR requirement, and the two sums are comparable when global access and long stays are required.

The main disadvantages of LPR are (1) the longer time required – 2-4 days as opposed to a few hours with LOR – to reach the Earth return vehicle from the lunar surface, (2) a moderate increase in the total transfer time between Earth and Moon, and (3) the addition of 4 major maneuvers to the number required for a simple LOR profile such as that used in the Apollo missions. With regard to the maneuver count, however, it is worthy of note that the LOR profiles discussed in the previous subsection of this report require, for the more difficult sites, as many or even more major maneuvers than the LPR profile.

8.3.3 Lunar Surface Rendezvous

The distinguishing features of the LSR profile, as defined for this study, are (1) predeployment and remote checkout of the surface habitat at the lunar landing site, along with all necessary equipment and expendables, before the flight crew departs Earth, and (2) subsequent landing of the flight crew, along with all assets needed for their return to Earth, within easy walking distance of the predeployed habitat. Carrying all of the Earth-return assets to the lunar surface eliminates the need for lunar orbit rendezvous, and allows the establishment of separate phasing orbits for arrival and departure. Each of these orbits can have an orbit plane that contains the lunar site at landing/liftoff time and the appropriate V_∞ vector at LOI/LOD time, thus eliminating all selenocentric plane-change penalties everywhere in the whole trajectory profile.

As discussed previously, transporting all return-to-Earth assets through a round trip between lunar orbit and lunar surface imposes a significant IMLEO penalty in itself. However, this is counterbalanced by the complete elimination of selenocentric plane-change penalties, which yields an equally significant reduction of the required propulsive velocity increments (and therefore IMLEO). Preliminary mass buildup calculations, which have yet to be verified, indicate that the IMLEO required for LSR is comparable to that required for LPR.

In terms of propulsive velocity requirements, LSR is completely insensitive to landing site location and the time spent there. LPR is also completely insensitive to duration of the surface stay, but its sensitivity to site location (although less severe than that of LOR) is significant.

LSR provides the following major advantages of over LOR and LPR:

(1) The time required to reach the Earth return vehicle from the lunar surface is measured in minutes, as compared to hours in the case of LOR and days in the case of LPR.

(2) Operationally, the mission profile is simplified considerably by reducing the number of propulsive maneuvers to the absolute minimum, and by eliminating the need for any docking operation after Earth orbit departure (EOD).

(3) Separate deployment of the lunar habitat, along with equipment and expendables needed on the lunar surface, decouples the CEV design from variations in surface stay time and specific mission objectives.

More study is needed to validate the preliminary IMLEO calculations for LSR, to fully assess the operational and spacecraft design implications of CEV egress and ingress on the lunar surface, and to establish the feasibility of automated landing with video monitoring by the flight crew – which probably would be necessary because of restricted direct visibility from within the CM.

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9.0 Element Overview

Four major elements have been identified for the proposed LDRM-2 lunar missions. They were selected to provide an initial point of departure for the trade reference mission, and consist of a Crew Exploration Vehicle (CEV), a Lunar Lander, an Earth Departure Stage (EDS), and a propulsive Kick Stage.

9.1 Crew Exploration Vehicle

As dictated in the LDRM-2 task request statement, the mission of the CEV is to transport the crew from the Earth's surface to lunar vicinity and back and to loiter unoccupied during the lunar landing portion of the mission. The CEV is injected towards the moon using a separate propulsive stage, an EDS. A baseline CEV was sized for the LDRM-2 Trade Reference Mission A2 (see Section 10.0) and was then modified for several variant mission architectures (Sections 11.0 through 17.0).

Analogous to the Apollo Command and Service Module, the CEV consists of a Crew Module (CM) and a Service Module (SM). The CM is envisioned as the crew transportation during launch, transfer to and from the moon's vicinity, and re-entry back at Earth. The SM provides resources to the CM during all of its mission phases except Earth re-entry.

The CEV design includes the following systems: vehicle command and control, main and reaction control propulsion, communications, life support, thermal control, power, crew accommodations, radiation protection, landing and recovery, thermal protection, structures, and mechanisms. These systems and the vehicle size and shape are sized for a specific mission scenario and can be scaled for alternative missions (including delta-V, crew size, duration, power levels, life support consumables, etc.).

9.2 Lunar Lander

The Lunar Lander functions as the crew transport vehicle from various staging points (dependent on mission architecture) to the lunar surface and back. On the lunar surface, the Lander serves as the crew living space and departure point for surface EVAs. An EDS transports the Lander to the staging point. A baseline Lander was sized for the LDRM-2 Trade Reference Mission A2 (see section 10.0) and was then modified for several variant mission architectures (see sections 11.0 through 17.0). The Lander is envisioned as a one-and-a-half-stage vehicle with the structural landing systems left on the surface of the moon when the ascent stage departs, but reusing the same descent engines for lunar ascent.

The Lunar Lander design includes the following systems: airlock, vehicle command and control, communications, life support, thermal control, power, crew accommodations, radiation protection, structures, and mechanisms.

9.3 Earth Departure Stage

The EDS's are propulsive elements used to inject the CEV and Lunar Lander towards the lunar vicinity. Depending on the architecture, there are two stage sizes required. The EDS's were not designed to a detailed level, but were sized to include the following systems: propulsion, power, command and control, limited guidance, structures, and mechanisms.

9.4 Kick Stage

The Kick Stage is a propulsive element designed to perform certain burns in each of the architecture variations. It was conceived to help reduce the differences in size of the two EDS's due to their payload inequalities without having to penalize the Lander design when performing those burns.

10.0 Trade Reference Mission

The LDRM-2 requirements formulation task called for the establishment of a trade reference mission (TRM) against which architecture trades could then be performed and evaluated. These trades were used to determine the relative impact of a particular architecture decision in the areas of safety & reliability, effectiveness & evolvability, schedule, and affordability. This section describes the trade reference mission, outlining the key assumptions made in the formulation of the TRM, and architecture trades performed to select a specific reference architecture and mission timeline including considerations of various allocations of major maneuvers among elements and different propulsion system types. Critical events for the TRM are outlined in this section and ranked by criticality, mission abort options are identified, and vehicle subsystem technology assumptions and mass properties are also described. Following sections of this report will describe the specific trades requested in the LDRM-2 study task using this TRM as a point of departure, such as changing mission rendezvous location(s), crew size, number of launches per mission, and several other key architecture parameters.

10.1 Major Assumptions/Clarifications

This section outlines the major architecture assumptions made in formulation of the LDRM-2 trade reference mission along with their supporting rationale. These assumptions, listed in the Requirements Formulation Task RFT 0001.04, were levied by the study's NASA HQ customer on the LDRM-2 study team to be used as an initial point of reference.

<u>One human lunar mission per year:</u> This is a programmatic assumption dictated in the LDRM-2 task statement. Flight rate has no impact on the analyses performed in this study.

<u>Return mass from the moon is 100 kg:</u> In the present absence of a clearly defined science strategy for the Vision for Space Exploration, the LDRM-2 trade reference mission will assume that a combination of 100 kg of payload and lunar surface material will be returned to Earth with the crew at the end of each mission. This assumed mass is comparable to the Apollo 17 "J-Type" mission, which returned 112 kg of lunar sample. Thirty-five years of advancement in instrumentation and measurement techniques have reduced required sample masses for analysis several orders of magnitude, from the kilogram-scale to gram-scale or less, thus a returned sample mass comparable to the Apollo program is thought to be entirely adequate. Other return mass needs such as returning system hardware from lunar surface assets may eventually drive this requirement to be greater than 100 kg. In addition, return samples may require conditioning during transit to Earth to preserve their scientific content (samples such as biological and planetary materials). A sensitivity study for a range of return payload masses has been performed and is described in Section 19.3.

<u>Payload to lunar surface is 500kg:</u> Lacking a surface exploration plan, the LDRM-2 Lunar Lander payload delivery capabilities to the lunar surface are assumed to be similar in mass, volume, and payload complement to the Apollo Lunar Module descent stage. The Apollo 17 "J-Type" mission delivered 558 kg of payload to the surface, which included an unpressurized lunar roving vehicle, a lunar surface experiment package, and various geology tools and equipment. Surface EVA equipment and sample return containers are not to be counted in this allocation. Lo-

gistics resupply requirements for lunar surface assets may eventually drive this requirement higher than 500 kg, however such considerations were outside the scope of this effort. A sensitivity study for a range of landed payload masses has been performed and is described in Section 19.4.

All mission elements placed in LEO (28.5^o 407 km circular): Due to launch vehicle constraints, lunar missions will require the mating of elements in Earth orbit prior to departure for the Moon. Launches into 28.5^o inclination orbits allow the maximum payload to orbit from the Eastern Test Range. Additionally, this inclination affords large planar launch windows required for rendezvous. The assembly altitude of 407 km is specified to minimize the effects of atmospheric drag on orbital lifetime while minimizing payload deployment altitude required on the launch vehicle upper stage. Future trades between the launch vehicle and orbital elements will be required to determine the optimum staging altitude.

4-launch solution: Without a clear understanding of potential launch vehicle cargo capabilities, an architecture that requires four launches per mission is considered to be a good initial balance between the desire to minimize launch vehicle size while minimizing number of launches and number of unique elements / element interfaces. A sensitivity study has been performed on the TRM examining 2-launch, 3-launch, and 25 t per launch architectures (see Sections 11.0-13.0 for additional information).

Consider the lunar mission elements to be "cargo" in terms of delivery to the LEO parking orbit: The launch vehicle will be responsible for delivering architecture elements to a 28.5^o 407 km circular orbit. This assumption puts the entire burden of cargo delivery on the launch vehicle, which helps to determine maximum launch vehicle capabilities. For this study, the propulsive capabilities of the lunar mission elements will not be employed for orbit insertion, but will likely be required for orbit maintenance. Future trades can be performed to optimize the allocation of the orbit insertion function between the launch vehicle and orbital elements.

Automated rendezvous and docking shall be used to assemble the elements: Several lunar mission elements (*e.g.* Earth Departure Stages, Lander) for the four-launch baseline architecture will be launched without crew onboard, therefore automated rendezvous and docking of these elements can reduce crew time in space by assembling, integrating, and testing the combined crew transportation system before the crew launches from Earth.

Assume 2 weeks between launches: This assumption is a balance between a desire to minimize total mission duration and vehicle lifetime while not severely impacting launch vehicle production, processing, and launch facilities for a four launch per mission baseline. A launch vehicle processing trade study will be required to determine the feasibility of meeting this assumption.

Crew must be launched on a human-rated launch system: This is dictated by the NASA human rating requirements document NPR 8705.2.

A dedicated lunar lander element with a separate crew module will be used to transfer the crew from the lunar vicinity to the lunar surface and back to lunar vicinity: The cost of landing the entire CEV (including parachutes, TPS, outer skin structure, Earth landing systems, etc.) on the lunar surface and operating out of that vehicle for the duration of the surface mission is deemed to be too severe for a L1 rendezvous architecture. As the mass of a lander ascent stage has the biggest leverage on total architecture mass, a separate, highly optimized lunar lander element like the Apollo Lunar Module will be used for the trade reference mission and the CEV will be used to deliver the crew to lunar vicinity.

Lunar surface stay time of 7 days: A surface stay time of seven days should provide crews with enough time to explore selected sites of scientific interest on the Moon while avoiding some of the thermal, lighting, and power generation challenges of the Moon's 27-day rotation period. A sensitivity study for a range of lunar surface stay times has been performed and is described in Sections 19.9 and 19.10.

Four crew with all crew going to the lunar surface: An architecture that allows all crew members to land on the Moon represents the highest overall architecture mass for that particular crew size and puts the highest burden on CEV autonomous/automatic operational requirements. Leaving crewmember(s) in orbit during the surface mission will enable future mass and cost savings if required by budgetary constraints. A parametric variation where three crewmembers land on the Moon and one crewmember remains in orbit is described in Section 19.5.

Daily EVAs will be conducted on the surface of the Moon from the Lunar Lander: Providing for a daily EVA capability while on the lunar surface will enable the greatest science return from a mission.

The CEV and Lunar Lander are not required to be reusable and will not be explicitly designed for reusability: Previous spaceflight experience has taught that reusability should not be dictated *a priori*, rather the decision to reuse vs. build new should be made based on cost and schedule trades for a given flight rate and total program duration. Future design efforts should examine vehicle reusability options.

The Lunar Lander will not be designed to provide functionality beyond that required for the planned lunar surface stay time: As the Lander has the greatest leverage on the total mass of the architecture, it is critical that in order to minimize mass it be designed to provide only the minimum amount of functionality required to safely meet the mission objectives.

The reference lunar surface environment for landing operations and the surface stay is a relatively benign, Apollo-type thermal and lighting condition: The capability to handle the entire span of lunar thermal and lighting extremes as required for anytime, anywhere landing and surface operation may have a severely negative impact on a lander design. Therefore, the study will assume a low sun angle of 7-20 degrees to aid crew visibility during descent and thermal conditions commensurate with lunar "daytime" operation. A detailed lander design study should be conducted to determine the cost of providing unrestricted lunar surface access.

After some consideration, it became clear that the "benign" lunar surface thermal environment described in the TRM assumptions does not truly exist for a seven-day period. A seven-day lunar surface stay represents approximately one-quarter of the lunar day/night cycle. Therefore, any mission restricted to daylight hours would necessarily include lunar noon. In addition, the lack of atmosphere and the composition of the lunar soil result in a fairly rapid day/night thermal transition from extremes heat to extreme cold. Each thermal environment has its own benefits and challenges in terms of lander subsystem design. The availability of sunlight enables the use of solar power, but also results in a hot surface environment that impacts EVA operations, active thermal control and propellant conditioning. The extreme cold of lunar night puts a premium on insulation and heaters and eliminates the option of using solar power. The availability of Earthshine, which has been likened to a dusk condition on Earth, may be sufficient to perform many surface operations.

The CEV will provide the crew habitation function from Earth's surface to lunar vicinity and back to Earth's surface: The cost of landing the entire CEV (including parachutes, TPS, outer skin, Earth landing systems, etc.) on the lunar surface and operating out of that vehicle for the duration of the surface mission is deemed to be too severe for a L1 rendezvous architecture. As the mass of a lander ascent stage has the biggest leverage on total architecture mass, a separate, highly optimized lunar lander element like the Apollo Lunar Module will be used for lunar exploration and the CEV will be used to deliver the crew to lunar vicinity.

The nominal Earth return for the CEV is a direct entry with a water landing: Crew safety requirements for aborted missions will require a direct entry capability, and with orbital mechanics restraints dictated with returns from the vicinity of the Moon, it may not be possible to guarantee a land landing in a favorable location every mission. Additionally, some aborts during ascent from Earth may result in water landings. A water landing and recovery will therefore be a capability required by the vehicle. Providing a capability to nominally perform both water and land landings will increase CEV and architecture mass, and CEV complexity (deployment of airbags, firing of retrorockets, etc.). To minimize impact to the CEV, the LDRM-2 study will assume that direct entry followed by water landing is the nominal mission mode, and an architecture trade will be performed in this study where a targeted land landing capability is assumed.

The CEV design will incorporate functionality for land landing as a contingency for an ascent abort: Some ascent aborts may result in the CEV landing on land. The ability to meet this requirement will be for crew survival only – *i.e.* the vehicle may be damaged beyond repair/reuse as long as the crew survives.

CEV shall include the capability for contingency EVAs: If the Lander Ascent Stage is unable to dock with the CEV upon return from the Moon, the CEV should be capable of supporting a contingency EVA transfer of the crew from that vehicle to the CEV. CEV EVA considerations are outlined in Section 19.11

Radiation shielding shall be incorporated into the design of the CEV and Lunar Lander crew modules to provide a core level of biological protection for the crew during transit and on the lunar surface: Radiation shielding is required to meet crew safety requirements during solar particle events (SPEs). Short-term and cumulative crew dose limits for exploration missions have not yet been defined.

Libration point L1 is used as the lunar vicinity rendezvous point to enable global lunar surface access: Due to the fixed geometry of the Earth-Moon-L1 system, using the Earth-Moon L1 point as a rendezvous location enables anytime lunar surface access with anytime ascent and return for the same total architecture cost. An architecture trade will be performed to determine the cost of the same capability using lunar orbit rendezvous.

Communications and tracking systems will be emplaced to support global human and robotic surface operations: The architecture will not be restricted to landing sites that have direct line of sight to Earth.

The Lunar Lander will be pre-deployed to lunar vicinity prior to initiation of the CEV mission: Deployment of the Lunar Lander prior to the crew launching in the CEV will simplify element interfaces and assembly requirements. An architecture trade will be performed where all elements are assembled as a single combined stack in low Earth orbit prior to departing for the Moon.

10.2 Architecture Description

As dictated by architecture assumptions laid out in the LDRM-2 task statement, assumptions that were made to provide an initial point of reference for architecture trade studies, the L1 trade reference mission is largely framed by the following eight key parameters:

– Four launches per mission
– Separate in-space transportation (CEV) and landing (Lunar Lander) vehicles
– Seven days on the lunar surface
– Four crew with all crew going to the surface
– Global lunar surface landing access (time-restricted)
– Anytime Earth return from the lunar surface
– Initial CEV/Lunar Lander rendezvous at Lunar L1
– Lunar Lander pre-deployed to Lunar L1

In addition to requiring a separate CEV and Lunar Lander, an Earth Departure Stage element will be used in the architecture to execute the necessary Earth orbit departure maneuvers. Several alternate architecture options functioning within these guidelines were initially constructed for selection of the LDRM-2 trade reference mission. These options focused on different propellant combinations and delta-V allocations for the Earth Departure Stages and Lander Descent Stage. In some options, a Kick Stage was added to the architecture to perform the libration point arrival, libration point departure, and lunar orbit insertion maneuvers, thereby off-loading the delta-V burden of the Lander Descent Stage. Seven independent architecture options were analyzed, and these seven were grouped into four categories according to the commonality of their features. They are as follows:

Group A:

1. This first option utilizes a single pump-fed oxygen (O_2) / hydrogen (H_2) Earth Departure Stage to perform Earth orbit departure and a pressure-fed O_2 / methane (CH_4) Kick Stage to perform libration point arrival. At L1, the crew transfers from the CEV to the Lander and undocks from the CEV. The Kick Stage then performs libration point departure, coasts to the Moon, and inserts the Lander into a 100 km circular lunar parking orbit. The pressure-fed O_2/CH_4 Descent Stage performs an in-plane descent to the nominal

landing site. After the seven-day mission on the lunar surface, a single-stage pressure-fed O_2/CH_4 Ascent Stage returns the crew to Lunar L1, where they re-dock with the loitering CEV. The crew transfers back to the CEV, undocks from the Ascent Stage, and returns to Earth via nominal direct entry. (Option A1)

2. This option differs from A1 in that a pump-fed O_2/H_2 Kick Stage is used instead of O_2/CH_4. (Option A2)
3. This option differs from A1 in that pump-fed O_2/H_2 systems are used in both the Kick Stage and Lander Descent Stage. (Option A3)

<u>Group B:</u>

4. For this option, two pump-fed O_2/H_2 Earth Departure Stages are stacked in serial to perform Earth orbit departure and libration point arrival for the Lunar Lander. These stages are sized to be identical in initial mass. The first stage performs ~50% of Earth orbit departure, that stage is then discarded, and the second Earth Departure Stage finishes the maneuver and performs L1 arrival. A single-stage pressure-fed O_2/CH_4 Lander Descent Stage instead of a Kick Stage is used for libration point departure, lunar orbit insertion, and descent. The remainder of the mission is identical to option A1. (Option B)

<u>Group C:</u>

5. Here, a single pump-fed O_2/H_2 Earth Departure Stage is used to perform both Earth orbit departure and libration point arrival. A single-stage pressure-fed O_2/CH_4 Lander Descent Stage instead of a Kick Stage is used for libration point departure, lunar orbit insertion, and descent. The remainder of the mission is identical to option A1. (Option C1)
6. This option differs from C1 in that a pump-fed O_2/H_2 Lander Descent Stage is assumed instead of a methane-based system. (Option C2)

<u>Group D:</u>

7. The final TRM option assumes a single pump-fed O_2/H_2 Earth Departure Stage is used to perform Earth orbit departure only. A single-stage pump-fed O_2/H_2 Lander Descent Stage is used for libration point arrival, libration point departure, lunar orbit insertion, and descent. The remainder of the mission is identical to option A1. (Option D)

The seven TRM architecture options were then analyzed to determine their approximate individual vehicle masses, mass required per launch, and overall architecture mass. Vehicle masses were estimated using the Envision parametric sizing tool. These results are summarized in Table 10.2-1.

	Launch #1	Launch #2			Launch #3	Launch #4	Total
	Lander EDS	Kick Stage	Descent Stage	Ascent Stage	CEV EDS	CEV	
Option A1: O2/CH4 Kick Stage (pressure-fed) + O2/CH4 Descent Stage (pressure-fed)	**99 T** O2/H2 ΔV=3,204 m/s	**36 T** O2/CH4 ΔV=1,834 m/s	**21 T** O2/CH4 ΔV=1,876 m/s	**18 T**	**36 T**	**25 T**	**235 T**
Option A2: O2/H2 Kick Stage (pump-fed) + O2/CH4 Descent Stage (pressure-fed)	**85 T** O2/H2 ΔV=3,204 m/s	**25 T** O2/H2 ΔV=1,834 m/s	**21 T** O2/CH4 ΔV=1,876 m/s	**18 T**	**36 T**	**25 T**	**210 T**
Option A3: O2/H2 Kick Stage (pump-fed) + O2/H2 Descent Stage (pump-fed)	**76 T** O2/H2 ΔV=3,204 m/s	**22 T** O2/H2 ΔV=1,834 m/s	**16 T** O2/H2 ΔV=1,876 m/s	**18 T**	**36 T**	**25 T**	**194 T**
Option B: 2 Identical Earth Departure Stages	**55 T x 2** O2/H2 ΔV=4,208 m/s	**N/A**	**35 T** O2/CH4 ΔV=2,756 m/s	**18 T**	**36 T**	**25 T**	**224 T**
Option C1: O2/CH4 Descent Stage (pressure-fed)	**110 T** O2/H2 ΔV=4,208 m/s	**N/A**	**35 T** O2/CH4 ΔV=2,756 m/s	**18 T**	**36 T**	**25 T**	**225 T**
Option C2: O2/H2 Descent Stage (pump-fed)	**90 T** O2/H2 ΔV=4,208 m/s	**N/A**	**25 T** O2/H2 ΔV=2,756 m/s	**18 T**	**36 T**	**25 T**	**194 T**
Option D: O2/H2 Descent Stage (pump-fed)	**74 T** O2/H2 ΔV=3,204 m/s	**N/A**	**37 T** O2/H2 ΔV=3,710 m/s	**18 T**	**36 T**	**25 T**	**189 T**

Table 10.2-1: TRM Downselection Trade Results

Option A1 was eliminated from further consideration because it required the highest total architecture mass and second-highest mass per launch for its Earth Departure Stage. Option A3 was attractive in that it offered a low total architecture mass and Earth Departure Stage size, however it was eliminated as a potential TRM because of suspected Lander packaging issues with large hydrogen tanks and its lower probability of mission success for lunar descent. A pressure-fed Descent Stage propulsion system using oxygen and hydrogen was not considered to be a reasonable option because of mass penalties incurred with large, high-pressure hydrogen tanks and the amount of pressurant needed for pressurizing hydrogen tanks to 250-350 psi (see Section 19.1 for additional trade details). Therefore, a pump-fed system was assumed, but because of this, the Lander Descent Stage of option A3 was thought to have a lower probability of mission success than other options as the pressure-fed propulsion systems of options A1 and A2 are inherently less complex and more reliable than a comparable pump-fed system. A more detailed analysis is recommended at some future point to determine the true costs and benefits of O_2/H_2 propulsion systems for landers.

Option B was removed as a potential TRM because it violated the four-launch constraint outlined in the LDRM-2 task statement, though it was noted that this option offered several positive features for a five-launch per mission architecture. Option B had the lowest mass per launch of all seven architectures and offered possible commonality between the CEV and Lander Earth Departure Stages. Next, Option C1 was discarded because it required the highest mass per launch for its departure stage, had a high total architecture mass, and required a large single-stage Lander Descent Stage to perform libration point departure through lunar descent. Positively, C1 did not require the development of the Kick Stage element, though this was not considered enough to offset its strongly negative features. Options C2 and D were also eliminated because of the difficulties with O_2/H_2 descent stages described above, though the benefits these two options offered were thought to be strong enough to warrant a future trade study.

The elimination of all other options led to A2 being selected as the LDRM-2 trade reference mission. This architecture offered both a moderate mass per launch requirement for its Earth Departure Stage and moderate total architecture mass. Further, the use of the Kick Stage to deliver the Lander to low lunar orbit allowed A2 to take advantage of high-performance O_2/H_2 propulsion for Lander's in-space transit to the Moon and higher-density O_2/CH_4 for powered descent to reduce the size of its Descent Stage. Requiring a smaller descent stage thereby minimized its required launch vehicle fairing diameter and crew height above the lunar surface. The trade reference mission, illustrated in Figure 10.2-1, was then used to perform all subsequent architecture and parametric variation trades.

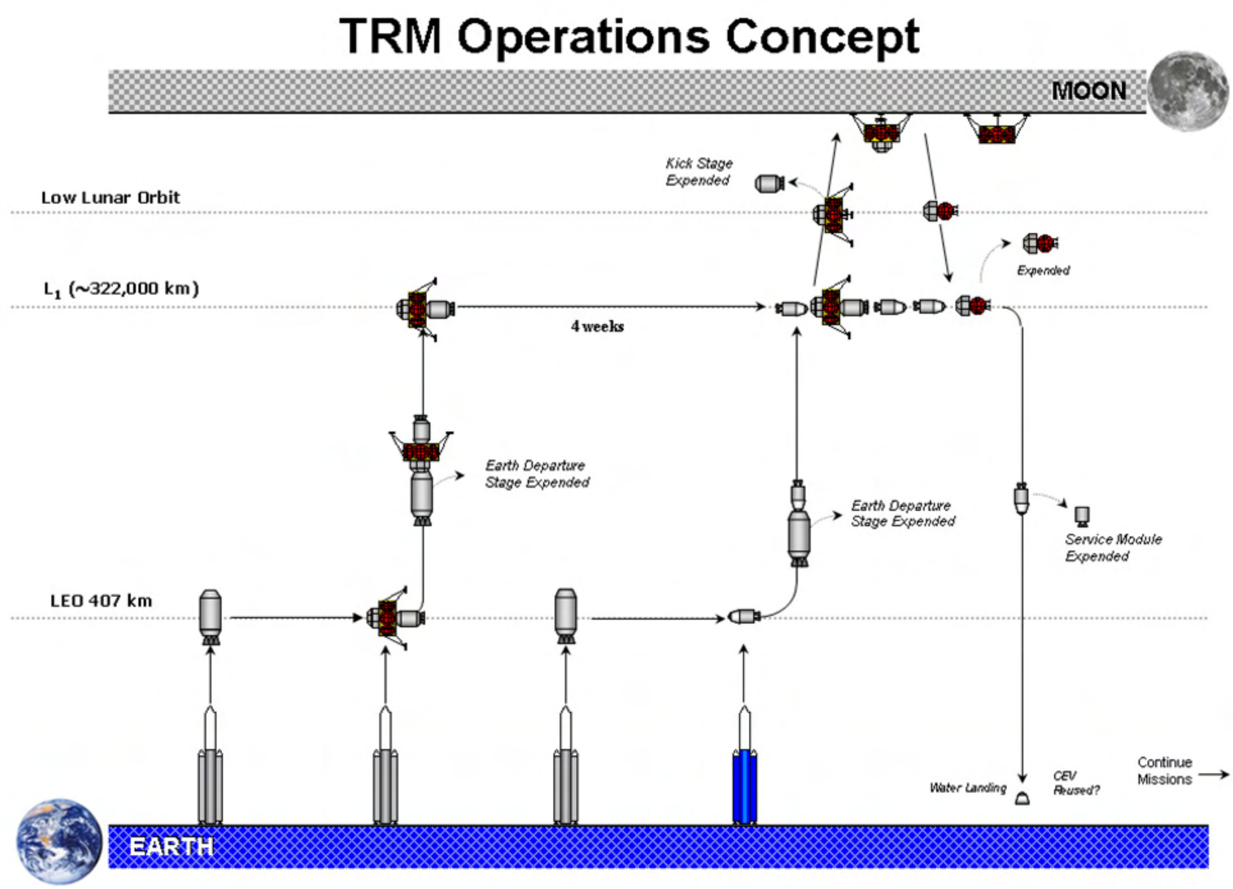


Figure 10.2-1: Trade Reference Mission Architecture Illustration

The trade reference mission begins with the launch of the Lander Earth Departure Stage. The assumed cargo launch vehicle for the architecture delivers that element to the LEO parking orbit previously assumed (28.5^{o} 407 km), where it loiters for assembly. Two weeks after the first launch, keeping with the assumed time between launches, the Lunar Lander and Kick Stage are delivered to LEO by a second cargo launch vehicle. The Lunar Lander and Kick Stage, acting as the chaser vehicle, perform a variable-length double coelliptic rendezvous maneuver profile to rendezvous and dock with the Earth Departure Stage (the target vehicle) within 50 hr after launch. A variable-length rendezvous profile enables a 360^{o} phase window for the Lander, which thereby affords a daily launch opportunity from the Eastern Test Range. After mating and vehicle checkout in LEO, the Earth Departure Stage performs the Earth orbit departure maneuver (3,104 m/s) for the Lunar Lander and Kick Stage, separates from the stack, and disposes itself. Several disposal options for the Earth Departure Stages are available for approximately equal delta-V costs, including Earth ocean disposal, lunar surface impact, and lunar fly-by to heliocentric orbit. After separating, the Lunar Lander and Kick Stage continue on and coast for 94 hr to the L1 libration point. Upon arrival in L1 vicinity, the Kick Stage orients the stack for the libration point arrival maneuver and performs the burn (954 m/s). The vehicles then loiter at the libration point until the CEV arrives with the crew several weeks later.

Two weeks after the Lunar Lander and Kick Stage are launched, a second Earth Departure Stage is launched to LEO to perform CEV injection to L1. Finally, two weeks later, the crew launch in the CEV (the 4th launch of a four launch per mission architecture) on a separate, human-rated launch vehicle. The CEV, as the chaser vehicle, performs a stable orbit rendezvous maneuver profile to rendezvous and dock with the CEV Earth Departure Stage within 50 hr after orbit insertion. Crew operational constraints such as required sleep periods preclude the use of a variable-length rendezvous profile, though as long as the time of theoretical braking occurs within 48 hr mission elapsed time, the CEV will have a 360^o phase window. As with the Lunar Lander, a 360^o phase window enables a daily launch opportunity. For the TRM, a 360^o phase window was chosen based on the assumed 407 km assembly orbit altitude to provide daily launch opportunities. Certain orbit altitudes exist, though, that provide 1-day phase-repeating orbits which would allow for flight day 1 rendezvous with the Earth Departure Stage instead of the 50-hr, flight day 3 rendezvous presently baselined. At 28.5^o inclination such orbits are found at 172 km and 474 km. The operational advantages of phase-repeating orbits should be analyzed in follow-on work.

In addition to the 50-hr rendezvous time, two extra days of on-orbit time were added in mission timeline budgeting for the trade reference mission. This time was added to the CEV's overall mission lifetime capabilities for weather-related launch delays prior to the opening of the Earth orbit departure window. The weather delay estimate was based on the historical 1.3-day delay average for the Space Shuttle program. The Space Shuttle historical average includes launch site, return to launch site (RTLS), and trans-Atlantic landing (TAL) weather restrictions, and it may be safe to assume that a vehicle that requires fewer abort site weather restrictions would have a smaller average delay. So while it was recognized that the human-rated launch vehicle developed for the CEV may be more impervious to launch delays than the Space Shuttle, given the present lack of abort scenarios for the CEV, and the criticality of performing Earth orbit departure on time, it was considered prudent to hold two days of launch delay protection at this time. However, assuming the CEV is able to successfully launch on the first available opportunity, the crew must still loiter for two additional days in LEO prior to Earth orbit departure. Injection opportunities to L1 arise when the Moon at the time of L1 arrival crosses the plane of the CEV Earth Departure Stage's orbit at the time of departure (*i.e.* the Moon's declination with respect to the Earth parking orbit equals zero). Regardless of how many extra days of timeline margin are carried, an on-time launch will not move up the opening of that injection window. The Lunar Lander and Kick Stage also carry 48 hr of weather delay protection to ensure those vehicles depart for L1 on time.

If launch delays do not allow the CEV to launch on one of the three daily launch opportunities bookkept in the mission timeline, and the first injection to L1 opportunity is missed, the orbital elements shall be designed to handle an extra 10 days of on-orbit lifetime. L1 injection opportunities from the reference LEO assembly orbit arise every 3-12 days (see Figure 10.2-2), with the average time between window openings being 10 days. Assuming a reasonable chance of missing one opportunity in a four launch per mission architecture, the Lander / Kick Stage and CEV Earth Departure Stage should each be capable of loitering for 10 additional days at Lunar L1 and LEO, respectively.

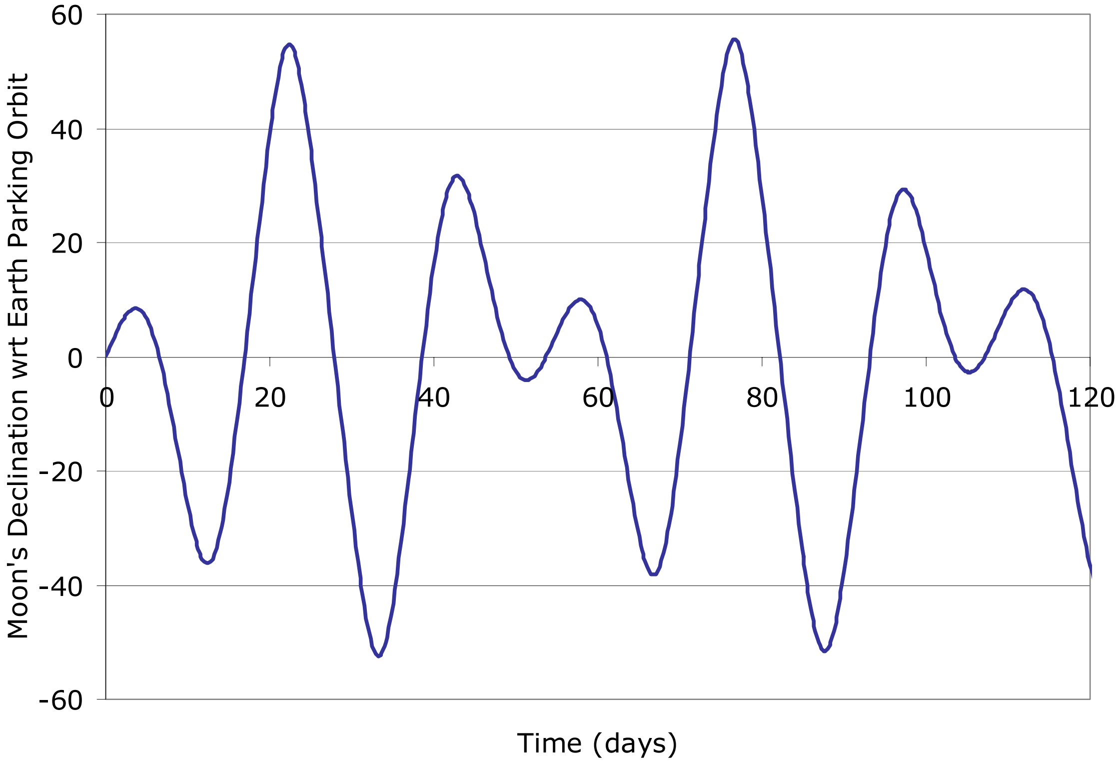


Figure 10.2-2: Frequency of Earth Orbit Departure Opportunities

Once the CEV mates with its Earth Departure Stage in LEO, the crew and mission control check out the vehicles and the EDS performs the Earth orbit departure maneuver (3,104 m/s) at the opening of the window. Once complete, the EDS separates from the CEV and disposes itself while the CEV coasts on a 94-hr transfer to L1. A 24-hr minimum delta-V injection window has been included in the sizing of the CEV Earth Departure Stage so flight time to L1 may vary between 94 hr for injection at the opening of the window to 70 hr for injection at window closing. Once at L1, the CEV performs a libration point arrival maneuver (954 m/s) to insert itself into an orbit in the vicinity of the Lunar Lander and Kick Stage, and then completes a series of rendezvous maneuvers to dock with those elements within 6 hr after arrival. All crewmembers then transfer over from the CEV to the Lunar Lander, complete any necessary vehicle check out tasks, and undock from the CEV. Next, the Kick Stage executes libration point departure (248 m/s) to target the Lander on a trajectory for insertion into a 100 km altitude low lunar orbit while the CEV loiters unoccupied at the libration point.

The selected L1-to-Moon trajectory is a near-minimum delta-V transfer with a flight time of 60 hr. At perilune, the Kick Stage will insert the Lunar Lander (632 m/s) into a temporary 100 x 100 km lunar parking orbit with an inclination appropriate for the selected landing site. A temporary parking orbit is used rather than direct hyperbolic descent for reasons of (1) crew safety, (2) to provide global access, and (3) a generally lower delta-V. For item (1), direct descent may require the perilune altitude of the inbound trajectory to be below the lunar surface, and engine ignition failure would result in lunar impact. After successful insertion into the lunar parking orbit, the Kick Stage separates from the Lunar Lander and disposes itself via lunar impact. The crew then executes deorbit and powered descent (1,876 m/s) to the surface with the Lander Descent Stage at the first available opportunity, which should occur within one orbit revolution. The availability of a given landing site for the trade reference mission was assumed to be restricted by lighting conditions during descent. The Apollo missions were conducted with a sun elevation angle between 7 to 20 degrees. The constraint was a derived requirement based on visual restrictions and landing fuel allotment. The first restriction was that the Sun had to be opposite the Lander's direction of travel to avoid having direct sunlight impair the crew's vision. The second restriction had two considerations. First, lunar surface photo resolution was unable to identify boulders or craters smaller than about 3 meters. Therefore the crew was required to pick an obstruction-free landing site. In order to pick a site, they had to visually identify the obstruction. Studies indicated that the contrast created by shadows was the best way to identify the obstructions, and in order for the crew to see the shadows, they had to approach their intended landing site with a "glide slope" greater than the sun angle that created the shadows. A higher sun angle therefore required an even higher glide slope. A higher glide slope in turn had the disadvantage in that a smaller portion of the descent engine thrust was used to slow the vehicle's horizontal velocity. The compromise was establishing a landing time that corresponded with a 7 to 20 degree sun angle at the intended landing site. Undoubtedly, other considerations played into this requirement as well. Although this trade reference mission was not constrained to follow the reported Apollo derived landing restrictions, it was considered to be a reasonable place from which to start. It would place the Lander on the surface starting 0.5 to 1.5 days after lunar dawn and ending 7.5 to 8.5 days after lunar dawn, allowing for a full 7 days of lighted surface activity. Advancements in landing aids, including high resolution surface imaging, laser- or camera-based visual aids, and emplaced surface targeting beacons may allow this visual landing restriction to be relaxed. If so, the Lander could theoretically land anytime in daylight or darkness.

Once on the lunar surface, the Lunar Lander provides the capability to operate and perform daily EVAs for up to 7 days as dictated in the LDRM-2 task statement. However, given the lack of a defined surface exploration strategy, no attempt was made to determine how the crew's time on the Moon might be spent. As the surface mission is expiring, the crew will prepare the Lander Ascent Stage for return to L1. The Ascent Stage separates from the Descent Stage on the lunar surface and ascends (1,829 m/s) to a 100 x 100 km temporary lunar parking orbit, where it loiters up to one orbit revolution until the lunar orbit departure window opens. The Lander Ascent Stage executes the lunar orbit departure maneuver (632 m/s), putting the vehicle and crew on a 60-hr transfer back to the CEV waiting at Lunar L1. Arriving in L1 vicinity, the stage performs a libration point arrival burn (248 m/s) and a series of rendezvous maneuvers to re-dock with the CEV within 6 hr.

After docking, the crew transfers back over to the CEV to start up and check out the vehicle, transfers over any cargo to be returned to Earth, and undocks from the Lander Ascent Stage. The CEV then executes a libration point departure burn (798 m/s) to target the CEV for atmospheric entry 82 hr later. Though the planned flight time is a near-minimum delta-V transfer lasting 82 hr, 360^{o} of landing site longitude control can be achieved by adding or subtracting 12 hr from the nominal flight time. The CEV upon reaching Earth directly enters the atmosphere and lands at a water landing site. While full landing site longitude control can be achieved by changing the return trip time, landing site latitude for direct entry will be dictated by the location of the Moon's antipode at the time of libration point departure. The Moon's antipode cycles between ±18.3^o and ±28.6^o latitude over its 27.3-day revolution about Earth, depending on how its rotation axis is oriented relative to Earth. The Moon's axis varies between ±5.1^o as measured relative to Earth's 23.5^o inclined rotation axis on an 18.6-year cycle, which gives rise to the 18.3^o to 28.6^o lunar antipode maximum latitude variation.

Three hours prior to atmosphere entry interface, the CEV separates the Crew Module (which contains the crew) from its service module for disposal and orients itself into the correct attitude for the entry phase. The service module is targeted for breakup in the atmosphere with a debris footprint uprange from the Crew Module's touchdown point. After landing successfully, the crew and cargo is assumed to be recovered within 2 hr of entry. Meanwhile, the Ascent Stage automatically performs a final maneuver for a controlled disposal. Even without this maneuver, such as with a dead Ascent Stage, the gravitational interactions from Earth, the Moon, and the Sun would eventually cause the stage to be disposed. The stage would either impact Earth or the Moon, or coast into a heliocentric orbit, though the specifics of the disposal would be uncontrollable.

Figure 10.2-3 and Tables 10.2-2 – 10.2-3 outline the assumed timelines and delta-V's for the trade reference mission as described above.

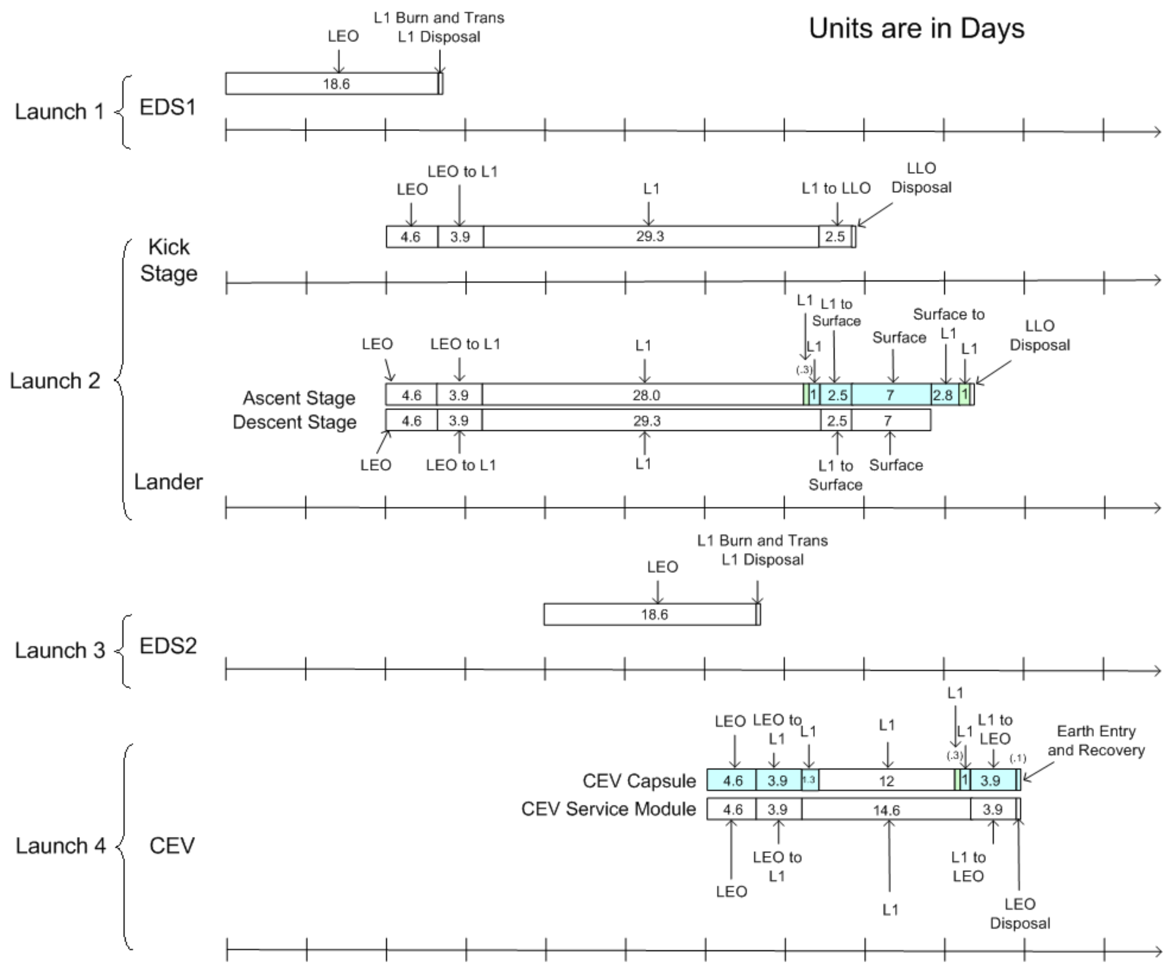


Figure 10.2-3: Nominal Timeline for the Trade Reference Mission

Vehicle	Phase Name	Phase Length	Mission Elapsed Time						
			Overall MET		EDS1	Kick Stage	Lander	EDS2	CEV
		(hr)	(hr)	(days)	(hr)				
EDS1	Launch from Earth/Loiter	2	2	0.1	2				
EDS1	Loiter in LEO	332	334	13.9	334				
Kick Stage/Lander	Launch Weather Delay	48	382	15.9	382	48	48		
Kick Stage/Lander	Launch from Earth/Loiter	2	384	16.0	384	50	50		
Kick Stage/Lander	Rendezvous & Dock w/ EDS	50	434	18.1	434	100	100		
EDS1/Kick Stage/Lander	Vehicle Checkout	12	446	18.6	446	112	112		
EDS1/Kick Stage/Lander	Missed EOD Opportunity	240	686	28.6	686	352	352		
EDS1	Earth Orbit Departure	0	686	28.6	686	352	352		
EDS1/Kick Stage/Lander	Coast	47	733	30.5	733	399	399		
EDS1	MCC & EDS Disposal	0	733	30.5	733	399	399		
Kick Stage/Lander	Coast	47	780	32.5		446	446		
Kick Stage/Lander	Libration Point Arrival	0	780	32.5		446	446		
Kick Stage/Lander	Loiter at L1	130	910	37.9		576	576		
EDS2	Launch from Earth/Loiter	2	912	38.0		578	578	2	
EDS2	Loiter in LEO	334	1246	51.9		912	912	336	
CEV	Launch Weather Delay	48	1294	53.9		960	960	384	
CEV	Launch from Earth/Loiter	2	1296	54.0		962	962	386	48
CEV	Rendezvous & Dock w/ EDS	50	1346	56.1		1012	1012	436	50
EDS2/CEV	Vehicle Checkout	12	1358	56.6		1024	1024	448	100
EDS2	Earth Orbit Departure	0	1358	56.6		1024	1024	448	112
EDS2/CEV	Coast	47	1405	58.5		1071	1071	495	112
EDS2	MCC & EDS Disposal	0	1405	58.5		1071	1071	495	159
CEV	Coast	47	1452	60.5		1118	1118		159
CEV	Libration Point Arrival	0	1452	60.5		1118	1118		206
CEV	Dock w/ Lander	6	1458	60.8		1124	1124		206
CEV/Kick Stage/Lander	Crew Transfer & Checkout	24	1476	61.5		1148	1148		212
Kick Stage/Lander	Undock from CEV	0	1476	61.5		1148	1148		236
Kick Stage	Libration Point Departure	0	1476	61.5		1148	1148		236
Kick Stage/Lander	Coast	60	1536	64.0		1208	1208		236
Kick Stage	Lunar Orbit Insertion	0	1536	64.0		1208	1208		296
Kick Stage	Kick Stage Disposal	0	1536	64.0		1208	1208		296
Lander	Powered Descent	0	1536	64.0			1208		296
Lander	Surface Mission	168	1704	71.0			1376		296
Lander	Ascent	0	1704	71.0			1376		464

Vehicle	Phase Name	Phase Length	Mission Elapsed Time						
			Overall MET		EDS1	Kick Stage	Lander	EDS2	CEV
		(hr)	(hr)	(days)	(hr)				
Lander	Lunar Orbit Departure	0	1704	71.0			1376		464
Lander	Coast	60	1764	73.5			1436		464
Lander	Libration Point Arrival	0	1764	73.5			1436		524
Lander	Rendezvous & Dock w/ CEV	6	1770	73.8			1442		524
Lander/CEV	Crew Transfer & Checkout	24	1794	74.8			1466		530
CEV	Undock from Lander	0	1794	74.8			1466		554
Lander	Ascent Stage Disposal	0	1794	74.8			1466		554
CEV	Libration Point Departure	0	1794	74.8					554
CEV	Coast	91	1885	78.5					554
CEV	Dispose Service Module	0	1885	78.5					645
CEV	Coast & Entry	3	1888	78.7					645
CEV	Recovery	1	1889	78.7					648

Table 10.2-2: TRM Mission Phase Description

Maneuver Name	Element	ΔV (m/s)	Comments
Earth Orbit Departure	Lander EDS	3,104	Co-planar departure from LEO assembly orbit (407 km, 28.5°) w/ 24-hr injection window. Nominal flt time to L1 = 82 hr. Moon @ perigee.
Libration Point Arrival	Kick Stage	954	L1 insertion w/ 57.1° plane change (worst-case inclination of transfer orbit w.r.t. Earth-Moon plane) & 24-hr EOD injection window.
Earth Orbit Departure	CEV EDS	3,104	Co-planar departure from LEO assembly orbit (407 km, 28.5°) w/ 24-hr injection window. Nominal flt time to L1 = 82 hr. Moon @ perigee.
Libration Point Arrival	CEV	954	L1 insertion w/ 57.1° plane change (worst-case inclination of transfer orbit w.r.t. Earth-Moon plane) & 24-hr EOD injection window.
Libration Point Departure	Kick Stage	248	Target for 100 km polar orbit (worst case). Nominal flt time to lunar orbit = 60 hr.
Lunar Orbit Insertion	Kick Stage	632	Insertion into 100x100 km polar orbit (worst case).
Descent	Descent Stage	1,876	Fuel-optimal powered descent design for in-plane descent from 100x100 km polar orbit (ref. First Lunar Outpost study)
Ascent	Ascent Stage	1,829	Fuel-optimal powered ascent design for in-plane ascent to 100x100 km polar orbit (ref. First Lunar Outpost study)
Lunar Orbit Departure	Ascent Stage	632	Departure from 100x100 km polar orbit (worst case). Nominal flt time to L1 = 60 hr.
Libration Point Arrival	Ascent Stage	248	L1 insertion from 100 km polar orbit (worst case).
Libration Point Departure	CEV	798	Earth return transfer w/ 40° transfer orbit inclination w.r.t. Earth equator for favorable landing latitude & 24-hr departure window. Earth vacuum perigee altitude = 38 km. Nominal flt time to Earth = 82 hr.

Table 10.2-3: Summary of Major Maneuvers for the TRM

10.3 Safety & Mission Success

The following sections detail the methodologies used to develop Safety and Mission Success products for the LDRM-2. The products generated are intended to support the Safety/Reliability Figures of Merit (FOMs) described in the LDRM-2 Requirements Formulation Task (RFT) Task Team Request RFT 0001.04.

10.3.1 Critical Event Identification

For the purpose of this LDRM-2 study, critical events are defined as any event during the mission timeline that is required to be successfully completed to avoid either a loss of the mission (LOM) or a loss of the crew (LOC). Fifty-six critical events were identified for the trade reference mission (TRM). Out of the fifty-six critical events identified, nineteen occurred during uncrewed portions of the mission while the remaining thirty-six occurred during the crewed portions of the mission. Each TRM critical event identified was assigned an identification number. The critical event identification numbers will be used to map the TRM and various architecture option critical events to related risks or hazards (refer to Section 20.16 Risks and Hazards Assessment). As the TRM critical events were identified, they were arranged in sequential order and reviewed with LDRM-2 team members. Once the sequence ordering and terminology of critical events were reviewed and approved by the participating team members, the TRM critical events were assigned a rank describing their importance. The critical event ranking methodology used is described in section 10.3.2.

In an effort to overlay the TRM critical events with the TRM mission abort opportunities, each TRM critical event was categorized into a mission phase. Section 10.4 discusses the TRM mission abort opportunities per mission phase in detail.

10.3.2 TRM Critical Event Ranking

Due to the TRM critical event descriptions being very general, it was decided to keep the critical event ranking criteria at a high-level for consistency purposes. A simplistic approach was used for determining the critical event ranking methodology. The TRM critical events were assigned a ranking of 1, 2, or 3, with 3 representing the most critical of mission events. The ranking definitions are defined as follows:

Rank of 1: *Failures during mission critical events that could lead to a Loss of Mission (LOM) but not a Loss of Crew (LOC).*

Rank of 2: *Failures during mission critical events that could lead to a LOC but would have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

Rank of 3: *Failures during mission critical events that would not have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

The LDRM-2 team members that reviewed and approved the sequence order and critical event terminology participated in ranking the TRM critical events. Seven of the fifty-six critical events received a rank of 3. Twenty-four of the fifty-six critical events received a rank of 2, while the remaining twenty-five critical events received a rank of 1. The complete set of identified and ranked critical events for the TRM is listed in Table 10.3.2-1.

	TRM Mission Phase	ID #	TRM Critical Events	TRM Critical Event Rank
Uncrewed Critical Events	EDS-1 Launch & Ascent to LEO	TRM-01	EDS-1 (for the LL) Launch	1
		TRM-02	EDS-1 Ascent	1
		TRM-03	EDS-1 Launch Shroud Separation	1
		TRM-04	EDS-1 Separation from Booster	1
	EDS-1 Orbit in LEO	TRM-05	EDS-1 Orbital Maneuvering	1
	LL & Kickstage Launch & Ascent to LEO	TRM-06	LL & Kickstage Launch	1
		TRM-07	LL & Kickstage Ascent	1
		TRM-08	LL & Kickstage Launch Shroud Separation	1
		TRM-09	LL & Kickstage Separation from Booster	1
	LL, Kickstage, & EDS-1 Orbit & Rendezvous	TRM-10	LL & Kickstage Orbital Maneuvering	1
		TRM-11	LL & Kickstage Docks to EDS-1	1
	LL, Kickstage, & EDS-1 LEO to L1 Transfer	TRM-12	EDS-1, Kickstage, & LL Burn for L1	1
		TRM-13	LL & Kickstage Separates from EDS-1	1
		TRM-14	Kickstage & LL Mid-course Correction Burn	1
	LL & Kickstage L1 Ops	TRM-15	Kickstage & LL Burn to Slow Near L1	1
	EDS-2 Launch & Ascent to LEO	TRM-16	EDS-2 (for CEV) Launch	1
		TRM-17	EDS-2 Ascent	1
		TRM-18	EDS-2 Launch Shroud Separation	1
		TRM-19	EDS-2 Separation from Booster	1
		TRM-20	EDS-2 Orbital Maneuvering	1
Crewed Critical Events	CEV Launch & Ascent to LEO	TRM-21	CEV (CM+SM) Launch	2
		TRM-22	CEV Ascent	2
		TRM-23	LAS Separation	2
		TRM-24	CEV Launch Shroud Separation	2
		TRM-25	CEV Separation from Booster	2
	CEV LEO Orbit & Rendezvous Ops	TRM-26	CEV Orbital Maneuvering	2
		TRM-27	CEV Docks to EDS-2	2
	CEV LEO to L1 Transfer	TRM-28	EDS-2 & CEV Burn for L1	2
		TRM-29	CEV Separates from EDS-2	2
		TRM-30	CEV Mid-course Correction Burn	1

	TRM Mission Phase	ID #	TRM Critical Events	TRM Critical Event Rank
	CEV, LL, & Kickstage L1 Ops	**TRM-31**	CEV Burn to Slow Near L1	2
		TRM-32	CEV Orbital Maneuvering	2
		TRM-33	CEV Docks to LL & Kickstage	2
		TRM-34	Crew Transfer from CEV to LL	1
		TRM-35	LL & Kickstage Separates from CEV	2
	LL & Kickstage L1 to LLO Transfer	**TRM-36**	LL & Kickstage Burns for Low Lunar Orbit	2
		TRM-37	LL & Kickstage Mid-course Correction Burn	1
	LL & Kickstage LLO Insertion	**TRM-38**	LL & Kickstage Lunar Orbit Insertion (LOI)	2
		TRM-39	Kickstage Separates from LL	2
	LL LLO to Powered Descent Initiation	**TRM-40**	LL Deorbit Burn to Moon	2
	LL Powered Descent Initiation to Lunar Surface	**TRM-41**	LL Powered Descent & Landing to Moon (Includes all Critical Burns)	3
	LL Ascent Stage Lunar Ascent to LLO	**TRM-42**	LL Ascent Stage Separation & Ascent	3
		TRM-43	LL Ascent Stage Orbital Maneuvering	3
	LL Ascent Stage LLO to L1 Transfer	**TRM-44**	LL Ascent Stage Lunar Orbit Departure	3
		TRM-45	LL Ascent Stage Mid-Course Correction Burn	1
	LL Ascent Stage & CEV L1 Ops	**TRM-46**	LL Ascent Stage L1 Arrival	3
		TRM-47	LL Ascent Stage Orbital Maneuvering	2
		TRM-48	LL Ascent Stage Docks with CEV	2
		TRM-49	Crew Transfer from LL to CEV	2
		TRM-50	CEV Separates from LL Ascent Stage	2
	CEV L1 to Earth Transfer	**TRM-51**	CEV Burn for Earth	3
		TRM-52	CEV Mid-course Correction Burn	1
		TRM-53	CM Separates & Maneuvers away from SM	2
	CM Earth Reentry to Touchdown	**TRM-54**	CM Entry	3
		TRM-55	CM Landing	2
		TRM-56	Crew Recovery	2

Table 10.3.2-1: TRM Critical Events Ranking

Due to the TRM critical event descriptions being generalized, assumptions were made and considered in the ranking. A blanket assumption was made for having redundant pyrotechnic separation provisions for all mechanical separation events (i.e. undocking). This assumption was the difference for some critical events receiving a rank of 2 as opposed to a rank of 3.

Five of the critical events received specific individual assumptions. Critical event TRM-31 (CEV Burns to Slow Near L1) was ranked a 2 assuming redundancy in the main propulsion system would help mitigate any failure resulting in a perigee change that might not allow the CEV to return to Earth. Critical event TRM-35 (LL & Kickstage Burns for Low Lunar Orbit) considers what happens to the Lunar Lander & Kickstage when there is a partial burn and early engine cutoff during the transfer to low lunar orbit. It was assigned a rank of 2 due to having the Lunar Lander Ascent Stage available for a mission abort and return to L1. Critical event TRM-40 (LL Powered Descent & Landing to Moon) was ranked a 3 due to having insufficient knowledge regarding black zones for a Lunar Lander powered descent and landing. Black zones indicate periods of time where the vehicle cannot respond to an engine failure and return to orbit during the time period that a catastrophic outcome would occur. Critical event TRM-52 (CM Separates & Maneuvers away from SM) was given a ranking of 2 based on the assumption the Crew Module (CM) could separate and maneuver away from the Service Module (SM) during re-entry. Lastly, critical event TRM-53 (CM Entry) was assigned a rank of 3 assuming the potential for a thermal protection system/heat shield (TPS) burn-through, without functional redundancy for this system, during re-entry.

10.3.3 Top TRM Safety Concerns Based on the Critical Event Ranking

Safety & Mission Assurance (S&MA) performed an evaluation of mission critical events and the hazards associated with the operations concept for the TRM. Hazard identification methodology was based on previous hazards analyses developed for the Space Shuttle Program and exploration studies. Hazardous conditions were identified for Contamination in the Habitable Volume, Electrical Shock, Environmental (temperature, humidity), Fire and Explosion, Impact/Collision, Loss of Habitable Environment, Physiological/Psychological, Loss of Vehicle Control, Radiation, Contingency EVA Operations, Inability to Dock, Transfer Crew, and Undock, Inability to Egress Vehicle after Contingency Earth Return, and Loss of Entry Capability. The complete set of hazardous conditions, causes, effects, and potential controls are located in Section 20.17 of the Appendix. In addition to the hazardous conditions, multiple subsystem risks were identified for the study. Example subsystems included, but are not limited to, avionics, propulsion, electrical power system, and active thermal control system. The documented subsystem risks are somewhat generalized in order to be consistent with the level of detail provided in this study. The complete set of system risks is also located in Section 20.17 of the Appendix.

The goal was to identify top TRM safety concerns based on the critical events and their ranking. Hazards and safety concerns are usually not isolated to one event phase of the mission. Safety concerns can be categorized into significant phases of the mission.

Major safety concerns can be attributed to hazards associated with *Launch Events and Environments*. The crew is subjected to high dynamic forces when the vehicle is transferring large amounts of energy into thrust for very short time periods during the on-orbit phases. The crew is also exposed to additional high dynamic loads and forces associated with the effect of leaving and re-entering the Earth's atmosphere.

Upon successful placement in low Earth orbit (LEO) (and other phases of the mission) the crew is subjected to mission specific risks and safety hazards associated with vehicle *Rendezvous,* *Docking, and Separation* events. During these events the crew must have controls in place to mitigate the higher risk of vehicle damage due to unintended contact between vehicles which could result in an injury to the crew, a loss of crew (LOC), or a loss of specific vehicle function resulting in a loss of mission (LOM).

The Insertion/Departure phases of the mission could be broken down into three sub-phases: *Earth Orbit Insertion/Departure*, *Lunar Orbit Insertion/Departure*, and *Libration Point Insertion/Departure*. During these phases, hazards associated with long duration physiological effects on the crew become significant safety concerns. These phases will subject the crew to long periods of minimal activity. Major safety controls will need to be in place to mitigate bone loss and other physiological effects. The insertion and departure phases will also subject the vehicle and crew to long sustained periods of radiation exposure. The vehicle and crew may also be subjected to additional environments that may increase the likelihood of certain hazards during the insertion and departure phases as well.

In preparation for the lunar surface phase of the mission, the crew will be subjected to hazards associated with *Lunar Descent, Ascent and Surface Ops*. This mission phase contains EVA and Vehicle Lunar Surface operations. Unique hazards will require controls that involve both Intravehicular Activities (IVA) as well as all Extravehicular Activities (EVA) operations.

To achieve a successful return to Earth the vehicle and crew will experience hazards associated with *Earth Arrival, Atmospheric Entry, and Recovery.* This phase involves events associated with the CEV propulsion burn required for return to Earth, CEV re-entry events, and crew recovery.

The final category where a significant amount of safety and risks exist, are the hazards associated with *Aborts and Crew Escape.* Unique hazards will be present in all Mission Abort and Crew Escape scenarios.

Based on the ranking criteria described in Section 10.3.2, the following table is a summary of the highest risk (Rank of 3) mission critical events and corresponding mission phases. All of these critical events could result in the loss of a crewmember. Upon comparison among the seven TRM critical events ranked as a 3, five occur during the timeframe between Lunar Lander descent to the lunar surface and the Lunar Lander arrival at L1.

ID #	Critical Event Description	Mission Phase
TRM-41	LL Powered Descent & Landing to Moon (includes all Critical Burns)	Lunar Descent and Surface Ops.
TRM-42	LL Ascent Stage Separation & Ascent	Lunar Ascent, Separation, and Surface Ops.
TRM-43	LL Ascent Stage Orbital Maneuvering	Lunar Ascent
TRM-44	LL Ascent Stage Lunar Orbit Departure	Lunar Ascent and LOD
TRM-46	LL Ascent Stage L1 Arrival	Lunar Ascent
TRM-51	CEV Burn for Earth	L1-D and EOI
TRM-54	CM Entry	Atmospheric Entry

Table 10.3.3-1: TRM Critical Events Receiving a Rank of 3

10.3.4 Top TRM Mission Success Concerns

From a Safety & Mission Assurance (S&MA) viewpoint, there are several operational issues in the TRM affecting the probability of mission success. The TRM contains five elements (Earth Departure Stage-1 (EDS-1), EDS-2, Kickstage, Lunar Lander, and the Crew Exploration Vehicle (CEV)) which are put into orbit on four separate launch vehicles. Inherently, the more elements or critical events contained in a mission architecture, the lower the probability of mission success will be. For example, increasing the number of high-energy transfer events will lower the probability of mission success. High-energy events are events such as launches, separations, and engine burns. The TRM contains a total of four launches, eleven element separations, and six element engine burns. Other critical events, such as automated docking and rendezvous of unmanned and manned elements, will contribute to lowering the probability of mission success.

Having multiple launches (total of four) in the TRM will also negatively affect the probability of mission success. Simply by having four launches, the chances of having a launch delay due to weather or a mechanical failure increases dramatically. Launch delays prior to the first element launch will not have as large an impact on the probability of mission success as a launch delay succeeding the first launch. Given the first launch occurs successfully and there is a launch delay for any of the remaining three elements, micrometeoroid & orbital debris (MMOD) strikes on the EDS-1 while loitering in LEO may become a concern. However, if any mission architecture element is stranded at L1 for a period of days, orbital debris strikes will become much less of a concern than micrometeoroid strikes. Thus the likelihood of a mission architecture element being struck by MMOD will never be any higher than what it is in LEO. Additional analyses should be completed to show the impact these critical events have on the probability of mission success.

10.3.5 Potential Mitigation Plans for TRM Critical Events

This analysis identified TRM critical events with a rank of 3 that present a challenge for the project to provide adequate mitigation options. The basic premise behind the success of any risk management system is to develop mitigation strategies that identify and implement actions and events needed to control a particular hazard or risk prior to it becoming a problem.

Certain situations might necessitate additional layers of *Redundancy* to mitigate a risk to success. Redundancy as a mitigation is not without its inherent costs. When redundancy is chosen, the mass and complexity of the system is virtually always increased. With a redundancy solution, added controllers and software must have a robust Integrated Vehicle Health Monitoring (IVHM) system to allow for switching parallel strings seamlessly, quickly, and without affecting the critical vehicle operations.

Whenever possible, the most logical path for mitigation is to increase the *Reliability* of the system or component. The level of reliability for components is based on the worst case environments the part must be required to function within for a specific amount of time.

Critical operations, for both on-orbit and lunar surface activities, will require *Contingency Plans* or Malfunction (MAL) procedures to be in place and ready to provide a work-around for any operational situation requiring mitigation. MAL procedures allow the crew to work from a pre-prepared plan to mitigate a problem, or improvise to adjust for unforeseen failures and/or anomalies.

In cases of failures or situations that would lead to a Loss of Crew (LOC), Loss of Vehicle (LOV), or Loss of Mission (LOM), the only viable option available for mitigation might be to activate *Crew Escape/Mission Abort* systems and return to either Earth or a safe haven location. Crew escape and mission aborts require a radical deviation in the mission plan. In addition to using crew escape and mission aborts for mitigating vehicle failures, this strategy can also be activated to return an injured or critically ill crewmember quickly back to Earth.

Based on the ranking criteria described in Section 10.3.2, Table 10.3.5-1 lists the High Risk TRM Critical Events (Rank of 3). These events were identified as mission phases that have the most amount of risk involved with a lack of mission abort or emergency procedure mitigation option to prevent LOC. The potential mitigation column is used to identify possible areas of development to lower the risk likelihood or consequence.

ID #	Critical Event Description	Potential Critical Event Mitigations
TRM-41	LL Powered Descent & Landing to Moon (includes all Critical Burns)	Increased Reliability, Redundancy, Contingency Plans
TRM-42	LL Ascent Stage Separation & Ascent	Increased Reliability, Redundancy, Contingency Plans
TRM-43	LL Ascent Stage Orbital Maneuvering	Increased Reliability, Redundancy, Contingency Plans
TRM-44	LL Ascent Stage Lunar Orbit Departure	Increased Reliability, Redundancy, Contingency Plans
TRM-46	LL Ascent Stage L1 Arrival	Increased Reliability, Redundancy, Contingency Plans
TRM-51	CEV Burn for Earth	Increased Reliability, Redundancy, Contingency Plans
TRM-54	CM Entry	Increased Reliability, Redundancy, Contingency Plans

Table 10.3.5-1: Potential Mitigation Areas for TRM Critical Events Assigned a Rank of 3

10.3.6 TRM Critical Events vs. Apollo 17 Mission Profile Critical Events

In an effort to better understand risk to mission success, a timeline from past missions can be used to identify the critical event similarities and differences between Apollo 17 and the TRM. A total of twenty-five critical events were identified for the Apollo 17 mission. All twenty-five events that occurred involved the crew. There were zero uncrewed critical events for this mission. Of the twenty-five crewed critical events, only four received a ranking of 3. Twenty critical events were ranked as a 2, while a single event was ranked 1. The same ranking methodology used for the TRM critical events was applied to Apollo 17's critical events.

Critical Event APOLLO-13 (Lunar Module Descent & Landing to the Moon) was ranked a 3 due to the Apollo 17 mission having a black zone where in the event of a descent engine failure, the descent stage could not be separated and the ascent stage engine started in time to prevent a catastrophic crash onto the lunar surface. Critical event APOLLO-16 (Lunar Module Ascent Stage Separation & Ascent) was ranked a 3 due to the Lunar Module having only a single engine during its ascent from the lunar surface to lunar orbit. The single ascent engine on the Lunar Module did not provide an engine-out capability during this phase to prevent a loss of crew (LOC). Critical event APOLLO-21 (Trans Earth Injection Burn) was necessary for returning the Apollo 17 crew back to Earth. Apollo 17 had no engine-out capability on the Service Module after the Lunar Module had been expended. Thus, APOLLO-21 received a ranking of 3. Critical event APOLLO-23 (CM Entry) received a ranking of 3 due to the Apollo 17 Command Module not having a redundant TPS system. If a TPS burn-through would have occurred during the Apollo 17 atmospheric entry, there were no additional resources to prevent a loss of crew (LOC).

As compared to the number of TRM critical events, Apollo 17 had twenty-nine fewer. Apollo 17 also had fewer critical events that received a ranking of 3. The majority of Apollo 17's critical events had some type of mitigating event feature(s) during crewed phases that would prevent a loss of crew (LOC) from occurring. Listed in Table 10.3.6-1 are the critical events and associated rankings from the Apollo 17 mission timeline.

	ID #	Derived Critical Events from Apollo 17 Technical Debrief	Apollo 17 Critical Event Rank
Crewed Vehicle Events	**APOLLO-01**	Launch	2
	APOLLO-02	Ascent	2
	APOLLO-03	Booster Separation	2
	APOLLO-04	LES Separation	2
	APOLLO-05	TLI Burn	2
	APOLLO-06	SLA Separation	2
	APOLLO-07	LM Dock	2
	APOLLO-08	LM Separation from Saturn-IVB	2
	APOLLO-09	Mid-course Correction Burn	2
	APOLLO-10	Lunar Orbit Insertion Burns	2
	APOLLO-11	Crew Transfers from CSM to LM	1
	APOLLO-12	LM Separates from CSM	2
	APOLLO-13	LM Powered Descent & Landing to the Moon	3
	APOLLO-14	CSM Plane Change Burn	2
	APOLLO-15	CSM Orbital Maneuvering	2
	APOLLO-16	LM Ascent Stage Separation & Ascent	3
	APOLLO-17	LM Orbital Maneuvering	2
	APOLLO-18	LM Docks with CSM	2
	APOLLO-19	Crew Transfers from LM to CSM	2
	APOLLO-20	CSM Separation from LM	2
	APOLLO-21	CSM Trans Earth Injection Burn	3
	APOLLO-22	CM Separates from SM	2
	APOLLO-23	CM Entry	3

	ID #	Derived Critical Events from Apollo 17 Technical Debrief	Apollo 17 Critical Event Rank
	APOLLO-24	CM Landing	2
	APOLLO-25	Crew Recovery	2

Table 10.3.6-1: Apollo 17 Critical Events Ranking

10.4 Mission Abort Options

Crew Survival encompasses all activities necessary to acquire a collective understanding of all threats to the life of a crewmember and provide for integrated crew survival solutions. These solutions are considered in terms of abort, escape, egress, safe haven, and rescue with respect to all human exploration systems and are applied across both flight and ground segments throughout all mission phases. Although it may not fit the classical definition of a subsystem, the primary function of the LDRM aborts are to ensure crew survival throughout the mission by the safe return of the crew to the Earth in the event that something occurs that precludes mission continuation. This includes the tasks necessary to identify, develop and test systems that ensure crew survival in the presence of catastrophic events.

10.4.1 Subsystem Description

a. Primary Functions

The NASA Human Rating Requirements and Guidelines for Space Flight Systems, NPR: 8705.2, (HRR) provides the following definitions of abort and crew escape:

> **Abort**: The successful recovery of the space flight system and its crew and passengers in the event of an anomaly that precludes mission continuance. One type of abort (intact) allows recovery without exceeding stability, control, thermal, or physiological limits, and the other type (contingency) may result in exceeding system limits in the process.
>
> **Crew Escape**: The successful recovery of the space flight system crew and passengers in the event of an anomaly that precludes mission continuance. The space flight system in this scenario is abandoned and presumably lost.

Subsequently, the Orbital Space Plane Human Rating Plan (OSP Plan-10) provided the following more specific definitions:

> **Abort**: In the event of an anomaly that precludes mission continuance, the safe return to Earth of the crew inside the spacecraft. Rescue of the crew is mandatory. The capability to recover and, if applicable, re-use the spacecraft is governed by the vehicle failure tolerance requirements in the System Requirements Document. On re-entry, "mission continuance" refers to the capability of the spacecraft to safely touchdown at the targeted site.

Crew Escape: 1. The safe return to Earth of the crew in the event of an anomaly that requires the crew to exit the spacecraft using an escape system (e.g. extraction, ejection, escape pod), or 2. The safe return to Earth of the crew inside a crew compartment that is no longer an integral part of the spacecraft element designed for nominal re-entry and touchdown.

The Human Rating Requirement (NPR 8705.2) also states that Beyond Earth Orbit (BEO) missions require unique abort and survival modes. Missions designed for BEO require sufficient power, consumables, and trajectory design to maximize abort capabilities to ensure crew survival. These abort modes include, but are not limited to, powered return, free return, pre-positioning capabilities, and safe haven. In general, this mission profile requires the space flight systems and its propulsion system to have sufficient propellant to fly off-nominal trajectories. Critical systems should also be designed so that failures do not result in a catastrophic event. The design should provide time for other systems or the crew to recover from a critical system failure. As a last resort, when abort modes are not feasible, a safe haven capability should be provided to ensure that survival capability and consumables exist to return the crew to a position from which a rescue can be conducted. Consideration should be given to pre-positioning consumables, spare parts, and other critical logistics and services to improve abort and safe haven capabilities.

The BEO mission must meet a high probability of safe crew return over the life of the program. However, the higher mission complexity and length is offset by the fact that there may be only a few missions conducted at that level of technical and safety risk. As experience with the mission grows and the possibility of establishing a permanent outpost or colony arises, the reliability goal for each individual mission must rise to account for the increased flight rate and consequent exposure. Autonomy, functional redundancy, and tools to deal with the unexpected are a critical part of the design for safety. Technology will likely pace the schedule for accomplishing this.

Loss of life will be prevented through the integrated efforts of accurate identification of critical and catastrophic hazards, along with understanding the likelihood of their occurrence; establishing a high confidence in the detection of the hazards and the corresponding hazard mitigation responses; accommodating complex interfaces; providing for the effective function of system solutions and the timely completion of manual or automated procedures. These aspects will be clearly related, traced and validated to provide an overall measure of crew survival for the program.

b. Key design parameters (design drivers)

The key design parameters that will drive the design of and options for LDRM aborts are the propulsive delta velocity required for any abort maneuvers, the overall trip time, the abort return to Earth or return to the Crew Exploration Vehicle (CEV) times, and any safe haven durations. These parameters will impact the overall mass of the LDRM architecture and the mass of each stage in the architecture. Additionally, crew volume and the accessibility to internal systems to effect system repairs and the accessibility, size and number of hatches to support egress and ingress activities will drive the design of the CEV and the Lander.

c. Typical redundancy/reliability design approach

To be compliant with the Human Rating Requirement all subsystems should be designed to be two failure tolerant to any loss of life event. However, this requirement is often set aside for systems where it may be impractical to implement such as primary structure and pressure vessels. In general all LDRM elements associated with crew transport and habitation during the mission should be designed to provide functional redundancy if physical redundancy is not practical.

d. Typical vehicle resource requirements

The crew transportation elements of the LDRM must be designed to provide the necessary resources to power the systems and provide adequate life support for the crew during the completion of the nominal mission and any potential aborts, whichever is larger. In addition, the elements may be required to provide the resources to provide a safe haven for the crew while they await an Earth based rescue mission. This safe haven time has not been determined but could range from weeks to months. Providing for safe haven will have significant impact on the resources required to support the crew.

e. Potential for resource conservation during coast or parking orbit mission phases

In order to reduce or minimize the resource requirements during any abort option the crew transportation elements should be designed to support both extensive power down of redundant systems and an increased time to the catastrophic effect such that the crewed vehicle requires fewer systems to remain operating at any given time in the mission.

f. Potential vehicle design interactions or synergy

The following vehicle design interactions could enhance crew survival by increasing resources available to the crew for life support:

1. Fuel cells produce potable water for crew consumption.
2. Cryogenic propellants common to power system and crew life support might support longer duration safe haven concepts.
3. Common propellants among the various stages with the appropriate propellant interconnects and transfer capability would allow the use of one stage's propellant with another stage's engines thus allowing for the successful completion of a major propulsive maneuver.

10.4.2 Technology Options

The following subsystem technologies have the potential to offer significant functional improvements for crew survival:

1. Advanced Integrated Vehicle Health Management (IVHM) and real time failure prognostics, detection and mitigation software for autonomous software control of critical subsystems
2. Advanced avionics prognostics capability
3. Low power, reduced size avionics
4. Advanced crew situational displays and controls
5. Autonomous flight manager
6. High speed atmospheric re-entry stabilization and deceleration devices
7. Light weight regenerative life support systems

Nominal Flight Sequence

1. Launch From KSC
2. EOD (Earth Orbit Departure) and Booster Separation
3. L1 Arrival
4. Crew Transfers to Lander Which Separates, Departs L1 and Lands on Lunar Surface
5. Lander Ascent Vehicle Returns to L1
6. Crew Transfers to CEV and Departs L1
7. Earth Arrival (Into Earth Orbit or Direct Earth Entry)

L1 Abort options

a. Launch Abort from KSC
b. LEO Deorbit
c. Powered L1 Transfer Abort, Early Return to Earth
d. L1 Swingby, Return to Earth
e. Powered Lunar Tranfer Abort, Return to L1
f. Lander Swingby of Moon, Early Return to L1
g. Lander Ascent Vehicle Early Return to L1
h. Crew Transfers to CEV and Departs L1, Shorten Transfer Time
i. Earth Arrival (Into Earth Orbit or Direct Earth Entry), Ballistic Reentry to Unplanned Water or Land Landing Site

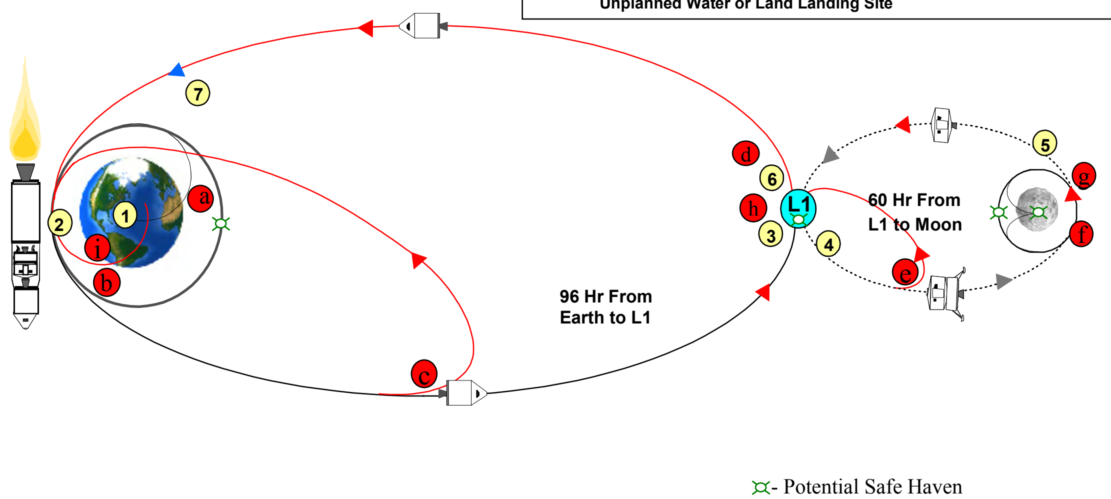


Figure 10.4.3-1: Trade Reference Mission Abort Options

10.4.3 Recommended Subsystem Design Approach for LDRM-2

LDRM–2 aborts will be developed and assessed for each mission phase from low Earth orbit to the surface of the Moon and the return to Earth's surface. Earth to orbit ascent aborts are out of the scope of this particular study. Possible abort and safe haven modes will be developed and assessed against the overall nominal mission requirements. Each mission phase may contain one or more critical events as identified in the TRM critical events table. The aborts selected for this L1 TRM only address those aborts occurring after CEV launch which result from an inability to complete a critical event required by the LDRM. Other system failures or problems with the crew may lead to a decision to abort the mission but those aborts can be readily accomplished by moving forward into the next mission phase or bypassing certain mission phases when necessary and completing a safe return to Earth transfer. The following nominal mission flight regimes have been identified along with the critical events previously identified in Section 10.3 (Safety and Mission Success section). Abort options are then described for each flight regime of the LDRM.

1. Launch and Ascent to Low Earth Orbit (LEO)

This mission phase begins with the launch from Earth surface and ends after the vehicle is established in the desired LEO.

a. Booster or major CEV system failure

i. CEV emergency separates and returns to Earth

During the CEV launch and ascent to LEO should the Human-Rated Launch Vehicle (HRLV) or the CEV suffer catastrophic failure the CEV can initiate the Launch Abort System, triggering an emergency separation from the HRLV and return to Earth using the CEV descent and touchdown systems.

2. LEO Orbit and Rendezvous Operations

This mission phase begins after the vehicle is in LEO and ends after the completion of any LEO rendezvous and mating of the Earth Departure Stage and the CEV or Lander.

a. CEV systems failure or failure to mate to Earth Departure Stage (EDS)

i. CEV de-orbit and return to Earth

Once the CEV has reached LEO, should the CEV suffer a significant system failure prior to initiating the EDS L1 transfer burn (TRM-27) the CEV must perform a standard de-orbit maneuver, reenter and touchdown on land or water. If the abort takes place after the CEV mates to the EDS (TRM-26) the CEV must separate from the EDS prior to re-entry. If CEV propulsion system failures preclude performing a de-orbit maneuver, the EDS could be used for that maneuver. Otherwise the CEV is stranded in LEO and an Earth based CEV rescue mission is required to prevent a loss of crew (LOC) event from occurring. The CEV would need the appropriate resources to provide this safe haven for the crew until that rescue mission is performed (x-weeks).

3. LEO to L1 Transfer

This phase begins at the start of the (L1) transfer burn and ends just before the start of the L1 arrival burn.

a. Early EDS shutdown and high elliptical orbit

i. CEV maneuver to desired orbit

ii. De-orbit burn and re-entry to touchdown

Should the EDS fail to fully complete the L1 departure burn the CEV can separate, perform any required transfer orbit adjustments within the limits of available CEV propulsion constraints, establish a return to Earth trajectory and perform a de-orbit and re-entry to touchdown. After completion of the L1 transfer burn the CEV can also abort by eliminating the L1 arrival burn and returning to Earth on the elliptical transfer orbit. The CEV can adjust this orbit within CEV propulsion constraints to ensure a safe Earth re-entry and touchdown.

4. L1 Operations

This phase begins at the start of the L1 arrival burn and includes all Lander/CEV rendezvous and mating operations. This phase ends after Lander/CEV separation just prior to the Lander departing for the lunar surface.

a. No CEV L1 arrival burn

i. CEV swing-by at L1 and return to Earth

If the CEV can not perform the L1 arrival burn the CEV can abort by continuing on the current elliptical transfer orbit and performing any maneuvers necessary to establish a safe return to Earth trajectory for a direct reentry or aerobraking pass.

b. L1 Rendezvous and mating with the Lander

i. Either CEV or Lander can be active vehicle for mating operations

If the CEV cannot perform the L1 rendezvous and mating with the Lander, the mission may be continued if the Lander is designed to perform the active rendezvous and mating. If the decision is made to abort the mission the CEV can abort back to Earth by performing a nominal L1 departure burn and establish an L1 to Earth transfer trajectory.

c. Crew transfer failure

i. CEV return to Earth

If there is a failure to transfer the crew from the CEV to the Lander the crew can abort the mission by separating the CEV from the Lander and performing the nominal L1 to Earth transfer burn. Another abort option is to use the Lander to perform the L1 to Earth transfer burn and then separate the CEV from the Lander once a safe return to Earth trajectory has been established.

5. L1 to Low Lunar Orbit (LLO)

This phase begins at the L1 to LLO departure burn and ends just prior to the lunar orbit insertion burn.

a. No Lander L1 departure burn

i. Remate to CEV and CEV return to Earth

Once the crew has transferred to the Lander and separated from the CEV, if the Lander is unable to perform the L1 to LLO transfer burn then the Lander must remate with the CEV to allow for crew transfer back to the

CEV and subsequent return to Earth. Either the Lander or the CEV should be capable of performing the required rendezvous and mating maneuvers. If Lander/CEV mating is impossible there must be a method to perform an extra-vehicular activity (EVA) transfer of the crew from the Lander to the CEV.

b. Bad L1 departure burn

i. Lander uses Descent or Ascent Stage to return to L1

If there is a failure of the Lander to successfully complete the L1 to LLO transfer burn then the Lander Ascent Stage can be used to either return to L1 or place the Lander in a Lunar swingby trajectory that will return to L1 with sufficient propellant to perform the L1 arrival burn. After returning to L1 the Lander and CEV will mate, transfer the crew and the CEV will return to Earth.

ii. CEV Rescue

If the Lander can successfully approach L1, the CEV may be designed to have the capability to perform a limited rescue rendezvous and mate with the Lander to allow for crew transfer to the CEV.

6. LLO Insertion

This phase begins at the start of the LLO insertion burn and continues until the Lander begins its descent to the lunar surface.

a. No LLO insertion burn

i. Lander swingby and return to L1

If the Lander Descent Stage is not successful in completing the LLO insertion burn then the Lander must be capable of performing a lunar swingby maneuver and returning to L1. This can be accomplished by the Lander Ascent Stage.

b. Partial insertion burn

i. Ascent Stage delta-V maneuver and return to L1

If the Lander Descent Stage partially completes the LLO insertion burn the Lander Ascent Stage could be used to complete the insertion and then perform the LLO to L1 transfer burn if within the Lander propellant budget. Otherwise the Lander Ascent Stage must be used to adjust the lunar trajectory to perform a lunar swingby and return to L1 for rendezvous with the CEV.

7. LLO to Powered Descent Initiation

This phase begins at the start of the lunar descent burn and ends just prior to the Powered Descent Initiation burn.

a. No de-orbit burn

i. Lander return to L1

During the Lander de-orbit and descent to the Lunar surface if any non-propulsion related failure causes an abort the Lander Descent Stage will be used to return to LLO where the Lander Ascent Stage can perform the LLO to L1 transfer burn. If the Lander cannot complete the de-orbit to the powered descent initiation point then the Lander can abort using the remainder of the Descent Stage or the Ascent Stage to initiate the LLO to L1 transfer and return the crew to the CEV.

b. Partial de-orbit burn

i. Lander ascent return to LLO or L1

If a partial de-orbit burn is performed, the Lander Ascent Stage to return the LLO, initiate the LLO to L1 transfer to return the crew to the CEV.

8. Powered Descent Initiation to Lunar Surface

This phase begins at the start of the powered descent initiation burn and ends at lunar surface touchdown.

a. No powered descent

i. Lander Ascent Stage return to LLO

If the powered descent maneuver is not started then the Lander can use either the Descent Stage or the Ascent Stage to return to LLO and initiate a LLO to L1 departure burn.

b. Descent abort

i. Lander Ascent Stage return to LLO

If the need to abort the landing occurs late in the powered descent phase, the Lander Ascent Stage will be used to return to LLO and perform the LLO to L1 transfer burn to return the crew to the CEV.

9. Lunar Surface Operations

This phase begins just after touchdown, encompasses all lunar surface activities and ends just prior to lunar ascent.

a. EVA suit failures

i. Emergency ingress from EVA

During Lunar surface operations the crew must have the ability to rapidly ingress the Lander from a lunar surface EVA to protect against EVA suit failures. This requires the ability to rapidly transit from any EVA site back to the Lander and reenter the Lander pressurized volume without extensive stays in any airlock. For long distance EVA sites a pressurized rover may be required to provide a habitable environment in the event of EVA suit failure.

10. Lunar Ascent to LLO

This phase begins at lunar ascent initiation and ends when the Lander has achieved the desired LLO.

a. No lunar liftoff

i. Long duration safe haven until Earth based rescue mission arrives (TBD weeks) or LOC

ii. Predeploy extended stay safe haven resources near landing site

If the Lander Ascent Stage fails to ignite then the crew is stranded on the lunar surface and must wait for an Earth based rescue mission. To prevent a LOC event requires the ability for a long duration (TBD weeks) safe haven on the lunar surface, which will require predeployment of safe haven resources near the landing site.

b. Failure to reach LLO

i. No failure allowed; Lander must reach safe lunar orbit or LOC, physical and functional redundancy is required

After liftoff from the lunar surface, the Lander must reach a safe LLO or a LOC event will occur. Physical or functional redundancy in the Lander Ascent Stage is required to ensure that the lunar ascent to LLO is successfully completed.

11. LLO to L1 Transfer

This phase begins when the Lander is in LLO, encompasses the LLO to L1 transfer burn and ends just prior to the L1 arrival burn.

a. No LLO departure burn

i. LLO safe haven operations until Earth based rescue

ii. CEV rescue (depart L1 for LLO)

Upon reaching LLO if the Lander is unable to perform the LLO to L1 transfer burn then the crew is stranded in LLO until an Earth based rescue mission arrives or a LOC event occurs. The Lander Ascent Stage will require enough resources to accommodate the long duration safe haven (TBD weeks) for the crew. Another option would be to use the CEV to leave L1, enter LLO, rendezvous and mate with the Lander to rescue the crew if there are enough propellant reserves onboard the CEV.

b. No L1 arrival burn

i. CEV capture and mate

After departing LLO for L1 the Lander must be able to perform the L1 arrival burn to set up for the rendezvous and mating with the CEV. If the Lander does not successfully complete the L1 arrival burn there will be a LOC unless the CEV is designed to allow it to rendezvous and mate with the Lander as it passes through L1 vicinity.

12. L1 Operations

This phase begins with the Lander L1 arrival burn and encompasses all Lander/CEV rendezvous and mating operations and crew transfer operations. This phase ends after CEV/Lander separation just prior to the CEV L1 departure burn.

a. Failure to mate Lander and CEV

i. EVA crew transfer to CEV

ii. CEV return to Earth

If the Lander and CEV are unable to mate and transfer the crew to the CEV then there must be a way to allow the crew to EVA translate to the CEV and reenter the CEV for the return to Earth. Otherwise a LOC will occur.

13. L1 to Earth Transfer

This phase begins with the CEV L1 departure burn and ends just prior to Earth atmospheric re-entry.

a. No L1 departure burn prior to CEV/Lander separation

i. Cross strap CEV propellant tanks to Lander engine and complete burn with Lander

If prior to CEV/Lander separation information is available that the CEV will be unable to perform the L1 to Earth transfer burn then the Lander and CEV could be designed to allow for CEV propellant to be used by the Lander engines to complete the maneuver. This would require the use of common engines and propellant between the CEV and Lander plus the addition of the appropriate propellant connection to allow CEV propellant to flow to the Lander propulsion system. The CEV must also carry enough propellant to accommodate the extra inert mass of the Lander Ascent Stage. The CEV and Lander can separate after the CEV is in a safe return to Earth trajectory.

b. No L1 burn post CEV/Lander separation

i. Return to Lander and do propellant cross strap or safe haven until Earth based rescue

ii. Physical or functional redundancy to prevent no L1 burn or LOC

If the CEV fails to perform the L1 to Earth transfer burn the CEV could return to the Lander, remate and use the Lander engines if the necessary propulsion system interconnects are available.

or

iii. CEV safe haven until Earth based rescue

Otherwise the CEV is stranded at L1 and must await an Earth based rescue mission or an LOC will occur. The CEV will require the necessary safe haven resources to provide the safe haven time for TBD weeks.

c. Post L1 departure burn

i. CEV burns to adjust trip time and touchdown site location

After completion of the L1 to Earth transfer burn the CEV is on a safe return to Earth trajectory. Minor adjustments in the trip time may be available to adjust the CEV landing site.

14. Earth Re-entry to Touchdown

This phase begins with the direct re-entry into Earth atmosphere and ends with CEV touchdown on the Earth surface.

a. Re-entry flight control failures

i. Ballistic re-entry (no lift vector control)

The only abort addressed for the Earth re-entry to touchdown phase is the possibility of performing a passive (zero lift) re-entry. This abort will be possible only if the Earth return trajectory allows the re-entry G levels to remain below the human tolerance limits during the passive re-entry. Otherwise a lift vector controlled trajectory would be required to lower the g loads in the crew and if the CEV lost all control during re-entry a LOC event might occur if the human limits are exceeded.

b. Entry targeting failures

i. Water or land touchdown

CEV equipped with appropriate crew survival and search and rescue gear for touchdown site

The CEV will be designed to support either land or water touchdown allowing for entry targeting failures to force the CEV to miss the desired landing site. The LDRM architecture is using 3 hr as the time required to find and recover the crew from the CEV after touchdown.

or

15. Earth Aerocapture to LEO

This phase begins with CEV re-entry into Earth atmosphere, encompasses CEV aerobraking into the desired LEO operations and ends just prior to the CEV final de-orbit burn.

a. Failure to aerocapture and circular burn (elliptical orbit)

i. Delta-V maneuver to appropriate orbit with physical or functional redundancy

ii. Safe haven until Earth based rescue or natural orbital decay

iii. Passive control/ballistic re-entry

For missions designed to use aerobraking to LEO instead of a direct entry a failure to successfully complete the aerocapture leads to the following aborts. If the aerocapture fails to produce the desired LEO, CEV propulsion can be used to provide the desired orbit. In addition, the CEV may be designed to allow for a passively controlled ballistic re-entry using the aerobrake heat shield in addition to the CEV. Once in LEO the CEV could provide a safe haven for TBD weeks until an Earth based rescue could be performed.

b. Failure to aerocapture (escape trajectory)

i. LOC

If the failure to aerocapture results in an atmospheric skip out and Earth escape trajectory there is a LOC. Physical or functional redundancy must be provided to ensure that the CEV is safely captured into LEO.

16. De-orbit and Re-entry to Touchdown

This phase begins with the CEV de-orbit burn and ends with CEV touchdown on Earth's surface.

a. No de-orbit

i. Safe haven until rescue or orbital decay or LOC

After reaching a safe LEO if the CEV fails to perform the de-orbit maneuver there is a LOC unless the CEV can provide a safe haven until an Earth based rescue can be performed.

b. Re-entry flight control failures

i. Passive re-entry (no lift vector control)

After a successful de-orbit burn the CEV will have the capability to perform a ballistic re-entry in the event a nominal re-entry is not possible.

c. Entry targeting failures

i. Water or land touchdown

CEV equipped with appropriate crew survival and search and rescue gear for touchdown site

The CEV will be designed to support either land or water touchdown allowing for entry targeting failures to force the CEV to miss the desired touchdown site. The LDRM architecture is using 3 hr as the time required to find and recover the crew from the CEV after touchdown.

10.4.4 Return Time to Earth and Return Time to CEV

Figures 10.4.4-1 and 10.4.4-2 depict the abort timelines for the L1 TRM of Lunar Design Reference Mission–2 (LDRM-2). The chart shows the maximum time required to return the crew to Earth or to the CEV as a function of when the abort is initiated during the nominal mission elapsed time. After launch of the Crew Exploration Vehicle (CEV) and while still in Low Earth Orbit (LEO) for the first 3-4 days the return to Earth time remains constant at close to 3.5 hr. This is the nominal time required to execute the de-orbit maneuver, re-enter the atmosphere, touchdown and be recovered by ground search and rescue forces. Upon completion of the Earth departure burn, the CEV is placed into a 94 hr transfer orbit to L1. Assuming no propulsive maneuvers to modify the transfer orbit period to reduce the transfer time, the CEV will take about twice the nominal transfer time to return to Earth. This is about 191.5 hr for the trade reference mission. As the CEV progresses toward L1 or the Moon the return to Earth time is correspondingly reduced until arrival at L1 the return to Earth time becomes the same as the initial transfer time plus the additional 3.5 hr of Earth recovery time. The L1 architecture requires the largest return to Earth abort time immediately after the Lunar Lander departs L1 for the Moon in a 60 hr transfer leg. At this point, the crew is some 247.5 hr away from Earth and 150 hr from the CEV. This time is comprised of a 120 hr transfer trajectory to and from the moon to L1, 30 hr of rendezvous, mating and crew transfer operations at L1, a 94 hr transfer back to Earth plus 3.5 hr recovery operations. For all abort timelines, it may be possible to use propulsion systems to reduce return times. The timelines shown in Figures 10.4.4-1 and 10.4.4-2 represent the worst-case abort scenarios, where no off-nominal maneuvers are executed to minimize return time.

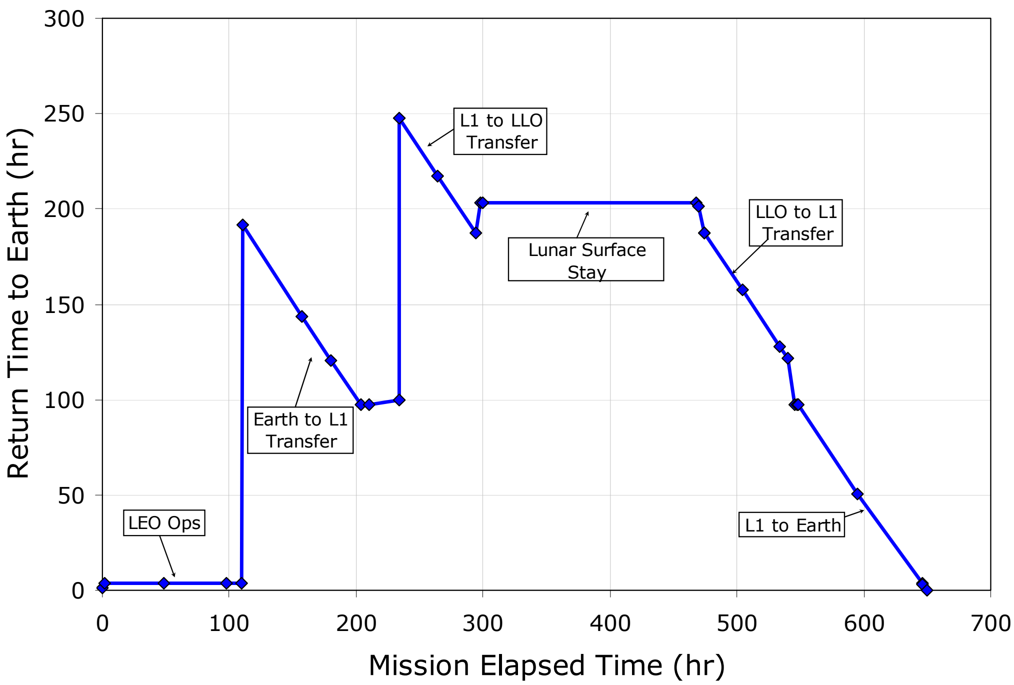


Figure 10.4.4-1: Maximum Return Time to Earth for TRM

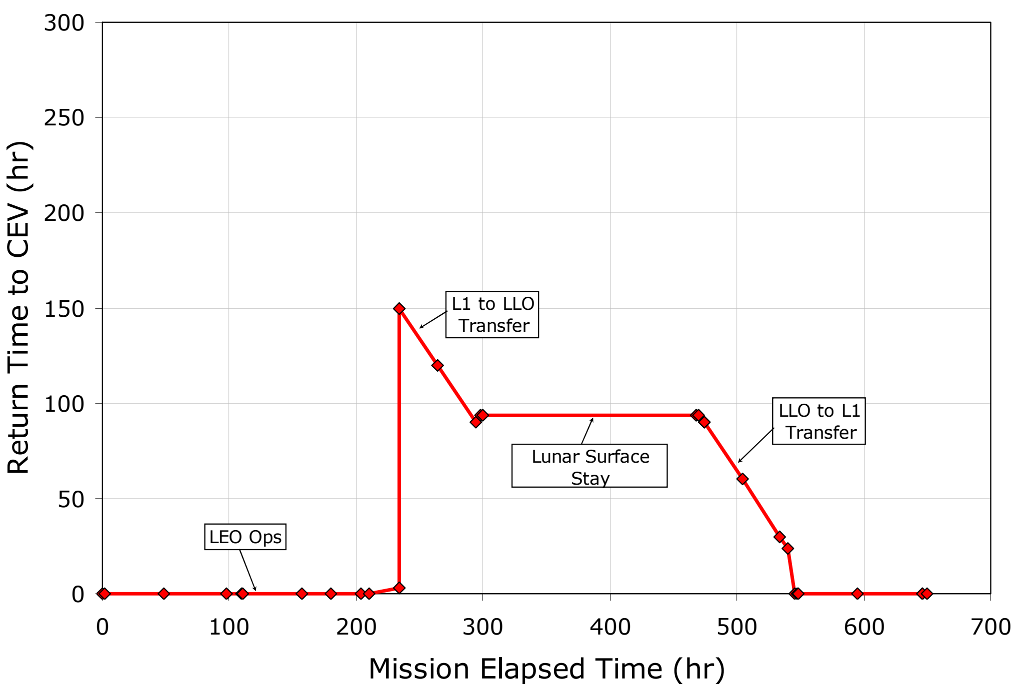


Figure 10.4.4-2: Maximum Return Time to CEV for TRM

10.5 Element Overview & Mass Properties

This section describes the key assumptions and subsystem selections used for sizing the TRM elements and the resulting vehicle mass properties. This section also provides a description and images of any CAD modeling done for the elements. For the trade reference mission option A2, the CEV, Lunar Lander, Earth Departure Stages, and Kick Stage mass properties were estimated using the Envision parametric sizing tool. The inputs for the sizing tool were selected to reflect the technology selections as closely as possible within the existing choices in the tool.

Technology selections for the TRM vehicles are generally not the results of detailed trade studies conducted for this architecture. Rather, they are derived from recommendations from subsystem experts and previous higher-fidelity design efforts for similar lunar exploration concepts. Instead of determining optimal subsystem designs for the TRM vehicles, the primary purpose of the LDRM-2 study was to select an initial architecture and set of vehicle configurations against which architecture trades could be performed to determine relative merits of different architecture options. The technology reports in section 20.0 of this document examine in greater detail the technology options for each subsystem and describe the technologies chosen for the trade mission.

10.5.1 Crew Exploration Vehicle

The CEV design was driven by several major assumptions. It must support a four-person crew for ~15 days (transit to and from L1), pressurized crew transfer to the Lunar Lander, ~12 days of loitering at L1 with no crew onboard, a possible contingency EVA without an airlock, and a direct entry at Earth with a nominal water landing (contingency land landing). In addition, the CEV must accommodate 100 kg of return cargo. To determine average vehicle power consumption for the CEV, an assumption was made of 6 kW for crew-occupied portions of the mission. A definitive mission profile and list of subsystem components was not immediately available during the study to create a CEV power profile and therefore derive a "bottoms-up" power estimate. Instead, the average power consumption for the X-38 flight test vehicle was used as a starting point for the CEV, with some provisions made for subsystem differences between the two vehicles. This approach led to the 6 kW average power estimate. Due to the significant differences between the X-38 and CEV mission profiles, though, more refined design efforts may show that the CEV average power requirement may be reduced below 6 kW by turning off non-critical systems during the long coasting phases of the mission. For unoccupied portions of the mission (loiter time at Lunar L1), it was assumed that the CEV average power could be reduced to 50% of the occupied power level. Again, future refinement efforts are needed to determine which systems can be turned off when the crew is not onboard and which systems need to remain powered for operational and crew safety needs. Using the mission profile and 6 kW/3 kW average power assumptions, an approximate total CEV energy requirement of 15 days*6 kW + 12 days*3 kW = 3,024 kW-hr was estimated.

The baseline CEV concept was modeled in the Pro/Engineer CAD system to demonstrate one feasible vehicle configuration. For the LDRM-2 study, a CEV consisting of a separate capsule-shaped Crew Module and cylindrical Service Module (analogous to the Apollo CSM) was selected as a reference configuration, though other concepts in which the functionality of those two elements are combined into a single vehicle were considered equally feasible. The outer mold line capsule shape of the Crew Module was chosen based on past experience with Apollo and work performed on the Orbital Space Plane project. Equipment packaging constraints and crew accommodations requirements led to the selection of a 5 m base diameter for the CM. The CM base shape was scaled from the Apollo CM, and the sidewall angle of 30^o was chosen to be similar to the Apollo CM sidewall angle of 32.5^o. The original Apollo CM sidewall angle was chosen to allow the vehicle to fly at an angle of attack such that a hypersonic L/D of 0.5 during Earth entry could be achieved, thereby increasing the vehicle's theoretical crossrange. Actual Apollo missions only flew at a hypersonic L/D of 0.3, though, meaning that a sidewall angle less than 32.5^o could have been selected. Future detailed vehicle design for a CEV concept of this type may show that the angle can be reduced which would result in increased internal volume. The CEV Service Module was configured as a cylindrical stage with a diameter of 5 m to match the CM base and a length of 3 m to accommodate initial estimates of the required radiator surface area and surface area for access panels and thrusters. The CEV shape and size chosen for this study may change in future studies based on differing mission architecture requirements and more detailed analyses (i.e. re-entry heating profiles, radiator sizing, etc.).

Representative subsystem components (mostly X-38 vintage) were packaged into the vehicle CAD model to show the feasibility of fitting the equipment assumed in the sizing into the available volume. However, the component size, shape, and packaging were not optimized for this particular mission.

There were several key considerations when laying out the CM equipment within the 22 m^3 of pressurized volume. Much of the equipment was placed low in the vehicle to help keep the vehicle center of gravity low. This provided the added benefit of maximizing crew habitable volume, which was estimated at 12 m^3 (for comparison, the Apollo CM provided 10.4 m^3 of total pressurized volume for its three passengers). The required open space for launch & entry suit donning and doffing was estimated to also fit within the habitable volume. The crew seats were arranged side by side to simplify secondary structure design with the heads in the "up" position during reentry. The seats shown will accommodate up to a 95^{th} percentile American male and were assumed to be removable for stowage while in space to increase the habitable volume. A keep-out zone corresponding to Apollo Block II data was created around the seats for landing seat stroke. Eighteen and a half inches were left at the crewmembers' feet, 5" at the head, 5.5" on each side, and 16.5" below the seats. A side hatch was placed on the vehicle by the crewmembers' heads for ease of ingress/egress of a suited crewmember. Additional overriding considerations were: logical component placement for access during flight, ease of wire and plumbing routing, and simplicity of secondary structure design.

Outside the pressurized volume, six high-pressure RCS Tridyne ($N_2/O_2/H_2$) propellant tanks and two gaseous nitrogen tanks are mounted near the base of the CM. Twelve 50 lbf RCS thrusters are distributed around the vehicle to control attitude during reentry. To provide radiation protection for the crew during transits to and from Lunar L1, a 5.5 cm thick blanket of polyethylene material is distributed around the vehicle sidewall and top of the main habitable volume. Near the top of the vehicle are found three round main parachutes, two drogue parachutes, and pilot parachutes to extract the main chutes. A LIDS-type docking adapter provides for low impact, fully androgynous docking to the Earth Departure Stage, and a tunnel from the main pressurized volume to the LIDS hatch allows for pressurized crew transfer to and from the Lunar Lander.

The SM layout was driven by the dimensions of its largest components, the propellant tanks. These tanks actually contain a combination of the OMS and RCS propellant (liquid oxygen/liquid methane), breathable oxygen for the crew, and oxygen for the fuel cells. Initially, the oxygen and methane was to be stored in a single common bulkhead tank; however, the resulting SM packaging was inefficient. The single tank was subsequently divided into 6 equal-volume common bulkhead tanks and packaged into the vehicle. The SM also contains the fuel cells which produce the power for the CEV. The water produced by the fuel cells is used to supply potable water for the crew and for heat rejection via a water evaporator following SM disposal. Thirty-two square meters of radiator panels were divided into four equal sections and placed between the four RCS thruster pods. Each of the four thruster pods includes six 50 lbf thrusters to provide vehicle attitude control. Finally, two 7,500 lbf OMS engines are mounted to the aft section of the Service Module to perform major orbital maneuvers such as libration point arrival and departure.

Table 10.5.1-1 describes the subsystem components selected for the TRM CEV concept.

Component Name	Qty	Description/Function Performed
Avionics		
Data Interface Units	8	Collect and transmit data
Flight Computer	4	Flight critical computers for implementing dual fault-op tolerant processing
GPS	4	Space-integrated GPS / INS computers perform vehicle guidance and navigation processing
GPS Combiner Unit	1	Combine GPS signals
LADAR	2	Laser detection and ranging for automated rendezvous & docking
Multi-Function Display Panel	2	Multifunction LCD displays to provide crew interface for system status and command input
Operations Data Recorder	1	Record vehicle data for post-mission processing
Rotational/Translational Hand Controller	2	Provide manual vehicle flight control
S-Band Comm Transponder / Power Amplifier / Switching Unit	2	Provide vehicle-to-vehicle and vehicle-to-ground communication
S-Band Dual Beam Antenna	4	Provide communication
Star Tracker	2	Provides on-orbit vehicle attitude determination data to augment GPS / INS
Switch Panel	2	Control switch panels to provide functions not controlled by multi-function displays
UHF Antenna	4	Provide communication
UHF Comm Transceiver/Switching Unit	2	Provide communication
Video System	1	Video cameras & video processing equipment
Crew Accommodations		
Clothing	0.46 kg/p/d	No clothes washing assumed
Commode	1	Waste collection and disposal
Cooking & Eating Supplies	0.5 kg/p	Prepare and consume food
Crew	4	Mass of a 95th percentile American male
Crew Health Care Kit	1	Medicine, basic medical equipment
Emergency Breathing Apparatus	4	Provide emergency oxygen for crew
Emergency Egress Kit	4	Launch pad egress
Food	2.3 kg/p/d	A combination of shelf-stable and dehydrated food.
Food Warmer	1	Prepare food
Hand Tools	1	Tools for in-flight maintenance
High-g Strokeable Couch Seat	4	For high-g's experienced on launch, entry, and landing. Includes 18" vertical stroke.
Personal Hygiene Kit	4	Washcloth, toothbrush, razor, etc.
Photography/Mission Documentation Kit	1	For mission documentation – digital still and video cameras, supplies, etc.
Recreational Equipment	5 kg/p	Misc. personal effects

Component Name	Qty	Description/Function Performed
Sighting Aid Kit	1	COAS + accessories, binoculars, LIDAR sighting aid, spot light, search lights, etc.
Sleep Accommodations	9 kg/p	Similar to Shuttle accommodations
Stowage	As needed	Soft stowage assumed
Vacuum	1	Housekeeping
Water Spigot	1	Provide potable water
Environment		
Ambient Temperature Catalytic Oxidizer w/ Charcoal Trace Contaminant Control	1	Remove trace gas contaminants from cabin atmosphere
Atmosphere Composition Monitoring	1	Monitor oxygen & carbon dioxide partial pressure
Cabin Fans	As needed	Cabin thermal conditioning
Combined CO2/Moisture Removal System	2	Remove carbon dioxide and moisture from cabin atmosphere. Each system is internally redundant.
EVA Tools	As needed	Handholds, tethers, etc.
EVA Umbilicals	4	For emergency full cabin depressurizations
Fire Detection and Suppression	1	Smoke detectors, fixed and portable halon extinguisher equipment
Flexible Body-Mounted Radiator	4	Radiate vehicle waste heat to deep space. 10 mil Ag-Teflon radiator coating (α/ε = 0.142)
Heat Collection Fluid Loop	2	Lines, valves, pumps, cold plates, heat exchangers. Single-phase 60% $C_3H_8O_2$ – 40% H_2O fluid. 9 kW total heat load.
Launch and Entry Suit	4	Pressure suit for launch and entry. Assumed to be functionally similar to Shuttle launch and entry suits.
Nitrogen Storage and Distribution System	1	Gaseous storage tanks. 70% N_2 – 30% O_2 atmosphere @ 9.5 psia
Oxygen Storage and Distribution System	1	Breathable oxygen shared storage w/ CEV oxidizer & fuel cell reactant. 70% N_2 – 30% O_2 atmosphere @ 9.5 psia
Potable Water Storage	1	Potable water produced by fuel cells. 50 kg storage.
Wastewater Storage		Excess assumed to be periodically vented. 25 kg storage.
Water Evaporator		For heat rejection following SM disposal
Other		
Docking Window	1	Window in docking adapter hatch to aid rendezvous & docking operations
Drogue Parachutes	2	Circular parachutes to orient and slow the spacecraft.

Component Name	Qty	Description/Function Performed
Hatch & Hatch Window	1	Provides primary crew ingress/egress path while on Earth and secondary path for contingency EVA
Low Impact Docking System	1	Fully androgynous low impact docking system for pressurized mating to other architecture elements (e.g. Lunar Lander)
Main Parachutes	3	Circular ringsail parachutes to slow the spacecraft to touchdown speed of 28 ft/s. 2 chutes required for safe landing.
Pilot Parachute System	1	Deploy the main parachutes
Pyros & Release Mechanisms	As needed	Provide for mechanical separation of vehicle components
Radiation Protection	As needed	5 gm/cm^2 of polyethylene radiation protection distributed around the outside of the CM pressure vessel
Water Flotation System	1	Inflatable airbags to right the spacecraft after landing
Passive Thermal Control		
Insulation Blankets	As needed	Vehicle passive thermal control. 2 kg/m^2 multi-layer insulation blankets
Lightweight Carbon-Based Charring Ablator	As needed	Thermal protection for areas of vehicle outer moldline w/ peak temperatures >2700°F
Reusable Surface Insulation	As needed	Thermal protection for areas of vehicle outer moldline w/ peak temperatures <2700°F
Power		
Electrical Power and Distribution Bus	3	28 Vdc bus. Includes remote power control units, wiring, inverters, wiring trays, etc.
Li-ion Primary Battery	4	Provides vehicle power following SM disposal. Total energy requirement = 28 kW-hr
PEM Fuel Cell	3	Provides vehicle power and potable water for all mission phases up to SM disposal. Peak power per FC stack = 6 kW. Total energy requirement = 3,012 kW-hr
Oxygen Reactant Accumulator Tank	3	Store 8 hr supply of O_2 reactant for fuel cell @ 1,000 psi. Reactant shared storage w/ CEV oxidizer. Graphite-epoxy overwrapped tanks w/ Inconel liner.
Hydrogen Reactant Tank	3	Store entire supply of supercritical H_2 reactant for fuel cell @ 500 psi. Graphite-epoxy overwrapped tanks w/ Inconel liner.
Propulsion		
CM RCS 50 lbf Thrusters	12	Provide attitude control during reentry. P_c = 125 psi, $\varepsilon = 25$, I_{sp} = 140 s.

Component Name	Qty	Description/Function Performed
CM RCS Tridyne Tanks	6	Store Tridyne propellant. Graphite-epoxy overwrapped tanks w/ Inconel liner @ 4,500 psi.
SM OMS 7,500 lbf Pressure-Fed Engines	2	Provide orbital maneuvering capability for libration point arrival and departure maneuvers. P_c = 175 psi, ε = 150, I_{sp} = 362 s, MR = 3.8:1.
SM OMS/RCS Fuel & Oxidizer Tanks	6	Al 7075 common-bulkhead tanks @ 275 psi for oxygen and methane storage. Methane OMS/RCS shared storage; Oxygen OMS/RCS/Fuel Cell/ECLSS O2 shared storage. Includes cryocoolers & MLI
SM OMS/RCS Helium Pressurization Tanks	2	Graphite-epoxy overwrapped tanks w/ Inconel liner @ 6,000 psi.
SM RCS 50 lbf Thrusters	24	Provide vehicle attitude control for all mission phases up to SM disposal. P_c = 125 psi, ε = 40, I_{sp} = 315 s, MR = 3.8:1.
Structure		
CM Outer Moldline Structure	1	Carbon fiber composite skin panels. 5 m base diameter Apollo capsule shape w/ 30° sidewall angle.
CM Pressure Vessel Structure	1	Al-Li 8090 primary/secondary structure. 22 m^3 internal pressurized volume
SM Structure	1	Al-Li 8090 primary/secondary structure. 5 m straight cylinder.

Table 10.5.1-1: CEV Subsystem Component Description

10.5.2 Lunar Lander

The role of the Lunar Lander (in conjunction with the Kick Stage) in the trade reference mission is to deliver four crewmembers from Lunar L1 to the lunar surface, allow the crew to conduct daily EVAs for up to 7 days while on the lunar surface, and return the crew to the CEV loitering at Lunar L1. The Lunar Lander is configured to be integrated with the Kick Stage on the ground and launched as a single combined element, and it includes the necessary avionics and mechanisms to mate with a pre-deployed Earth Departure Stage in low Earth orbit.

The Lunar Lander was assumed to be a one-and-a-half-stage lander, meaning that its descent stage structure and propellant tanks for lunar descent remain on the surface while its main engines are reused for lunar ascent. The Apollo LM, on the other hand, was considered to be a two-stage lander because it had two fully independent propulsion stages operating in serial, one stage for lunar descent and one for lunar ascent. Lander stage configuration will have a major impact on vehicle packaging, overall reliability, abort scenarios, mass, and other key considerations. Although the one-and-a-half stage configuration promises some mass savings advantages by eliminating the mass of separate descent stage engines, this concept's effect on Lander mission success and crew survival should be examined closely in the future. For propulsion, both the Descent and Ascent Stages use space-storable oxygen and methane propellants, stored in 275 psi common bulkhead tanks, and burned through four 7,500 lbf pressure-fed main engines with 20% throttling capability. This propellant combination offers both relatively high packaging efficiency and performance, and enables shared fluid storage between OMS, RCS, power generation, and ECLSS. Earth-storable propellant options such as NTO/MMH were initially considered but eventually discarded as they offer higher bulk densities than oxygen and methane at the price of lower specific impulse. Earth-storable propellants are also generally highly-toxic and require active heating to prevent freezing. A hydrogen-based propulsion system offers significantly higher performance than methane but it was not considered further because of its very low boiling point and liquid density. Packaging efficiency is particularly important for landers, as they are generally have the highest diameters of all elements in a mission and have a requirement for safe crew transfer from the egress hatch down to the lunar surface and back. Propellant tanks can be made with narrow diameters to minimize overall vehicle size and required launch vehicle shroud diameter, but with traditional lander configurations, this requires high crew heights above the lunar surface which may jeopardize crew safety. Likewise, minimizing lander height generally comes at the cost of wider vehicle diameters and launch shrouds. For these reasons and others, the relatively high performance and bulk density combination of oxygen and methane was chosen as an initial reference against which future trades could be evaluated.

A four-engine approach was chosen for the Lunar Lander to support the required thrust-to-weight requirements (see Table 10.5.2-1) and provide an engine-out capability during ascent and descent. In the case of an engine failure, the engine opposite the failed engine may be shut off, allowing the two other main engines to thrust through the CG at full throttle without large engine gimbaling requirements. Engines on the Lander are assumed to be identical to the CEV's engines, though it is possible that a non-throttling version be initially developed for use on the CEV. The maximum thrust requirement for the Lunar Lander is dictated by the start of the powered descent phase. To perform a fuel-optimal descent trajectory, an initial thrust-to-weight (T/W) of at least 0.33 Earth g's is desired. Gravity loss costs increase with lower initial T/W ratios. With the current TRM Lander configuration, the four engines are required to throttle to 103% of 7,500 lbf to achieve the desired thrust at powered descent initiation. Minimum Lander thrust is driven by the need to hover and slowly accelerate downwards just prior to touchdown on the lunar surface. The required thrust in this phase is assumed to be 80% of the Lander's landed weight in the lunar gravity field. Achieving this level requires that all four engines throttle to 24% of their maximum capability. This appears to be well within the range of throttling capability of pressure-fed main engines. The Apollo LM descent engine was capable of throttling to 13% of its maximum 9,900 lbf thrust capability.

Phase Name	Desired Total Thrust	Lander T/W (Earth g's)	Required Engine Throttling
Powered Descent Initiation	30,939 lbf	0.33	103%
Hover & Lunar Touchdown	7,306 lbf	0.13	24%
Ascent Ignition	27,437 lbf	0.625	92%

Table 10.5.2-1: Lunar Lander Engine Thrust Requirements

Much like powered descent initiation, required thrust for lunar ascent is also driven by the need to minimize gravity losses. According to the curve in Figure 10.5.2-1, an initial T/W of 0.625 at ascent ignition enables the most fuel-optimal lunar ascent trajectory. With the current four-engine, 30,000 lbf capability, each engine would be throttled to 92% to fly that T/W profile. Assuming one engine is lost at ignition and the remaining engines (at 100% throttle) can gimbal to thrust through the vehicle center of gravity, the vehicle still has a T/W of 0.51, which only increases ascent delta-V by 5 m/s above the minimum. If two engines are lost, though, vehicle T/W decreases to 0.34, resulting in 57 m/s of additional gravity loss delta-V above the minimum.

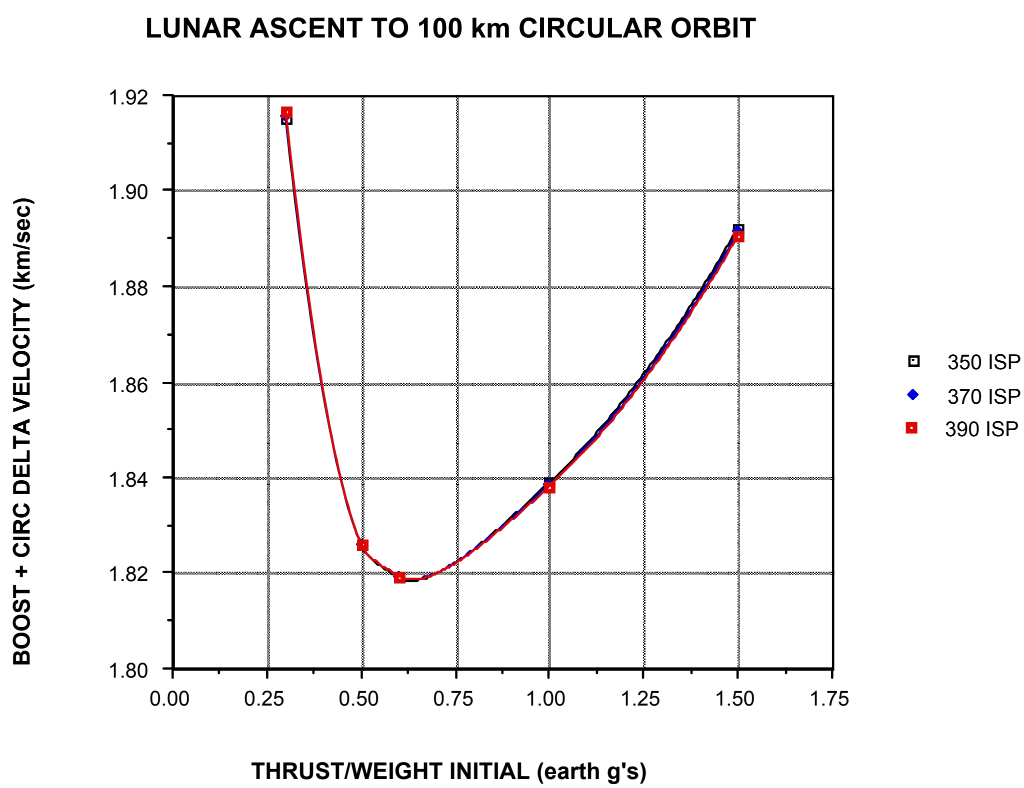


Figure 10.5.2-1: Ascent Delta-V as a Function of Initial Thrust-to-Weight

The Lunar Lander's liquid oxygen/liquid methane RCS is integrated with the vehicle's OMS as was done with the TRM CEV. Twelve 50 lbf engines are mounted on the Lander Ascent Stage to provide rendezvous and mating capability with the CEV at the end of the mission, and sixteen 50 lbf engines are included on the Descent Stage for mating with the Earth Departure Stage in LEO and vehicle attitude control during powered descent. Each engine is assumed to be identical to the RCS engines on the CEV Service Module. RCS functionality was split between the Ascent and Descent Stages to minimize overall vehicle mass, though more study is needed to understand the merits of including a single reaction control systems on the Ascent Stage only.

For EVA capability on the lunar surface, an externally-attached deployable airlock is assumed to be included on the Lander Descent Stage. An airlock allows EVAs to be conducted by two or three crewmembers while the remaining person(s) remain in the vehicle for coordinating the EVA and monitoring vehicle status. A full cabin depressurization strategy where the entire cabin atmosphere gas is vented down to near-vacuum conditions, on the other hand, would require that all crewmembers don their pressure suits to conduct an EVA. An airlock also provides a separate staging area for doffing EVA suits. In the Apollo missions, the accumulation of lunar dust within the LM Ascent Stage crew cabin was an important consideration. With the inclusion of an airlock, it may be easier to contain the bulk of the dust within that volume and prevent dust from being transferred to the Ascent Stage volume. However, further study will definitely be required in this area to better understand the trade of airlocks vs. full cabin depressurization for various crew sizes, surface durations, and EVA strategies.

The internal pressurized volume of the Lander Ascent Stage was sized at 22 m^3 based on initial estimates of hardware volumes and desired habitable volume/open floor space. There is currently no NASA-accepted guidance on floor area requirements for planetary surface elements. However, using data on EVA suit donning/doffing and examining the types of activities the crew will participate in while in the Lunar Lander, open floor dimensions of 2 meters x 2.8 meters were used and a cabin height of 2.28 meters was used.

As the avionics, life support, and environmental control requirements for the Lunar Lander were considered to be similar to those of the CEV, the complement of subsystem components for the two vehicles (Table 10.5.2-2 lists components assumed for the Lander) were assumed to share as much commonality as practical. Therefore, the same average power requirement used for the CEV, 6 kW, was used for initial sizing of the Lunar Lander. Of course, a power profile analysis will be needed to determine the vehicle's true average power requirement. A PEM fuel cell system identical to the CEV's was used to generate the ~1,900 kW-hr of total energy needed to power the Lander from initial crew ingress to disposal. Fuel cells, as opposed to photovoltaic systems, have the advantage of being able to operate independent of sun lighting conditions which is particularly important for the long lunar night, and are able to produce potable water for the crew. Fuel cell oxygen reactant has been assumed to be stored with the Ascent Stage OMS & RCS oxidizer, and hydrogen reactant is stored in separate power reactant storage and distribution (PRSD) tanks. For mission phases prior to crew ingress, such as the several week-long loiter periods at Lunar L1 and in LEO, a separate, non-consumable based solar array and battery system for power generation has been included on the Lander Descent Stage. This system will be left with the Descent Stage on the lunar surface to minimize total lander mass.

Various configurations of the Lunar Lander were used to examine launch vehicle packaging requirements. Pro/Engineer was used to explore the various Lunar Lander configurations, taking into account habitable volume placement, airlock placement, propellant tank placement, and the desire to keep the main engines close together. The results from these variations indicate that a payload launch shroud of approximately 6.5 to 7.0 meters is needed to accommodate the Lander. A final high-fidelity CAD model was not developed for the Lunar Lander.

Component Name	Qty	Description/Function Performed
Avionics		
Data Interface Units	8	Collect and transmit data
Flight Computer	4	Flight critical computers for implementing dual fault-op tolerant processing
GPS	2	Space-integrated GPS / INS computers perform vehicle guidance and navigation processing
GPS Combiner Unit	1	Combine GPS signals
K-Band Phased Array Antenna	1	Provide communication
K-Band Signal Processor	1	Process comm. Signal
K-Band Transponder/Signal Amplifier	2	Provide communication
LADAR	2	Laser detection and ranging for automated rendezvous & docking
Multi-Function Display Panel	2	Multifunction LCD displays to provide crew interface for system status and command input
Operations Data Recorder	1	Record vehicle data for post-mission processing
Rotational/Translational Hand Controller	2	Provide manual vehicle flight control
S-Band Comm Transponder / Power Amplifier / Switching Unit	2	Provide vehicle-to-vehicle and vehicle-to-ground communication
S-Band Dual Beam Antenna	4	Provide communication
Star Tracker	2	Provides on-orbit vehicle attitude determination data to augment GPS / INS
Switch Panel	2	Control switch panels to provide functions not controlled by multi-function displays
UHF Antenna	2	Provide communication
UHF Comm Transceiver/Switching Unit	2	Provide communication
Video System	1	Video cameras & video processing equipment
Crew Accommodations		
Clothing	0.46 kg/p/d	No clothes washing assumed
Commode	1	Waste collection and disposal
Cooking & Eating Supplies	0.5 kg/p	Prepare and consume food
Crew	4	Mass of a 95^{th} percentile American male
Crew Health Care Kit	1	Medicine, basic medical equipment
Emergency Breathing Apparatus	4	Provide emergency oxygen for crew
Food	2.3 kg/p/d	A combination of shelf-stable and dehydrated food.
Food Warmer	1	Prepare food
Hand Tools	1	Tools for in-flight maintenance
Lightweight Recumbent Seats	4	Reclinable seats for zero-g transits, lunar surface operations, and ascent/descent phases. May also serve as sleeping surface.

Component Name	Qty	Description/Function Performed
Personal Hygiene Kit	4	Washcloth, toothbrush, razor, etc.
Photography/Mission Documentation Kit	1	For mission documentation – digital still and video cameras, supplies, etc.
Recreational Equipment	5 kg/p	Misc. personal effects
Sleep Accommodations	9 kg/p	Similar to Shuttle accommodations
Stowage	As needed	Soft stowage assumed
Vacuum	1	Housekeeping and lunar dust cleanup
Water Spigot	1	Provide potable water
Environment		
Ambient Temperature Catalytic Oxidizer w/ Charcoal Trace Contaminant Control	1	Remove trace gas contaminants from cabin atmosphere
Atmosphere Composition Monitoring	1	Monitor oxygen & carbon dioxide partial pressure
Cabin Fans	As needed	Cabin thermal conditioning
Combined CO2/Moisture Removal System	2	Remove carbon dioxide and moisture from cabin atmosphere. Each system is internally redundant.
EVA Tools	As needed	Handholds, tethers, etc.
EVA Umbilicals	4	For emergency full cabin depressurizations
Fire Detection and Suppression	1	Smoke detectors, fixed and portable halon extinguisher equipment
Flexible Body-Mounted Radiator	1	Radiate vehicle waste heat to deep space. 10 mil Ag-Teflon radiator coating (α/ε = 0.142)
Heat Collection Fluid Loop	2	Lines, valves, pumps, cold plates, heat exchangers. Single-phase 60% $C_3H_8O_2$ – 40% H_2O fluid. 9 kW total heat load.
Exploration EVA Suit & Spares	4	EVA suit for lunar surface exploration. Seven 2-person EVAs assumed.
Nitrogen Storage and Distribution System	1	Gaseous storage tanks. 70% N_2 – 30% O_2 atmosphere @ 9.5 psia
Oxygen Storage and Distribution System	1	Breathable oxygen shared storage w/ Ascent Stage oxidizer & fuel cell reactant. 70% N_2 – 30% O_2 atmosphere @ 9.5 psia
Potable Water Storage	1	Potable water produced by fuel cells. 25 kg maximum storage.
Wastewater Storage	2	Excess assumed to be periodically vented. 25 kg maximum storage.
Water Evaporator	1	For peak load heat rejection using excess water produced by fuel cells
Other		
Inflatable Airlock	1	Airlock on Lander Descent Stage to support daily EVAs on the lunar surface.
Docking Window	1	Window in docking adapter hatch to aid rendezvous & docking operations

Component Name	Qty	Description/Function Performed
Landing Gear	1	4-legged strokeable landing gear on Lander Descent Stage for landing on the lunar surface
Low Impact Docking System	1	Fully androgynous low impact docking system for pressurized mating to other architecture elements (e.g. CEV)
Pyros & Release Mechanisms	As needed	Provide for mechanical separation of vehicle components
Radiation Protection	As needed	5 gm/cm^2 of polyethylene radiation protection distributed around the outside of the Ascent Stage pressure vessel
Passive Thermal Control		
Insulation Blankets	As needed	Vehicle passive thermal control. 2 kg/m^2 multi-layer insulation blankets
Power		
Descent Stage Body-Mounted Solar Arrays	3	Provide vehicle power from launch through crew ingress at L1. 2 kW peak power generated per array. Triple-junction GaAs solar cells at 25% AM0 efficiency.
Electrical Power and Distribution Bus	3	28 Vdc bus. Includes remote power control units, wiring, inverters, wiring trays, etc.
Li-ion Primary Battery	4	Provides keep-alive vehicle power during eclipse periods in low Earth orbit. Total energy requirement = 2 kW-hr
PEM Fuel Cell	3	Provides vehicle power and potable water for all crewed mission phases up to Ascent Stage disposal. Peak power per FC stack = 6 kW. Total energy requirement = 1,908 kW-hr
Oxygen Reactant Accumulator Tank	2	Store 8 hr supply of O_2 reactant for fuel cell @ 1,000 psi. Reactant shared storage w/ Ascent Stage oxidizer. Graphite-epoxy overwrapped tanks w/ Inconel liner.
Hydrogen Reactant Tank	2	Store entire supply of supercritical H_2 reactant for fuel cell @ 500 psi. Graphite-epoxy overwrapped tanks w/ Inconel liner.
Propulsion		
Ascent Stage OMS 7,500 lbf Pressure-Fed Engines	4	Provide orbital maneuvering capability. P_c = 175 psi, ε = 150, I_{sp} = 362 s, MR = 3.8:1. Throttle range = 5:1. Ascent Stage engines also used for descent maneuvers.
Ascent Stage OMS/RCS Fuel & Oxidizer Tanks	2	Al 7075 common-bulkhead tanks @ 275 psi for oxygen and methane storage. Methane OMS/RCS shared storage; Oxygen OMS/RCS/Fuel Cell/ECLSS O2 shared storage. Includes cryocoolers & MLI

Component Name	Qty	Description/Function Performed
Ascent Stage OMS/RCS Helium Pressurization Tanks	2	Graphite-epoxy overwrapped tanks w/ Inconel liner @ 6,000 psi.
Ascent Stage RCS 50 lbf Thrusters	12	Provide vehicle attitude control for all mission phases. P_c = 125 psi, ε = 40, I_{sp} = 315 s, MR = 3.8:1.
Descent Stage OMS/RCS Fuel & Oxidizer Tanks	2	Al 7075 common-bulkhead tanks @ 275 psi for oxygen and methane storage. Methane and oxygen OMS/RCS shared storage. Includes cryocoolers & MLI
Descent Stage OMS/RCS Helium Pressurization Tanks	2	Graphite-epoxy overwrapped tanks w/ Inconel liner @ 6,000 psi.
Descent Stage RCS 50 lbf Thrusters	16	Provide vehicle attitude control for all mission phases through descent. P_c = 125 psi, ε = 40, I_{sp} = 315 s, MR = 3.8:1.
Structure		
Ascent Stage Pressure Vessel Structure	1	Al-Li 8090 primary/secondary structure. 21.5 m^3 internal pressurized volume
Descent Stage Structure	1	Al-Li 8090 primary/secondary descent stage platform structure

Table 10.5.2-2: Lunar Lander Subsystem Component Description

10.5.3 Kick Stage

As was described in the architecture description section, a Kick Stage has been included in the trade reference mission to insert the Lunar Lander and CEV into orbit at Lunar L1 and to deliver the Lunar Lander and crew from L1 to low lunar orbit. This was done to minimize the size of the Lunar Lander Descent Stage and total architecture IMLEO. The Kick Stage is assume to be pre-integrated with the Lunar Lander prior to launch, which allows the stage's required functionality to be simplified as much as possible. Vehicle control and power generation capabilities are assumed to be provided by the Lunar Lander.

To reduce architecture mass, an oxygen/hydrogen main propulsion system is used. Vehicle volumetrics are not as critical as with the Lunar Lander, as this stage is not being used to land on the Moon, therefore, the low packaging efficiency of hydrogen was not considered to be a significant issue. Two 25,000 lbf pump-fed engines, identical to the Earth Departure Stage engines, are mounted to the aft end of the Kick Stage. Only one engine was required to achieve the desired thrust-to-weight ratio, however two engines were used to support an engine-out capability. Like the EDS, the Kick Stage includes two coaxial cylindrical tanks to store the liquid oxygen and liquid hydrogen with aluminum intertank structure. An entirely passive MLI-based thermal control system is included to minimize propellant boiloff, and propellant tanks have been sized to accommodate the volume of propellant lost to boiloff during the mission. The oxygen tank is pressurized to 50 psi with gaseous helium, and the hydrogen tank is autogenously pressurized using gaseous hydrogen drawn from the main engines. Table 10.5.3-1 lists components assumed for the TRM Kick Stage.

Component Name	Qty	Description/Function Performed
Avionics		
Data Interface Units	8	Collect and transmit data. Includes wiring and sensors as needed.
Other		
Attachment System	1	Structural mating to Lunar Lander Descent Stage
Pyros & Release Mechanisms	As needed	Provide for mechanical separation of vehicle components
Passive Thermal Control		
Insulation Blankets	As needed	Vehicle passive thermal control. 2 kg/m^2 multi-layer insulation blankets
Power		
Electrical Power and Distribution Bus	3	28 Vdc bus. Distribute power to Kick Stage components. Power provided by Lunar Lander.
Propulsion		
OMS 25,000 lbf Pump-Fed Engines	2	Provide orbital maneuvering capability. P_c = 1,100 psi, ε = 200, I_{sp} = 459 s, MR = 6:1.
OMS Fuel & Oxidizer Tanks	1 per fluid	Al 7075 tanks @ 50 psi for oxygen and hydrogen storage. Includes MLI for passive cooling
OMS Oxidizer Helium Pressurization Tanks	2	Graphite-epoxy overwrapped tanks w/ Inconel liner @ 6,000 psi for oxygen pressurization.
Structure		
Intertank Structure	1	Al-Li 8090 skin-and-stringer construction intertank structure

Table 10.5.3-1: Kick Stage Subsystem Component Description

10.5.4 Earth Departure Stages

The TRM Earth Departure Stages are assumed to be analogous in function and design to the high-energy upper stages of the Saturn V launch vehicle and Evolved Expendable Launch Vehicle (EELV) fleets. The stages execute Earth orbit departure maneuvers to deliver the Lunar Lander/Kick Stage and CEV, respectively, to the vicinity of Lunar L1. Prior to mating in LEO, the EDS's are required to maintain vehicle attitude and transmit health status data to Earth, and following completion of the EOD maneuver, to separate from their payloads and safely dispose themselves. Due to the imposed two-week launch spacing assumption, the stages must loiter in LEO for several weeks before the next architecture element is launched.

To minimize Earth Departure Stage size and overall architecture mass, high-performance liquid oxygen (O_2) and liquid hydrogen (H_2) were selected as the main propellant combination. Pump-fed engines, as opposed to the pressure-fed systems on the Lunar Lander and CEV, were chosen to maximize performance and reduce the mass of the voluminous liquid H2 tanks. Each stage consists of a single cylindrical H_2 tank oriented coaxially with a cylindrical O_2 tank and connected to the O_2 tank by a skin-and-stringer type aluminum intertank structure. The tanks are lined with multi-layer insulation to minimize propellant boil-off during launch and loiter phases. Figure 10.5.4-1 shows how hydrogen boil-off varies as function of loiter time on orbit and number of MLI layers. Oxygen and hydrogen on the EDS lost due to boil-off is assumed to be periodically vented overboard to control tank pressure, and tanks were sized to accommodate the extra volume required for this propellant. No means for active propellant cooling on the EDS have been assumed.

At the aft end of the stage are mounted four 25,000 lbf pump-fed engines. These engines are assumed to be similar in design (maximum thrust, engine weight, and specific impulse) to the RL-10 family of engines found on the Atlas V and Delta IV launch vehicle upper stages. Future refinement efforts should determine if the vehicle's design requirements would allow the actual RL-10 engines to be used on the EDS for potential reductions in development time and DDT&E/production cost. In determining the number of engines required for the stage, total engine thrust for the Lander EDS was driven by the amount of gravity loss during the Earth orbit departure maneuver that would be considered acceptable. Figure 10.5.4-2 illustrates how gravity losses for the Earth Departure Stage decrease with higher combined-element thrust-to-weight ratios. With four engines operating at full thrust, the combined Earth Departure Stage, Lunar Lander, and Kick Stage have a thrust-to-weight of 0.3 at engine ignition resulting in ~50 m/s of gravity loss delta-V. This extra delta-V must be added to the theoretical minimum delta-V of 3,054 m/s for the maneuver to determine the actual design requirement. The Lander EDS could suffer a loss of one engine during the maneuver without a significant rise in gravity loss, but as the figure shows, gravity losses rapidly increase with thrust-to-weight ratios less than 0.2 at ignition.

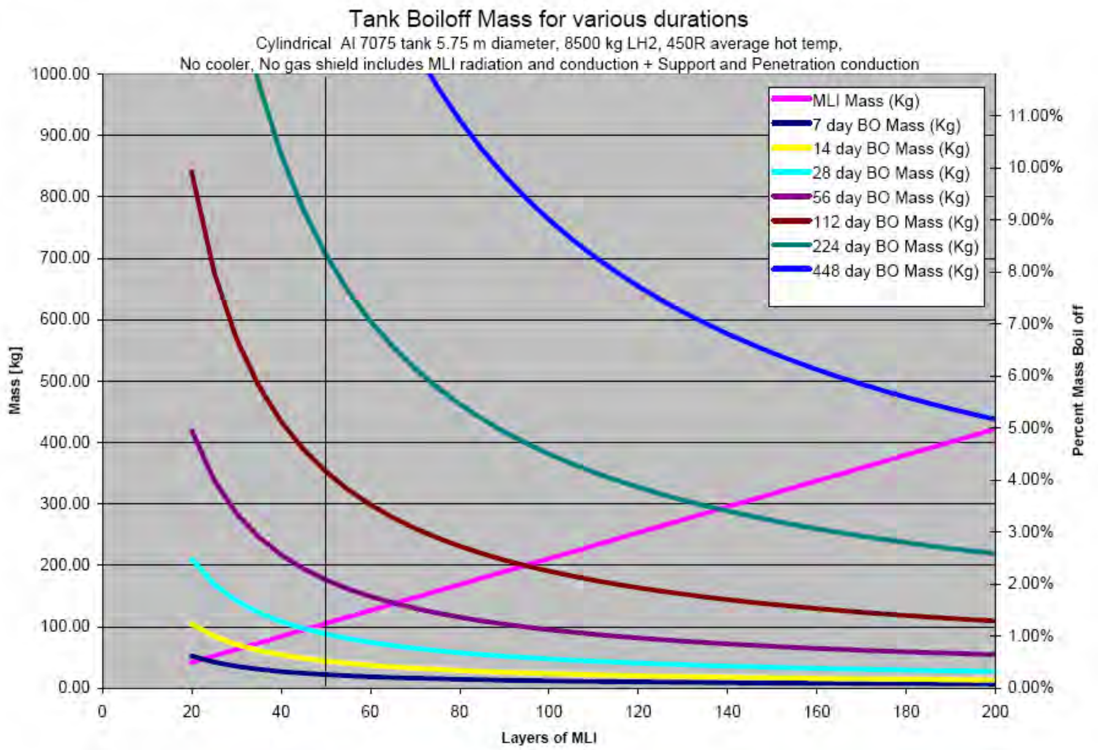


Figure 10.5.4-1: Hydrogen Boiloff for Earth Departure Stages

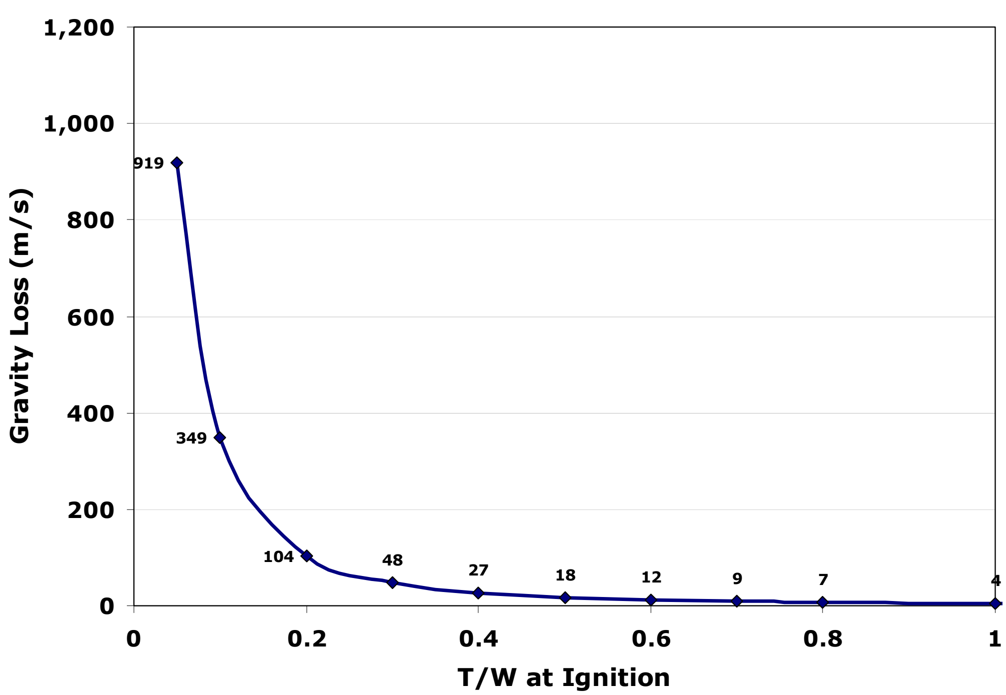


Figure 10.5.4-2: Gravity Loss During Earth Orbit Departure

Also located at the aft end of the Earth Departure Stage is the reaction control system. A gaseous hydrogen/gaseous oxygen propellant combination was initially selected. Evaporated propellants from the main engines are assumed to be tapped off and temporarily stored in four high pressure accumulator tanks (two per fluid). The RCS then uses this propellant with its sixteen 50 lbf RCS thrusters to stabilize and change vehicle attitude. Moving forward, on the exterior of the tank walls and intertank structure are mounted three radiator panels and three solar array panels. These panels allow the EDS to operate independently in LEO by rejecting vehicle waste heat and producing the necessary electrical power required. Avionics, power distribution, and heat collection equipment is mounted in the intertank area. See Table 10.5.4-1 for further definition of these components. Finally, an interstage structure with a low impact docking system is connected to the forward part of the stage to allow for mating to the Lunar Lander or CEV.

Component Name	Qty	Description/Function Performed
Avionics		
Data Interface Units	8	Collect and transmit data
Flight Computer	2	Flight critical computers for implementing dual fault-op tolerant processing
GPS	2	Space-integrated GPS / INS computers perform vehicle guidance and navigation processing
GPS Combiner Unit	1	Combine GPS signals
LADAR	2	Laser detection and ranging for automated rendezvous & docking
S-Band Comm Transponder / Power Amplifier / Switching Unit	2	Provide vehicle-to-vehicle and vehicle-to-ground communication
S-Band Dual Beam Antenna	2	Provide communication
Star Tracker	2	Provides on-orbit vehicle attitude determination data to augment GPS / INS
UHF Antenna	2	Provide communication
UHF Comm Transceiver/Switching Unit	2	Provide communication
Environment		
Flexible Body-Mounted Radiator	3	Radiate vehicle waste heat to deep space. 10 mil Ag-Teflon radiator coating (α/ε = 0.142)
Heat Collection Fluid Loop	2	Lines, valves, pumps, cold plates. Single-phase 60% $C_3H_8O_2$ – 40% H_2O fluid. 2 kW total heat load.
Other		
Low Impact Docking System	1	Fully androgynous low impact docking system for structural mating to other architecture elements (e.g. CEV, Lunar Lander)
Pyros & Release Mechanisms	As needed	Provide for mechanical separation of vehicle components
Passive Thermal Control		
Insulation Blankets	As needed	Vehicle passive thermal control. 2 kg/m^2 multi-layer insulation blankets
Power		
Body-Mounted Solar Arrays	3	Provide vehicle power from launch through disposal. 2 kW peak power generated per array. Triple-junction GaAs solar cells at 25% AM0 efficiency.
Electrical Power and Distribution Bus	3	28 Vdc bus. Includes remote power control units, wiring, inverters, wiring trays, etc.
Li-ion Primary Battery	4	Provides keep-alive vehicle power during eclipse periods in low Earth orbit. Total energy requirement = 2 kW-hr

Component Name	Qty	Description/Function Performed
Propulsion		
OMS 25,000 lbf Pump-Fed Engines	4	Provide orbital maneuvering capability. P_c = 1,100 psi, ε = 200, I_{sp} = 459 s, MR = 6:1.
OMS Fuel & Oxidizer Tanks	1 per fluid	Al 7075 tanks @ 50 psi for oxygen and hydrogen storage. Includes MLI for passive cooling
OMS Oxidizer Helium Pressurization Tanks	2	Graphite-epoxy overwrapped tanks w/ Inconel liner @ 6,000 psi for oxygen pressurization.
RCS 50 lbf Thrusters	16	Provide vehicle attitude control for all mission phases. GO2/GH2 propellant. P_c = 125 psi, ε = 40, I_{sp} = 370 s, MR = 6:1.
RCS GO2/GH2 Tanks	2 per fluid	Graphite-epoxy overwrapped tanks w/ aluminum liner
Structure		
Intertank Structure	1	Al-Li 8090 skin-and-stringer construction intertank structure

Table 10.5.4-1: Earth Departure Stage Subsystem Component Description

10.5.5 Vehicle Mass Properties for Trade Reference Mission

TRM vehicle mass properties as generated by the Envision parametric sizing tool are listed in Table 10.5.5-1. Subsystem components are categorized according the mass properties reporting standard outlined in JSC-23303 *Design Mass Properties: Guidelines and Formats for Aerospace Vehicles*. All estimates include 20% margin applied to categories one through eight of the vehicle mass properties for dry mass growth.

	CEV CM	CEV SM	CEV Earth Dep. Stage	Ascent Stage	Descent Stage	Kick Stage	Lander Earth Dep. Stage
1.0 Structure	1,523	1,455	932	839	553	621	1,972
2.0 Protection	822	0	0	73	50	0	0
3.0 Propulsion	117	1,408	2,318	1,631	1,413	1,530	4,361
4.0 Power	482	661	190	813	137	100	190
5.0 Control	0	0	0	0	0	0	0
6.0 Avionics	737	0	171	738	0	0	175
7.0 Environment	691	110	104	851	530	0	105
8.0 Other	835	100	455	455	708	405	455
9.0 Growth	1,041	747	834	1,080	678	531	1,452
DRY MASS	6,249 kg	4,481 kg	5,004 kg	6,479 kg	4,071 kg	3,188 kg	8,710 kg
10.0 Cargo	966	305	1,355	1,483	464	953	3,109
11.0 Non-Cargo	1,478	0	0	227	500	0	0
INERT MASS	8,692 kg	4,786 kg	6,359 kg	8,190 kg	5,035 kg	4,141 kg	11,819 kg
12.0 Non-Propellant	55	1,442	0	1,014	0	0	0
13.0 Propellant	64	11,332	32,896	10,703	17,573	23,323	82,289
GROSS MASS	8,812 kg	17,560 kg	39,255 kg	19,906 kg	22,608 kg	27,465 kg	94,109 kg

Table 10.5.5-1: TRM Vehicle Mass Properties

The largest single element to be launched is the Earth Departure Stage for the Lunar Lander and Kick Stage with an initial mass in low Earth orbit (IMLEO) of 94 t. This stage is the first element launched in the architecture, and it executes the Earth orbit departure burn for the Lander and Kick Stage. The launch of this element will drive the payload delivery capabilities of the cargo launch vehicle. The Kick Stage, Lander Ascent Stage, and Lander Descent Stage are launched next with a combined launch mass of 60 t. Following arrival of these vehicles at Lunar L1, a 39 t Earth Departure Stage for the CEV is launched, and finally, the CEV is launched with the crew on a human-rated launch vehicle capable of delivering 26 t to LEO. The combined architecture elements of the trade reference mission have a total IMLEO of 230 t.

L1 TRM Mass Properties (kg)	CEV CM	CEV SM	CEV EDS	Ascent Stage	Descent Stage	Kick Stage	Lander EDS
1.0 Structure	**1,523**	**1,455**	**932**	**839**	**553**	**621**	**1,972**
Primary Structure	1522.1	0.0	0.0	737.1	400.0	0.0	0.0
Stowage Equipment	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Chemical Propulsion Stage Structure	1.0	1454.6	931.8	102.4	153.4	621.4	1971.6
2.0 Protection	**822**	**0**	**0**	**73**	**50**	**0**	**0**
Thermal Protection System	738.8	0.0	0.0	0.0	0.0	0.0	0.0
Insulation	82.7	0.0	0.0	72.6	50.0	0.0	0.0
3.0 Propulsion	**117**	**1,408**	**2,318**	**1,631**	**1,413**	**1,530**	**4,361**
OMS Engines & Installation	0.0	212.9	688.7	485.7	0.0	463.0	914.5
RCS Engines & Installation	63.0	163.0	153.1	81.5	108.7	0.0	153.1
OMS Fuel Tanks & Feed/Fill/Drain System	0.5	442.2	739.8	469.5	580.4	687.9	1866.3
OMS Oxidizer Tanks & Feed/Fill/Drain System	0.5	421.6	450.6	441.9	522.0	308.9	1003.3
RCS Fuel Tanks & Feed/Fill/Drain System	53.0	11.8	89.3	6.7	8.4	0.0	114.0
RCS Oxidizer Tanks & Feed/Fill/Drain System	0.0	13.2	97.5	7.5	9.4	0.0	139.9
Pressurization System	0.0	143.6	98.9	137.8	184.4	70.4	170.3
4.0 Power	**482**	**661**	**190**	**813**	**137**	**100**	**190**
Fuel Cell	0.0	210.1	0.0	210.1	0.0	0.0	0.0
Regenerative Fuel Cells	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Hydrogen PRSD Tanks	0.0	253.7	0.0	207.1	0.0	0.0	0.0
Oxygen PRSD Tanks	0.0	77.7	0.0	91.9	0.0	0.0	0.0
Photovoltaic Arrays	0.0	0.0	84.0	0.0	47.7	0.0	84.0
Battery Type #1	171.1	0.0	6.1	0.0	12.2	0.0	6.1
Battery Type #2	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Power Management & Distribution	311.3	120.0	100.0	304.3	77.5	100.0	100.0
Nuclear Reactor	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5.0 Control	**0**	**0**	**0**	**0**	**0**	**0**	**0**
Flight Control Surface Actuation	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6.0 Avionics	**737**	**0**	**171**	**738**	**0**	**0**	**175**
Command, Control, and Data Handling	161.5	0.0	38.9	161.5	0.0	0.0	38.9
Guidance & Navigation	145.1	0.0	40.7	145.1	0.0	0.0	40.7
Communications	79.5	0.0	36.0	117.7	0.0	0.0	36.0
Vehicle Health Management	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Cabling and Instrumentation	351.2	0.0	55.1	313.3	0.0	0.0	59.8
7.0 Environment	**691**	**110**	**104**	**851**	**530**	**0**	**105**
Environmental Control & Life Support System							
Nitrogen Storage	30.2	0.0	0.0	44.1	0.0	0.0	0.0
Oxygen Storage	17.7	0.0	0.0	18.1	0.0	0.0	0.0
Atmosphere Supply Reg, Dist, and Control	56.9	0.0	0.0	60.3	0.0	0.0	0.0
Atmosphere Contaminant Control	122.1	0.0	0.0	97.6	0.0	0.0	0.0
Fire Detection and Suppression	20.3	0.0	0.0	20.3	0.0	0.0	0.0
Venting and Thermal Conditioning	47.5	0.0	0.0	54.9	0.0	0.0	0.0
Water Management	29.1	0.0	0.0	29.7	0.0	0.0	0.0

Airlock/EVA Support	0.0	0.0	0.0	18.5	18.5	0.0	0.0
Airlock	0.0	0.0	0.0	0.0	449.0	0.0	0.0
Umbilicals and Support	21.6	0.0	0.0	62.4	62.4	0.0	0.0
Thermal Control System							
ETCS	63.3	0.0	20.7	56.4	0.0	0.0	21.6
ITCS	140.8	0.0	59.1	140.8	0.0	0.0	59.1
Radiator	0.0	109.7	24.4	119.5	0.0	0.0	24.4
Fluid Evaporator System	20.2	0.0	0.0	20.0	0.0	0.0	0.0
Phase Change Heat Rejection	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Heat Pump	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Crew Accommodations							
Galley	38.9	0.0	0.0	38.5	0.0	0.0	0.0
Waste Collection System	29.4	0.0	0.0	28.9	0.0	0.0	0.0
Seats & Tables	53.3	0.0	0.0	41.0	0.0	0.0	0.0
8.0 Other	**835**	**100**	**455**	**455**	**708**	**405**	**455**
Parafoil Assembly	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Main Parachutes	196.2	0.0	0.0	0.0	0.0	0.0	0.0
Drogue Parachutes	48.2	0.0	0.0	0.0	0.0	0.0	0.0
Landing, Flotation, & Misc Chutes	91.3	0.0	0.0	0.0	608.3	0.0	0.0
Shell Heaters	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Doors, Hatches, Pyros, and Docking Adapters	499.3	100.0	455.2	455.2	100.0	405.2	455.2
9.0 Growth	**1,041**	**747**	**834**	**1,080**	**678**	**531**	**1,452**
10.0 Non-Cargo	**966**	**305**	**1,355**	**1,483**	**464**	**953**	**3,109**
Personnel Provisions							
Recreational Equipment	20.0	0.0	0.0	20.0	0.0	0.0	0.0
Crew Health Care	54.9	0.0	0.0	54.9	0.0	0.0	0.0
Personal Hygiene	11.7	0.0	0.0	11.2	0.0	0.0	0.0
Clothing	27.6	0.0	0.0	24.4	0.0	0.0	0.0
Housekeeping Supplies	25.0	0.0	0.0	23.6	0.0	0.0	0.0
Operational Supplies	72.7	0.0	0.0	52.0	0.0	0.0	0.0
Maintenance Equipment	25.0	0.0	0.0	25.0	0.0	0.0	0.0
Photography Supplies	45.0	0.0	0.0	45.0	0.0	0.0	0.0
Sleep Accommodations	36.0	0.0	0.0	36.0	0.0	0.0	0.0
EVA Suits and Spares	90.8	0.0	0.0	381.8	0.0	0.0	0.0
EVA Tools	17.7	0.0	0.0	0.0	0.0	0.0	0.0
Food	138.0	0.0	0.0	121.9	0.0	0.0	0.0
Crew	400.0	0.0	0.0	400.0	0.0	0.0	0.0
Reserve, Residual Fluids, and Gases							
Pressurant	0.0	78.1	107.9	73.3	112.5	74.7	266.2
Unused Fuel	0.3	47.2	540.6	44.6	73.2	273.3	1139.8
Unused Oxidizer	1.0	179.4	706.8	169.5	278.2	605.3	1703.0
11.0 Cargo	**1,478**	**0**	**0**	**227**	**500**	**0**	**0**
Ballast & Other Misc. Mass	100.0	0.0	0.0	0.0	0.0	0.0	0.0
Radiation Protection	1377.9	0.0	0.0	227.0	0.0	0.0	0.0
Payload	0.0	0.0	0.0	0.0	500.0	0.0	0.0
12.0 Non-Propellant	**55**	**1,442**	**0**	**1,014**	**0**	**0**	**0**

Fuel Cell Oxygen	0.0	1287.1	0.0	846.4	0.0	0.0	0.0
Fuel Cell Hydrogen	0.0	155.4	0.0	98.4	0.0	0.0	0.0
Oxygen (Life Support)	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Nitrogen (Life Support)	29.3	0.0	0.0	42.8	0.0	0.0	0.0
Fluid Evaporator System Water	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Potable Water	26.0	0.0	0.0	26.0	0.0	0.0	0.0
13.0 Propellant	**64**	**11,332**	**32,896**	**10,703**	**17,573**	**23,323**	**82,289**
Usable OMS Fuel (None)	0.0	2263.5	4661.1	2223.0	3632.4	3331.9	11659.4
Usable OMS Oxidizer (None)	0.0	8601.2	27966.4	8447.5	13803.3	19991.2	69956.6
Usable RCS Fuel (None)	13.3	97.4	38.4	6.7	28.6	0.0	96.2
Usable RCS Oxidizer (None)	50.7	370.2	230.4	25.6	108.6	0.0	577.0
Dry Mass	**6,249**	**4,481**	**5,004**	**6,479**	**4,071**	**3,188**	**8,710**
Inert Mass	**8,692**	**4,786**	**6,359**	**8,190**	**5,035**	**4,141**	**11,819**
Total Vehicle	**8,812**	**17,560**	**39,255**	**19,906**	**22,608**	**27,465**	**94,109**

Table 10.5.5-2: TRM Detailed Vehicle Mass Properties

Finally, when performing architecture-level trade studies similar as LDRM-2, understanding how sensitive total architecture mass is to increases in vehicle inert mass can be valuable. These sensitivity factors are known as architecture "gear ratios". For example, a Lander Ascent Stage gear ratio of 15 to 1 means that for every 1 kg increase in vehicle inert mass, total architecture mass increases by 15 kg. Gear ratios also exist for architecture parameters other than discrete vehicles, such as one for cargo carried with the crew through the entire round trip across multiple vehicles. Gear ratios calculated for the TRM are below.

Lander Earth Departure Stage:	*2.3:1*	*CEV Earth Departure Stage:*	*2.3:1*
Kick Stage:	*3.5:1*	*CEV Service Module:*	*4.3:1*
Descent Stage:	*6.7:1*	*CEV Crew Module:*	*5.0:1*
Ascent Stage:	*16.7:1*	*Round Trip Cargo:*	*22.7:1*

10.6 System Technologies and Programmatic Risks

The following section describes the key system technology needs and programmatic risks for the LDRM-2 trade reference mission.

10.6.1 TRM Technology Assessment

Across all of the TRMs, the up-mass requirements will drive the need for mass efficient solutions. There are a number of technologies that by themselves are not an enabling technology. When multiple technologies (power, thermal, structures, etc) that address mass efficient solutions are aggregated, the sum in turn may become an enabler. There were a number of technologies utilized as in developing the LDRM-2 TRM but the following were key enabling technologies.

- Capsule RCS (TRL5) - A mono-propellant RCS is the simplest, low cost, and risk solution. A stable non-toxic mono-propellant system, such as GN2/Tridyne, simplifies ground processing and recovery operations
- Service Module Propulsion/RCS (TRL 5) – A pressure-fed Lox/Ethanol and Lox/Methane with an integrated RCS using sub-cooled liquid propellants require prototype engine development, flight weight cryogenic valves, flight weight ignition system technology development
- Power (TRL 4-5) PEM Fuel Cells – Enables the use of common tankage and leverages terrestrial commercial development activities
- Advanced ECLSS for a combined regenerable CO2 & Humidity control (TRL 5) – Significant mass & volume savings with increased reliability over separate systems
- Lightweight Radiators (TRL 5) - Flexible metal fabric radiators with silver Teflon coating (TRL 5) utilized to reduce weight
- Single Loop Active Thermal Control System (TRL 5) - Use of a single heat transfer fluid simplifies the ATCS system that reduces weight, risk, cost, and increases reliability. Human rating of the non-toxic heat transfer fluid is required
- Advanced EVA (TRL 2-6) - A number of suit subsystem technologies (light weight life support, batteries, dust mitigation, etc) are needed along with dust resistant materials. Additional investigation is required in the airlock/dust lock designs and a suit port (rear entry, externally mounted EVA suits)
- Avionics (TRL 3-9) – Electronics that are protected from faults/failures due to space radiation are critical to the architecture. Advanced Avionics (computational speed, bus speed, networking, packaging, etc) would greatly enhance avionics effectiveness, potentially enable other technologies (IVHM), and enhance mission success
- GN&C AR&D Sensor Development (TRL 4) – RF based combined communications and navigation system, LADAR based system (maximum flexibility with high accuracy, dual use for AR&D and landing), and Natural Feature Recognition
- Structures & Mechanisms (TRL 4) – Low Impact Docking System to eliminate the need for high velocity mating operations (also supports AR&D and in-flight assembly)
- Capsule Ablative TPS (TRL 5) – Apollo Ablative TPS (AVCOAT-5061) is no longer available, a family of ablative materials exists but human spaceflight rating is required

Candidate Technology Demonstration Items for 2008 Test Flight

- Capsule Ablative TPS
- Capsule/SM RCS

Each of the subsystem technology reports in Section 20 of this report provides a detailed description of the subsystem design and technology options considered with rationale for candidate selection. Table 10.6-1 lists the technologies requiring maturation (i.e. TRL currently less than 6) to support the architecture associated with the TRM.

Name	TRL	Comments
1. Propulsion		
Capsule RCS	5	A mono-propellant RCS is the simplest, low cost, and risk solution. A stable non-toxic mono-propellant system, such as GN2/Tridyne, simplifies ground processing and recovery operations
Service Module Integrated OMS/RCS	5	A pressure-fed Lox/Ethanol and Lox/methane with an integrated RCS using sub-cooled liquid propellants require prototype engine development, flight weight cryogenic valves, flight weight ignition system technology development
2. Power		
PEM fuel cells	4-5	Commercial stack development for H2 & air; space development ongoing from NGLT; allows common tankage with propellants
3. Environmental Control and Life Support System (ECLSS)		
CO2 & Humidity Control - CMRS	5	Regenerable combined CO2 & moisture removal system w/solid amine swing beds save mass, volume, decreases complexity, decreases radiator size, & increases reliability
Reactive Plastic LiOH	5	Saves volume shape constraints vs regular LiOH 25+ % more LiOH into the same volume while not reducing the LiOH CO2 removal efficiency or capability
Motor-Settable Regulators	5	Continues to regulate atmosphere w/a loss of power to PCS have been used in the aircraft industry
4. Habitation System		
N/A		No technology development required for LDRM 2 (Mars testbed technologies still relevant - see report)
5. Active Thermal Control System		
Lightweight Radiators	5	Mass savings for radiators and for mounting structure Several technologies currently in mid-TRL range
Alternative Heat Transfer Fluid	5	Allows for a single loop design and greatly reduces ATCS risk and cost Potential fluids include aqueous propylene glycol, Galden, HFE 7100, Fluorinert 72

Name	TRL	Comments
6. Extravehicular Activity System (EVAS)		
Advanced EVA	2-6	A number of suit subsystem technologies (light weight life support, CO2 removal, thermal control, power, electronics and data management, space suit pressure garment, dust mitigation, Integrated EVA/robotic interfaces, rovers, etc) needed along with dust resistant materials. Additional investigation is required in the air-lock/dust lock designs and a suit port (rear entry, externally mounted EVA suits).
7. Avionics		
Advanced modular computer units	3-7	Small, distributed processors performing specific tasks (versus larger centralized computers). Redundant and highly reliable. This would help eliminate having to design with hardware that carries extra resources not needed that are commonly found on big single-board-computers (i.e., integrated I/O ports, bus I/Fs that will never be used). - Distributed, networked devices and computers may be more efficient than larger centralized computers. - Could be implemented in programmable logic (i.e., FPGAs) - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
High-speed, fault-tolerant data bus	5-7	100 Mbps to over 1 Gbps bandwidth required, using either copper or optical core. Built-in fault-tolerance and low-power consumption are desired. - IEEE 1394b and Fibre-Channel are good candidates. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
Wireless networked systems	5-7	Needed for crew cabin communication devices, laptops, PDAs, etc. - Bluetooth variant could be used for non-secure cabin communication.
Advanced wireless instrumentation sensors	1-6	A network of MEMS/ nanotechnology based sensors used to monitor vehicle health (i.e., temp, strain, pressure). The RF module could be integrated onto the sensor, which could wirelessly communicate with the data acquisition device. - Would eliminate a lot of vehicle wiring - Very small packaging would allow sensors to be easily located on vehicle. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
Radiation-hardened electronic technologies	3-9	New materials and fabrication processes may yield better rad-hard/tolerant electronics. New materials may also provide better shielding solutions for electronics. - Parts would be latchup immune, have very low SEE rates, & be total dose tolerant - Expensive option, but would greatly allow for stronger design performance, capability & flexibility - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.

Name	TRL	Comments
Autonomous Recon-figurable computing	3-5	Programmable hardware that detects and isolates failures, and reconfigures itself to continue operating successfully (i.e., “self-healing” machines) - Used to provide system fault-tolerance. - Designers must ensure inadvertent reconfiguration changes are not possible.
Embedded Avionics packaging	1-2	An embedded three- dimensional packaging technology with all EEE parts inside printed wiring boards which are bonded to a thermally conductive core to eliminate need for active cooling. The approach will also eliminate most solder connections, eliminate one third of the total electrical connections and metal housings to yield weight/volume savings - New concept with great potential for spacecraft electronics. - Could lead to substantially smaller, lower power avionics that generate less heat. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
New display and con-trol technologies	3-9	Useful for efficient crew-machine interfaces. - Includes touch screens, heads-up displays, LCD flat panels - New technologies under commercial development include Field-Emission Dis-plays and Organic Light Emitting Diodes displays
Speech recognition technology	2-4	Potential crew-machine interface with improved human factors. - Uncertain of current technology and/or use in spacecraft applications.
Artificial Intelligence	1-4	Potential for great improvements in spacecraft operations, ranging from mundane crew tasks to eventually flight control. - Uncertain of current technology and/or use in spacecraft applications. - Uncertain of development challenges.
Advanced error detec-tion & correction schemes	2-4	Would allow for greater fault detection, isolation & recovery, thereby increasing the system’s reliability. - Uncertain of current technology or development challenges. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
Advanced data com-pression techniques	2-4	Would allow more vehicle data to be sent to the ground or to onboard data re-corders. - Uncertain of current technology or development challenges.
Advanced encryp-tion/decryption codes	2-4	Would allow for more secure command uplink and telemetry downlink. - Uncertain of current technology or development challenges.
Wire Integrity	2-3	Technology that will determine the condition of installed wiring and cable har-nesses inside a spacecraft. - Could be used for production verification on the ground, and possibly on-board in space for in-flight troubleshooting.

Name	TRL	Comments
8. Guidance, Navigation, and Control		
Radio Frequency (RF) Based Navigation (combined w/ communication)	3 – 9**	Uses communication signal between spacecraft to provide long range relative state information Uses vehicle subsystem equipment (communication) that is required and present independent of relative navigation requirement State (position, velocity, bearing) information may be available to both halves of the interface (chaser and target) Potential for relative attitude measurement capability, so may be applicable through to docking **Proven in space; however, certain aspects have been developed/used by Russia only; h/w, s/w and detailed results are not accessible, so TRL is lower based on availability
LADAR ("Laser radar")	4	Laser based detection and ranging system that processes a scanned signal into three-dimensional relative state information Flexible -- no need for any retro-reflectors or other devices on target vehicle Potential for use from long range (50 to 100 km) up to docking No lighting constraints Potential for dual use as landing sensor for altitude measurement and terrain mapping
NFIR (Natural Feature Image Recognition)	4	Processes video camera image into three-dimensional relative state information; software based solution No need for any retro-reflectors or other devices on target vehicle Uses subsystem equipment (camera) that will likely be present to meet human-rating requirements May impose natural or artificial lighting requirements Useable range is a function of camera focal length (in practice max range ~1 km)
GN&C & FDIR Algorithms	2 - 4	Perform both nominal and contingency functions to ensure safe/successful docking for crewed and uncrewed elements. Technology emphasis is on contingency capabilities, especially when crew is present Must be coordinated with AFM which will balance ground/onboard and human/computer responsibilities
Automated & Precision Landing – guidance, trajectory management, and hazard avoidance	3	Algorithm and corresponding software that provide the solutions necessary to support Automated & Precision Landing coupled with the LADAR and NFIR hardware capabilities
Autonomous Flight Manager	4	AFM is considered to be enabling for complex operations where no crew is present and time lags make ground operations impractical

Name	TRL	Comments
9. Communications		
Ka-band	4-5	High data rates possible with smaller antennas Power amplifiers are not very efficient at this frequency.
Software Defined Radio	4-5	Reduces size and weight of communication subsystems by having one box that can communicate with multiple networks (TDRSS, DSN, GPS, etc.) Software defined radios have been tested in space but are not available yet for very high data rate communications.
UWB	2	Can be used for tracking, Can operate at higher data rates than 802.11 Designed for short distances currently Will need to look at using higher gain antennas and higher power amplifiers that are not allowed on Earth in order to increase range of system. May not be able to use near Earth because of interference.
10. Structures		
Lightweight Structures	5	AL 2097 & AL 2099 are candidate materials that show promise for decreasing density and increasing stiffness, but more work is needed to characterize the materials.
11. Passive Thermal Control System		
N/A		Many currently available passive thermal control technologies will satisfy the needs for future lunar missions.
12. Thermal Protection System		
Ablative TPS		The original Apollo ablator material AVCOAT-5061 is no longer available. A human-rated material does not exist today which is suitable for lunar returns although significant progress in the development of a replacement material has been made by Applied Research Associates (ARA). ARA has developed a family of ablative materials which should be able to meet mission requirements. Further testing is required to human-rate the material system.
13. Mechanisms		
Low Impact Mating System	4	The primary goal of this technology is the: • Elimination of the need for high velocity docking • Robust and safe operation for deep space missions • Androgynous, modular design that is re-configurable for multiple operations and applications • Incorporates an active load-sensing system to realign the soft-capture ring automatically rather than requiring force to realign The elimination of the two force requirements is key to realizing all of the benefits offered by a low-impact docking system. This technology alleviates the requirement to ram-mate vehicles together with large closing velocities.

Table 10.6-1: Required Technology Developments for TRM Vehicles

10.6.2 Programmatic Risks

During the study process, potential programmatic risks were identified. The risk analysis and mitigation planning is still required. Below are the identified programmatic/developmental risks:

- Technology Development – Given the number of technology development needs, there is the possibility that some of the technologies will not be adequately matured to a TRL 6 by 2009 or may not be as effective as promised resulting in performance, cost, and schedule impacts.
- Weight Problems - Given all previous spacecraft developments encountered weight problems, there is a high probability that weight issues will be identified after the booster configuration has been defined necessitating significant and costly lander and spacecraft redesign efforts.
- Formulation Uncertainty - Given the lack of well-defined mission requirements, there is a possibility that the team's concept will not satisfy the customer's expectations. Exploration studies at this stage of project development are typically only given a few top-level requirements and mission objectives from which to develop a concept.
- Launch Vehicle Uncertainty - Given the Launch Vehicle approach has not been determined; there is the possibility the crew escape and abort approaches developed will not adequately provide for crew survival.
- L1 & Lunar Environments – Given the L1 & lunar environments, there is a possibility that the environment is "worse than expected" (radiation, thermal, MMOD, dust, etc) which may lead to system component failures jeopardizing mission success.
- Fault Tolerance - Assuming a two-fault tolerance just for redundancy sake may result in unavoidable increase in mass; may result in the lack of innovative functional redundancy measures (unlike redundancy).
- Test & Verification – Given in past programs that and verification test facility (e.g avionics hardware and software integration) have not been planned for, there is a possibility that the exploration program may also not plan for and develop an integration and verification test facility resulting in cost and schedule impacts. System integration should begin early in the program and not be an afterthought when the system is developed.
- Mission Analysis and Toolset – Given that mission analysis tools may not have been validated for the exploration missions including both nominal and off-nominal (e.g. abort, escape) scenarios, there is a possibility that the performance figures used for sizing (CEV, lander, etc) are inadequate for an actual mission.

11.0 TRM with 2-Launch Solution

This first architecture variant examines the impact of changing the number of launches required per mission. Instead of the four launches required for the trade reference mission, this variant launches all architecture elements in two separate launches. This section of the report examines the impact of such a change.

11.1 Major Assumptions/Clarifications

The "TRM with 2-Launch Solution" architecture variant affects the following TRM assumptions from the original LDRM-2 task request statement. Assumptions from Section 10.0 not explicitly listed here are still applicable to the architecture.

<u>4-launch solution:</u> This variant requires only two launches per mission. The first launch of the architecture combines the Lunar Lander, Kick Stage, and Lander Earth Departure Stage into a single launch. The second launch combines the CEV and CEV Earth Departure Stage.

<u>Automated rendezvous and docking shall be used to assemble the elements:</u> As the Lunar Lander and Kick Stage are pre-deployed to Lunar L1, and the Lunar Lander/Kick Stage and CEV launch with their respective Earth Departure Stages, there is no need to assemble any architecture elements in low Earth orbit. Therefore, this assumption is not necessary.

<u>Crew must be launched on a human-rated launch system:</u> This assumption is still valid, however in the two launch per mission architecture, the CEV will now launch with its Earth Departure Stage which may affect the human rating of the system.

11.2 Architecture Description

The two launch per mission architecture seen in Figure 11.2-1 begins with the launch of the Lander Earth Departure Stage, Kick Stage, and Lunar Lander as a single combined element. The assumed cargo launch vehicle for the architecture delivers that element to the LEO parking orbit previously assumed (28.5^{o} 407 km), where it loiters 1-2 orbit revolutions for vehicle checkout. Within 3 hr after launch, the Earth Departure Stage performs the Earth orbit departure maneuver for the Lunar Lander and Kick Stage. Unlike the TRM where the Lunar Lander and Kick Stage were launched separate from the EDS and needed to launch and mate with the EDS within the three-day launch window to avoid missing the first available Earth orbit departure opportunity, the direct injection strategy employed here enables much greater launch flexibility. The TRM pre-deployed and assembled multiple architecture assets in LEO prior to departing for Lunar L1, and injection opportunities arose on average once per every ten days on orbit. If that first opportunity were missed, the Kick Stage, Lander, and EDS needed to loiter in LEO between 3-12 (average of 10) days until the next opportunity. With single launch architectures where no LEO rendezvous or assembly is necessary, two distinct opportunities for coplanar departure to L1 occur in any given 25-hr period, one on a northerly launch azimuth and one on a southerly azimuth. Further, if the cargo launch vehicle has sufficient performance to accept launch azimuths between 72^{o} and 108^{o}, as was the case with the Apollo missions, each of the two daily launch opportunities has an associated 4.5-hr launch window. Launch delays will not cause the elements to begin accumulating on-orbit time.

Two weeks after the EDS, Lunar Lander and Kick Stage initially launch the CEV and its Earth Departure Stage launch to the LEO parking orbit on a human-rated launch vehicle. As with the previous launch, the CEV and EDS loiter in LEO up to 3 hr for vehicle checkout before the EDS performs the Earth orbit departure maneuver. Using a direct injection architecture again allows for two CEV launch and L1 departure opportunities per day. The TRM, on the other hand, required 109 hr of additional crew time in space for rendezvous, mating, and weather delay protection. If the CEV could not launch from Earth within its 3-day window, a departure opportunity was missed and the Lunar Lander/Kick Stage needed to loiter longer at L1. If the CEV did launch, but was unable to execute Earth orbit departure on time, the entire mission would likely be lost. Direct injection avoids these complications.

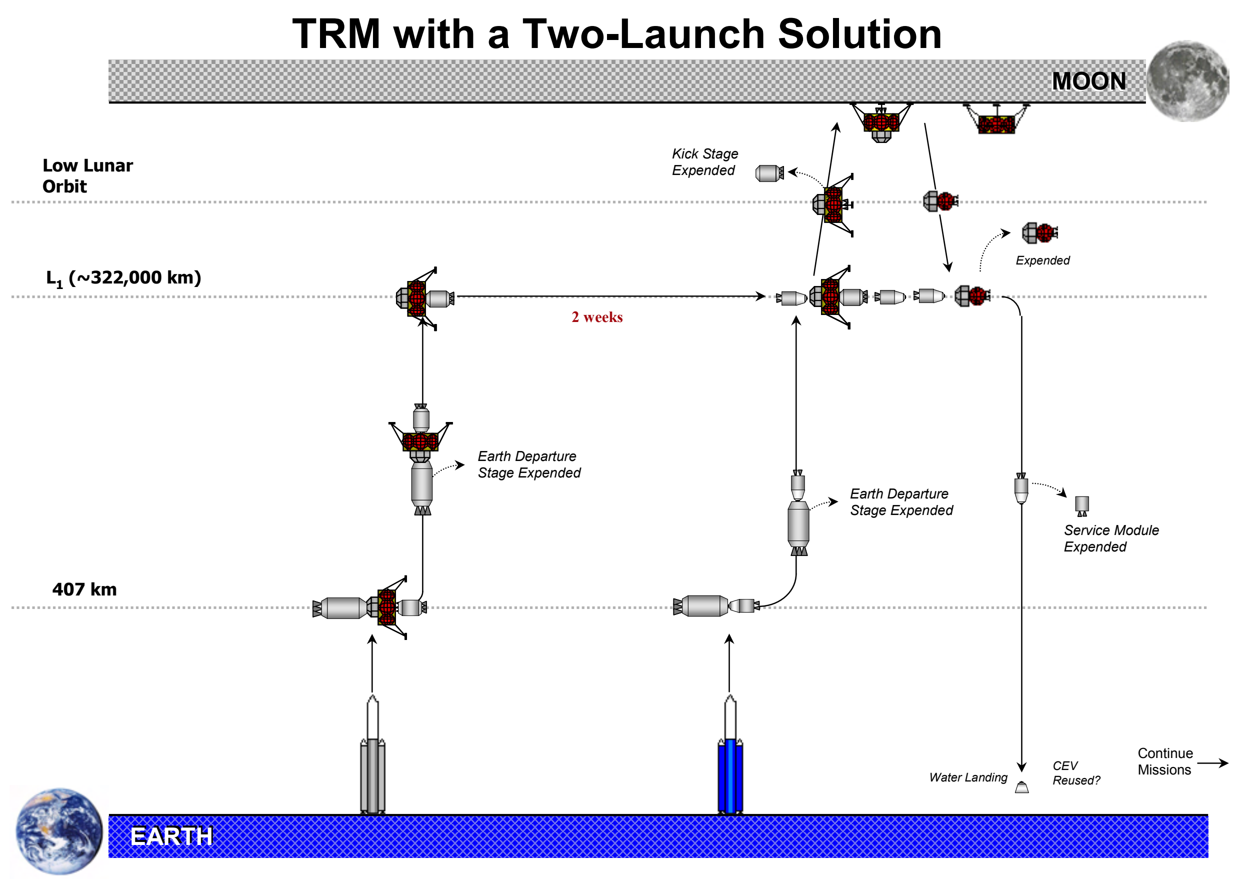


Figure 11.2-1: TRM with 2-Launch Solution Architecture Illustration

Once the CEV EDS completes Earth departure, the remainder of the mission functions identically to the trade reference mission. Tables 11.2-1 outlines the assumed timelines for the two launch per mission architecture as just described.

Vehicle	Phase Name	Phase Length	Mission Elapsed Time						
			Overall MET		EDS1	Kick Stage	Lander	EDS2	CEV
		(hr)	(hr)	(days)	(hr)				
EDS1/Kick Stage/Lander	Launch from Earth/Loiter	2	2	0.1	2				
EDS1/Kick Stage/Lander	Loiter in LEO	1	3	0.1	3				
EDS1	Earth Orbit Departure	0	3	0.1	3	0	0		
EDS1/Kick Stage/Lander	Coast	47	50	2.1	50	47	47		
EDS1	MCC & EDS Disposal	0	50	2.1	50	47	47		
Kick Stage/Lander	Coast	47	97	4.0		94	94		
Kick Stage/Lander	Libration Point Arrival	0	97	4.0		94	94		
Kick Stage/Lander	Loiter at L1	239	336	14.0		333	333		
EDS2/CEV	Launch Weather Delay	48	384	16.0		381	381		
EDS2/CEV	Launch from Earth/Loiter	2	386	16.1		383	383	2	2
EDS2/CEV	Loiter in LEO	1	387	16.1		384	384	3	3
EDS2	Earth Orbit Departure	0	387	16.1		384	384	3	3
EDS2/CEV	Coast	47	434	18.1		431	431	50	50
EDS2	MCC & EDS Disposal	0	434	18.1		431	431	50	50
CEV	Coast	47	481	20.0		478	478		97
CEV	Libration Point Arrival	0	481	20.0		478	478		97
CEV	Dock w/ Lander	6	487	20.3		484	484		103
CEV/Kick Stage/Lander	Crew Transfer & Checkout	24	511	21.3		508	508		127
Kick Stage/Lander	Undock from CEV	0	511	21.3		508	508		127
Kick Stage	Libration Point Departure	0	511	21.3		508	508		127
Kick Stage/Lander	Coast	60	571	23.8		568	568		187
Kick Stage	Lunar Orbit Insertion	0	571	23.8		568	568		187
Kick Stage	Kick Stage Disposal	0	571	23.8		568	568		187
Lander	Powered Descent	0	571	23.8			568		187
Lander	Surface Mission	168	739	30.8			736		355
Lander	Ascent	0	739	30.8			736		355
Lander	Lunar Orbit Departure	0	739	30.8			736		355
Lander	Coast	60	799	33.3			796		415
Lander	Libration Point Arrival	0	799	33.3			796		415
Lander	Rendezvous & Dock w/ CEV	6	805	33.5			802		421
Lander/CEV	Crew Transfer & Checkout	24	829	34.5			826		445
CEV	Undock from Lander	0	829	34.5			826		445
Lander	Ascent Stage Disposal	0	829	34.5			826		445
CEV	Libration Point Departure	0	829	34.5					445

Vehicle	Phase Name	Phase Length (hr)	Mission Elapsed Time: Overall MET (hr)	Overall MET (days)	EDS1 (hr)	Kick Stage (hr)	Lander (hr)	EDS2 (hr)	CEV (hr)
CEV	Coast	91	920	38.3					536
CEV	Dispose Service Module	0	920	38.3					536
CEV	Coast & Entry	3	923	38.5					539
CEV	Recovery	1	924	38.5					540

Table 11.2-1: Mission Phase Description

11.3 Safety & Mission Success

The 2-Launch Solution approach only differs from the TRM during the uncrewed portions of the mission. The TRM identified twenty uncrewed critical events while nine uncrewed critical events were identified for the 2-Launch Solution. The crewed phases of the 2-Launch Solutions and the TRM are nearly identical to each other. The TRM identified thirty-six crewed critical events while the 2-Launch Solution identified thirty-five crewed critical events. The difference is that with the 2-Launch Solution approach, the docking between the CEV and EDS-2 does not occur on-orbit rather the elements are mated together prior to launch. The TRM launches the CEV and EDS-2 separately and then the spacecraft dock with each other while in low Earth orbit.

Of the forty-four total critical events identified for the 2-Launch Solution, seven received a ranking of three, twenty-four received a ranking of two, and the remaining thirteen received a ranking of one. The complete set of identified and ranked critical events for the 2-Launch Solution is listed in the table below.

	ID #	TRM w/2-Launch Solution Critical Events	TRM w/2-Launch Solution Critical Event Rank
Uncrewed Critical Events	VAR-01-01	EDS-1, Kickstage, & LL Launch	1
	VAR-01-02	EDS-1, Kickstage, & LL Ascent	1
	VAR-01-03	EDS-1, Kickstage, & LL Launch Shroud Separation	1
	VAR-01-04	EDS-1, Kickstage, & LL Separation from Booster	1
	VAR-01-05	EDS-1, Kickstage, & LL Orbital Maneuvering	1
	VAR-01-06	EDS-1, Kickstage, & LL Burn for L1	1
	VAR-01-07	LL & Kickstage Separate from EDS-1	1
	VAR-01-08	Kickstage, & LL Mid-course Correction Burn	1
	VAR-01-09	Kickstage, & LL Burn to Slow Near L1	1
Crewed Critical Events	VAR-01-10	EDS-2 & CEV (CM+SM) Launch	2
	VAR-01-11	EDS-2 & CEV Ascent	2
	VAR-01-12	LAS Separation	2

	ID #	TRM w/2-Launch Solution Critical Events	TRM w/2-Launch Solution Critical Event Rank
	VAR-01-13	EDS-2 & CEV Launch Shroud Separation	2
	VAR-01-14	EDS-2 & CEV Separation from Booster	2
	VAR-01-15	EDS-2 & CEV Orbital Maneuvering	2
	VAR-01-16	EDS-2 & CEV Burn for L1	2
	VAR-01-17	CEV Separates from EDS-2	2
	VAR-01-18	CEV Mid-course Correction Burn	1
	VAR-01-19	CEV Burn to Slow Near L1	2
	VAR-01-20	CEV Orbital Maneuvering	2
	VAR-01-21	CEV Docks to LL & Kickstage	2
	VAR-01-22	Crew Transfers from CEV to LL	1
	VAR-01-23	LL & Kickstage Separates from CEV	2
	VAR-01-24	LL & Kickstage Burns for Low Lunar Orbit	2
	VAR-01-25	LL & Kickstage Mid-course Correction Burn	2
	VAR-01-26	LL & Kickstage Lunar Orbit Insertion (LOI)	2
	VAR-01-27	Kickstage Separates from LL	2
	VAR-01-28	LL Deorbit Burn to Moon	2
	VAR-01-29	LL Powered Descent & Landing on Moon	3
	VAR-01-30	LL Ascent Stage Separation & Ascent	3
	VAR-01-31	LL Ascent Stage Orbital Maneuvering	3
	VAR-01-32	LL Ascent Stage Lunar Orbit Departure	3
	VAR-01-33	LL Ascent Stage Mid-Course Correction Burn	1
	VAR-01-34	LL Ascent Stage Arrival L1 Arrival	3
	VAR-01-35	LL Ascent Stage Orbital Maneuvering	2
	VAR-01-36	LL Ascent Stage Docks with CEV	2
	VAR-01-37	Crew Transfers from LL to CEV	2
	VAR-01-38	CEV Separates from LL Ascent Stage	2
	VAR-01-39	CEV Burn for Earth	3
	VAR-01-40	CEV Mid-course Correction Burn	1
	VAR-01-41	CM Separates & Maneuvers away from SM	2
	VAR-01-42	CM Entry	3
	VAR-01-43	CM Landing	2
	VAR-01-44	Crew Recovery	2

Table 11.3-1: 2-Launch Solution Critical Events and Ranking

In terms of mission success, the 2-Launch Solution reduces the total number of critical events. Reducing the total number of launches from four to two, the number of dockings from four to two, and the number of separations from eleven to nine inherently increases the likelihood of achieving mission success. However, launching the crew and CEV mated with the EDS-2 may decrease the level of crew safety. Launching the crew, CEV, and EDS-2 together exposes the crew to more risk because of the size of the launch vehicle required. The larger launch vehicle will carry a significant increase in propellant for the launcher, possibly increasing the risk of fire and explosion, and may increase the risk due to launcher engine failures. Abort and crew escape may be significantly more difficult from the larger launch vehicle increasing the probability of loss of the crew. Potential mitigations for the aforementioned hazards and risks could include a very reliable Integrated Vehicle Health Monitoring (IVHM) system, a pressurized volume designed for the crew to withstand a certain level of blast overpressure, and a very reliable full-coverage crew escape system (CES).

11.4 Mission Abort Options

As the Crew Exploration Vehicle functions identically following Earth orbit departure for the two options under consideration, mission aborts are unaffected by changing the TRM from a four launch per mission to a two launch per mission architecture.

11.5 Element Overview & Mass Properties

This section describes any changes made in sizing the trade reference mission elements and compares the resulting vehicle mass properties. The total architecture mass for the two launch per mission architecture variant is estimated at 216 metric tons, a 14 metric ton savings from the TRM.

11.5.1 Crew Exploration Vehicle

For the 2-Launch Solution variant, the CEV launches with the Earth Departure Stage, whereas with the TRM, those two elements launch separately and assemble in LEO. Therefore, the extra 109 hr of on-orbit time (weather delays, rendezvous, and checkout) and rendezvous & mating propellant required previously is unnecessary here, which should reduce CEV initial mass in LEO by reducing crew provisions, propellant quantity, and propellant tank size.

The only other modification made to the TRM CEV is that the vehicle now requires a second interface on the aft end (Service Module side) for attaching to the Earth Departure Stage. The TRM assumed that the CEV would dock to the EDS using the same interface used for docking and transferring to the Lunar Lander, the low impact docking system located on the forward end (Crew Module side) of the vehicle. Since the CEV is now launching with the EDS, launch abort considerations dictate that the CEV be stacked above the EDS for launch, therefore for the assumed CEV capsule shape, the CEV needs a second interface through the Service Module.

CEV Crew Module's System Mass Changes				
System	**TRM**	**2-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	815	(7)	(0.9)
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	680	(11)	(1.6)
Other	835	832	(3)	(0.4)
Growth	1041	1037	(4)	(0.4)
Non-Cargo	966	916	(50)	(5.2)
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	52	(3)	(5.5)
Propellant	64	64	No Change	0.0
Total	8812	8735	(77)	(0.9)

Table 11.5.1-1: Variation in CEV CM Mass with 2-Launch Solution

CEV Service Module's System Mass Changes				
System	**TRM**	**2-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	1455	1437	(18)	(1.2)
Protection	0	0	No Change	0.0
Propulsion	1408	1272	(136)	(9.7)
Power	661	624	(37)	(5.6)
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	708	(39)	(5.2)
Non-Cargo	305	256	(49)	(16.1)
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1171	(271)	(18.8)
Propellant	11332	9512	(1820)	(16.1)
Total	17560	15190	(2370)	(13.5)

Table 11.5.1-2: Variation in CEV SM Mass with 2-Launch Solution

11.5.2 Lunar Lander

The Lunar Lander, like the CEV, now launches with its Earth Departure Stage attached in the 2-Launch Solution instead of launching separately and mating in LEO. The 2-Launch Solution also eliminates two additional launches required in the TRM. These changes eliminate any propellant previously required for rendezvous and docking in LEO and 640 hr of extra propellant boiloff in the Ascent and Descent Stages.

Lander's Ascent Stage's System Mass Changes				
System	**TRM**	**2-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	839	839	No Change	0.0
Protection	73	73	No Change	0.0
Propulsion	1631	1621	(10)	(0.6)
Power	813	813	No Change	0.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	851	No Change	0.0
Other	455	455	No Change	0.0
Growth	1080	1078	(2)	(0.2)
Non-Cargo	1483	1482	(1)	(0.1)
Cargo	227	227	No Change	0.0
Non-Propellant	1014	1014	No Change	0.0
Propellant	10703	10688	(15)	(0.1)
Total	19906	19879	(27)	(0.1)

Table 11.5.2-1: Variation in Lander Ascent Stage Mass with 2-Launch Solution

Lander's Descent Stage's System Mass Changes				
System	**TRM**	**2-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	553	553	No Change	0.0
Protection	50	50	No Change	0.0
Propulsion	1413	1403	(10)	(0.7)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	707	(1)	(0.1)
Growth	678	676	(2)	(0.3)
Non-Cargo	464	462	(2)	(0.4)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	17543	(30)	(0.2)
Total	22608	22562	(46)	(0.2)

Table 11.5.2-2: Variation in Lander Descent Stage Mass with 2-Launch Solution

11.5.3 Kick Stage

The only change made to the Kick Stage in the 2-Launch Solution is its on-orbit lifetime, which like the Lunar Lander reduces the lifetime by 640 hr. This results in less propellant boiloff and a lower IMLEO.

Kick Stage's System Mass Changes				
System	**TRM**	**2-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	621	615	(6)	(1.0)
Protection	0	0	No Change	0.0
Propulsion	1530	1492	(38)	(2.5)
Power	100	100	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	0	0	No Change	0.0
Other	405	405	No Change	0.0
Growth	531	522	(9)	(1.7)
Non-Cargo	953	750	(203)	(21.3)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	23323	23259	(64)	(0.3)
Total	27465	27143	(322)	(1.2)

Table 11.5.3-1: Variation in Kick Stage Mass with 2-Launch Solution

11.5.4 Earth Departure Stage

The most significant vehicle modifications made for the 2-Launch Solution were made to the CEV and Lunar Lander Earth Departure Stages. In the TRM, the stages are required to maintain vehicle attitude and transmit health status to Earth for weeks prior to mating in LEO. Therefore, an extensive suite of avionics, thermal control, power generation, and attitude control equipment are included on the vehicle to meet this requirement. By launching with its payload attached and directly injecting to L1, the EDS design can either eliminate this functionality or have it provided by the CEV or Lander. This change greatly simplifies the EDS design and reduces the vehicle's inert mass. Further, the elimination of separate launches for the CEV and Lander reduces the EDS lifetime to less than 50 hr, which eliminates a substantial fraction of the propellant boiloff factored into the TRM.

Lander Earth Departure Stage's System Mass Changes				
System	**TRM**	**2-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	1972	1881	(91)	(4.6)
Protection	0	0	No Change	0.0
Propulsion	4361	3834	(527)	(12.1)
Power	190	106	(84)	(44.2)
Control	0	0	No Change	0.0
Avionics	175	0	(175)	(100.0)
Environment	105	0	(105)	(100.0)
Other	455	100	(355)	(78.0)
Growth	1452	1184	(268)	(18.5)
Non-Cargo	3109	2294	(815)	(26.2)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	82289	80191	(2098)	(2.5)
Total	94109	89591	(4518)	(4.8)

Table 11.5.4-1: Variation in Lander EDS Mass with 2-Launch Solution

CEV Earth Departure Stage's System Mass Changes				
System	**TRM**	**2-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	932	799	(133)	(14.3)
Protection	0	0	No Change	0.0
Propulsion	2318	2001	(317)	(13.7)
Power	190	106	(84)	(44.2)
Control	0	0	No Change	0.0
Avionics	171	0	(171)	(100.0)
Environment	104	0	(104)	(100.0)
Other	455	100	(355)	(78.0)
Growth	834	601	(233)	(27.9)
Non-Cargo	1355	847	(508)	(37.5)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	32897	28853	(4044)	(12.3)
Total	39256	33306	(5950)	(15.2)

Table 11.5.4-2: Variation in CEV EDS Mass with 2-Launch Solution

11.5.5 Vehicle Mass Properties

Table 11.5.5-1 lists vehicle mass properties for the two launch per mission architecture variant, and Figure 11.5.5-1 compares individual vehicle gross mass to the trade reference mission. The largest single element, the Earth Departure Stage for the Lunar Lander and Kick Stage, has an initial mass in LEO of 89.6 t as compared to 94.1 t for the TRM. This savings is largely due to the reduction in vehicle hardware mass and propellant boiloff for the stage. Next, the Kick Stage and Lunar Lander have initial masses of 27.1 t and 42.4 t, respectively, slightly less than the 27.5 t and 42.5 t in the TRM. The CEV Earth Departure Stage is reduced by 5.9 t below the previous estimate of 39.3 t, again reflecting the elimination of EDS hardware components for direct injection architectures. Finally, the CEV total mass is estimated at 23.9 t, a reduction of 2,400 kg. The combined architecture elements of the 2-Launch Solution architecture variant have a total IMLEO of 216 t, compared to 230 t for the trade reference mission.

	CEV CM	CEV SM	CEV Earth Dep. Stage	Ascent Stage	Descent Stage	Kick Stage	Lander Earth Dep. Stage
1.0 Structure	1,523	1,437	799	839	553	615	1,881
2.0 Protection	815	0	0	73	50	0	0
3.0 Propulsion	117	1,272	2,001	1,621	1,403	1,492	3,834
4.0 Power	482	624	106	813	137	100	106
5.0 Control	0	0	0	0	0	0	0
6.0 Avionics	737	0	0	738	0	0	0
7.0 Environment	680	110	0	851	530	0	0
8.0 Other	832	100	100	455	707	405	100
9.0 Growth	1,037	708	601	1,078	676	522	1,184
DRY MASS	6,224 kg	4,251 kg	3,607 kg	6,468 kg	4,057 kg	3,134 kg	7,106 kg
10.0 Cargo	916	256	847	1,482	462	750	2,294
11.0 Non-Cargo	1,478	0	0	227	500	0	0
INERT MASS	8,619 kg	4,506 kg	4,454 kg	8,177 kg	5,019 kg	3,884 kg	9,399 kg
12.0 Non-Propellant	52	1,171	0	1,014	0	0	0
13.0 Propellant	64	9,512	28,853	10,688	17,543	23,259	80,191
GROSS MASS	8,735 kg	15,190 kg	33,306 kg	19,879 kg	22,562 kg	27,143 kg	89,591 kg

Table 11.5.5-1: Vehicle Mass Properties

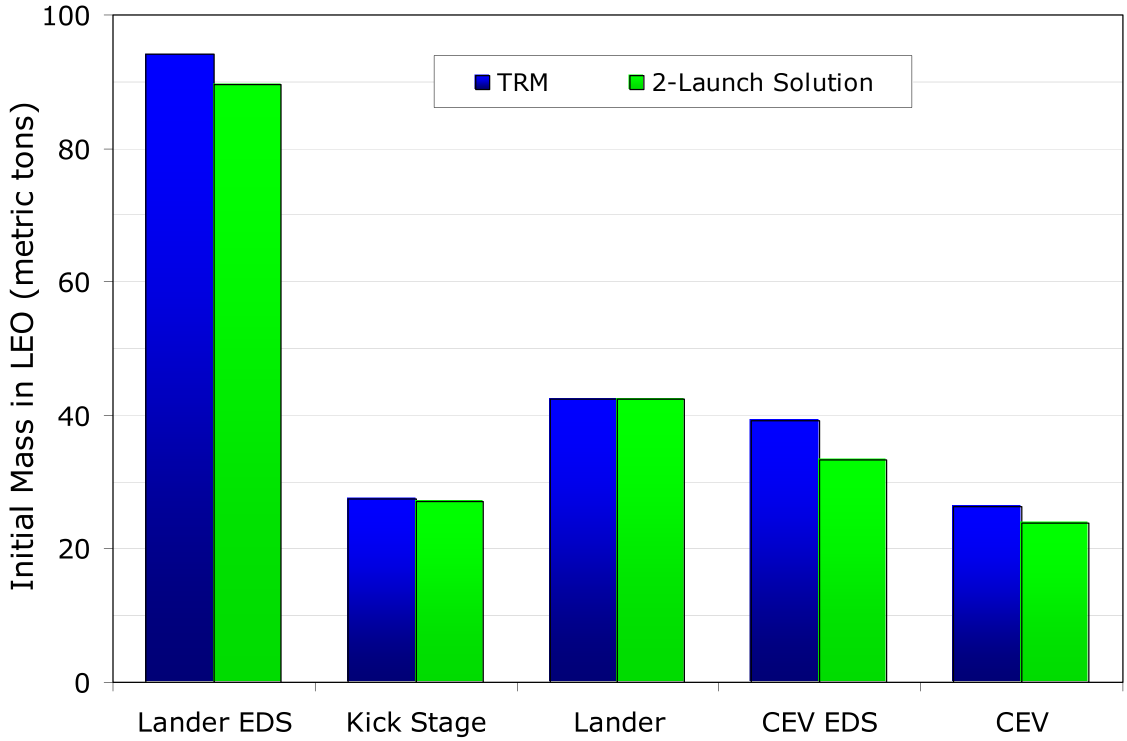


Figure 11.5.5-1: Vehicle Mass Properties Comparison

Figure 11.5.5-2 compares individual launch package masses to the trade reference mission. This architecture variant requires a 160 t payload capability by the cargo launch vehicle to deliver the Lunar Lander, Kick Stage, and Earth Departure Stage to LEO. The TRM, on the other hand, only required a maximum capability of 94 t as the EDS launched separately. As for the human-rated launch vehicle required by the CEV, its maximum size increases to 57 t to accommodate both an EDS and CEV in one launch. This is an increase from the 26 t, CEV-only capability with the TRM.

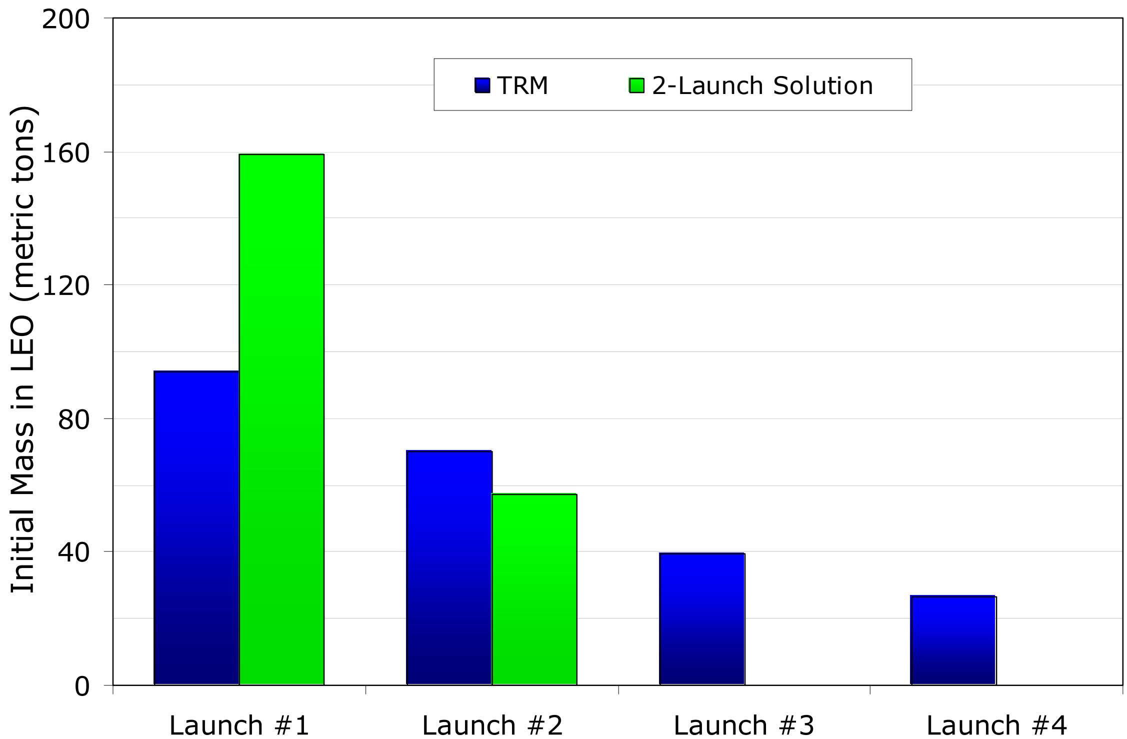


Figure 11.5.5-2: Mass per Launch Comparison

11.6 System Technologies and Programmatic Risks

The analysis performed for this architecture variant did not modify any vehicle system technology assumptions made for the trade reference mission.

11.7 Pros/Cons Summary

Reducing the number of launches per mission from four to two affords a number of benefits to the trade reference mission. By eliminating the need to assemble elements in LEO, architecture elements can launch from Earth and immediately inject to L1. Two distinct opportunities for launch and departure will be available in any 25-hr period during the year. With LEO assembly as in the TRM, the CEV and Lander/Kick Stage must each launch within a 3-day window to rendezvous and dock with the EDS, and then immediately perform Earth orbit departure to meet the first available departure opportunity. If that window is missed, the vehicles must loiter in LEO for 3-12 days for the next opportunity. Avoiding such complications with direct injection could be very beneficial in cases of extended launch delays due to weather or launch vehicle problems.

The flexibility offered by the 2-Launch Solution further reduces mission duration, mission risk, and total architecture mass. Without rendezvous and mating in LEO for the CEV and Lunar Lander/Kick Stage, the total crew time in space and CEV lifetime decreases by 4.5 days. The

EDS lifetime drops from 4 weeks to less than 50 hr, and the Lander and Kick Stage loiter time at Lunar L1 decreases by 18 days. Reducing vehicle lifetime, number of launches, and number of dockings will likely improve probability of mission success.

Direct injection will also simplify the Earth Departure Stage design. In the TRM, the EDS requires extensive power generation, avionics, active thermal control, and on-orbit mating equipment for assembling with the CEV and Lander in LEO. Integrating those elements on the ground eliminates much of that equipment or transfers the functionality to the CEV or Lander. This decreases EDS inert mass, which in turn reduces the quantity of propellant needed to execute Earth orbit departure. Combining these savings with the elimination of 18 crew-days in the CEV, CEV rendezvous propellant for docking with the EDS, and less propellant boil-off with shorter on-orbit lifetimes, the 2-Launch Solution variant has a total architecture mass of ~13 t less than the four-launch TRM for a savings of 5.8%.

Conversely, performing the TRM with only two launches per mission requires a cargo launch vehicle capable of delivering 159 t of payload to LEO in a single launch, as opposed to the 94 t requirement for the four-launch TRM. Such a requirement is far beyond the performance of any launch vehicle developed to date (the most capable launcher, the Saturn V, could deliver ~135 t to LEO) and exceeds the capability of reasonable Shuttle- or EELV-derived launch vehicle concepts.

12.0 TRM with 3-Launch Solution

This first architecture variant examines the impact of changing the number of launches required per mission. Instead of the four launches required for the trade reference mission, this variant launches all architecture elements in three separate launches. This section of the report examines the impact of such a change.

12.1 Major Assumptions/Clarifications

The "TRM with 3-Launch Solution" architecture variant affects the following TRM assumptions from the original LDRM-2 task request statement. Assumptions from Section 10.0 not explicitly listed here are still applicable to the architecture.

4-launch solution: This variant requires only three launches per mission. The first two launches of the architecture are identical to the trade reference mission. The third launch combines the CEV and CEV Earth Departure Stage.

Crew must be launched on a human-rated launch system: This assumption is still valid, however in the three launch per mission architecture, the CEV will now launch with its Earth Departure Stage which may affect the human rating of the system.

12.2 Architecture Description

The three launch per mission architecture seen in Figure 12.2-1 begins identically to the TRM with the launch of the Lander Earth Departure Stage followed two weeks later by the Lunar Lander and Kick Stage. The elements mate in LEO as in the TRM and depart for Lunar L1. Where the 3-Launch Solution differs is two weeks after the Lunar Lander and Kick Stage initially launch, instead of launching separately, the CEV and its Earth Departure Stage launch to the LEO parking orbit as a single combined element on a human-rated launch vehicle. With the 2-Launch Solution architecture variant, the CEV and EDS loiter in LEO up to 3 hr for vehicle checkout before the EDS performs the Earth orbit departure maneuver. The primary advantage of the 3-Launch Solution over the 2-Launch Solution is that it divides its largest elements, the EDS and Lander/Kick Stage into two separate launches instead of one. As seen in Section 11.0, combining these elements into one launch requires a cargo launch vehicle capable of delivering 160 metric tons to LEO. This variant divides the pre-deployed cargo elements into more manageable launch packages while retaining the benefits of direct injection for the crewed mission phases.

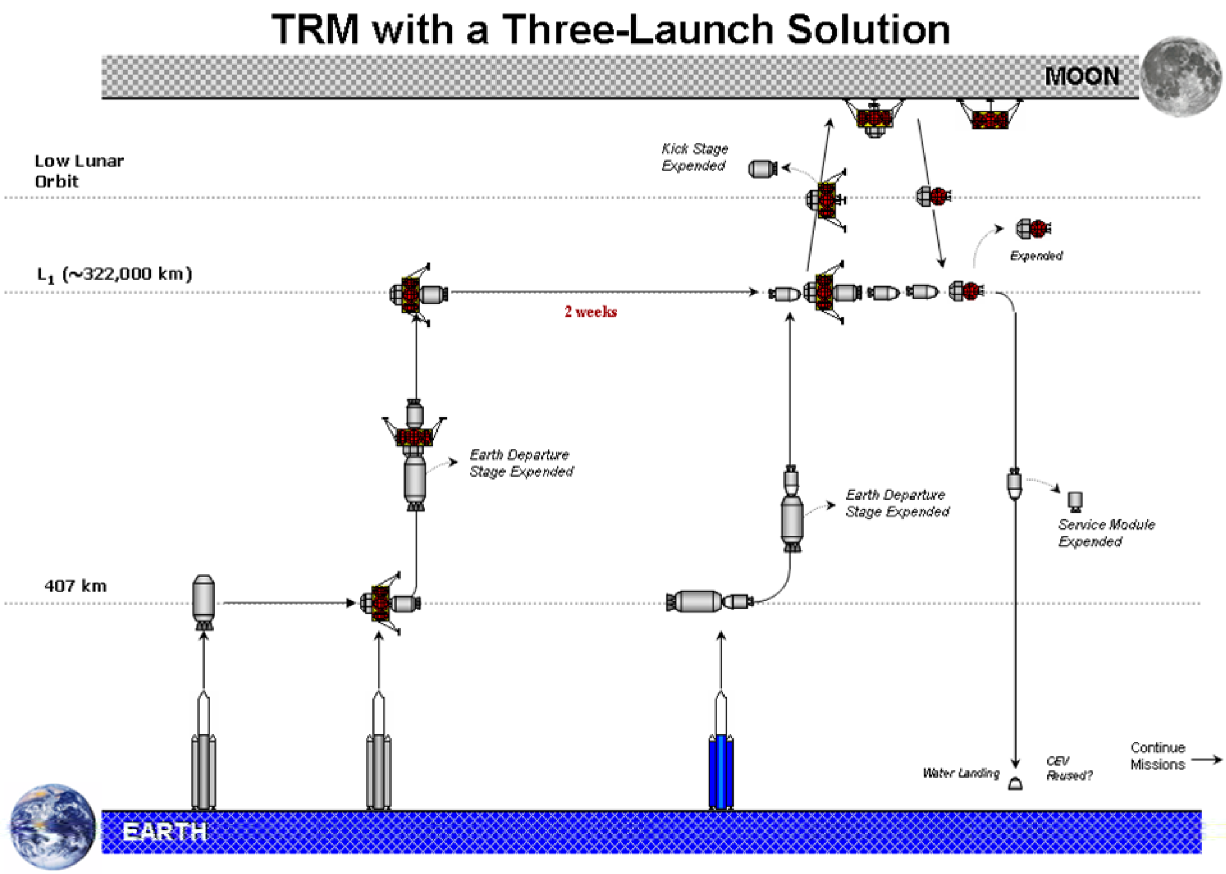


Figure 12.2-1: TRM with 3-Launch Solution Architecture Illustration

Once the CEV EDS completes Earth departure, the remainder of the mission functions identically to the trade reference mission. Table 12.2-1 outlines the assumed timelines for the three launch per mission architecture as just described.

Vehicle	Phase Name	Phase Length	Mission Elapsed Time						
			Overall MET		EDS1	Kick Stage	Lander	EDS2	CEV
		(hr)	(hr)	(days)	(hr)				
EDS1	Launch from Earth/Loiter	2	2	0.1	2				
EDS1	Loiter in LEO	332	334	13.9	334				
Kick Stage/Lander	Launch Weather Delay	48	382	15.9	382	48	48		
Kick Stage/Lander	Launch from Earth/Loiter	2	384	16.0	384	50	50		
Kick Stage/Lander	Rendezvous & Dock w/ EDS	50	434	18.1	434	100	100		
EDS1/Kick Stage/Lander	Vehicle Checkout	12	446	18.6	446	112	112		
EDS1/Kick Stage/Lander	Missed EOD Opportunity	240	686	28.6	686	352	352		
EDS1	Earth Orbit Departure	0	686	28.6	686	352	352		
EDS1/Kick Stage/Lander	Coast	47	733	30.5	733	399	399		
EDS1	MCC & EDS Disposal	0	733	30.5	733	399	399		
Kick Stage/Lander	Coast	47	780	32.5		446	446		
Kick Stage/Lander	Libration Point Arrival	0	780	32.5		446	446		
Kick Stage/Lander	Loiter at L1	130	910	37.9		576	576		
EDS2/CEV	Launch Weather Delay	48	958	39.9		624	624		
EDS2/CEV	Launch from Earth/Loiter	2	960	40.0		626	626	2	2
EDS2/CEV	Loiter in LEO	1	961	40.0		627	627	3	3
EDS2	Earth Orbit Departure	0	961	40.0		627	627	3	3
EDS2/CEV	Coast	47	1008	42.0		674	674	50	50
EDS2	MCC & EDS Disposal	0	1008	42.0		674	674	50	50
CEV	Coast	47	1055	44.0		721	721		97
CEV	Libration Point Arrival	0	1055	44.0		721	721		97
CEV	Dock w/ Lander	6	1061	44.2		727	727		103
CEV/Kick Stage/Lander	Crew Transfer & Checkout	24	1085	45.2		751	751		127
Kick Stage/Lander	Undock from CEV	0	1085	45.2		751	751		127
Kick Stage	Libration Point Departure	0	1085	45.2		751	751		127
Kick Stage/Lander	Coast	60	1145	47.7		811	811		187
Kick Stage	Lunar Orbit Insertion	0	1145	47.7		811	811		187
Kick Stage	Kick Stage Disposal	0	1145	47.7		811	811		187
Lander	Powered Descent	0	1145	47.7			811		187
Lander	Surface Mission	168	1313	54.7			979		355
Lander	Ascent	0	1313	54.7			979		355
Lander	Lunar Orbit Departure	0	1313	54.7			979		355
Lander	Coast	60	1373	57.2			1039		415
Lander	Libration Point Arrival	0	1373	57.2			1039		415

Vehicle	Phase Name	Phase Length	Mission Elapsed Time						
			Overall MET		EDS1	Kick Stage	Lander	EDS2	CEV
		(hr)	(hr)	(days)	(hr)				
Lander	Rendezvous & Dock w/ CEV	6	1379	57.5			1045		421
Lander/CEV	Crew Transfer & Checkout	24	1403	58.5			1069		445
CEV	Undock from Lander	0	1403	58.5			1069		445
Lander	Ascent Stage Disposal	0	1403	58.5			1069		445
CEV	Libration Point Departure	0	1403	58.5					445
CEV	Coast	91	1494	62.3					536
CEV	Dispose Service Module	0	1494	62.3					536
CEV	Coast & Entry	3	1497	62.4					539
CEV	Recovery	1	1498	62.4					540

Table 12.2-1: Mission Phase Description

12.3 Safety & Mission Success

Like the 2-Launch Solution approach, the 3-Launch Solution approach differs significantly from the TRM during the uncrewed and slightly during the crewed portions of the mission. The TRM identified twenty uncrewed critical events while a total of fifteen were identified for the 3-Launch Solution. Unlike the TRM, the 3-Launch Solution approach launches the CEV and EDS-2 spacecraft together instead of separately. The crewed phases of the 3-Launch Solution and the TRM are nearly identical to each other. The TRM identified thirty-six crewed critical events while the 3-Launch Solution identified thirty-five crewed critical events. Similar to the 2-Launch Solution approach, the 3-Launch Solution approach launches the CEV and EDS-2 spacecraft mated together on the ground. Thus, there is no on-orbit docking for the two elements. Whereas, the TRM launches the CEV and EDS-2 separately, and then the spacecraft docks with each other while in low Earth orbit (LEO).

Of the fifty total critical events identified for the 3-Launch Solution, seven received a ranking of three, twenty-four received a ranking of two, and the remaining nineteen received a ranking of one. The complete set of identified and ranked critical events for the 3-Launch Solution approach is listed in the table below.

	ID #	TRM w/3-Launch Solution Critical Events	TRM w/3-Launch Solution Critical Event Rank
Uncrewed Critical Events	VAR-02-01	EDS-1 (for the LL) Launch	1
	VAR-02-02	EDS-1 Ascent	1
	VAR-02-03	EDS-1 Launch Shroud Separation	1
	VAR-02-04	EDS-1 Separation from Booster	1

	ID #	TRM w/3-Launch Solution Critical Events	TRM w/3-Launch Solution Critical Event Rank
	VAR-02-05	EDS-1 Orbital Maneuvering	1
	VAR-02-06	LL & Kickstage Launch	1
	VAR-02-07	LL & Kickstage Ascent	1
	VAR-02-08	LL & Kickstage Launch Shroud Separation	1
	VAR-02-09	LL & Kickstage Separation from Booster	1
	VAR-02-10	LL & Kickstage Orbital Maneuvering	1
	VAR-02-11	LL & Kickstage Docks to EDS-1	1
	VAR-02-12	EDS-1, Kickstage, & LL Burn for L1	1
	VAR-02-13	LL & Kickstage Separates from EDS-1	1
	VAR-02-14	Kickstage, & LL Mid-course Correction Burn	1
	VAR-02-15	Kickstage, & LL Burn to Slow Near L1	1
Crewed Critical Events	**VAR-02-16**	EDS-2 & CEV (CM+SM) Launch	2
	VAR-02-17	EDS-2 & CEV Ascent	2
	VAR-02-18	LAS Separation	2
	VAR-02-19	EDS-2 & CEV Launch Shroud Separation	2
	VAR-02-20	EDS-2 & CEV Separation from Booster	2
	VAR-02-21	EDS-2 & CEV Orbital Maneuvering	2
	VAR-02-22	EDS-2 & CEV Burn for L1	2
	VAR-02-23	CEV Separates from EDS-2	2
	VAR-02-24	CEV Mid-course Correction Burn	1
	VAR-02-25	CEV Burn to Slow Near L1	2
	VAR-02-26	CEV Orbital Maneuvering	2
	VAR-02-27	CEV Docks to LL & Kickstage	2
	VAR-02-28	Crew Transfer from CEV to LL	1
	VAR-02-29	LL & Kickstage Separates from CEV	2
	VAR-02-30	LL & Kickstage Burns for Low Lunar Orbit	2
	VAR-02-31	LL & Kickstage Mid-course Correction Burn	2
	VAR-02-32	LL & Kickstage Lunar Orbit Insertion (LOI)	2
	VAR-02-33	Kickstage Separates from LL	2
	VAR-02-34	LL Deorbit Burn to Moon	2
	VAR-02-35	LL Powered Descent & Landing to Moon	3
	VAR-02-36	LL Ascent Stage Separation & Ascent	3
	VAR-02-37	LL Ascent Stage Orbital Maneuvering	3
	VAR-02-38	LL Ascent Stage Lunar Orbit Departure	3
	VAR-02-39	LL Ascent Stage Mid-Course Correction Burn	1
	VAR-02-40	LL Ascent Stage L1 Arrival	3
	VAR-02-41	LL Ascent Stage Orbital Maneuvering	2
	VAR-02-42	LL Ascent Stage Docks with CEV	2
	VAR-02-43	Crew Transfer from LL to CEV	2
	VAR-02-44	CEV Separates from LL Ascent Stage	2
	VAR-02-45	CEV Burn for Earth	3
	VAR-02-46	CEV Mid-Course Correction Burn	1
	VAR-02-47	CM Separates & Maneuvers away from SM	2

	ID #	TRM w/3-Launch Solution Critical Events	TRM w/3-Launch Solution Critical Event Rank
	VAR-02-48	CM Entry	3
	VAR-02-49	CM Landing	2
	VAR-02-50	Crew Recovery	2

Table 12.3-1: 3-Launch Solution Critical Events and Ranking

In terms of mission success, the 3-Launch Solution approach will be slightly better than the TRM since the 3-Launch Solution reduces the total number of critical events. Reducing the total number of launches from four to three, the number of dockings from four to three, and the number of separations from eleven to ten, inherently increases the likelihood of achieving mission success. However, as with the 2-Launch Solution, launching the crew and CEV mated with the EDS-2 may decrease the level of crew safety. Launching the crew, CEV, and EDS-2 spacecraft together exposes the crew to more risk because of the size of the launch vehicle required. The larger launch vehicle will carry a significant increase in propellant in the launcher, possibly increasing the risk of fire and explosion, and may increase the risk due to launcher engine failures (depending on the launcher configuration, possibly increasing the risk during engine out failures). Abort and crew escape may be significantly more difficult from the larger launch vehicle increasing the probability of loss of the crew. Potential mitigations for the aforementioned hazards and risks could include a very reliable Integrated Vehicle Health Monitoring (IVHM) system, a pressurized volume designed for the crew to withstand a certain level of blast overpressure, and a very reliable full-coverage crew escape system (CES).

12.4 Mission Abort Options

As the Crew Exploration Vehicle functions identically following Earth orbit departure for the two options under consideration, mission aborts are unaffected by changing the TRM from a four launch per mission to a three launch per mission architecture.

12.5 Element Overview & Mass Properties

This section describes any changes made in sizing the trade reference mission elements and compares the resulting vehicle mass properties. The total architecture mass for the three launch per mission architecture variant is estimated at 220 t, a 10 t savings from the TRM.

12.5.1 Crew Exploration Vehicle

For the 3-Launch Solution variant, the CEV launches with the Earth Departure Stage, whereas with the TRM, those two elements launch separately and assemble in LEO. Therefore, the extra 109 hr of on-orbit time (weather delays, rendezvous, and checkout) and rendezvous & mating propellant required previously is unnecessary here, which should reduce CEV initial mass in LEO by reducing crew provisions, propellant quantity, and propellant tank size.

The only other modification made to the TRM CEV is that the vehicle now requires a second interface on the aft end (Service Module side) for attaching to the Earth Departure Stage. The TRM assumed that the CEV would dock to the EDS using the same interface used for docking and transferring to the Lunar Lander, the low impact docking system located on the forward end (Crew Module side) of the vehicle. Since the CEV is now launching with the EDS, launch abort considerations dictate that the CEV be stacked above the EDS for launch, therefore for the assumed CEV capsule shape, the CEV needs a second interface through the Service Module.

CEV Crew Module's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	815	(7)	(0.9)
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	680	(11)	(1.6)
Other	835	832	(3)	(0.4)
Growth	1041	1037	(4)	(0.4)
Non-Cargo	966	916	(50)	(5.2)
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	52	(3)	(5.5)
Propellant	64	64	No Change	0.0
Total	8812	8735	(77)	(0.9)

Table 12.5.1-1: Variation in CEV CM Mass with 3-Launch Solution

CEV Service Module's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	1455	1437	(18)	(1.2)
Protection	0	0	No Change	0.0
Propulsion	1408	1272	(136)	(9.7)
Power	661	624	(37)	(5.6)
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	708	(39)	(5.2)
Non-Cargo	305	256	(49)	(16.1)
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1171	(271)	(18.8)
Propellant	11332	9512	(1820)	(16.1)
Total	17560	15190	(2370)	(13.5)

Table 12.5.1-2: Variation in CEV SM Mass with 3-Launch Solution

12.5.2 Lunar Lander

The Lunar Lander differs from the TRM only in on-orbit lifetime. The TRM Lander has a total mission duration of 1,466 hr, which compares to 1,069 hr for the 3-Launch Solution. This difference is due to the elimination of a separate launch for the CEV Earth Departure Stage, and it results in slightly less propellant boiloff for the Ascent and Descent Stages.

Lander's Ascent Stage's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	839	839	No Change	0.0
Protection	73	73	No Change	0.0
Propulsion	1631	1625	(6)	(0.4)
Power	813	813	No Change	0.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	851	No Change	0.0
Other	455	455	No Change	0.0
Growth	1080	1079	(1)	(0.1)
Non-Cargo	1483	1483	No Change	0.0
Cargo	227	227	No Change	0.0
Non-Propellant	1014	1014	No Change	0.0
Propellant	10703	10695	(8)	(0.1)
Total	19906	19891	(15)	(0.1)

Table 12.5.2-1: Variation in Lander Ascent Stage Mass with 3-Launch Solution

Lander's Descent Stage's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	553	553	No Change	0.0
Protection	50	50	No Change	0.0
Propulsion	1413	1408	(5)	(0.4)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	708	No Change	0.0
Growth	678	677	(1)	(0.1)
Non-Cargo	464	463	(1)	(0.2)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	17556	(17)	(0.1)
Total	22608	22582	(26)	(0.1)

Table 12.5.2-2: Variation in Lander Descent Stage Mass with 3-Launch Solution

12.5.3 Kick Stage

Like the Lunar Lander, the only change made to the Kick Stage in the 3-Launch Solution is its mission duration, which is reduced by 397 hr. This results in less propellant boiloff and a lower IMLEO.

Kick Stage's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	621	615	(6)	(1.0)
Protection	0	0	No Change	0.0
Propulsion	1530	1492	(38)	(2.5)
Power	100	100	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	0	0	No Change	0.0
Other	405	405	No Change	0.0
Growth	531	522	(9)	(1.7)
Non-Cargo	953	750	(203)	(21.3)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	23323	23266	(57)	(0.2)
Total	27465	27151	(314)	(1.1)

Table 12.5.3-1: Variation in Kick Stage Mass with 3-Launch Solution

12.5.4 Earth Departure Stages

The Lander EDS is unchanged from the TRM.

The most significant vehicle modifications made for the 3-Launch Solution were made to the CEV Earth Departure Stage. In the TRM, the stage is required to maintain vehicle attitude and transmit health status to Earth for two weeks prior to mating in LEO. By launching with the CEV attached and directly injecting to L1, the EDS design can either eliminate this functionality or have it provided by the CEV. This change greatly simplifies the EDS design and reduces the vehicle's inert mass. Further, the elimination of a separate launch for the CEV reduces the EDS lifetime to less than 50 hr, which eliminates a substantial fraction of the propellant boiloff factored into the TRM.

Lander Earth Departure Stage's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	1972	1923	(49)	(2.5)
Protection	0	0	No Change	0.0
Propulsion	4361	4074	(287)	(6.6)
Power	190	190	No Change	0.0
Control	0	0	No Change	0.0
Avionics	175	175	No Change	0.0
Environment	105	105	No Change	0.0
Other	455	455	No Change	0.0
Growth	1452	1384	(68)	(4.7)
Non-Cargo	3109	3082	(27)	(0.9)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	82289	81481	(808)	(1.0)
Total	94109	92869	(1240)	(1.3)

Table 12.5.4-1: Variation in Lander EDS Mass with 3-Launch Solution

CEV Earth Departure Stage's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	932	799	(133)	(14.3)
Protection	0	0	No Change	0.0
Propulsion	2318	2001	(317)	(13.7)
Power	190	106	(84)	(44.2)
Control	0	0	No Change	0.0
Avionics	171	0	(171)	(100.0)
Environment	104	0	(104)	(100.0)
Other	455	100	(355)	(78.0)
Growth	834	601	(233)	(27.9)
Non-Cargo	1355	847	(508)	(37.5)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	32897	28853	(4044)	(12.3)
Total	39256	33306	(5950)	(15.2)

Table 12.5.4-2: Variation in CEV EDS Mass with 3-Launch Solution

12.5.5 Vehicle Mass Properties

Table 12.5.5-1 lists vehicle mass properties for the three launch per mission architecture variant, and Figure 12.5.5-1 compares individual vehicle gross mass to the trade reference mission. The largest single element, the Earth Departure Stage for the Lunar Lander and Kick Stage, has an initial mass in LEO of 92.9 t as compared to 94.1 t for the TRM. This savings is largely due to the reduction in Lunar Lander and Kick Stage mass. Next, the Kick Stage and Lunar Lander have initial masses of 27.1 t and 42.5 t, respectively, slightly less than the 27.5 t and 42.5 t in the TRM. The CEV Earth Departure Stage is reduced by 5.9 t below the TRM estimate of 39.3 t, reflecting the elimination of EDS hardware components for direct injection architectures. Finally, the CEV total mass is estimated at 23.9 t, a reduction of 2,400 kg. The combined architecture elements of the 3-Launch Solution architecture variant have a total IMLEO of 220 t, compared to 230 t for the trade reference mission.

	CEV CM	CEV SM	CEV Earth Dep. Stage	Ascent Stage	Descent Stage	Kick Stage	Lander Earth Dep. Stage
1.0 Structure	1,523	1,437	799	839	553	615	1,923
2.0 Protection	815	0	0	73	50	0	0
3.0 Propulsion	117	1,272	2,001	1,625	1,408	1,492	4,074
4.0 Power	482	624	106	813	137	100	190
5.0 Control	0	0	0	0	0	0	0
6.0 Avionics	737	0	0	738	0	0	175
7.0 Environment	680	110	0	851	530	0	105
8.0 Other	832	100	100	455	708	405	455
9.0 Growth	1,037	708	601	1,079	677	522	1,384
DRY MASS	6,224 kg	4,251 kg	3,607 kg	6,473 kg	4,063 kg	3,135 kg	8,306 kg
10.0 Cargo	916	256	847	1,483	463	750	3,082
11.0 Non-Cargo	1,478	0	0	227	500	0	0
INERT MASS	8,619 kg	4,506 kg	4,454 kg	8,183 kg	5,026 kg	3,885 kg	11,388 kg
12.0 Non-Propellant	52	1,171	0	1,014	0	0	0
13.0 Propellant	64	9,512	28,853	10,695	17,556	23,266	81,481
GROSS MASS	8,735 kg	15,190 kg	33,306 kg	19,891 kg	22,582 kg	27,151 kg	92,869 kg

Table 12.5.5-1: Vehicle Mass Properties

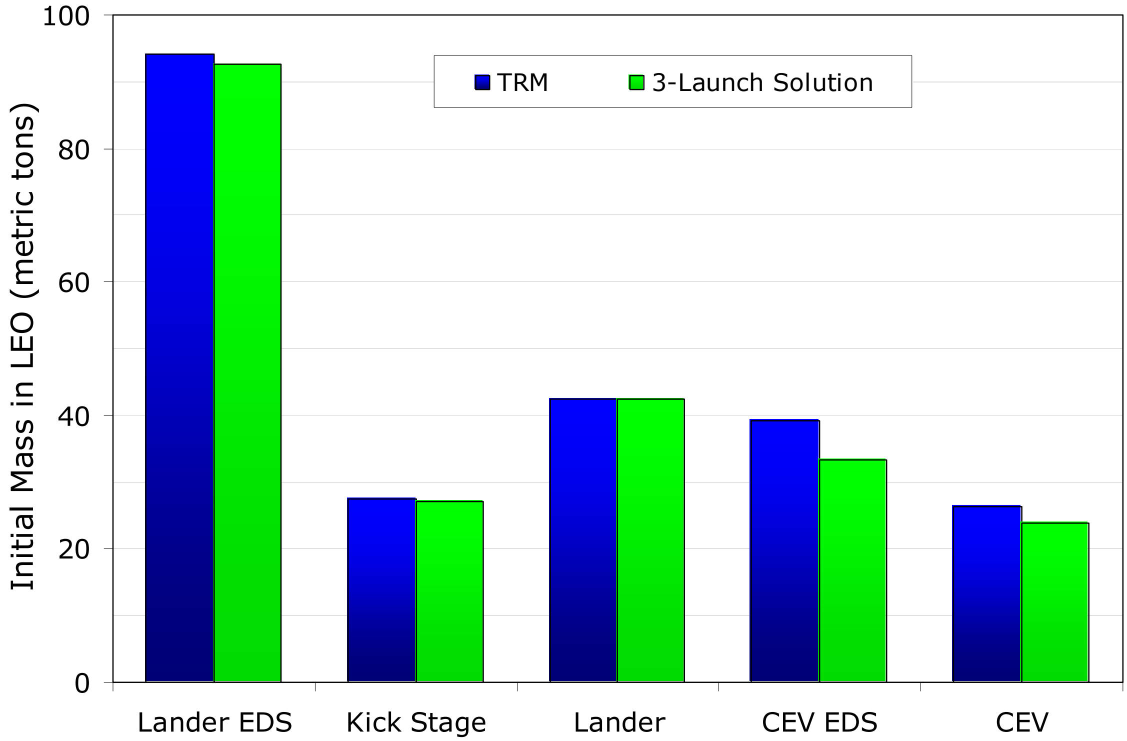


Figure 12.5.5-1: Vehicle Mass Properties Comparison

Figure 12.5.5-2 compares individual launch package masses to the trade reference mission. This architecture variant requires a 93 t payload capability by the cargo launch vehicle to deliver the Lander EDS to LEO, just slightly less than the 94 t required maximum capability for the TRM. As for the human-rated launch vehicle required by the CEV, its maximum size increases to 57 t to accommodate both an EDS and CEV in one launch. This is a significant increase from the 26 t, CEV-only capability with the TRM.

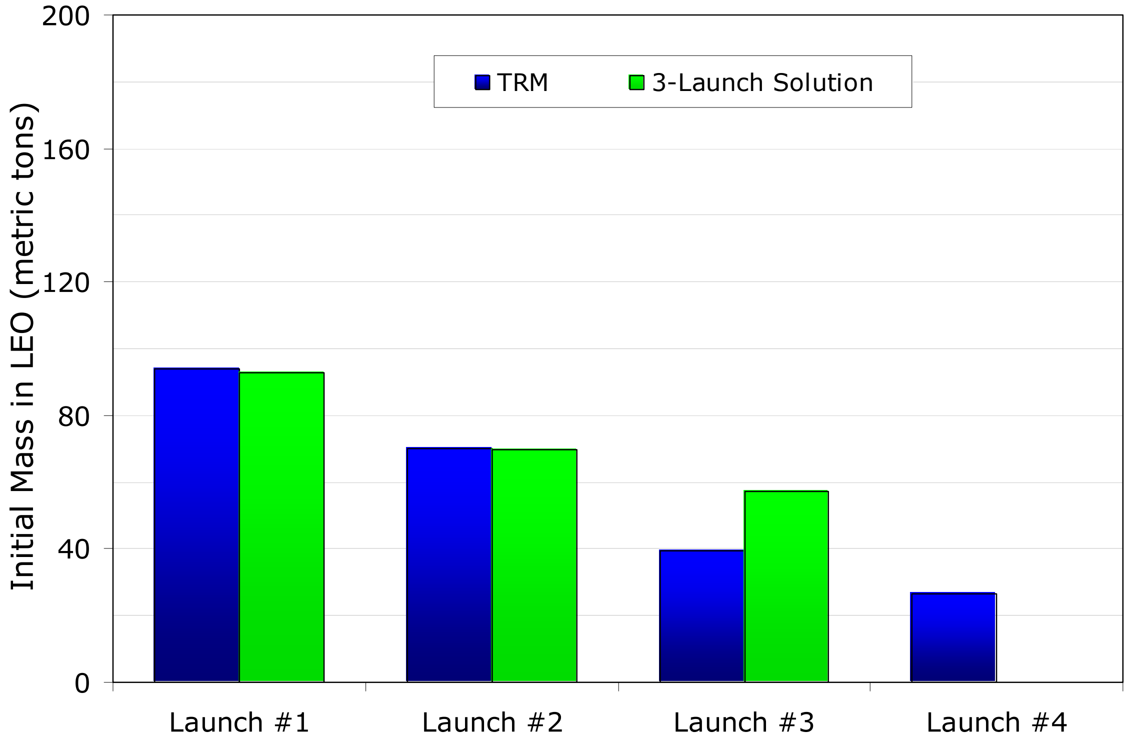


Figure 12.5.5-2: Mass per Launch Comparison

12.6 System Technologies and Programmatic Risks

The analysis performed for this architecture variant did not modify any vehicle system technology assumptions made for the trade reference mission.

12.7 Pros/Cons Summary

The primary benefit of reducing the number of launches per mission for the TRM from four to three that the 2-Launch Solution did not offer is the ability to retain the direct injection feature for the CEV without requiring a 159 t cargo launch vehicle. The 3-Launch Solution incorporates LEO assembly of the Lander/Kick Stage with its EDS to keep the maximum launch vehicle size below 100 t, as in the TRM. While losing direct injection capability for the Lander EDS will increase the mass and complexity of that stage by including dedicated avionics, power generation, on-orbit mating, and thermal control equipment that the 2-Launch variant does not require, direct injection is more valuable for the crewed phases of the architecture. If the Lander and Kick Stage miss the first L1 departure opportunity, the primary impact will be additional propellant boil-off while awaiting the next opportunity. However, if the CEV misses its first departure opportunity, the entire mission may be scrubbed rather than leaving the crew to loiter up to 12 days in LEO. A combined CEV/EDS launch affords two daily opportunities to launch and depart for L1.

Eliminating a dedicated launch for the CEV EDS reduces total crew time in space and CEV lifetime by 4.5 days, and decreases Kick Stage and Lunar Lander loiter time at L1 by 14 days. Shorter vehicle lifetimes, one fewer launch per mission, and no CEV rendezvous and mating maneuver in LEO should improve probability of mission success relative to the four-launch TRM. Combining these factors into the vehicle sizing, along with a simplified CEV EDS design, results in a total architecture mass for the 3-Launch Solution variant of ~10 t less than the four-launch TRM for a savings of 4.3%.

While this architecture does not affect cargo launch vehicle size relative to the TRM, it does require a significantly larger human-rated launch vehicle. Direct injection capability means that the CEV launches with the EDS which increases the size of that launcher to 57 t. This is a negative feature of the 3-Launch architecture as larger launch vehicles may be more expensive to operate, more difficult and costly to human-rate, and due to their explosive potential, may pose a greater hazard to the crew in a catastrophic launch failure.

13.0 TRM with 25 t Solution

This first architecture variant examines the impact of changing the number of launches required per mission. Instead of the four launches required for the trade reference mission, this variant assumes all architecture elements are limited to a 25 t launch package. This section of the report examines the impact of such a change.

13.1 Major Assumptions/Clarifications

The "TRM with 25 t Solution" architecture variant affects the following TRM assumptions from the original LDRM-2 task request statement. Assumptions from Section 10.0 not explicitly listed here are still applicable to the architecture.

<u>4-launch solution:</u> This variant places no restrictions on the number of launches per mission. Each launch is limited to a 25 t payload size.

13.2 Architecture Description

With the trade reference mission, architecture elements such as the Lunar Lander and Earth Departure Stages launched as single indivisible elements and the total mass of those elements determined the required cargo launch vehicle capabilities. This architecture variant assumes that future launch capabilities are limited to 25 t per launch, which is roughly equivalent to the heavy-lift versions of present day EELVs. Such a restriction will require that architecture elements divide into multiple launch packages to fit within the 25 t limit.

The 25 t per launch architecture seen in Figure 13.2-1 begins with the launch of the first of four consecutive equal-mass Lander Earth Departure Stages with two-week spacing between launches. These elements automatically rendezvous and mate at the reference LEO assembly orbit. For the TRM, the Lander Earth Departure Stage was a single-stage propulsive element with a total mass of 94 t. The EDS for this architecture variant is divided into four individual stages to duplicate the functionality of the TRM EDS with a 25 t launch limit. Two weeks after the fourth EDS initially launches and the stages have assembled in LEO, the Kick Stage launches and it docks with the assembled elements, followed two weeks thereafter by the Lander Descent Stage. The Lunar Lander is also too massive to be launched with a single vehicle, therefore its Ascent and Descent Stages are launched separately. Once the Ascent Stage launches and mates with the assembled stack, the vehicles are checked out and depart for L1 at the first available opportunity. Given the high likelihood of missing the planned departure opportunity, ten days of LEO loiter padding are held in reserve.

For Earth orbit departure, two of the four Earth departure stages initially ignite and burn their supply of propellant. These two stages then separate from the stack, dispose themselves, and the remaining two EDS's ignite to complete the maneuver. Because of this staging strategy and having equal-mass stages, the first two stages perform 40% of the total departure delta-V and the second pair completes the remaining 60%. This particular strategy was selected as some efficiency is gained by staging the EDS's rather than burning all propellant simultaneously like the

TRM EDS. However, introducing a separation event and engine start event during a critical maneuver like Earth orbit departure will add risk to the mission.

Two weeks following launch of the Ascent Stage, the first of two equal-mass Earth Departure Stages for the CEV is delivered to the LEO assembly orbit. The second stage then launches, once again with a two-week spacing, and mates with the first EDS loitering in orbit. Like the Lander EDS, the CEV EDS is divided into two separate elements to fit within the 25 t launch limit. The tenth and final launch for the mission contains the CEV and crew. As in the TRM, the CEV mates with the EDS in LEO and executes Earth orbit departure when the window opens. The vehicles also stage during the maneuver like the Lander EDS for mass efficiency.

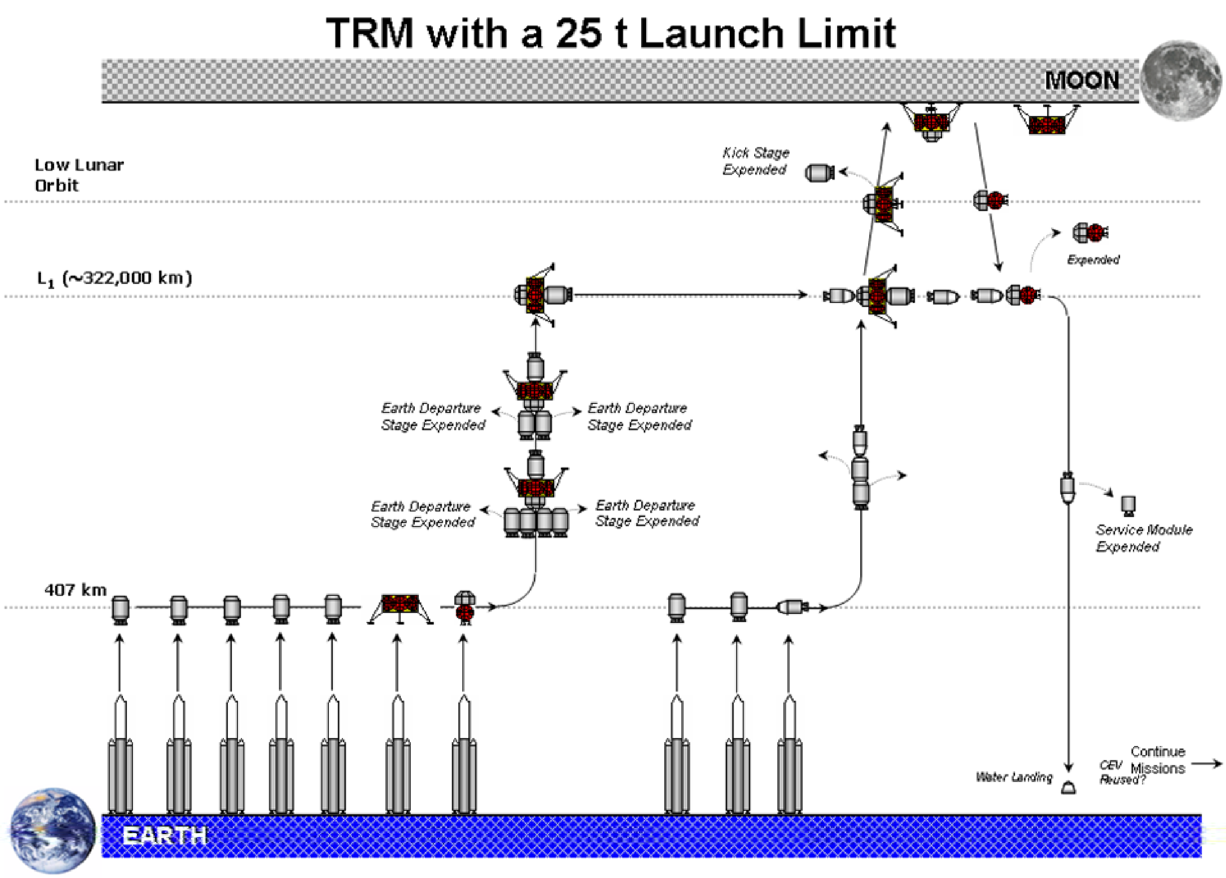


Figure 13.2-1: TRM with 25 t Solution Architecture Illustration

Once the second CEV EDS completes Earth departure, the remainder of the mission functions identically to the trade reference mission with one notable exception. Fitting within the 25 t launch limit for the Kick Stage requires that the Lander Descent Stage be used to execute part of lunar orbit insertion. For the 25 t Solution architecture, the Kick Stage performs the first 482 m/s of that maneuver and the Lander Descent Stage performs the remaining 150 m/s. This introduces another staging event to the mission.

Table 13.2-1 outlines the assumed timelines for the 25 t per launch mission architecture as just described.

Vehicle	Phase Name	Phase Length	Mission Elapsed Time						
			Overall MET		LL EDS	Kick Stage	Lander	CEV EDS	CEV
		(hr)	(hr)	(days)	(hr)				
EDS1	Launch from Earth/Loiter	2	2	0.1	2				
EDS1	Loiter in LEO	334	336	14.0	336				
EDS2	Launch from Earth/Loiter	2	338	14.1	338				
EDS2	Dock w/ EDS1	50	388	16.2	388				
EDS2	Loiter in LEO	284	672	28.0	672				
EDS3	Launch from Earth/Loiter	2	674	28.1	674				
EDS3	Dock w/ EDS's	50	724	30.2	724				
EDS3	Loiter in LEO	284	1008	42.0	1008				
EDS4	Launch from Earth/Loiter	2	1010	42.1	1010				
EDS4	Dock w/ EDS's	50	1060	44.2	1060				
EDS4	Loiter in LEO	284	1344	56.0	1344				
Kick Stage	Launch from Earth/Loiter	2	1346	56.1	1346	2			
Kick Stage	Dock w/ EDS's	50	1396	58.2	1396	52			
Kick Stage	Loiter in LEO	284	1680	70.0	1680	336			
Descent Stage	Launch from Earth/Loiter	2	1682	70.1	1682	338	2		
Descent Stage	Dock w/ EDS's	50	1732	72.2	1732	388	52		
Descent Stage	Loiter in LEO	284	2016	84.0	2016	672	336		
Ascent Stage	Launch from Earth/Loiter	2	2018	84.1	2018	674	338		
Ascent Stage	Dock w/ EDS's	50	2068	86.2	2068	724	388		
EDS's/Kick Stage/Lander	Vehicle Checkout	12	2080	86.7	2080	736	400		
EDS's/Kick Stage/Lander	Missed EOD Opportunity	240	2320	96.7	2320	976	640		
EDS's	Earth Orbit Departure	0	2320	96.7	2320	976	640		
EDS's/Kick Stage/Lander	Coast	47	2367	98.6	2367	1023	687		
EDS's	MCC & EDS Disposal	0	2367	98.6	2367	1023	687		
Kick Stage/Lander	Coast	47	2414	100.6		1070	734		
Kick Stage/Lander	Libration Point Arrival	0	2414	100.6		1070	734		
Kick Stage/Lander	Loiter at L1	178	2592	108.0		1248	912		
EDS5	Launch from Earth/Loiter	2	2594	108.1		1250	914	2	
EDS5	Loiter in LEO	334	2928	122.0		1584	1248	336	
EDS6	Launch from Earth/Loiter	2	2930	122.1		1586	1250	338	
EDS6	Dock w/ EDS5	50	2980	124.2		1636	1300	388	

Vehicle	Phase Name	Phase Length	Mission Elapsed Time						
			Overall MET		LL EDS	Kick Stage	Lander	CEV EDS	CEV
		(hr)	(hr)	(days)	(hr)				
EDS6	Loiter in LEO	284	3264	136.0		1920	1584	672	
CEV	Launch Weather Delay	48	3312	138.0		1968	1632	720	48
CEV	Launch from Earth/Loiter	2	3314	138.1		1970	1634	722	50
CEV	Dock w/ EDS's	50	3364	140.2		2020	1684	772	100
CEV	Vehicle Checkout	12	3376	140.7		2032	1696	784	112
EDS's	Earth Orbit Departure	0	3376	140.7		2032	1696	784	112
EDS's/CEV	Coast	47	3423	142.6		2079	1743	831	159
EDS's	MCC & EDS Disposal	0	3423	142.6		2079	1743	831	159
CEV	Coast	47	3470	144.6		2126	1790		206
CEV	Libration Point Arrival	0	3470	144.6		2126	1790		206
CEV	Dock w/ Lander	6	3476	144.8		2132	1796		212
CEV/Kick Stage/Lander	Crew Transfer & Checkout	24	3500	145.8		2156	1820		236
Kick Stage/Lander	Undock from CEV	0	3500	145.8		2156	1820		236
Kick Stage	Libration Point Departure	0	3500	145.8		2156	1820		236
Kick Stage/Lander	Coast	60	3560	148.3		2216	1880		296
Kick Stage	Lunar Orbit Insertion	0	3560	148.3		2216	1880		296
Kick Stage	Kick Stage Disposal	0	3560	148.3		2216	1880		296
Lander	Powered Descent	0	3560	148.3			1880		296
Lander	Surface Mission	168	3728	155.3			2048		464
Lander	Ascent	0	3728	155.3			2048		464
Lander	Lunar Orbit Departure	0	3728	155.3			2048		464
Lander	Coast	60	3788	157.8			2108		524
Lander	Libration Point Arrival	0	3788	157.8			2108		524
Lander	Rendezvous & Dock w/ CEV	6	3794	158.1			2114		530
Lander/CEV	Crew Transfer & Checkout	24	3818	159.1			2138		554
CEV	Undock from Lander	0	3818	159.1			2138		554
Lander	Ascent Stage Disposal	0	3818	159.1			2138		554
CEV	Libration Point Departure	0	3818	159.1					554
CEV	Coast	91	3909	162.9					645
CEV	Dispose Service Module	0	3909	162.9					645
CEV	Coast & Entry	3	3912	163.0					648
CEV	Recovery	1	3913	163.0					649

Table 13.2-1: Mission Phase Description

13.3 Safety & Mission Success

The design approach to meet the 25 t launch limit is drastically different from the TRM. Due to having a launch limit of 25 t and given the predicted mass properties for the elements, more total elements are required for a successful mission. An increase in the number of elements from the TRM increases the number of separation events and dockings. Thus, the total number of critical events for the 25 t launch limit architecture far exceeds those of the TRM. This variant identified 103 critical events. This is in comparison to the 55 critical events identified for the TRM. Of the 103 critical events, 66 were for the uncrewed phases of the mission while the remaining 37 were for the crewed phases of the mission. The TRM contained 20 uncrewed critical events and 36 crewed critical events.

Of the 103 identified critical events for the 25 t Launch Limit Approach, seven were ranked a three, 24 were ranked a two, and the remaining 72 were ranked a one. The complete set of identified and ranked critical events for the 25 t Launch Limit Approach is listed in the table below.

	ID #	TRM with 25 t Launch Limit Critical Events	TRM with 25 t Launch Limit Critical Event Rank
Uncrewed Critical Events	VAR-03-01	EDS-1A (for the LL) Launch	1
	VAR-03-02	EDS-1A Ascent	1
	VAR-03-03	EDS-1A Launch Shroud Separation	1
	VAR-03-04	EDS-1A Separation from Booster	1
	VAR-03-05	EDS-1A Orbital Maneuvering	1
	VAR-03-06	EDS-1B (for the LL) Launch	1
	VAR-03-07	EDS-1B Ascent	1
	VAR-03-08	EDS-1B Launch Shroud Separation	1
	VAR-03-09	EDS-1B Separation from Booster	1
	VAR-03-10	EDS-1B Orbital Maneuvering	1
	VAR-03-11	EDS-1A Docks with EDS-1B	1
	VAR-03-12	EDS-1A & EDS-1B Orbital Maneuvering	1
	VAR-03-13	EDS-1C (for the LL) Launch	1
	VAR-03-14	EDS-1C Ascent	1
	VAR-03-15	EDS-1C Launch Shroud Separation	1
	VAR-03-16	EDS-1C Separation from Booster	1
	VAR-03-17	EDS-1C Orbital Maneuvering	1
	VAR-03-18	EDS-1C Docks with EDS-1A & EDS-1B	1
	VAR-03-19	EDS-1A, EDS-1B, & EDS-1C Orbital Maneuvering	1
	VAR-03-20	EDS-1D (for the LL) Launch	1
	VAR-03-21	EDS-1D Ascent	1
	VAR-03-22	EDS-1D Launch Shroud Separation	1
	VAR-03-23	EDS-1D Separation from Booster	1
	VAR-03-24	EDS-1D Orbital Maneuvering	1
	VAR-03-25	EDS-1D Docks with EDS-1A, EDS-1B, & EDS-1C	1

	ID #	TRM with 25 t Launch Limit Critical Events	TRM with 25 t Launch Limit Critical Event Rank
	VAR-03-26	EDS-1A, EDS-1B, EDS-1C, & EDS-1D Orbital Maneuvering	1
	VAR-03-27	Kickstage Launch	1
	VAR-03-28	Kickstage Ascent	1
	VAR-03-29	Kickstage Launch Shroud Separation	1
	VAR-03-30	Kickstage Separation from Booster	1
	VAR-03-31	Kickstage Orbital Maneuvering	1
	VAR-03-32	LL Descent Stage Launch	1
	VAR-03-33	LL Descent Stage Ascent	1
	VAR-03-34	LL Descent Stage Launch Shroud Separation	1
	VAR-03-35	LL Descent Stage Separation from Booster	1
	VAR-03-36	LL Descent Stage Orbital Maneuvering	1
	VAR-03-37	LL Descent Stage docks with Kickstage	1
	VAR-03-38	LL Descent Stage & Kickstage Orbital Maneuvering	1
	VAR-03-39	LL Ascent Stage Launch	1
	VAR-03-40	LL Ascent Stage Ascent	1
	VAR-03-41	LL Ascent Stage Launch Shroud Separation	1
	VAR-03-42	LL Ascent Stage Separation from Booster	1
	VAR-03-43	LL Ascent Stage Orbital Maneuvering	1
	VAR-03-44	LL Ascent Stage Docks with LL Descent Stage & Kickstage	1
	VAR-03-45	LL (Ascent + Descent Stages) & Kickstage Orbital Maneuvering	1
	VAR-03-46	LL + Kickstage Docks to EDS-1A, EDS-1B, EDS-1C, & EDS-1D	1
	VAR-03-47	EDS-1A/1B/1C/1D, Kickstage & LL Burn for L1	1
	VAR-03-48	EDS-1A/1B/1C/1D, Kickstage & LL Mid-course Correction Burn	1
	VAR-03-49	EDS-1C & EDS-1D Separates from EDS-1A, EDS-1B, Kickstage, & LL	1
	VAR-03-50	LL, Kickstage, EDS-1A, & EDS-1B Mid-course Correction Burn	1
	VAR-03-51	EDS-1A & EDS-1B Separates from Kickstage, & LL	1
	VAR-03-52	LL & Kickstage Mid-course Correction Burn	1
	VAR-03-53	LL & Kickstage Burn to Slow Near L1	1
	VAR-03-54	LL & Kickstage Orbital Maneuvering	1
	VAR-03-55	EDS-2A (for CEV) Launch	1
	VAR-03-56	EDS-2A Ascent	1
	VAR-03-57	EDS-2A Launch Shroud Separation	1
	VAR-03-58	EDS-2A Separation from Booster	1
	VAR-03-59	EDS-2A Orbital Maneuvering	1
	VAR-03-60	EDS-2B Launch	1
	VAR-03-61	EDS-2B Ascent	1
	VAR-03-62	EDS-2B Launch Shroud Separation	1

	ID #	TRM with 25 t Launch Limit Critical Events	TRM with 25 t Launch Limit Critical Event Rank
	VAR-03-63	EDS-2B Separation from Booster	1
	VAR-03-64	EDS-2B Orbital Maneuvering	1
	VAR-03-65	EDS-2A Docks with EDS-2B	1
	VAR-03-66	EDS-2A & EDS-2B Orbital Maneuvering	1
Crewed Critical Events	**VAR-03-67**	CEV (CM+SM) Launch	2
	VAR-03-68	CEV Ascent	2
	VAR-03-69	LAS Separation	2
	VAR-03-70	CEV Launch Shroud Separation	2
	VAR-03-71	CEV Separation from Booster	2
	VAR-03-72	CEV Orbital Maneuvering	2
	VAR-03-73	CEV Docks to EDS-2A & EDS-2B	2
	VAR-03-74	EDS-2A, EDS-2B & CEV Burn for L1	2
	VAR-03-75	EDS-2A, EDS-2B & CEV Mid-course Correction Burn	1
	VAR-03-76	EDS-2A & EDS-2B Separates from CEV	2
	VAR-03-77	CEV Mid-course Correction Burn	1
	VAR-03-78	CEV Burns to Slow Near L1	2
	VAR-03-79	CEV Orbital Maneuvering	2
	VAR-03-80	CEV Docks to LL & Kickstage at L1	2
	VAR-03-81	Crew Transfer from CEV to LL & Kickstage	1
	VAR-03-82	LL & Kickstage Separates from CEV	2
	VAR-03-83	LL & Kickstage Burns for Low Lunar Orbit	2
	VAR-03-84	LL & Kickstage Mid-course Correction Burn	1
	VAR-03-85	LL & Kickstage Lunar Orbit Insertion (LOI)	2
	VAR-03-86	Kickstage Separates from LL	2
	VAR-03-87	LL Deorbit Burn to Moon	2
	VAR-03-88	LL Powered Descent & Landing to Moon	3
	VAR-03-89	LL Ascent Stage Separation & Ascent	3
	VAR-03-90	LL Ascent Stage Orbital Maneuvering	3
	VAR-03-91	LL Ascent Stage Lunar Orbit Departure	3
	VAR-03-92	LL Ascent Stage Mid-Course Correction Burn	1
	VAR-03-93	LL Ascent Stage L1 Arrival	3
	VAR-03-94	LL Ascent Stage Orbital Maneuvering	2
	VAR-03-95	LL Ascent Stage Docks with CEV	2
	VAR-03-96	Crew Transfer from LL to CEV	2
	VAR-03-97	CEV Separates from LL Ascent Stage	2
	VAR-03-98	CEV Burn for Earth	3
	VAR-03-99	CEV Mid-course Correction Burn	1
	VAR-03-100	CM Separates & Maneuvers away from SM	2
	VAR-03-101	CM Entry	3
	VAR-03-102	CM Landing	2
	VAR-03-103	Crew Recovery	2

Table 13.3-1: 25 t Solution Critical Events and Ranking

From a qualitative standpoint, the TRM is a much less risky approach than the 25 t Launch Limit approach in terms of Mission Success. The TRM has four elements that are launched (EDS-1, EDS-2, Lunar Lander, & CEV). By imposing a 25 t limit on the mission architecture, the number of launches increases from four to ten. The launch mass limit forces the EDS-1 single element from the TRM to be launched as four smaller stages and the EDS-2 to be launched as two smaller stages. The Lunar Lander Descent Stage, Lunar Lander Ascent Stage, and the Kick Stage are all launched as separate elements. The increase in number of separate elements to be launched has a ripple effect. For each additional element launched, several critical events are added such as separations and dockings. For example, the TRM has eleven separation events and four docking events. Imposing the launch limit increases the number of separation events from eleven to eighteen and the number of docking events from four to ten.

There is also another aspect of Mission Success that should be discussed, and that is the two-week interval between launches. If this groundrule were followed in the 25 t Launch Limit Approach, there would be an eighteen-week period between the first launch of the mission (uncrewed) and the launch of the crew aboard the CEV spacecraft. In order to have achieved mission success, there can be no unrecoverable system or subsystem failures on any of the mission architecture elements while they are loitering in LEO. However, if major system or subsystem failures occur on one or more of the mission architecture elements while loitering in LEO, there must be a maintenance strategy built into all the mission architecture elements. For example, there could be built-in redundancy, an increase in the system and/or subsystem reliability, or an autonomous repair capability. There should also be contingency operational plans for losing an element during a launch or while it on-orbit, after the first element launch occurs. For example, if one of the EDS elements is lost due to a catastrophic launch vehicle failure or a major subsystem failure, a spare EDS should be available for immediate launch to replace the failed EDS.

In terms of crew safety, the TRM and 25 t Launch Limit Approach are relatively the same. The only major difference would occur during the L1 transit. The TRM had a single EDS to carry the CEV out to L1. With the 25 t limit, two stages would be involved in the transit of the CEV to L1. Having two stages during the L1 CEV spacecraft transit to L1 may allow for additional mission abort opportunities as opposed to having a single EDS element.

13.4 Mission Abort Options

As the Crew Exploration Vehicle functions identically following Earth orbit departure for the two options under consideration, mission aborts are unaffected by changing the TRM from a four launch per mission to a 25 t per launch mission architecture.

13.5 Element Overview & Mass Properties

This section describes any changes made in sizing the trade reference mission elements and compares the resulting vehicle mass properties. The total architecture mass for the 25 t per launch architecture variant is estimated at 240 t, a 10 t increase over the TRM.

13.5.1 Crew Exploration Vehicle

The CEV for the 25 t Solution variant is identical to the TRM CEV.

13.5.2 Lunar Lander

The combined mass for the Lunar Lander in this architecture option is 44.8 t, which is significantly higher than the imposed 25 t per launch limit. Therefore, to stay under the limit, the Ascent Stage is separated from the Descent Stage for launch and the two stages are assembled with the Kick Stage and Earth Departure Stages on orbit. This requirement adds rendezvous propellant mass to the Ascent Stage and mating hardware mass to both elements that was not previously necessary. With an extra launch in the architecture after the Lander is assembled, the Lander also requires a longer on-orbit loiter time, which adds additional propellant boil-off. The final modification made is that the Lander Descent Stage in this variant performs the final 150 m/s of lunar orbit insertion, which further increases tank size and propellant. A Kick Stage sized to perform the entire maneuver as was done in the TRM exceeded the 25 t limit.

Lander's Ascent Stage's System Mass Changes				
System	**TRM**	**25 t-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	839	839	No Change	0.0
Protection	73	73	No Change	0.0
Propulsion	1631	1631	No Change	0.0
Power	813	813	No Change	0.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	851	No Change	0.0
Other	455	455	No Change	0.0
Growth	1080	1080	No Change	0.0
Non-Cargo	1483	1483	No Change	0.0
Cargo	227	227	No Change	0.0
Non-Propellant	1014	1014	No Change	0.0
Propellant	10703	10703	No Change	0.0
Total	19906	19906	No Change	0.0

Table 13.5.2-1: Variation in Lander Ascent Stage Mass with 25 t Launch Solution

Lander's Descent Stage's System Mass Changes				
System	**TRM**	**25 t-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	553	571	18	3.3
Protection	50	50	No Change	0.0
Propulsion	1413	1551	138	9.8
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	714	6	0.8
Growth	678	711	33	4.9
Non-Cargo	464	518	54	11.6
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	19610	2037	11.6
Total	22608	24893	2285	10.1

Table 13.5.2-2: Variation in Lander Descent Stage Mass with 25 t Launch Solution

13.5.3 Kick Stage

The Kick Stage in this architecture launches separately from the Lunar Lander; therefore, it requires the capability to automatically rendezvous and dock with the assembled Earth Departure Stages in LEO. The Lander had provided that functionality in the TRM, so for this option, the same suite of avionics, power generation, attitude control, and thermal control equipment included on the EDS is included on the Kick Stage. The Kick Stage also includes provisions for a longer on-orbit loiter duration.

Kick Stage's System Mass Changes				
System	**TRM**	**25 t-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	621	589	(32)	(5.2)
Protection	0	0	No Change	0.0
Propulsion	1530	1490	(40)	(2.6)
Power	100	190	90	90.0
Control	0	0	No Change	0.0
Avionics	0	175	175	0.0
Environment	0	105	105	0.0
Other	405	405	No Change	0.0
Growth	531	591	60	11.3
Non-Cargo	953	1058	105	11.0
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	23323	21373	(1950)	(8.4)
Total	27465	25976	(1489)	(5.4)

Table 13.5.3-1: Variation in Kick Stage Mass with 25 t Launch Solution

13.5.4 Earth Departure Stages

Fitting within the 25 t per launch limit requires that the 94 t Lander EDS and 39 t CEV EDS from the TRM divide into four and two separate equal-mass stages, respectively. Some mass savings can be realized by staging the Earth orbit departure maneuver, though these savings are offset by the duplication of structure, propulsion, avionics, and other miscellaneous inert mass. The assumed Earth Departure Stages are simply scaled versions of the TRM EDS design, except that each stage contains one 25,000-lbf OMS engine instead of four engines in the TRM. In scaling the propellant tanks, provisions were made to accommodate propellant lost due to boil-off during the longer LEO loiter time of this architecture. The EDS's for this architecture variant also require additional structural provisions to mate with one or more other stages, and additional propellant to perform the rendezvous and mating maneuvers. However, given the limited analysis put into the EDS design, the amount of mass included in the EDS mass properties is likely insufficient. More work is necessary to formulate a credible concept for assembling four stages in low Earth orbit.

Lander Earth Departure Stage's System Mass Changes				
System	**TRM**	**25 t-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	1972	631	(1341)	(68.0)
Protection	0	0	No Change	0.0
Propulsion	4361	1542	(2819)	(64.6)
Power	190	190	No Change	0.0
Control	0	0	No Change	0.0
Avionics	175	175	No Change	0.0
Environment	105	105	No Change	0.0
Other	455	455	No Change	0.0
Growth	1452	620	(832)	(57.3)
Non-Cargo	3109	1291	(1818)	(58.5)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	82289	20162	(62127)	(75.5)
Total	94109	25172	(68937)	(73.3)

Table 13.5.4-1: Variation in Lander EDS Mass with 25 t Launch Solution

CEV Earth Departure Stage's System Mass Changes				
System	**TRM**	**3-Launch Solution**	**Mass Change (kg)**	**% Change**
Structure	932	542	(390)	(41.8)
Protection	0	0	No Change	0.0
Propulsion	2318	1863	(455)	(19.6)
Power	190	190	No Change	0.0
Control	0	0	No Change	0.0
Avionics	171	171	No Change	0.0
Environment	104	104	No Change	0.0
Other	455	455	No Change	0.0
Growth	834	665	(169)	(20.3)
Non-Cargo	1355	835	(520)	(38.4)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	32897	16620	(16277)	(49.5)
Total	39256	21444	(17812)	(45.4)

Table 13.5.4-2: Variation in CEV EDS Mass with 25 t Launch Solution

13.5.5 Vehicle Mass Properties

Table 13.5.5-1 lists vehicle mass properties for the 25 t per launch architecture variant, and Figure 13.5.5-1 compares individual vehicle mass to the trade reference mission. All elements are limited to a 25 t gross mass. While some elements slightly exceed this limit (the CEV is estimated at 26.4 t), there was sufficient uncertainty in the vehicle design and mass estimating techniques to warrant not adding additional launches at this time. The combined architecture elements of this variant have a total IMLEO of 240 t, compared to 230 t for the trade reference mission.

	CEV CM	CEV SM	CEV Earth Dep. Stage (x 2)	Ascent Stage	Descent Stage	Kick Stage	Lander Earth Dep. Stage (x 4)
1.0 Structure	1,523	1,455	542	839	571	589	631
2.0 Protection	822	0	0	73	50	0	0
3.0 Propulsion	117	1,408	1,863	1,631	1,551	1,490	1,542
4.0 Power	482	661	190	813	137	190	190
5.0 Control	0	0	0	0	0	0	0
6.0 Avionics	737	0	171	738	0	175	175
7.0 Environment	691	110	104	851	530	105	105
8.0 Other	835	100	455	455	714	405	455
9.0 Growth	1,041	747	665	1,080	711	591	620
DRY MASS	6,249 kg	4,481 kg	3,990 kg	6,479 kg	4,265 kg	3,545 kg	3,719 kg
10.0 Cargo	966	305	835	1,483	518	1,058	1,291
11.0 Non-Cargo	1,478	0	0	227	500	0	0
INERT MASS	8,692 kg	4,786 kg	4,825 kg	8,190 kg	5,283 kg	4,603 kg	5,010 kg
12.0 Non-Propellant	55	1,442	0	1,014	0	0	0
13.0 Propellant	64	11,332	16,620	10,703	19,610	21,373	20,162
GROSS MASS	8,812 kg	17,560 kg	21,444 kg	19,906 kg	24,893 kg	25,976 kg	25,172 kg

Table 13.5.5-1: Vehicle Mass Properties

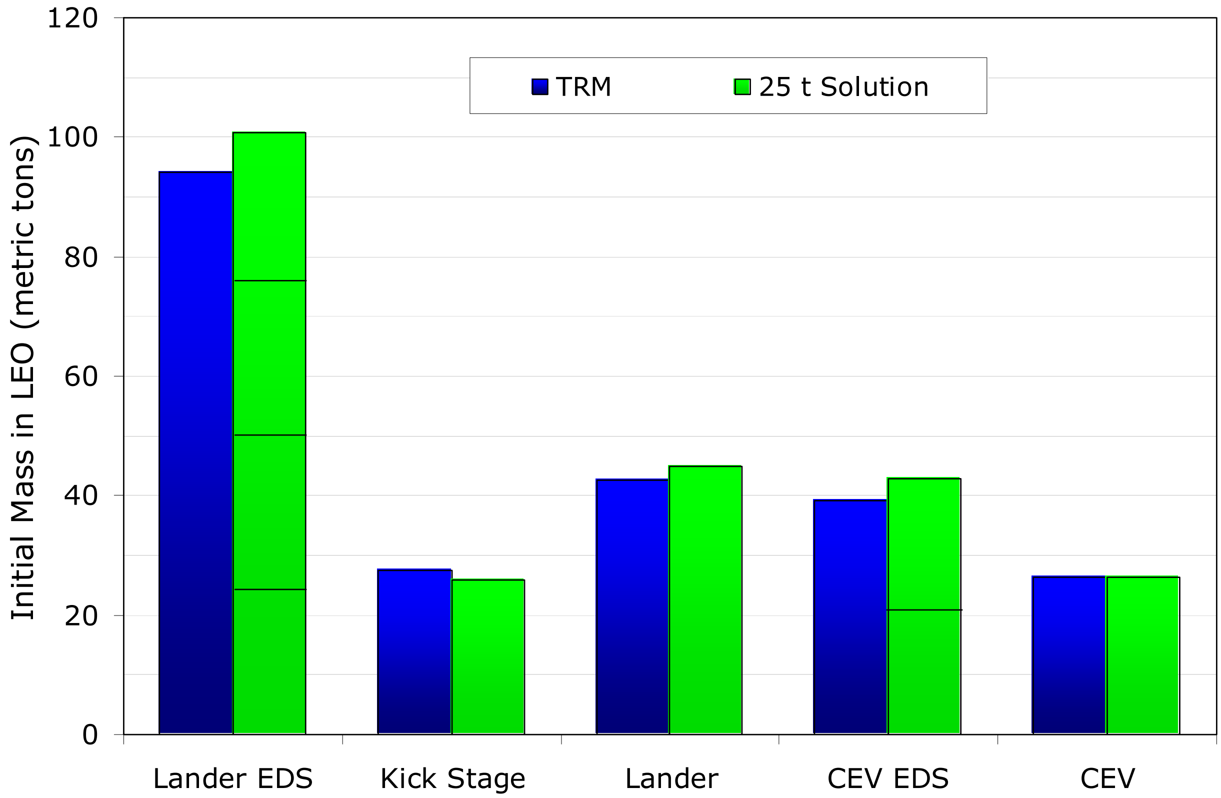


Figure 13.5.5-1: Vehicle Mass Properties Comparison

Figure 13.5.5-2 compares individual launch package masses to the trade reference mission. This architecture variant requires a 25 t payload capability by the cargo launch vehicle to deliver the elements to LEO, significantly less than the 94 t required maximum capability for the TRM. Size requirements for the CEV's human-rated launch vehicle are identical. However, as the figure shows, the 25 t Solution requires six more launches than the TRM to perform the same mission.

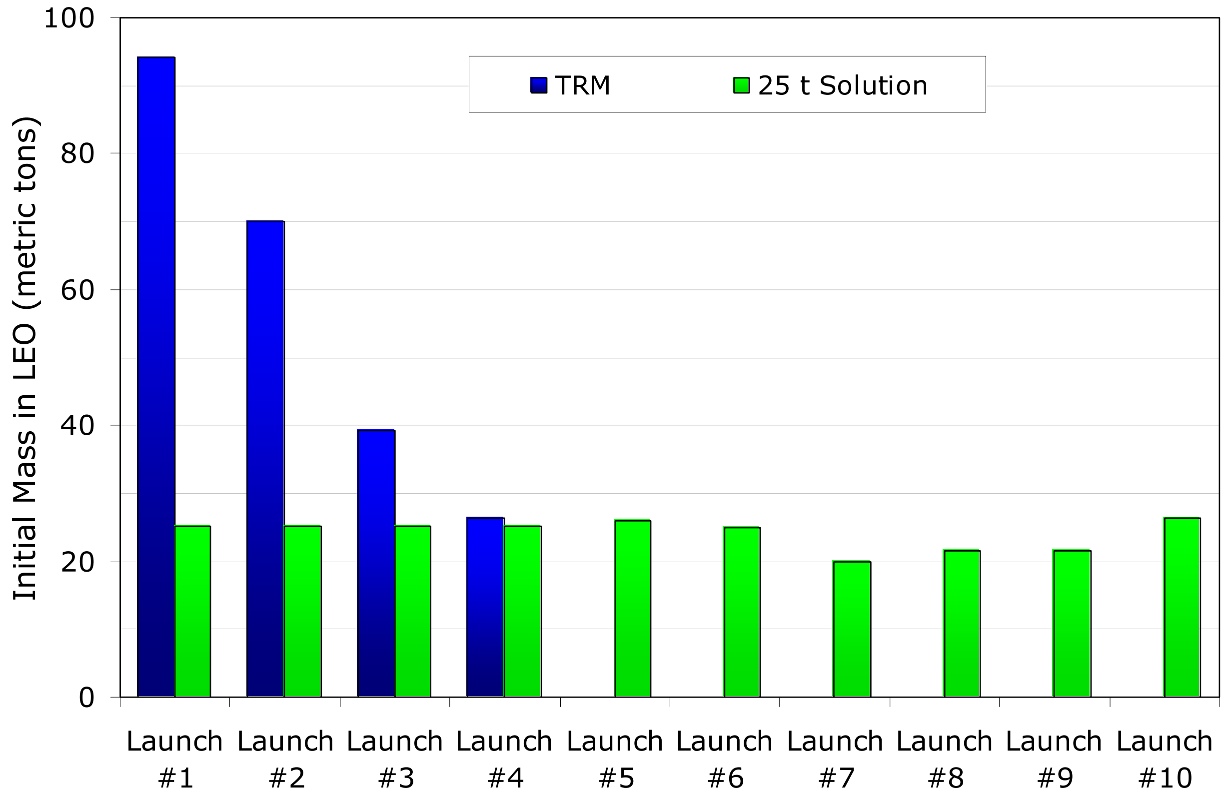


Figure 13.5.5-2: Mass per Launch Comparison

13.6 System Technologies and Programmatic Risks

The analysis performed for this architecture variant did not modify any vehicle system technology assumptions made for the trade reference mission.

13.7 Pros/Cons Summary

The foremost positive aspect of the 25 t-Solution architecture variant in comparison to the trade reference mission is that it utilizes the existing fleet of commercial EELV-heavy launchers rather than developing a new heavy-lift launcher whose only user may be NASA human exploration. This will provide a new market for those vehicles, and the resulting development cost savings could potentially be used instead for investments in enhancing or enabling spacecraft technologies.

Restricting an architecture with a total mass well over 200 t to individual launch sizes of only 25 t, however, has a significant impact on mission risk and complexity. Whereas the previous three architectures described required four, two, and three launches per mission respectively, this variant requires ten. Assuming a launch probability of success of 98% (the approximate historical average), the 92.2% probability of simply delivering all architecture elements to LEO for the TRM drops to 81.7% here. Once on orbit, this option requires that four Earth Departure Stages, an Ascent Stage, a Descent Stage, and a Kick Stage assemble from individual elements into one combined stack, and two Earth Departure Stages and a CEV into another stack. The TRM only required two such matings. Another negative aspect for this variant is that during the Earth orbit departure maneuver, an EDS divided into multiple equal mass pieces is staged to save mass, which also adds risk to the mission. Burning all Lander Earth Departure Stages simultaneously would increase the size of those elements above the 25 t limit. Finally, with this ten launch, 25 t per launch architecture, the total Lander/Kick Stage and maximum EDS loiter times are increased by 36 days and 70 days, thereby increasing propellant boil-off and risk of vehicle failure.

The total architecture mass estimate for the 25 t-Solution option represents an increase of ~11 or 4.8% above the four-launch trade reference mission. This estimate, though, is likely too low given the very limited analysis put into the LEO assembly aspects of the mission. Additional effort will likely show that propellant, avionics, and structural mass required for mating multiple elements on-orbit is underestimated.

14.0 TRM with Lunar Orbit Rendezvous

This first architecture variant examines the impact of changing the rendezvous point for the mission. Instead of CEV/Lunar Lander rendezvous at Lunar L1 after returning from the Moon for the trade reference mission, this variant assumes the CEV and Lander will rendezvous in low lunar orbit (LOR). This section of the report examines the impact of such a change.

14.1 Major Assumptions/Clarifications

The "TRM with LOR" architecture variant affects the following TRM assumptions from the original LDRM-2 task request statement. Assumptions from Section 10.0 not explicitly listed here are still applicable to the architecture.

<u>Libration point L1 is used as the lunar vicinity rendezvous point to enable global lunar surface access</u>: This variant uses lunar orbit rendezvous while retaining the capability to land at any latitude and longitude on the lunar surface.

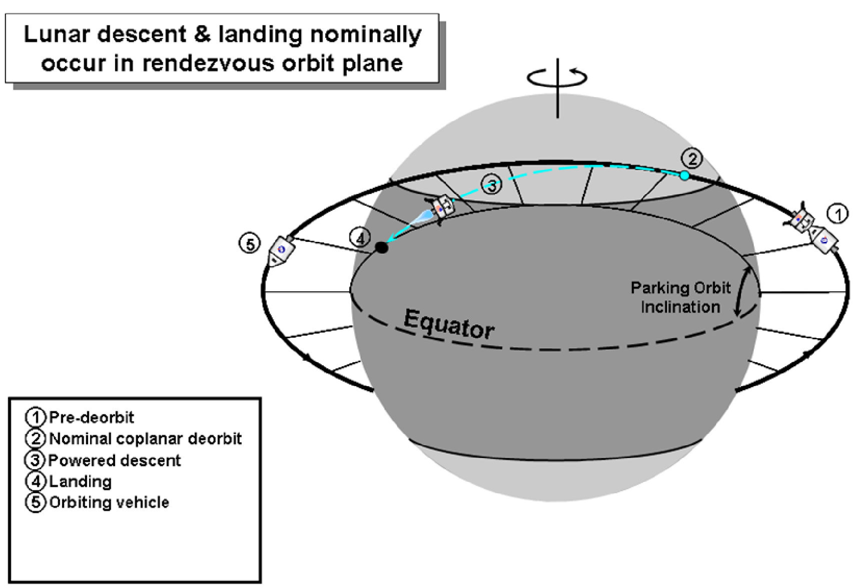
<u>The Lunar Lander will be pre-deployed to lunar vicinity prior to initiation of the CEV mission:</u> As discussed below, pre-deploying the Lunar Lander to lunar vicinity in a global access LOR architecture severely restricts the frequency of possible mission opportunities. Therefore, this variant will assume initial CEV/Lander assembly takes place in LEO, not in lunar vicinity.

14.2 Architecture Description

The LOR mission begins with two consecutive launches of equal-mass Earth Departure Stages with two-week spacing between launches. The assumed cargo launch vehicle for the architecture delivers the elements to the LEO parking orbit previously assumed (28.5^{o} 407 km), where they loiter for assembly. Two weeks after the second launch, the Lunar Lander is delivered to LEO by a third cargo launch vehicle. The Lander performs a variable-length double coelliptic rendezvous maneuver profile to rendezvous and dock with the assembled Earth Departure Stages (the target vehicle) within 50 hr after launch. Recall that for the TRM, a Kick Stage was included and it launched with the Lunar Lander. The Kick Stage performed the libration point arrival, libration point departure, and lunar orbit insertion maneuvers for the Lander. With LOR, there are no libration point-related maneuvers and the EDS, the CEV, or the Lander can perform lunar orbit insertion. Therefore, this variant does not need a Kick Stage.

Finally, two weeks after the Lander, the crew launch in the CEV (the 4^{th} launch of a four launch per mission architecture) on a separate, human-rated launch vehicle. The CEV, as the chaser vehicle, performs a stable orbit rendezvous maneuver profile to rendezvous and dock with the Lander and EDS's within 50 hr after orbit insertion. This represents a different mission strategy than the TRM, where the Lander is pre-deployed to Lunar L1 prior to initiation of the CEV mission. However, given the limitations that pre-deploying the Lander to lunar orbit places on possible mission opportunities, it is more prudent to assemble the elements in LEO and depart for the Moon as a single integrated stack. Pre-deployment works well for libration point architectures because of the fixed geometry of the Earth-Moon-L1 system, which means that any point on the lunar surface will be continuously available from L1. With lunar orbit rendezvous, the Lander would be positioned in a parking orbit tailored for a specific landing site, but if CEV were unable to reach the Lander when originally planned, that landing site would not be theoretically accessible until 27.3 days later when the slow-rotating Moon moved the landing site back under the parking orbit. These complications can be eliminating by transporting the crew and CEV to the Moon together with the Lander.

Once the CEV mates with the assembled stack in LEO, the crew and mission control check out the vehicles and the first EDS performs part of the Earth orbit departure maneuver (52%) at the opening of the window. That stage separates, disposes itself, and the second EDS ignites to complete the burn. The selected Earth-to-Moon trajectory is a near-minimum delta-V transfer with a flight time of 96 hr. A 24-hr minimum delta-V injection window has been included in the sizing of the Earth Departure Stages so flight time to lunar orbit may vary between 108 hr for injection at the opening of the window to 84 hr for injection at window closing. At perilune, the Earth Departure Stage will execute the first of three impulsive maneuvers to insert the CEV and Lunar Lander into a 100 x 100 km lunar parking orbit with an inclination and longitude of ascending node appropriate for the selected landing site. The first maneuver inserts the stack into a 24-hr orbit, the second maneuver performs up to a 90^o plane change at apolune to tailor the orbit's inclination and ascending node, and the third circularizes the orbit at 100 km altitude. This is the same approximate parking orbit altitude used in the Apollo missions. The 3-impulse sequence of events, which adds 24 hr and 538 m/s delta-V to the mission, is necessary to meet global access and anytime return requirements and is more efficient than a comparable single-impulse event. By choosing a particular inclination and ascending node for a given landing site latitude and longitude, in-plane descent and ascent trajectories can be planned for the start and end of the nominal surface duration, and the cost of ascending back to the parking orbit before the nominal time can be kept small (see the sequence of illustrations in Figure 14.2-1). For 7-day missions such as the TRM, the maximum out-of-plane ascent has a wedge angle of 6.7^o.



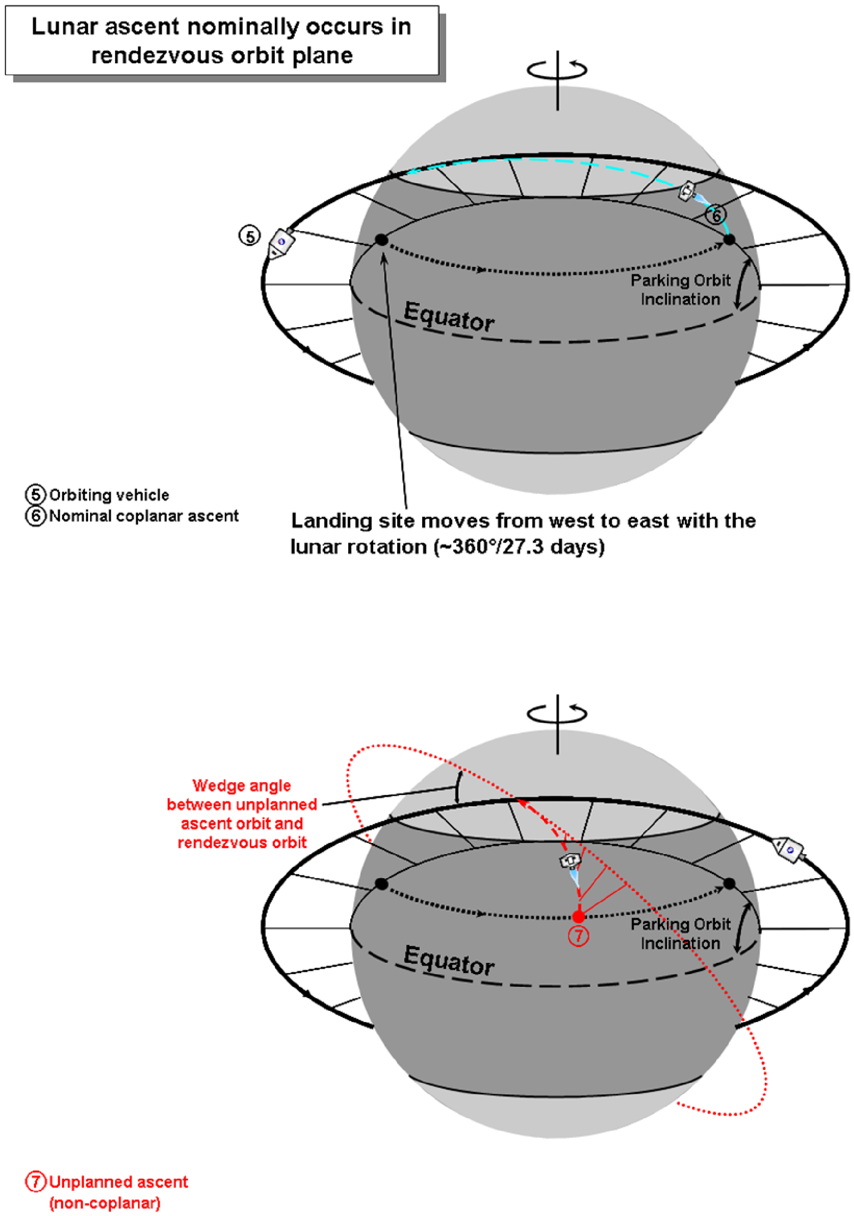


Figure 14.2-1: Tailoring the Parking Orbit for Short-Stay LOR

After successful insertion into the lunar parking orbit, the Earth Departure Stage separates from the Lunar Lander and CEV and disposes itself via lunar impact. The crew then transfers to the Lander, checks out the vehicle, and undocks from the CEV. Deorbit and powered descent (1,881 m/s) to the surface with the Lander Descent Stage follows at the first available opportunity, which should occur within one orbit revolution. The lunar surface exploration strategy for lunar orbit rendezvous is identical to the TRM. As the surface mission is expiring, the crew will prepare the Lander Ascent Stage for return to lunar orbit. The Ascent Stage separates from the Descent Stage on the lunar surface and ascends (1,834 m/s) to the 100 x 100 km CEV parking orbit.

The vehicle also includes an extra 191 m/s of delta-V for anytime ascent. Arriving in lunar orbit, the Ascent Stage performs a series of rendezvous maneuvers to re-dock with the CEV within 6 hr.

After docking, the crew transfers back over to the CEV to start up and check out the vehicle, transfers over any cargo returned to Earth, and undocks from the Lander Ascent Stage. The CEV then executes another 3-impulse, 24-hr sequence of maneuvers to return to Earth. After 7 days on the surface, the lunar parking orbit may not be aligned for Earth return and may require a plane change to meet the anytime return requirement. In a worst-case, the plane may be up to 90^{o} out of alignment. Here, the first impulse of the sequence raises the apolune altitude to create a 24-hr period orbit, the second performs the plane change at apolune where it is most efficient, and the third burn departs the Moon and targets the CEV for Earth atmospheric entry 96 hr later. The end of the CEV mission is the same as in the trade reference mission. The Ascent Stage, left unoccupied in low lunar orbit, is disposed on the lunar surface.

Figures 14.2-2 – 14.2-3 and Tables 14.2-1 – 14.2-2 outline the assumed timelines and delta-V's for the lunar orbit rendezvous variant as described above.

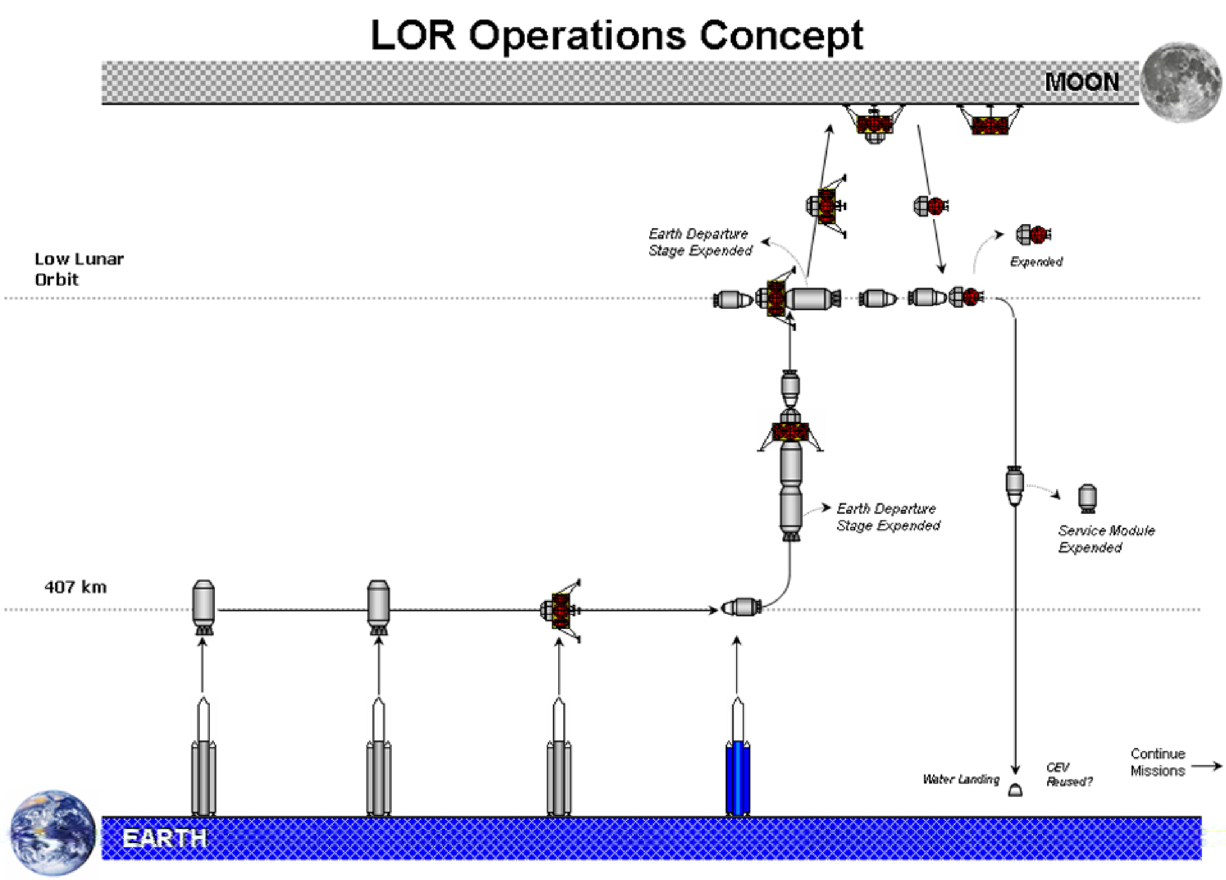


Figure 14.2-2: TRM with Lunar Orbit Rendezvous Architecture Illustration

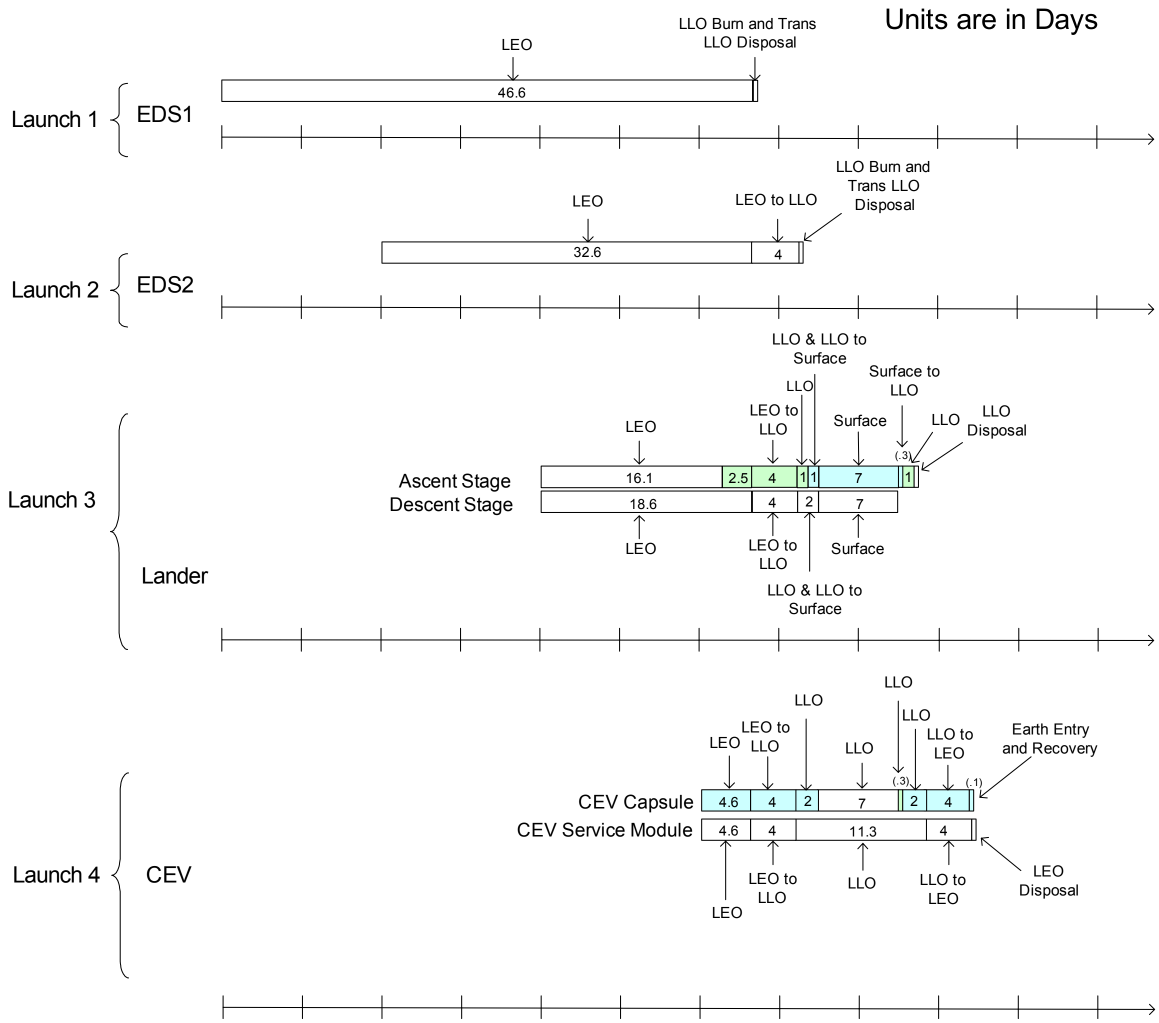


Figure 14.2-3: Nominal Timeline for Lunar Orbit Rendezvous

Vehicle	Phase Name	Phase Length (hr)	Mission Elapsed Time Overall MET (hr)	Overall MET (days)	EDS1 (hr)	EDS2 (hr)	Lander (hr)	CEV (hr)
EDS1	Launch from Earth/Loiter	2	2	0.1	2			
EDS1	Loiter in LEO	334	336	14.0	336			
EDS2	Launch from Earth/Loiter	2	338	14.1	338	2		
EDS2	Rendezvous & Dock w/ EDS	50	388	16.2	388	52		
EDS2	Loiter in LEO	284	672	28.0	672	336		
Lander	Launch from Earth/Loiter	2	674	28.1	674	338	2	
Lander	Rendezvous & Dock w/ EDS	50	724	30.2	724	388	52	
EDS/Lander	Vehicle Checkout	12	736	30.7	736	400	64	
EDS/Lander	Loiter in LEO	272	1008	42.0	1008	672	336	
EDS/Lander	Missed EOD Opportunity	240	1248	52.0	1248	912	576	
CEV	Launch Weather Delay	48	1296	54.0	1296	960	624	48
CEV	Launch from Earth/Loiter	2	1298	54.1	1298	962	626	50
CEV	Rendezvous & Dock w/ Stack	50	1348	56.2	1348	1012	676	100
EDS/Lander/CEV	Vehicle Checkout	12	1360	56.7	1360	1024	688	112
EDS	Earth Orbit Departure	0	1360	56.7	1360	1024	688	112
EDS/Lander/CEV	Coast	48	1408	58.7		1072	736	160
EDS	MCC & EDS Disposal	0	1408	58.7			736	160
EDS/Lander/CEV	Coast	48	1456	60.7			784	208
EDS2	Lunar Orbit Insertion	0	1456	60.7			784	208
CEV	3-Impulse Plane Change	24	1480	61.7			808	232
Lander/CEV	Crew Transfer & Checkout	24	1504	62.7			832	256
Lander	Undock from CEV	0	1504	62.7			832	256
Lander	Powered Descent	0	1504	62.7			832	256
Lander	Surface Mission	168	1672	69.7			1000	424
Lander	Ascent	0	1672	69.7			1000	424
Lander	Rendezvous & Dock w/ CEV	6	1678	69.9			1006	430
Lander/CEV	Crew Transfer & Checkout	24	1702	70.9			1030	454
CEV	Undock from Lander	0	1702	70.9			1030	454
Lander	Ascent Stage Disposal	0	1702	70.9			1030	454
CEV	3-Impulse Plane Change	24	1726	71.9				478
CEV	Lunar Orbit Departure	0	1726	71.9				478
CEV	Coast	93	1819	75.8				571
CEV	Dispose Service Module	0	1819	75.8				571
CEV	Coast & Entry	3	1822	75.9				574
CEV	Recovery	1	1823	76.0				575

Table 14.2-1: Mission Phase Description

Maneuver Name	Element	ΔV (m/s)	Comments
Earth Orbit Departure	EDS1 & EDS2	3,104	Co-planar departure from LEO assembly orbit (407 km, 28.5°) w/ 24-hr injection window. Nominal flt time to lunar orbit = 96 hr. Moon @ perigee.
Lunar Orbit Insertion	EDS2	1,416	Insertion into 100x100 km orbit tailored to landing site (V_∞ = 986 m/s). Includes 3-impulse insertion maneuver w/ 24-hr intermediate orbit for 90° worst-case relative declination angle.
Descent	Descent Stage	1,881	Fuel-optimal powered descent design for in-plane descent from 100x100 km orbit (ref. First Lunar Outpost study)
Ascent	Ascent Stage	2,025	Fuel-optimal powered ascent design for ascent to 100x100 km orbit. Includes 191 m/s for 6.7° on-orbit plane change (anytime ascent for 7-day surface stay)
Lunar Orbit Departure	CEV	1,410	Departure from 100x100 km orbit tailored to landing site (V_∞ = 952 m/s). Includes 3-impulse departure maneuver w/ 24-hr intermediate orbit for 90° worst-case relative declination angle. Nominal flt time to Earth = 96 hr.

Table 14.2-2: Summary of Major Maneuvers for the LOR Architecture

14.3 Safety & Mission Success

The Lunar Orbit Rendezvous approach is very comparable to the trade reference mission (TRM) based on the number of critical events and when during the mission profile, whether crewed or uncrewed, they occur. Fifty critical events were identified for the Lunar Orbit Rendezvous approach while the TRM had fifty-six critical events identified. Out of the fifty critical events identified for the Lunar Orbit Rendezvous approach, nineteen occurred during uncrewed portions of the mission while the remaining thirty-one occurred during the crewed portions of the mission. The TRM contained twenty critical events during the uncrewed portion of the mission and thirty-six critical events during crewed portions of the mission. The number of uncrewed critical events identified for the Lunar Orbit Rendezvous approach and the TRM differed by only one total event (all the elements get mated together in LEO and transit to lunar orbit together, versus the TRM where the Lunar Lander and CEV transit to the L1 point separately). However, the Lunar Orbit Rendezvous approach had five fewer crewed critical events than the TRM. This was due to the Lunar Orbit Rendezvous approach collecting and mating all the elements, both crewed and uncrewed, in LEO and then transiting directly to the Moon. The Kick Stage was not necessary for braking purposes at low lunar orbit. From a qualitative standpoint, the omission of using the Kick Stage for the L1 rendezvous point combined with the separate paths the crewed and uncrewed elements take to transit there should increase the Lunar Orbit Rendezvous approach's probability of mission success. The omission of using the Kick Stage for the L1 rendezvous point also inherently increases the crew safety for the Lunar Orbit Rendezvous approach by decreasing the number of crewed critical events. The reduction of other critical events such as additional rendezvous' and maneuverings with Earth Departure Stages and at the L1 point, both going to and from the lunar surface, may increase crew safety.

Of the fifty total critical events identified for the Lunar Orbit Rendezvous approach, seven received a rank of 3, nineteen received a rank of 2, and the remaining twenty-four received a rank of 1. The complete set of identified and ranked critical events for the Lunar Orbit Rendezvous approach is listed in the table below.

	ID #	TRM with Lunar Orbit Rendezvous Critical Events	TRM with Lunar Orbit Rendezvous Critical Event Rank
Uncrewed Critical Events	VAR-04-01	EDS-1 Launch	1
	VAR-04-02	EDS-1 Ascent	1
	VAR-04-03	EDS-1 Launch Shroud Separation	1
	VAR-04-04	EDS-1 Separation from Booster	1
	VAR-04-05	EDS-1 Orbital Maneuvering	1
	VAR-04-06	EDS-2 Launch	1
	VAR-04-07	EDS-2 Ascent	1
	VAR-04-08	EDS-2 Launch Shroud Separation	1
	VAR-04-09	EDS-2 Separation from Booster	1
	VAR-04-10	EDS-2 Orbital Maneuvering	1
	VAR-04-11	EDS-1 & EDS-2 Dock	1
	VAR-04-12	EDS-1 & EDS-2 Orbital Maneuvering	1
	VAR-04-13	LL Launch	1
	VAR-04-14	LL Ascent	1
	VAR-04-15	LL Launch Shroud Separation	1
	VAR-04-16	LL Separation from Booster	1
	VAR-04-17	LL Orbital Maneuvering	1
	VAR-04-18	LL Docks to EDS-1 & EDS-2	1
	VAR-04-19	EDS-1, EDS-2, & LL Orbital Maneuvering	1
Crewed Critical Events	VAR-04-20	CEV (CM+SM) Launch	2
	VAR-04-21	CEV Ascent	2
	VAR-04-22	LES Separation	2
	VAR-04-23	CEV Launch Shroud Separation	2
	VAR-04-24	CEV Separation from Booster	2
	VAR-04-25	CEV Orbital Maneuvering	2
	VAR-04-26	CEV Docks to EDS-1, EDS-2, & LL	2
	VAR-04-27	EDS-1, EDS-2, LL, & CEV Burn for Lunar Orbit	2
	VAR-04-28	EDS-1 Separates from EDS-2, LL, & CEV	2

	ID #	TRM with Lunar Orbit Rendezvous Critical Events	TRM with Lunar Orbit Rendezvous Critical Event Rank
	VAR-04-29	EDS-2, LL, & CEV Mid-course Correction Burn	1
	VAR-04-30	EDS-2 Separates from LL & CEV	2
	VAR-04-31	LL & CEV Mid-course Correction Burn	1
	VAR-04-32	LL & CEV Lunar Orbit Insertion (LOI)	2
	VAR-04-33	LL & CEV Orbital Maneuvering	2
	VAR-04-35	Crew Transfers from the CEV to LL	1
	VAR-04-36	LL Separates from CEV	2
	VAR-04-37	LL Powered Descent & Landing to the Moon	3
	VAR-04-38	LL Ascent Stage Separation & Ascent	3
	VAR-04-39	LL Ascent Stage Orbital Maneuvering	3
	VAR-04-40	LL Ascent Stage Lunar Orbit Departure	3
	VAR-04-41	LL Ascent Stage Mid-course Correction Burn	3
	VAR-04-42	LL Ascent Stage Docks with CEV	2
	VAR-04-43	Crew Transfers from the LL to CEV	2
	VAR-04-44	CEV Separates from LL Ascent Stage	2
	VAR-04-45	CEV Burn for Earth	3
	VAR-04-46	CEV Mid-course Correction Burn	1
	VAR-04-47	CM Separates & Maneuvers away from SM	2
	VAR-04-48	CM Entry	3
	VAR-04-49	CM Landing	2
	VAR-04-50	Crew Recovery	2

Table 14.3-1: TRM with Lunar Orbit Rendezvous Critical Events and Ranking

Given that the Lunar Orbit Rendezvous approach has a fewer number of critical events than the TRM, it will inherently increase the probability of achieving mission success. Not having a Kick Stage attached to the Lunar Lander increases both the probability of achieving mission success as well as crew safety. There may also be additional mission abort opportunities during transit since the EDS-2 will be mated to the CEV and Lunar Lander. Having additional mission abort opportunities will increase crew safety as well.

14.4 Mission Abort Options

LOR aborts will be developed and assessed for each mission phase from low Earth orbit to the lunar surface and the return to the Earth's surface. Earth to orbit ascent aborts are out of the scope of this particular study. Nominal mission flight regimes have been identified along with the critical events previously identified in Section 14.3 (Safety & Mission Success).

14.4.1 LOR Abort Selection

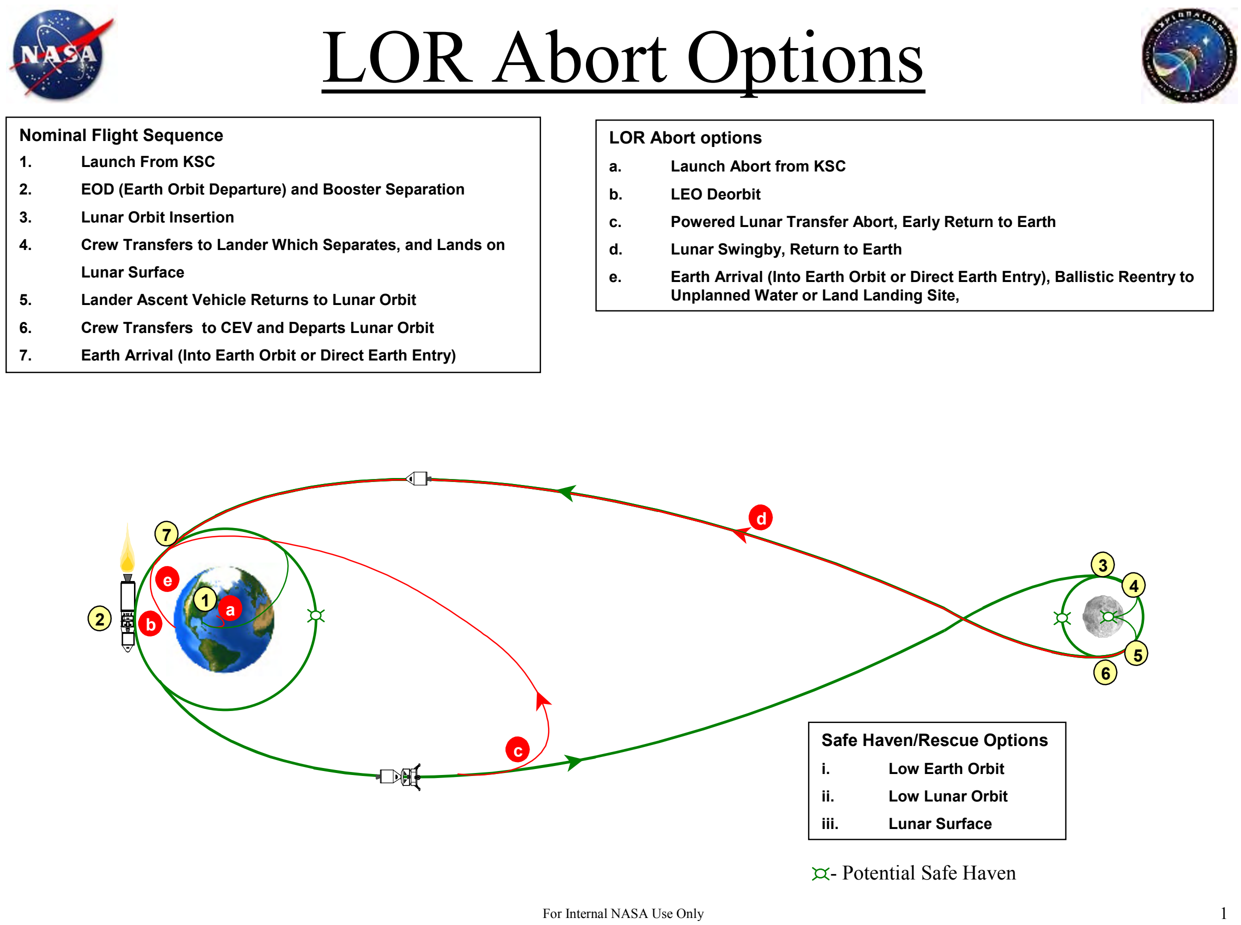


Figure 14.4.1-1: LOR Abort Options

The chart above depicts the crew survival options for the lunar orbit rendezvous mission architecture. The aborts selected for this LOR Trade Reference Mission (TRM) addresses those aborts occurring after CEV launch which result from an inability to complete a critical event required by the LOR mission architecture. Other system failures or problems with the crew may lead to a decision to abort the mission but those aborts can be readily accomplished by moving forward into the next mission phase or bypassing certain mission phases when necessary and completing a safe return to Earth transfer. The following aborts are described for each flight regime of the LOR architecture.

1. Launch and Ascent to Low Earth Orbit (LEO)

This mission phase begins with the launch from Earth surface and ends after the Crew Exploration Vehicle (CEV) is established in the desired LEO.

a. Booster or major CEV system failure

i. CEV emergency separates and returns to Earth

During the CEV launch and ascent to LEO should the Expendable Launch Vehicle (ELV) booster or the CEV suffer catastrophic failure the CEV or ground control can initiate the Launch Abort System, triggering an emergency separation from the ELV and return to Earth using the CEV descent and touchdown systems.

2. LEO Orbit And Rendezvous Operations

This mission phase begins after the CEV is in LEO and ends after the completion of any LEO rendezvous and mating of the Earth Departure Stages, CEV and Lunar Lander.

a. CEV systems failure or failure to mate to Lunar Lander and Earth Departure Stage (EDS)

i. CEV de-orbit and return to Earth

Once the CEV has reached LEO, should the CEV suffer a significant system failure prior to initiating the EDS L1 transfer burn the CEV must perform a standard de-orbit maneuver, re-enter the Earth's atmosphere and successfully touchdown on land or water. If the abort takes place after the CEV mates to the EDS the CEV must separate from the Lunar Lander and EDS prior to re-entry. If CEV propulsion system failures preclude performing a de-orbit maneuver, the Lander or the EDS could be used for that deorbit maneuver. Otherwise the CEV is stranded in LEO and an Earth based CEV rescue mission is required to prevent a loss of crew (LOC) event from occurring. The CEV would need the appropriate resources to provide this safe haven for the crew until that rescue mission is performed (TBD-weeks).

3. LEO to Lunar Transfer

This phase begins at the LEO to LLO departure burn and ends just prior to the lunar orbit insertion burn.

a. Early EDS shutdown and high elliptical orbit

i. CEV maneuver to desired orbit

ii. De-orbit burn and re-entry to touchdown

Should the EDS fail to fully complete the lunar departure burn the CEV/Lander can separate, perform any required transfer orbit adjustments within the limits of available CEV or Lander propulsion constraints, establish a return to Earth trajectory and perform a CEV de-orbit and re-entry to touchdown. After completion of the lunar transfer burn the CEV can also abort by eliminating the Lunar Orbit Insertion (LOI) burn, completing a lunar swingby and returning to Earth on a return transfer orbit. The CEV can adjust this orbit within CEV or Lunar Lander propulsion constraints (if still mated) to ensure a safe Earth re-entry and touchdown.

4. Lunar Orbit Insertion

This phase begins at the start of the LOI burn and ends with the circularization of the lunar orbit.

a. No LOI burn

i. CEV/Lander swingby and return to Earth

If the combined CEV/Lander is not successful in completing the LOI burn then the vehicle must be capable of performing a lunar swingby maneuver and returning to Earth. This can be accomplished by using either the Lander or CEV propulsion systems.

b. Partial LOI burn

i. Ascent stage Delta-V maneuver and return to Earth

If the CEV/Lander partially completes the LLO insertion burn the Lander propulsion stages could be used to complete the insertion burn and then perform the LLO to Earth transfer burn if within the Lander descent or ascent stage propellant budgets. The CEV could also be used to perform the return to Earth maneuver. If CEV failures are known that would preclude the safe execution of a return to earth transfer maneuver, the Lander must be used to adjust the lunar trajectory to perform a lunar swingby and establish the CEV on a safe return to Earth transfer.

5. Lunar Orbit Operations

This phase begins with crew transfer from the CEV to the Lunar Lander and ends continues through CEV/Lander demate and separation from the CEV.

a. Inability to transfer crew from CEV to Lander

i. The CEV separates from the Lander and performs a nominal return to Earth burn. The Lander could also perform the return to earth burn with CEV/Lander separation occurring sometime before CEV reentry.

b. Inability to demate CEV and Lander

i. CEV returns to Earth

The crew returns to the CEV, performs the return to Earth burn, and performs and emergency CEV/Lander separation. Either the CEV or the Lander can perform the return to Earth burn.

c. Inability to perform Lander separation maneuver

i. CEV re-rendezvous and mate with Lander

The CEV, as the active vehicle, will re-rendezvous and mate with the Lander. The crew transfers back to the CEV and performs a nominal return to Earth burn.

6. LLO to Powered Descent Initiation

This phase begins at the start of the lunar descent transfer orbit maneuver and ends just prior to the Powered Descent Initiation burn.

a. No lunar descent transfer orbit burn

i. Lander returns to CEV

During the Lander de-orbit and descent to the lunar surface if any non-propulsion related failure causes an abort, the Lander descent stage will be used to return to LLO and rejoin the CEV. The CEV can then perform the nominal LLO to Earth transfer burn. If the Lander cannot complete the de-orbit to the powered descent initiation point then the Lander can abort using the remainder of the descent stage or the ascent stage to return to LLO and rejoin the CEV.

b. Partial lunar descent transfer orbit burn

i. Lander ascent return to LLO

If a partial de-orbit burn is performed, the Lander ascent stage will return to LLO and rejoin the CEV.

7. Powered Descent Initiation to Lunar Surface

This phase begins at the start of the powered descent initiation burn and ends at lunar surface touchdown.

a. No powered descent

i. Lander ascent stage return to LLO

If the powered descent maneuver is not initiated then the Lander can use either the remainder of the descent stage or the ascent stage to return to LLO and rejoin the CEV.

b. Descent abort

i. Lander ascent stage return to lunar descent transfer orbit

If the need to abort the lunar touchdown occurs late in the powered descent phase, the Lander ascent stage will be used to return to the lunar descent transfer orbit and rejoin the CEV.

8. Lunar Surface Operations

This phase begins just after touchdown, encompasses all lunar surface activities and ends just prior to lunar ascent.

a. EVA suit failures

i. Emergency ingress from EVA

During Lunar surface operations the crew must have the ability to rapidly ingress the Lander from a lunar surface EVA to protect against EVA suit failures. This requires the ability to rapidly transit from any EVA site back to the Lander and re-enter the Lander pressurized volume without extensive stays in any airlock. For long distance EVA sites a pressurized rover with rapid ingress capability may be required to provide a habitable environment in the event of EVA suit failure. In addition, the Lander must be capable of supporting crew medical emergencies resulting from lunar surface operations including the ability to ingress, treat and transport injured crewmembers.

9. Lunar Ascent to LLO

This phase begins at lunar ascent initiation and ends when the Lander has achieved the desired Lunar Orbit.

a. No lunar liftoff

i. Long duration safe haven until Earth based rescue mission arrives (TBD weeks) or LOC

ii. Predeploy extended stay safe haven resources near touchdown site

If the Lander ascent stage fails to ignite then the crew is stranded on the lunar surface and must wait for an Earth based rescue mission. To prevent a LOC event requires the ability for a long duration (TBD weeks) safe haven on the lunar surface, which will require predeployment of safe haven resources near the touchdown site.

b. Failure to reach LLO

i. No functional failure allowed; the Lander ascent stage must reach safe lunar orbit or LOC will occur, physical and functional redundancy is required

After lift off from the lunar surface, the Lander ascent stage must reach a safe LLO or a LOC event will occur. Physical or functional redundancy in the Lander ascent stage is required to ensure that the lunar ascent to LLO is successfully completed.

10. LLO Orbit and Rendezvous Operations

This mission phase begins after the vehicle is in lunar orbit and ends after the completion of any rendezvous and mating of the Lunar Ascent stage with the CEV.

a. Inability to maneuver onorbit (Lander)

i. Passive and Active vehicle exchange roles

If either the Lander ascent stage is unable to maneuver then it becomes the passive vehicle and the CEV becomes the active maneuvering vehicle.

b. Failure to mate Lander and CEV

i. EVA crew transfer to CEV

If the Lander ascent stage and CEV are unable to mate there must be a way to allow the crew to perform an EVA transfer to the CEV for the return to Earth transfer. Otherwise a LOC event will occur.

11. LLO to Earth Transfer

This phase begins with the CEV lunar orbit departure burn and ends just prior to Earth atmospheric re-entry.

a. No LLO departure burn

i. LLO safe haven operations until Earth based rescue

Upon reaching LLO if the CEV is unable to perform the LLO to Earth transfer burn then the crew is stranded in LLO until an Earth based rescue mission arrives or a LOC event occurs. The CEV will require enough resources to accommodate the long duration safe haven (TBD weeks) for the crew.

12. Earth Re-entry to Touchdown (direct entry)

This phase begins with the direct re-entry into Earth atmosphere and ends with CEV touchdown on the Earth surface.

a. Re-entry flight control failures

i. Ballistic re-entry (no lift vector control)

The only abort addressed for the Earth re-entry to touchdown phase is the possibility of performing a passive (zero lift) re-entry. This abort will be possible only if the Earth return trajectory allows the re-entry g-levels to remain below the human tolerance limits during the passive re-entry. Otherwise a lift vector controlled trajectory would be required to lower the g loads on the crew and if the CEV lost all control during re-entry a LOC event might occur if the human limits are exceeded.

b. Entry targeting failures

i. Water or land touchdown

CEV equipped with appropriate crew survival and search and rescue equipment for touchdown site

The CEV will be designed to support either land or water touchdown allowing for entry targeting or control system failures that may force the CEV to miss the desired touchdown site. The LOR architecture study is using 3 hr as the time required to find and rescue the crew from the CEV after touchdown on the Earth.

- OR -

13. Earth Aerocapture to LEO

This phase begins with CEV re-entry into Earth atmosphere, encompasses CEV aerobraking into the desired LEO operations and ends just prior to the CEV final de-orbit burn.

a. Failure to aerocapture and circular burn (elliptical orbit)

i. Delta-V maneuver to appropriate orbit with physical or functional redundancy

ii. Safe haven until Earth based rescue or natural orbital decay

iii. Passive control/ballistic re-entry

For missions designed to use aerobraking to LEO instead of a direct entry, a failure to successfully complete the aerocapture leads to the following aborts. If the aerocapture fails to produce the desired LEO, available CEV propulsion can be used to provide the desired orbit. In addition, the CEV may be designed to allow for a passively controlled ballistic re-entry using the aerobrake heat shield in addition to the CEV. Once in LEO the CEV could provide a safe haven for TBD weeks until an Earth based rescue could be performed.

b. Failure to aerocapture (escape trajectory)

i. LOC

If aerocapture failure results in an atmospheric skip out and corresponding Earth escape trajectory, a LOC event will occur. Physical or functional redundancy must be provided to ensure that the CEV is safely captured into LEO.

14. De-orbit to Touchdown

This phase begins with the CEV de-orbit burn and ends with CEV touchdown on the Earth's surface.

a. No de-orbit

i. Safe haven until rescue, orbital decay, or LOC

After reaching a safe LEO if the CEV fails to perform the de-orbit maneuver there is a LOC unless the CEV can provide a safe haven for TBD weeks until an Earth based rescue can be performed.

b. Re-entry flight control failures

i. Passive re-entry (no lift vector control

After a successful de-orbit burn the CEV will have the capability to perform a ballistic re-entry in the event a nominal re-entry is not possible.

c. Entry targeting and control failures

i. Water or land touchdown

CEV with appropriate crew survival and search and rescue equipment for touchdown site.

The CEV will be designed to support either land or water touchdown allowing for entry targeting failures to force the CEV to miss the desired touchdown site. The LOR architecture is using 3 hr as the time required to find and rescue the crew from the CEV after touchdown.

14.4.2 Abort Timelines

The figures below depict the abort timelines for both the L1 and Lunar Orbit Rendezvous (LOR) variant of LDRM–2. The chart shows the time required to return the crew to Earth and to the CEV as a function of when the abort is initiated during the nominal mission elapsed time.

After launch of the CEV and while still in LEO, for the first 3-4 days the return to Earth time remains constant at close to 3.5 hr. This is the nominal time required to execute the de-orbit maneuver, re-enter the atmosphere, touchdown and be rescued by ground search and rescue forces. Upon completion of the Earth departure burn, the CEV is placed into a 94-hr transfer orbit to L1 or a 96-hr direct transfer to the Moon in the LOR variant. Assuming no propulsive maneuvers to modify the transfer orbit period to reduce the transfer time, the CEV will take about twice the nominal transfer time to return to Earth. This is about 191.5 hr for the L1 option and 195.5 hr for the LOR variant. As the CEV progresses toward L1 or the moon, the return to Earth time is correspondingly reduced until arrival at either L1 or the moon. The return to Earth time becomes the same as the initial transfer time plus the additional 3.5 hr of Earth recovery time.

The L1 architecture requires the largest return to Earth abort time immediately after the Lunar Lander departs L1 for the Moon in a 60-hr transfer leg. At this point, the crew is some 247.5 hr away from the Earth. This time is comprised of 120-hr transfer trajectory to and from the Moon to L1, 30 hr of rendezvous, mating and crew transfer operations at L1, a 94-hr transfer back to Earth plus 3.5 hr recovery operations.

As can be seen from the chart the LOR variant does not have this dramatically increased return to Earth abort time since the CEV remains in low lunar orbit (about 30 hr away, lunar ascent plus LLO rendezvous and mating time) while the Lunar Lander completes the surface mission. After leaving the lunar surface in the L1 architecture, the crew remains about 190 hr away from Earth. This is comprised of the 60-hr Moon to L1 transfer time, 30 hr of L1 operations time, 94 hr of L1 to Earth transfer time plus the 3.5 hr of Earth rescue time. For the LOR variant after leaving the lunar surface the crew are 30 hr away from the CEV and then 99.5 hr away from Earth recovery.

The conclusion is that the LOR variant provides much better return to Earth abort time than the L1 architecture. In addition, the LOR variant has a shorter nominal crew mission elapsed time than the L1 architecture. From a crew survival standpoint the LOR variant is more favorable to crew survival than the L1 option since the crew is exposed to the mission for fewer hours total and the abort return to Earth times are generally lower across the various mission phases.

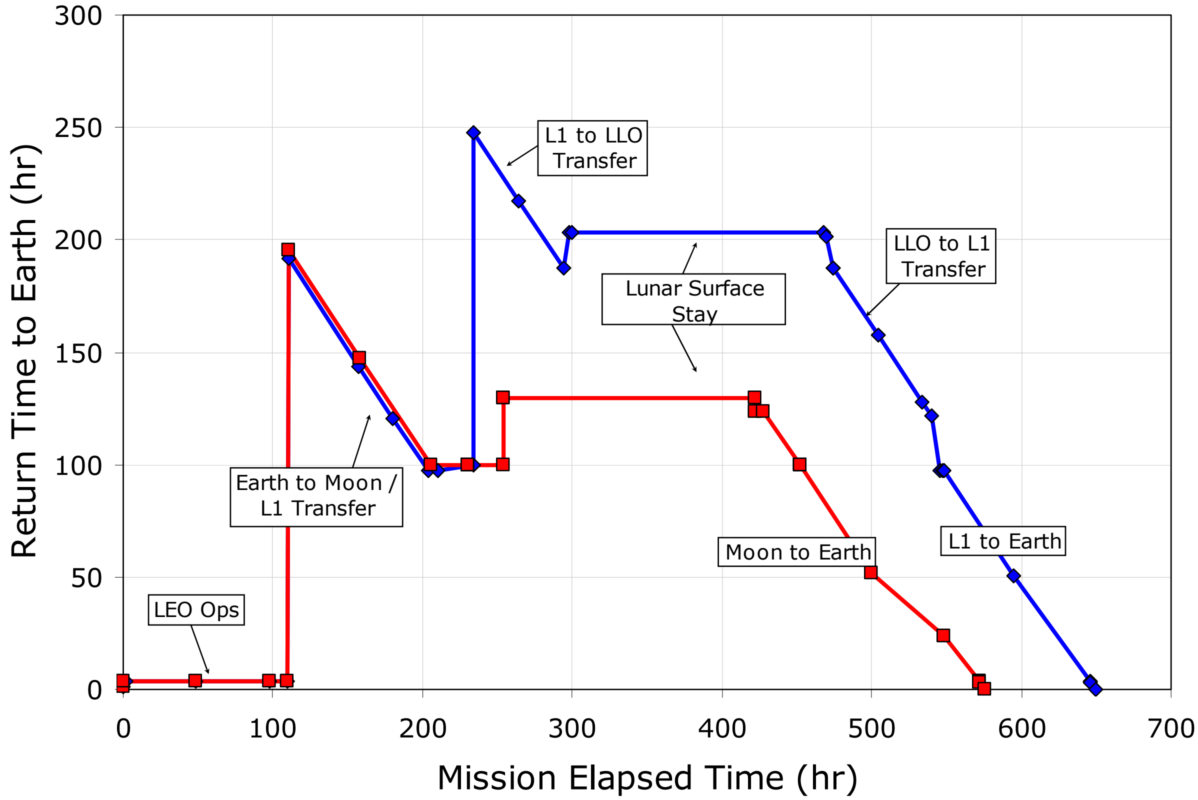


Figure 14.4.2-1: Maximum Return Time to Earth Comparison

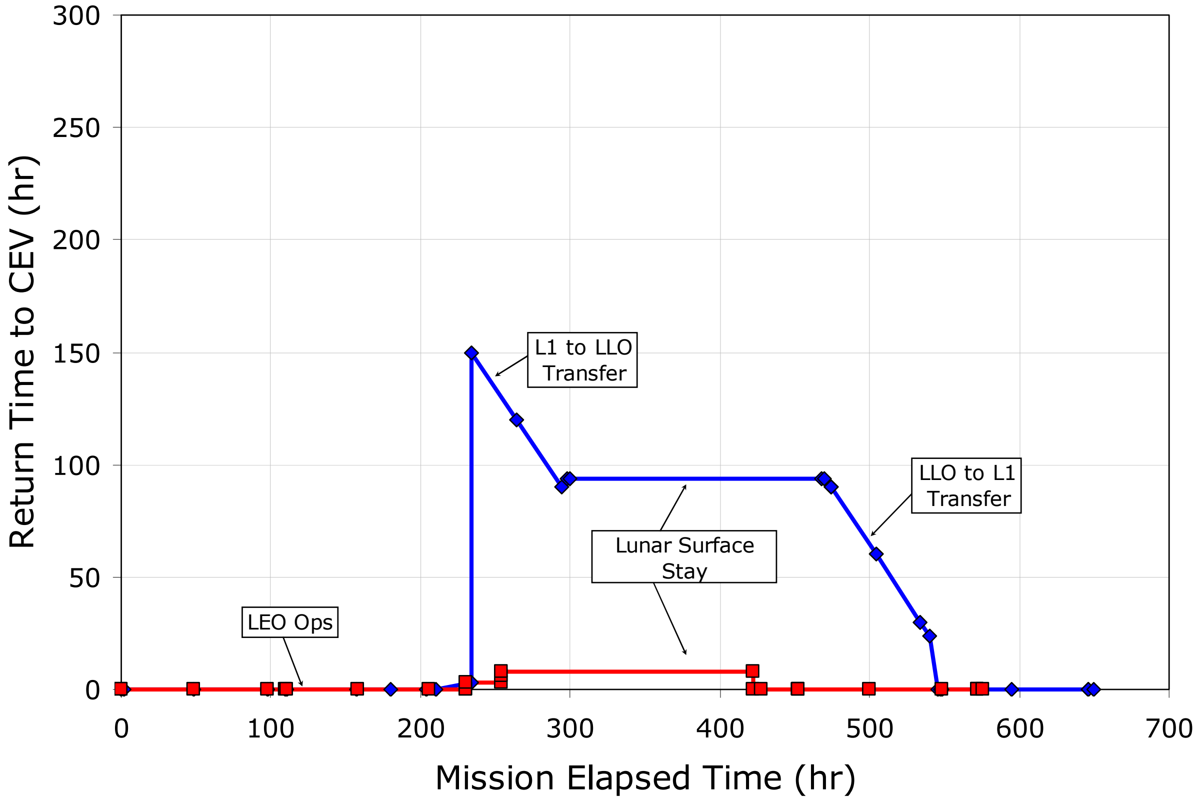


Figure 14.4.2-2: Maximum Return Time to CEV Comparison

14.5 Element Overview & Mass Properties

This section describes any changes made in sizing the trade reference mission elements and compares the resulting vehicle mass properties. The total architecture mass for the lunar orbit rendezvous architecture variant is estimated at 199 t, a 31 t decrease from the TRM.

14.5.1 Crew Exploration Vehicle

The primary differences in the CEV for the lunar orbit rendezvous architecture compared to the TRM are the total mission duration, crew time in the vehicle, and total delta-V. The total mission duration for the TRM CEV is 73 hr longer than the LOR CEV, which increases the consumables required for power generation. However, the total crew time in the CEV is 46 hr longer for the LOR variant due to the two 3-impulse, 24-hour sequences on the outbound and inbound transfer legs. This increases the consumables required for crew support on the LOR CEV. Finally, the TRM CEV carries more OMS delta-V than in the LOR variant. The total delta-V in the TRM is 1,911 m/s, which includes rendezvous delta-V, libration point arrival, and libration point departure. The total delta-V in the LOR architecture is only 1,569 m/s for rendezvous in LEO and lunar orbit departure. Therefore, the LOR CEV should require significantly less propellant and propulsion mass than in the TRM. No other hardware changes were made to the CEV.

CEV Crew Module's System Mass Changes				
System	**TRM**	**LOR**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	812	(10)	(1.2)
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	670	(21)	(3.0)
Other	835	831	(4)	(0.5)
Growth	1041	1035	(6)	(0.6)
Non-Cargo	966	896	(70)	(7.2)
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	57	2	3.6
Propellant	64	64	No Change	0.0
Total	8812	8702	(110)	(1.2)

Table 14.5.1-1: Variation in CEV CM Mass with LOR

CEV Service Module's System Mass Changes				
System	**TRM**	**LOR**	**Mass Change (kg)**	**% Change**
Structure	1455	1430	(25)	(1.7)
Protection	0	0	No Change	0.0
Propulsion	1408	1217	(191)	(13.6)
Power	661	653	(8)	(1.2)
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	97	(13)	(11.8)
Other	100	100	No Change	0.0
Growth	747	700	(47)	(6.3)
Non-Cargo	305	230	(75)	(24.6)
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1396	(46)	(3.2)
Propellant	11332	8509	(2823)	(24.9)
Total	17560	14332	(3228)	(18.4)

Table 14.5.1-2: Variation in CEV SM Mass with LOR

14.5.2 Lunar Lander

Like the CEV, the Lunar Lander differs from the TRM primarily in total mission duration, total crew time in the vehicle, and descent/ascent delta-V. The TRM Lander has a total mission duration of 1,466 hr, which compares to 1,030 hr for LOR. This difference is due to reordering of the launch sequence and elimination of 120 hr for transit to and from Lunar L1. With the TRM, the Lunar Lander was pre-deployed to L1, so it was launched prior to the CEV Earth Departure Stage. Here, all elements assemble in LEO and the two Earth Departure Stages are mated prior to the Lander launch. The TRM Lander also has a total crew time 120 hr longer than the LOR Lander, which is equivalent to the time required for transit to and from L1. However, the most substantial change between the two vehicle designs is in the total delta-V required. With the TRM, the Descent Stage only performed in-plane powered descent from a 100 km lunar parking orbit, and this variant allocates the same functionality to that stage. The Ascent Stage, on the other hand, performed in the TRM powered ascent, lunar orbit departure, and libration point arrival. For lunar orbit rendezvous, only powered ascent is required. This greatly reduces the amount of propellant carried on the stage, which decreases the Ascent Stage mass and results in a lower Descent Stage mass even though the delta-V's are equal.

One final change was made to the Ascent Stage configuration. In the TRM, required engine-out capability and the desire to minimize gravity losses resulted in a four-engine, 7,500 lbf per engine configuration. With a lower stage mass at ignition in the LOR variant, only three engines are required to provide equivalent performance capabilities.

Lander's Ascent Stage's System Mass Changes				
System	**TRM**	**LOR**	**Mass Change (kg)**	**% Change**
Structure	839	804	(35)	(4.2)
Protection	73	73	No Change	0.0
Propulsion	1631	1194	(437)	(26.8)
Power	813	763	(50)	(6.2)
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	924	73	8.6
Other	455	455	No Change	0.0
Growth	1080	990	(90)	(8.3)
Non-Cargo	1483	1310	(173)	(11.7)
Cargo	227	227	No Change	0.0
Non-Propellant	1014	857	(157)	(15.5)
Propellant	10703	6993	(3710)	(34.7)
Total	19906	15328	(4578)	(23.0)

Table 14.5.2-1: Variation in Lander Ascent Stage Mass with LOR

Lander's Descent Stage's System Mass Changes				
System	TRM	LOR	Mass Change (kg)	% Change
Structure	553	522	(31)	(5.6)
Protection	50	50	No Change	0.0
Propulsion	1413	1172	(241)	(17.1)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	583	(125)	(17.7)
Growth	678	599	(79)	(11.7)
Non-Cargo	464	369	(95)	(20.5)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	13991	(3582)	(20.4)
Total	22608	18453	(4155)	(18.4)

Table 14.5.2-2: Variation in Lander Descent Stage Mass with LOR

14.5.3 Kick Stage

The Kick Stage has been eliminated from the lunar orbit rendezvous architecture.

14.5.4 Earth Departure Stages

For the reasons discussed previously, the LOR architecture variant will incorporate element assembly in LEO rather than Lander pre-deployment as is used in the TRM. This means that two equal-mass Earth Departure Stages are used to perform Earth orbit departure for the mated Lander and CEV, whereas in the TRM one EDS was used for the mated Lander and Kick Stage and one EDS for the CEV. Additionally, the second of the two EDS's in this variant performs lunar orbit insertion for the mated elements, while in the TRM, the Kick Stage and CEV performed libration point arrival themselves. This change was made because it is more efficient to use for insertion the higher performance oxygen/hydrogen EDS for the 57 t combined Lander and CEV mass than either the lower performance propulsion systems on the Lander or CEV. In the TRM, it was less efficient to use the EDS for libration point arrival as an oxygen/hydrogen system was included on the Kick Stage and with the CEV, staging benefits offset the loss of propulsive performance – the CEV mass was sufficiently small such that the decrease in specific impulse had minimal impact. Using the EDS for lunar orbit insertion will add another engine restart to the stage requirements, though.

Another impact on the Earth Departure Stage design for this variant comes with Lander/CEV mating in LEO. Having two stages assembled together results in at least one of those two requiring two mating interfaces – one interface to the other EDS and one interface to the Lunar Lander. The TRM EDS only required one interface to either the Lander or the CEV. Finally, the maximum total on-orbit mission duration for the EDS in this option is 627 hr longer than in the TRM. This is primarily due to the reordering of the launch sequence and the use of the EDS for lunar orbit insertion, and it results in slightly higher propellant boil-off.

The stages in this variant are scaled in length to accommodate the necessary changes in propellant quantities from the TRM, however no changes were made in hardware selection other than those described above.

Lander Earth Departure Stage's System Mass Changes				
System	**TRM**	**LOR**	**Mass Change (kg)**	**% Change**
Structure	1972	1606	(366)	(18.6)
Protection	0	0	No Change	0.0
Propulsion	4361	3737	(624)	(14.3)
Power	190	190	No Change	0.0
Control	0	0	No Change	0.0
Avionics	175	175	No Change	0.0
Environment	105	105	No Change	0.0
Other	455	455	No Change	0.0
Growth	1452	1254	(198)	(13.6)
Non-Cargo	3109	2766	(343)	(11.0)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	82289	60622	(21667)	(26.3)
Total	94109	70909	(23200)	(24.7)

Table 14.5.4-1: Variation in Lander EDS Mass with LOR

CEV Earth Departure Stage's System Mass Changes				
System	**TRM**	**LOR**	**Mass Change (kg)**	**% Change**
Structure	932	1606	674	72.3
Protection	0	0	No Change	0.0
Propulsion	2318	3737	1419	61.2
Power	190	190	No Change	0.0
Control	0	0	No Change	0.0
Avionics	171	175	4	2.3
Environment	104	105	1	1.0
Other	455	455	No Change	0.0
Growth	834	1254	420	50.4
Non-Cargo	1355	2766	1411	104.1
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	32897	60622	27725	84.3
Total	39256	70909	31653	80.6

Table 14.5.4-2: Variation in CEV EDS Mass with LOR

14.5.5 Vehicle Mass Properties

Table 14.5.5-1 lists vehicle mass properties for the lunar orbit rendezvous architecture variant, and Figure 14.5.5-1 compares individual vehicle gross mass to the trade reference mission. The largest single element, the two equal-mass Earth Departure Stages for the Lunar Lander and CEV, each have an initial mass in LEO of 70.9 t as compared to 94.1 t and 39.3 t for the TRM. Recall that the TRM pre-deploys the Lander to L1 prior to launching the CEV, whereas the LOR variant assembles the Lander and CEV in low Earth orbit. Next, the Lunar Lander has an initial mass of 33.8 t, respectively, significantly less than the 42.5 t estimate in the TRM. This variant does not require a Kick Stage. Finally, the CEV total mass is estimated at 23.0 t, a reduction of 3,300 kg. The combined architecture elements of the LOR architecture variant have a total IM-LEO of 199 t, compared to 230 t for the trade reference mission. It should be noted that total LOR architecture mass can be reduced by trading anytime return capability for on-orbit loiter time. In this case, the crew could loiter in the CEV in lunar orbit until favorable trans-Earth injection opportunities arise. The CEV presently includes the capability to perform a 90^{o} plane change around the Moon in order to align the orbital plane for Earth return. Without this plane change delta-V, the CEV propellant savings would offset the additional crew consumables required during the wait time.

Figure 14.5.5-2 compares individual launch package masses to the trade reference mission. This architecture variant requires a 71 t payload capability by the cargo launch vehicle to deliver the elements to LEO, less than the 94 t required maximum capability for the TRM. Size requirements for the CEV's human-rated launch vehicle are 23 t, 3 t less than the TRM. The LOR variant requires the same number of launches as the TRM.

	CEV CM	CEV SM	Ascent Stage	Descent Stage	Earth Dep. Stage (x 2)
1.0 Structure	1,523	1,430	804	522	1,606
2.0 Protection	812	0	73	50	0
3.0 Propulsion	117	1,217	1,194	1,172	3,737
4.0 Power	482	653	763	137	190
5.0 Control	0	0	0	0	0
6.0 Avionics	737	0	738	0	175
7.0 Environment	670	97	924	530	105
8.0 Other	831	100	455	583	455
9.0 Growth	1,035	700	990	599	1,254
DRY MASS	6,208 kg	4,197 kg	5,940 kg	3,593 kg	7,522 kg
10.0 Cargo	896	230	1,310	369	2,766
11.0 Non-Cargo	1,478	0	227	500	0
INERT MASS	8,582 kg	4,427 kg	7,478 kg	4,462 kg	10,288 kg
12.0 Non-Propellant	57	1,396	857	0	0
13.0 Propellant	64	8,509	6,993	13,991	60,622
GROSS MASS	8,702 kg	14,332 kg	15,328 kg	18,453 kg	70,909 kg

Table 14.5.5-1: Vehicle Mass Properties

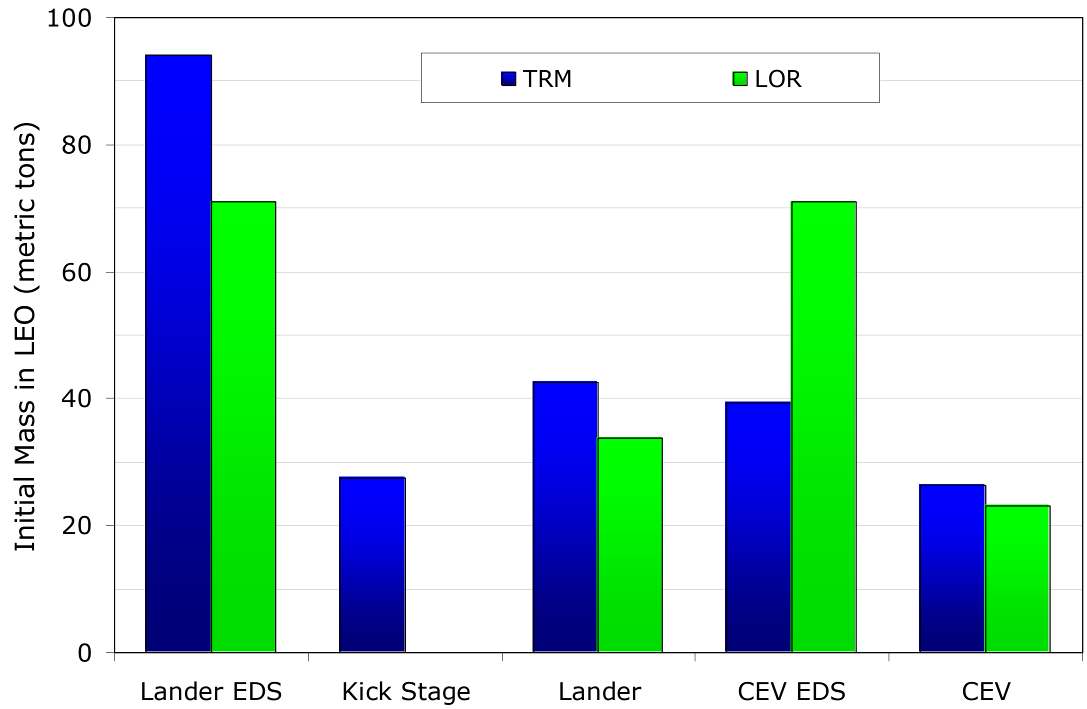


Figure 14.5.5-1: Vehicle Mass Properties Comparison

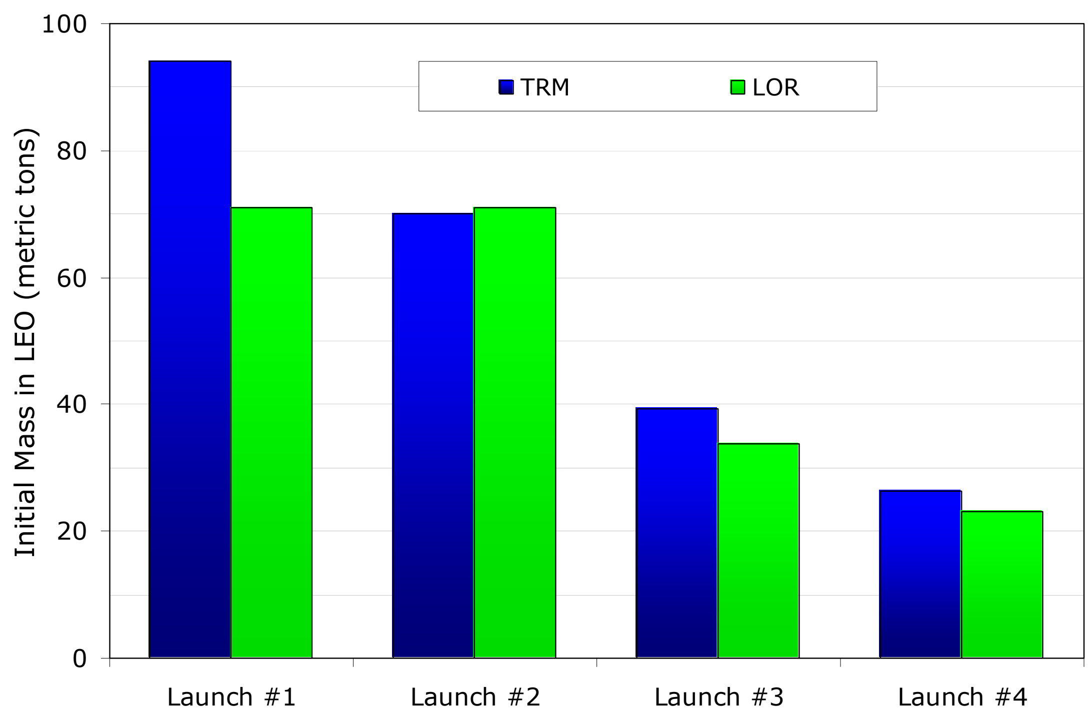


Figure 14.5.5-2: Mass per Launch Comparison

Finally, architecture "gear ratios" for the lunar orbit rendezvous variant have been calculated and are listed below.

Earth Departure Stage #1:	*1.5:1*	*Earth Departure Stage #2:*	*3.2:1*
CEV Service Module:	*5.4:1*	*CEV Crew Module:*	*6.4:1*
Descent Stage:	*6.2:1*	*Ascent Stage:*	*12.5:1*
Round Trip Cargo:	*20.6:1*		

14.6 System Technologies and Programmatic Risks

The analysis performed for this architecture variant did not modify any vehicle system technology assumptions made for the trade reference mission.

14.7 Pros/Cons Summary

The lunar orbit rendezvous architecture variant examined here provides the same benefits as the trade reference mission (global lunar surface access with anytime return) at a lower overall architecture cost. The initial mass in LEO for LOR is 199 t, approximately 31 t (13.5%) less than the TRM. The minimum payload mass required per launch of the cargo launch vehicle is also reduced from 94 t to 71 t, and the human-rated launch vehicle requirement is lowered from 26 t to 23 t. Lunar orbit rendezvous also provides advantages with total mission duration, total crew time in space, and abort timelines. The total mission duration, from launch of the first element to crew touchdown on Earth, is 66 hr shorter, and the total crew time in space is 74 hr shorter. Perhaps more importantly, the length of time required for the crew to return to the CEV in a worst-case abort is 120 hr less than the TRM. In the TRM, a Lander failure shortly after departing L1 may require to fly around the Moon to return to the CEV. With LOR, the longest return time only requires a powered ascent and rendezvous with the CEV in low lunar orbit. The worst-case return time to Earth for the two options is comparable – LOR is 24 hr shorter.

Another possible benefit that this variant provides over the TRM is that assembly of the CEV and Lunar Lander in LEO may enable Apollo 13-esque "Lander lifeboat" abort options. In the event of a CEV failure on the outbound transfer, an attached Lander may provide a life support and communications backup to save the crew. This is not possible with the TRM as the Lander is pre-deployed to Lunar L1. Assembly in LEO also means that Lander checkout and troubleshooting is performed in LEO rather than lunar vicinity, so the crew does not have to leave the relative safety of low Earth orbit until the Lander is operating properly. An attached Lunar Lander also provides the potential for using its volume for additional crew space crew during the outbound transfer.

The lunar orbit rendezvous architecture also enables smaller Lander ascent and descent stages, which may reduce the required launch vehicle payload shroud diameter, may improve lunar landing dynamics, and may reduce development costs.

However, one benefit that the TRM provides is a far simpler assembled stack configuration, primarily a result of pre-deploying the Lander to L1. The on-orbit assembled interfaces for the TRM are one between the Lander EDS and Lunar Lander/Kick Stage, and one between the CEV EDS and the CEV. The LOR architecture has assembled interfaces between EDS1 and EDS2, between EDS2 and the Lunar Lander, and one between the Lunar Lander and the CEV. The four launch per mission nature of the LOR variant requires automatic assembly of multiple uncrewed propulsion stages in LEO. This complex assembled stack configuration adds a staging event in the middle of the critical Earth orbit departure maneuver. The stack dynamics during that maneuver have not been analyzed at this time.

Finally, the LOR strategy selected to enable global lunar surface access with anytime Earth return is highly sensitive in total architecture mass to increasing lunar stay times. For a 7-day surface mission such as baselined for LDRM-2, the required ascent plane change for anytime aborts off the lunar surface is reasonably small – only 6.7^{o} or 191 m/s of extra delta-V. This cost rapidly increases for longer stay missions. A 14-day mission requires a worst-case plane change of 28.1^{o} (794 m/s), and the maximum plane change case, a 21-day surface mission, requires a plane change of 43.9^{o} (1,222 m/s). Libration point rendezvous may be as competitive as or better than LOR in total mass with longer-stay missions such as these.

15.0 TRM with Initial CEV/Lander Mating in LEO

This architectural variation examined the major differences between mating the CEV to a pre-deployed Lander at L1 vs. mating the two elements and their associated EDS's in LEO prior to departure for L1. This approach will be referred to as the Earth Orbit Rendezvous architecture.

15.1 Major Assumptions/Clarifications

This section outlines the major architectural assumptions that differ from the trade reference mission defined in section 10.1. Unless otherwise stated, all other major assumptions remained the same as outlined in section 10.1.

Changed Assumption:

New: The Lunar Lander will be pre-deployed to LEO prior to initiation of the CEV mission.

Old: The Lunar Lander will be pre-deployed to lunar vicinity prior to initiation of the CEV mission.

15.2 Architecture Description

This architecture begins similar to the trade reference mission (TRM) with the launch of an Earth Departure Stage (EDS). This EDS is launched to a 28.5°, 407 km orbit where it loiters for assembly with other elements. Two weeks after this first launch, a second EDS is launched and mates to the first EDS. This launch marks the beginning of the architectural differences with the TRM. The strategy adopted by the TRM was to use the second launch to send the Lunar Lander to LEO, after which it would mate with its EDS and depart for L1. The TRM approach was designed to retire as much risk as possible with deployed assets prior to launching additional assets. Since the TRM relied on a pre-deployed Lunar Lander at L1, which required several critical maneuvers (LEO mating, Earth orbit departure burns, Lunar Lander/EDS separation, and libration point arrival maneuvers), it was desirable to ensure these maneuvers were completed prior to committing to a third launch – the second EDS. In contrast, the Earth Orbit Rendezvous approach requires that the Lunar Lander and CEV perform these maneuvers together; therefore, it is not possible to retire this risk prior to the fourth launch. Thus, given a choice in the order of the second EDS's launch and the Lunar Lander's launch, it was felt that it would be preferable to launch the second EDS first. This would allow the vehicle that would eventually contain humans (therefore more complex and possibly more delicate) to remain on the ground for as long as possible. The issue of H2 boil-off was considered in this approach; however, early data indicated that it did not seem to be a distinguishing factor. Given certain assumptions regarding the design of the H2 tanks, the boil-off was calculated to be on the order of a few 10's of kilograms per week.

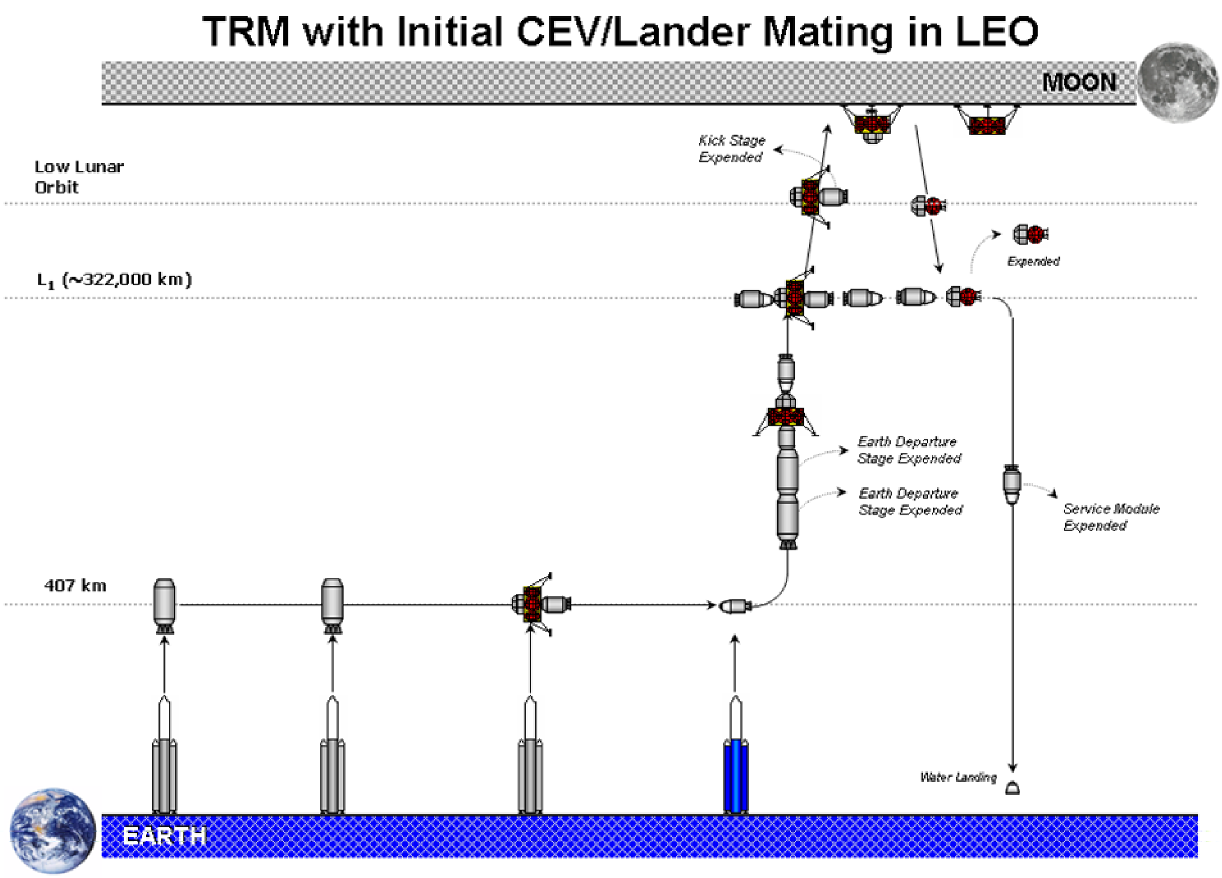


Figure 15.2-1: Earth Orbit Rendezvous Architecture Illustration

Two weeks after the launch of the second EDS the Lunar Lander is launched. As with all LEO rendezvous maneuvers in this architecture, the Lunar Lander has been designed to accommodate a 50 hour, 360° phasing window rendezvous maneuver. Acting as the chaser vehicle, the Lunar Lander will maneuver to the proximity of the EDS stack to execute the mating procedures.

Finally, two weeks after the launch of the Lunar Lander the CEV is launched. Consistent with the TRM, the CEV is nominally launched 4.6 days earlier than the first Earth Orbit Departure (EOD) opportunity in order to accommodate the 50 hour rendezvous procedures and protect against launch/on-orbit delays. Acting as the chaser vehicle, the CEV will maneuver to the proximity of the EDS/Lunar Lander stack to execute the mating procedures.

If launch delays do not allow the CEV to launch on one of the three daily launch opportunities bookkept in the mission timeline, and the first injection to L1 opportunity is missed, the orbital elements shall be designed to handle an extra 10 days of on-orbit lifetime. L1 injection opportunities from the reference LEO assembly orbit arise every 3-12 days, with the average time between window openings being 10 days. Assuming a reasonable chance of missing one opportunity in a four launch per mission architecture, the two EDS's, Lunar Lander, and Kick Stage will be capable of loitering for 10 additional days in LEO.

Subsequent to successful mating, check out, and L1 injection window opening, EDS1 will perform its EOD burn (1,229 m/s). After expending nearly all of its propellants, EDS1 will be jettisoned and dispose of itself. At this point, EDS2 will be ignited and continue to perform the EOD burn (1,875 m/s). Again, after expending nearly all of its propellants, EDS2 will be jettisoned and dispose of itself. At this point, the CEV/Lunar Lander stack will coast on an approximately 94 hour journey to L1. Upon L1 arrival, the stack's Kick Stage will perform the libration point arrival burn (954 m/s). At this point the crew will transfer themselves and any necessary cargo over to the Lunar Lander. From this point forward, the mission is exactly the same as the TRM, described in section 10.2. Figure 15.2-2 outlines the nominal timelines for this mission.

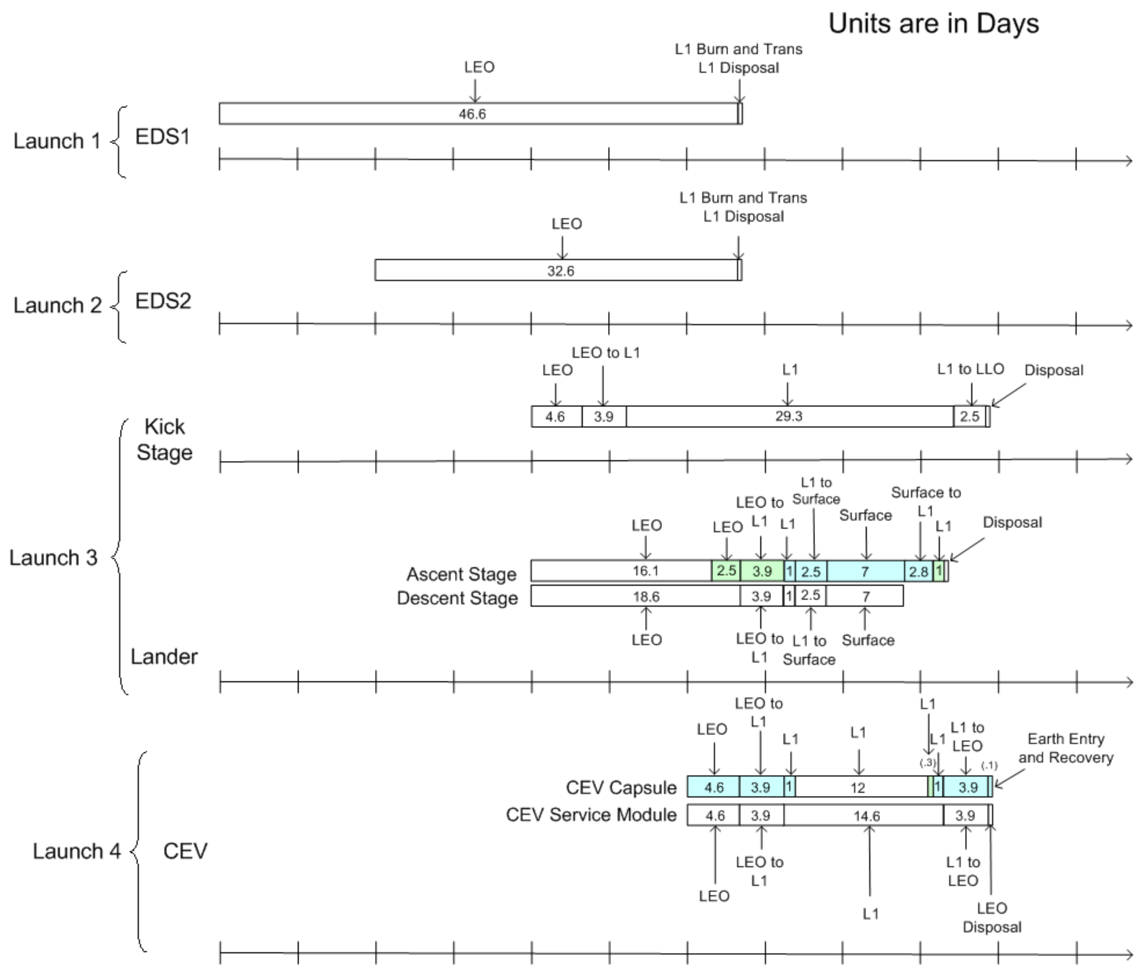


Figure 15.2-2: Nominal Timeline for the Earth Orbit Rendezvous Architecture

15.3 Safety & Mission Success

15.3.1 Critical Event Identification

The Earth Orbit Rendezvous approach has exactly the same number of critical events as the TRM architecture. Both architectures have a total of fifty-six critical events. Twenty of the fifty-six critical events for the TRM occur during uncrewed portions of the mission while the remaining thirty-six occur during the crewed portions of the mission. However, in the Earth Orbit Rendezvous approach, only nineteen of these occur during uncrewed portions of the mission, while the remaining thirty-seven occur during crewed portions of the mission.

15.3.2 Earth Orbit Rendezvous Critical Event Ranking

The critical events for this architectural approach were ranked, in order to understand their severity. Following the ranking scheme of the TRM, the following numbers were assigned to each critical event to signify its level of inherent risk.

Rank of 1: *Failures during mission critical events that could lead to a Loss of Mission (LOM) but not a Loss of Crew (LOC).*

Rank of 2: *Failures during mission critical events that could lead to a LOC but would have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

Rank of 3: *Failures during mission critical events that would not have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

Of the fifty-four critical events identified for the Earth Orbit Rendezvous approach, seven received a rank of three, twenty-four received a rank of two, and the remaining twenty-five received a rank of one. This is the exact same distribution as the TRM's critical event ranking. The complete set of identified and ranked critical events for the Earth Orbit Rendezvous approach is listed in Table 15.3.2-1.

	ID #	TRM with CEV-LL Mating in LEO Critical Events	TRM with CEV-LL Mating in LEO Critical Event Rank
Uncrewed Critical Events	**VAR-04-01**	EDS-1 Launch	1
	VAR-04-02	EDS-1 Ascent	1
	VAR-04-03	EDS-1 Launch Shroud Separation	1
	VAR-04-04	EDS-1 Separation from Booster	1
	VAR-04-05	EDS-1 Orbital Maneuvering	1
	VAR-04-06	EDS-2 Launch	1
	VAR-04-07	EDS-2 Ascent	1
	VAR-04-08	EDS-2 Launch Shroud Separation	1
	VAR-04-09	EDS-2 Separation from Booster	1
	VAR-04-10	EDS-2 Orbital Maneuvering	1
	VAR-04-11	EDS-1 & EDS-2 Dock	1
	VAR-04-12	EDS-1 & EDS-2 Orbital Maneuvering	1
	VAR-04-13	LL Launch	1
	VAR-04-14	LL Ascent	1
	VAR-04-15	LL Launch Shroud Separation	1
	VAR-04-16	LL Separation from Booster	1
	VAR-04-17	LL Orbital Maneuvering	1
	VAR-04-18	LL Docks to EDS-1 & EDS-2	1
	VAR-04-19	EDS-1, EDS-2, & LL Orbital Maneuvering	1
Crewed Critical Events	**VAR-04-20**	CEV (CM+SM) Launch	2
	VAR-04-21	CEV Ascent	2
	VAR-04-22	LAS Separation	2
	VAR-04-23	CEV Launch Shroud Separation	2
	VAR-04-24	CEV Separation from Booster	2
	VAR-04-25	CEV Orbital Maneuvering	2
	VAR-04-26	CEV Docks to EDS-1, EDS-2, & LL	2
	VAR-04-27	EDS-1, EDS-2, LL, & CEV Burn for L1	2
	VAR-04-28	EDS-1 Separates from EDS-2, LL, & CEV	2
	VAR-04-29	EDS-2, LL, & CEV Mid-course Correction Burn	1
	VAR-04-30	EDS-2 Separates from LL & CEV	2
	VAR-04-31	LL & CEV Mid-course Correction Burn	1
	VAR-04-32	LL & CEV Burn to Slow Near L1	2
	VAR-04-33	LL & CEV Orbital Maneuvering	2
	VAR-04-34	Crew Transfers from the CEV to LL	1
	VAR-04-35	LL Separate from CEV	2
	VAR-04-36	LL Burns for Low Lunar Orbit	2
	VAR-04-37	LL Mid-course Correction Burn	1
	VAR-04-38	LL Lunar Orbit Insertion (LOI)	2
	VAR-04-39	Kick Stage Separates from LL	2
	VAR-04-40	LL Deorbit Burn to Moon	2
	VAR-04-41	LL Powered Descent & Landing to the Moon	3
	VAR-04-42	LL Ascent Stage Separation & Ascent	3

	ID #	TRM with CEV-LL Mating in LEO Critical Events	TRM with CEV-LL Mating in LEO Critical Event Rank
	VAR-04-43	LL Ascent Stage Orbital Maneuvering	3
	VAR-04-44	LL Ascent Stage Lunar Orbit Departure	3
	VAR-04-45	LL Ascent Stage Mid-course Correction Burn	1
	VAR-04-46	LL Ascent Stage L1 Arrival	3
	VAR-04-47	LL Ascent Stage Orbital Maneuvering	2
	VAR-04-48	LL Ascent Stage Docks with CEV	2
	VAR-04-49	Crew Transfers from the LL to CEV	2
	VAR-04-50	CEV Separates from LL Ascent Stage	2
	VAR-04-51	CEV Burn for Earth	3
	VAR-04-52	CEV Mid-course Correction Burn	1
	VAR-04-53	CM Separates & Maneuvers away from SM	2
	VAR-04-54	CM Entry	3
	VAR-04-55	CM Landing	2
	VAR-04-56	Crew Recovery	2

Table 15.3.2-1: Earth Orbit Rendezvous Critical Events and Rank

There are only very slight differences that can be noted when the Earth Orbit Rendezvous critical event list is compared against the TRM's critical event list. Critical event VAR-04-28, which has a rank of "2", received a rank of only "1" in the TRM listing. This is because this operation is performed with the crew on-board the CEV in the Earth Orbit Rendezvous approach; whereas, the analogous procedure was performed prior to the launch of the crew in the TRM. There was also a slight difference between the two approaches in that the TRM required two different docking procedures before the crew obtained access to the Lunar Lander; whereas, the Earth Orbit Rendezvous approach only requires one docking procedure. In the TRM approach, the first mating occurs in LEO where the CEV mates to EDS2 (TRM-27). After arriving at L1, the CEV mates to the Lunar Lander, which was pre-deployed to L1 (TRM-33). The Earth Orbit Rendezvous approach allows this procedure to be accomplished in one event, VAR-04-26, in which the CEV mates to the entire stack that is pre-deployed to LEO.

Aside from these slight differences, which do not seem to provide any strong rationale for performing the mission one way versus the other, all other critical events are the same. All critical events with a rank of "3" were exactly the same and have the same concerns and mitigation plans as outlined in sections 10.3.4 and 10.3.5 of this report.

15.4 Mission Abort Options

15.4.1 Mission Abort Comparison to the TRM

The mission abort options for the Earth Orbit Rendezvous approach differ for flight phases two through four, as defined in section 10.4.3. They remain constant for phase one and phases five through fourteen.

Flight phase two is defined as the "LEO Orbit and Rendezvous Operations" phase in section 10.4.3. The two failures described for this phase in section 10.4.3 are possible CEV system failures and the failure to mate with EDS2. The only operational difference between the TRM and the Earth Orbit Rendezvous approach for this phase is in the brief discussion of CEV propulsion system failures. In this case, section 10.4.3 indicates that EDS2 could be used in order to perform the de-orbit maneuvers necessary to allow the CEV to enter the Earth's atmosphere (after the CEV and EDS2 re-mate to each other), if the TRM approach were employed. However, in the Earth Orbit Rendezvous approach, with the same CEV propulsion system failure, either the combined CEV/Lunar Lander/Kick Stage/EDS1/EDS2 stack would be required to perform this maneuver or one of the EDS's would need to break away from the rest of the stack, mate to the CEV, and then perform this maneuver.

A second failure that could occur during this phase of the Earth Orbit Rendezvous approach, that does not have the opportunity to occur until flight phase four of the TRM is in regards to the mating of the CEV with the Lunar Lander. In the event that the CEV and Lunar Lander cannot successfully mate, the CEV would perform the appropriate de-orbit maneuvers as previously discussed.

Flight phase three is defined as the "LEO to L1 Transfer" phase in section 10.4.3. The failure described for this phase in section 10.4.3 is an early EDS shutdown, leaving the CEV in a highly elliptical orbit. In this situation, the Earth Orbit Rendezvous approach provides a greater amount of flexibility in returning the crew to Earth. The Earth Orbit Rendezvous approach uses two EDS's to perform the complete EOD burn. If the failure occurs during the burn of the first EDS, the crew could jettison this EDS and use the second EDS, Lunar Lander, or Service Module to perform transfer orbit adjustments to bring the CEV back to the desired orbit at Earth. If this failure occurred during the burn of the second EDS, the crew could jettison this EDS and then use either the Lunar Lander or the Service Module to perform the desired transfer orbit adjustments. In either case, there is a greater amount of delta V available to the crew to inject into the desired Earth return orbit.

Another scenario that could be protected in the Earth Orbit Rendezvous architecture is an Apollo 13-type contingency. In the Earth Orbit Rendezvous approach, the Crew Module is attached to the Lunar Lander prior to EOD. Therefore, the crew has two fully independent human-rated vehicles at their disposal. Although it is impossible to anticipate exactly what failure would occur, it is a reasonable expectation that this extra spacecraft could be used temporarily to provide redundant crew volume and supplemental resources.

Flight phase four is defined as the "L1 Operations" phase in section 10.4.3. The first failure outlined in section 10.4.3 for this phase is the failure of the CEV's Service Module to perform the L1 arrival burn. In the Earth Orbit Rendezvous approach, the Kick Stage performs the L1 arrival burn, as opposed to the Service Module in the TRM. If the Kick Stage fails to perform this maneuver, the CEV could perform a swing-by at L1, jettison the Lunar Lander/Kick Stage and use the Service Module to establish a safe Earth-return trajectory.

Finally, section 10.4.3 describes a failure in which the crew is unable to transfer from the CEV to the Lunar Lander in order to begin their transfer to the Moon. Most likely, this failure is a non-issue in the Earth Orbit Rendezvous approach. It is likely that the crew would have already double-checked the hatch and corridor that connects the CEV to the Lunar Lander prior to EOD. If an anomaly were to arise during that check, the problem would have either have been fixed in LEO or the mission would have been aborted. However, if an anomaly that prevented the crew from transferring over to the Lunar Lander were encountered during this mission phase, the crew would return to the CEV, separate from the Lander, and perform the nominal L1 departure burn.

15.4.2 Return Time to Earth and Return Time to CEV

The timelines for returning the crew to Earth and to the CEV are identical to those specified in section 10.4.4 and illustrated in Figures 10.4-1 and 10.4-2.

15.5 Element Overview & Mass Properties

This section describes any changes that were made to the TRM elements in order to accommodate this architectural variation. The changes due to these modifications, if any, were evaluated with the use of the Envision sizing tool. Unless otherwise stated in this section, there were no changes to the TRM elements described in section 10.5.

15.5.1 Crew Exploration Vehicle

As described in section 10.5.1, the CEV is comprised of a Crew Module and a Service Module. The CEV Crew Module did not require any modifications in order to meet the requirements of this architectural variation. However, there was a significant change in the size of the Service Module, due to a change in the maneuvers it is required to perform in the Earth Orbit Rendezvous architecture.

The L1 TRM first deployed the Lunar Lander to L1. In order to perform this deployment, the Lunar Lander mated with an EDS in LEO. The EDS was used to perform the EOD burn with a delta V of 3,104 m/s. After disposal of the EDS, the Lunar Lander continued to L1, where it used its Kick Stage to perform an L1 arrival maneuver of 954 m/s. A similar approach was used to deliver the CEV to L1. The main difference between the two vehicles was that the CEV used its Service Module to perform the L1 arrival maneuver, whereas, the Lunar Lander used its Kick Stage.

The main difference between the L1 TRM approach and the Earth Orbit Rendezvous approach is that the latter assembles all elements into one large stack in LEO and deploys the entire stack to L1 in one series of maneuvers. Therefore, the 954 m/s L1 arrival maneuver is performed by the Kick Stage for the entire stack. Thus, this means that the Service Module is no longer required to perform this maneuver.

This architectural change results in a significantly smaller Service Module. Table 15.5.1-1 outlines the system mass changes due to this new approach.

As can be seen in Table 15.5.1-1, there are significant mass savings associated with removing the L1 arrival maneuver from the list of burns that the Service Module needs to perform. The propellant and propulsion system hardware that is saved by removing this maneuver from the set of burns that the Service Module must perform, totals nearly 7,000 kg. The resulting structural and dry mass growth mass changes account for the remainder of the mass delta between the two versions of the CEV's Service Module.

CEV Service Module's System Mass Changes				
System	TRM (kg)	EOR (kg)	Mass Change (kg)	% Change
Structure	1455	1397	(58)	(4.0)
Protection	0	0	No Change	0.0
Propulsion	1408	968	(440)	(31.3)
Power	661	661	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	647	(100)	(13.4)
Non-Cargo	305	131	(174)	(57.0)
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1442	No Change	0.0
Propellant	11332	4742	(6590)	(58.1)
Total	17560	10199	(7361)	(41.9)

Table 15.5.1-1: CEV Service Module's System Mass Changes, Resulting From Employing An Earth Orbit Rendezvous Approach

15.5.2 Lunar Lander

As described in section 15.2, the Lunar Lander is deployed to LEO on the third launch. This is in contrast to the TRM where it was deployed on the second launch. This architectural change results in a decreased Lunar Lander lifetime of approximately two weeks. As can be seen in Tables 15.5.2-1 and 15.5.2-2, this architectural variation has essentially no impact on the Lander's mass properties. The small mass deltas between the TRM stages of the Lander and the EOR stages is attributable to the boil-off of propellants.

Lander Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**EOR (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	839	No Change	0.0
Protection	73	73	No Change	0.0
Propulsion	1631	1622	(9)	(0.6)
Power	813	813	No Change	0.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	851	No Change	0.0
Other	455	455	No Change	0.0
Growth	1080	1078	(2)	(0.2)
Non-Cargo	1483	1482	(1)	0.0
Cargo	227	227	No Change	0.0
Non-Propellant	1014	1014	No Change	0.0
Propellant	10703	10690	(13)	(0.1)
Total	19906	19882	(24)	(0.1)

Table 15.5.2-1: Lander Ascent Stage's System Mass Changes, Resulting From Employing An Earth Orbit Rendezvous Approach

Lander Descent Stage's System Mass Changes				
System	**TRM (kg)**	**EOR (kg)**	**Mass Change (kg)**	**% Change**
Structure	553	553	No Change	0.0
Protection	50	50	No Change	0.0
Propulsion	1413	1404	(9)	(0.6)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	707	(1)	(0.1)
Growth	678	676	(2)	(0.3)
Non-Cargo	464	462	(2)	(0.4)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	17546	(27)	(0.2)
Total	22608	22566	(42)	(0.2)

Table 15.5.2-2: Lander Descent Stage's System Mass Changes, Resulting From Employing An Earth Orbit Rendezvous Approach

15.5.3 Kick Stage

As described in section 15.2, the Kick Stage is used to perform the libration point arrival burn for the entire stack of elements: Lunar Lander, CEV Crew Module, and CEV Service Module. Therefore, the mass of this element grew significantly in this architecture when compared to the Kick Stage in the TRM. As can be seen in Table 15.5.3-1, the propellant and propulsion system hardware increased by nearly 4,700 kg in order to accommodate the extra mass of the elements included in this stack during the L1 arrival maneuver. The resulting structural and dry mass growth mass changes account for the remainder of the mass delta between the two versions of the Kick Stage.

Kick Stage's System Mass Changes				
System	**TRM (kg)**	**EOR (kg)**	**Mass Change (kg)**	**% Change**
Structure	621	702	81	13.0
Protection	0	0	No Change	0.0
Propulsion	1530	1646	116	7.6
Power	100	100	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	0	0	No Change	0.0
Other	405	405	No Change	0.0
Growth	531	571	40	7.5
Non-Cargo	953	913	(40)	(4.2)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	23323	27922	4599	19.7
Total	27465	32259	4794	17.5

Table 15.5.3-1: Kick Stage's System Mass Changes, Resulting From Employing An Earth Orbit Rendezvous Approach

15.5.4 Earth Departure Stage

The EDS's in this architecture were re-sized such that they were of approximately equal size. As indicated in section 15.2, the EOD burn is shared between the two EDS's. In order to achieve two EDS's of approximately the same size, the EOD maneuver (3,104 m/s total) had to be shared: EDS1 provides 1,229 m/s, EDS2 provides 1,875 m/s. This resulted in two EDS's, each sized at approximately 62,500 kg. The strategy behind this sizing was twofold. First, it resulted in two equally sized EDS's, which will have similar manufacturing processes. Second, it decreased the mass of the largest launch required to perform this architecture. The mass of the largest launch in the TRM was approximately 94 metric tons, the mass of EDS1. The mass of the largest launch in this Earth Orbit Rendezvous approach was calculated to be 75 metric tons, the combined mass of the Lunar Lander and Kick Stage.

EDS1's and EDS2's System Mass Changes					
Element	**System**	**TRM (kg)**	**6 Crew (kg)**	**Mass Change (kg)**	**% Change**
EDS1	Structure	1972	1437	(535)	(27.1)
	Protection	0	0	No Change	0.0
	Propulsion	4361	3485	(876)	(21.1)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1170	(282)	(19.4)
	Non-Cargo	3109	2472	(637)	(20.5)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	53045	(29244)	(35.5)
	Total	94109	62534	(31575)	(33.6)
EDS2	Structure	932	1418	486	52.1
	Protection	0	0	No Change	0.0
	Propulsion	2318	3020	702	30.3
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	1072	238	28.5
	Non-Cargo	1355	2222	867	64.0
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	53806	20909	63.6
	Total	39256	62458	23202	59.1

Table 15.5.4-1: EDS1's and EDS2's System Mass Changes, Resulting From Employing An Earth Orbit Rendezvous Approach

15.5.5 Vehicle Mass Properties for the Earth Orbit Rendezvous Architecture

Earth Orbit Rendezvous vehicle mass properties as generated by the Envision parametric sizing tool are listed in Table 15.5.5-1. Subsystem components are categorized according the mass properties reporting standard outlined in JSC-23303 Design Mass Properties: Guidelines and Formats for Aerospace Vehicles. All estimates include 20% margin applied to categories one through eight of the vehicle mass properties for dry mass growth.

The largest single element to be launched is the Earth Departure Stage with an initial mass in low Earth orbit (IMLEO) of nearly 62.5 tons. However, since the Lunar Lander (both ascent and descent stages) and the Kick Stage are launched as an integrated stack, they combine to make the largest launch with an IMLEO of nearly 75 tons. The launch of this stack will drive the payload delivery capabilities of the cargo launch vehicle. Finally, the CEV is launched with the crew on a human-rated launch vehicle capable of delivering 19 tons to LEO. The combined architecture elements of the Earth Orbit Rendezvous architecture have a total IMLEO of 219 tons, compared to 230 tons for the TRM architecture.

	CEV CM	CEV SM	Ascent Stage	Descent Stage	Kick Stage	EDS1	EDS2
1.0 Structure	1,523	1,397	702	839	553	1,437	1,418
2.0 Protection	822	0	0	73	50	0	0
3.0 Propulsion	117	968	1,646	1,622	1,404	3,485	3,020
4.0 Power	482	661	100	813	137	190	190
5.0 Control	0	0	0	0	0	0	0
6.0 Avionics	737	0	0	738	0	175	171
7.0 Environment	691	110	0	851	530	105	104
8.0 Other	835	100	405	455	707	455	455
9.0 Growth	1,041	647	571	1,078	676	1,170	1,072
DRY MASS	6,249 kg	3,883 kg	3,424 kg	6,469 kg	4,058 kg	7,017 kg	6,430 kg
10.0 Cargo	1,478	0	0	227	500	0	0
11.0 Non-Cargo	966	131	913	1,482	462	2,472	2,222
INERT MASS	8,693 kg	4,015 kg	4,337 kg	8,178 kg	5,020 kg	9,489 kg	8,652 kg
12.0 Non-Propellant	55	1,442	0	1,014	0	0	0
13.0 Propellant	64	4,742	27,922	10,690	17,546	53,045	53,806
GROSS MASS	8,812 kg	10,199 kg	32,259 kg	19,882 kg	22,566 kg	62,534 kg	62,458 kg

Table 15.5.5-1: Element Mass Properties for the Earth Orbit Rendezvous Architecture

15.6 System Technologies and Programmatic Risks

There were no new technologies introduced into this architecture that were not already employed in the TRM. Therefore, section 10.6.1 should be referred to for an assessment of current Technology Readiness Levels (TRLs) for the required technologies.

Similarly, there were no new programmatic risks introduced into this architecture that were not already present in the TRM. Therefore, section 10.6.2 should be referred to for a description of the programmatic risks associated with this architecture.

15.7 Pros/Cons Summary

This section contains a discussion of the pros/cons of this architecture relative to the TRM.

15.7.1 Pros of Earth Orbit Rendezvous Architecture

There are three major benefits that arise from the tandem EOD approach adopted by this architecture. The tandem EOD helps to provide flexibility in the allocation of the delta V maneuvers. This flexibility allows the EDS's to shrink in size, since they are able to more evenly share the burden of deploying the Lander and CEV to L1. The resulting EDS's have a mass of approximately 62.5 metric tons each. This is in contrast to the TRM approach in which one EDS had a mass of 39 metric tons and the second had a mass of 94 metric tons.

The first major benefit of this "burden-sharing" is that the size of the largest launch in this architecture shrinks from 94 metric tons (the size of one of the largest EDS in the TRM) to 75 metric tons, which is the mass of the Lunar Lander/Kick Stage stack.

The second major benefit regards the manufacturing of the EDS's. In the TRM, since the two EDS's had a significant difference between their required sizes, it is likely that their physical dimensions would be different upon manufacturing (although, there is always the option to create a common EDS that can hold the maximum quantity of propellants that will be needed by an EDS and then only fill the tanks to the amount that is needed to perform the required maneuvers for the other EDS). The Earth Orbit Rendezvous approach, on the other hand, was deliberately designed to achieve the EOD maneuver with two evenly sized EDS's. This commonality may allow the EDS manufacturing and testing costs to remain lower than would be possible if the TRM were adopted.

The third major benefit regards the total architecture mass. This approach allows the total architecture mass to drop by 11 metric tons (4.8%). Therefore, there is a possibility that this architecture can be operated at a lower cost relative to the TRM.

Additionally, two benefits arise from logistics of maintaining the elements in/staging the mission from LEO as opposed to pre-deploying some of the assets to L1. First, the Lander can be checked out and any problems troubleshot from LEO rather than L1. Therefore, upon launch of the CEV, it may be possible for the crew to address existing problems with the Lunar Lander prior to committing the crew to the journey to L1. If the problem can not be resolved, the crew can return home from LEO without being required to transit back from L1.

Finally, a second logistical benefit arises from the fact that upon LEO departure, the Lunar Lander's and CEV's habitable volumes are physically connected. Depending on the design of the Lander and CEV, this may allow the crew to utilize the Lunar Lander as extra cabin space, if they are permitted to use the Lander's volume. As during Apollo 13, there may also be benefits of having a second, completely independent vehicle that can be used by the crew in the case of an emergency.

15.7.2 Cons of Earth Orbit Rendezvous Architecture

Two major cons are apparent in this architecture. First, at least one of the two EDS's must have a much more complex suite of avionics and docking hardware as compared to the EDS's in the TRM. This arises from the requirement for the second EDS to actively chase the first EDS and dock to it prior to the launch of the Lunar Lander. In contrast, the two EDS's in the TRM were deployed to LEO and then the Lunar Lander chased one EDS and the CEV chased the other, allowing both EDS's to remain passive.

A second possible major con stems from the configuration of the stack prior to EOD in the Earth Orbit Rendezvous architecture. The stack in this architecture contains five major elements: 2 EDS's, the Lunar Lander, the Kick Stage, and the CEV. This is a large, very complex stack. Not only will the mechanics of this stack most likely be more complex than any of the stacks in the TRM, but it will also require complex maneuvers, such as splitting the EOD burn between two different elements.

16.0 TRM with Aerocapture, Phasing, and Land-Landing

This architectural variation examined the major differences between ending the mission with direct entry followed by landing in water vs. performing an aerocapture, phasing, and landing on land.

16.1 Major Assumptions/Clarifications

This section outlines the major architectural assumptions that differ from the trade reference mission defined in section 10.1. Unless otherwise stated, all other major assumptions remained the same as outlined in section 10.1.

Changed Assumption:

New: The nominal Earth return for the CEV is aerocapture, phasing, and land landing.

Old: The nominal Earth return for the CEV is a direct entry with a water landing.

16.2 Architecture Description

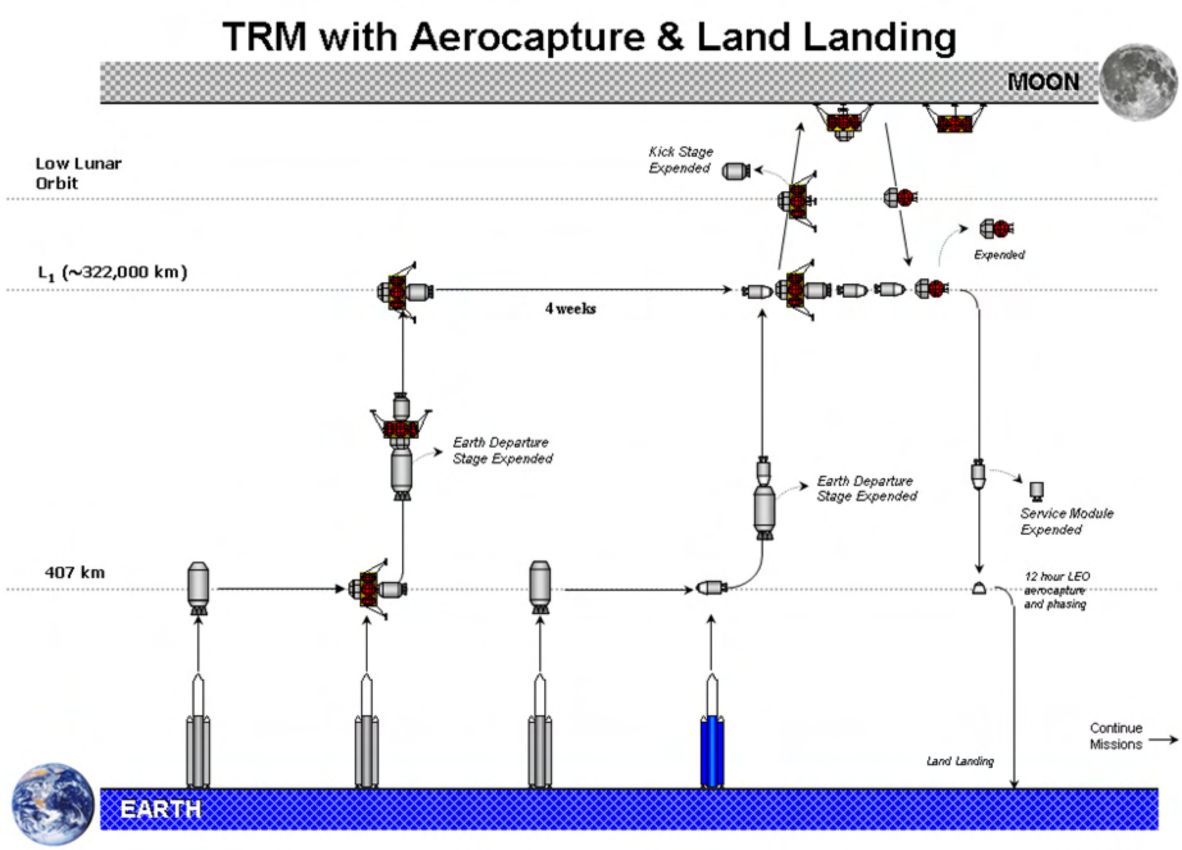


Figure 16.2-1: Aerocapture, Phasing, and Land-Landing Architecture Illustration

This architecture follows the exact same operational plan as the TRM until the point when the crew departs from L1 in the CEV to return to Earth. Refer to section 10.2 for a description of the architecture prior to this point of departure.

At the time of Earth-bound L1 departure, the CEV Service Module will need to adjust its burn such that it can capture into the desired inclination and minimize the amount of time that it is required to loiter in orbit. After the L1 departure burn, the CEV will coast on a 94-hour return trajectory. Three hours prior to atmospheric interface, the Service Module will separate from the Crew Module. The Service Module will be targeted to enter the atmosphere for disposal. The CEV Crew Module, on the other hand, will continue on its trajectory and perform its aerocapture maneuvers upon arrival at Earth.

Figure 16.2-2 depicts possible Earth return profiles. Any of these profiles can be achieved by slightly modifying the L1 departure burn. An important feature to note in Figure 16.2-2 is that if a direct entry return were to be used, the CEV would be forced to land near the antipode of the Moon. Note: Figure 16.2-2 depicts departure from lunar orbit; however, the same principles apply for departure from L1.

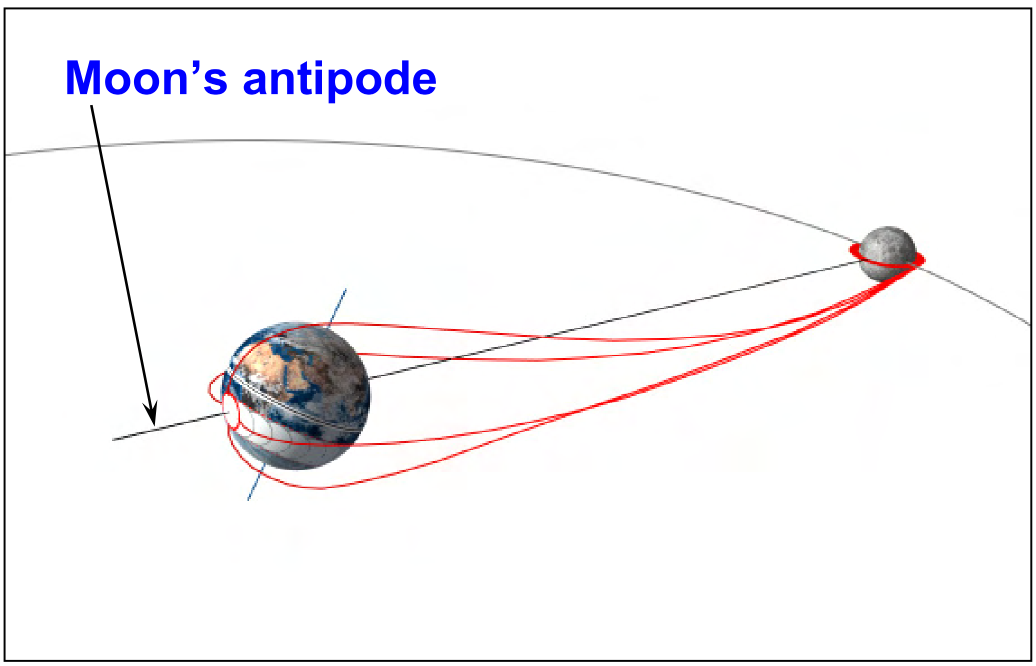


Figure 16.2-2: CEV Earth Return Profiles

There are two important features that should be noted in Figure 16.2-2. First, if a direct entry return were to be used, the CEV would be forced to land near the antipode of the Moon. Second, the inclination of the return trajectory can be chosen based upon tailoring the Earth-return burn. Using this second characteristic, an architecture that incorporates aerocapture can capture into any inclination.

In the case of an architecture that incorporates aerocapture, the CEV would have access to landing sites within the band of latitudes whose absolute values are less than or equal to the magnitude of the arrival inclination. For example, if the CEV were to capture into an inclination of 28.5°, it would have access to landing sites that have latitudes between +28.5° to -28.5°. This is depicted in Figure 16.2-3. The shaded areas are the accessible latitudes.

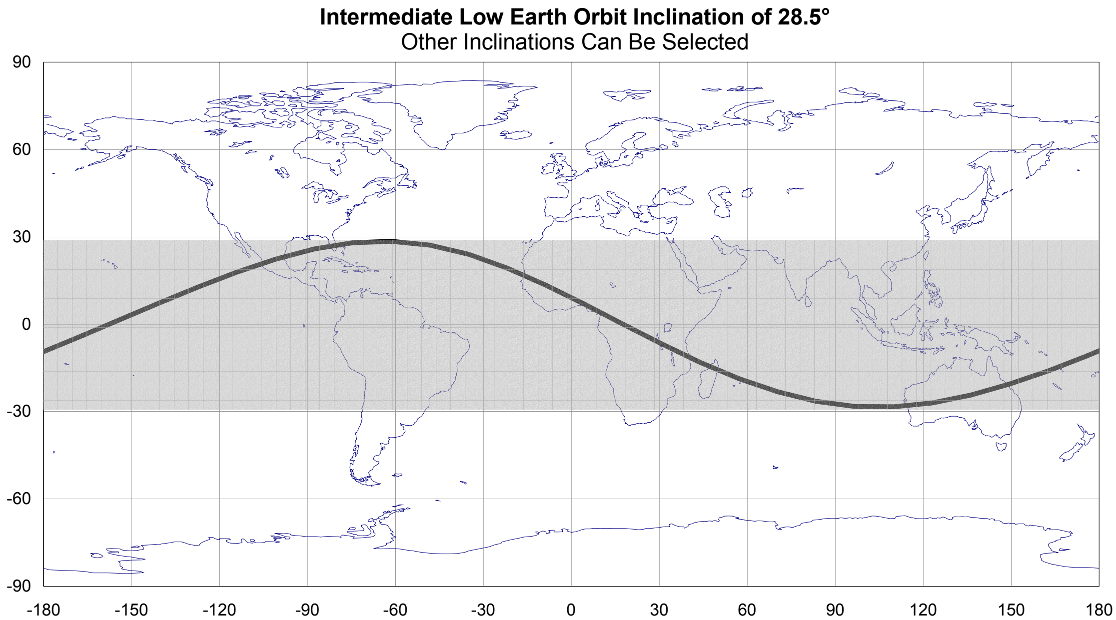


Figure 16.2-3: Accessible Latitudes From a Fixed 28.5° Inclination

Once the CEV is captured into the desired inclination, it will need to phase with the appropriate landing site. Normally, this will be performed mainly through loitering until the appropriate time and then performing de-orbit burns. Loitering in orbit, combined with the timing of the L1 departure burn, will allow the CEV to adjust which landing sites at various latitudes and longitudes can be reached (the CEV was only designed to accommodate 12 hours of on-orbit loitering; a greater variety of landing sites at various latitudes/longitudes could be accessed if the CEV were to loiter for a longer period of time).

Upon arrival at Earth, the CEV will briefly skim through the atmosphere at an altitude of 75 km in order to lower the apogee of its orbit to 315 km. Next, the CEV will use its RCS thrusters to raise its perigee to 185 km, which is the altitude of a standard 24-hour, unimpeded orbital altitude. The CEV will remain in this 185 km x 315 km orbit until it has aligned itself with the proper landing site. Once aligned, the CEV will use its RCS thrusters in order to perform the final Earth entry maneuvers where its perigee will be lowered to 15 km. At this point, the CEV will be slowed sufficiently to perform landing. Figure 16.2-4 outlines the nominal timelines for this mission.

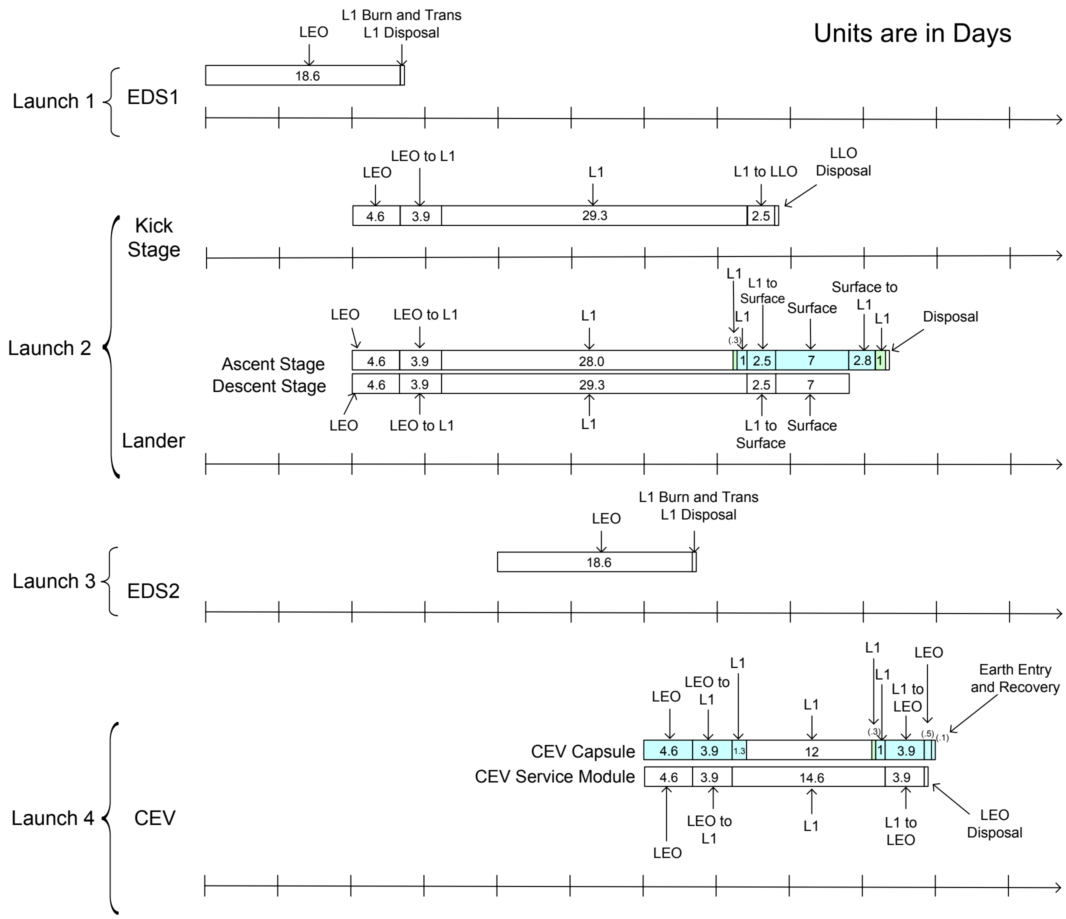


Figure 16.2-4: Nominal Timeline for the Aerocapture, Phasing, and Land-Landing Architecture

16.3 Safety & Mission Success

16.3.1 Critical Event Identification

The TRM has a total of fifty-six identified critical events. Of those fifty-six critical events, twenty occur during the uncrewed portions of the mission. The remaining thirty-six critical events occur during the crewed portions of the mission. The Aerocapture, Phasing, and Land-Landing approach has fifty-seven identified critical events. Of those fifty-seven critical events, twenty occur during the uncrewed portions of the mission. The remaining thirty-seven critical events occur during the crewed portions of the mission. The additional critical event during the crewed portion of the mission for the Aerocapture, Phasing, and Land-Landing approach is the Aerocapture maneuver.

16.3.2 TRM with Aerocapture, Phasing, and Land-Landing Critical Event Ranking

The critical events for this architectural approach were ranked, in order to understand their severity. Following the ranking scheme of the TRM, the following numbers were assigned to each critical event to signify its level of inherent risk.

Rank of 1: *Failures during mission critical events that could lead to a Loss of Mission (LOM) but not a Loss of Crew (LOC).*

Rank of 2: *Failures during mission critical events that could lead to a LOC but would have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

Rank of 3: *Failures during mission critical events that would not have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

Of the fifty-seven total critical events identified for the Aerocapture, Phasing, and Land-Landing approach, eight received a rank of 3, twenty-five received a rank of 2, and the remaining twenty-four received a rank of 1. The complete set of identified and ranked critical events for the Aerocapture, Phasing, and Land-Landing approach is listed in Table 16.3.2-1.

	ID #	TRM with Aerocapture, Phasing, and Land TD Critical Events	TRM with Aerocapture, Phasing and Land TD Critical Event Rank
Uncrewed Critical Events	VAR-06-01	EDS-1 (for the LL) Launch	1
	VAR-06-02	EDS-1 Ascent	1
	VAR-06-03	EDS-1 Launch Shroud Separation	1
	VAR-06-04	EDS-1 Separation from Booster	1
	VAR-06-05	LL & Kick Stage Launch	1
	VAR-06-06	LL & Kick Stage Ascent	1
	VAR-06-07	LL & Kick Stage Launch Shroud Separation	1
	VAR-06-08	LL & Kick Stage Separation from Booster	1
	VAR-06-09	LL & Kick Stage Orbital Maneuvering	1
	VAR-06-10	LL & Kick Stage Docks to EDS-1	1
	VAR-06-11	EDS-1, Kick Stage, & LL Burn for L1	1
	VAR-06-12	LL & Kick Stage Separates from EDS-1	1
	VAR-06-13	Kick Stage, & LL Mid-course Correction Burn	1
	VAR-06-14	Kick Stage, & LL Burn to Slow Near L1	1
	VAR-06-15	EDS-2 (for CEV) Launch	1
	VAR-06-16	EDS-2 Ascent	1
	VAR-06-17	EDS-2 Launch Shroud Separation	1
	VAR-06-18	EDS-2 Separation from Booster	1

	ID #	TRM with Aerocapture, Phasing, and Land TD Critical Events	TRM with Aerocapture, Phasing and Land TD Critical Event Rank
	VAR-06-19	EDS-2 Orbital Maneuvering	1
Crewed Critical Events	VAR-06-20	CEV (CM+SM) Launch	2
	VAR-06-21	CEV Ascent	2
	VAR-06-22	LES Separation	2
	VAR-06-23	CEV Launch Shroud Separation	2
	VAR-06-24	CEV Separation from Booster	2
	VAR-06-25	CEV Orbital Maneuvering	2
	VAR-06-26	CEV Docks to EDS-2	2
	VAR-06-27	EDS-2 & CEV Burn for L1	2
	VAR-06-28	EDS-2 & CEV Mid-course Correction Burn	1
	VAR-06-29	EDS-2 & CEV Burn to Slow Near L1	2
	VAR-06-30	CEV Separates from EDS-2	2
	VAR-06-31	CEV Orbital Maneuvering	2
	VAR-06-32	CEV Docks to LL & Kick Stage	2
	VAR-06-33	Crew Transfers from CEV to LL & Kick Stage	1
	VAR-06-34	LL & Kick Stage Separates from CEV	2
	VAR-06-35	LL & Kick Stage Burns for Low Lunar Orbit	2
	VAR-06-36	LL & Kick Stage Mid-course Correction Burn	1
	VAR-06-37	LL & Kick Stage Lunar Orbit Insertion (LOI)	2
	VAR-06-38	Kick Stage Separates from LL	2
	VAR-06-39	LL De-orbit Burn to Moon	2
	VAR-06-40	LL Powered Descent & Landing on Moon	3
	VAR-06-41	LL Ascent Stage Separation & Ascent	3
	VAR-06-42	LL Ascent Stage Orbital Maneuvering	3
	VAR-06-43	LL Ascent Stage Lunar Orbit Departure	3
	VAR-06-44	LL Ascent Stage Mid-course Correction Burn	1
	VAR-06-45	LL Ascent Stage L1 Arrival	3
	VAR-06-46	LL Ascent Stage Orbital Maneuvering	2
	VAR-06-47	LL Ascent Stage Docks with CEV	2
	VAR-06-48	Crew Transfers from LL to CEV	2
	VAR-06-49	CEV Separates from LL Ascent Stage	2
	VAR-06-50	CEV Burn for Earth	3
	VAR-06-51	CEV Mid-course Correction Burn	1
	VAR-06-52	CM Separates & Maneuvers away from SM	2
	VAR-06-53	CM Performs Aerocapture	3
	VAR-06-54	CM Orbital Maneuvering	2
	VAR-06-55	CM Entry	3
	VAR-06-56	CM Land-Landing	2
	VAR-06-57	Crew Recovery	2

Table 16.3.2-1: TRM with Aerocapture, Phasing, and Land-Landing Critical Events and Rank

Post-aerocapture (critical event VAR-06-53), the CEV Crew Module will loiter in LEO for 12 hours. This extra on-orbit duration will increase the probability of the Crew Module sustaining a Micrometeoroid and/or Orbital Debris (MMOD) penetration. A MMOD strike that penetrates the pressurized crew volume could damage the spacecraft and, in the worst-case scenario, ultimately lead to the loss of one or more of the crewmembers.

Crew safety will be a large concern if the Crew Module is unable to perform the aerobrake maneuver. For example, if for some reason the Crew Module is unable to perform the aerobrake maneuver, it may skip off of the Earth's atmosphere into an orbit from which it can not recover. It would also be possible for the Crew Module to re-enter the Earth's atmosphere steeper and with more energy than expected. The increase in energy could expose the crew to excessive G-loads and/or the Crew Module could exceed its upper thermal heating limits. This could result in the loss of one or more of the crewmembers. The inability to perform the correct aerobraking maneuver will also affect the ability to land in the correct landing zone. Therefore, there is a possibility that the CEV could land in the water or in a location on land where recovery equipment is not nearby. Thus, to increase the probability of crew survival, there would need to be provisions built into the Crew Module to protect for these situations.

16.4 Mission Abort Options

16.4.1 Mission Abort Comparison to the TRM

The mission abort options for this scenario are outlined in section 10.4. Note that section 10.4.3 lists critical events that could occur on this mission and their mitigating solutions (if there is one).

As explained in section 15.2, the Aerocapture, Phasing, and Land-Landing architecture starts to diverge from the TRM at the point of L_1 departure. However, from a mission-risk point of view, the major departure from the TRM begins at the time of aerocapture upon arrival at Earth. Section 10.4.3 covers this subject, but the information has been repeated below for convenience.

Earth Aerocapture to LEO

This phase begins with CEV re-entry into Earth atmosphere, encompasses CEV aerobraking into the desired LEO operations and ends just prior to the CEV final de-orbit burn.

a. Failure to aerocapture and circular burn (elliptical orbit)

i. Delta-V maneuver to appropriate orbit with physical or functional redundancy

ii. Safe haven until Earth based rescue or natural orbital decay

iii. Passive control/ballistic re-entry

For missions designed to use aerobraking to LEO instead of a direct entry a failure to successfully complete the aerocapture leads to the following aborts. If the aerocapture fails to produce the desired LEO, CEV propulsion can be used to provide the desired orbit. In addition, the CEV may be designed to allow for a passively controlled ballistic re-entry using the aerobrake heat shield in addition to the CEV. Once in LEO the CEV could provide a safe haven for TBD weeks until an Earth based rescue could be performed.

b. Failure to aerocapture (escape trajectory)

i. LOC

If the failure to aerocapture results in an atmospheric skip out and Earth escape trajectory there is a LOC. Physical or functional redundancy must be provided to ensure that the CEV is safely captured into LEO.

De-orbit and Re-entry to Touchdown

This phase begins with the CEV de-orbit burn and ends with CEV touchdown on Earth's surface.

a. No de-orbit

i. Safe haven until rescue or orbital decay or LOC

After reaching a safe LEO if the CEV fails to perform the de-orbit maneuver there is a LOC unless the CEV can provide a safe haven until an Earth based rescue can be performed.

b. Re-entry flight control failures

i. Passive re-entry (no lift vector control)

After a successful de-orbit burn the CEV will have the capability to perform a ballistic re-entry in the event a nominal re-entry is not possible.

c. Entry targeting failures

i. Water or land touchdown

CEV equipped with appropriate crew survival and search and rescue gear for touchdown site

The CEV will be designed to support either land or water touchdown allowing for entry targeting failures to force the CEV to miss the desired touchdown site. The LDRM architecture is using 3 hr as the time required to find and recover the crew from the CEV after touchdown.

16.4.2 Return to Earth and Return Time to CEV

The timelines for returning the crew to Earth and to the CEV are identical to those specified in section 10.4.4 and illustrated in Figures 10.4-1 and 10.4-2 except a possible deviation at the very end of the mission, if a specific landing site is to be targeted.

16.5 Element Overview & Mass Properties

This section describes any changes that were made to the TRM elements in order to accommodate this architectural variation. The changes due to these modifications, if any, were evaluated with the use of the Envision sizing tool. Unless otherwise stated in this section, there were no changes to the TRM elements described in section 10.5.

16.5.1 Crew Exploration Vehicle

As described in section 10.5.1, the CEV is comprised of a Crew Module and a Service Module. The Crew Module required modification to its Entry, Descent, and Landing (EDL) sub-systems and its Reaction Control System (RCS) to accommodate this architectural variation. To accommodate these changes, which resulted in an increase in the mass of the Crew Module, the Service Module's propulsion system and propellant masses were scaled upwards.

The Crew Module had to be modified in order to accommodate the aerocapture maneuver and the nominal land-landing approaches. The aerocapture maneuver increased the heat load on the CEV's TPS. Although the peak heating rates are decreased, the overall duration of the atmospheric heating will be increased, which led to the requirement for scaling up the amount of TPS attached to the Crew Module. The original scaling factor for the TPS, 8.5% of the total CEV mass, was scaled up to 9.5% to accommodate this extra heat load. These percentages are partially based on historical values and partially on recent estimates for capsule-shaped bodies.

In order to protect the Crew Module for a nominal land-landing, the landing systems were modified such that they could protect the CEV from damage. The TRM also protected for the situation during an abort where the Crew Module would land on land. However, this was an off-nominal situation in the TRM, during which the recovery of the CEV was not a top priority. Therefore, although the TRM Crew Module was designed to return the crew safely to Earth in the event of a land-landing, it was not designed to withstand damage. Since the aerocapture architecture uses land-landings as the nominal situation, and since it is desirable to recover the spacecraft fully intact after a nominal mission, the landing systems were improved.

One sub-system that was modified in order to achieve this was the parachute system. The TRM CEV Crew Module parachute system was sized such that it would provide a sink rate of 28 ft/s. This value was based on historical spacecraft designs that were designed for similar landings. In order to accommodate a nominal land-landing, the sink rate was decreased to 22 ft/s. This value was based on studies that were performed under the NASA Space Launch Initiative program for capsule-shaped bodies that performed land-landings. This decreased sink rate caused an increase in mass of ~175 kg for the parachute system.

A second modification that was made in order to protect the Crew Module, was that airbags were chosen as a replacement for the crushable rib structure, which was used by the TRM Crew Module. This resulted in a mass increase of ~560 kg.

Finally, the last modification made to the Crew Module was to its RCS. The TRM Crew Module was designed with a very small RCS that was designed to supply 10 m/s delta V, for attitude control. In contrast, as stated in section 16.2, the aerocapture architecture requires the CEV to propulsively maneuver to a 185 km x 315 km orbit after skimming through the Earth's atmosphere at 75 km. Therefore, the Crew Module in this scenario required 112 m/s delta V. The TRM Crew Module RCS drew approximately 65 kg of fuel and oxidizer (combined) from the LO2/LCH4 main propellant tanks. Due to the higher performance requirements of the aerocapture architecture's Crew Module RCS, a separate propellant system was used that burned LO2/Ethanol. This caused an increase in the mass of both the propulsion system hardware and propellants. The net increase due to this change was 500 kg.

As previously stated, the CEV's Service Module was modified only in order to accommodate the increased mass of the Crew Module. The mass of the propulsion system and propellants increased approximately 2,000 kg in order to achieve the required performance. The resulting structural and dry mass growth changes account for the remainder of the mass delta between the two versions of the Service Module.

CEV Crew Module's System Mass Changes				
System	**TRM (kg)**	**A/P/LL (kg)**	**Mass Change (kg)**	**% Change**
Structure	1523	1527	4	0.3
Protection	822	1100	278	33.8
Propulsion	117	296	179	153.0
Power	482	880	398	82.6
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	726	35	5.1
Other	835	1608	773	92.6
Growth	1041	1375	334	32.1
Non-Cargo	966	981	15	1.6
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	59	4	7.3
Propellant	64	435	371	580.0
Total	8812	11202	2390	27.1

Table 16.5.1-1: CEV Crew Module's System Mass Changes, Resulting From Employing An Aerocapture, Phasing, and Land-Landing Approach

CEV Service Module's System Mass Changes				
System	**TRM (kg)**	**A/P/LL (kg)**	**Mass Change (kg)**	**% Change**
Structure	1455	1472	17	1.2
Protection	0	0	No Change	0.0
Propulsion	1408	1538	130	9.2
Power	661	661	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	776	29	3.9
Non-Cargo	305	357	52	17.0
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1442	No Change	0.0
Propellant	11332	13299	1967	17.4
Total	17560	19756	2196	12.5

Table 16.5.1-2: CEV Service Module's System Mass Changes, Resulting From Employing An Aerocapture, Phasing, and Land-Landing Approach

16.5.2 Lunar Lander

There were no changes made to the Lunar Lander to accommodate this architectural variation.

16.5.3 Kick Stage

There were no changes made to the Kick Stage to accommodate this architectural variation.

16.5.4 Earth Departure Stage

There were no changes made to EDS1 to accommodate this architectural variation. However, similar to the CEV Service Module, EDS2's propulsion hardware and propellants were scaled upwards in order to accommodate the increased mass of the CEV. The resulting structural and dry mass growth changes account for the remainder of the mass delta between the two versions of EDS2.

EDS2's System Mass Changes				
System	**TRM (kg)**	**A/P/LL (kg)**	**Mass Change (kg)**	**% Change**
Structure	932	1053	121	13.0
Protection	0	0	No Change	0.0
Propulsion	2318	2470	152	6.6
Power	190	190	No Change	0.0
Control	0	0	No Change	0.0
Avionics	171	171	No Change	0.0
Environment	104	104	No Change	0.0
Other	455	455	No Change	0.0
Growth	834	889	55	6.6
Non-Cargo	1355	1537	182	13.4
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	32897	38025	5128	15.6
Total	39256	44895	5639	14.4

Table 16.5.4-1: EDS2's System Mass Changes, Resulting From Employing An Aerocapture, Phasing, and Land-Landing Approach

16.5.5 Vehicle Mass Properties for the Aerocapture, Phasing, and Land-Landing Architecture

The Aerocapture, Phasing and Land-Landing architecture's vehicle mass properties, as generated by the Envision parametric sizing tool, are listed in Table 16.5.5-1. Subsystem components are categorized according the mass properties reporting standard outlined in JSC-23303 Design Mass Properties: Guidelines and Formats for Aerospace Vehicles. All estimates include 20% margin applied to categories one through eight of the vehicle mass properties for dry mass growth.

The largest single element to be launched is the Earth Departure Stage with an initial mass in low Earth orbit (IMLEO) of nearly 94.1 metric tons. The launch of this element will drive the payload delivery capabilities of the cargo launch vehicle. The Lunar Lander (both ascent and descent stages) and the Kick Stage are launched as an integrated stack, which combine to make an IMLEO of nearly 70 tons. Finally, the CEV is launched with the crew on a human-rated launch vehicle capable of delivering 31 tons to LEO. The combined architecture elements of the aerocapture architecture have a total IMLEO of 240 tons, compared to 230 tons for the TRM architecture.

	CEV CM	CEV SM	Kick Stage	Ascent Stage	Descent Stage	EDS1	EDS2
1.0 Structure	1,527	1,472	621	839	553	1,972	1,053
2.0 Protection	1,100	0	0	73	50	0	0
3.0 Propulsion	296	1,538	1,530	1,631	1,413	4,361	2,470
4.0 Power	880	661	100	813	137	190	190
5.0 Control	0	0	0	0	0	0	0
6.0 Avionics	737	0	0	738	0	175	171
7.0 Environment	726	110	0	851	530	105	104
8.0 Other	1,608	100	405	455	708	455	455
9.0 Growth	1,375	776	531	1,080	678	1,452	889
DRY MASS	**8,249**	**4,657**	**3,188**	**6,479**	**4,071**	**8,710**	**5,332**
10.0 Cargo	1,478	0	0	227	500	0	0
11.0 Non-Cargo	981	357	953	1,483	464	3,109	1,537
INERT MASS	**10,708**	**5,014**	**4,141**	**8,190**	**5,035**	**11,819**	**6,869**
12.0 Non-Propellant	59	1,442	0	1,014	0	0	0
13.0 Propellant	435	13,299	23,323	10,703	17,573	82,289	38,025
GROSS MASS	**11,202**	**19,756**	**27,465**	**19,906**	**22,608**	**94,109**	**44,895**

Table 16.5.5-1: Element Mass Properties for the Aerocapture, Phasing, and Land-Landing Architecture

16.6 System Technologies and Programmatic Risks

16.6.1 Technology Assessment

This architecture required three different technologies compared to the TRM. First, the RCS was switched from a Tridyne system to an O2/Ethanol system. Second, the crushable rib landing system of the TRM Crew Module was replaced with an airbag system in order to accommodate nominal land-landings. Third, aerocapture could be considered a technology (although it is actually more of a technique) that was not included in the TRM and would need further development.

The Tridyne system that was replaced in this architecture was rated at TRL 5 in section 10.6.1 for the TRM architecture. The replacement O2/Ethanol system is currently rated at TRL 5 for this application.

The crushable landing system that was replaced in this architecture did not require any new technologies, so it would be considered to be at a very high TRL and would only require tailoring for this particular application. The replacement airbag system is currently rated at TRL 5 for this application.

Aerocapture is an unproven technique for human-rated missions at Earth. Because of the fact that Earth-based aerocapture for human applications has been studied, but never tested, it would probably only be given a TRL of 3, possibly 4.

16.6.2 Programmatic Risks

There were no new programmatic risks introduced into this architecture that were not already present in the TRM. Therefore, section 10.6.2 should be referred to for a description of the programmatic risks associated with this architecture.

16.7 Pros/Cons Summary

This section contains a discussion of the pros/cons of this architecture relative to the TRM.

16.7.1 Pros of the TRM with Aerocapture, Phasing, and Land-Landing Architecture

The major benefit of this architecture is the flexibility it allows the CEV to reach convenient landing sites in the northern hemisphere. It is possible to target landing sites in the Continental United States (CONUS) with this architecture, which will simplify the size and logistics of the Earth-based recovery assets (such as a fleet of ships) that must be on-hand in order to recover the CEV and crew.

A second benefit is the fact that the crew will land on land, which allows them the flexibility to egress the capsule post-landing. This is a great safety benefit and allows an extra level of comfort for the crew.

16.7.2 Cons of the TRM with Aerocapture, Phasing, and Land-Landing Architecture

One of the major cons of the this architecture is the extra mass that must be carried in order to accommodate the performance requirements associated with the aerocapture maneuver and the nominal land-landing. The extra mass in the CEV's Crew Module and Service Module placed the CEV in the 30 metric ton range rather than the 20 metric ton range, which was achieved in the TRM. This is of major strategic importance, when trying to make decisions about launch infrastructure investments. It also pushed the overall architectural mass from 230 metric tons up to 240 metric tons.

A second major con is that this architecture relies upon the successful development and testing of aerocapture technologies and techniques that are currently at a low level of maturity. This will add schedule risk to the program.

17.0 TRM with Direct Return

This architecture variant examines the impact of eliminating the spacecraft rendezvous and crew transfer on the inbound leg of a lunar mission. In the direct return architecture the propellant and systems required for Earth return and recovery are carried to the lunar surface. Because the direct return architecture is particularly well-suited to lunar missions that leverage emplaced assets for lunar surface operations, it is also often referred to as the 'Lunar Surface Rendezvous' architecture.

17.1 Major Assumptions/Clarifications

The direct return architecture variant affects the following TRM assumptions from the LDRM-2 task statement. Any other mission assumptions from Section 10.0 that are not explicitly discussed in this section are unchanged for the direct return architecture.

<u>CEV crew habitation functionality</u>: In the direct return architecture the Crew Exploration Vehicle is the ascent stage of the lander. Therefore, the CEV must provide the crew habitation function for all mission phases from launch through the recovery of the crew.

<u>Dedicated lunar lander element with a separate crew module</u>: In the direct return architecture the Crew Exploration Vehicle and lander form an integrated vehicle. A single crew module is used for all mission phases.

<u>Lunar lander will be pre-deployed to the lunar vicinity</u>: The lunar lander is the descent stage of the combined CEV/lander flight element. The advantages of the direct return architecture are maximized using LEO assembly with a tandem Earth orbit departure strategy.

<u>Libration point L1 is used as the rendezvous point to enable global lunar surface access</u>: By definition the direct return architecture does not require a rendezvous other than for LEO assembly of flight elements in a multi-launch strategy. Global lunar access is inherently provided in the direct return approach.

<u>Assume CH4/LO2 propellants for all other propulsive stages</u>: The direct return architecture results in the highest IMLEO of the three basic lunar architectures when sized using a pressure-fed liquid oxygen/methane propulsion system. The use of higher performance pump-fed engines on the lander descent stage with propellant combinations of liquid oxygen/methane or liquid hydrogen/oxygen propellants are required to significantly reduce the IMLEO of the direct return mission.

Lunar Surface EVA Considerations

Although the subject of an airlock is not explicitly covered in the task statement requirements, it was felt that an airlock would provide significant operational benefits for the L1 TRM and its variants. Unlike the Apollo missions, which sent two astronauts to the lunar surface for up to three days, LDRM-2 requirements specify four crewmembers over a seven-day lunar surface mission, and may also include a mixture of two, three and four person EVA's as well as surface activities with both EVA and IVA components. The presence of an airlock permits EVA's without depressurizing the entire lander crew module, thereby eliminating the need for all crewmembers to don pressure suits. The smaller volume of the airlock relative to the lander crew module also reduces the mass impact of venting breathable atmosphere to vacuum. It is also believed that an airlock can play a major role in reducing the contamination of the lander crew module with lunar dust. Since the mass of the airlock has a significant impact on IMLEO, however, the decision process on whether to include or exclude a lander airlock should balance the mass and cost of the airlock relative to the operational benefits based on number of crew, number of days on the lunar surface, and the EVA objectives.

There are two challenges to the inclusion of an airlock on the CEV Crew Module in a direct return architecture. The first challenge is the packaging and integration of an airlock in a single crew module approach that satisfies the functional requirements for all mission phases from launch through atmospheric re-entry. The second challenge is the mass impact to the lander. An external airlock maximizes the usable volume of the single crew module and can be jettisoned prior to departing the lunar surface to save propellant mass. It is difficult, however, to conceive of an external airlock that integrates cleanly with a blunt body crew module and provides reasonable utility on the lunar surface. An external airlock may also prove to be problematic for launch aborts. An internal airlock will significantly eat into the habitable volume of the crew module and cannot be jettisoned at the conclusion of lunar surface operations. Because the IMLEO of the direct return architecture is extremely sensitive to changes in crew module mass, the addition of an airlock will have a significant impact on the mass of the lander and the Earth Departure Stages. If an airlock is included in a direct return mission then it should be jettisoned at the earliest opportunity.

17.2 Architecture Description

The launch sequence and top-level operations concept for the direct return architecture is shown in Figure 17.2-1. Because the direct return approach employs a tandem Earth orbit departure, all four of the flight elements are launched in two-week intervals and assembled in low Earth orbit – two Earth Departure Stages followed by a lander descent stage and the CEV. The launch and assembly sequence for the direct return architecture is matches that of the TRM architecture variants described in Sections 14.0 and 15.0.

Both of the Earth Departure Stages are burned in series to complete the Earth orbit departure maneuver. The first EDS is expended part way through the EOD maneuver and is targeted for disposal at Earth. After completing the EOD maneuver, the second EDS shuts down during the outbound coast phase that typically lasts around four days. After performing the lunar orbit arrival maneuver for the combined lander/CEV, the second EDS is jettisoned in low lunar orbit. The lander descent stage provides the propulsion to deliver the entire CEV to the lunar surface. No flight assets are left in lunar orbit, thus eliminating a major orbital mechanics rendezvous constraint on the ascent and lunar orbit departure maneuvers for Earth return. At the conclusion of the lunar surface mission the CEV performs an ascent maneuver to a low lunar orbit. Depending on the design of the CEV – single stage, one-and-a-half stage or two stage – CEV propellant tanks or the ascent propulsion stage may be jettisoned in low lunar orbit. Following the lunar orbit departure burn, the CEV coasts for approximately four days in transit to the Earth. Shortly before reaching entry interface, the CEV service module is jettisoned and targeted for disposal in the open ocean. The CEV Crew Module re-enters the Earth's atmosphere separately and is targeted for an ocean recovery.

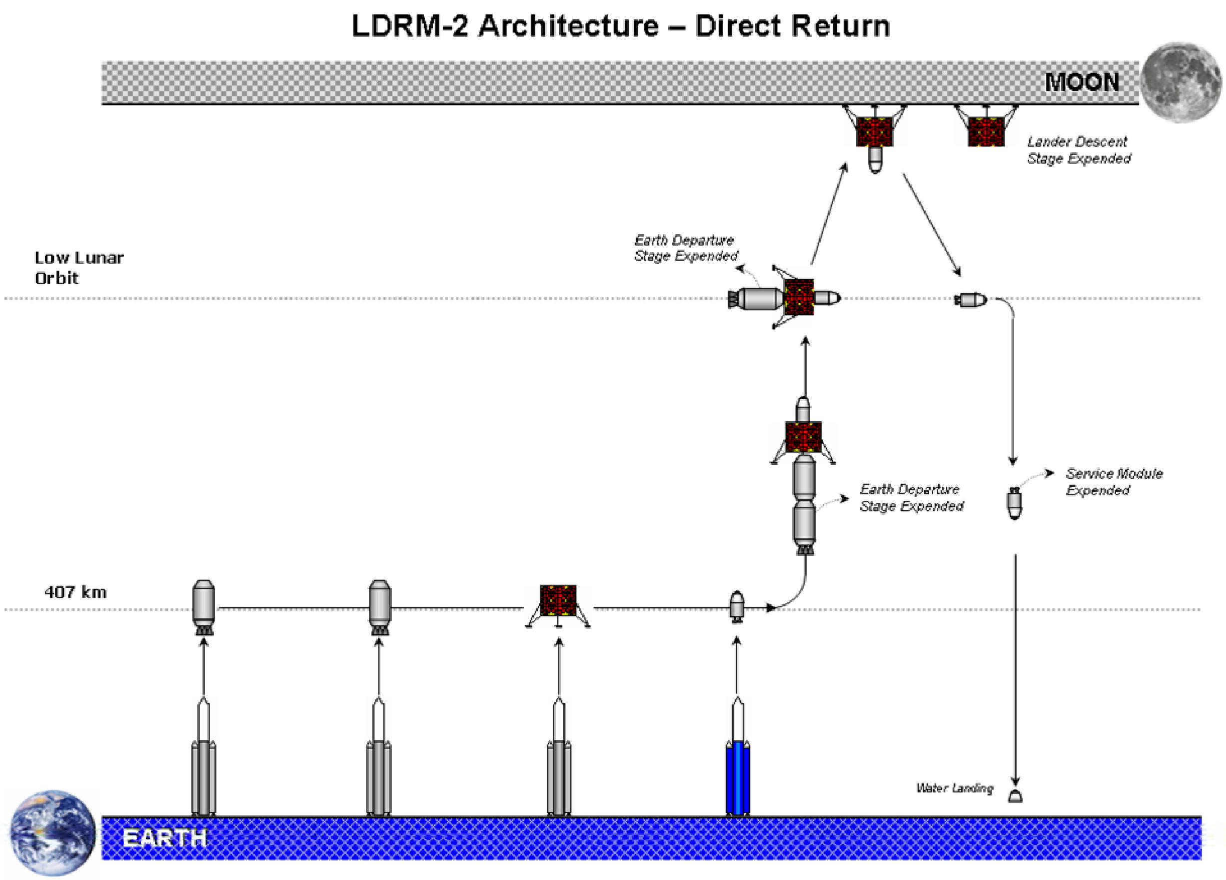


Figure 17.2-1: Mission Flow of the TRM with Direct Return

The nominal timeline for the direct return mission is shown graphically in Figure 17.2-2. The nominal mission duration from the launch of the first EDS through the recovery of the crew is approximately 62 days. The first 42 days of the mission are used to launch and assemble the two EDS elements and the lander descent stage in LEO. At the end of the sixth week the CEV is launched. The crew spends nearly 20 days in the CEV for a nominal direct return lunar mission including several days in LEO for phasing and rendezvous, eight days in transit between the Earth and the Moon, seven days on the lunar surface and up to two days of weather padding for the CEV launch to protect the Earth orbit departure opportunity. A single launch approach could reduce the crew mission duration for the direct return architecture by up to five days.

The two-week spacing between launches specified in the LDRM-2 task statement drives the total mission duration for a four-launch strategy, as well as the design lifetimes of the EDS and lander descent stage elements. The length of the LEO assembly sequence will directly influence the element design with respect to micrometeoroid and orbital debris exposure and thermal issues, such as the boiloff of cryogenic propellants.

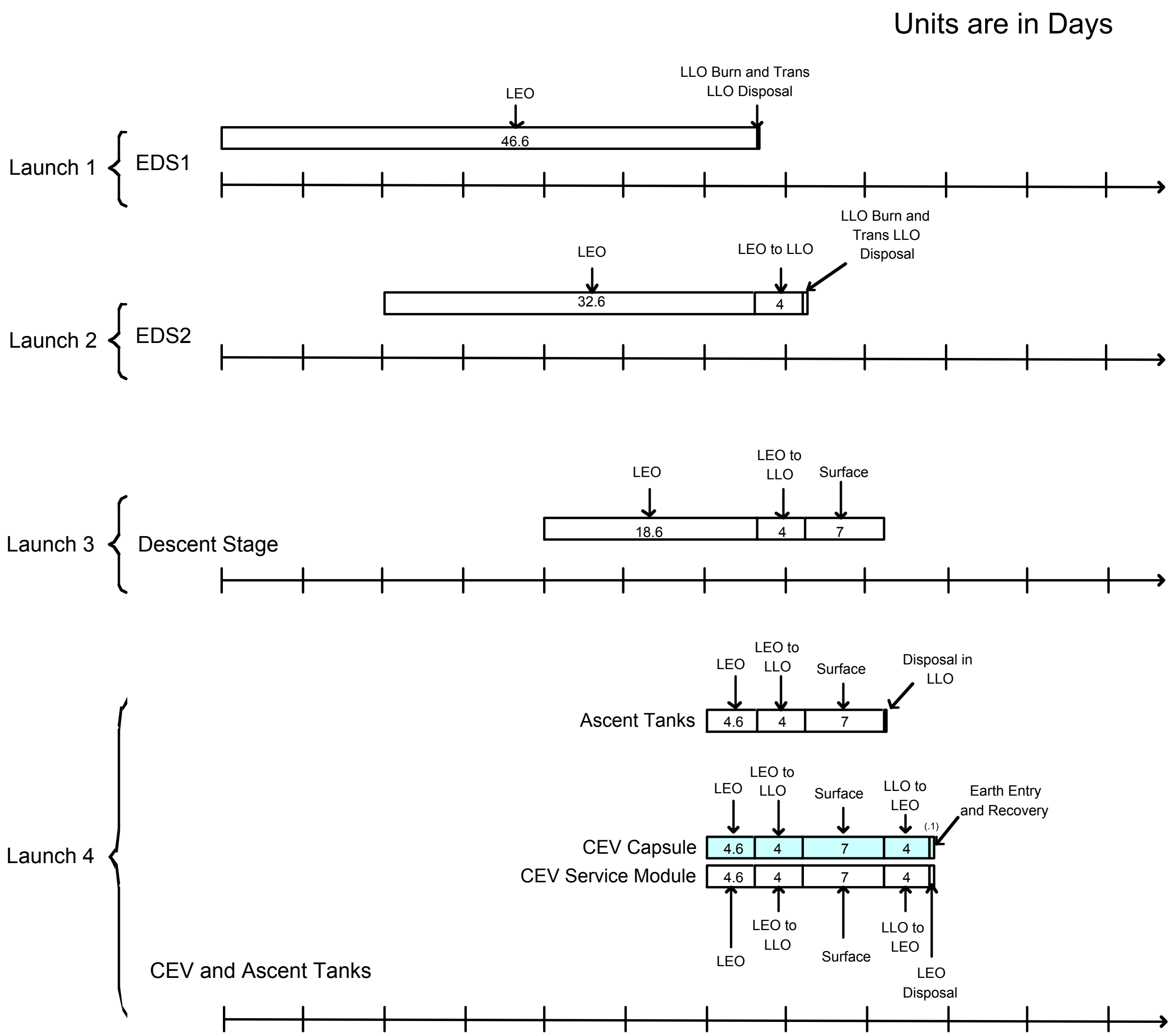


Figure 17.2-2: Nominal Mission Timeline for the Direct Return Architecture

17.3 Safety & Mission Success

The direct return approach is significantly different from the L1 TRM in terms of operations and critical events. The L1 TRM includes two Earth Departure Stages (EDS-1 & EDS-2), a lander plus a Kick Stage, and a CEV comprised of a Crew Module (CM) and Service Module (SM). The direct return approach also includes EDS-1 and EDS-2, but eliminates the additional Kick Stage. In addition, the lander consists of a combined lander descent stage and CEV.

Operationally, the L1 TRM specifies the pre-deployment of the lander to L1 with a subsequent rendezvous and mating of the CEV. In contrast, the direct return approach employs a tandem Earth orbit departure in which all four elements are mated in LEO. Because of the tandem EOD and the lack of a rendezvous requirement on the Earth return phase of the mission, the direct return approach has fewer critical events than identified for the L1 TRM. The L1 TRM has a total of fifty-six critical events. Twenty of the fifty-six critical events for the TRM are considered to occur during uncrewed portions of the mission while the remaining thirty-six are considered to occur during the crewed portions of the mission. The direct return approach has a total of forty-five critical events identified. Nineteen of the forty-five critical events for the Lunar Surface Rendezvous approach occur during the uncrewed portions of the mission. The remaining twenty-six critical events occur during the crewed portions of the mission.

Of the total forty-five critical events identified for the Lunar Surface Rendezvous approach, five received a rank of three, seventeen received a rank of two, and the remaining twenty-three received a rank of one. The complete set of identified and ranked critical events for the CEV/Lunar Lander Mating in LEO approach is listed in Table 17.3-1.

The probability of mission success for the direct return architecture is enhanced by the fact that it has eleven fewer critical events than the L1 TRM. The decrease in the number of critical events for the direct return architecture is due to the fact that the lunar elements travel to and from lunar orbit without stopping or rendezvousing at L1, thus eliminating one vehicle mating operation, two crew transfer operations, two vehicle de-mating operations, and several spacecraft maneuvers associated with the L1-to-Moon transfers.

	ID #	TRM with Lunar Surface Rendezvous Critical Events	TRM with Lunar Surface Rendezvous Critical Event Rank
Uncrewed Critical Events	VAR-07-01	EDS-1 Launch	1
	VAR-07-02	EDS-1 Ascent	1
	VAR-07-03	EDS-1 Launch Shroud Separation	1
	VAR-07-04	EDS-1 Separation from Booster	1
	VAR-07-05	EDS-1 Orbital Maneuvering	1
	VAR-07-06	EDS-2 Launch	1
	VAR-07-07	EDS-2 Ascent	1
	VAR-07-08	EDS-2 Launch Shroud Separation	1
	VAR-07-09	EDS-2 Separation from Booster	1
	VAR-07-10	EDS-2 Orbital Maneuvering	1
	VAR-07-11	EDS-1 Docks with EDS-2	1
	VAR-07-12	EDS-1 & EDS-2 Orbital Maneuvering	1

	ID #	TRM with Lunar Surface Rendezvous Critical Events	TRM with Lunar Surface Rendezvous Critical Event Rank
	VAR-07-13	Lander Descent Stage (LDS) Launch	1
	VAR-07-14	LDS Ascent	1
	VAR-07-15	LDS Launch Shroud Separation	1
	VAR-07-16	LDS Separation from Booster	1
	VAR-07-17	LDS Orbital Maneuvering	1
	VAR-07-18	LDS Docks to EDS-1 & EDS-2	1
	VAR-07-19	LDS, EDS-1, & EDS-2 Orbital Maneuvering	1
Crewed Critical Events	VAR-07-20	CEV (CM+SM) Launch with Launch Abort System	2
	VAR-07-21	CEV Ascent with Launch Abort System	2
	VAR-07-22	Launch Abort System Separation	2
	VAR-07-23	CEV Launch Shroud Separation	2
	VAR-07-24	CEV Separation from Booster	2
	VAR-07-25	CEV Orbital Maneuvering	2
	VAR-07-26	CEV Docks to LDS, EDS-1, & EDS-2	2
	VAR-07-27	EDS-1, EDS-2, LDS & CEV Earth Orbit Departure Burn for Low Lunar Orbit	2
	VAR-07-28	EDS-1, EDS-2, LDS & CEV Mid-course Correction Burn	1
	VAR-07-29	EDS-1 Separates from EDS-2, LDS, & CEV	2
	VAR-07-30	EDS-2, LDS, & CEV Mid-course Correction Burn	1
	VAR-07-31	EDS-2, LDS, & CEV Lunar Orbit Arrival Burn	2
	VAR-07-32	EDS-2 Separates from LDS & CEV	2
	VAR-07-33	LDS & CEV Orbital Maneuvering	2
	VAR-07-34	LDS & CEV Powered Lunar Descent & Landing	3
	VAR-07-35	CEV Depress/Repress	2
	VAR-07-36	CEV Separates from LDS and Ascends to LLO	3
	VAR-07-37	CEV Orbital Corrections	1
	VAR-07-38	Ascent Stage separation from the CEV (only applies to a two-stage CEV design)	2
	VAR-07-39	CEV Lunar Orbit Departure	3
	VAR-07-40	CEV Burn for Earth	3
	VAR-07-41	CEV Mid-course Correction Burn	1
	VAR-07-42	CM Separates & Maneuvers away from SM	2
	VAR-07-43	CM Entry	3
	VAR-07-44	CM Landing	2
	VAR-07-45	Crew Recovery	2

Table 17.3-1: Direct Return Architecture Critical Events and Rank

With respect to crew safety, the advantages of the direct return approach relative to the L1 TRM clearly outweigh the disadvantages. The simplicity of the direct return approach is evident in the reduced number of crewed critical events relative to the L1 TRM. The reduction from 36 to 26 crewed critical events should translate to an improvement in reliability and overall crew safety. Another advantage is that the crew is always in close proximity to the CEV in the direct return approach, even while on the lunar surface, and can more rapidly return to Earth without an orbital rendezvous. In the case of the L1 TRM, the crew spends nearly half of their time in the lander and may be several days away from a rendezvous opportunity with the CEV.

The single crew module design of the direct return architecture does not provide the same range of functional redundancy during the outbound transit that is offered in a tandem EOD approach with two separate crew modules. However, the split mission L1 TRM also lacks crew module redundancy on the outbound transit due to the pre-deployment of the lander element to L1.

17.4 Mission Abort Options

The abort options for the direct return architecture during the low Earth orbit assembly and outbound transit mission phases are similar to the abort options for the Lunar Orbit Rendezvous and CEV/Lander Mating in LEO variants to the L1 TRM. All three of these architecture variants employ an integrated lunar stack with a tandem Earth orbit departure that offers propulsive redundancy at the element level for the safe return of the crew to Earth. This propulsive redundancy is gradually depleted through successive maneuvers until the lander reaches lunar orbit and begins the descent burn. The opportunities for functional redundancy at the element level are largely ended once the lander reaches the terminal descent and landing phase of the mission.

Because the direct return architecture uses a single crew module, it does not have the element level of redundancy for habitation and life support that was employed with great success during the Apollo 13 mission. However, it should be noted that the direct return architecture provides the distinct advantage that the crew is always in close proximity to the CEV, even while on the lunar surface, and can return to Earth without an orbital rendezvous. It should also be noted that the functional redundancy offered in a dual crew module architecture with a tandem EOD only exists during the outbound transit phase prior to the separation of the lander and CEV in the lunar vicinity.

17.5 Element Overview & Mass Properties

The definition of the flight elements for the direct return architecture is primarily driven by the allocation of the major propulsive maneuvers. The preferred design approach is to shift as much ΔV as possible from the lander to the Earth Departure Stage to reduce the architecture IMLEO as well as to minimize the volume and dimensions of the lander descent stage. In its simplest form the direct return architecture can be accomplished with only two propulsive stages – an EDS and a single-stage lander. From a practical standpoint, however, the lander is typically divided into two propulsive stages for the lunar descent and lunar ascent/Earth return maneuvers. The typical maneuver allocation for the direct return mission is as follows:

- Earth Departure Stage – Earth orbit departure and lunar orbit arrival
- Lander Descent Stage – lunar descent
- CEV – lunar ascent and lunar orbit departure (Earth return)

It should be noted that the lunar orbit departure maneuver is relatively small (ΔV~850 m/s) and is essentially identical to the ΔV total allocated to the lander ascent stage in the L1 TRM for the lunar orbit departure (600 m/s) and L1 arrival (248 m/s) maneuvers. Staging of inert mass is typically worthwhile only when the maneuver ΔV exceeds a substantial multiple of the propulsive efficiency (Isp).

In the TRM four-launch scenario, the EDS is divided into two equal-mass stages that are mated in LEO to reduce the maximum launch vehicle payload requirement. Additional staging is possible, but results in rapidly diminishing performance advantages coupled with increasing spacecraft complexity and cost.

The direct return architecture provides the ability to orient the lunar arrival and departure orbits to minimize total mission ΔV while preserving the capability for global lunar access with anytime Earth return. The Earth orbit departure and lunar orbit arrival maneuvers place the lander in an orbit that provides a coplanar descent to the landing site. At the conclusion of the lunar surface operations, the CEV performs an ascent maneuver to a low lunar orbit that is oriented to minimize the ΔV for the lunar orbit departure maneuver. Despite this ΔV advantage, however, the direct return approach results in the highest architecture IMLEO due to the burden of transporting the Earth return propellant to and from the lunar surface.

Due to resource limitations, the sizing of flight elements for the direct return architecture was deferred until after the completion of the LDRM-2 Phase 1 and Phase 2 studies. However, some initial sizing estimates performed during the LDRM-2 study indicate that a direct return architecture that is sized to meet the basic L1 TRM requirements will have an IMLEO in the range of 230 to 260t. The higher end of the IMLEO range reflects the use of a pressure-fed liquid oxygen and methane propulsion system for the lander descent stage. The lower end of the IMLEO range reflects a higher performance, pump-fed propulsion system using liquid oxygen/methane or liquid oxygen/hydrogen. Due to the volumetric advantages of liquid methane relative to liquid hydrogen, it is very possible that the combination of oxygen and methane will result in a lower IMLEO despite its somewhat lower Isp (see Section 19.1).

17.6 System Technologies and Programmatic Risks

The direct return architecture primarily involves a reconfiguration and blending of the L1 TRM functional requirements for the lander and CEV. As a result, the majority of the vehicle system technology assumptions made for the L1 TRM also apply to the direct return architecture variant. However, the emphasis on higher propulsive efficiency for the lander descent stage in the direct return architecture will likely require technology development for a throttleable, pump-fed lander engine.

17.7 Pros/Cons Summary

The direct return architecture provides the same general functionality as the L1 rendezvous architecture in terms of global lunar access with anytime Earth return capability for short or long duration lunar surface missions. In the case of the direct return approach, however, the one-way transit time from the Earth to the Moon is approximately four days rather than the seven days required for an L1 rendezvous approach. The elimination of the L1 rendezvous and associated propulsive maneuvers also greatly decreases the number of critical mission events relative to the L1 TRM.

The direct return architecture offers an interesting potential for reducing the number of separate propulsion stages and dynamic on-orbit interfaces. Through the use of on-orbit fueling, a direct return mission can be packaged in as few as three flight elements – an EDS, lander descent stage and CEV – within a reasonable range of launch vehicle payload capacity. The L1 rendezvous and lunar orbit rendezvous architectures are better suited to a four element approach – an EDS, two-stage lander and CEV.

The direct return architecture does have some disadvantages. The most obvious drawback to the direct return architecture is a high initial mass in low Earth orbit, even with the use of a high performance propulsion system for the lander descent stage. Another disadvantage is the fact that the single CEV crew module must envelope the habitation functionality for all mission phases including launch, in-space transit, lunar landing and ascent, and lunar surface operations. A related concern is that the CEV will be exposed to a wide range of thermal and lighting conditions on the lunar surface defined by the latitude of the selected lunar landing site and the local terrain.

18.0 Architecture Comparison

The Trade Reference Mission developed using the L1 rendezvous architecture is the core of the LDRM-2 Phase 1 study. The L1 TRM operations concept and timeline development, flight element definition and mass properties estimates provide a framework from which to compare alternative architectures and mission options.

The TRM variants that utilize the L1 rendezvous approach provide an increased depth of understanding of the mass and operational sensitivities to incremental changes in architecture definition or variations in mission design parameters. It should be noted that the mass sensitivity results are directly related to the specific flight element configurations developed for the L1 TRM. Variations in the propellant selection or ΔV allocations for these flight elements, for example, may have a significant effect on the mass sensitivity results.

Although the LDRM-2 Phase 1 study focuses on the L1 TRM, it also provides a framework from which to compare the three basic lunar architectures – libration point rendezvous, lunar orbit rendezvous and direct return – using a common set of lunar mission requirements. A higher priority was placed on analyses of the lunar orbit rendezvous architecture because it offers a broad range of optimization options. A lower priority was placed on the direct return architecture during the LDRM-2 Phase 1 study, primarily because of the availability of detailed data from previous studies such as the Lunar Lander Design for the First Lunar Outpost study (JSC-25896) in 1992.

18.1 L1 TRM and Architecture Variants

A comparison chart of the total architecture mass (IMLEO) and element mass results for the L1 TRM and its associated L1 rendezvous architecture variants is provided in Figure 18.1-1. The reference L1 TRM data is provided at the left side of the chart followed by five separate architecture variants, all of which employ the basic L1 rendezvous approach. One mission design variable was altered for each of the architecture variants. The mass data shows relatively little variation in IMLEO for the L1 variants in comparison to the L1 TRM estimate of 230t. The lowest IMLEO estimate is 216t (-6.1%) for the two-launch variant, while the highest IMLEO estimate is 241t (+4.8%) for the variant constrained by a 25t launch vehicle payload capacity. This small range of variation in IMLEO indicates that the architectural framework, itself, is the primary determinant of IMLEO for the trade reference mission. Other significant IMLEO drivers identified during the development of the L1 TRM include the flight element propellant selection and ΔV allocation.

The mass distribution among the flight elements and its resulting impact on launch vehicle packaging is likely to be a more important figure of merit for a human exploration mission than the overall IMLEO. The L1 TRM, for example, includes six separate propulsion stages ranging from 18t for the CEV Service Module to 94t for the lander Earth Departure Stage. Three of these stages – the Kick Stage (27t), Descent Stage (23t) and Ascent Stage (20t) – combine to form the lander element with a gross mass of 70t. The difference in the gross masses of the lander EDS (94t) and CEV EDS (39t) is driven by the requirement to pre-deploy the lander

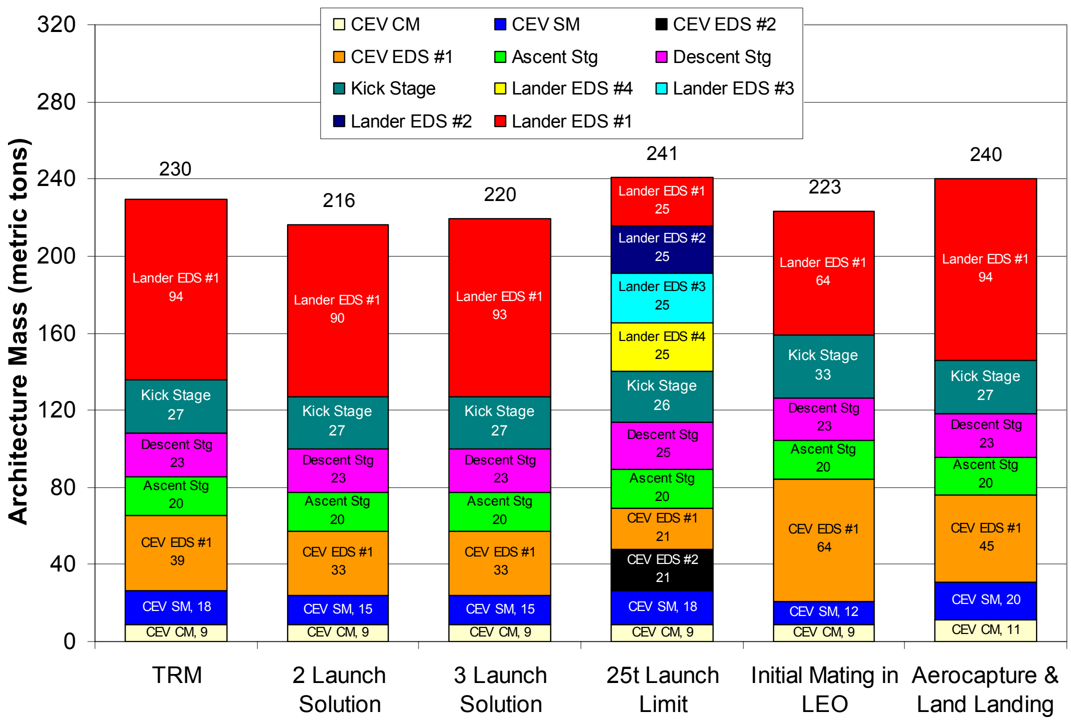


Figure 18.1-1: IMLEO for the L1 TRM and Architecture Variants

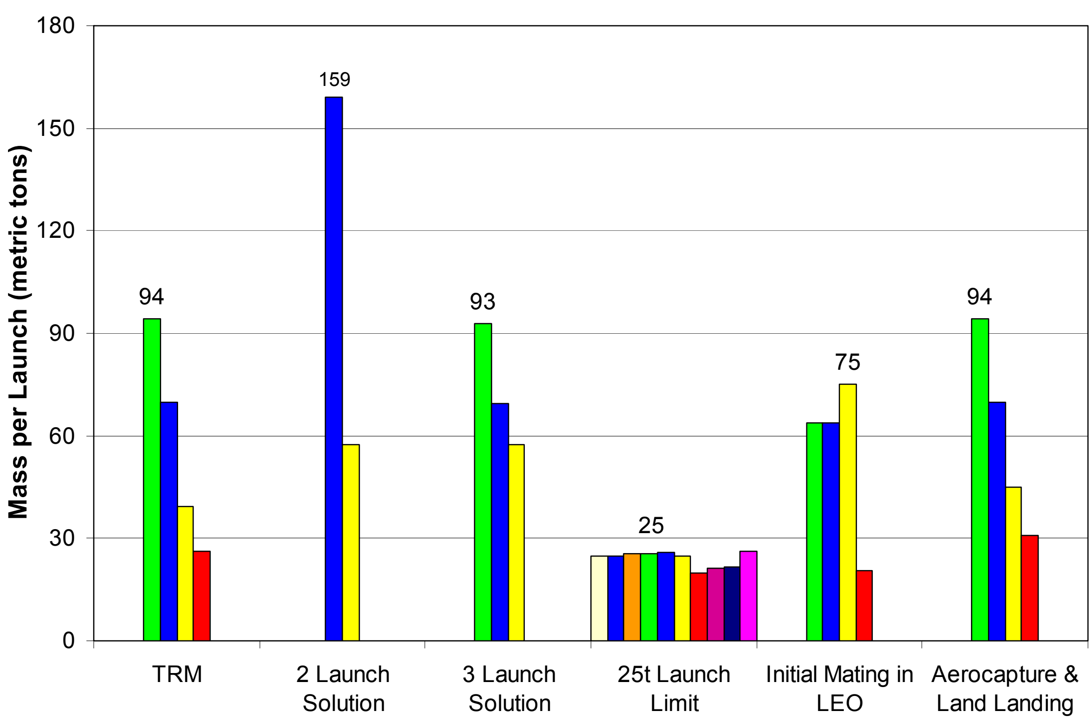


Figure 18.1-2: Mass per Launch for the L1 TRM and Architecture Variants

to L1 in combination with the high mass of the lander element in the L1 rendezvous architecture. A comparison chart illustrating the mass per launch for the L1 TRM and the L1 variants is provided in Figure 18.1-2. The primary drivers of the maximum required launch vehicle payload capacity are the number of launches, Earth orbit departure strategy and the natural design breakpoints for the flight elements, particularly the lander. If the maximum launch vehicle payload mass is specified rather than number of launches, then the primary consideration is the largest single element or stage that must be launched in one piece. On-orbit fueling of propulsive stages is potentially a very powerful tool for addressing launch vehicle payload limitations.

18.1.1 L1 TRM Variant: Two-Launch

Because the lander and lander EDS are combined into a single launch in the two-launch variant, the maximum launch vehicle payload mass requirement increases from 94t to 159t and the two launches are highly unbalanced. The combined CEV and CEV EDS launch mass of 57t is the driver for the payload capacity of the human-rated launch vehicle.

The primary operational benefits of the two-launch approach are the result of the integrated launch of the CEV and CEV EDS. Because the lander is pre-deployed to L1, the integrated launch eliminates the need for LEO rendezvous and assembly, thus enabling daily launch and Earth orbit departure opportunities for the crewed vehicle. The elimination of the LEO rendezvous operations also results in a 4.5-day reduction in the crew mission duration.

The two-launch variant offers several other benefits relative to the L1 TRM. The elimination of two launches and two LEO rendezvous and docking operations increases the probability of mission success. The ground integration of the lander and CEV with their EDS elements also eliminates two dynamic mating interfaces and simplifies testing. The two-launch approach also reduces the mission duration for the lander and EDS elements as well as minimizing their exposure to the environmental hazards in LEO.

18.1.2 L1 TRM Variant: Three-Launch

In the three-launch variant the CEV and CEV EDS are combined into a single launch, but the lander and lander EDS are launched separately. As a result, most of the CEV operational benefits noted for the two-launch variant are retained even though the maximum launch vehicle payload mass drops back to the original L1 TRM value of 94t for the lander EDS. The three launches are also more closely balanced in terms of launch mass with the lander at 70t and the combined CEV/CEV EDS at 57t. The payload requirement for the human-rated launch vehicle is unchanged from the two-launch variant.

The three-launch strategy eliminates one launch and one LEO rendezvous and docking operation in comparison to the L1 TRM. Therefore, the probability of mission success is likely to fall between the values for the L1 TRM and the two-launch variant. Ground integration and testing can still be applied to the critical CEV/CEV EDS launch package. The three-launch approach reduces the mission duration for the lander by 14 days, but does not provide any operational benefits for the lander EDS.

18.1.3 L1 TRM Variant: 25 t Launch Limit

The application of a 25t launch limit resulted in a total of ten launches. The lander portion of the mission requires seven launches including four Earth Departure Stages and a Kick Stage, as well as separate launches for the lander descent and ascent stages. In order to bring the Kick Stage mass down to the 25t launch limit, a portion of the lunar orbit arrival ΔV had to be shifted from the Kick Stage to the lander descent stage. The CEV portion of the mission requires three launches including two Earth Departure Stages and the CEV. At 27t, the CEV slightly exceeds the specified launch constraint.

The sole benefit of this approach relative to the L1 TRM is the reduction in the maximum required launch vehicle payload capacity from 94t to 25t. The disadvantages, however, are numerous. Chief among those is the reduction in the probability of mission success for the seven launches required to deploy the lander to L1 followed by three more launches to assemble the CEV stack. Assuming that the launches are serial and spaced at two-week intervals, the mission duration from first launch through crew recovery is approximately five months, resulting in increased on-orbit loiter times for the EDS and lander elements. In addition, the complexity of the mission will be increased by the multi-stage Earth orbit departure maneuvers and the need to safely dispose of the expended EDS stages.

On-orbit fueling should be seriously considered for lunar architectures that are highly constrained in terms of payload mass. The offloading of propellant can significantly reduce the number of launches required to deliver the flight element hardware to orbit. In addition, the aggregation of the offloaded propellant at a LEO depot will decouple the propellant launch schedule from the launch and assembly schedule for the flight elements.

18.1.4 L1 TRM Variant: Initial CEV/Lander Mating in LEO (Tandem EOD)

The mating of all of the flight elements in LEO, which results in a tandem Earth orbit departure, has several design and operational impacts on a lunar mission. First, the Earth Departure Stages can be divided on an equal-mass basis rather than being sized to independently deliver the lander and CEV to the lunar vicinity. Since the lander EDS is the launch vehicle driver for the L1 TRM, the switch to equal-mass Earth Departure Stages significantly reduces the required launch vehicle payload capacity. Second, the tandem EOD enables more flexibility in the allocation of maneuver ΔV among the flight elements. This enables the CEV, for example, to perform maneuvers for the combined CEV/lander configuration, just as the CSM performed the lunar orbit insertion for the CSM/LEM in the Apollo Program. Third, the mated CEV and lander provide redundant functionality for crew habitation, power generation and thermal control during the outbound leg of the lunar mission. This type of redundancy was invaluable during the Apollo 13 mission. Finally, the mated stack can be fully checked out in LEO prior to initiating the Earth orbit departure burn.

Three different element combinations were developed for the tandem EOD variant to the L1 TRM. The IMLEO and mass per launch results for these options are provided in Figures 18.1.4-1 and 18.1.4-2, respectively. The L1 TRM has an IMLEO of 230t and a launch mass of 94t.

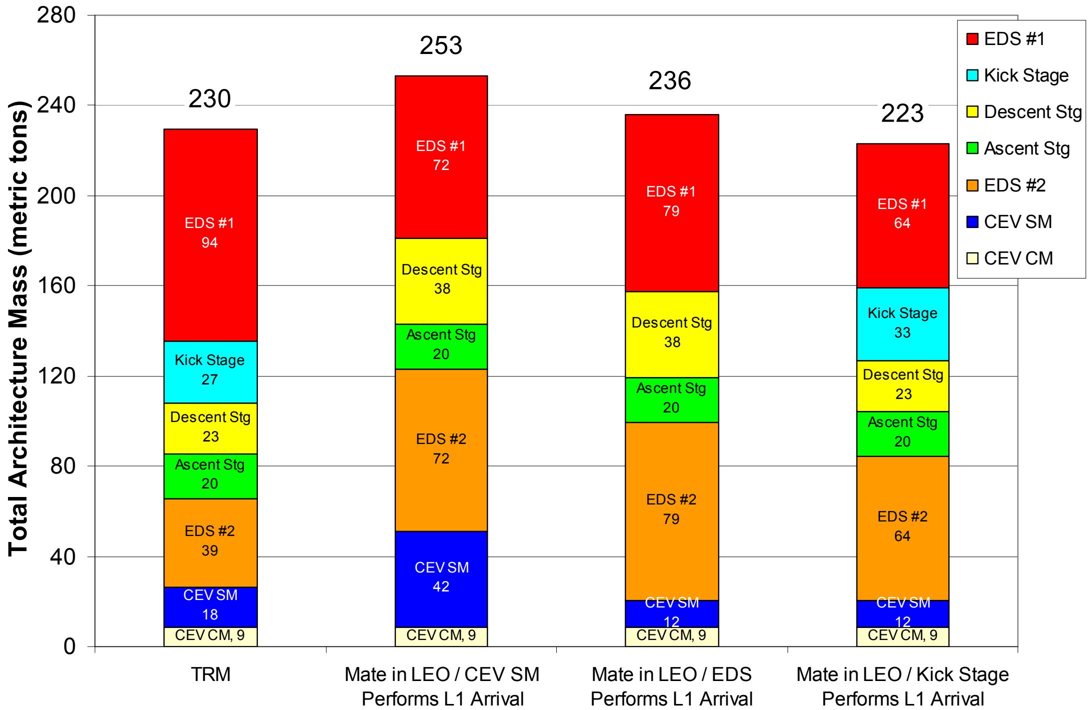


Figure 18.1.4-1: IMLEO for the L1 TRM and Tandem EOD Options

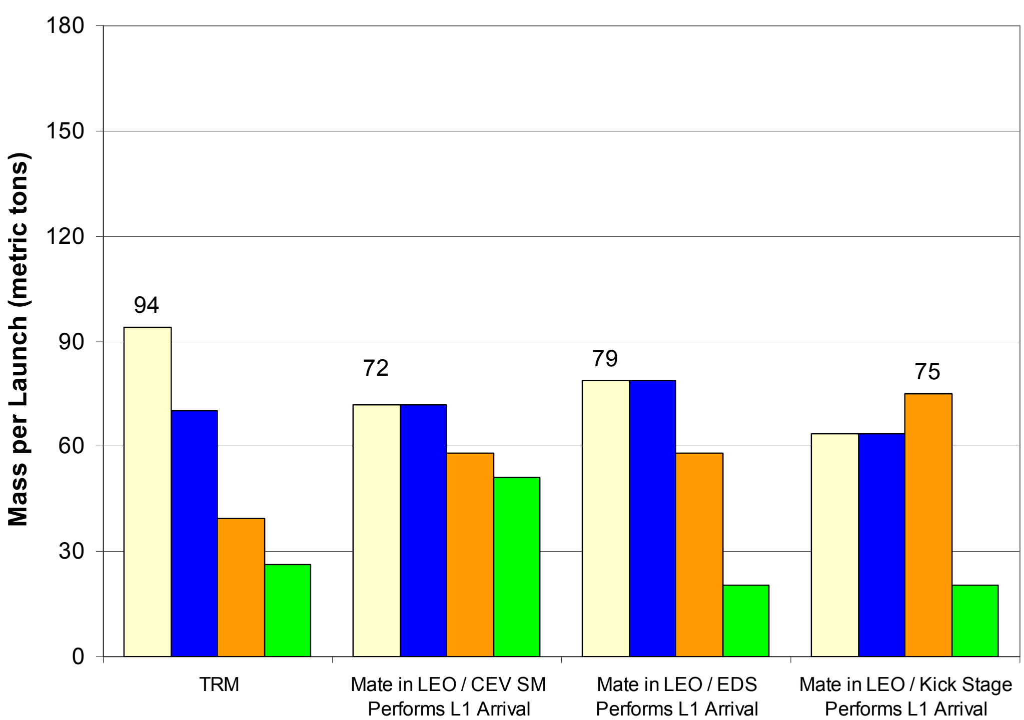


Figure 18.1.4-2: Mass per Launch for the L1 TRM and Tandem EOD Options

The first tandem EOD option eliminates the lander Kick Stage and allocates the L1 arrival maneuver to the CEV. This option provides the lowest launch vehicle payload requirement (72t), but also increases the IMLEO by 23t (10%) relative to the L1 TRM due to the lower Isp of the CEV propulsion system. The second option also eliminates the lander Kick Stage, but allocates the L1 arrival maneuver to an EDS, instead. This option reduces the mass penalty of eliminating the Kick Stage to only 6t (2.6%). Because the EDS is the largest flight element, however, this option is not as effective at reducing the launch vehicle payload requirement (79t). The third option retains the lander Kick Stage from the L1 TRM and uses it to perform the L1 arrival maneuver for the combined CEV/lander. The staging efficiency provided by this option actually reduces the IMLEO by 7t (3%) relative to the L1 TRM. However, the sharp decrease in the mass of the EDS along with the associated increase in the mass of the Kick Stage results in the lander becoming the largest element with a combined mass of 75t.

The primary drawback of the tandem EOD approach in a four-launch strategy is the increased complexity of docking the elements in LEO. The split mission EOD approach enables a single dynamic docking interface on each flight element. The tandem EOD approach requires two docking interfaces on the EDS and lander to permit coaxial assembly. The Earth Departure Stages must also loiter in LEO for up to four weeks longer in the tandem EOD launch sequence. The lander exchanges loiter time at L1 for loiter in LEO, but has a lower total mission duration because of the tandem EOD launch sequence.

18.1.5 L1 TRM Variant: Aerocapture and Land Landing

The most straightforward Earth return strategy is the direct entry approach that was used in the Apollo Program. In the direct return approach the latitude of the Crew Module recovery site at Earth is defined by the region around the lunar antipode at the time of lunar departure. The longitude of the CM recovery site is controlled by adjusting the period of the Earth return trajectory to allow the Earth to rotate to the desired orientation. Assuming that the Service Module is expendable, its disposal footprint will limit the range of viable recovery sites for the CM. For this reason, the direct entry approach is generally linked with an ocean recovery of the CM.

In the aerocapture variant the CEV CM employs aerodynamic braking to achieve a low Earth orbit rather than performing a direct entry. The CM then loiters in LEO to phase with a recovery site before executing the de-orbit maneuver. A CM loiter capability of twelve hours was selected for the aerocapture variant to the L1 TRM. However, detailed landing site analyses are required to accurately define a loiter time that envelopes a range of CM phasing orbits, number and locations of landing sites, CM hypersonic L/D, and recovery site constraints, such as lighting conditions. The recovery site for the CM is not directly tied to the disposal footprint of the SM, and land recovery is an option given a sufficient number and distribution of suitable landing sites. The aerocapture approach also enhances the capability of the CM to respond to adverse weather conditions at a primary landing site.

The aerocapture and de-orbit approach results in several design impacts to the CEV. First, assuming an expendable SM, the CEV CM must independently provide power generation, thermal control and life support functionality during the LEO loiter period. Second, the CM design must incorporate the additional ΔV required to perform the LEO de-orbit maneuver. In response to the

100 m/s of additional ΔV, the CM Tridyne RCS was changed to a higher efficiency, but more complex, LO2/ethanol system. Third, the CM recovery system was enhanced with the addition of deployable airbags to attenuate the impact loads for land landing.

Relative to the L1 TRM, the aerocapture variant results in a 10t increase in IMLEO. The increase in the mass of the CEV from 27t to 31t affects the launch capacity of the human-rated launch vehicle. The increase in the mass of the CEV CM also increases the mass of the launch escape system. Since the lander EDS is the largest element, the aerocapture variant does not affect the launch capacity for the cargo launch vehicle.

It should be noted that a 'skip entry' is currently being analyzed that offers some of the benefits of the aerocapture approach while reducing the impacts to the CEV design.

18.2 Lunar Orbit Rendezvous Variant to the L1 TRM

The lunar orbit rendezvous variant developed for the LDRM-2 Phase 1 study involves three mission design changes relative to the L1 TRM. First, the CEV/lander rendezvous location on the return leg of the mission occurs in low lunar orbit rather than L1. This reduces the ΔV allocated to the LOR lander element relative to the L1 TRM. Second, a tandem Earth orbit departure with equal-mass Earth Departure Stages is employed for the LOR variant. Third, the lander Kick Stage is eliminated.

The IMLEO and mass per launch results for the L1 TRM and two LOR options are provided in Figures 18.2-1 and 18.2-2, respectively. The LOR architecture provides the functional capability required for the TRM – 7 days on the lunar surface with anytime Earth return – with an IMLEO of 199t, or 31t (13.5%) less than the L1 rendezvous approach. The tandem EOD and reduced IMLEO also lower the required launch vehicle capacity to 71t from the L1 TRM value of 94t. Mass results are also provided for an LOR architecture that substitutes loiter capability for propulsive anytime Earth return. From a mass standpoint the use of loiter in place of propulsion further reduces the IMLEO and required launch vehicle capacity for the LOR variant to 169t and 57t, respectively. Intermediate combinations of loiter and propulsive plane change are also possible. These results clearly demonstrate the potential for mass optimization using the LOR architecture.

In terms of probability of mission success and crew safety, the LOR architecture with propulsive anytime return provides some noteworthy benefits relative to the L1 TRM. Aspects of the LOR architecture favorable to mission success include a modest reduction in the total number of critical events and the elimination of the Kick Stage. Crew safety is enhanced by the elimination of five crewed critical events as well as a reduction in the time required to return to Earth from the lunar surface.

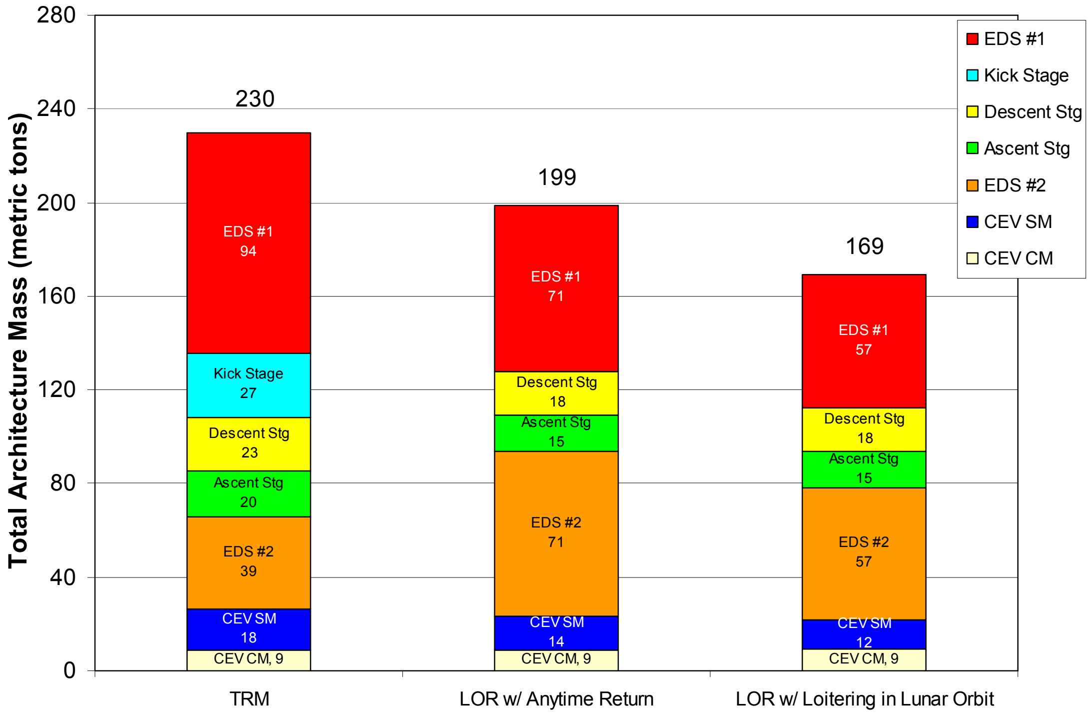


Figure 18.2-1: IMLEO for the L1 TRM and LOR Variants

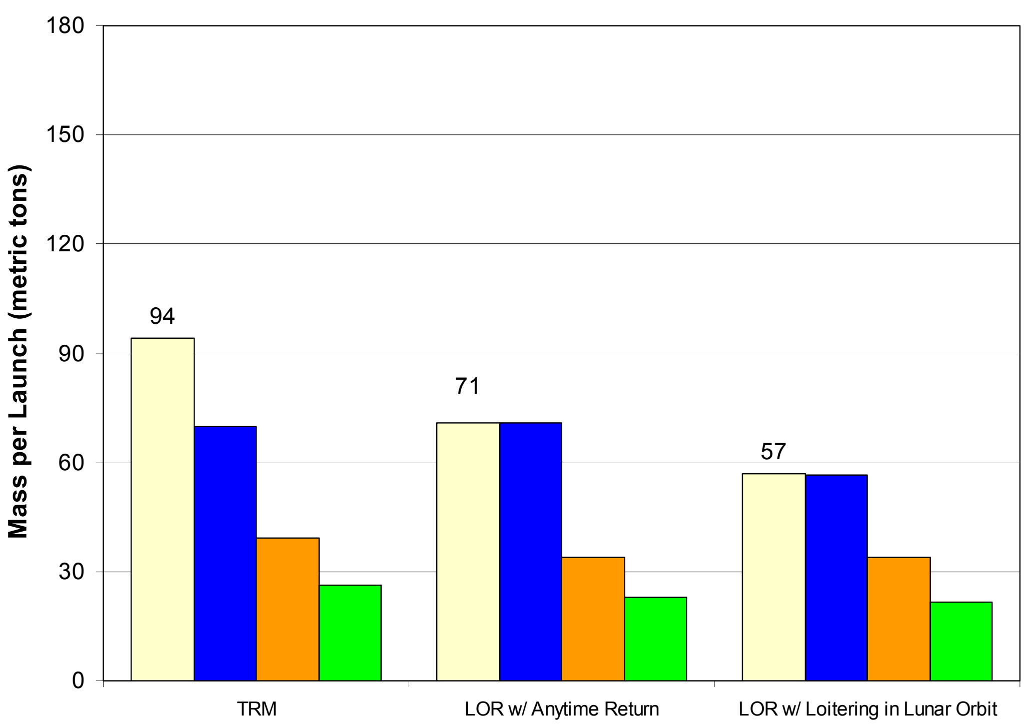


Figure 18.2-2: Mass per Launch for the L1 TRM and LOR Variants

19.0 Parametric Variations

Previous sections of this report examined how key architecture parameters selected for the LDRM-2 trade reference mission (TRM) such as rendezvous nodes and number of launches per mission could be modified, and each section measured the effect of those changes relative to the TRM in four Figure of Merit categories. This section looks at how different vehicle design or technology parameters vary and determines their effects on the mission, again measured relative to the TRM. The particular parametric variations detailed here (crew size, propellant type, mission duration, etc.) were selected because the impacts of their modification are not isolated to a single vehicle. Rather, the impacts have a ripple effect through most or all vehicles in the architecture. The parametric trades performed for the LDRM-2 study are below.

1. Alternate Propellants
2. Alternate Power Sources
3. Return Payload Mass
4. Landed Payload Mass
5. All vs. Partial Crew to the Surface
6. Crew Size of 2
7. Crew Size of 6
8. Launch Interval Delay Between 7 and 30 Days
9. Surface Duration of 3 Days
10. Surface Duration of 14 Days
11. Elimination of Contingency CEV EVA Requirement
12. Recommended Cabin Design Pressure and Mass Effects

19.1 Alternate Propellants

This section describes the figures of merit, data, and trade study for alternate propellants for the CEV (Crew Exploration Vehicle) and Lander stages. Earth-storable propellants were successfully used in Apollo, Gemini, and Shuttle. However during the Shuttle program, long term issues with Earth-storable propellant such as valve corrosion and leakage, toxic propellant leakage, heater power, propellant freezing, and propellant cost and availability became more pronounced. These are not desirable characteristics as the basis for future robust exploration. A higher performance, more operationally efficient, reliable, and safe propulsion system is needed for the Lunar and Mars mission. Furthermore, using propellants compatible with in-situ resource utilization, power, and life support systems will increase flexibility for future mission architectures. The challenge to is to determine which propellant best meets future needs and which can be implemented with minimal risk to the program to support 2008, 2011 and 2014 CEV and Lander missions. As shown in Figure 19.1-1 it is useful to consider the duty cycle, thrust level, and total impulse of the different vehicles.

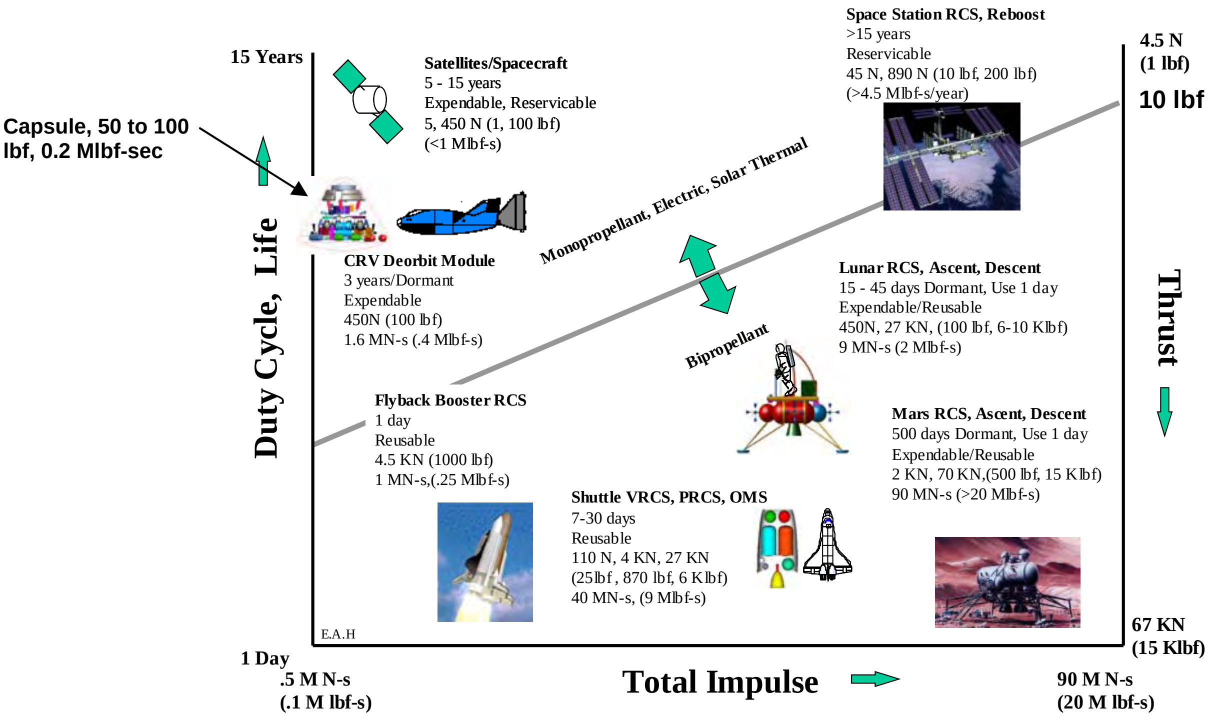


Figure 19.1-1: Duty Cycle – Thrust – Total Impulse Considerations

A recommendation provided is to pursue technology for a pressure-fed liquid oxygen and methane propulsion for the CEV Service Module or Lander. The LO2 is common with life support, power, and thermal control systems. A pressure-fed LO2/methane saves 2600 lbm compared to MMH/NTO, and offers additional cost, operational, safety, vehicle integration benefits. Due to mass, safety, reliability, complexity, packaging, and performance reasons, a pump-fed LO2/LH2 system is not recommended for a Service Module or Lander.

A recommendation provided is to pursue technology for a monopropellant (GN2, Tridyne, Nitrous Oxide) system for the CEV entry vehicle. A monopropellant reaction control system (RCS) is the simplest, low cost and risk solution to support an early 2008 demonstration, such as a CEV capsule tests, if desired. The schedule for technology for a pressure-fed LO2/methane propulsion system could support a 2011 unmanned vehicle (if needed) and 2014 Service Module. A pressure-fed deep throttling cryogenic engine could use a pintle concept such as used in Apollo.

19.1.1 CEV Service Module / Lander Alternate Propellant Trade Study

A number of propellants have been evaluated for a Service Module or Lander type vehicle; oxygen, hydrogen peroxide, nitrogen tetroxide (NTO), with ethanol, methane, mono-methyl hydrazine (MMH), and hydrogen. Only clean burning hydrocarbons were evaluated.

> **Trade #1** is a pressure-fed MMH/NTO. This type of system was used on Apollo and space shuttle.
>
> **Trade #2** is pressure-fed Hydrogen Peroxide (H2O2)/Ethanol system.

Trade #3 is a pressure-fed LO2/ethanol OMS and RCS. The LO2 is fed as a sub-cooled liquid. This option uses ethanol, which is a high density, storable clean burning fuel, which also provides a reliable coolant for RCS and OMS engines. Since fuel does not undergo density changes, when LO2 density decreases or becomes two phase, the combustion becomes more fuel rich and temperatures decrease.

Trade #4 is a pressure-fed LO2/LCH4 OMS and RCS. The cryogenic RCS propellants are kept conditioned in the lines by a combination of passive insulation, nominal propellant usage, and cryocoolers. This option shares some technology with the LO2/ethanol system, except that the methane is also cryogenic.

Trade #5 is pump-fed LO2/ethanol system

Trade #6 is a pump-fed hydrogen peroxide system.

Trade #7 is a pressure-fed LO2/LCH4 OMS with an integrated cryogenic RCS feedsystem system.

Trade #8 is a pump-fed LO2/LCH4 system.

Trade #9 is a LO2/LH2 pump-fed OMS with an integrated RCS. The main OMS tanks are at a low pressure to avoid the mass penalty of a large high pressure LH2 tank. The RCS then takes the OMS tank propellants and boosts the pressure to 250, and gasifies the propellants using heat from a gas generator. The primary issue with this system is the large amount of heat required to gasify the propellants and the complex controls required. There is a higher number of critical failure modes.

Trade #10 is a pressure-fed LO2/LH2. To provide the lowest mass possible, the He pressurant was stored as a supercritical liquid. The Ghe was then warmed to LH2 temperature using HEX with the LO2 to pressurize the LH2 tank. The RCS is integrated with OMS. The liquid oxygen and hydrogen propellants are delivered directly to the RCS engine as a subcooled liquid. Cryocoolers are used to keep the lines conditioned and remove heat. A more detailed discussion of this technique is provided in the RCS feedsystem design discussion.

19.1.1.1 Evaluation of the Propellants

The total mass, dry mass, complexity, and volume trades are summarized in Figures 19.1.2.1-1, 2, 3, and 4 for a hypothetical Service Module type propulsion stage with a 19700 lbm inert (capsule, equipment) mass and 6500 ft/sec (1975 m/sec) delta-V. For the LO2 based propellants, the power system reactants were stored in the propellants tanks which eliminated the high pressure PRSD tank. Also the amount of power reactants was reduced since the 2000 W of heaters are eliminated. This also results in a smaller fuel cell and thermal control system. The net result is 1100 lbm payload mass saving for all the LO2 based propulsion systems. As shown in figure the overall savings for LO2/methane versus MM/NTO is approx 2600 lbms, which is significant. The analysis considered the mass of the vehicle structure as a function of vehicle size or volume.

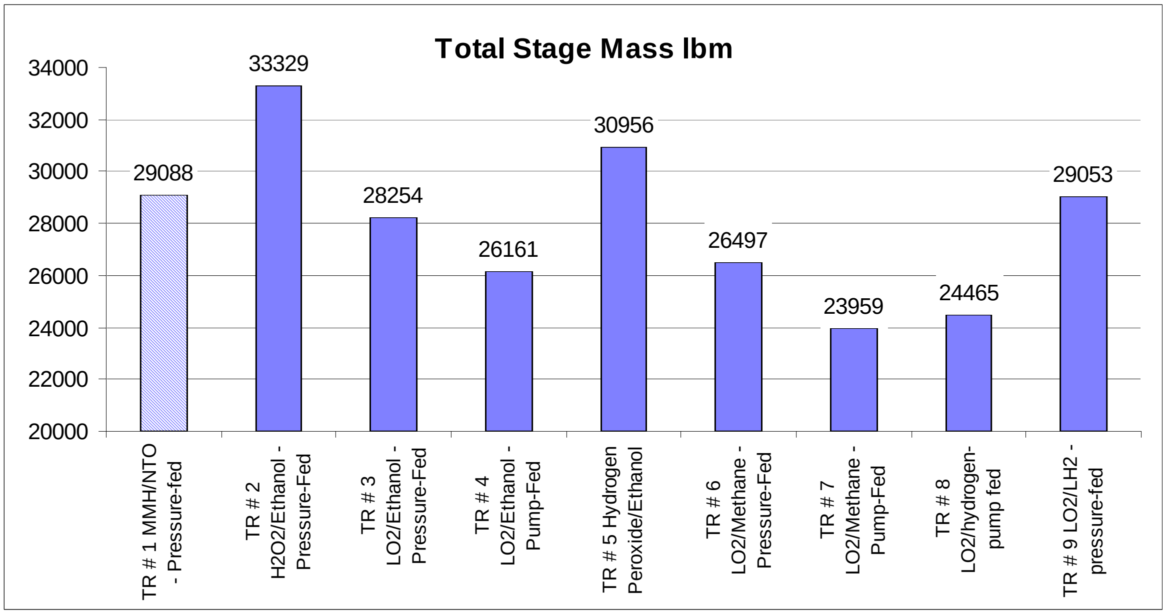


Figure 19.1.1.1-1: Total Stage Mass

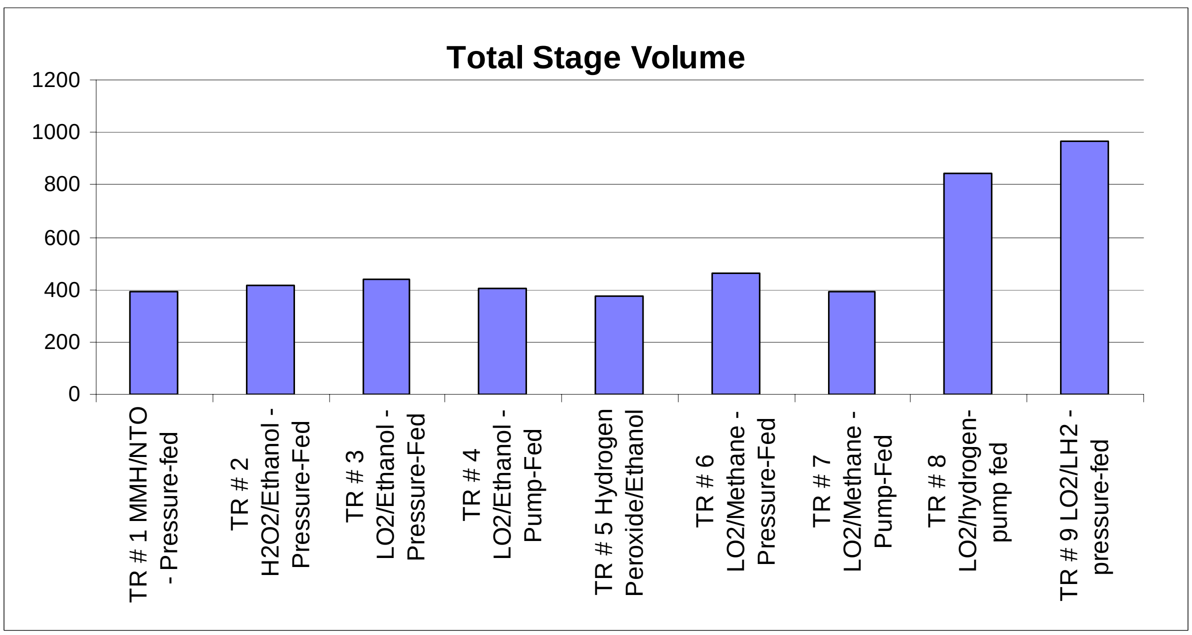


Figure 19.1.1.1-2: Total Stage Volume

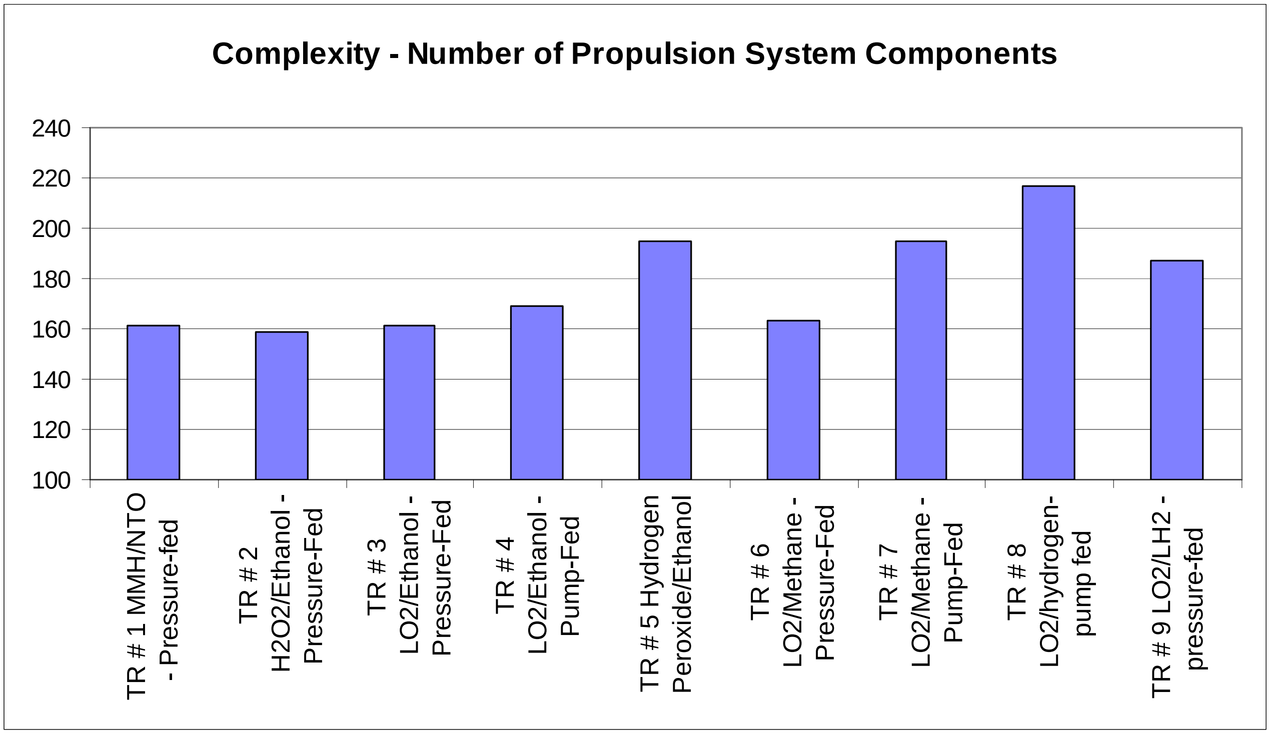


Figure 19.1.1.1-3: Complexity – Number of Components

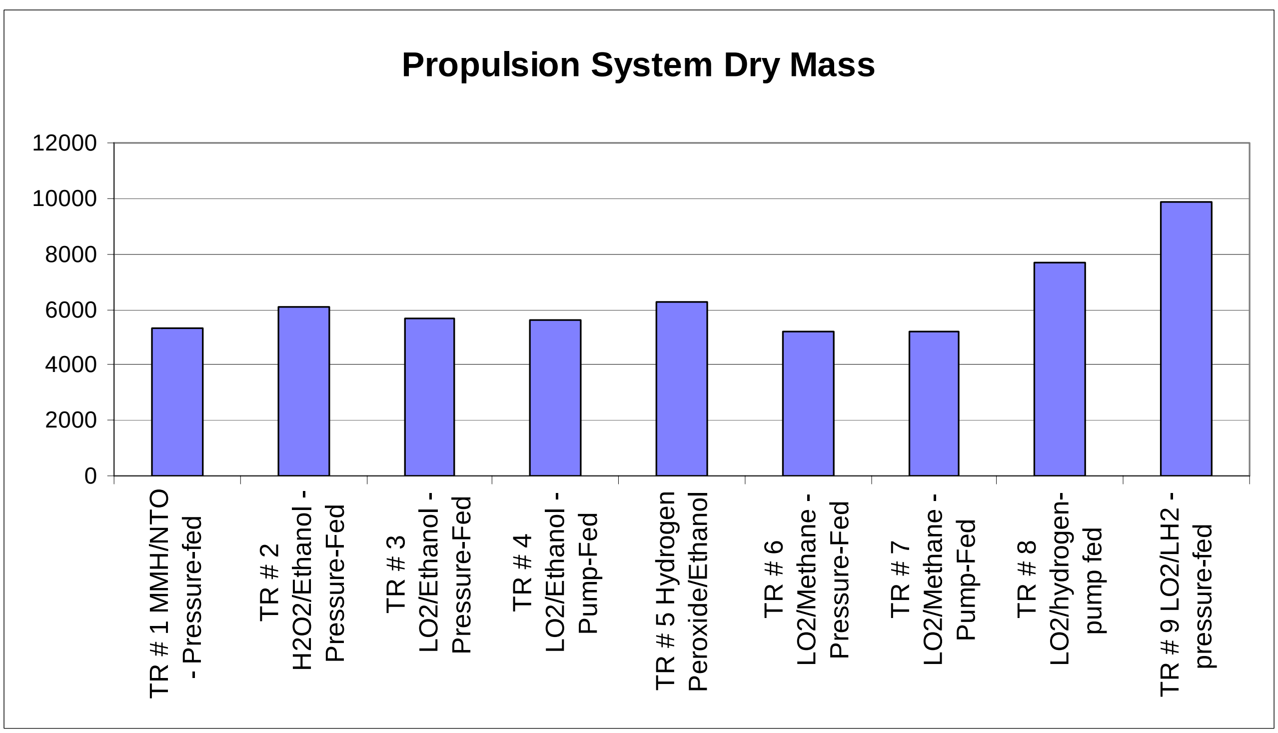


Figure 19.1.1.1-4: Propulsion System Dry Mass

The performance of LO2/methane exceeds that of LO2/LH2 and MMH/NTO. From a volume standpoint it is slight larger, but not significantly. LO2/methane is also capable of being pressure-fed for reliability. The dry mass of LO2/methane is also comparable to MMH/NTO.

LO2/LH2 does not offer any advantages from a mass, complexity, or volume perspective. The size of the vehicle is twice that of other propellants. The dry mass is also 50% higher which will effect cost. The higher dry mass will affect the Lander at touchdown. The reason LO2/LH2 does not perform well is due to the high volume which increases the structural mass (primary, protection, etc). The stage dry mass offsets the I_{sp} benefits in the rocket equation.

19.1.1.2 Service Module / Lander – Rationale for Selection of Pressure-Fed LO2/Methane or LO2/Ethanol

Liquid oxygen based propellants for CEV Orbital Maneuvering System (OMS), RCS, Lander/ascent descent have been identified as good candidates. The fuels best suited are ethanol and methane. This is due to the higher density, clean burning, space-storable characteristics. The I_{sp} advantage of LH2 does not offset the negatives associated with LH2 storage. This and other trades have shown that liquid hydrogen results in a spacecraft that is twice as large, and 33% more complex. Pressure-fed LO2/methane actually performs comparable to the LO2/LH2 pump-fed. The reason for this is the higher dry mass of a LO2/LH2 system caused by the tank and structure mass. The hazards of hydrogen systems are a significant impact to safety of the mission, and worth a separate discussion. Hydrogen is prone to leakage due to its low temperature, small molecule, and difficulty in conducting leak tests. The shuttle Main Propulsion System has shown the difficulty in verifying leak tight systems and finding hydrogen leaks. Furthermore, since LO2/LH2 is generally pump-fed, the RCS gasification and OMS engine gas generators, heat exchangers will have more failure modes. One such failure mode would be leakage between shutdown of a propellant into the RCS gas generators. Restart would be hazardous unless purged well between runs. This will be a major safety concern for RCS which performs a variety of duty cycles. Table 19.1.1.2-1 shows a qualitative comparison of the propellants.

Performance	**MMH/NTO** Pressure-Fed	**H2O2/H-C** Pressure-Fed	**LOX/Alcohol** Pressure-Fed	**LOX/Methane** Pressure-Fed	**LOX/Methane** Pump-Fed	**LOX/LH2** Pump-Fed
Total Mass (Isp)	**SOA**	**-**	**+**	**+**	**+**	**+**
Power Required (Heaters)	**SOA**	**-**	**+**	**+**	**+**	**+**
Volume, (Density Isp)	**SOA**	**+**	**+**	**-**	**+**	**-**
Reliability and Safety						
Number of Components	**SOA**	**+**	**+**	**+**	**-**	**-**
Explosive Residues	**Need Imp**	**+**	**+**	**+**	**+**	**+**
Plume Contamination		**-**	**+**	**+**	**+**	**+**
Non-Corrosive	**Need Imp**	**-**	**+**	**+**	**+**	**+**
Low Leakage	**Need Imp**	**+**	**+**	**+**	**+**	**-**
Fast Response	**SOA**	**+**	**+**	**+**	**-**	**-**
Toxicity	**Need Imp**	**+**	**+**	**+**	**+**	**+**
Flammability	**Need Imp**	**+**	**+**	**+**	**+**	**-**
Cost						
Inert (Dry) Mass	**SOA**	**+**	**+**	**+**	**-**	**-**
Propellant Cost	**Need Imp**	**-**	**+**	**+**	**+**	**-**
Number of Components	**SOA**	**+**	**+**	**+**	**-**	**-**
Operations						
Long Term Storability (Years)	**SOA**	**-**	**-**	**-**	**-**	**-**
Propellant Management	**SOA**	**-**	**+**	**+**	**-**	**-**
Ground Propellant Handling	**Need Imp**	**-**	**+**	**+**	**+**	**+**
Integration w/Power/ECLSS	**Need Imp**	**-**	**+**	**+**	**+**	**+**
Commonality with HEDS Roadmap	**Need Imp**	**-**	**+**	**+**	**+**	**+**
Total +		**9**	**18**	**17**	**13**	**9**

Table 19.1.1.2-1: Qualitative Comparison of Propellants

19.1.1.3 Service Module RCS Feedsystem – Cryogenic vs. Gaseous Feedsystem for RCS

There are basically two options for feeding RCS propellants to the engines; gaseous or sub-cooled liquid.

Gaseous RCS - The selection of cryogenic versus gaseous RCS is a strong function of the duty cycle requirements. The source of heat to gasify the propellants must come either from combustion of propellants or from the spacecraft thermal system. Typical RCS usage is 3 to 10 lbm/hr for a spacecraft of this size. The Apollo Service Module averaged 3 lbm/hr propellant usage during attitude hold. A typical CEV Service Module thermal system would reject approximately 6000 to 8500 watts. This is enough to gasify about 85 lb/hr of oxygen and methane propellants on the average. However, a 100 lbf engine however uses propellants at a peak flowrate of 1200 lbm/hr, which would require 87 kW.

A gaseous system is only sufficient for on-orbit attitude hold. However, for any other maneuvers, such as mid-course corrections or entry or descent RCS, a gaseous system is not sufficient. A gaseous system would not allow the RCS to be used as a back-up to OMS. For example, an entry or descent may use 300 to 1000 lbm in 20 minutes. An accumulator sized to operate in blowdown results in an impractical accumulator size. A compressor would be required to raise the pressure in the accumulator to reduce its size. Then pressure regulators are required to deliver propellants to the engines. For DC-X and X-33, Aerojet worked on a gasification system using for Gox/GH2 thruster. These systems proved to be complex and heavy (900 lbms). It was at this point that Aerojet switched to a stored Gox/GCH4 for the RCS. The primary issues with gasification of OMS propellants for RCS are the large amount of power required to gasify the propellants, the complexity, and the large valve flow capabilities required for gas. Furthermore, the gasification system adds a significant number of criticality 1 failure modes.

Integrated Cryogenic RCS Feedsystem - The high duty cycle requirements are the reason a liquid cryogenic RCS is recommended for an integrated system for a vehicle performing landing or entry. The advantages of a cryogenics RCS feedsystem are 1) the reduction in size of valves and piping, 2) the elimination of criticality 1 failures modes of gasification equipment, 3) the reduction in mass, and 4) the commonality of hardware and technologies to cryogenic tank storage. The disadvantage of course is the need to further develop engine technologies which allow the engine to rapidly start-up from ambient with warm gas and engine injector-to-valve thermal isolation. Other technologies to keep the valves pre-chilled can aid in a fast engine start-up.

Based on the energy that it takes to gasify propellants, it is simpler to insulate and deliver as a liquid. In space, the vacuum is ideal. The key to using cryogenics RCS feedsystem is to highly sub-cool the propellants. A sub-cooled cryogenic RCS feedsystem uses multilayer insulation, flow of propellants caused by thruster usage, and possibly cryocoolers to keep the manifolds conditioned. The properties of LO2 and methane allow it to be transferred and remain liquid even after absorbing much heat. As shown in Figure 19.1.2.3-1, liquid methane that is stored at 275 psia and 163 R, is sub-cooled by 140 deg R. Actually turning liquid methane to a gas requires another 219 Btu/lbm.

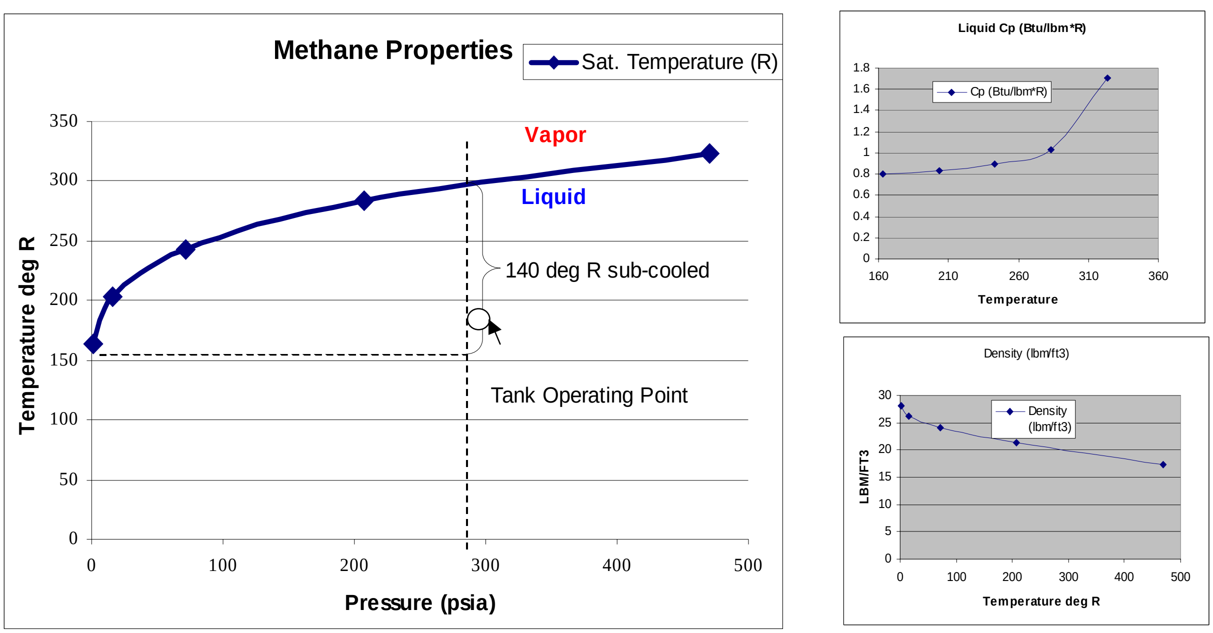


Figure 19.1.1.3-1: Methane Properties

By comparing the thruster propellant usage rate and heat leak, it can be determined if the feedsystem will remain chilled with minimal venting or cryocoolers. The heat leak into the feedsystem for a spacecraft needs to include lines, supports, valves, and engines. A heat leak per linear foot of pipe including supports and valves of much less than 0.5 Btu/hr/ft, assuming 20 layers of MLI, is readily achievable. Heat soak-back from thrusters for a 100 lbf engine of 1 Btu/hr should be achievable based on a thermal isolator analysis. For 24 thrusters and 100 feet of line, this would be approximately 74 Btu/hr. At a usage rate of 1 lbm/hr, subcooled liquid methane can absorb 126 Btu/hr before it turns to a gas by the time it reaches the engines. The Apollo Service Module used on average about 3 lbm/hr of propellant. It is not unreasonable to expect that thruster usage will keep the lines chilled, and that mimimal venting will be required. ***However, even if propellant is budgeted for venting (at 10 lbm/day), a cryogenic feedsystem would still be lighter than a gasification system (900 lbm).***

Cryogenic RCS feedsystem and engines have been under development since Shuttle Upgrades and Next Generation Launch Technologies (NGLT). Breadboard testing of a cryogenic LO2 RCS feedsystem at JSC/Energy Systems Test Area (ESTA) has demonstrated the capability to maintain subcooled propellants in the manifold near the thruster inlets. The design uses a semi-ring manifold to distribute propellants. This semi-ring manifold allows firing of any engine to keep the manifold chilled down. A vent valve at the end of the manifold keeps the line conditioned if necessary. Another significant finding is that the pipe wall is effective in transferring heat. This means that a cryocooler installed at the end of the pipe can be designed to keep the entire pipe chilled as shown in the schematic in Figure 19.1.1.3-2.

CEV Service Module Propulsion Schematic (Lander is Similar*)

Assumptions
1) *Lander has different number of engines and tanks
2) Number of tanks from 1 (common bulkhead) to 6 for packaging as required
3) Fail Op / Fail Safe
4) RCS provides back-up OMS as a fail safe mode
5) RCS uses liquid and is FFC with long-life chamber to allow long burns
6) Tank structural failure not credible due to design for minimum risk, however tank isolation is possible
7) Tank designed for 60 day passive storage with active cooling as DTO
8) Tank include 2 compartments allowing integrated OMS / RCS (vane device in blue, screens in green)
9) Cryogenic Feedsystem - Thruster Usage is able to keep line conditioned. Feedsystem also uses active cooling with thermodynamic venting as back-up
10) Cryocooler uses conduction along feedline to remove heat from feedline
11) Only gas is vented – no liquid

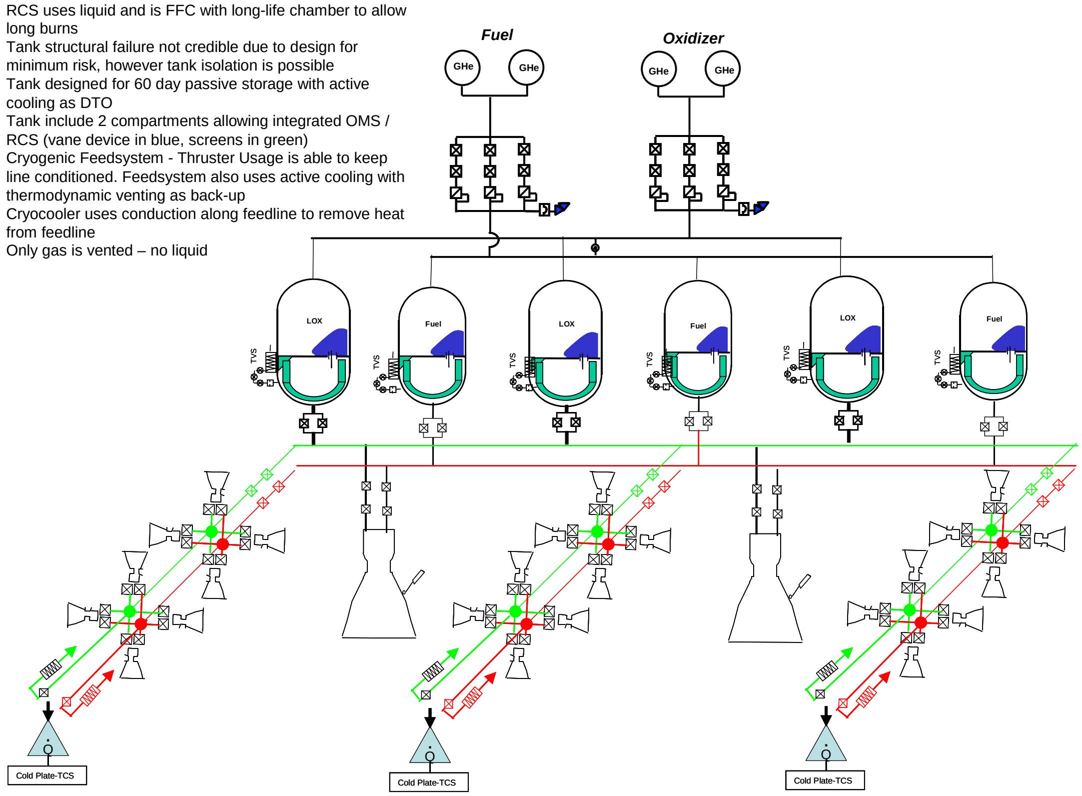


Figure 19.1.1.3-2: CEV SM Propulsion Schematic

In Shuttle Upgrade, the Advance Space Transportation Program (ASTP), and NGLT, the engines recently tested for LO2/ethanol included an 870 lbf LO2/ethanol and a 6000 lbf LO2/ethanol Rocketdyne engine. Many of the technologies for LO2/ethanol are applicable to LO2/methane. This same technology could be applied to a liquid methane thruster. For example, engine hardware for LO2/ethanol may be operated with LO2/methane to obtain design data for LO2/methane engines. For robustness, the ability of an engine to start on gas or liquid during start-up is desired. Engine testing has also shown that at >50 lbf, there is little impact to thrust response as could be caused by two phase LO2 flow, even when the engine starts from ambient temperatures. MSFC's NGLT Aerojet 870 lbf engine has shown the ability of the igniter to start with gaseous propellants. At low flowrates (igniter only), or thrust levels of ~10 lbf, there is significant delay in thrust ramp-up. However, for attitude hold this will not be an issue, since the vehicle control system is looking at rates and simply fires the engine longer.

19.1.2 CEV Entry Vehicle (Capsule) RCS Alternate Propellants

The CEV entry vehicle RCS is used only for entry in order to keep it pristine for this critical phase of the mission The design remains inactive until entry (up to 3 months) after launch.. The thrust required ranges from 50 to 100 lbf. The RCS is sized to provide 10 m/s delta-V. This system should remain un-wetted and pristine for entry. This approach was used in Apollo. There are 12 thrusters which are required for fail op, fail safe redundancy. The entry vehicle is ballistically stable, such that the primary purpose of the RCS is to provide roll control for entry. Since this system is only used for entry, a passive system (minimal power, crew interaction, etc) is desired. Also considered is dual use of the propellant for other functions, such as air flotation bag inflation or for Environmental Control and Life Support Systems (ECLSS).

Several monopropellants and bipropellants were considered as shown in Table 19.1.2-1. The design philosophy is to choose the simplest system that provides an acceptable mass and packaging efficiency.

Propellant	**Cold Gas GN2**	**Warm Gas Tridyne (GN2< GO2, GH2)**	**N2O Nitrous Oxide**	**Hydrazine**	**Hydrogen Peroxide**	**Nitrous oxide, ethanol**	**GO2 Ethanol**	**MMH NTO**	**HAN based**
Isp	70.0	140.0	196.0	234.0	160.0	220.0	285.0	285.0	223.0
Mass Prop	210.0	105.0	75.0	62.8	91.9	66.8	51.6	51.6	65.9
Volume of prop (ft3)	10.1	5.1	1.0	1.0	1.0	1.1	1.3	0.7	0.8
Complexity	mono-prop	mono-prop with catalyst beds	mono-prop with catalyst beds	mono-prop with catalyst beds	mono-prop with catalyst beds	bipropellant 2x complexity	bipropellant 2x complexity	bipropellant	mono-prop with catalyst beds
Prop hazrds	non flammable, non-toxic	non flammable, non-toxic	non flammable, non-toxic	flammable, toxic, detonable	toxic (slight) unstable	non-toxic flammable	non-toxic, flammable	toxic flammable	non-flammable low toxicity
Spacecraft fluid Commonality	Air bag inflation, ECLSS	Air bag inflation	can make O2 and N2 if decomposed	none	none	none	ECLSS, O2 for crew	none	none
Heater Power	none	none	cat bed	cat bed, prop	prop, (protect against freezing)	none	none	propellant	propellant, cat beds

Table 19.1.2-1: Comparison of Capsule RCS Propellants

All options examined have been or are being developed. The X-38 completed development of a 25 lbf cold gas RCS system using GN2. If higher performance than cold gas is required, a warm gas or Tridyne system could meet the requirements. Tridyne is a safe, non-flammable, non-explosive mixture of nitrogen, oxygen, and hydrogen. The O2 and hydrogen are at very low concentrations that can be catalyzed to produce a warm gas. The development of Tridyne was started during the Apollo program by Rocketdyne. Recently a 10 lbf Tridyne thruster was tested at WSTF. This system is essentially the same as a cold gas system, except that Shell 405 (or it replacement) catalyst beds are used at the thruster. The system and catalyst beds do not need to be heated, so this system is entirely passive, except for instrumentation. Three approx. 500 psia regulators provide a redundant supply to the manifolds. Three manifolds with isolation valves are used for redundancy. Each thruster consists of a single valve, thermal isolator, catalyst bed, and nozzle.

Nitrous Oxide has been tested in monopropellant and bipropellant modes. XCOR has demonstrated nitrous oxide/ethane and alcohol thrusters from 15 to 50 lbf. Nitrous Oxide is space storable if kept at a high enough pressure. Another benefit for life support is that nitrous oxide can also be decomposed to O2 and N2 at the right concentrations. Gaseous Oxygen and ethanol thrusters have been tested since 1984. The technology is promising for exploration because of its higher performance potential and commonality with oxygen storage for life support. Gox/ethanol are completely space storable in that no heaters are required. Tests have demonstrated using ethanol down to –120 deg F. Several engines from 25 lbf to 870 lbf have been demonstrated. Hydroxyl Ammonium Nitrate (HAN) based propellants offer close to hydrazine type performance without the toxicity. Hydrogen Peroxide was used on Mercury capsule. In the Mercury Program, hydrogen peroxide was not entirely successful in that thruster failures due to rapid decomposition occurred. It is an unstable oxidizer that is not very tolerant. Hydrazine and MMH/NTO are fallback options for RCS. However for a capsule with recovery operations, there will be safety, cost, and reusability or refurbishment impacts.

19.1.2.1 Capsule RCS Selection Rationale

The propellants were compared on the basis of mass, volume/packaging efficiency, power, number of components, and hazards. The number of components affects cost and reliability. Reliability can also be affected by propellant characteristics, such as stability, corrosiveness, and residues. The three most promising technologies are Tridyne, gaseous oxygen/ethanol RCS, and nitrous oxide for a CEV. These propellants provide a simple, safe, and cost effective RCS propulsion system that can support a 2008 demo of the CEV. The CEV is the cornerstone of Project Constellation, and a successful, on schedule, on cost development will help exploration. Choosing the simplest, safest, and lowest life cycle cost propellants will ensure 2008 demo flight tests. Choosing MMH/NTO for the capsule RCS will add significant development risk and cost.

19.1.3 Conclusions and Recommendations

Capsule - The conclusion of the trade study is to recommend warm gas Tridyne mixture as the first choice for a capsule RCS. This monopropellant provides acceptable mass and volume. It uses no power for storage or catalyst bed pre-heat. The Tridyne system has the fewest number of components. The propellant is stable, non-toxic, non-flammable, and non-explosive. The propellant can also be used to inflate flotation bags. Although the volume is higher, it offers good packaging efficiency because it has only one storage commodity. Using a gas also simplifies acquisition and gauging, and thus eliminates many tank components. The current TRL of a Tridyne Warm Gas Propulsion System is currently 5. All of the components are high TRL items; valves, catalyst beds, tanks, and regulators. The primary work to reach TRL 6 is build a prototype system and test in space environment. The technology readiness level is sufficient to support a 2008 demo flight and 2014 manned flight.

Service Module and Lander - It is recommended that Code T pursue technology for oxygen and methane propulsion for the CEV Service Module or Lander. A pressure-fed LO2/methane saves 2600 lbm compared to MMH/NTO, and offers additional cost, operational, safety, vehicle integration benefits. LO2/LH2 does not offer any advantages from a mass, complexity, or volume perspective. The size of a LO2/LH2 vehicle is twice that of other propellants. The LO2/LH2 system dry mass is also 50% higher which will effect cost. The higher dry mass will effect the Lander at touchdown. The LO2/methane work can be done to support a 2011 demonstration. It is important that time with the technology be spent to gain the most benefits. In addition it is recommended that further work in subcooled cryogenic integrated RCS feedsystem and engines be pursued due the safety, reliability, and performance advantages over gaseous RCS feedsystems. The critical engine technologies are engine thermal injector-to-valve thermal isolator designs and injector designs for cryogenic propellants.

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19.2 Alternate Power Sources

This parametric variation examines the impact of changing the power generation technology for the Crew Exploration Vehicle. The selected technology for the LDRM-2 trade reference mission was a PEM fuel cell system to generate electrical power from launch to Service Module (SM) separation, and a battery system provided power from separation to crew recovery. For comparison, an equivalent system incorporating deployable solar arrays plus batteries is estimated here. CEV masses have been generated for lunar surface stay times of 3-90 days and solar cell efficiencies from 10-35%. Trade results show that for all possible lunar stay times, the solar array system is less massive than the PEM system. For the TRM's 7-day surface mission, a solar array and battery system with state of the art 26% efficient solar cells provides 3,100 kg of CEV mass savings over the PEM fuel cell and battery system. A similar power system trade has not yet been performed for the other elements in the architecture.

<u>PEM Fuel Cell Description</u>

The assumed PEM fuel cell system includes three fuel cell power plants each sized to produce 6 kW of continuous peak power. As was previously discussed in the TRM description section, the CEV's average power is 6 kW during occupied phases of the mission and the vehicle is powered down to 3 kW average power during unoccupied phases. Therefore, any single 6 kW power plant is capable of supplying the entire power needs for the mission. Oxygen reactant for the fuel cell is stored in the SM oxidizer tanks, and supercritical hydrogen reactant is stored in two separate graphite-epoxy overwrapped Inconel-lined tanks at 500 psia. Water produced by the fuel cell is used for crew consumption and water-boiler heat rejection after SM separation. Each fuel cell has a mass of 70 kg.

<u>Solar Array System Description</u>

The assumed solar array system includes two deployable sun-tracking array panels each sized to produce 10.8 kW peak power at end-of-life. As the CEV spends part of its mission in low Earth orbit and solar arrays are not capable of generating power during orbital eclipse periods, the arrays must generate power during orbital daylight periods for both the vehicle's regular power needs and an energy storage system. A worst-case eclipse period of 36 minutes for a 407 km circular orbit coupled with expected array pointing losses results in a 10.8 kW end-of-life power requirement. The array panels stow in the Service Module during launch and deploy after the CEV reaches orbit. For energy storage in LEO, the CEV will use the Lithium-ion batteries included in the Crew Module (CM) that were sized to provide peak power for 3.5 hr after SM separation. With an eclipse period less than 1 hr, the batteries do not need to be resized to handle this requirement. Unlike the PEM fuel cells, the solar array and battery system is not a source for potable water production. Therefore, separate potable water tanks fully loaded at launch must be included here to provide water during the mission.

Five solar cell types are examined in this trade. They are:

1. 10% Air Mass Zero (AM0) efficiency thin-film CuInS2

2. 14% Si
3. 18% GaAs/Ge
4. 26% 3-Junction
5. 35% 4-Junction

Cell efficiencies assumed above represent the energy conversion of a single cell at AM0, 28^{o}C. AM0 describes solar irradiance in deep space where it is unaffected by Earth's atmosphere, and the vehicle sizing includes a number of degradation factors to determine actual array efficiency at the end of the CEV mission. For example, current state of the art cells, the 26% 3-Junction cells, are only 21% efficient at the array level at end-of-life.

19.2.1 Operational, Safety, and Mission Assurance Impacts

Some of the primary benefits of using PEM fuel cells instead of solar arrays for CEV power generation are that they do not require deployment and tracking mechanisms and can generate power regardless of vehicle attitude and solar visibility. A CEV with solar arrays, on the other hand, must deploy its arrays within a few hours after reaching orbit and the arrays must track the Sun throughout the vehicle's useful lifetime. Mechanisms such as these to a vehicle add failure mores, design complexity, and typically decrease the overall probability of mission success.

Other impacts in using solar arrays comes with the three major propulsive maneuvers the CEV is involved in with this scenario – Earth orbit departure, libration point arrival, and libration point departure. Solar arrays and their tracking mechanisms are typically constructed to be as lightweight as possible as they are subjected very low accelerations through their missions. However, these three maneuvers are relatively high acceleration events, and large area panels will impart high bending moments at the attachment points. For these phases of the mission, deployed arrays must be either retracted and then redeployed after the burn is complete, secured for the burn, or designed to be sufficiently strong to handle the accelerations.

One potential safety impact that solar array systems can minimize is that the fuel cell requires more high-pressure fluid tanks in the SM for its supercritical hydrogen, and possibly additional tanks for supercritical oxygen if that reactant cannot be stored in the propellant tanks. Solar arrays are consumable-free technology.

19.2.2 Architecture Sizing Impacts

Figure 19.2.2-1 illustrates how total CEV mass changes with power system technology and lunar surface stay time. A range of stay times from 3 days – 90 days is shown with the corresponding CEV total mission duration, which determines the vehicle's total energy requirement. The selected solar cell technology is 26% 3-Junction cells. For the TRM's 7-day surface mission duration, the CEV with PEM fuel cells has a total mass of 29,638 kg, and the corresponding solar array and battery system has a total mass of 26,539 kg, a decrease of 3,099 kg (10.5%). For a long-stay lunar mission such as a 90-day stay, the solar array and battery system provides a savings of 9,707 kg (26.8%). The mass of the CEV with a solar array system is independent of stay time.

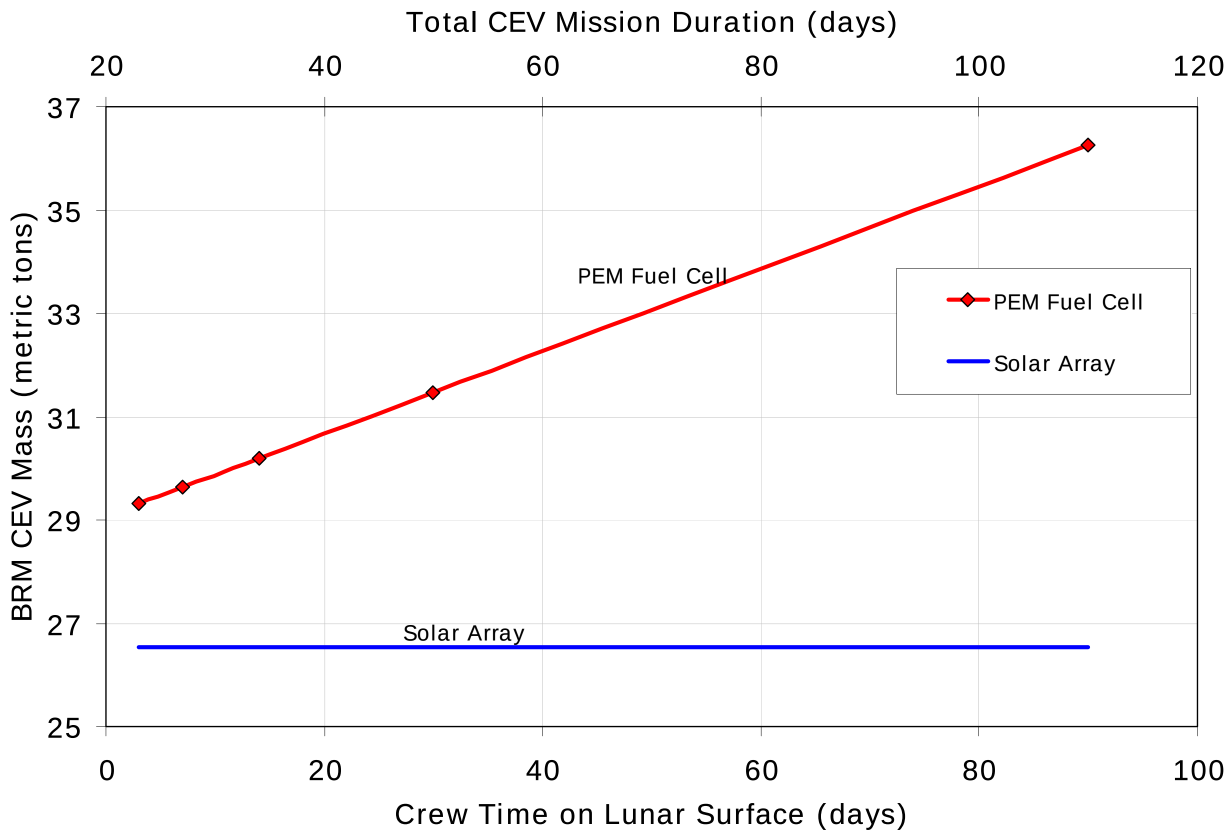

Figure 19.2.2-1: Variation in CEV Mass with Power Technology and Lunar Stay Time

Table 19.2.2-1 provides the raw sizing data for all cases analyzed in this parametric trade, including Service Module internal equipment volume, solar array area per panel (where applicable), and required radiator area. With fuel cells, SM equipment volume includes propellant tanks, pressurant tanks, fuel cell reactant tanks, fuel cell power plants, and power management and distribution equipment. The SM equipment volume for solar arrays includes propellant tanks, pressurant tanks, stowed solar array panels, potable water tanks, and power management and distribution equipment.

Average CEV Occupied Power = 6 kW									
	Surface Stay (d)	Total Mission (d)	CEV CM Mass (kg)	CEV SM Mass (kg)	Total Mass (kg)	CEV Total Energy (W-hr)	SM Equip Vol (m^3)	PV Area per Panel (m^2)	Radiator Area (m^2)
PEM Fuel Cell	3	23	10,084	19,246	29,330	2,723,645	21.0	0	31
	7	27	10,084	19,554	29,638	3,011,645	21.5	0	31
	14	34	10,084	20,101	30,185	3,515,645	22.4	0	31
	30	50	10,084	21,368	31,452	4,667,645	24.6	0	31
	90	110	10,084	26,162	36,246	8,987,645	32.6	0	31
PV + Battery, 10% CuInS2 Cells	90	110	10,059	16,694	26,753	8,987,645	27.1	107	23
PV + Battery, 14% Silicon Cells	90	110	10,059	16,733	26,792	8,987,645	24.0	77	23
PV + Battery, 18% GaAs/Ge Cells	90	110	10,059	16,740	26,799	8,987,645	22.3	60	23
PV + Battery, 26% 3-J Cells	90	110	10,059	16,480	26,539	8,987,645	20.4	41	23
PV + Battery, 35% 4-J Cells	90	110	10,059	16,330	26,389	8,987,645	19.2	31	23

Table 19.2.2-1: Power Source Trade Raw Sizing Data

19.2.3 System Impacts

This section examines how CEV element and subsystem mass changes with solar arrays instead of fuel cells as the primary power generation technology. As will be described, this modification not only affects the power system design, it also has an impact on most other major CEV subsystems, particularly active thermal control and life support.

CEV Crew Module's System Mass Changes				
System	PEM Fuel Cell (7 days)	26% Solar Array & Battery	Mass Change (kg)	% Change
Structure	1873	1873	No Change	0.0
Protection	887	882	(5)	(0.6)
Propulsion	117	117	No Change	0.0
Power	550	550	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	895	848	(47)	(5.3)
Other	949	946	(3)	(0.3)
Growth	1202	1191	(11)	(0.9)
Non-Cargo	967	967	No Change	0.0
Cargo	1784	1784	No Change	0.0
Non-Propellant	60	101	41	68.3
Propellant	64	64	No Change	0.0
Total	10084	10059	(25)	(0.2)

Table 19.2.3-1: Variation in CEV CM Mass with Fuel Cells and Solar Arrays

CEV Service Module's System Mass Changes				
System	**PEM Fuel Cell (7 days)**	**26% Solar Array & Battery**	**Mass Change (kg)**	**% Change**
Structure	1469	1446	(23)	(1.6)
Protection	0	0	No Change	0.0
Propulsion	1734	1544	(190)	(11.0)
Power	673	450	(223)	(33.1)
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	168	58	52.7
Other	100	100	No Change	0.0
Growth	817	742	(75)	(9.2)
Non-Cargo	463	401	(62)	(13.4)
Cargo	0	0	No Change	0.0
Non-Propellant	1454	224	(1230)	(84.6)
Propellant	12733	11404	(1329)	(10.4)
Total	19554	16480	(3074)	(15.7)

Table 19.2.3-2: Variation in CEV SM Mass with Fuel Cells and Solar Arrays

Active Thermal Control System

In addition to generating electricity through the reaction of oxygen and hydrogen, fuel cells have inherent conversion inefficiencies that produce waste heat, and this heat must be collected and rejected to maintain fuel cell operation. The assumed PEM fuel cells of the TRM generate 3 kW of waste heat for every 6 kW of electricity produced. The impact of this on the active thermal control system is that it adds heat exchangers to collect the waste heat from the fuel cell power plants, higher power requirements to pump the heat-carrying fluid, and larger radiator panels to reject the extra heat to space.

Though a solar array system does not produce waste heat like fuel cells, it too may have an adverse impact on the active thermal control system. Depending on the placement on the vehicle, the array panels may interfere with the radiators' ability to reject heat by adding a high temperature, infrared-emitting obstruction to the radiator view to deep space. However, this is highly configuration-dependent and requires a detailed analysis.

Avionics

No significant impacts identified.

Descent and Landing System

No significant impacts identified.

Environmental Control and Life Support System

With a solar array system, the environmental control and life support system no longer has a source for potable water production. Instead, the entire water supply required for the mission must be loaded in dedicated water tanks prior to launch. The fuel cell system can produce water throughout the mission and constantly "top-off" a smaller potable water accumulator tank. Excess water produced by fuel cells may also have ancillary benefits such as radiation protection or water-evaporation heat rejection.

Extra-Vehicular Activity System

No significant impacts identified.

Human Factors and Habitability System

No significant impacts identified.

Propulsion System

The primary impacts on the propulsion system for a solar array system is that it does not require the storage of oxygen reactant in its propellant tanks, allowing for smaller, lighter-weight oxidizer tanks. The CEV with solar arrays and batteries is also lower overall in mass, therefore it requires less propellant to perform the same mission delta-V, which also reduces tank size. Negatively, this system may require the use of the reaction control system more often to maintain the CEV in an attitude capable of pointing arrays at the Sun.

Structures and Thermal Protection Systems

With smaller radiators in a solar array system, the current Service Module length can be reduced slightly. Any decrease in SM dimensions will decrease structural mass.

19.2.4 Impact on Mars Preparations

No significant Mars preparation impacts identified.

19.3 Return Payload Mass

This parametric variation examines the impact of changing the amount of payload returned to Earth from the lunar surface. The payload, possibly lunar material, biological samples, or hardware from surface equipment, is collected on the lunar surface and will not be carried from Earth to the Moon. After collection, the crew will transfer the payload to the Lander Ascent Stage, which carries the mass from the surface to Lunar L1. Once docked to the CEV, the crew transfers the payload from the Ascent Stage module to the CEV Crew Module, and returns with the crew to Earth. The LDRM-2 trade reference mission called for a returned payload mass of 100 kg, and this trade will examine a range of masses from 0 kg to 1,000 kg. No provisions are made for changes in stowed volume of the payload, though. The payload has an assumed constant volume of 1 m^3.

The conclusion of this trade study is that architecture initial mass in LEO (IMLEO) increases by 11 kg for every 1 kg of returned payload mass. The required payload delivery capability of the cargo launch vehicle also increases by 5.5 kg for every 1 kg of payload. While the mass of the payload is not included in the architecture IMLEO, an increase in that mass will dramatically increase the amount of propellant required by the Ascent Stage and CEV to return it to Earth. As these vehicle masses increase, the Descent Stage, Kick Stage, and Earth Departure Stages must likewise carry more propellant by the nature of the rocket equation. This extra propellant mass (and extra propulsion mass to store it) accounts for the 11:1 ratio of architecture mass to returned payload mass.

19.3.1 Operational, Safety, and Mission Assurance Impacts

No significant operational, safety, or mission assurance impacts identified.

19.3.2 Architecture Sizing Impacts

Figure 19.3.2-1 demonstrates how total architecture mass changes with returned payload mass. A range of payload mass from 0 kg – 1,000 kg is shown. By reducing the TRM payload mass from 100 kg to 0 kg, architecture IMLEO decreases by 1,100 kg (0.5%), and increasing payload mass from 100 kg to 1,000 kg increases IMLEO by 9,900 kg (4.3%). Therefore, total architecture mass increases by 11 kg for every extra kilogram of payload mass returned from the Moon.

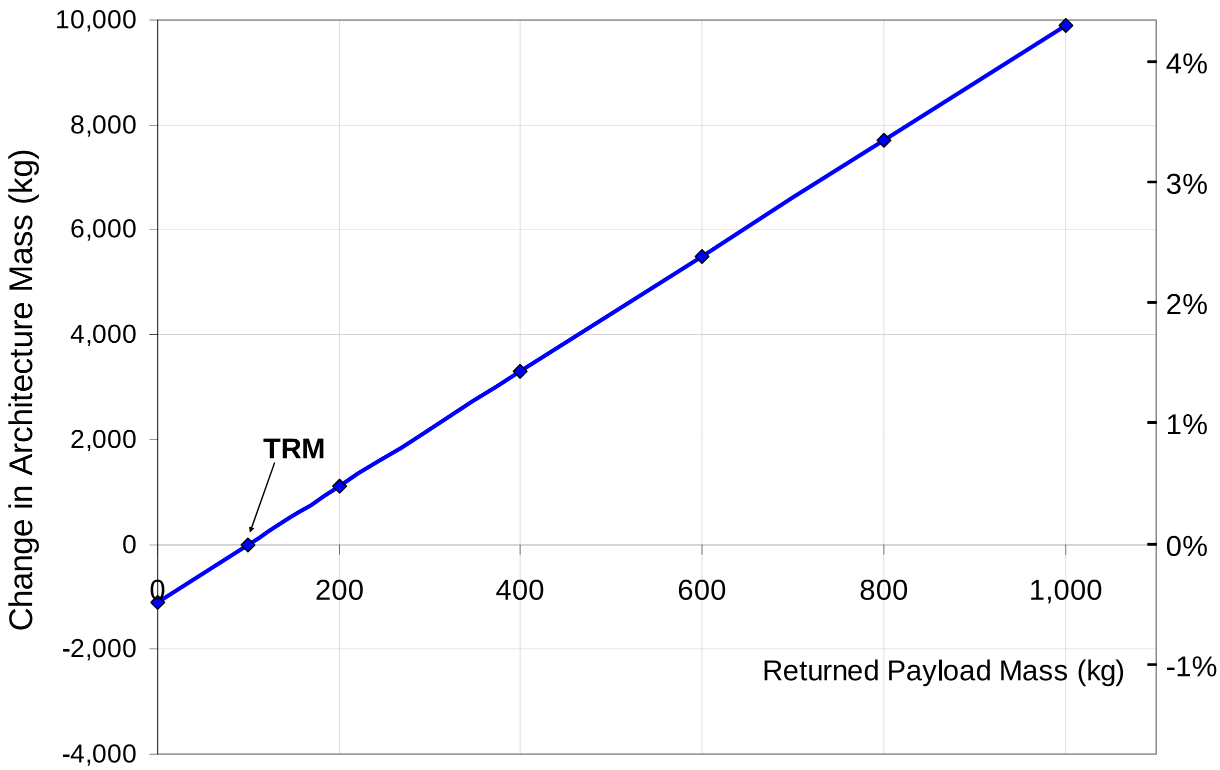

Figure 19.3.2-1: Variation in Total Architecture Mass with Returned Payload Mass

19.3.3 System Impacts

This section examines how TRM element and subsystem mass changes with an increase in returned payload mass from 100 kg to 1,000 kg. No subsystem technologies are modified to accommodate this increase in mass. However, as the returned payload may require conditioning during the return to Earth, the vehicles may require additional accommodations such as more power for a greater payload size. Given the presently undefined nature of the returned payload, though, such accommodations are not modeled in this analysis.

CEV Crew Module's System Mass Changes				
System	**100 kg Payload**	**1,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	822	No Change	0.0
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	691	No Change	0.0
Other	835	835	No Change	0.0
Growth	1041	1041	No Change	0.0
Non-Cargo	966	966	No Change	0.0
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	55	No Change	0.0
Propellant	64	64	No Change	0.0
Total	8812	8812	No Change	0.0

Table 19.3.3-1: Variation in CEV CM Mass with Returned Payload Mass

CEV Service Module's System Mass Changes				
System	**100 kg Payload**	**1,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	1455	1458	3	0.2
Protection	0	0	No Change	0.0
Propulsion	1408	1432	24	1.7
Power	661	661	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	752	5	0.7
Non-Cargo	305	314	9	3.0
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1442	No Change	0.0
Propellant	11332	11688	356	3.1
Total	17560	17957	397	2.3

Table 19.3.3-2: Variation in CEV SM Mass with Returned Payload Mass

Lander's Ascent Stage's System Mass Changes				
System	**100 kg Payload**	**1,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	839	850	11	1.3
Protection	73	73	No Change	0.0
Propulsion	1631	1711	80	4.9
Power	813	813	No Change	0.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	851	No Change	0.0
Other	455	455	No Change	0.0
Growth	1080	1098	18	1.7
Non-Cargo	1483	1515	32	2.2
Cargo	227	227	No Change	0.0
Non-Propellant	1014	1014	No Change	0.0
Propellant	10703	11901	1198	11.2
Total	19906	21245	1339	6.7

Table 19.3.3-3: Variation in Lander Ascent Stage Mass with Returned Payload Mass

Lander's Descent Stage's System Mass Changes				
System	**100 kg Payload**	**1,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	553	563	10	1.8
Protection	50	50	No Change	0.0
Propulsion	1413	1483	70	5.0
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	745	37	5.2
Growth	678	702	24	3.5
Non-Cargo	464	492	28	6.0
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	18634	1061	6.0
Total	22608	23836	1228	5.4

Table 19.3.3-4: Variation in Lander Descent Stage Mass with Returned Payload Mass

Propulsive Stages' System Mass Changes

Element	System	100 kg Payload	1,000 kg Payload	Mass Change (kg)	% Change
Kick Stage	Structure	621	645	24	3.9
	Protection	0	0	No Change	0.0
	Propulsion	1530	1576	46	3.0
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	545	14	2.6
	Non-Cargo	953	998	45	4.7
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	24674	1351	5.8
	Total	27465	28944	1479	5.4
EDS1	Structure	1972	2048	76	3.9
	Protection	0	0	No Change	0.0
	Propulsion	4361	4521	160	3.7
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1499	47	3.2
	Non-Cargo	3109	3263	154	5.0
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	86807	4518	5.5
	Total	94109	99064	4955	5.3

Propulsive Stages' System Mass Changes					
Element	**System**	**100 kg Payload**	**1,000 kg Payload**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	941	9	1.0
	Protection	0	0	No Change	0.0
	Propulsion	2318	2331	37	0.6
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	838	4	0.5
	Non-Cargo	1355	1371	16	1.2
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	33339	442	1.3
	Total	39256	39741	485	1.5

Table 19.3.3-5: Variation in Propulsive Stage Mass with Returned Payload Mass

19.3.4 Impact on Mars Preparations

No significant Mars preparation impacts identified.

19.4 Landed Payload Mass

This parametric variation examines the impact of changing the amount of payload landed on the lunar surface. The payload, possibly science equipment, system spares, or logistics resupply, launches from Earth and is left on the lunar surface. The payload mass is assumed to launch with the Lander Descent Stage. The LDRM-2 trade reference mission called for a landed payload mass of 500 kg, and this trade will examine a range of masses from 0 kg to 3,000 kg. No provisions are made for changes in stowed volume of the payload, though. The payload has an assumed constant volume for the range of masses examined.

The conclusion of this trade study is that architecture initial mass in LEO (IMLEO) increases by ~7 kg for every 1 kg of landed payload mass. The required payload launch capability of the cargo launch vehicle also increases by 3.7 kg for every 1 kg of payload. In this variation, the mass of the payload is included in the architecture IMLEO as it launches with the Lander. Therefore, for every extra kilogram of payload, the remaining 6 kg of increased architecture mass is propellant for the Descent Stage, Kick Stage, and Earth Departure Stage and extra propulsion mass to store the propellant.

19.4.1 Operational, Safety, and Mission Assurance Impacts

No significant operational, safety, or mission assurance impacts identified.

19.4.2 Architecture Sizing Impacts

Figure 19.4.2-1 demonstrates how total architecture mass changes with landed payload mass. A range of payload mass from 0 kg – 3,000 kg is shown. By reducing the TRM payload mass from 500 kg to 0 kg, architecture IMLEO decreases by 3,400 kg (1.5%), and increasing payload mass from 500 kg to 3,000 kg increases IMLEO by 16,800 kg (7.3%). Therefore, total architecture mass increases by 6.7 kg for every extra kilogram of payload mass landed on the Moon.

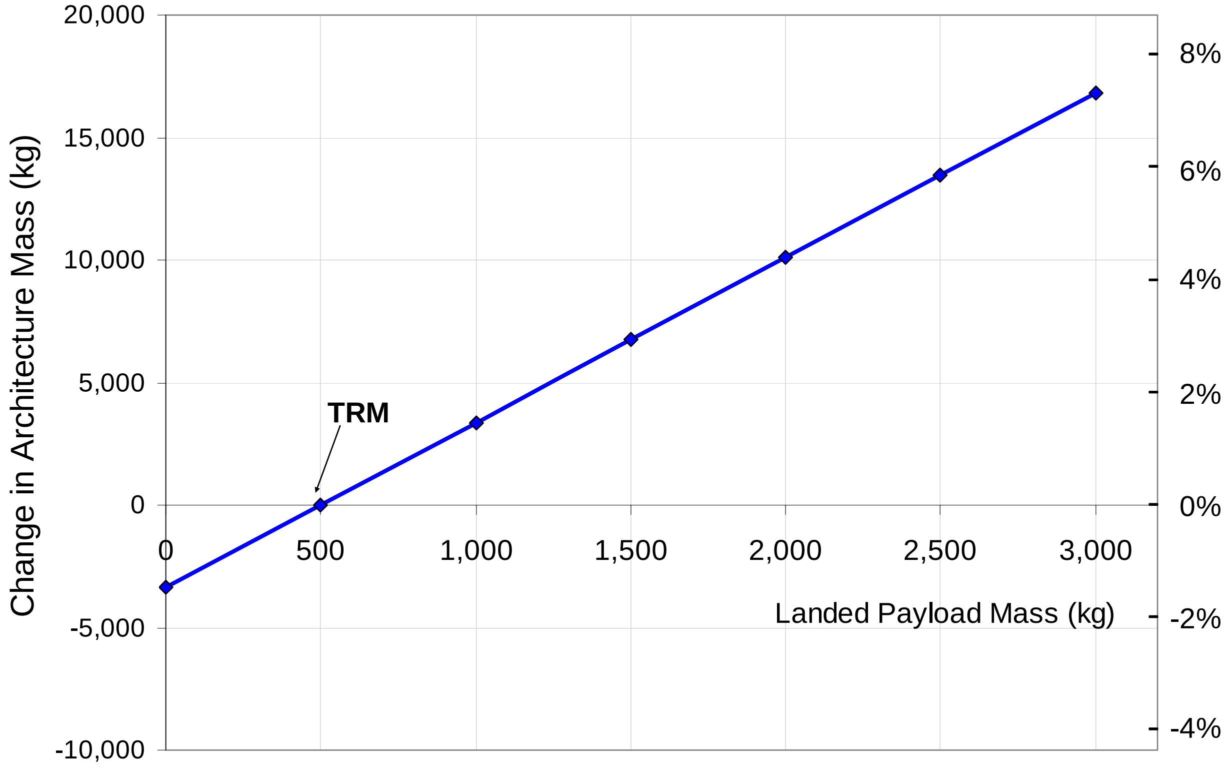

Figure 19.4.2-1: Variation in Total Architecture Mass with Landed Payload Mass

19.4.3 System Impacts

This section examines how TRM element and subsystem mass changes with an increase in landed payload mass from 500 kg to 3,000 kg. No subsystem technologies are modified to accommodate this increase in mass.

CEV Crew Module's System Mass Changes				
System	**500 kg Payload**	**3,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	822	No Change	0.0
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	691	No Change	0.0
Other	835	835	No Change	0.0
Growth	1041	1041	No Change	0.0
Non-Cargo	966	966	No Change	0.0
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	55	No Change	0.0
Propellant	64	64	No Change	0.0
Total	8812	8812	No Change	0.0

Table 19.4.3-1: Variation in CEV CM Mass with Landed Payload Mass

CEV Service Module's System Mass Changes				
System	**500 kg Payload**	**3,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	1455	1455	No Change	0.0
Protection	0	0	No Change	0.0
Propulsion	1408	1408	No Change	0.0
Power	661	661	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	747	No Change	0.0
Non-Cargo	305	305	No Change	0.0
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1442	No Change	0.0
Propellant	11332	11332	No Change	0.0
Total	17560	17560	No Change	0.0

Table 19.4.3-2: Variation in CEV SM Mass with Landed Payload Mass

Lander's Ascent Stage's System Mass Changes				
System	**500 kg Payload**	**3,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	839	839	No Change	0.0
Protection	73	73	No Change	0.0
Propulsion	1631	1631	No Change	0.0
Power	813	813	No Change	0.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	851	No Change	0.0
Other	455	455	No Change	0.0
Growth	1080	1080	No Change	0.0
Non-Cargo	1483	1483	No Change	0.0
Cargo	227	227	No Change	0.0
Non-Propellant	1014	1014	No Change	0.0
Propellant	10703	10703	No Change	0.0
Total	19906	19906	No Change	0.0

Table 19.4.3-3: Variation in Lander Ascent Stage Mass with Landed Payload Mass

Lander's Descent Stage's System Mass Changes				
System	**500 kg Payload**	**3,000 kg Payload**	**Mass Change (kg)**	**% Change**
Structure	553	571	18	3.3
Protection	50	50	No Change	0.0
Propulsion	1413	1544	131	9.3
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	777	69	9.7
Growth	678	722	44	6.5
Non-Cargo	464	516	52	11.2
Cargo	500	3000	2500	500.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	19554	1981	11.3
Total	22608	27401	4793	21.2

Table 19.4.3-4: Variation in Lander Descent Stage Mass with Landed Payload Mass

Propulsive Stages' System Mass Changes

Element	System	500 kg Payload	3,000 kg Payload	Mass Change (kg)	% Change
Kick Stage	Structure	621	667	46	7.4
	Protection	0	0	No Change	0.0
	Propulsion	1530	1615	85	5.5
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	557	26	4.9
	Non-Cargo	953	1038	85	8.9
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	25847	2524	10.8
	Total	27465	30229	2764	10.1
EDS1	Structure	1972	2114	142	7.2
	Protection	0	0	No Change	0.0
	Propulsion	4361	4661	300	6.9
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1540	88	6.1
	Non-Cargo	3109	3396	287	9.2
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	90729	8440	10.3
	Total	94109	103365	9256	9.8

Propulsive Stages' System Mass Changes					
Element	**System**	**500 kg Payload**	**3,000 kg Payload**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	932	No Change	0.0
	Protection	0	0	No Change	0.0
	Propulsion	2318	2318	No Change	0.0
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	834	No Change	0.0
	Non-Cargo	1355	1355	No Change	0.0
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	32897	No Change	0.0
	Total	39256	39256	No Change	0.0

Table 19.4.3-5: Variation in Propulsive Stage Mass with Landed Payload Mass

19.4.4 Impact on Mars Preparations

Decreasing the amount of payload mass landed on the lunar surface may have an adverse effect on preparations for human expeditions to Mars. With less payload delivery capability, fewer and/or less complex systems for testing Mars-related technologies can be landed in a single mission, which means that additional flights to the Moon may be required. This may delay the date at which Mars preparation testing on the Moon is complete, and could delay the date for the first Mars mission. In addition, if the Lander Descent Stage supplies consumables to emplaced surface assets such as habitats, a lesser payload delivery capability may mean that the crew will be able to operate for a shorter time on the surface, which may also adversely affect testing goals.

19.5 All vs. Partial Crew to the Surface

This trade study examined the impact of sending three crewmembers to the lunar surface while leaving one crew member on board the CEV in orbit at Earth-Moon L1.

This change from the TRM approach results in a reduction of 6,222 kg (2.7%) in total architecture mass. These reductions are the result of reduced consumables and reduced quantities of propellants to deliver those consumables to the lunar surface (and ascend them in the case of an aborted surface mission). The reduction of the size of the lunar surface crew saved mass in the Lander's Ascent and Descent stages, the Kick Stage, and EDS1. Although these mass savings resulted in a net mass savings in the total architectural mass, there were slight increases in the mass of EDS2, and the CEV's crew module and service module in order to accommodate the fourth crewmember's loiter at L1 during the course of the lunar surface mission.

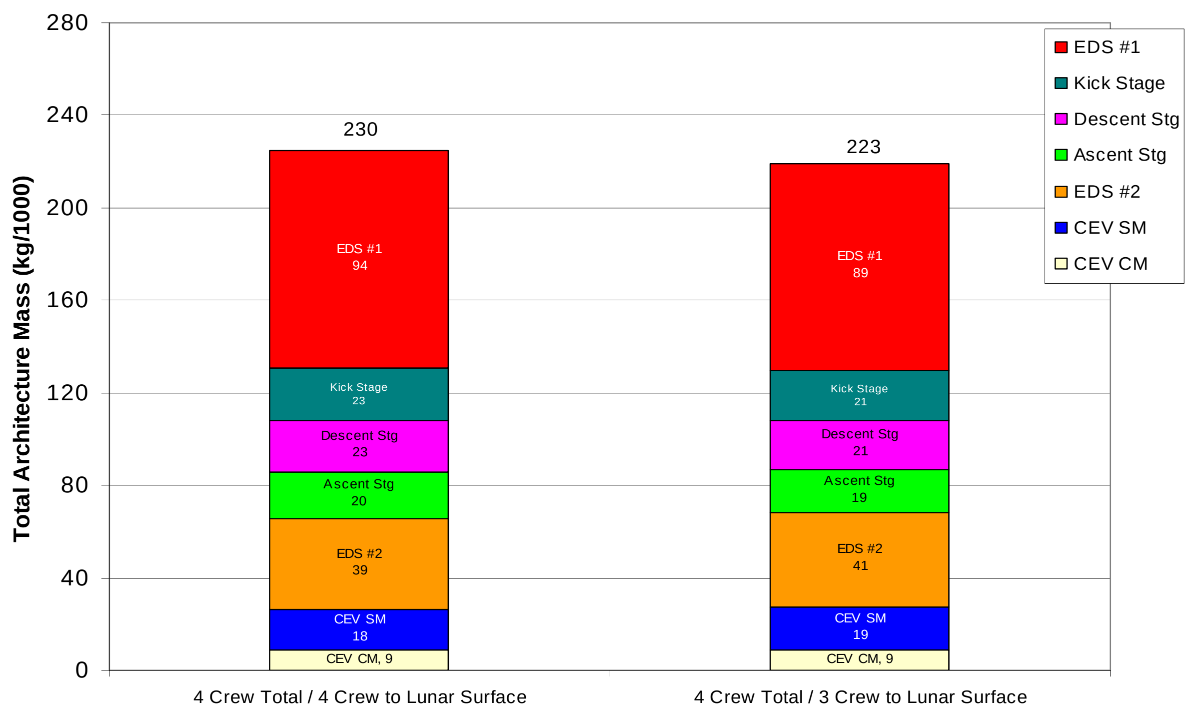

Figure 19.5-1: Effects of Sending Partial Crew to the Surface

This variation was designed to deliver three crewmembers to the lunar surface. The resulting main operational differences between this variation and the TRM would be the way EVAs are approached and the role of the fourth crewmember. It is likely that one of two EVA approaches would be adopted: a three-member EVA approach in which the three crewmembers work as a group on the lunar surface or a two-member EVA approach in which the IVA role rotates among the three crewmembers throughout the surface mission.

The fourth crewmember that remains on the CEV could provide an extra level of redundancy during certain operations and possibly perform certain corrective actions during off-nominal situations. For example, during docking operations, it would be possible to have the fourth crewmember manually override the automated docking sequence from within the CEV, if problems developed. The fourth crewmember could also visually inspect the Lunar Lander after separation and prior to docking. Finally, it is possible that this person could participate in the surface activities remotely, if the appropriate systems were on board the CEV to interact with the surface crew and other assets.

19.5.1 Operational, Safety, and Mission Assurance Impacts

At the current level of detail for which these missions are defined, there are only two major operational differences between this variation and the TRM: surface EVA approach and the role/tasks of the fourth crewmember that remains at L1.

EVA time will be at a premium due to the short surface duration of these missions (7 days). If the assumption is made that the crew will begin performing EVAs during their first day on the lunar surface, but dedicate the final day towards preparations for Earth-return, there will be a total of six full days on the lunar surface for EVAs. The actual number of hours of EVA for a given mission will depend on the approach adopted by the program for conducting EVAs. If the assumption is made that a three-member EVA approach will be adopted and each EVA will last approximately 8 hours, it is likely that the EVA team will be able to perform between 120 crew-hours and 144 crew-hours of EVA, depending on whether the crew takes a day of rest during one of those six days. If the assumption is made that a two-member EVA approach will be adopted and each EVA lasts approximately 8 hours, it is likely that the EVA team will be able to perform ~96 crew-hours of EVA, by using a rotating schedule for IVA responsibilities. Leaving one crewmember IVA (without a “buddy”) raises certain safety considerations that will require programmatic policy guidance.

At this point in time, there has been no requirement indicating the need to keep a crewmember in the CEV during the course of the lunar mission. In fact, it may be prudent to strive to design the missions and the spacecrafts such that they avoid this, since the benefits of this approach are not clear. However, if it is decided that leaving a spacecraft uncrewed is too unpalatable, there are certainly some useful functions that this person could perform. For example, the fourth crewmember could possibly visually inspect the Lander after the lunar surface crew has separated from the CEV. A visual inspection could also be performed prior to the docking of the Lander to the CEV after the lunar surface mission. It is hard to outline exactly what inspections could be performed at this point in time, since the hardware designs have not been defined, but it is possible to imagine that such a capability could be useful. The fourth crewmember could also provide an active capability within the CEV to “wave-off” and perhaps maneuver out of the way during a rendezvous in which the Lander is not operating properly. In this off-nominal situation, the Lander could become the passive vehicle while the CEV becomes the active chaser, maneuvered by the fourth crewmember.

In the case of leaving a crewmember in the CEV at L1 it would be prudent to avoid long idle periods during the course of the surface mission. One could envision that the fourth crewmember might be able to still play a role in the surface mission in a number of ways. For example, if the CEV had the capability to communicate with the surface, the fourth crewmember could interact with the surface crew and possibly manipulate robotic systems on the surface, similar to the role that an IVA surface crewmember would serve. The fourth crewmember could also perform checks and inspections of the CEV throughout the surface mission, ensuring that the CEV is operating properly. During off-nominal situations inside the CEV, it is possible that the fourth crewmember could provide the ability to solve problems. However, since this would occur during off-nominal situations, rather than as part of normal operations, it should not be relied on as a source of tasks.

19.5.2 Architecture Sizing Impacts

Sending partial crew to the surface (3 to the surface and keeping 1 at L1 in this example) had mixed results for the elements. The CEV's Crew Module, Service Module, and EDS (EDS2), all grew in mass due to the increased consumables required for the crewmember that loiters at L1 in the CEV. Conversely, the Lander stages, the Kick Stage, and EDS1 all decreased in mass due to the decreased crew size and consumables that had to be descended to and ascended from the lunar surface. The net architectural mass savings was 6,222 kg (2.7%). The mass of the largest launch decreased by 4,797 kg (5.1%).

Element	TRM (kg)	Partial Crew (kg)	Mass Change (kg)	Percentage Change
CEV Crew Module	8,812	8,874	62	0.7
CEV Service Module	17,560	18,617	1,057	6.0
Ascent Stage	19,906	18,611	(1,295)	(6.5)
Descent Stage	22,608	21,419	(1,189)	(5.3)
Kick Stage	27,465	26,033	(1,432)	(5.2)
EDS1	94,109	89,312	(4,797)	(5.1)
EDS2	39,256	40,628	1,372	3.5
Largest Launch	**94,109**	**89,312**	**(4,797)**	**(5.1)**
Total Architecture	**229,716**	**223,494**	**(6,222)**	**(2.7)**

Table 19.5.2-1: Effects of Sending Partial Crew to the Surface

19.5.3 System Impacts

The major conclusion after examining each system was that, with the possible exception of the EVA system, none of the technologies would change. Only the system masses and quantity of consumables would vary relative to the TRM.

CEV Crew Module's System Mass Changes				
System	**TRM (kg)**	**Partial Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	827	5	0.6
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	700	9	1.3
Other	835	837	2	0.2
Growth	1041	1045	4	0.4
Non-Cargo	966	1006	10	1.0
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	58	3	5.5
Propellant	64	64	No Change	0.0
Total	8812	8874	62	0.7

Table 19.5.3-1: CEV Crew Module's System Mass Changes, Resulting from Sending Partial Crew to the Surface

CEV Service Module's System Mass Changes				
System	**TRM (kg)**	***Partial Crew (kg)***	**Mass Change (kg)**	**% Change**
Structure	1455	1462	7	0.5
Protection	0	0	No Change	0.0
Propulsion	1408	1461	53	3.8
Power	661	738	77	11.6
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	774	27	3.6
Non-Cargo	305	319	14	4.6
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1841	399	27.7
Propellant	11332	11812	480	4.2
Total	17560	18617	1057	6.0

Table 19.5.3-2: CEV Service Module's System Mass Changes, Resulting from Sending Partial Crew to the Surface

Lander's Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**Partial Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	741	(98)	(11.7)
Protection	73	73	No Change	0.0
Propulsion	1631	1581	(50)	(3.1)
Power	813	800	(13)	(1.6)
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	790	(61)	(7.2)
Other	455	455	No Change	0.0
Growth	1080	1036	(44)	(4.1)
Non-Cargo	1483	1198	(285)	(19.2)
Cargo	227	227	No Change	0.0
Non-Propellant	1014	962	(52)	(5.1)
Propellant	10703	10010	(693)	(6.5)
Total	19906	18611	(1295)	(6.5)

Table 19.5.3-3: Lander Ascent Stage's System Mass Changes, Resulting from Sending Partial Crew to the Surface

Lander's Descent Stage's System Mass Changes				
System	**TRM (kg)**	**Partial Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	553	545	(8)	(1.4)
Protection	50	50	No Change	0.0
Propulsion	1413	1345	(68)	(4.8)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	673	(35)	(4.9)
Growth	678	656	(22)	(3.2)
Non-Cargo	464	437	(27)	(5.8)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	16546	(1027)	(5.8)
Total	22608	21419	(1189)	(5.3)

Table 19.5.3-4: Lander Descent Stage's System Mass Changes, Resulting from Sending Partial Crew to the Surface

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**Partial Crew (kg)**	**Mass Change (kg)**	**% Change**
Kick Stage	Structure	621	598	(23)	(3.7)
	Protection	0	0	No Change	0.0
	Propulsion	1530	1486	(44)	(2.9)
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	518	(13)	(2.4)
	Non-Cargo	953	910	(43)	(4.5)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	22015	(1308)	(5.6)
	Total	27465	26033	(1432)	(5.2)
EDS1	Structure	1972	1897	(75)	(3.8)
	Protection	0	0	No Change	0.0
	Propulsion	4361	4207	(154)	(3.5)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1406	(46)	(3.2)
	Non-Cargo	3109	2960	(149)	(4.8)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	77917	(4372)	(5.3)
	Total	94109	89312	(4797)	(5.1)

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**Partial Crew (kg)**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	960	28	3.0
	Protection	0	0	No Change	0.0
	Propulsion	2318	2355	37	1.6
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	847	13	1.6
	Non-Cargo	1355	1400	45	3.3
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	34146	1249	3.8
	Total	39256	40628	1372	3.5

Table 19.5.3-5: Propulsive Stages' System Mass Changes, Resulting from Sending Partial Crew to the Surface

Active Thermal Control System

Sending only partial crew to the surface would require the CEV systems to perform at nearly full capacity for the entire mission duration (at least at the current level of system understanding), as opposed to powering down to a dormant state in the TRM concept. Therefore, the estimated 9 kW heat load from the CEV's systems would need to be rejected throughout the lunar mission. Since the metabolic heat generated by a crewmember is a small fraction of the total heat load (120 W metabolic heat load per crewmember vs. ~9 kW system heat load), few savings would be achieved in the Lander's ATCS by reducing the surface crew from four to three.

Avionics System

Sending only partial crew to the surface would have no effect on the avionics systems.

Descent and Landing Systems

Sending only partial crew to the surface would have very little effect on the descent and landing systems of the CEV. The CEV's mass only increased by 57 kg (excluding the extra TPS required for CEV entry), causing an increase of ~5 kg of required extra TPS. In turn, this slightly increased the mass of the parafoil that was chosen. All CEV descent and landing system technology selections remained constant.

The Lander's descent and landing systems were scaled down due to the decreased mass of the crew and consumables. A mass savings of nearly 1.1 metric tons occurred between the propulsion system hardware and propellants. All Lander descent and landing system technology selections remained constant.

Environmental Control and Life Support System

Sending only partial crew to the surface requires that the CEV remain active during the period of surface operations. Therefore, the CEV will need to continue to provide life support (maintain pressures, provide breathable air, provide temperature and humidity control, remove trace contaminants, provide for waste management, etc.) for the resident crew. This impact was calculated to be only on the order of 10 kg. All CEV ECLSS technology selections remained constant. The Lander's life support consumables decreased by approximately 62 kg. All Lander ECLSS technology selections remained constant.

Extra-Vehicular Activity System

Sending only partial crew to the surface reduced the EVA system mass in the Lander by approximately 90 kg compared to the TRM due to the fact that one less suit and PLSS was required. This was the only change in the EVA system at this point in time. However, during future iterations, it may be useful to examine concepts in which the airlock is removed from the Lander in order to understand the mass benefits that this decision might provide.

Human Factors and Habitability System

Sending only partial crew to the surface will have little effect on the habitation system. The overall mass and volume of the system as a whole should not change significantly, but the allocation of the mass/volume between the CEV and Lander will change. All consumables that are linked directly to crew size and mission duration (e.g. food, food prep supplies, waste collection supplies, hygiene supplies, clothing, housekeeping supplies, operational supplies, medical consumables) will need to be reallocated for only 3 crew for the Lander for 12 days (7 days on the surface and 5 days transit to and from the surface) and an additional 1 crew in the CEV for 12 days. For crew provisions that are linked only to crew size and that are provided on both the CEV and the Lander (e.g. personal hygiene kits, recreational equipment, sleep accommodations, seats), reductions in mass and volume can be made on the Lander to eliminate one crewmember's provisions. In addition to consumables, this situation will require that all components of the habitation system in the CEV be independent from the components of the habitation system in the Lander. For example, it cannot be assumed that the crew can move the food warmer from the CEV to the Lander for their surface stay. However, the current baseline sizing already assumes completely independent systems for the Lander and CEV so this is not a change, but just needs to be considered as a requirement rather than an option in this scenario.

Power System

Sending partial crew to the surface will somewhat increase the amount of consumables carried in the Service Module to support the greater power requirements caused by the necessity of providing life support for a crewmember in the CEV during the surface phase of the mission.

Propulsion System

Sending partial crew to the surface had a modest impact on the Service Module and Lander's stages. The Service Module's propulsion system hardware increased by approximately 53 kg, primarily attributable to the increased tank size that was needed for the increased quantity of propellants. Likewise, the Descent and Ascent stages' propulsion system hardware masses were increased by approximately 68 kg and 50 kg, respectively. The technology selections remained constant between the TRM and this scenario.

Structures and Thermal Protection Systems

Sending partial crew to the surface had a moderate impact on the structures and thermal protection systems in this scenario. This study assumed that the Lander's pressurized volume decreased from 32 m3 to 24 m3 due to the requirement to support 3 crewmembers instead of 4 crewmembers. This decreased the primary structural mass by approximately 98 kg.

19.5.4 Impact on Mars Preparations

Operations, Safety, and Mission Assurance

This variation on the lunar mission does not seem to provide any major advantages in preparing for a human landing on Mars when compared to the TRM. It is possible that a similar "partial crew to the surface" could be adopted for a mission to Mars, but has not been fully explored. The role that the fourth crewmember would fulfill in the lunar mission may have parallels to the role that the crew (or part of the crew) would fulfill in a mission to Mars, if the Mars program adopted a strategy in which the surface mission were conducted from Mars orbit through the use of surface robots controlled by telepresence. Variations of such a mission to Mars includes placing the crewed vehicle in areosynchronous orbit, so that the vehicle is always over the same hemisphere and has continuous line-of-sight, real-time communication with several rovers, or placing the vehicle in the orbit of Deimos (above areosynchronous), which would allow up to 40 hours of line-of-sight communication with a single rover. If this type of Mars mission were ever seriously considered, a crewmember could test the operation of robotic equipment during a lunar mission from lunar orbit or L1. One reference for the above mission scenario is entitled, "An Opposition Class Piloted Mission to Mars Using Telerobotics for Landing Site Reconnaissance and Exploration," Philip J. Burley, Steven E. Fredrickson, Ph.D., Darby F. Magruder, and John D. Rask, October, 2000.

Active Thermal Control System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

Avionics System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars except if there are parallels in space-to-surface robotic control technologies and processes in the two architectures.

Descent and Landing System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

Environmental Control and Life Support System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

Extra-Vehicular Activity System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

Human Factors and Habitability System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

Power System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

Propulsion System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

Structures and Thermal Control System

There would be no significant effect of sending partial crew to the surface on the development of systems for a mission to Mars, as compared to the TRM.

19.6 Crew Size of 2

This trade study examined the impact of reducing the crew size from the TRM approach of four crewmembers to two crewmembers.

This change from the TRM approach results in a reduction of 24,031 kg (10.5%) in total architecture mass. These reductions are the result of reduced consumables required by the crew and systems, the scaling of systems down in size, and a reduction in required habitable volume. Additionally, reducing the crew size to two crewmembers may negate some of the main issues that made the inclusion of the Lander's airlock attractive in the TRM. If the airlock were to be removed, the total architectural mass could be reduced further by an additional 6,018 kg (2.6%), resulting in a total architecture mass savings of 30,049 kg (13.1%).

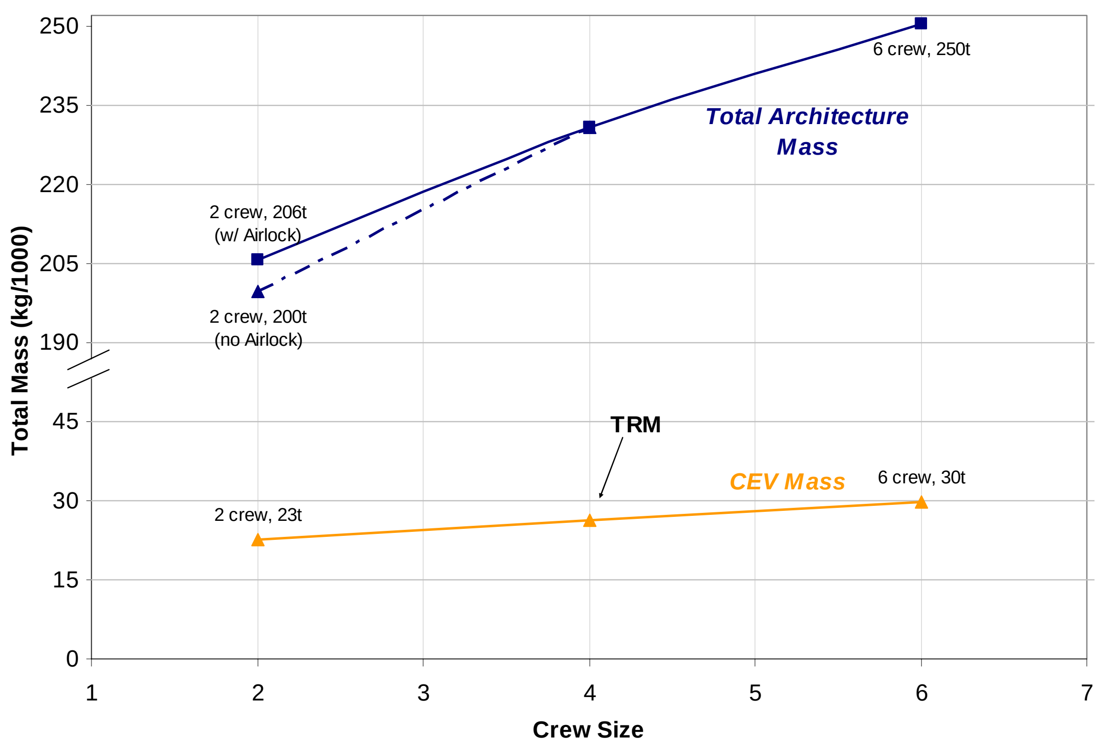

Figure 19.6-1: Effects of Varying the Crew Size on Total Architecture Mass

The reduction in crew size may result in the need to decrease the scope of surface mission objectives. The final level to which the mission objectives are decreased in scope would be a trade between higher crew workloads (greater chance of fatigue) and a decrease in the amount of planned exploration/Mars preparation tasks. Reducing the crew size to two crewmembers would also result in losing redundancy in critical skills and crew-related functions.

This variation deviates from the likely crew sizes that would be used during a mission to Mars. It may prevent the opportunity of training and performing operations with larger crews that would be more representative of those for missions to Mars.

19.6.1 Operational, Safety, and Mission Assurance Impacts

The magnitude of the operational impacts will depend on the type of tasks the crew is expected to perform. However, there are some general areas where conclusions can be drawn. Reducing the size of the crew to two crewmembers will result in a decreased skill mix, fewer crew-hours, fewer crewmembers to accomplish a given task, less redundancy in the case of a crewmember becoming incapacitated, and a requirement for more training on a larger number of skills for each crewmember to ensure that the missions can be accomplished safely.

Regardless of the ratio of time spent performing lunar specific science or training for missions to Mars on the Moon, reducing the crew size to two crewmembers significantly decreases the ability to perform meaningful simulations and tests. The tasks will most likely be required to be smaller in scope so as to accommodate a decreased skill mix. Although Apollo missions proved limited exploration can be performed with two crewmembers, studies on crew sizes for Mars missions have found that a minimum crew size of four to five people is needed to assure a proper skill mix. However, when operational scenarios and workload were taken into account, a crew size of six to eight people offered much better benefits as far as operational safety, task distribution, and redundancy. It is probably reasonable to assume that future lunar missions will require more extensive activities than those of Apollo, but probably no more than those needed to complete a mission to Mars. Therefore, the training requirements would most likely be increased for each crewmember, relative to Apollo missions, in order to ensure that the proper skill mix is present for meeting mission objectives and performing the critical mission functions. Ultimately, the mission will need to have to be scoped such that crew training, crew scheduling, and mission objectives can all be achieved.

Coupling a two-crewmember approach with the TRM surface stay of seven days, results in an extremely small number of crew-hours available to accomplish tasks and experiment with multiple operational approaches. Furthermore, some tasks may go beyond the issue of number of available crew-hours; instead, the limiting factor may be the actual number of crewmembers that are needed to perform a given task. This point may become a safety issue in the event that a crewmember becomes incapacitated. Certainly the mission would be cut short in this case, but the Lander and CEV would still need to accommodate critical procedures that can be performed by only one person (e.g. lunar surface landing/ascent, docking, earth entry/landing, post flight ops, etc…). Additionally, during earth-based mission simulations, certain tasks (e.g. surface exploration EVAs) have been shown to be much more productive and safe with three crewmembers present.

Reducing the crew size to two crewmembers will also introduce specific safety considerations. The TRM approach allowed for teams of two crewmembers to perform EVAs on alternating days, thus allowing each team to rest between days of EVAs. This becomes of a factor during longer duration surface missions. Apollo missions proved that two crewmembers can perform three consecutive days of EVA, but there are concerns in the safety community to plan much more than this without a day of rest. Although this could be planned for in the mission planning of the two-crewmember approach, it was inherent in the TRM approach. Additionally, if both crewmembers perform the EVA in the two-crewmember approach, which is recommended, it will leave the Lander unoccupied. While the Apollo missions proved that this is possible, the risks associated with leaving a spacecraft unoccupied will have to be mitigated.

19.6.2 Architecture Sizing Impacts

As stated previously, two scenarios were considered for the two-crewmember approach: with and without an airlock. For the approach that included an airlock, there was a reduction of 10.5% from 229,716 kg to 205,685 kg. In this scenario, the mass of the largest launch was reduced by 8.6% from 94,109 kg to 86,036 kg. For the approach without an airlock, there was a reduction of 13.1% from 229,716 kg to 199,667 kg. In this scenario, the mass of the largest launch was reduced by 12.1% from 94,109 kg to 82,723 kg.

Element	TRM (kg)	2 Crew (Airlock) (kg)	Mass Change (kg)	Percentage Change
CEV Crew Module	8,812	7,109	(1,703)	(19.3)
CEV Service Module	17,560	15,407	(2,153)	(12.3)
Ascent Stage	19,906	17,727	(2,179)	(10.9)
Descent Stage	22,608	20,608	(2,000)	(8.8)
Kick Stage	27,465	25,054	(2,411)	(8.8)
EDS1	94,109	86,036	(8,073)	(8.6)
EDS2	39,256	33,744	(5,512)	(14.0)
Largest Launch	**94,109**	**86,036**	**(8,073)**	**(8.6)**
Total Architecture	**229,716**	**205,685**	**(24,031)**	**(10.5)**

Table 19.6.2-1: Effects of reducing the crew size to two crewmembers, with an airlock kept in the concept of the Lander

Element	TRM (kg)	2 Crew (No Airlock) (kg)	Mass Change (kg)	Percentage Change
CEV Crew Module	8,812	7,109	(1,703)	(19.3)
CEV Service Module	17,560	15,407	(2,153)	(12.3)
Ascent Stage	19,906	17,469	(2,437)	(12.2)
Descent Stage	22,608	19,152	(3,456)	(15.3)
Kick Stage	27,465	24,063	(3,402)	(12.4)
EDS1	94,109	82,723	(11,386)	(12.1)
EDS2	39,256	33,744	(5,512)	(14.0)
Largest Launch	**94,109**	**82,723**	**(11,386)**	**(12.1)**
Total	**229,716**	**199,667**	**(30,049)**	**(13.1)**

Table 19.6.2-2: Effects of reducing the crew size to two crewmembers, removing the airlock from the concept of the Lander

19.6.3 System Impacts

The major conclusion after examining each system was that, with the possible exception of the EVA system, none of the technologies would change. Only the system masses and quantity of consumables would vary relative to the TRM.

CEV Crew Module's System Mass Changes				
System	**TRM (kg)**	**2 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	1523	1136	(387)	(25.4)
Protection	822	677	(145)	(17.6)
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	706	(31)	(4.2)
Environment	691	522	(169)	(24.5)
Other	835	777	(58)	(6.9)
Growth	1041	884	(157)	(15.1)
Non-Cargo	966	562	(404)	(41.8)
Cargo	1478	1133	(345)	(23.3)
Non-Propellant	55	49	(6)	(10.9)
Propellant	64	64	No Change	0.0
Total	8812	7109	(1703)	(19.3)

Table 19.6.3-1: CEV Crew Module's System Mass Changes, Resulting From Decreasing the Crew Size from Four to Two Crewmembers

CEV Service Module's System Mass Changes				
System	**TRM (kg)**	***2 Crew (kg)***	**Mass Change (kg)**	**% Change**
Structure	1455	1286	(169)	(11.6)
Protection	0	0	No Change	0.0
Propulsion	1408	1294	(114)	(8.1)
Power	661	639	(22)	(3.3)
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	104	(6)	(5.5)
Other	100	100	No Change	0.0
Growth	747	685	(62)	(8.3)
Non-Cargo	305	261	(44)	(14.4)
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1360	(82)	(5.7)
Propellant	11332	9679	(1653)	(14.6)
Total	17560	15407	(2153)	(12.3)

Table 19.6.3-2: CEV Service Module's System Mass Changes, Resulting From Decreasing the Crew Size from Four to Two Crewmembers

In tables 19.6.3-1 and 19.6.3-2 it is noteworthy that decreasing the crew size from four to two crewmembers has a moderate impact on the CEV Crew Module and Service Module. Most of the mass changes in the Crew Module portion of the CEV are due to the decreased structural and habitability requirements. These modifications trickle down to the Service Module in the form of decreased propellant requirements.

Airlock Included in Lander Concept

Lander Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**2 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	760	(79)	(9.4)
Protection	73	73	No Change	0.0
Propulsion	1631	1546	(85)	(5.2)
Power	813	787	(26)	(3.2)
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	748	(103)	(12.1)
Other	455	455	No Change	0.0
Growth	1080	1021	(59)	(5.5)
Non-Cargo	1483	921	(562)	(37.9)
Cargo	227	227	No Change	0.0
Non-Propellant	1014	914	(100)	(9.9)
Propellant	10703	9537	(1166)	(10.9)
Total	19906	17727	(2179)	(10.9)

Table 19.6.3-3: Lander Ascent Stage's System Mass Changes, Resulting From Decreasing the Crew Size From Four to Two Crewmembers

Lander Descent Stage's System Mass Changes				
System	**TRM (kg)**	**2 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	553	538	(15)	(2.7)
Protection	50	50	No Change	0.0
Propulsion	1413	1299	(114)	(8.1)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	649	(59)	(8.3)
Growth	678	641	(37)	(5.5)
Non-Cargo	464	418	(46)	(9.9)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	15846	(1727)	(9.8)
Total	22608	20608	(2000)	(8.8)

Table 19.6.3-5: Lander Descent Stage's System Mass Changes, Resulting From Decreasing the Crew Size From Four to Two Crewmembers

Airlock Removed from Lander Concept

Lander Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**2 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	759	(80)	(9.5)
Protection	73	73	No Change	0.0
Propulsion	1631	1537	(94)	(5.8)
Power	813	787	(26)	(3.2)
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	651	(200)	(23.5)
Other	455	455	No Change	0.0
Growth	1080	1000	(80)	(7.4)
Non-Cargo	1483	917	(566)	(38.2)
Cargo	227	227	No Change	0.0
Non-Propellant	1014	927	(87)	(8.6)
Propellant	10703	9399	(1304)	(12.2)
Total	19906	17469	(2437)	(12.2)

Table 19.6.3-4: Lander Ascent Stage's System Mass Changes, Resulting From Decreasing the Crew Size From Four to Two Crewmembers

Lander Descent Stage's System Mass Changes				
System	**TRM (kg)**	***2 Crew (kg)***	**Mass Change (kg)**	**% Change**
Structure	553	532	(21)	(3.8)
Protection	50	50	No Change	0.0
Propulsion	1413	1252	(161)	(11.4)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	0	(530)	(100.0)
Other	708	624	(84)	(11.9)
Growth	678	519	(159)	(23.5)
Non-Cargo	464	400	(64)	(13.8)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	15137	(2436)	(13.9)
Total	22608	19152	(3456)	(15.3)

Table 19.6.3-6: Lander Descent Stage's System Mass Changes, Resulting From Decreasing the Crew Size From Four to Two Crewmembers

Airlock Included in Lander Concept

Element	System	TRM (kg)	2 Crew (kg)	Mass Change (kg)	% Change
Propulsive Stages' System Mass Changes					
Kick Stage	Structure	621	581	(40)	(6.4)
	Protection	0	0	No Change	0.0
	Propulsion	1530	1457	(73)	(4.8)
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	509	(22)	(4.1)
	Non-Cargo	953	880	(73)	(7.7)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	21122	(2201)	(9.4)
	Total	27465	25054	(2411)	(8.8)
EDS1	Structure	1972	1846	(126)	(6.4)
	Protection	0	0	No Change	0.0
	Propulsion	4361	4102	(259)	(5.9)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1375	(77)	(5.3)
	Non-Cargo	3109	2858	(251)	(8.1)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	74930	(7359)	(8.9)
	Total	94109	86036	(8073)	(8.6)

Airlock Removed from Lander Concept

Element	System	TRM (kg)	2 Crew (kg)	Mass Change (kg)	% Change
Propulsive Stages' System Mass Changes					
Kick Stage	Structure	621	564	(57)	(9.2)
	Protection	0	0	No Change	0.0
	Propulsion	1530	1427	(103)	(6.7)
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	499	(32)	(6.0)
	Non-Cargo	953	849	(104)	(10.9)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	20220	(3103)	(13.3)
	Total	27465	24063	(3402)	(12.4)
EDS1	Structure	1972	1793	(179)	(9.1)
	Protection	0	0	No Change	0.0
	Propulsion	4361	3996	(365)	(8.4)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1343	(109)	(7.5)
	Non-Cargo	3109	2755	(354)	(11.4)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	71910	(10379)	(12.6)
	Total	94109	82723	(11386)	(12.1)

Airlock Included in Lander Concept

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**3 Days (kg)**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	821	(111)	(11.9)
	Protection	0	0	No Change	0.0
	Propulsion	2318	1899	(419)	(18.1)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	728	(106)	(12.7)
	Non-Cargo	1355	1183	(172)	(12.7)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	28193	(4704)	(14.3)
	Total	39256	33744	(5512)	(14.0)

Table 19.6.3-7: Propulsive Stages' System Mass Changes, Resulting From Decreasing the Crew Size From Four to Two Crew-members

Airlock Removed from Lander Concept

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**3 Days (kg)**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	821	(111)	(11.9)
	Protection	0	0	No Change	0.0
	Propulsion	2318	1899	(419)	(18.1)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	728	(106)	(12.7)
	Non-Cargo	1355	1183	(172)	(12.7)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	28193	(4704)	(14.3)
	Total	39256	33744	(5512)	(14.0)

Table 19.6.3-8: Propulsive Stages' System Mass Changes, Resulting From Decreasing the Crew Size From Four to Two Crew-member

Active Thermal Control System

Reducing the crew size to two crewmembers would reduce the ATCS heat load by about 240 W. This is a small percentage of the total heat load, which has been approximated as 9 kW for this study. If the power requirements are decreased, the ATCS will have a reduced thermal load allowing for reductions in ATCS mass, power, and volume. The heat load has been approximated as a sum of the total vehicle power, fuel cell waste heat, and metabolic heat from the crew. The total vehicle power is the largest portion and would have the greatest impact on ATCS size.

Another aspect of the Lander that might change regards the possible option to eliminate the airlock from the Lander concept. If this option were chosen, the systems inside the Lander would have to be designed to accommodate a cabin depressurization/re-pressurization for each EVA. This would mean the ATCS would be required to accommodate the thermal operating requirements of all spacecraft systems.

Avionics

Reducing the crew size to two crewmembers may have a moderate impact on the avionics technologies and system designs. This would necessitate the ability for all crewed spacecrafts in the architecture to be operated by two crewmembers. This may place a greater requirement on the avionics system to provide more crew-aids and streamlined controls/displays, especially for critical events such as docking and lunar landing. Many of these impacts were beyond the scope of this study, but merit further consideration in the future.

Descent and Landing System

Decreasing the crew size to two crewmembers would decrease the mass of the sub-systems that perform Descent and Landing due to the decreased vehicle mass. For this study, the same technology selection was made for this variation as was made for the TRM approach. This decision merits further evaluation in the future.

Environmental Control and Life Support System

Reducing the crew size to two crewmembers will reduce the required life support consumables and allow the size/quantity of the ECLSS hardware’s mass, power, and volume to shrink. If fuel cells remain the power source in the various elements, more excess water will be required to be vented since crew consumption will be reduced. Additionally, if the option to eliminate the airlock from the Lander concept were chosen, the ECLSS system would be required to survive after being exposed to vacuum conditions and provide the re-pressurization gases for the crew cabin.

Extra-Vehicular Activity System

Reducing the crew size to two would affect both the EVA system mass and operational approaches for EVAs. In the TRM, the approach was adopted to carry four ascent/entry suits and four surface suits/PLSSs. The numbers for their associated masses would be reduced by a factor of two, if the crew size were cut in half. Second, it may be possible to remove the airlock from the Lander concept, if a replacement dust mitigation measure could be found. This would save ~450 kg on the mass of the Lander, which ultimately saves ~6 metric tons in the total architecture. Operational impacts may include performing fewer EVAs and limiting the types of activities that are attempted. If the airlock were removed, the egress and ingress operations would be altered to those associated with cabin depressurization and re-pressurization.

Human Factors and Habitability System

It is not anticipated that decreasing the crew size to two would have an impact on the technologies chosen for any of the elements, but this variation will reduce all consumables that are sized on a per-person basis (e.g. food, food prep supplies, waste collection supplies, hygiene supplies, clothing, housekeeping supplies, operational supplies, medical consumables, personal hygiene kits, recreational equipment, sleep accommodations, seats). In addition, consideration needs to be given from a Human Factors perspective to designing a vehicle that is fully operable by two crewmembers. This may require more automation than the baseline vehicle that is operated by four crewmembers. If the vehicle is to be at least one-fault tolerant, systems should be designed such that the “one-fault” could be the loss of functionality of one crewmember (due to illness, injury, etc.). Therefore, the vehicle systems should all be operable by only a single crewmember.

Power System

Reducing the crew size to two would decrease the overall power requirement. This would result in a smaller power plant to supply the power needs. Additionally, if the vehicles were to shrink in size due to the decreased crew size, there may also be mass savings in the power management and distribution subsystem.

Propulsion System

It is not anticipated that decreasing the crew size to two would have an impact on the technologies chosen for any of the elements, but this variation will scale the size and mass of the propellant systems downwards due to the decreased vehicle masses.

Structures and Thermal Protection Systems

It is not anticipated that decreasing the crew size to two would have an impact on the technologies chosen for any of the elements, but this variation will scale the size and mass of all the structures downwards due to the decreased system masses on all of the elements and the decreased pressurized volumes on the habitable elements.

19.6.4 Impact on Mars Preparations

Operations, Safety, and Mission Assurance

It is difficult to imagine an operational advantage to be gained from reducing the lunar crew size to two in preparing for missions to Mars, especially as it relates to preparation for Mars. The Operations Concept for the Human Exploration of Mars (DV-00-014), Second Edition, May 17, 2000, Appendix A, Recommendations on Crew Size, cites six people as the best practical crew size for a Mars mission, with a minimum recommended crew size of four. Other exploration mission studies have recommended a minimum crew size of five with a recommendation between six to eight. Therefore, this variation would deviate from the likely lunar surface preparations for missions to Mars as compared to the TRM.

Reducing the crew size to two limits the amount of science, increases fatigue levels, increases the risk for contingency scenarios, and potentially increases the risk of psychological impact to the crew. Overall, reducing the crew size to two could lower the probability of mission success for the lunar landing, and provide less relevant experience for a potential Mars mission than the TRM.

Active Thermal Control System

There would be no significant effect of reducing the crew size to two on the development of systems for a Mars mission.

Avionics System

There would be no significant effect of reducing the crew size to two on the development of systems for a Mars mission, except in the cases where new technologies are adopted to provide greater ease in operation of the spacecrafts due to the reduced crew size. These technologies may also provide benefits during a mission to Mars, even though the Mars mission crew sizes are expected to be larger.

Descent and Landing System

Although no different technologies were selected to accommodate the entry, descent, and landing of the two-crew version of the CEV, because scale of the systems decrease, the descent and landing systems probably deviates from that used during a mission to Mars. Entry speed differences between lunar missions and Mars missions remain a source of uncertainty that will impact landing system technology choices and design.

Environmental Control and Life Support System

There would be no significant effect of reducing the crew size to two on the development of systems for a Mars mission.

Extra-Vehicular Activity System

The benefits of training for Mars missions would be significantly limited. Training associated with EVA/IVA interactions, multiple EVA teams, and any tasks that require more than two EVA crewmembers would be lost. There could also be impacts on system design. For example, in the case of reducing the crew size to two crewmembers, it may be deemed appropriate to perform the lunar EVAs without the use of an airlock. This would change the system hardware and operations associated with ingress and egress of the Lander. All the suit and PLSS designs could remain the same for a mission to Mars, if the suit architecture and technologies were designed such that they could accommodate a go anywhere-anytime approach. If the suit and PLSS designs deviated from a go anywhere-anytime approach, the magnitude of the difference between the lunar mission EVA design and the Mars mission EVA design would be a result of the specific deviances. A study that examines the specific differences between Mars and lunar mission EVA operations and technologies should be pursued in the future, once the two architectures are more thoroughly understood.

Human Factors and Habitability System

There would be no significant effect of reducing the crew size to two on the development of systems for a Mars mission.

Power System

There would be no significant effect of reducing the crew size to two on the development of systems for a Mars mission.

Propulsion System

There would be no significant effect of reducing the crew size to two on the development of systems for a Mars mission.

Structures and Thermal Protection Systems

There would be no significant effect of reducing the crew size to two on the development of systems for a Mars mission.

19.7 Crew Size of 6

This trade examined the impact of increasing the crew size from the TRM approach of four crewmembers to six crewmembers.

This change from the TRM approach results in an increase of 20,572 kg (9.0%) in total architecture mass. These increases are the result of increased consumables required by the systems, scaling the system sizes up, and an increase in habitable volume.

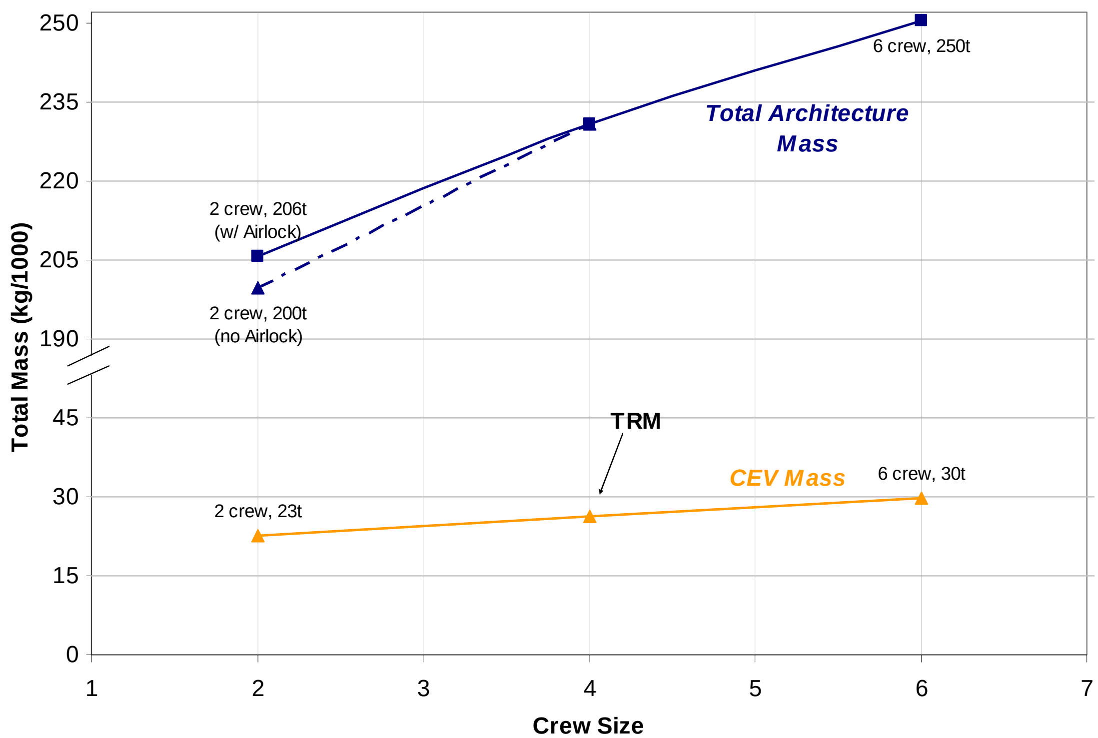

Figure 19.7-1: Effects of Varying the Crew Size on Total Architecture Mass

A larger crew would allow more specialization among the crewmembers, allowing a greater variety as well as a greater number of tasks to be accomplished during the course of a single mission. A larger crew size would allow a greater skill mix, greater redundancy in critical skills, provide for a more even redistribution of tasks in the case of an ill or injured crewmember(s), and allow for a greater variety of EVA approaches/activities.

This variation most likely brings the mission crew size closer to that which will be used during a mission to Mars. Therefore, increasing the crew size to six would provide additional operational experience beneficial to preparing for a mission to Mars.

19.7.1 Operational, Safety, and Mission Assurance Impacts

The magnitude of the operational impacts will depend on the type of tasks the crew is expected to perform. However, there are some general areas where conclusions can be drawn. Increasing the crew to six crewmembers will result in an increased skill mix, a greater number of available crew-hours, a greater number of crewmembers to accomplish a given task, greater redundancy in the case of a crewmember becoming incapacitated, and the ability to have more specialized training for each crewmember resulting in the ability for the crew to perform a greater number of specialized tasks.

An advantage of this variation is its likely similarity to long-term exploration missions to destinations beyond the Earth-Moon system, such as missions to Mars. Although Apollo missions proved limited exploration can be performed with two crewmembers, studies on crew sizes for missions to Mars have found that a minimum crew size of four to five people is needed to assure a proper skill mix. However, when operational scenarios and workload were taken into account, a crew size of six to eight people offered much better benefits as far as operational safety, task distribution, and redundancy. It is probably reasonable to assume that future lunar missions will require more extensive activities than those of Apollo, but probably no more than those needed to complete a mission to Mars. Therefore, using six crewmembers for lunar missions would seem to provide excellent operational experience with crews of similar skill mixes for preparation for missions to Mars. If the crew size were increased to six crewmembers, the remaining two limiting factors in preparing for missions to Mars would be the duration the crew remains on the lunar surface and the environmental differences between the Moon and Mars.

Coupling a six-crewmember approach with the TRM surface stay of seven days, results in a relatively large number of crew-hours compared to that of the Apollo missions. Additionally, depending on the EVA capabilities, it could provide up to six crewmembers to perform a given task, thus allowing relatively large tasks to be accomplished. It would also provide the possibility for the crew to gain experience in coupling EVA and IVA activities to accomplish tasks. In the case of having six crewmembers on the lunar surface to accomplish tasks, the limiting factor may be the number of hours on the lunar surface rather than the number of crew.

Increasing the crew size to six crewmembers would allow the mission planners to focus the crew members' time on a greater number of more specific tasks due to the increased skill mix. The crew could have a diverse skill mix, including a geologist, biologist, medical doctor, others with EMT level medical training, and other science and technology specialties appropriate to specific mission objectives. This would enable a much broader scope of tasks to be accomplished in a single mission, including testing remote medical technology and techniques, maintenance and construction methods, and in-situ resource utilization (ISRU) experiments similar to what might be used on a mission to Mars. It would also allow crewmembers to provide better aid to an ill or injured crewmember, while still allowing a large portion of the crewmembers to perform critical procedures needed to finish the mission safely.

The obvious disadvantage of using six crewmembers is the extra ~9% in total architecture mass that is needed beyond what is required for the TRM approach. Depending on launch vehicle capabilities and individual element masses, extra launches may be needed. Extra launches would add extra rendezvous with associated on-orbit procedures/checkouts, extend the total duration of the architecture, and decrease the probability of mission success.

19.7.2 Architecture Sizing Impacts

Increasing the crew size to six would significantly increase the mass of the CEV, Lander, and all propulsion stages. Increasing the crew size to six increases the total architecture mass by 9.0% from 229,716 kg to 250, 288 kg. The mass of the largest launch increases by 7.6% from 94,109 kg to 101,279 kg.

Element	TRM (kg)	6 Crew (kg)	Mass Change (kg)	Percentage Change
CEV	8,812	10,424	1,612	18.3
Service Module	17,560	19,337	1,777	10.1
Ascent Stage	19,906	21,841	1,935	9.7
Descent Stage	22,608	24,382	1,774	7.8
Kick Stage	27,465	29,604	2,139	7.8
EDS1	94,109	101,279	7,170	7.6
EDS2	39,256	43,421	4,165	10.6
Largest Launch	**94,109**	**101,279**	**7,170**	**7.6**
Total	**229,716**	**250,288**	**20,572**	**9.0**

Table 19.7.2-1: Effects of Increasing the Crew Size to Six Crewmembers

19.7.3 System Impacts

The major conclusion after examining each system was that, with the possible exception of the EVA system, none of the technologies would change. Only the system masses and quantity of consumables would vary relative to the TRM.

CEV Crew Module's System Mass Changes				
System	**TRM (kg)**	**6 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	1523	1917	394	25.9
Protection	822	958	136	16.5
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	753	16	2.2
Environment	691	805	114	16.5
Other	835	890	55	6.6
Growth	1041	1184	143	13.7
Non-Cargo	966	1822	856	88.6
Cargo	1478	1822	344	23.2
Non-Propellant	55	61	6	10.9
Propellant	64	64	No Change	0.0
Total	8812	10424	1612	18.3

Table 19.7.3-1: CEV Crew Module's System Mass Changes, Resulting From Increasing the Crew Size from Four to Six Crewmembers

CEV Service Module's System Mass Changes				
System	**TRM (kg)**	**6 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	1455	1545	90	6.2
Protection	0	0	No Change	0.0
Propulsion	1408	1507	99	7.0
Power	661	672	11	1.7
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	113	3	2.7
Other	100	100	No Change	0.0
Growth	747	787	40	5.4
Non-Cargo	305	343	38	12.5
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1484	42	2.9
Propellant	11332	12786	1454	12.8
Total	17560	19337	1777	10.1

Table 19.7.3-2: CEV Service Module's System Mass Changes, Resulting From Increasing the Crew Size from Four to Six Crewmembers

It is noteworthy that increasing the crew size from four to six crewmembers has a moderate impact on the CEV Crew Module and Service Module. Most of the mass changes in the crew module portion of the CEV are due to the increased structural and habitability requirements. These modifications trickle down to the Service Module in the form of increased propellant requirements.

Lander Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**6 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	1112	273	32.5
Protection	73	73	No Change	0.0
Propulsion	1631	1703	72	4.4
Power	813	826	13	1.6
Control	0	0	No Change	0.0
Avionics	738	775	37	5.0
Environment	851	909	58	6.8
Other	455	455	No Change	0.0
Growth	1080	1171	91	8.4
Non-Cargo	1483	1722	239	16.1
Cargo	227	300	73	32.2
Non-Propellant	1014	1056	42	4.1
Propellant	10703	11738	1035	9.7
Total	19906	21841	1935	9.7

Table 19.7.3-3: Lander Ascent Stage's System Mass Changes, Resulting From Increasing the Crew Size from Four to Six Crewmembers

Lander Descent Stage's System Mass Changes				
System	**TRM (kg)**	**6 Crew (kg)**	**Mass Change (kg)**	**% Change**
Structure	553	567	14	2.5
Protection	50	50	No Change	0.0
Propulsion	1413	1514	101	7.1
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	530	No Change	0.0
Other	708	761	53	7.5
Growth	678	712	34	5.0
Non-Cargo	464	504	40	8.6
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	19106	1533	8.7
Total	22608	24382	1774	7.8

Table 19.7.3-4: Lander Descent Stage's System Mass Changes, Resulting From Increasing the Crew Size from Four to Six Crewmembers

Propulsive Stages' System Mass Changes

Element	System	TRM (kg)	6 Crew (kg)	Mass Change (kg)	% Change
Kick Stage	Structure	621	657	36	5.8
	Protection	0	0	No Change	0.0
	Propulsion	1530	1596	66	4.3
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	552	21	4.0
	Non-Cargo	953	1019	66	6.9
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	25276	1953	8.4
	Total	27465	29604	2139	7.8
EDS1	Structure	1972	2084	112	5.7
	Protection	0	0	No Change	0.0
	Propulsion	4361	4593	232	5.3
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1521	69	4.8
	Non-Cargo	3109	3332	223	7.2
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	88824	6535	7.9
	Total	94109	101279	7170	7.6

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**6 Crew (kg)**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	1020	88	9.4
	Protection	0	0	No Change	0.0
	Propulsion	2318	2431	113	4.9
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	874	40	4.8
	Non-Cargo	1355	1490	135	10.0
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	36686	3789	11.5
	Total	39256	43421	4165	10.6

Table 19.7.3-5: Propulsive Stages' System Mass Changes, Resulting From Increasing the Crew Size from Four to Six Crewmembers

Active Thermal Control System

Increasing the crew size to six would increase the ATCS heat load by about 240 W. This is a small percentage of the total heat load, which has been approximated as 9 kW for this study. If the power requirements for the elements are increased, the ATCS will require increases in the ATCS mass, power, and volume in order to accommodate the increased thermal loads. The heat load has been approximated as a sum of the total vehicle power, fuel cell waste heat, and metabolic heat from the crew. The total vehicle power is the largest portion and would have the greatest impact on ATCS size.

Avionics System

Increasing the crew size to six would have minimal impact on the avionics technologies or system designs. Small variations such as the number of controls and displays may change, but any changes would be relatively minor.

Descent and Landing System

Increasing the crew size to six would increase the mass of the sub-systems that perform Entry, Descent, and Landing due to the increased vehicle mass. For this study, the same technology selection was made for this variation as was made for the TRM approach. This decision merits further evaluation in the future.

Environmental Control and Life Support System

Increasing the crew size to six would increase the amount of life support consumables, equipment (gases, food, emergency O2 and masks, habitation equipment, etc.), and associated storage volume. The fuel cells will likely be able to provide enough water for two additional crewmembers. The size of the proposed CO2 removal systems for the CEV and Lander will mostly likely remain the same.

Extra-Vehicular Activity System

Increasing the crew size to two would affect the EVA system mass and possibly the operational approaches used for EVAs. In the TRM, the approach was adopted to carry four ascent/entry suits and four surface suits/PLSSs. In the case of conducting a lunar mission with six crewmembers, it is likely that the crew would bring six surface suits, but only four PLSSs. Each suit would need to be custom fitted to each crewmember; however, the PLSSs could be interchangeable. This would limit the number of EVA crewmembers to a maximum of four at a given time, but would save storage volume and system mass. Mission planners would have much more flexibility in planning EVA operations. The crewmembers could split into two-member, three-member, or a four and two-member team. As with the TRM approach, it would be possible to have each team perform separate tasks, work on a combined task, and mix in various levels of EVA and IVA activities.

Human Factors and Habitability System

Increasing the crew size to six will not change any of the Habitation system hardware components, but will increase all consumables that are sized on a per-person basis (e.g. food, food prep supplies, waste collection supplies, hygiene supplies, clothing, housekeeping supplies, operational supplies, medical consumables, personal hygiene kits, recreational equipment, sleep accommodations, seats). Additional habitable volume will also be required.

Power System

Increasing the crew size to six will increase the overall power requirement. This will result in a larger power plant and a greater quantity of consumables to supply the power needs. Additionally, if the vehicles were to increase in size due to the increased crew size, there may also be an increase in mass for the power management and distribution subsystem.

Propulsion System

Increasing the crew size to six would not have an impact on the technologies chosen for any of the elements, but will scale the size and mass of the propellants upwards due to the increased vehicle mass.

Structures and Thermal Protection Systems

Increasing the crew size to six would not have an impact on the technologies chosen for any of the elements, but will scale the size and mass of all the structures upwards due to the increased system masses on all elements and the increased pressurized volumes on the habitable elements.

19.7.4 Impact on Mars Preparations

Operations, Safety, and Mission Assurance

This variation to the TRM has the most direct operational application to preparing for missions to Mars, especially in the respects of allowing for a variety of skill mixes, real-time mission planning and execution, group dynamics and leadership structure of a six-person crew, EVA operations, and exploration objectives. Inserting time delays would allow mission planners the opportunity to test the different roles the crew and the MCC would play on a mission to Mars. Although the MCC would still provide long-term mission planning and non-time-critical troubleshooting guidance, more responsibility for the detailed crew activity scheduling could be relegated to the crew. All time-critical decision making would be the crew's responsibility, because immediate help from the MCC would not be possible. Six-person crews are appropriate for special long-duration lunar missions that are heavily focused on preparation for missions to Mars. Transferring the crew from the CEV to the Lander at L1 is analogous to the high earth orbit crew transfer approach that could occur for a mission to Mars and would provide good operational experience for that activity.

Active Thermal Control System

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission.

Avionics System

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission.

Descent and Landing System

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission, except in the respect that developing landing systems to accommodate likely Mars-mission crew sizes may be beneficial. Entry speed differences between lunar missions and Mars missions remain a source of uncertainty that will impact landing system technology choices and design.

Environmental Control and Life Support System

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission.

Extra-Vehicular Activity System

Increasing the crew size to six would improve the ability to train crewmembers and learn the operational aspects of performing a mission to Mars. A greater number of simulations that incorporate training associated with EVA/IVA interactions, multiple EVA teams, and any tasks that require more than two EVA crewmembers could be integrated into the lunar mission design relative to the TRM approach. All the suit and PLSS designs could remain the same for a mission to Mars, if the suit architecture and technologies were designed such that they could accommodate a go anywhere-anytime approach. If the suit and PLSS designs deviated from a go anywhere-anytime approach, the magnitude of the difference between the lunar mission EVA design and the Mars mission EVA design would be a result of the specific deviances. A study that examines the specific differences between Mars and lunar mission EVA operations and technologies should be pursued in the future, once the two architectures are more thoroughly understood.

Human Factors and Habitability Systems

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission.

Power System

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission.

Propulsion System

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission.

Structures and Thermal Protection Systems

There would be no significant effect of increasing the crew size to six on the development of systems for a Mars mission.

19.8 Launch Delay Interval Between 7 and 28 Days

This trade study examined the impact of varying the duration between launches from 7 to 28 days.

19.8.1 Mass Impacts

Reducing the duration of the launch interval from 14 to 7 days decreased the architectural mass by approximately 1,000 (0.4%). Increasing the duration from 14 to 21 days increased the architectural mass by approximately 800 kg (0.4%). Finally, increasing the duration from 14 to 28 days increased the architectural mass by approximately 1,400 kg (0.6%). These reductions and increases are mainly the result of boil-off associated with the Kick Stage and EDS's. Note: small changes were made in the trade reference mission mass estimates relative to those reported in other sections of this report as the approach evolved over the course of the study.

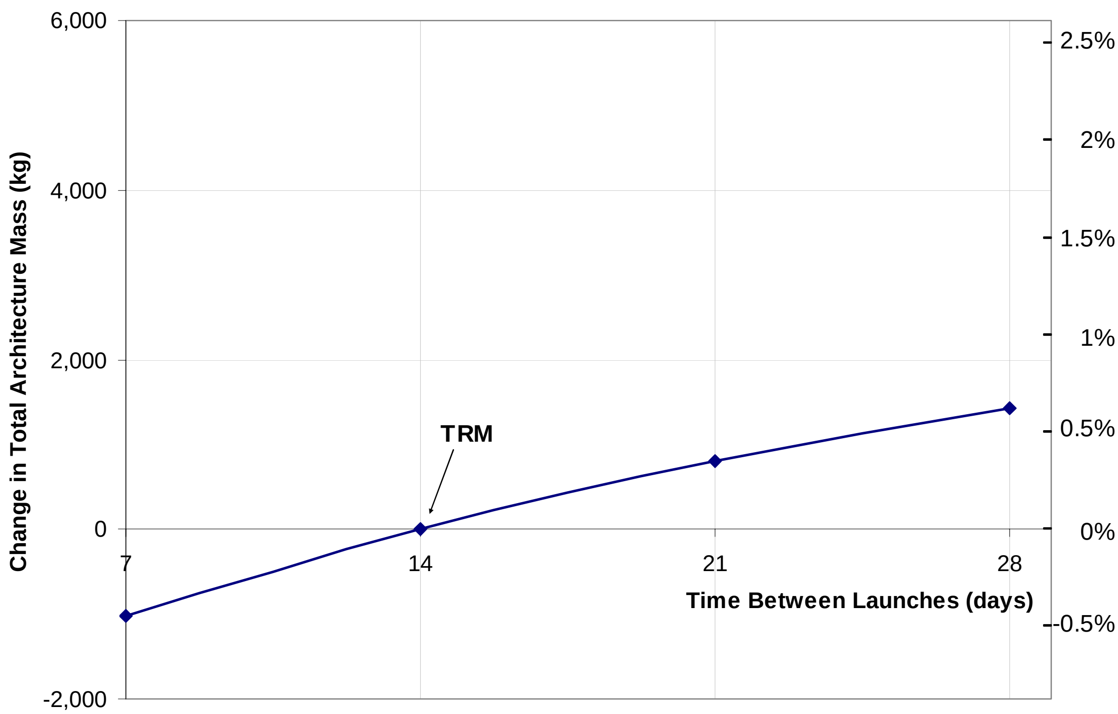

Figure 19.8.1-1: Mass Impacts to Architecture from Varying Launch Interval

It would be possible to minimize the losses due boil-off through the use of systems that either actively cool the propellants or insulate them from the environment. For example, in order to actively cool the propellants, hydrogen cryo-coolers could be implemented into the propulsion system designs. However, cryo-coolers that can operate to the proper specifications would need to be developed over the upcoming years. Surrounding the tanks with extra layers of MLI would be a low-tech alternative. However, this study did not examine the mass impacts of these alternatives. Instead, it only sought to assess the baseline designs as the launch interval was varied.

19.8.2 Timeline Impacts

Changing the launch interval from 14 to 7 days decreases the total duration of the mission from 69.1 days to 48.1 days. However, the crewed portion of the mission remains at 27.1 days.

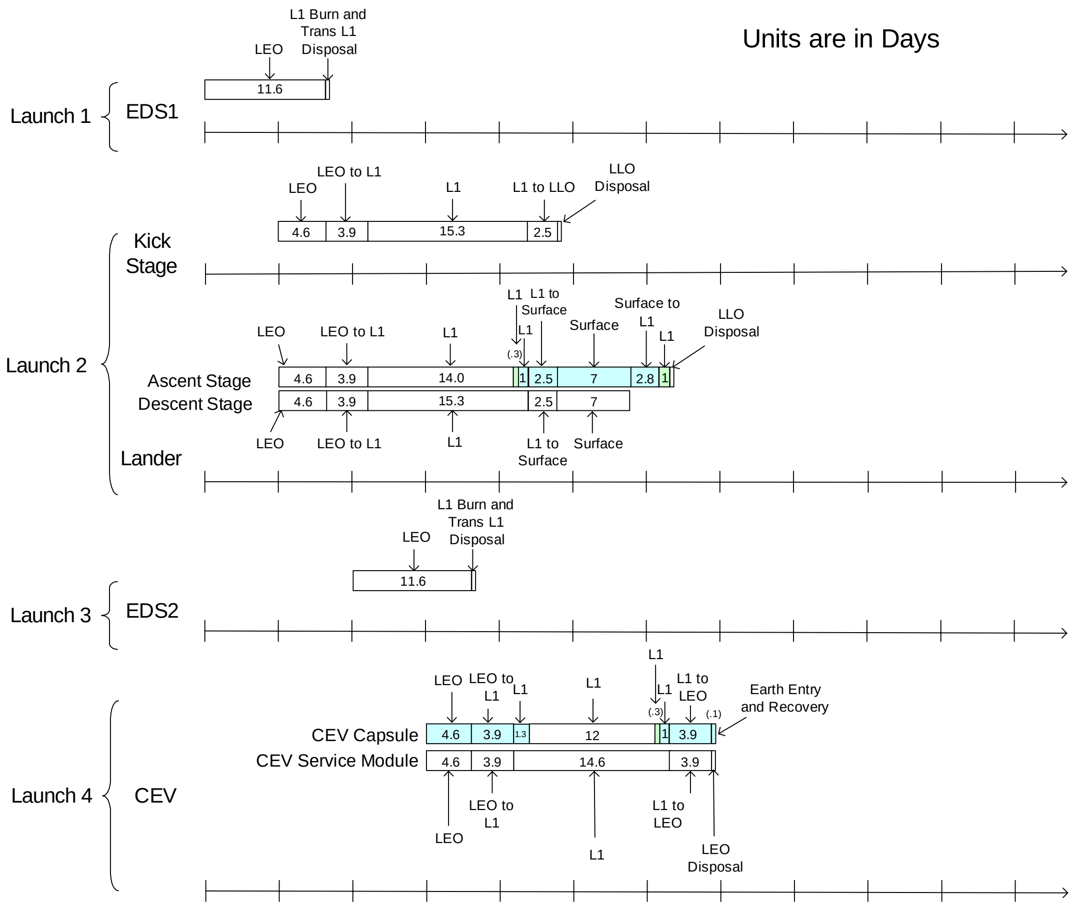

Figure 19.8.2-1: Nominal Mission Timeline with Launch Interval Decreased from 14 to 7 Days

On the other extreme, changing the launch interval from 14 to 28 days increases the total duration of the mission from 69.1 days to 111.1 days, as depicted in figure 19.8.2-1.

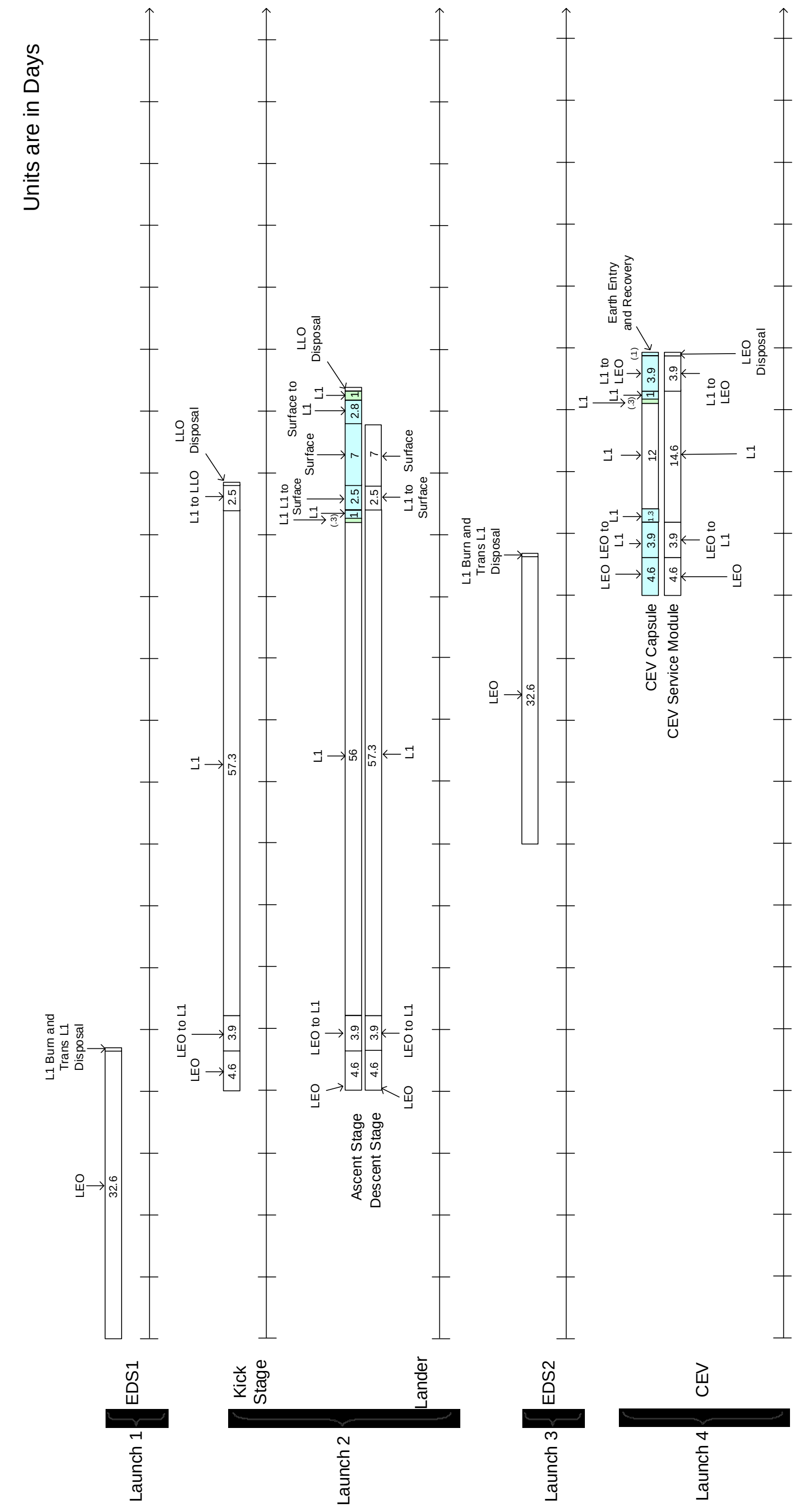

Figure 19.8.2-2: Nominal Mission Timeline with Launch Interval Increased from 14 to 28 Days

19.9 Surface Duration of 3 Days

This trade examined the impacts of reducing the duration of the lunar surface mission from the TRM approach of seven days to three days.

This change from the TRM approach results in a reduction of 5,958 kg (2.6%) in total architecture mass. These reductions are the result of reduced consumables required by the systems, reduced consumables for the crew, and system downscaling. Additionally, reducing the duration of the surface mission to three days may negate some of the main issues that made the inclusion of the Lander's airlock attractive in the TRM. If the airlock were to be removed, the total architecture mass could be reduced further by an additional 3,319 kg (1.4%), resulting in a total architecture mass savings of 9,277 kg (4.0%).

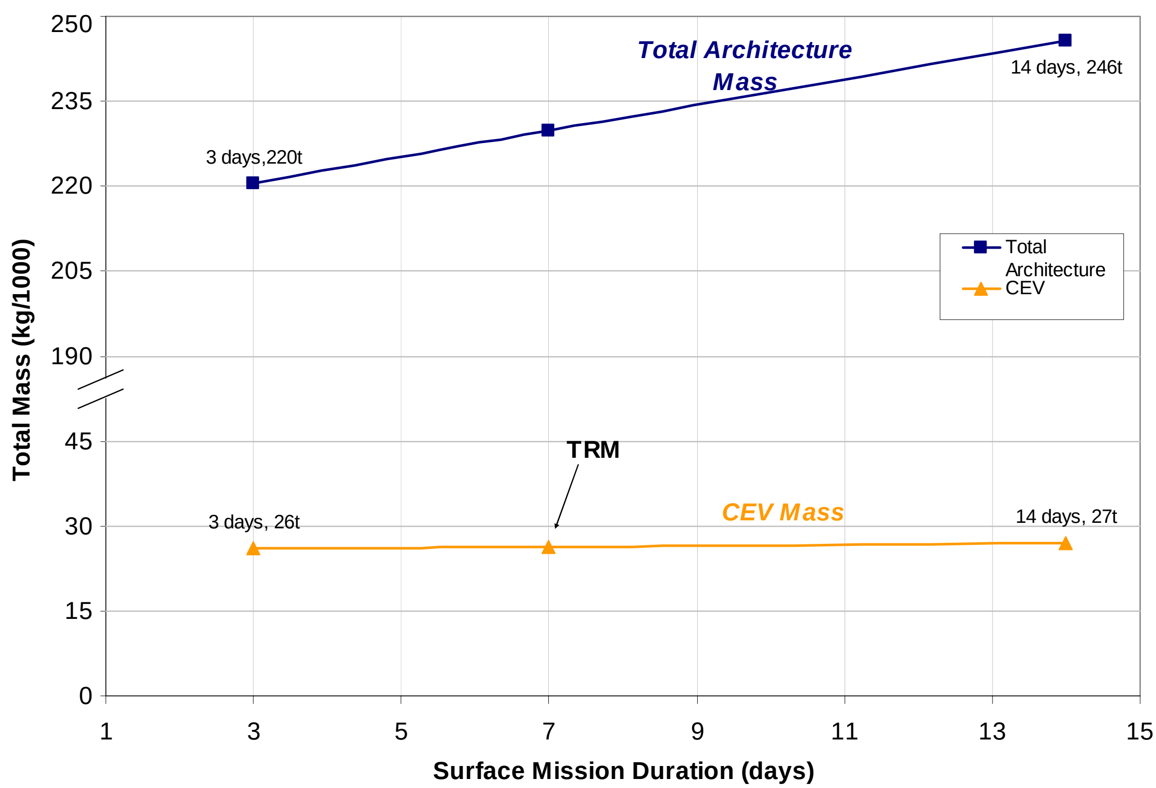

Figure 19.9-1: Effects of Varying the Duration of the Surface Mission on Total Architecture Mass

Significantly fewer tasks could be accomplished during the shorter stay. The reduction in the duration of the surface mission may result in the need to decrease the scope of the surface mission objectives. The final level to which the mission objectives are decreased in scope would be a trade between higher crew workloads (greater chance of fatigue) and a decrease in the amount of planned exploration/training work.

A three-day mission deviates from the surface mission durations that would be present in a mission to Mars (although, the TRM surface mission duration of seven days is also significantly shorter than would be present in a mission to Mars). Although the shortened duration of the surface mission might provide enough time to perform limited lunar exploration, its usefulness in preparing for the surface portion of a Mars mission might be extremely limited.

19.9.1 Operational, Safety, and Mission Assurance Impacts

One advantage of shortening the duration of the surface mission is that it might permit a mission that has been slightly delayed to proceed to the originally planned landing site and still satisfy any landing and operational constraints (e.g. lighting and thermal environments). During the course of studying the TRM approach of seven days, it was found that it might be difficult to balance appropriate lighting conditions for landing (Apollo imposed 7° to 20° sun angles during the lunar morning so the pilots could use shadows to distinguish features on the lunar surface) with "benign" thermal environments (EVA engineers suggest trying to avoid ±2 days around solar noon).

This variation has the disadvantage of severely limiting the amount of work and/or training that could be accomplished during each mission. It would be difficult to fit a meaningful number of activities such as lunar exploration, deployment/testing of technologies on the surface, and operations training/testing into a three-day mission.

From a safety and mission assurance perspective, a scenario where lunar surface time is reduced to three days would have little effect on the types of hazards and failures that could occur, or their respective effects. The effect on reliability prediction and estimation calculations between a seven-day mission and a three-day mission would result in a reduced minimum duty cycle used in calculating Mean Time Between Failure (MTBF). However, the reliability goal for the Lander and lunar system cannot be reduced based on surface time.

A shorter stay on the lunar surface would not lead to any safety or mission assurance based benefits. The same number of critical events (other than EVA tasks) will be required to perform the lunar surface phase of the mission. EVA time will be reduced and would result in a lower probability of an injury occurring during an EVA. However a shorter lunar surface time will most likely eliminate more risky long duration EVAs. A shorter stay would expose the crew to less radiation, however after the margin of safety is applied for shielding the difference between three to seven days might not equate into a change in design. A shorter lunar surface duration means less time for Mars mission related operations and technology testing, which may have an impact on the safety and reliability of future missions to Mars.

19.9.2 Architecture Sizing Impacts

As stated previously, two scenarios were considered for the two-crewmember approach: with and without an airlock. For the approach that included an airlock, there was a reduction of 2.6% from 229,716 kg to 223,758 kg. In this scenario, the mass of the largest launch was reduced by 3.1% from 94,109 kg to 91,196 kg. For the approach without an airlock, there was a reduction of

4.0% from 229,716 kg to 220, 439 kg. In this scenario, the mass of the largest launch was reduced by 5.0% from 94,109 kg to 89,367 kg.

Element	TRM	3 Day Duration (Airlock)	Mass Change	Percentage Change
CEV	8,812	8,812	0	0.0
Service Module	17,560	17,264	(296)	(1.7)
CEV Earth Departure Stage	39,256	38,888	(368)	(0.9)
Ascent Stage	19,906	19,124	(782)	(3.9)
Descent Stage	22,608	21,880	(728)	(3.2)
Kick Stage	27,465	26,594	(871)	(3.2)
Lander Earth Departure Stage	94,109	91,196	(2,913)	(3.1)
Largest Launch	**94,109**	**91,196**	**(2,913)**	**(3.1)**
Total	**229,716**	**223,758**	**(5,958)**	**(2.6)**

Table 19.9.2-1: Effects of Reducing the Duration of the surface mission to three days, with an airlock kept in the concept of the Lander

Element	TRM	3 Day Duration (No Airlock)	Mass Change	Percentage Change
CEV	8,812	8,812	0	0.0
Service Module	17,560	17,264	(296)	(1.7)
CEV Earth Departure Stage	39,256	38,888	(368)	(0.9)
Ascent Stage	19,906	19,262	(644)	(3.2)
Descent Stage	22,608	20,797	(1,811)	(8.0)
Kick Stage	27,465	26,049	(1,416)	(5.2)
Lander Earth Departure Stage	94,109	89,367	(4,742)	(5.0)
Largest Launch	**94,109**	**89,367**	**(4,742)**	**(5.0)**
Total	**229,716**	**220,439**	**(9,277)**	**(4.0)**

Table 19.9.2-2: Effects of reducing the duration of the surface mission to three days, removing the airlock from the concept of the Lander

19.9.3 System Impacts

The major conclusion after examining each system was that, with the possible exception of the EVA system, none of the technologies would change. Only the system masses and quantity of consumables would vary relative to the TRM.

CEV Crew Module's System Mass Changes				
System	**TRM (kg)**	**3 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	822	No Change	0.0
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	691	No Change	0.0
Other	835	835	No Change	0.0
Growth	1041	1041	No Change	0.0
Non-Cargo	966	966	No Change	0.0
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	55	No Change	0.0
Propellant	64	64	No Change	0.0
Total	8812	8812	No Change	0.0

Table 19.9.3-1: CEV Crew Module's System Mass Changes, Resulting From Decreasing the Surface Mission Duration from 7 to 3 Days

CEV Service Module's System Mass Changes				
System	**TRM (kg)**	**3 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	1455	1452	(3)	(0.2)
Protection	0	0	No Change	0.0
Propulsion	1408	1393	(15)	(1.1)
Power	661	652	(9)	(1.4)
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	741	No Change	0.0
Non-Cargo	305	301	(4)	(1.3)
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1310	(132)	(9.2)
Propellant	11332	11205	(127)	(1.1)
Total	17560	17264	(296)	(1.7)

Table 19.9.3-2: CEV Service Module's System Mass Changes, Resulting from Decreasing the Surface Duration from 7 to 3 Days

In tables 19.9.3-1 and 19.9.3-2 it is noteworthy that decreasing the duration of the surface mission does not significantly affect the CEV Crew Module or Service Module to a large degree. Most of the mass deltas in the service module arise due to the decreased boil-off resulting from the decreased time spent loitering at L1.

Airlock Included in Lander Concept

Lander Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**3 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	834	(5)	(0.6)
Protection	73	73	No Change	0.0
Propulsion	1631	1587	(44)	(2.7)
Power	813	780	(33)	(4.1)
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	826	(25)	(2.9)
Other	455	455	No Change	0.0
Growth	1080	1058	(22)	(2.0)
Non-Cargo	1483	1422	(61)	(4.1)
Cargo	227	227	No Change	0.0
Non-Propellant	1014	839	(175)	(17.3)
Propellant	10703	10284	(419)	(3.9)
Total	19906	19124	(782)	(3.9)

Table 19.9.3-3: Lander Ascent Stage's System Mass Changes, Resulting from Decreasing the Surface Duration from 7 to 3 Days

Lander Descent Stage's System Mass Changes				
System	**TRM (kg)**	**3 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	553	548	(5)	(0.1)
Protection	50	50	No Change	0.0
Propulsion	1413	1372	(41)	(2.9)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	526	(4)	(0.7)
Other	708	687	(21)	(3.0)
Growth	678	664	(14)	(2.1)
Non-Cargo	464	448	(16)	(3.4)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	16949	(624)	(3.6)
Total	22608	21880	(728)	(3.2)

Table 19.9.3-5: Lander Descent Stage's System Mass Changes, Resulting from Decreasing the Surface Duration from 7 to 3 Days

Airlock Removed from Lander Concept

Lander Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**3 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	834	(5)	(0.6)
Protection	73	73	No Change	0.0
Propulsion	1631	1592	(39)	(2.4)
Power	813	780	(33)	(4.1)
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	840	(11)	(1.3)
Other	455	455	No Change	0.0
Growth	1080	1063	(17)	(1.6)
Non-Cargo	1483	1424	(59)	(4.0)
Cargo	227	227	No Change	0.0
Non-Propellant	1014	877	(137)	(13.5)
Propellant	10703	10358	(345)	(3.2)
Total	19906	19262	(644)	(3.2)

Table 19.9.3-4: Lander Ascent Stage's System Mass Changes, Resulting from Decreasing the Surface Duration from 7 to 3 Days

Lander Descent Stage's System Mass Changes				
System	**TRM (kg)**	**3 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	553	545	(8)	(1.4)
Protection	50	50	No Change	0.0
Propulsion	1413	1346	(67)	(4.7)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	0	(530)	(100.0)
Other	708	673	(35)	(4.9)
Growth	678	550	(128)	(18.9)
Non-Cargo	464	437	(27)	(5.8)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	16558	(1015)	(5.8)
Total	22608	20797	(1811)	(8.0)

Table 19.9.3-6: Lander Descent Stage's System Mass Changes, Resulting from Decreasing the Surface Duration from 7 to 3 Days

Airlock Included in Lander Concept

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**3 Days (kg)**	**Mass Change (kg)**	**% Change**
Kick Stage	Structure	621	607	(14)	(2.3)
	Protection	0	0	No Change	0.0
	Propulsion	1530	1504	(26)	(1.7)
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	523	(8)	(1.5)
	Non-Cargo	953	927	(26)	(2.7)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	22528	(795)	(3.4)
	Total	27465	26594		
EDS1	Structure	1972	1928	(44)	(2.2)
	Protection	0	0	No Change	0.0
	Propulsion	4361	4268	(93)	(2.1)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1424	(28)	(1.9)
	Non-Cargo	3109	3019	(90)	(2.9)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	79633	(2656)	(3.2)
	Total	94109	91196	(2913)	(3.1)

Airlock Removed from Lander Concept

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**3 Days (kg)**	**Mass Change (kg)**	**% Change**
Kick Stage	Structure	621	598	(23)	(3.7)
	Protection	0	0	No Change	0.0
	Propulsion	1530	1487	(43)	(2.8)
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	518	(13)	(2.4)
	Non-Cargo	953	910	(43)	(4.5)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	22030	(1293)	(5.5)
	Total	27465	26049	(1416)	(5.2)
EDS1	Structure	1972	1898	(74)	(3.8)
	Protection	0	0	No Change	0.0
	Propulsion	4361	4209	(152)	(3.5)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1406	(46)	(3.2)
	Non-Cargo	3109	2962	(147)	(4.7)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	77967	(4322)	(5.3)
	Total	94109	89367	(4742)	(5.0)

Airlock Included in Lander Concept

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**3 Days (kg)**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	923	(9)	(1.0)
	Protection	0	0	No Change	0.0
	Propulsion	2318	2308	(10)	(0.4)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	830	(4)	(0.5)
	Non-Cargo	1355	1343	(12)	(1.0)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	32563	(334)	(1.0)
	Total	39256	38888	(368)	(1.1)

Table 19.9.3-7: Propulsive Stages' System Mass Changes, Resulting from Decreasing the Surface Duration from 7 to 3 Days

Airlock Removed from Lander Concept

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**3 Days (kg)**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	923	(9)	(1.0)
	Protection	0	0	No Change	0.0
	Propulsion	2318	2308	(10)	(0.4)
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	830	(4)	(0.5)
	Non-Cargo	1355	1343	(12)	(1.0)
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	32563	(334)	(1.0)
	Total	39256	38888	(368)	(1.1)

Table 19.9.3-8: Propulsive Stages' System Mass Changes, Resulting from Decreasing the Surface Duration from 7 to 3 Days

Active Thermal Control System

Reducing the duration of the surface mission to three days would not have an impact on the technologies or system designs chosen for any of the elements, except for the possible exception of the Lander. The current Lander design has an evaporator supplementing the radiators for heat rejection from the vehicle. Currently, the Lander is designed to have the evaporator rejecting 1.5 kW of heat throughout the surface portion of the mission. This equates to approximately 52.7 kg of water per day. However, since the water used in the current design is a by-product of the fuel cells, it does not impact launch mass. If a future design of the Lander were switch to a system other than fuel cells for power generation, the decrease in the mass of water could be taken into account.

Another aspect of the Lander that might change regards the possible option to eliminate the air-lock from the Lander concept. If this option were chosen, the systems inside the Lander would have to be designed to accommodate a cabin depressurization/re-pressurization for each EVA. This would mean the ATCS would be required to accommodate the thermal operating requirements of all spacecraft systems.

Avionics System

Reducing the duration of the surface mission to three days would not have an impact on the technologies chosen for any of the elements. However, if the decision were made to remove the airlock from the Lander's EVA system concept, it would be necessary to design the Lander's avionics system such that it could withstand cabin depressurizations/re-pressurizations.

Descent and Landing System

Reducing the duration of the surface mission to three days would not have an impact on the technologies chosen for any of the elements.

Environmental Control and Life Support System

Reducing the duration of the surface mission to three days would reduce the required life support consumables and allow the size/quantity of the Lander's ECLSS hardware's mass, power, and volume to shrink. Additionally, if the option to eliminate the airlock from the Lander concept were chosen, the ECLSS system would be required to survive after being exposed to vacuum conditions and provide the re-pressurization gases for the crew cabin.

Extra-Vehicular Activity System

Reducing the duration of the surface mission to three days opens the possibility of removing the airlock from the Lander concept. If this option were chosen, the ingress/egress operations for the crew would change. It is likely that all four crewmembers would perform EVAs simultaneously in this approach, rather than the two-in/two-out option that was used in the EVA concept for the TRM approach. Additionally, if the airlock were removed, there would be a requirement for all four crewmembers to have a suit and PLSS. Although, the decision was made in the TRM to carry four suits and four PLSSs as the baseline approach, there was flexibility in this decision which would not exist for an approach where all four crewmembers are forced to don their suits at the same time. If the airlock were removed from the concept, a mass of ~450 kg would be saved in the Lander EVA system, which ultimately saves ~3.5 metric tons in the total architecture. Note: The 3.5 metric ton architectural mass savings for this variation is different than the 6 metric ton architectural mass savings in the "Crew Size Reduced to 2" variation because of the difference in cabin size and quantity of re-pressurizations (due to the differences in surface duration), which leads to differences in the amount of cabin make-up gases and the propellant needed to descend those gases to the lunar surface (and ascend them in the case of a surface abort).

Human Factors and Habitability System

Reducing the surface stay to 3 days will not change any of the Habitation system hardware components, but will reduce all consumables that are sized by duration (e.g. food, food prep supplies, waste collection supplies, hygiene supplies, clothing, housekeeping supplies, operational supplies, medical consumables). In addition, reducing the surface stay could allow for some reduction in habitable volume required for the Lander.

Another aspect that merits further investigation is the required floor area for donning/doffing surface suits and carrying out daily operations. If the option to eliminate the airlock were chosen, there would need to be adequate room for four crewmembers to don and doff their suits. Using data generated during the First Lunar Outpost study for donning/doffing surface suits (JSC-26019), it appears that the current design of the four-person Lander has nearly exactly the correct floor space for allowing four crewmembers to simultaneously perform this activity. However, NASA-accepted requirements for floor area for surface elements analogous to habitable volume requirements in NASA STD-3000 do not currently exist. Efforts should be made to define these requirements for future design activity guidance.

Power System

Reducing the duration of the surface mission to three days would reduce the quantity of consumables needed for the Lander. The current concept of the Lander uses fuel cells to generate the required electricity. If the duration of the surface mission were decreased, the quantity of hydrogen and oxygen used as fuel cell reactants could be reduced.

Propulsion System

Reducing the duration of the surface mission to three days would not have an impact on the technologies chosen for any of the elements. However, the quantities of propellants would shrink slightly due to the decreased on-orbit durations resulting in less propellant boil-off.

Structures and Thermal Protection Systems

Reducing the duration of the surface mission to three days would not have an impact on the technologies of the structures for any of the elements. Some of the structural masses would decrease slightly due to the decreased masses of the other systems that would need to be supported.

19.9.4 Impact on Mars Preparations

Operations, Safety and Mission Assurance

Decreasing the duration of the lunar surface mission to three days would significantly limit the available time for testing technologies and training/experimenting with various ops plans. This decreased amount of time spent on testing technologies and developing ops procedures may translate to decreased safety and reliability of future missions to Mars. All missions to Mars involve surface stays of at least 30 days, so the crews would not experience a daily work routine that would in any way resemble a Mars mission.

Active Thermal Control System

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission. Some of the operational experience with certain technologies may be lost due to the decreased duration, but since no changes in technologies were recommended for this architectural approach, the development of system technologies that are applicable for missions to Mars is unaffected relative to the TRM.

Avionics System

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission. Some of the operational experience with certain technologies may be lost due to the decreased duration, but since no changes in technologies were recommended for this architectural approach, the development of system technologies that are applicable for missions to Mars is unaffected relative to the TRM.

Descent and Landing System

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission.

Environmental Control and Life Support System

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission. Some of the operational experience with certain technologies may be lost due to the decreased duration, but since no changes in technologies were recommended for this architectural approach, the development of system technologies that are applicable for missions to Mars is unaffected relative to the TRM.

Extra-Vehicular Activity System

A potential mass benefit of decreasing the mission duration to three days was the possibility of removing an airlock from the Lander's EVA system concept, resulting in the requirement to de-pressurize the entire module for EVAs. If this option were employed, it would remove the benefit of testing airlock-related dust abatement system technology important for future longer lunar surface durations and missions to Mars.

Human Factors and Habitability System

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission. Some of the operational experience with certain technologies may be lost due to the decreased duration, but since no changes in technologies were recommended for this architectural approach, the development of system technologies that are applicable for missions to Mars is unaffected relative to the TRM.

Power System

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission. Some of the operational experience with certain technologies may be lost due to the decreased duration, but since no changes in technologies were recommended for this architectural approach, the development of system technologies that are applicable for missions to Mars is unaffected relative to the TRM.

Propulsion System

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission.

Structures and Thermal Protection Systems

There would be no significant effect of decreasing the duration of the surface mission to three days on the development of systems for a Mars mission.

19.10 Surface Duration of 14 Days

This trade examined the impact of extending the duration of the lunar surface mission from the TRM approach of seven days to fourteen days.

This change from the TRM approach results in an increase of 15,961 kg (6.9%) in total architecture mass. These increases are the result of increased consumables required by the systems and scaling some of the system sizes upwards.

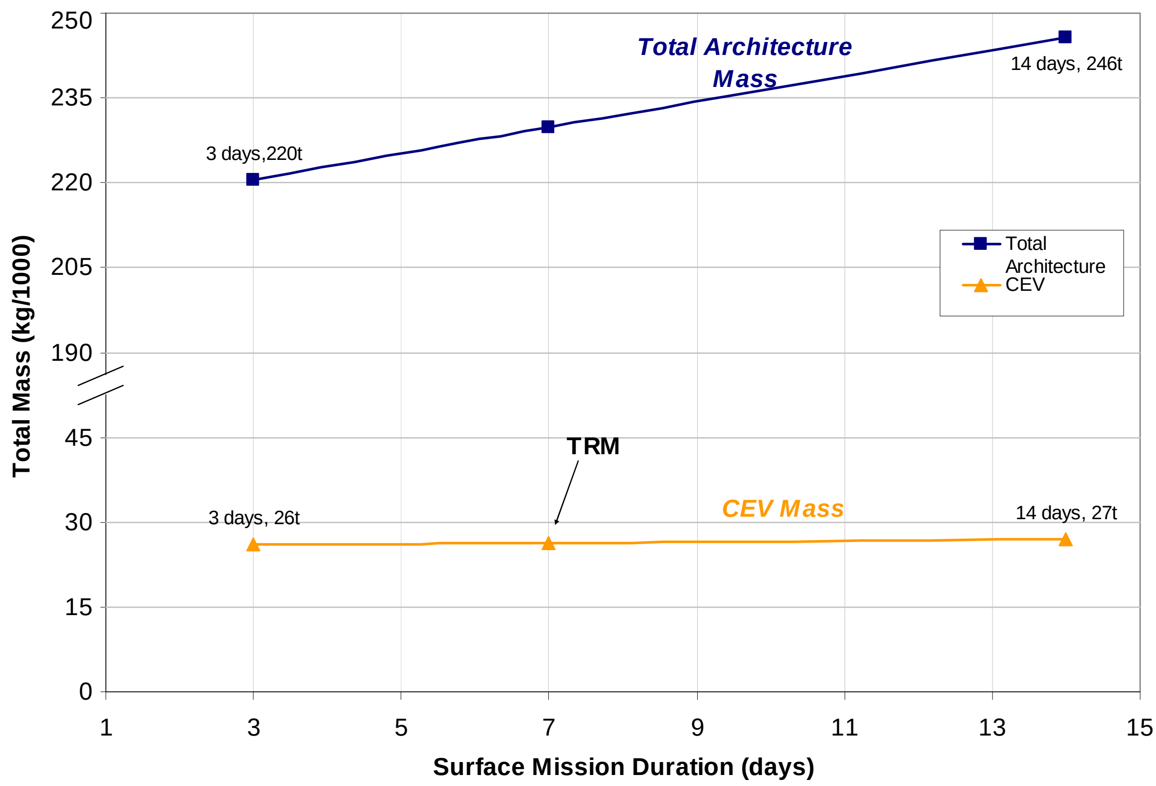

Figure 19.10-1: Effects of Varying the Duration of the Surface Mission on Total Architecture Mass

Extending the mission duration to fourteen days has both positives and negatives. The fourteen-day surface mission would allow a significant increase in the number of EVAs and time for testing/training with systems/ops concepts that could be used on a mission to Mars. The need to deal with increased dust, radiation, and more extreme temperatures could help develop strategies applicable to a mission to Mars.

However, most of the increase in mass/volume to extend the duration of the surface mission would be placed on the Lander and its Earth Departure Stage. These two elements were the most massive elements in the TRM approach and will grow larger in this approach. Although the packaging of these two elements remains an open issue (dependent on their final design), these two elements will drive the launch vehicle mass and volume requirements. Ultimately, this issue will require a trade study that balances Lander design with Habitat functionality.

Additionally, extending the surface duration will require the systems to operate through a greater variety of thermal and lighting extremes. Of primary concerns are landing on the lunar surface, ascent from the lunar surface, and surface EVAs. Landing/Ascent systems and procedures would likely need to be designed to accommodate a much wider array of conditions. Likewise, performing surface EVAs during lunar noon and during lunar night conditions become much more likely. Current inputs from the EVA engineers recommend avoiding lunar noon EVAs near the equator. Given proper lighting, EVAs during lunar night are not seen as difficult.

19.10.1 Operational, Safety, and Mission Assurance Impacts

Increasing the duration of the surface mission to fourteen days would allow time for longer range exploration of the lunar surface (if the appropriate hardware were available), allow human-tended science or technology testing experiments to operate in the lunar environment for a longer period of time, and provide more time for a wider variety of experiments and other activities. Extending the duration of the surface mission would allow a crew to establish a work routine closer to what they would experience on a longer exploration mission. Concepts on delegating greater responsibility to the crew for managing their daily schedule could be explored.

Extending the surface duration would force systems to be designed to operate over a wider variety of surface environments, the most limiting of which seem to be periods of high sun angles during lunar landing and surface operations at equatorial latitudes during lunar noon. If it is too costly or unfeasible to design around these environments, it should be expected that certain operational restrictions may be imposed, placing limitations on periods of surface operations or mission opportunities.

A scenario where lunar surface time is increased to 14 days would have little effect on what hazards and failures could occur, or their respective effects. The effect on reliability prediction and estimation calculations between a 7-day mission and a 14-day mission would result in an increased duty cycle used in calculating Mean Time Between Failure (MTBF). If the probability of a successful mission were to remain the same for the 14 day mission as for the 7 day mission, the cost of the systems and perhaps their complexity would increase to accommodate their longer duty cycle. If the same MTBF were to be used for the 14-day mission as for 7-day mission, a greater probability for required maintenance should be expected. In analyzing the critical events list that was created during the LDRM-2 study, it was felt that the types of critical events would remain the same, but the quantity of the surface events would increase, thus increasing the probability of an event.

19.10.2 Architecture Sizing Impacts

Increasing the surface duration from 7 to 14 days increased the total architecture mass by 6.9% from 229,716 kg to 245, 677 kg. The mass of the largest launch is increased by 8.7% from 94,109 kg to 102,252 kg.

Element	TRM (kg)	14 Day Duration (kg)	Mass Change (kg)	Percentage Change
CEV Crew Module	8,812	8,812	0	0.0
CEV Service Module	17,560	18,087	527	3.0
Ascent Stage	19,906	22,030	2,124	10.7
Descent Stage	22,608	24,700	2,092	9.3
Kick Stage	27,465	29,896	2,431	8.9
Lander EDS (EDS1)	94,109	102,252	8,143	8.7
CEV EDS (EDS2)	39,256	39,900	644	1.6
Largest Launch	94,109	102,252	8,143	8.7
Total	229,716	245,677	15,961	6.9

Table 19.10.2-1: Effects of increasing the surface duration from 7 to 14 days

19.10.3 System Impacts

Extending the surface duration mainly had an impact on the quantity of consumables that were required. The support of a surface dust remediation system was felt to be strongly needed in this scenario. An airlock was chosen, as it was for the 7-day mission, but this is a function for which competing technologies should be examined in the future. Extending the surface duration also increases the attractiveness of solar technologies as opposed to fuel cells, assuming that all operations occur during the day-time; however, this variant continued to use fuel cells due to the lack of data that will need to be generated through a power source trade.

CEV Crew Module's System Mass Changes				
System	**TRM (kg)**	**14 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	822	822	No Change	0.0
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	691	691	No Change	0.0
Other	835	835	No Change	0.0
Growth	1041	1041	No Change	0.0
Non-Cargo	966	966	No Change	0.0
Cargo	1478	1478	No Change	0.0
Non-Propellant	55	55	No Change	0.0
Propellant	64	64	No Change	0.0
Total	8812	8812	No Change	No Change

Table 19.10.3-1: CEV Crew Module's System Mass Changes, Resulting from Increasing the Surface Duration from 7 to 14 Days

CEV Service Module's System Mass Changes				
System	**TRM (kg)**	**14 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	1455	1458	3	0.2
Protection	0	0	No Change	0.0
Propulsion	1408	1435	27	1.9
Power	661	682	21	3.2
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	110	110	No Change	0.0
Other	100	100	No Change	0.0
Growth	747	757	No Change	0.0
Non-Cargo	305	312	7	1.6
Cargo	0	0	No Change	0.0
Non-Propellant	1442	1675	233	16.2
Propellant	11332	11558	226	2.0
Total	17560	18087	527	3.0

Table 19.10.3-2: CEV Service Module's System Mass Changes, Resulting from Extending the Surface Duration from 7 to 14 Days

Lander's Ascent Stage's System Mass Changes				
System	**TRM (kg)**	**14 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	839	853	14	1.7
Protection	73	73	No Change	0.0
Propulsion	1631	1734	103	6.3
Power	813	870	57	7.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	851	962	No Change	0.0
Other	455	455	No Change	0.0
Growth	1080	1137	57	5.3
Non-Cargo	1483	1600	117	7.9
Cargo	227	227	No Change	0.0
Non-Propellant	1014	1542	528	52.1
Propellant	10703	11839	1136	10.6
Total	19906	22030	2124	10.7

Table 19.10.3-3: Lander Ascent Stage's System Mass Changes, Resulting from Extending the Surface Duration from 7 to 14 Days

Lander's Descent Stage's System Mass Changes				
System	**TRM (kg)**	**14 Day (kg)**	**Mass Change (kg)**	**% Change**
Structure	553	569	16	2.9
Protection	50	50	No Change	0.0
Propulsion	1413	1528	115	8.1
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	530	592	62	11.7
Other	708	769	61	8.6
Growth	678	729	51	7.5
Non-Cargo	464	510	46	9.9
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17573	19316	1743	9.9
Total	22608	24700	2092	9.3

Table 19.10.3-4: Lander Descent Stage's System Mass Changes, Resulting from Extending the Surface Duration from 7 to 14 Days

Propulsive Stages' System Mass Changes

Element	System	TRM (kg)	14 Day (kg)	Mass Change (kg)	% Change
Kick Stage	Structure	621	661	40	6.4
	Protection	0	0	No Change	0.0
	Propulsion	1530	1605	75	4.9
	Power	100	100	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	0	0	No Change	0.0
	Environment	0	0	No Change	0.0
	Other	405	405	No Change	0.0
	Growth	531	554	23	4.3
	Non-Cargo	953	1027	74	7.8
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	23323	25543	2220	9.5
	Total	27465	29896	2431	8.9
EDS1	Structure	1972	2098	126	6.4
	Protection	0	0	No Change	0.0
	Propulsion	4361	4625	264	6.1
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	175	175	No Change	0.0
	Environment	105	105	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	1452	1530	78	5.4
	Non-Cargo	3109	3362	253	8.1
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	82289	89712	7423	9.0
	Total	94109	102252	8143	8.7

Propulsive Stages' System Mass Changes					
Element	**System**	**TRM (kg)**	**14 Day (kg)**	**Mass Change (kg)**	**% Change**
EDS2	Structure	932	945	13	1.4
	Protection	0	0	No Change	0.0
	Propulsion	2318	2335	17	0.7
	Power	190	190	No Change	0.0
	Control	0	0	No Change	0.0
	Avionics	171	171	No Change	0.0
	Environment	104	104	No Change	0.0
	Other	455	455	No Change	0.0
	Growth	834	840	6	0.7
	Non-Cargo	1355	1376	21	1.5
	Cargo	0	0	No Change	0.0
	Non-Propellant	0	0	No Change	0.0
	Propellant	32897	33484	587	1.8
	Total	39256	39900	644	1.6

Table 19.10.3-5: Propulsive Stages' System Mass Changes, Resulting from Extending the Surface Duration from 7 to 14 Days

Active Thermal Control System

Extending the lunar surface duration from 7 to 14 days would not have an impact on the CEV's ATCS. However, it would have an impact on surface systems due to the increase in the range over which the surface elements' thermal control systems would be required to operate. Although this may not pose a great threat to the Lander, due to its ability to deploy space-facing radiators with the proper surface coatings, this may pose a technological challenge to the surface suits.

ATCS consumables may be affected in each surface element, depending on their design. For example, the Lander's ATCS system currently employs an evaporator that rejects approximately 1.5 kW of heat throughout the surface portion of the mission. This equates to approximately 52.7 kg of water per day. This mass may be increased if the sink temperature of the environment were to increase.

Avionics System

Extending the lunar surface duration from 7 to 14 days will have no effect on the avionics systems.

Descent and Landing System

Extending the lunar surface duration from 7 to 14 days will have no effect on the CEV's descent and landing systems. However, it may have an impact on the types of systems employed for the

Lander. If it is desired to operate on the lunar surface while avoiding lunar noon, landing on the lunar surface may be forced to occur during the lunar afternoon or evening. If this were the case, the ability of the crew to perform landings based on visual cues on the lunar surface may not be possible. Therefore, highly reliable automated landing systems or computer-aided visualization equipment may be needed.

Environmental Control and Life Support System

Extending the lunar surface duration from 7 to 14 days will begin to drive the mission towards adding a surface habitat separate from the Lander. Life support and habitation equipment consumables (gases, food, habitation equipment, water etc.) will increase to the point that mass and volumes required for them may be prohibitive for a Lander. With longer surface stays it is likely the power system may trade away from fuel cells and the ECLS system would lose the benefit of the water that the fuels cells provide. Substantial amounts of potable water would then need to be added as a stored consumable and its potability would need to be maintained for the longer durations. The increased number of EVAs that would be preformed during a longer surface mission would translate into an increased number of airlock repressurizations and thus the need to store more makeup gasses.

Extra-Vehicular Activity System

Extending the lunar surface duration from 7 to 14 days would bring with it an increased number of EVAs. In this case, a dust mitigation strategy would have to be employed. An airlock was carried in the Lander's design to satisfy this requirement for this design iteration, but alternatives should be examined more closely in the future. An increased number of EVAs would increase the number of consumables that are needed for both the airlock's operation (the maximum amount of EVAs are still below the number that trades for carrying a depress pump) and the EVA suits. Additionally, EVA suits and maintenance procedures would have to be designed to accommodate the greater number of EVAs and their associated wear and tear. It should be noted that although increasing the number of EVAs requires a greater degree of reliability in the system designs and perhaps a greater emphasis on maintenance and repair, it tends to push system and operational concepts towards what would be used during longer duration lunar missions and missions to Mars.

Human Factors and Habitability System

Extending the lunar surface duration from 7 to 14 days will not change any of the Habitation system hardware components, but will increase all consumables that are sized by duration (e.g. . food, food prep supplies, waste collection supplies, hygiene supplies, clothing, housekeeping supplies, operational supplies, medical consumables). Additionally, increasing the surface stay may require an increase in habitable volume required for the Lander, but better guidance on surface element habitable volume and floor area is needed.

Power System

Extending the lunar surface duration from 7 to 14 days under the current power system concept will increase the quantity of consumables that are needed for the power source for the Lander. The increase in calculated O2 and H2 in this study needed for power generation was 500.7 kg (53%). However, it was felt during the course of the study that other power source technologies such as solar arrays would become more attractive if the surface duration was extended. A power source trade is an item that should be performed during future iterations of this study. Extending the surface duration would have little effect on the other Lander's power systems, such as the power management and distribution system, and would have little effect on the CEV's power systems.

Propulsion System

Extending the lunar surface duration from 7 to 14 days would have significant impacts on the mass of propellants and propellant tanks that are needed for the Lander and its associated Earth Departure Stage, but will have little effect on the system concept or technology selections. The increase in propellant mass was calculated as 1,743 kg in the descent stage (an increase of 9.9% of the total descent stage's propellant mass), 1,136 kg in the ascent stage (an increase of 10.6% of the total ascent stage's propellant mass), 7,423 kg in EDS1 (an increase of 9.0% of EDS1's total propellant mass), and 587 kg in EDS2 (an increase of 1.8% of EDS2's total propellant mass). The increase in other propulsion system hardware, including propellant tanks, was calculated as 115 kg in the descent stage (an increase of 8.1% of the total descent stage's propulsion system's hardware mass), 103 kg in the ascent stage (an increase of 6.3% of the total ascent stage's propulsion system's hardware mass), 264 kg in the EDS1 (an increase of 6.1% of EDS1's total propulsion system's hardware mass), and 17 kg in EDS2 (an increase of 0.7% of EDS2's total propulsion system's hardware mass).

Structures and Thermal Protection Systems

Extending the lunar surface duration from 7 to 14 days would have little effect on the CEV structures and TPS. The Lander's structural masses would increase slightly in order to support the increased masses of all the other systems and the proposed increase in pressurized volume. As can be seen in the mass breakout tables, this structural mass increase was minimal: 14 kg (1.7% of the total structural mass).

19.10.4 Impact on Mars Preparations

Operations, Safety, and Mission Assurance

Extending the lunar surface duration from 7 to 14 days has several advantages that apply to preparation for missions to Mars. This mission is long enough to simulate the real-time mission planning, execution roles, the group dynamics, and leadership structure of a four-person Mars crew. This variation allows a greater amount of time for testing the effectiveness of dust control techniques, surface communication systems, navigation systems, and for significant surface exploration with and without rovers. Operationally, this is a better option for Mars mission preparation relative to the 7-day mission. When compared to the eventual results of six-person missions, the lessons learned could help to validate or invalidate earlier conclusions about the relative merits of four vs. six-person Mars crews.

Due to the increased duty cycle of all systems in the architecture, this variation would tend to provide a greater amount of data that would be useful in understanding system weaknesses. Due to the increased duty cycle, the system design philosophy may tend towards systems that are designed to withstand longer-term operation, which would be needed during missions to Mars. However, there is still a significant difference between Mars mission duration and a 14-day lunar mission duration. Therefore, there are most likely significant differences in the pieces of surface infrastructure and requirements that would be employed in a mission to Mars.

Active Thermal Control System

There would be no significant effect of extending the lunar surface duration from 7 to 14 days on the development of systems for a Mars mission.

Avionics System

There would be no significant effect of extending the lunar surface duration from 7 to 14 days on the development of systems for a Mars mission.

Descent and Landing System

There would be no significant effect of extending the lunar surface duration from 7 to 14 days on the development of systems for a Mars mission.

Environmental Control and Life Support System

Extending the lunar surface duration from 7 to 14 days would have little effect on the preparation for missions to Mars in this scenario. However if longer lunar surface stays also drove the decision to include a separate habitat in the architecture, ECLSS subsystems with more applicability to Mars missions (water recycling, ISRU, small vegetation production units) could be tested.

Extra-Vehicular Activity System

Extending the lunar surface duration from 7 to 14 days increases the number of EVAs that could be attempted. Therefore, system design reliability requirements may be more reflective of the type of requirements that would be needed to design systems for missions to Mars. Additonally, the operational concepts used to perform EVAs may tend to look more similar to those used on Mars, due to the extended duration. However, it should be noted that two of the major environmental factors that affect EVA systems, dust (or regolith) and environmental temperature, are significantly different between the Moon and Mars.

Human Factors and Habitability System

There would be no significant effect of extending the lunar surface duration from 7 to 14 days on the development of systems for a Mars mission. However, extending the surface duration may allow for the ability to test new Habitation System hardware (e.g. laundry facilities, full body cleansing facilities) in a "testbed" setting for a future Mars mission. However, these advanced technologies are not required for the 14-day Lunar mission itself.

Power System

In this particular scenario, given the current power system concept, there would be no significant effect of extending the lunar surface duration from 7 to 14 days on the development of systems for a Mars mission. However, after performing the recommended power source trade, extending the surface duration may provide greater insight into the technologies and their associated operation for a mission to Mars. In order to explore this issue more completely a power source trade would have to be performed for both the lunar and Mars surface missions.

Propulsion System

There would be no significant effect of extending the lunar surface duration from 7 to 14 days on the development of systems for a Mars mission.

Structures and Thermal Protection Systems

There would be no significant effect of extending the lunar surface duration from 7 to 14 days on the development of systems for a Mars mission.

19.11 Elimination of Contingency CEV EVA Requirement

In the lunar Trade Reference Missions presented here, we have assumed that the crew might need to carry out spacewalks (extravehicular activity, EVA) from the Crew Exploration Vehicle to recover from a small set of specific failure cases. These "contingency EVAs" would be accomplished during orbital flight, and share little in common with the planned lunar surface EVAs originating from the lander.

Although we have designed our notional systems to be compatible with contingency EVA capability, neither the design reference mission nor any of its variants specifically require it. It therefore makes sense to weigh the advantages and disadvantages of keeping or dropping the option. Although a quantitative cost-risk-benefit analysis is beyond the scope of this work, it is possible to lay out the considerations on both sides of the issue and make a preliminary decision.

<u>What Are the Benefits of Contingency EVA Capability?</u>

The benefit of contingency EVA capability is that it provides an additional, independent layer of redundancy (or, equivalently, an extra leg of fault tolerance) for certain systems, some of which could endanger the mission or the crew if they fail. Illustrative examples from existing spacecraft include:

1. Emergency crew transfer. After the Columbia accident, it was determined that a rescue mission might have been able to save the crew of Columbia in orbit, had the extent of the damage to its wing leading edge been known early enough. Such a mission could succeed only if there was a means of transferring crewmembers from ship to ship across the intervening space. In some lunar architectures, an analogous capability could potentially save the mission or the crew by giving them access to other flight elements (e.g., the CEV or the Lander) in the event that the docking mechanisms cannot be engaged or there is a problem with the pressure seals or pressurized tunnel. It could also enable rescue flights where such a tunnel might not exist.

2. Emergency repairs. The International Space Station has, and the Space Shuttle is developing, plans for spacewalking astronauts to inspect, repair, and replace critical equipment on the outside of each vehicle. An analogous capability may be desirable for lunar missions, especially given the long dwell times of some flight elements in space.

3. Failed Dock or Undock. The Shuttle has contingency plans in place for spacewalking astronauts to manually unbolt the Orbiter from the Space Station in the case that a mating adapter failure prevents undocking. Similar EVA capability may be desirable for the lunar missions discussed here, especially given the large number of critical spacecraft separation events in each scenario. The successful crew transfer from the Lander to the CEV on the return leg of a lunar mission is particularly critical to crew safety.

4. Closing the Payload Bay Doors. The Shuttle has contingency plans in place for spacewalking astronauts to manually shut the payload bay doors in case the mechanisms that normally close them should fail. Such a malfunction would effectively strand the Orbiter in space, unable to reenter and land. Similar EVA capability may be desirable for lunar missions if there are critical deployable mechanisms outside the crew compartment.

What Are the Costs of Contingency EVA Capability?

For the CEV, the ability to carry out a contingency EVA demands the following:

1. EVA-capable suits. But it is likely that the suits worn by the crew during ascent and entry will be vacuum-rated anyway, as a protection against cabin leaks. Adding true EVA capability (air and cooling for hours rather than minutes, gloves designed for extended work in vacuum, etc.) to launch-and-entry suits will require additional effort. Alternatively, the crew could use lunar surface suits. In this case, the suits would have to be carried in the CEV (rather than sent ahead in the lander) and the CEV must have enough internal volume to allow the crew to don and doff those suits.

2. Avionics that can tolerate vacuum. This generally means that critical avionics must reject waste heat to coldplates rather than flowing air. But it is likely that the avionics will be coldplated anyway, to protect against cabin leaks. An alternative to coldplated avionics is an airlock, so that the CEV cabin need not be depressurized for an EVA. The weight and complexity of an airlock, however, are likely to far exceed its usefulness.

3. A hatch large enough for suited crewmembers to pass through. But if the crew uses launch-and-entry suits augmented for EVA, the hatch must readily accommodate them anyway, to permit fast emergency egress in a pad abort.

4. Resources (e.g., air and water) for the EVA, either from CEV via umbilicals (complicating crew transfer and rescue scenarios), or from backpacks attached to the suits. But the quantity of resources needed will not differ much from those needed by a crew inside the CEV for the same duration. Additional air will be needed to repressurize the crew compartment after the EVA.

5. EVA handrails and other mobility aids along outside translation paths. This will require minor additional mass and design work.

What Are the Benefits of Deleting Contingency EVA Capability?

Because most of the major design features needed for contingency EVA capability are likely to be driven by requirements other than EVA, deleting that capability will result in only minor savings in weight and complexity. As noted above, the savings will be realized by omitting suit augmentation, some air reserves, and EVA mobility aids.

What Are the Costs of Deleting Contingency EVA Capability?

Removing contingency EVA capability deletes one level of failure tolerance from functions such as crew transfer, external equipment replacement or maintenance, docking and separation, and external deployables. If those functions are critical to mission success or flight safety then that layer of fault tolerance must be provided by another method, presumably equipment redundancy. This will add mass (in the case of docking and separation mechanisms, possibly substantial mass) and complexity to the systems that provide those functions.

Although a final answer must await a full, quantitative cost-risk-benefit analysis, it appears that the cost of retaining contingency EVA capability is low. This is especially true in light of the capabilities the system is likely to need in order to meet other requirements. Furthermore, the cost to add reliability to some functions that currently use EVA as a layer of fault tolerance seems potentially high in comparison to the minor benefits to be gained if contingency EVA capability is deleted. Until a comprehensive analysis is available, it appears that the best and safest approach is to retain contingency CEV EVA capability for the missions we treat in this report.

19.12 Recommended Cabin Design Pressure and Mass Effects

The selection of a cabin atmosphere includes defining the total cabin pressure and the chemical composition of the atmosphere. The dominant factors in choosing a cabin atmosphere were the flammability of materials, ensuring healthy oxygen levels, managing the risk of decompression sickness, and providing sufficient mobility for extra-vehicular activities. Numerous studies have been performed to address this issue in the past. Therefore, a literature review supplemented by discussions with key technical experts and a few minor calculations were sufficient to make a selection. This trade study resulted in the selection a cabin atmosphere of 9.5 +/- 0.5 psia that was made of oxygen and nitrogen. Oxygen levels could range from 27 – 30%.

19.12.1 Purpose and Description

The purpose of this trade study was to define an optimum cabin pressure and chemical make up. Cabin pressure and composition have multiple impacts on a several different aspects of a vehicle. The selection of an optimum cabin atmosphere must balance the following major vehicle design and mission impacts:

- Flammability of materials
- Appropriate oxygen levels to ensure crew health
- Risk of decompression sickness (DCS) during Extra Vehicular Activities (EVAs)
- Spacesuit Design

In addition to these design drivers, the following minor impacts deserve consideration:

- Environmental Control and Life Support System (ECLSS) performance
- Active Thermal Control System (ATCS) performance
- Vehicle Structure
- Mass of gases required to make up the cabin atmosphere

19.12.2 Evaluation

Flammability of Materials

Flammability of materials is a complex characteristic to quantify. Oxygen concentration, total pressure, and gravity all effect how a material combusts and burns. Materials are generally more flammable as the oxygen concentration increases. A wide range of materials are currently certified for use in a 30% O_2 environment. This is necessary because the Space Shuttle operates at 10.2 psia and 30% O_2 (3.06 psia partial pressure of oxygen) [1] for periods prior to EVAs. This was chosen as a boundary for maximum oxygen concentration. In order to expand this boundary, additional flammability research must be performed and new material certification programs must be implemented.

Appropriate Oxygen Levels to Ensure Crew Health

Humans must maintain a certain oxygen partial pressure in their lungs. A hypoxic condition is one where the partial pressure is too low. Oxygen becomes toxic to humans if the partial pressure in the lungs is too high. Figure 19.12-1 shows combinations of total pressure and oxygen concentration that define hypoxic, normal oxygen levels (normoxic), and oxygen toxicity values. It should be noted that humans live at elevations on Earth greater than 10,000 ft above sea-level. When combined with the Earth's 21% oxygen concentration, this produces hypoxic conditions according to this NASA standard.

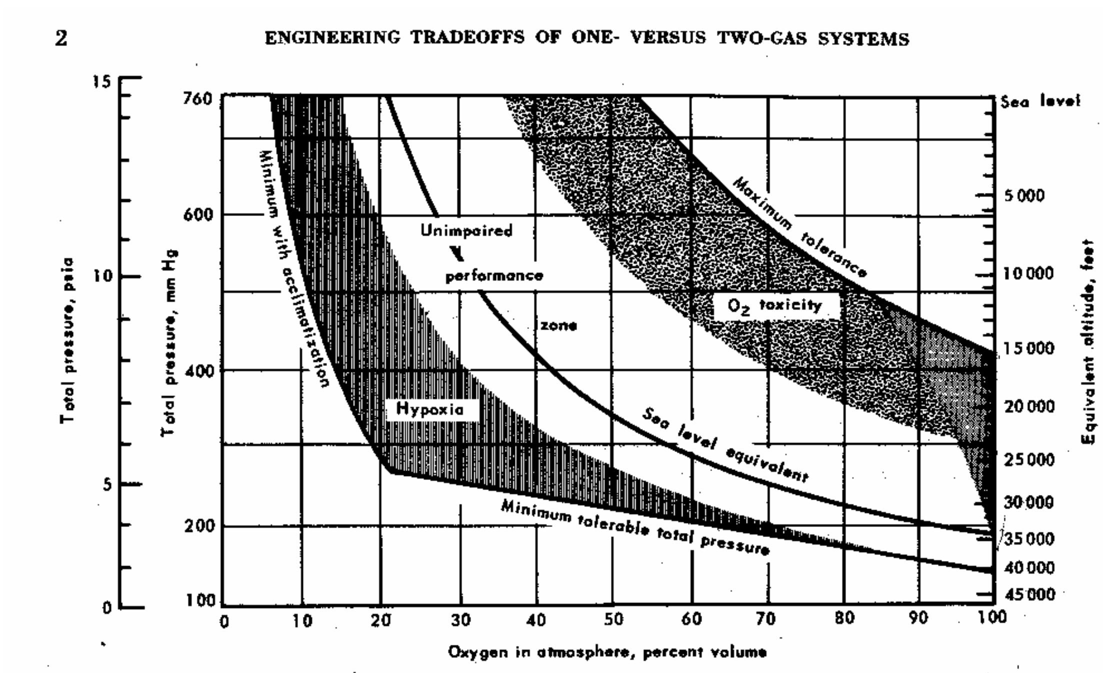

Figure 19.12.2-1: Limits for Oxygen Concentrations (Webb, 1964)

Risk of Decompression Sickness

"Decompression sickness (DCS) takes place when the inert gas (generally nitrogen) that normally is dissolved in body tissues at one pressure forms a gas phase ("bubbles") at a lower ambient pressure, when the tissues become supersaturated (Powell, 1993)." In other words, when an astronaut dons a spacesuit, his or her body undergoes a decrease in ambient pressure. This pressure change can allow gas bubbles to form in the astronaut's body. These bubbles cause symptoms that range from minor discomforts, classified as Type I DCS, to neurological disorders, Type II DCS. Common Type I symptoms include joint pain, pain in the limbs, and skin manifestations. Type II symptoms include neurological problems, visual disturbances, sensory/motor problems, problems with speech or memory, headaches, inappropriate fatigue, limb parasthesias, spinal cord involvement, and chokes (Bendrick et al., 1996 and Pickard 2003).

The “R” ratio is often used to characterize the risk of DCS given a change in ambient pressure. For the current purpose, R is defined by the following equation. PP_{N2} is the partial pressure of Nitrogen in the vehicle / habitat and P_{SUIT} is the space suit pressure, assuming it is 100% oxygen.

$$R = \frac{PP_{N2}}{P_{SUIT}}$$

Significant amounts of ground test data have been collected in an effort to relate the R value to a risk of DCS. However, ground test data and flight experience have not correlated well. Ground test data generally shows more incidents of DCS than the on-orbit experience. Limited lower body motion, microgravity, and accurate symptom reporting are some issues that could account for these differences. First, the physics of gases coming out of solution, bubble nucleation, bubble growth, and bubble dynamics are all gravity dependent phenomena. In addition, the microgravity environment of space limits lower body activity. Conkin and Powell (2001) showed that limiting lower body activity decreased the risk of DCS. These results would apply to the CEV, but might not apply to EVA activity on the Lunar surface. The Moon has low levels of gravity and astronauts would be required to do a lot of walking. Lastly, Bendrick et al. (1996) found that U-2 fighter pilots did not accurately report cases of DCS during active duty. An anonymous survey given to both active and retired pilots showed a much higher incidence rate than had been previously reported. DCS reporting by astronauts might also be skewed. Current space suits are often uncomfortable and could mask minor DCS symptoms like joint pain. Astronauts might strongly desire to complete their high profile tasks and not report minor symptoms for fear of being “grounded,” or having to miss an EVA. These factors must be considered when evaluating the risk associated with specific R values. A Lunar surface mission will likely consist of multiple EVAs. Multiple EVAs also decrease DCS risk according to the EVA-Integrated Product Team (IPT) (Fitzpatrick 2004).

Prebreathe, or breathing 100% O2, for a prescribed period before an EVA decreases the risk of DCS. Currently, the Space Shuttle uses a 10.2 psia, 26.5 % O2 atmosphere and requires a prebreathe duration of approximately 40 minutes. Currently, the EVA-IPT would recommend a longer prebreathe than specified in the flight rules for the Space Shuttle. At this time it is thought that EVA preparations could also take approximately 40 minutes. This period of time could be combined with prebreathe operations to minimize any impact to the mission timeline. Prebreathe protocols can be accelerated by exercising while breathing oxygen, but between 30 and 60 minutes are required to denitrify an astronaut’s brain and spinal cord in order to prevent severe Type II DCS. These data lead to using a one-hour prebreathe minimum as another boundary to the cabin atmosphere operational envelope. The EVA-IPT believed that a one-hour prebreathe for a 9.5 psia and 30% oxygen environment was a realistic goal.

Spacesuit Design

Spacesuit pressure and cabin pressure are linked by the risk of DCS. It is desirable to minimize the spacesuit pressure because it increases the astronaut dexterity while decreasing suit mass. The combined effect of these trends is to make the spacesuit easier and less tiring to operate. Lower pressures increase dexterity by decreasing the amount of work required for an astronaut to move. When an astronaut flexes their fingers or bends their knees the pressure vessel of the suit generally changes shape and volume. Thermodynamically, the change in volume of a gas at a given pressure is associated with energy. This is commonly called expansion or compression work (Moran and Shapiro, 1992).

$$W = \int_{V_1}^{V_2} p * dV$$

Progress has been made to develop constant volume joints for space suits that require little energy. However, glove fatigue resulting from finger intensive tasks is still particularly sensitive to suit pressure.

In addition to an increase in mobility, space suit mass decreases as suit pressure decreases. This is due to mass savings in the pressure vessel structure and in the Portable Life Support System (PLSS). Lighter spacesuits will also decrease astronaut fatigue allowing for improved crew performance and longer EVAs.

Impacts to ECLSS and ATCS

ECLSS and ATCS will be impacted by the cabin pressure selection because the density of the gas flowing through air revitalization or heat transfer equipment will decrease at lower pressures. Lower density gas decreases the mass flow rate through these devices and can decrease performance. In order to maintain the same mass flow rate through these devices, systems must be designed with larger ducts or must operate at increase fan speeds. These changes either increase system volume or power, respectively.

Impacts to Structures

Decreasing the cabin pressure decreases the stresses in the vehicle pressure vessel and potentially enables structural mass savings. It should be noted that a pressure vessel optimized for a lower operating pressure could limit a vehicle's ability to dock to the ISS or other higher pressure spacecraft. Figure 19.12.2-2 shows an estimated structural mass versus cabin pressure correlation for a CEV type vehicle. This estimate linearly scales the structure mass versus pressure. As the vehicle structure is optimized to take advantage of decreased cabin pressure, a structural design limit may be encountered due to minimum gage material thickness considerations. The data presented in the following figure does not take this into account.

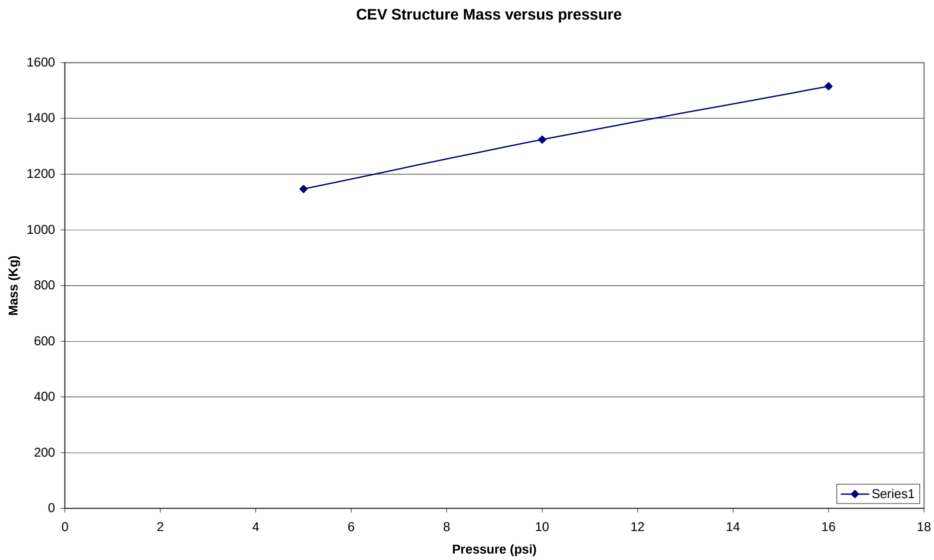

Figure 19.12.2-2: CEV Cabin Pressure Vessel Mass as a Function of Design Pressure

Make-up Gas Requirements

Lower pressure atmospheres require a smaller mass of gas. The baseline CEV has a pressurized volume of 22 m^3 and is required to support up to two cabin repressurization cycles during a mission. A lower operating pressure would provide some minor additional mass savings on the order of the tens of kilograms.

19.12.3 Tools Used for Evaluation

Numerous studies have been performed on this subject. The References section contains several good sources of information. Lange, et al. (2004) provide a thorough compilation of several references. This presentation was used as a primary reference for this trade study. Several discussions were held with JSC personnel that have expertise in decompression sickness, materials for space vehicles, EVA technologies, EVA operations, Environmental Control and Life Support Systems (ECLSS), Active Thermal Control Systems (ATCS), and structures. The following experts can serve as points of contact for their respective areas of expertise.

Name	Area of Expertise
Dr. Johnny Conkin Dr. Dan Fitzpatrick	Decompression Sickness
Rajib Dasgupta	Materials
Joe Kosmo	EVA Technologies
Robert Trevino	EVA Operations
Dr. Kevin Lange	ECLSS
Mike Ewert	ATCS
Gregg Edeen	Structures

19.12.4 Affected Requirements

For LDRM-2, the CEV has a requirement only for contingency EVA. Since EVAs are not part of the nominal mission operation of the vehicle, a higher risk of decompression sickness can be tolerated. However, a requirement to perform multiple EVAs applies to the Lunar Lander. EVAs will be one of the primary functions that this vehicle will be designed around. The Lunar Lander design must minimize the risk of DCS. Frequent EVAs from the Lander also amplify the need to minimize prebreathe duration. All interested parties would like to minimize the amount of time spent performing prebreathe procedures and maximize the time astronauts can spend exploring the surface of the moon.

19.12.5 Results and Recommendations

This trade study resulted in the selection of a nitrogen/oxygen cabin atmosphere of 9.5 +/- 0.5 psia with oxygen levels ranging from 27 – 30 %.

The final recommendation for this study was to use a 9.5 +/- 0.5 psia, 27 – 30 % oxygen atmosphere. Lange, et al. recommended a slightly lower pressure and was generally accepted by the multiple disciplines that this trade incorporates.

Work needs to continue to understand hypoxic limits, DCS, and flammability of materials to enable missions that require no prebreathe procedure prior to EVA.

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20.0 Appendix

Section	Title	Author
20.1	*Propulsion Subsystem Technology Report*	Eric Hurlbert
20.2	*Power Subsystem Technology Report*	Karla Bradley
20.3	*Environmental Control and Life Support System (ECLSS) Technology Report*	Kathy Daues
20.4	*Habitation System Technology Report*	Susan Baggerman
20.5	*Active Thermal Control System Technology Report*	David Westheimer
20.6	*Extravehicular Activity System (EVAS) Technology Report*	Robert Trevino
20.7	*Avionics Subsystem Technology Report*	Coy Kouba David Jih Helen Neighbors
20.8	*GN&C Subsystem Technology Report*	Thomas Moody Brian Rishikof David Strack Tim Crain Howard Hu
20.9	*Communications and Tracking Subsystem Technology Report*	Laura Hood
20.10	*Structures Technology Report*	Gregg Edeen
20.11	*Passive Thermal Control System Technology Report*	Steve Rickman
20.12	*Thermal Protection System Technology Report*	Chris Madden
20.13	*Advanced Mating System Technology Report*	James Lewis
20.14	*Thermal Environment for a Lunar Mission*	Steve Rickman
20.15	*Space Radiation Protection*	Francis Cucinotta
20.16	*Micrometeoroid and Orbital Debris (MMOD) Technology Assessment*	Eric Christiansen
20.17	*Risks and Hazards Assessments*	Jan Railsback Randy Rust Bryan Fuqua Clint Thornton

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20.1 Propulsion Subsystem Technology Report

Eric Hurlbert, NASA/JSC/EP – Energy Systems Division

20.1.1 CEV Capsule RCS

20.1.1.1 CEV Capsule RCS Subsystem Description

The primary function of the CEV capsule RCS is attitude control (primarily roll control) during Earth atmospheric re-entry at the end of the lunar mission. A ballistic re-entry mode (slow roll used to null the lift vector) is assumed as a backup in the event of a loss of control. The capsule RCS system is assumed to remain inactive until the re-entry phase of the mission. It is recommended that the CEV capsule RCS remain un-wetted and pristine for Earth re-entry, as was done in the Apollo Program.

Twelve thrusters are required for fail operational/fail safe redundancy. The assumed thrust required is 220N (50 lbf) per thruster with a total delta-V of 10 m/s (32.8 ft/s). Figure 1 shows the thruster locations. Fail safe is provided by a ballistic re-entry.

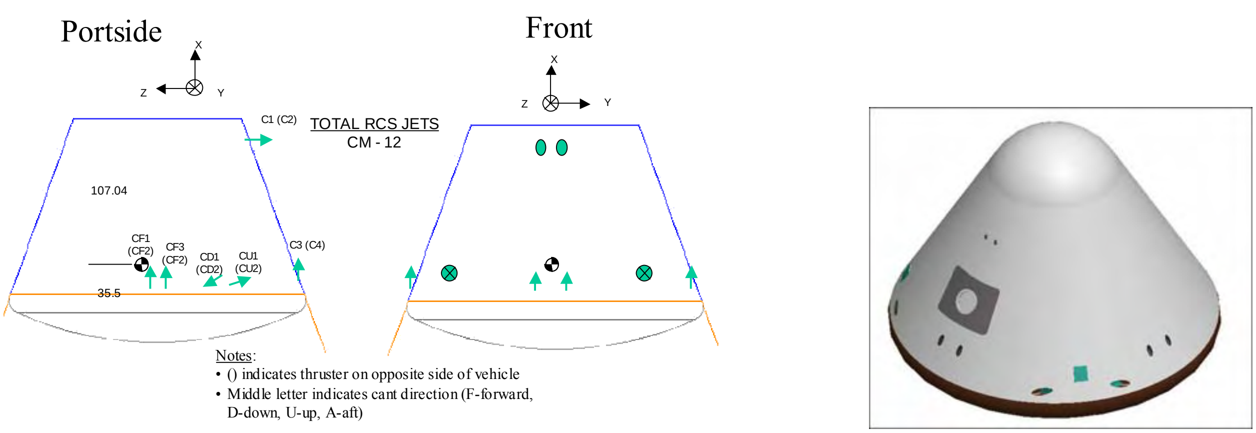


Figure 20.1.1.1-1: RCS Jet Configuration

The propulsion system will require interfaces to other capsule subsystems including electrical power, guidance, navigation and control, data management, vehicle health management, etc. Since this system is only used for re-entry, a passive system (minimal power, crew interaction, etc) is desired.

Consideration was also given to the dual use of the capsule RCS propellant for other functions, such as air flotation bag inflation or ECLSS consumables.

20.1.1.2 Capsule RCS Technology Options

Section 19.1 Alternate Propellants discusses the details of the capsule RCS trades. The three most promising technologies for a CEV capsule RCS are Tridyne, gaseous oxygen/ethanol, or nitrous oxide. A mono-propellant system can provide a simple, safe, and cost effective RCS propulsion system that can support a 2008 demo of a CEV capsule. The use of a more complex bi-propellant system, such as MMH/NTO, for the CEV capsule RCS will add significant development risk and operational cost.

The propellants were compared on the basis of mass, volume/packaging efficiency, power, number of components, and hazards. The number of components affects cost and reliability. Reliability can also be affected by propellant characteristics such as stability, corrosiveness, and residues. On the basis of this trade study the warm gas Tridyne mixture was selected as the best choice. This monopropellant provides acceptable mass and volume and uses no power for storage or catalyst bed pre-heat. The Tridyne system also has the fewest number of components and the propellant is stable, non-toxic, non-flammable, and non-explosive and can also be used to inflate flotation bags. Although the volume is higher then liquid propellant options, it can offer good packaging efficiency because it has only one storage commodity, whereas a bipropellant like MMH/NTO requires at least four tanks - oxidizer, fuel, and two gaseous helium pressurant tanks – along with the associated secondary structure and plumbing. Using a gas propellant also simplifies acquisition and gauging, thus eliminating a number of tank components.

The X-38 Program completed development of a 25 lbf cold gas RCS system using GN2. Many of these components are applicable to a Tridyne system. All of the components associated with the Tridyne system are high TRL items - valves, catalyst beds, tanks and regulators. The Tridyne warm gas propulsion system is currently at TRL 5. The primary work required to reach TRL 6 is the assembly and testing of a prototype system in a space environment. All of the RCS options listed above have been or are currently being developed. The development of the Tridyne system by Rocketdyne began during the Apollo program. Recently a 10 lbf Tridyne thruster was tested at WSTF.

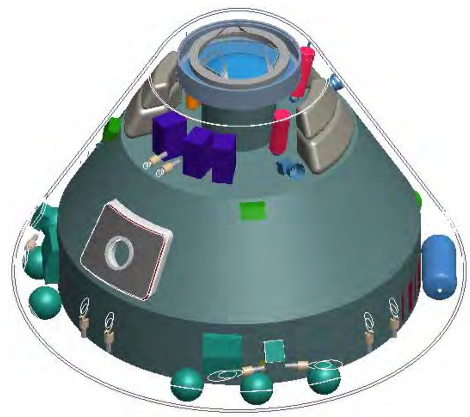
The other candidates are nitrous oxide and gaseous oxygen with ethanol. Nitrous oxide has been tested in monopropellant and bipropellant modes. Several gaseous oxygen and ethanol thrusters (25 lbf, 600 lbf, 870 lbf) have been tested since 1984. These technologies are promising for small vehicle (ie capsule) because of its higher performance potential, passive storage, and ECLSS integration.

Hydroxyl Ammonium Nitrate (HAN) based propellants offer performance close to that of hydrazine without the toxicity. Hydrogen peroxide was used on the Mercury capsules. In the Mercury Program, hydrogen peroxide was not entirely successful because of thruster failures due to rapid propellant decomposition. Hydrogen peroxide is an unstable oxidizer that is not very tolerant of contamination and heating that can cause thermal runaway. The Soyuz, which uses hydrogen peroxide, has a life limit on-orbit due to the decomposition of hydrogen peroxide. Monopropellant hydrazine and MMH/NTO are fallback options for RCS. However, the use of hydrazine or MMH/NTO will impact the safety and cost of recovery operations.

20.1.1.3 CEV Capsule RCS Design Approach

The recommended design approach for the CEV capsule RCS is to use Tridyne gas. This system is essentially the same as a cold gas system, except that Shell 405 (or comparable replacement) catalyst beds are used at the thruster. The system and catalyst beds do not need to be heated, so this system is entirely passive except for instrumentation. The gas is stored in 5.67 ft^3 of volume at 4500 psia. Two 500 psia regulators provide a redundant supply to the manifolds. Four manifolds with isolation valves are used for redundancy. Each thruster consists of a single valve, thermal isolator, catalyst bed, and nozzle. The number of GHe bottles is a function of packaging. The figure 20.1.1.3-1 shows six spherical tanks.

Volume of tank required					
Volume	0.154668 m3				
Volume	5.462854 ft3				
Component Masses					
Mass of Tridyne Tank		PV/Mass_tank =	270.9343 psia*ft3/lbm		
Mass_tank =	100.8151 lbm	conservative value based on			
Mass_tank =	45.74189 kg	shuttle RCS tank (4500 psia,			
	unit mass	qty	total mass		
thruster	2 kg	12	24 kg	based on 1	
regulator	2 kg	3	6		
iso-valves	2 kg	6	12		
gas lines	1 kg/m	15	15		
accumulator	2 kg	3	6		
supports mass (assume some %)					
Prop System Total Dry Mass		108.7419 kg			
Useable Propellant		55 kg			
Unusable		10.73397 kg			
	total	174.4759 kg			



Command Module RCS (12)

	+Roll	-Roll	+Pitch	-Pitch	+Yaw	-Yaw
C1				X		
C2				X		
C3			X			
C4			X			
CF1						X
CF2					X	
CF3						X
CF4					X	
CD1	X					
CD2		X				
CU1		X				
CU2	X					

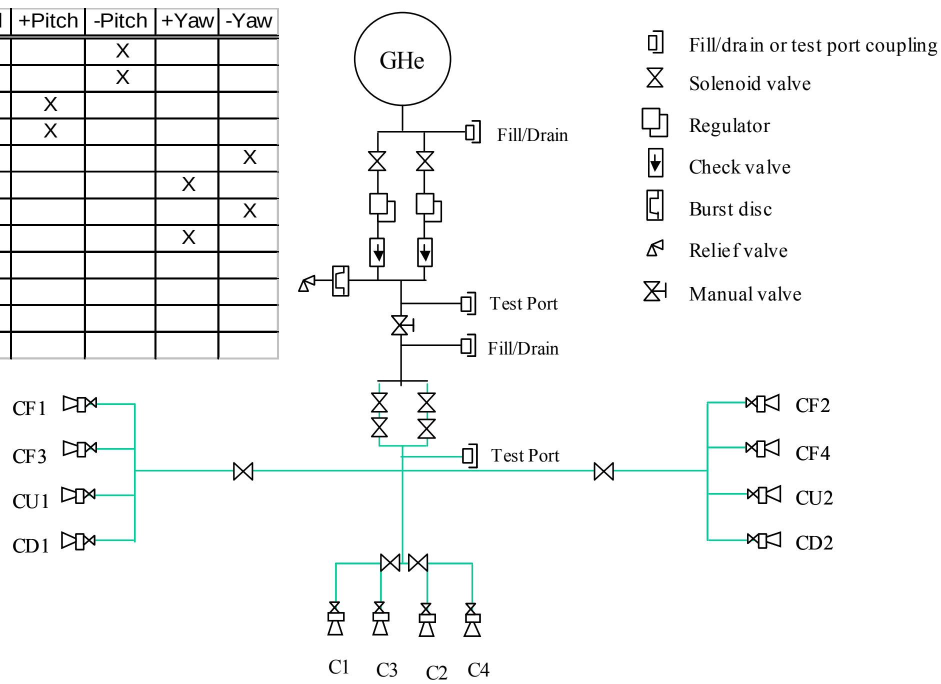


Figure 20.1.1.3-1: CEV Entry Vehicle RCS Concept

It is recommended that the CEV capsule RCS remain unwetted and pristine for Earth re-entry, as was done in the Apollo Program. If, however, the capsule RCS is employed during other mission phases (e.g., aerocapture) resulting in a significant increase in delta-V, then a higher performing, but more complex monopropellant or bipropellant RCS system may be necessary. For smaller packaging with some increase in complexity, a supercritical nitrous oxide system would be the next choice. If a bipropellant system is required for performance, then GO2/ethanol would be the next best choice. The GO2 propellant tanks can be sized to accommodate ECLSS oxygen storage requirements, and do not require any heater power for storage. Ethanol is a safe, non-toxic fluid that can be stored easily in a metal bellows, bladder, or diaphragm tank, and also does not require any heater power for storage.

20.1.2 CEV Service Module and Lander OMS/RCS

20.1.2.1 CEV Service Module OMS/RCS Subsystem Description

The Service Module is required to perform the L1 insertion and departure burns as well as any maneuvers associated with earth return (e.g., midcourse corrections) or service module disposal. The Service Module OMS is required to provide 1901 m/s (6237 ft/s) of delta-V and the Service Module RCS is required to provide 75 m/s (246 ft/s). The lifetime on orbit is approximately 60 days. The majority of time will be spent at L1 in a dormant but ready state.

The Service Module OMS and RCS are designed to provide fail operational/fail safe redundancy.

The propulsion system will require interfaces to other subsystems including electrical power, guidance, navigation and control, active thermal control, data management, vehicle health management, etc. The system will be designed to minimize power usage during the L1 station keeping and can share resources with life support, thermal, and power.

The OMS/RCS systems shall provide flexible propellant utilization to allow many different missions.

20.1.2.2 CEV Service Module OMS/RCS Technology Options

Section 19.1 Alternate Propellants provides the details on the propellant selection. There are numerous combinations of OMS propellants, RCS propellants, and integration options between OMS and RCS. Many of these have been evaluated in the past for First Lunar Outpost, Shuttle, Shuttle Upgrades, X-33 and NGLT vehicles. The recommend design concept is a pressure-fed LO2/methane system because of the high performance, small package, and simplicity/reliability, and fewer critical failure modes. The OMS and RCS are integrated to allow flexible propellant utilization to support many different possible CEV missions.

The technology options for an integrated OMS/RCS pressure-fed LO2/LCH4 system are related to the tank, feedsystem, and engine. Each of these will be discussed in more detail.

Tank - The propellant tank has several options. The tank material can be either all composite, composite over-wrapped metal, or all metal. It is recommended for CEV that either the composite over-wrapped or all metal tank be selected due to the complex amount of attachments and penetrations.

The all metal tank could be either aluminum lithium or Inconel. The over-wrapped tank could use Invar as the metal, which has a zero coefficient of thermal expansion, so it is more compatible with a composite over-wrap. The fuel tank could be either the same as the Lox tank or could be made from titanium to provide the lightest weight.

The key question is how to integrate the RCS acquisition system into the tank. Integrated OMS/RCS tanks were studied in NAS9- and NAS9- for Shuttle Upgrades. The two tank designs are shown in Figure 20.1.2.2-1. The key to integrating these functions is to have an upper and lower compartment to the tank. The RCS screen galleries are contained in the lower compartment. This prevents screen breakdown during high-G acceleration caused by OMS maneuvers.

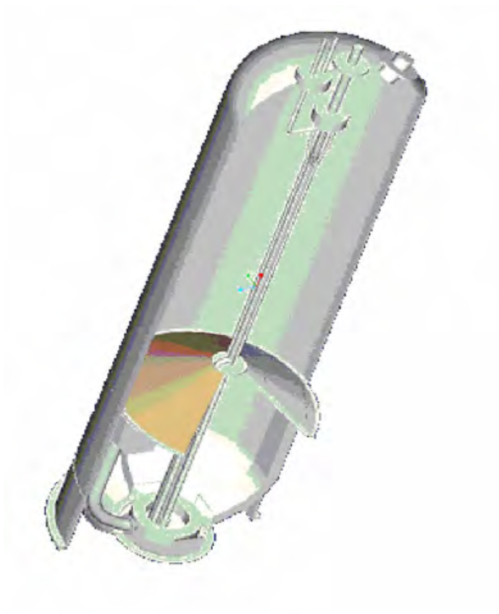


Figure 20.1.2.2-1: Integrated LO2 OMS/RCS Tank Designs

Feedsystem - Section 19.1 Alternate Propellant discussed the selection of integrated cryogenic liquid fed RCS. Two options for the RCS feedsystem were examined: 1) integrated RCS and 2) separate RCS. The recommendation is to integrate the RCS with the OMS. This provided the simplest, most operationally flexible, reliable option. This allows the RCS to perform OMS type maneuvers, or to budget OMS propellant for more RCS attitude hold or maneuvers. This will greatly increase the flexibility of the CEV or lander to meet different mission needs.

Another issue is whether to choose a cryogenic liquid feed of the RCS engines versus gaseous feed. This is also discussed in detail in Section 19.01 Alternate Propellants. The primary issue with gasification of OMS propellants for RCS is the large amount of power (>84 KW for 100 lbf engine) or heat required to gasify the propellants, the complexity of the system, the higher hardware mass (450 to 900 lbms), and the large flow areas of the valve required for gas. The gasification system also adds a significant number of Criticality 1 failure modes. In shuttle upgrades, it was for this reason that a sub-cooled cryogenic RCS feedsystem was selected as the optimal technology using multilayer insulation, propellant flow dues to thruster usage, and possibly cryocoolers to keep the manifolds conditioned.

The recommended option is to use liquid-fed because it is the simplest and most robust solution. Liquid-fed RCS also allows the RCS to perform long duration burns using OMS propellant, whereas gas-fed would be limited by the duty cycle capability of the gasification system. The technology options for a liquid-fed RCS center around how to maintain liquid up to the thruster interface. The options to maintain liquid in feedlines are 1) propellant flow due to thruster usage, 2) thermodynamic venting of propellant to vapor cool the lines, 3) using a cryocooler to maintain liquid in the lines, and 4) recirculation of fluid back to tank.

1) The simplest approach is a combination of 1) thruster usage and 2) thermodynamic venting. They key here is to match the heat leak into the lines with the ability of thrust usage to keep the lines chilled. The thrusters on redundant manifolds are cycled through such that each manifold is kept chilled. The manifold is designed as a semi-ring manifold with the most active thruster on the end. Since four thrusters are at the end of the manifold on a service module, this easy to do. This will mean that venting is only required in an off-nominal condition or if the system has been inactive at L1. *Note: Even if it assumed that propellant is budgeted for venting continuously (~10 lbm/day), a cryogenic feedsystem is still lighter than a system using gasification and also lighter than an MMH/NTO OMS/RCS vehicle*.

3) A cryocooler cold head can be installed at the end of the feedsystem line and can remove heat from the line. This relies on the conduction of heat through the pipe wall. A key technology is the development of multi-cold head cryocoolers.

4) Recirculation requires the use of pumps and lines back to the tanks. The heat from the tank will eventually need to be removed and will result in some boil-off, although the tank can absorb some heat before requiring venting. Recirculation adds the complexity of the pumps, however if the tank requires mixing then a pump may be needed anyway.

Engines – The RCS engines are film-cooled chambers with radiation cooled nozzles. This provides the life and duty cycle capability to provide delta-v maneuvers. The chamber materials could be C103 with R512 coating or a platinum alloy chamber that does not require coatings. It is recommended to look at Pt alloy chambers, since it eliminates coating failures. For subcooled LO2/ LCH4, the injector design can be either liquid/liquid pintle or impinging elements. A capacitive discharge spark ignition system is recommended as the baseline. Laser ignition has been examined in the past, but it does not appear to offer any advantages. The issues are higher mass (laser, power supply, optics). Damage to the fiber due to the high power requirements for ignition and damage to the optics are issues.

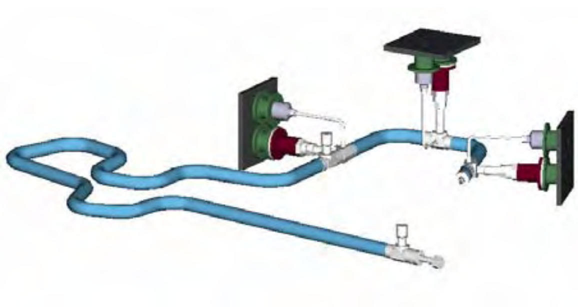
The OMS engines are regeneratively cooled for life and performance, with possible film cooling augmentation. The lander engine differs in that throttling is required. This could be accomplished 1) by having a modular engine that allows cells or injector circuits to be shutdown or 2) by utilizing a sliding pintle design as TRW did on the Apollo LEM descent engine.

20.1.2.3 Description of OMS/RCS Research and Development Activities

Tank – Cryogenic liquid oxygen OMS/RCS tanks have been developed and tested starting in the Shuttle Program in 1969. Ball Aerospace demonstrated a 1 year passive storage capability for a LO2 OMS tank in 1972. In 1997, the Shuttle Upgrades program designed a LO2 tank for OMS/RCS. MSFC/GRC have been working towards performing LO2 screen tests to verify results obtained using liquid nitrogen and some limited LO2 testing the past. Some of the future actitivites to investigate are integration of the cryocooler onto the tank. A common bulkhead LO2/LCH4 tank, Figure 20.1.2.3-1, has been developed to test cryocooler integration and storage of liquid methane oxygen at a common temperature and pressure. There should also be some effort made in tank pressure shell material and design selections. The storage and acquisition of sub-cooled LO2 in space is viewed by the CFM community as a higher TRL than other cryogens (ie LH2).

Figure 20.1.2.3-1: Common Bulkhead LO2 / LCH4 Tank

Feedsystem - Breadboard testing of a cryogenic LO2 RCS feedsystem has demonstrated the capability to maintain subcooled propellants near the thruster inlets. The tests at Energy Systems Test Area in 2001-2003 showed subcooled liquid could be maintained in the manifolds that feed the engine using a semi-ring manifold at a length of 140 ft.



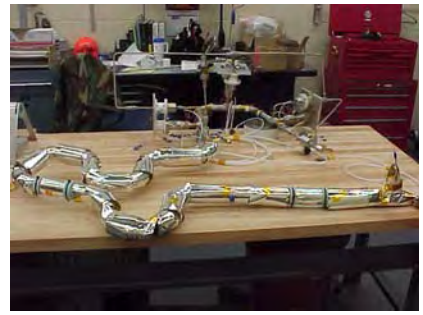


Figure 20.1.2.3-2: RCS Feedsystem breadboard test article (outside of 8 ft vacuum chamber)

The other approach of using gaseous propellants has been examined since the 1970's for shuttle prior to using MMH/NTO. For DC-X and X-33, Aerojet worked on gasification systems for GO2/GH2 thruster. These systems proved to impractical and complex. It was at this point that Aerojet switched to GO2/GCH4 for the RCS. The primary issue with gasification of OMS propellants for RCS is the large amount of power (84 Kw) required to gasify the propellants, the complexity of the system, and the large flow areas of the valve required for gas. The gasification system also adds a significant number of Criticality 1 failure modes. In shuttle upgrades, it was for this reason that a sub-cooled cryogenic RCS feedsystem was selected as the optimal technology using multilayer insulation, minimal thruster usage, and possibly cryocoolers to keep the manifolds conditioned.

Pressure-fed RCS and OMS Engine – Interest in LO2/hydrocarbon propellants for OMS/RCS started when a replacement for MMH/NTO was examined by McDonnell-Douglas and JSC in 1980. Subscale injector tests and engine point design studies for LO2/methane were performed by Aerojet in the 1980's[1]. Some recommended point designs were provided for engines to fit the shuttle engine envelopes as shown in Table 20.1.2.3-1.

Engine	Pc	Injection	Mixture ratio O/F	Cooling	Isp (sec)
RCE	150 psia	Liquid/Liquid	2.49	17% FFC	314
OMS	150 psia	Liquid/vapor	3.4	Vapor regen	346
OMS	400 psia	Supercritical CH4	3.4	regen	360.8

Table 20.1.2.3-1: LO2 / Methane Engine Designs for Shuttle

Aerojet tested a GO2/ethanol for JSC in 1986. When Kistler needed a non-toxic OMS for their reusable vehicle, Aerojet developed the 870 lbf LO2/ethanol engine, based on the previous etha-

[1] Hart, S. W. , "Combustion Performance and Heat Transfer Characterization of LOX/Hydrocarbon Type Propellants", Contract NAS 9-15958, July 1982

nol work. The JSC Shuttle Upgrade and the MSFC ASTP and NGLT program have performed LO2 based RCS and OMS engine testing since 1996 at TRW, Aerojet, and Rocketdyne. LO2/ethanol engines have been recently tested including an two 870 lbf LO2/ethanol engines, TRW and Aerojet, and a 6000 lbf LO2/ethanol Rocketdyne / DASA engine. The NGLT program for LO2/ethanol is building a test stand compatible with ethanol, methane, and hydrogen fuels.

LO2/methane has seen some development activities, but has not been carried through to completing an engine. However, many of the technologies for LO2/ethanol are applicable to LO2/methane, with modification. For example, engine hardware for LO2/ethanol may be operated with LO2/methane to obtain design data for LO2/methane engines. It is also important to note that MMH/NTO technologies are compatible to some degree with subcooled pressure-fed LO2/ethanol and methane. The 6000 lbf Rocketdyne/DASA Aestus engine was an unmodified MMH/NTO engine, but was operated successfully on LO2/ethanol as shown in figure 20.1.2.3-3.

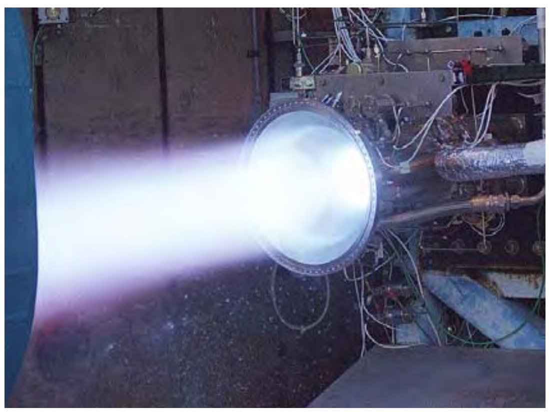


Figure 20.1.2.3-3: 6000 lbf Unmodified Rocketdyne / DASA Aestus MMH/NTO Engine Running on Lox / Ethanol

For pulsing applications, it is likely that two-phase propellant will exist at the inlet to the engine. However, igniter and engine testing has also shown that at greater than ~50 lbf, there is little impact to thrust response and repeatability as caused by small amounts of two phase LO2 flow, even when the engine starts from ambient temperatures. At high enough propellant flowrates, the vapor passes quickly and the subcooled cryogenic liquid flow overwhelms the ability of heat transfer from the injector to significantly boil the LO2 flow. This same approach could be applied to a liquid methane thruster. It is critical that injector designs minimize surface area for heat transfer into the propellants.

Pintle engines are possible technologies for OMS and lander engines. The pintle was a liquid/liquid injector used on the LEM descent engine to provide 10:1 throttling. This same approach could be used for LO2/liquid methane.

20.1.2.4 Recommended CEV Service Module Propulsion System Design Approach

The recommended system is a pressure-fed LO2/methane or LO2/ethanol with an integrated RCS using sub-cooled liquid propellants. This selection is based on the best combination of performance, volume/packaging, safety, and reliability. Pressure-fed systems are robust, highly reliable systems. There will be 24 RCS jets and 2 OMS engines as shown in Figure 20.1.2.4-1. The RCS engines are designed to operate on liquid, therefore they are able to performs long steady state burns as an OMS engine back-up, although at a lower Isp and mixture ratio. This is a fail-safe mode and would require a return to earth. [A RCS operating on a gasification system would not be able to perform the OMS function].

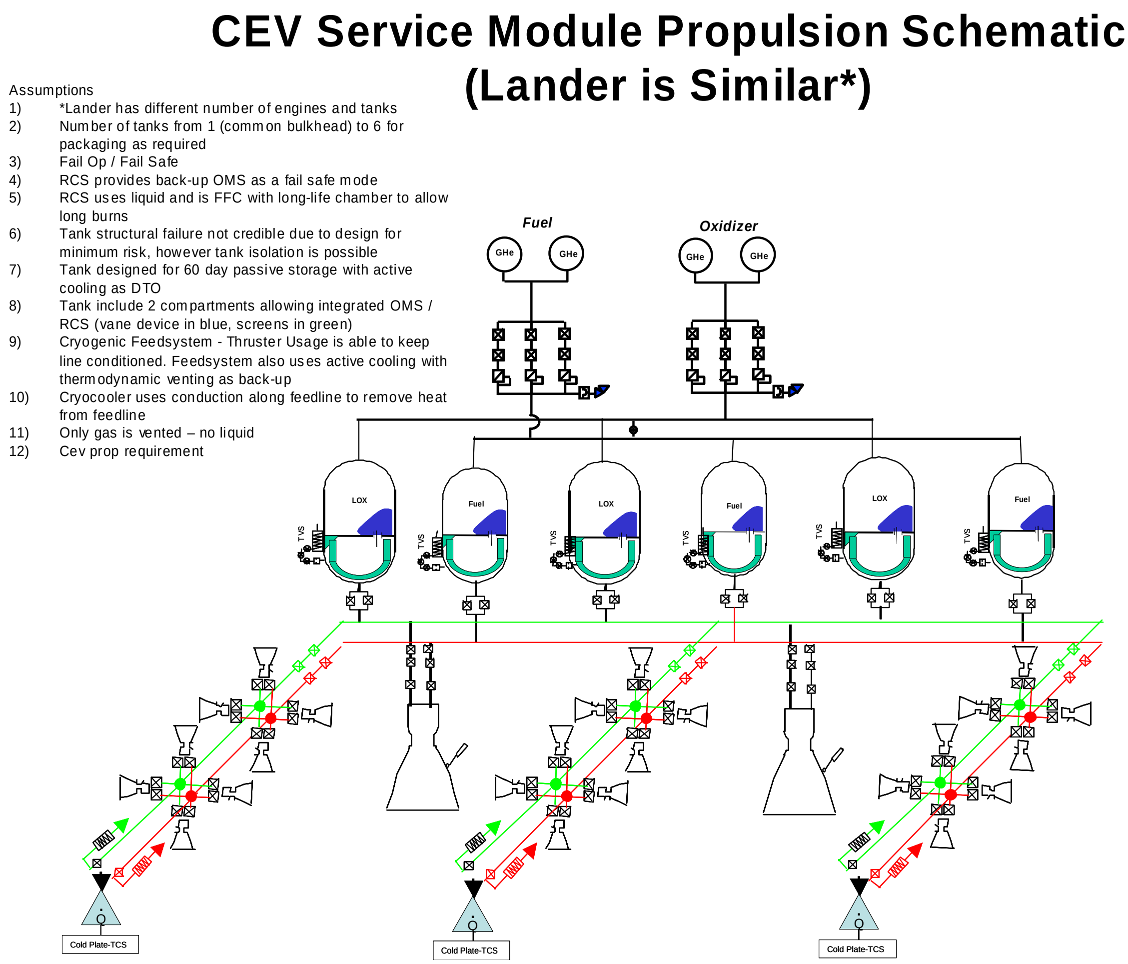


Figure 20.1.2.4-1: Integrated OMS/RCS Schematic

The LO2 and methane are stored subcooled at 163 R. The tank pressure is 350 psia. This raises the boiling point to 240 R for LO2 and 300 R for LCH4 as shown in Figure 20.1.2.4-2, which provides >77 deg R of subcooling. The configuration chosen for this study is 3 LO2 and 3 LCH4 tanks as shown in Figure 20.1.2.4-3a,b. This allows passive storage of the propellants with 50 layers of MLI for a 60 day mission. A passive thermodynamic vent system is provided to cool the tank wall. A cryocooler is not required, however it is recommended that cryocoolers be integrated as a development test objective (DTO) for eventual Mars application.

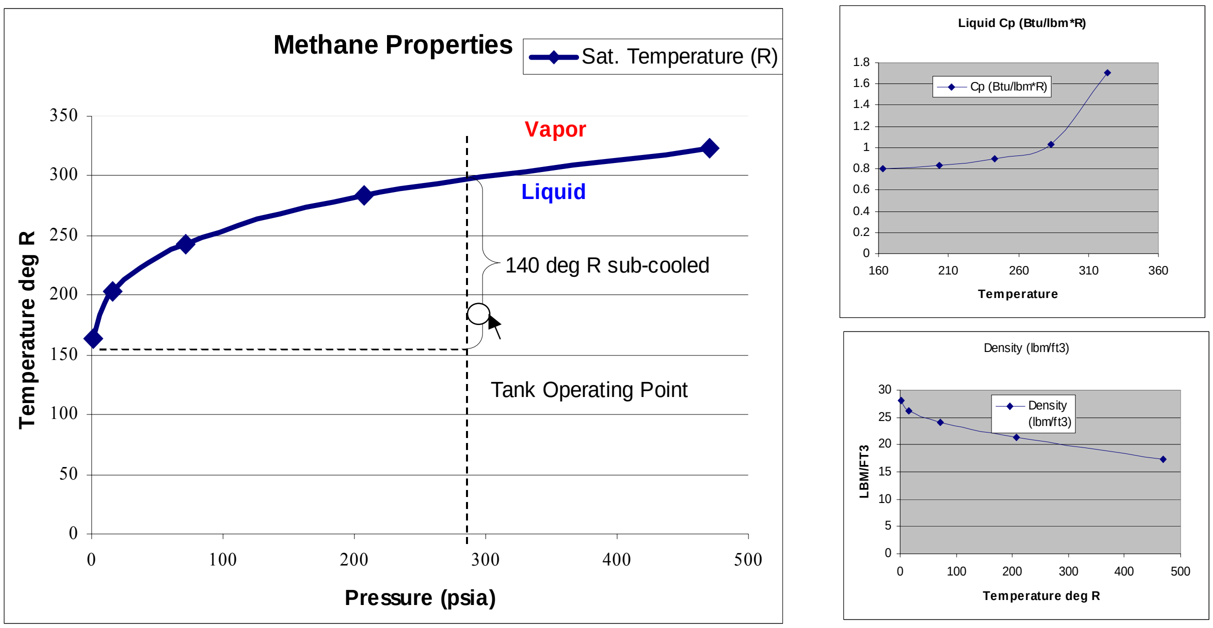


Figure 20.1.2.4-2: Properties of Methane

Another option for LO2/methane is to have a single common bulkhead tank. This simplifies the propulsion system and may allow a single tank to be packaged on the CEV or Lander as shown in figure 20.1.2.4-3c.

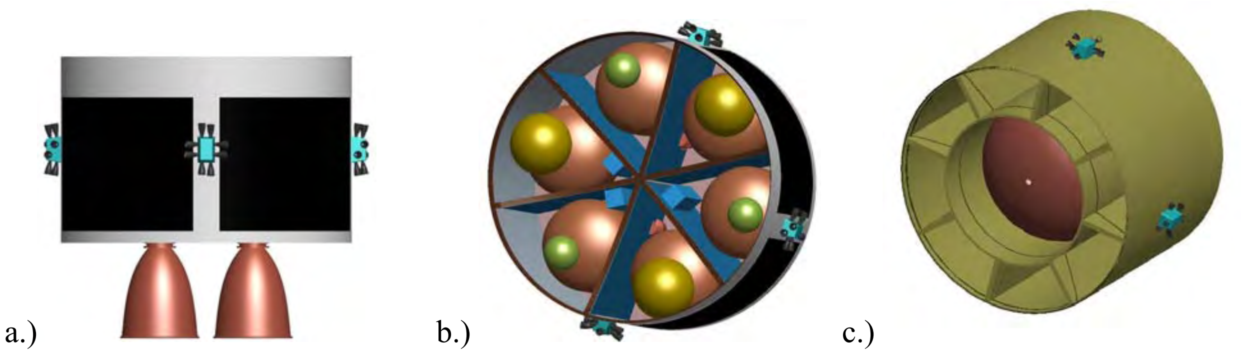


Figure 20.1.2.4-3: a) Side View of SM, b) Six-Tank Configuration, c) Single Common Bulkhead Tank

Figure 20.1.2.4-1 shows the 3 redundant RCS manifolds. The 3 RCS feedline manifold are approximately 25 ft long each. The heat leak into each manifold could be designed to approximately 20 btu/hr for both propellants. The two propellant lines (LO2 + LCH4) could be insulated together using 30 layers of MLI and G10 supports. (Each redundant manifold would take separate routes.) The design would use 3 independent methods to maintain condition in the lines. This would provide the most robust operation. Recirculation is another option that would move the cryo-cooler to the tank, but would add the complexity of a pump. If a pump is required for mixing the tank (which may not be necessary if the cryogenic tank is highly subcooled), then the feedsystem recirculation pump could help with tank mixing.

1) Thruster usage would need to be 1.8 lbm/hr for all three manifolds to keep lines sub-cooled. This is within expected RCS usage for a service module during the active periods. This means thruster usage could keep the line chilled, thus venting is not usually required.
2) Thermodynamic venting of 0.5 lbm/hr using a vapor cooled shield on the line would be used in the event thruster usage does not keep the lines conditioned. <u>If the vented gaseous oxygen is used for life support or power then it is not wasted</u>. Vapor cooling the line use the heat of vaporization to intercept the heat leak (H_{fg} = 200 btu/lbm for LCH4 and 90 btu/lbm for LO2). This would require 0.5 lbm/hr. A valve at the end of the manifold would cycle if the temperature exceed a preset limit (~220 R).
3) A cryocooler could also be sized to remove the 20 btu/hr of heat leak. The cryocooler would need to cool 6 watts per manifold. This would be recommended as a DTO for Mars application.

The power requirement is estimated to be 200 to 300 W for propellant conditioning using cryo-coolers. However, it is possible to passively maintain tank conditions with no boil-off of LO2/LCH4 for several months, given sufficient ullage and the high degree of sub-cooling obtained during propellant loading and pressurization. If the lines are allowed to warm, such during the L1 loiter period, they can be re-chilled thermo-dynamically using roughly 40 lbm of propellant.

The RCS engines are 100 lbf liquid oxygen / liquid methane engines. The engines are film-cooled. The Isp is 315 sec at mixture ratio of 2.5:1. The expansion ratio is 40:1. For robustness, the RCS engines would be designed to operate for a short time on cold gaseous propellants if the line conditioning was not functioning properly.

The OMS engines are 5000 lbf thrust, 150:1 area ratio engines at 175 psia Pc with an MR = 3.5. The assumed C* efficiency is 94%. The Isp is 362 sec. The lander engine are similar except that they can be throttled 5:1. The engine is assumed to be liquid regeneratively cooled, perhaps with both propellants, and cooled possibly with some film cooling. The critical area to study is the regeneratively cooling of this engine using LO2/methane.

The key LO2/LCH4 technologies for development by 2009 are prototype engine and system level tests in a simulated space environment. The critical areas requiring additional work before 2009 are flight-weight cryogenic valves and ignition systems. In a pressure-fed system, valves are probably the biggest problem area.

No design breakpoints were identified for the design of the CEV Service Module OMS/RCS within the parameters identified for LDRM-2.

20.1.3 Mars Spiral Development

A Mars mission will require a high performance lander descent/ascent propulsion system. These systems must survive the cold/hot temperatures on the Mars surface. Earth storable propellants will require high power heaters on the surface of Mars to keep the propellants from freezing and rupturing lines. Cryogenic propellants will not freeze, but will require lightweight vacuum jacketing (fortunately there is low delta-P across the jacket in Mars atmosphere) and power for cryocoolers. From the standpoint of in-situ resource utilization, the availability of CO2 at Mars also favors the use of oxygen and methane propellants.

It is not reasonable to expect, if the CEV uses earth storable propellants, that NASA can then fly cryogenic systems on a lunar mission. Experience with cryogenics in low earth orbit is critical to future lunar vehicles. The CEV (ie capsule, service module) are critical steps in using cryogenics on landers. The CEV and lunar missions should test LO2/methane systems and cryocoolers for eventual use on Mars. A pressure-fed LO2/methane system can be designed to be robust (high design margins) for a service modules (ie high boil-off allocations, etc) for 2011 vehicle and then 2014.

The key LO2/LCH4 technologies for development by 2008 are prototype engine and system level tests in a simulated space environment. The critical areas requiring additional work before 2008 are flight-weight cryogenic valves and ignition systems. In a pressure-fed system, valves are probably the biggest problem area, so this area must be emphasized. Future efforts for Mars would include higher performing LO2/methane engines (pump-fed or high Pc pressure-fed).

Recommend Technology Maturations

- Develop integrated pressure-fed cryogenic OMS and RCS tanks and feedsystems to allow flexible propellant utilization to support many different missions. The use of pressurized subcooled cryogenics simplifies cryogenic fluid management technologies.
- Focus efforts on pressure-fed LO2 methane due to its overall performance, simplicity relative to pump-fed LO2/LH2, fewer critical failure modes, small packaging volume.
- The key LO2/LCH4 technologies for development by 2009 are prototype engine and system level tests in a simulated space environment. The other critical areas requiring additional work before 2009 are flight-weight tanks, cryogenic valves and ignition systems. In a pressure-fed system, valves are probably the biggest problem area.
- Focus efforts on Tridyne for the capsule RCS. Alternate technologies would be nitrous oxide or GOx/ethanol.
- Focus efforts on Mars surface storage of LO2/LCH4.
- Use CEV service module flights with DTO’s for cryocoolers to demonstrate for Mars. Lunar missions can be done passively, but cryocoolers must be flown to test for Mars applications.

20.2 Power Subsystem Technology Report

Karla Bradley, NASA/JSC/EP – Energy Systems Division

20.2.1 Power Subsystem Description

The primary functions of the power subsystem are power generation, power distribution, energy storage and energy conversion. The design driver for the power subsystem is the combined vehicle subsystem loads. The typical redundancy and reliability approach for power systems is fault tolerance. For a human-rated vehicle two-fault tolerance (three or more string power systems) allow for safe return of the crew. Each string has its own power source and control source that is distributed by multiple switching units. The main bus switch units will be fully cross strapped to provide the maximum redundancy during nominal and off-nominal operations.

One method of reducing the vehicle mass and volume is to conserve the vehicle resources. Power usage can be reduced on the CEV when the crew is in the Lander. During this time the CEV life support functions would be reduced. Also, depending upon the avionics design, some portions of the avionics may be either put in a reduced power mode and/or a reduced data rate.

One byproduct of fuel cell power generation is potable water. This water can be utilized for crew consumption and/or cooling purposes. Proton exchange membrane (PEM) fuel cells can also utilize O2 reactant which can share storage with the propulsion O2 and the ECLSS O2. This shared storage reduces the number of O2 tanks on the vehicles.

20.2.2 Power Subsystem Technology Options

Name	TRL	Comments
Alkaline fuel cells	9	No commercial applications; facing obsolescence issues; safe & reliable performance on Shuttle; requires high purity O2 &H2; no common tankage with propellants
Proton Exchange Membrane fuel cells	4-5	Commercial stack development for H2 & air; space development ongoing from NGLT; allows common tankage with propellants; requires external reforming of hydrocarbon fuels
Solid Oxide fuel cells	2-3	Commercial development; allows commonality with propellant; allows external or internal reforming of hydrocarbon fuels; operates at high temperatures (700-1000 ^{o}C)
Li-ion rechargeable small cell batteries	7-9	Multiple suppliers; safer than large cell batteries; commercial development for further weight reduction; flown on commercial satellites; developed for Shuttle electric auxiliary power unit program
LiMnO2	7-8	Primary battery; Better long-term storage than AgZn; qualified at 28V for X-38
GaAs Solar Arrays	7-9	Flown on commercial satellites

Figure 20.2.2-1: Power Subsystem Technologies

20.2.3 Current Research and Development for Power Systems

20.2.3.1 Fuel Cells

A fuel cell is an electrochemical power generation device that converts hydrogen and oxygen reactants into electrical power, heat, and potable water. A fuel cell power plant has a power section and an accessory section. The power section is the fuel-cell stack. The accessory section consists of hardware that provides reactant supply and control, thermal management, water management, and instrumentation and control.

Fuel cells have been used during the Gemini, Apollo, and Space Shuttle programs for space vehicle power. An early form of the proton exchange membrane fuel cell (PEMFC) was used for Gemini, while the Apollo and Shuttle programs have used alkaline fuel cell (AFC) technology.

Although the AFC technology presently used on the Shuttle has been highly effective and reliable, it faces serious obsolescence issues in the near future—around 2010. AFC technology has no widespread commercial applications, has seen little development over the past 20 years, and is supported for space applications by a single vendor. Furthermore, the AFC requires higher purity reactants than those used for the propulsion systems. Therefore, the AFC cannot share O2 reactant storage with the propellant storage, unless the costlier fuel cell grade O2 reactant is utilized, whereas the PEMFC can utilize common tankage.

Code T has assumed responsibility for the NGLT program for hydrogen and oxygen PEMFC research and development. Under this program, one vendor has successfully designed and built a breadboard PEMFC system. An engineering model (EM), which will undergo performance testing, vibration testing and thermal vacuum testing is currently underway. EM delivery for testing is expected to be in March 2005. Currently, hydrogen and air PEMFC's are under development for transportation and stationary power applications.

20.2.3.2 Batteries

Batteries are energy storage devices used in numerous space applications. For space applications batteries have been used for vehicle power, ground flight equipment, crew flight equipment, Shuttle and ISS payloads and satellites. Battery cell chemistry is selected based upon the duty cycle, operational environments, specific power and power density, specific energy and energy density, cycle life, charge retention, maintenance capabilities and discharge efficiency.

For the CEV, a rechargeable Li-ion small cell chemistry has been selected and sized. Small cell batteries are safer for space applications than large cell batteries. These small cell batteries have multiple suppliers. Commercial development is also underway to potentially further reduce the weight of these Li-ion batteries. Li-ion batteries have flown on satellites and were under development by NASA for the Shuttle electric auxiliary power unit program.

For the injection stage and the lunar Lander descent stage, Lithium Manganese Oxide (LiMnO2) batteries were selected and sized. These batteries have high specific energy, a low drain rate, a long wet life and good charge retention. LiMnO2 batteries were designed and qualified for the X-38 program.

20.2.3.3 Solar Arrays

Solar arrays have been used on various NASA programs, most recently including the ISS. Solar arrays are also commonly utilized on commercial satellites. State-of-the-art solar array technology is gallium arsenide (GaAs). This same technology can be used for both deployable arrays and body-mounted arrays. To meet the objectives of a CEV lunar mission, no technology development of solar arrays themselves would be necessary.

20.2.3.4 Power Management and Distribution

Power management and distribution is the channelization of available power throughout the vehicle to meet the various loads. The power distribution system may need to distribute both DC and AC power to the loads. The power distribution system must have redundant power channels. Each channel should have a separate power source with its own control unit. The main bus switch units should be fully cross-strapped to provide the maximum redundancy during nominal and off-nominal operations. This design is also fault tolerant to system failures. While the various power management and distribution technologies are already in existence, the distribution architecture for the CEV differs based upon the loads, the power sources, the mission profiles and environments and the safety and reliability requirements. Further trade studies will have to be performed to determine the best methods of power transmission while maintaining the crew safety.

20.2.4 Recommended LDRM-2 Power Subsystem Design Approach

The power system recommended for LDRM-2 consists of a combination of power generation and energy storage technologies, which includes PEMFCs, rechargeable Lithium Ion batteries, non-rechargeable Lithium Manganese Oxide batteries, and Gallium Arsenide solar arrays. The CEV service module (SM) will have three 6kW PEM fuel cells fed by multiple reactant storage tanks and will provide power to both the command module (CM) and the SM. The O2 reactants will share tanks with the ECLSS and the propulsion system on the SM, but there will be a limited quantity of O2 stored in redundant accumulator tanks. The H2 reactant will be stored in multiple dedicated reactant tanks. The quantities of O2 and H2 reactants required are dependent upon the number of days that the SM will be on–orbit as well as the power profiles of the vehicle. Assuming 16 days at 6 kW and 13 days at 3 kW, the H2 reactant quantity needed for the CEV is 130 kg. The CEV SM power system will also include redundant power distribution and conversion boxes.

The CEV CM will have three primary 28V distribution buses to distribute the power generated by the fuel cells. Additionally, there will be three 28 kW-hr Li-ion batteries to generate up to 8 kW of power for 3.5 hours after separation from the SM. Each of these three Li-ion batteries will weigh 488 pounds and be approximately 105 L. This assumes that the three battery sets are fully two-fault tolerant. If two strings of batteries fail, the third battery set can fulfill the energy requirements of the mission.

The Lander ascent stage power system will be similar to that of the CEV SM. There will be three 6kW PEM fuel cells generating power. It will be distributed through three primary distribution buses. The O2 reactant will again share storage with the ECLSS and propulsion systems, but there will also be limited quantities of O2 reactants stored in dedicated redundant accumulator tanks. The H2 reactant will again be stored in multiple dedicated reactant tanks. The quantities of reactants required is again dependent upon the number of days that the ascent stage will be on–orbit as well as the power profiles of the vehicle. Assuming 12 days at 5 kW, the H2 reactant quantity needed for the lander ascent stage is 59 kg.

The lander descent stage power system will have deployable triple-junction Ga-As solar arrays for quiescent periods. There will also be three non-rechargeable Lithium Manganese Oxide batteries, which store 2 kW-hrs of energy each. These batteries will be 15.6 kg and 13 liters each.

The injection stage will utilize Ga-As body-mounted solar arrays and three non-rechargeable Lithium Manganese Oxide batteries, which store 2 kW-hrs of energy each. These batteries will also be 15.6 kg and 13 liters each.

For all of these power subsystems, the only recommended technology not yet at a TRL of 6 is the PEM fuel cell. Additional funding will need to be allocated to assure that PEM fuel cells can achieve a TRL of 6 by 2009 and a TRL of 9 by 2014.

20.2.5 Mars Spiral Development

The most promising power generation or energy storage device for a Mars transit vehicle utilizes nuclear power as a primary source, and a combination of solar arrays, batteries and fuel cells as a secondary source. Nuclear energy systems have long lives and can be used in harsh space environments. Radioisotope power system (RPS) development is underway in conjunction with the Department of Energy for Project Prometheus. Fission reactor power systems for space are significantly smaller (~10,000 times smaller than terrestrial applications), and are designed to remain inactive until arriving at their startup location and receiving a signal to initiate operation. NASA will have to design and safely implement RPS and nuclear reactor power systems for both the transit vehicle power as well as the surface power generation. Supplemental power generation and energy storage subsystems on the surface of Mars will be regenerative fuel cells, which consist of fuel cells, electrolysis units, rechargeable Lithium-based batteries, and possibly solar arrays. Portable power systems for use in tools, rovers and portable equipment will use fuel cells, solar arrays, rechargeable batteries and RPS. The exclusive use of solar arrays and rechargeable batteries as a secondary power source on the surface of Mars is unlikely due to the planetary-wide dust storms and dust devils on Mars. These conditions are likely to introduce solar flux and mechanical load challenges that would preclude the utilization of solar arrays. Furthermore, day/night cycles would also affect the solar array performance.

PEM and solid oxide fuel cells are a likely candidate for Mars supplemental sustained surface power generation. Fuel cells are modular and scalable and have the additional benefit of producing potable water as a by-product. Solid oxide fuel cells do not require the use of external reformers to utilize hydrocarbon fuels. They also operate at potentially very high efficiency levels. However, the low TRL of solid oxide fuel cell precludes their use for the lunar applications.

20.3 Environmental Control and Life Support System (ECLSS) Technology Report

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20.3.1 Description of ECLSS Functions

Spacecraft and planetary habitats will require environmental control and support systems (ECLSS) to support crews. Typically, the ECLSS provides the crew a habitable environment that includes:

- breathable air (nominal and emergency air)
- temperature, pressure and humidity control
- environmental monitoring
- fire detection and suppression
- waste management
- potable and hygiene water management
- habitation systems (including food systems) [discussed in a separate technology report]

Key ECLSS interfaces include the crew and human factors, the active thermal control system (ATCS), power systems, extra-vehicular activity (EVA) systems (including airlocks), displays and controls, integrated health management (IHM) systems, ground processing, operations and training. ATCS to support the habitable environment and equipment functionality will be addressed in a separate technology report. A primary dependence for ECLSS is selection of power system technology. If fuels cells are used, water is provided as a by-product. If solar arrays or another power source is used, potable water for the mission duration would need to be carried on the crew exploration vehicle (CEV) and Lander and its potability would need to be maintained for mission durations. For LDRM-2 another key dependency for ECLSS is the oxygen (O2) supplied by the propellant tanks.

20.3.2 Driving ECLSS Requirements

LDRM-2 life support considerations are largely tied to the mission durations and crew sizes for the CEV and Lander; more specifically, the length of time the crew will occupy the CEV and Lander during any given mission phase. Current LDRM-2 operations concepts estimate the approximate crew residence times in the CEV and Lander as listed in Table 20.3.2-1. ECLSS functioning during the quiescent (loiter) phases for the CEV and Lander also need to be considered (e.g., subsystem health status, timing of power-up before crew occupies to the vehicle, power requirements while quiescent, etc.). The crew size for LDRM-2 is four for both the CEV and the Lander. This dictates the amount of ECLSS and habitation consumables needed per day per crew person. Crew size will also impact crew cabin volume utilization requirements. Due to the relatively low internal crew cabin volume assumptions for the CEV and Lander, optimization of storage, seating, suit options, subsystems sizing and habitation elements assumptions is critical. EVA Systems requirements (including the airlock on the Lander) will also drive ECLSS consumables and subsystems sizing. For LDRM-2, early implementation of in-situ resource utilization (ISRU) for life support systems is not part of the picture. Hence, the maturation of current regenerative life support systems technologies could benefit CEV and Lander applications by reducing mass and volume, providing higher reliability and reducing radiator size.

LDRM-2 Element/Phase	With Crew (~days)	Without Crew (~days)
CEV Outbound	8.75	
CEV Rendezvous w/Lander	1	
CEV L1 Loiter		12
CEV Rendezvous w/Lander	1.25	
CEV Earth Return	4	
CEV Totals	**15**	**12**
Lander (in space)	7.5	54
Lander (on surface)	7	
Lander Totals	**14.5**	**54**

Table 20.3.2-1: Crew CEV and Lander Residence Times

Another driving requirement for CEV and Lander ECLSS is the availability of potable water for the crew. As mentioned in 20.3.1, this requirement also affects the power system technology selection. Other driving requirements for CEV and Lander ELCSS design considerations include LDRM-2 safety assumptions. Redundancy and reliability requirements will drive ECLSS subsystem mass and volume requirements. For the CEV and Lander two carbon dioxide (CO2) and moisture removal systems were assumed for redundancy. Contingency assumptions drive emergency life support design. For example, emergency O2 and quick disconnect masks were assumed for both the CEV and Lander.

Additional driving requirements for CEV and Lander ECLSS include anticipated landing conditions (determines the extent of post-landing environmental control required), the number of crew cabin repressurizations assumed for the CEV and Lander (drives amount of gas consumables - two full cabin repressurizations were assumed for the CEV and eight for the Lander), and crew medical requirements (drives habitation system equipment requirements).

20.3.3 Technology Selection and Assessments for CEV and Lander

Technologies considered for the LDRM-2 CEV and Lander are addressed below for each major ECLSS functional element.

20.3.3.1 Atmosphere and Pressure Control

ECLSS functional elements provided for CEV and Lander to support an acceptable atmosphere and pressure for the crew include:

- Pressure Control System (PCS)
- CO2 removal system
- Humidity control system
- Trace Contaminant Control System (TCCS)
- Air distribution system, Air mixing system, Avionic cooling support (with air)
- Emergency oxygen system
- Environmental monitoring systems

A top-level concept for LDRM-2 CEV and Lander atmosphere revitalization is provided in Figure 20.3.3.1-1.

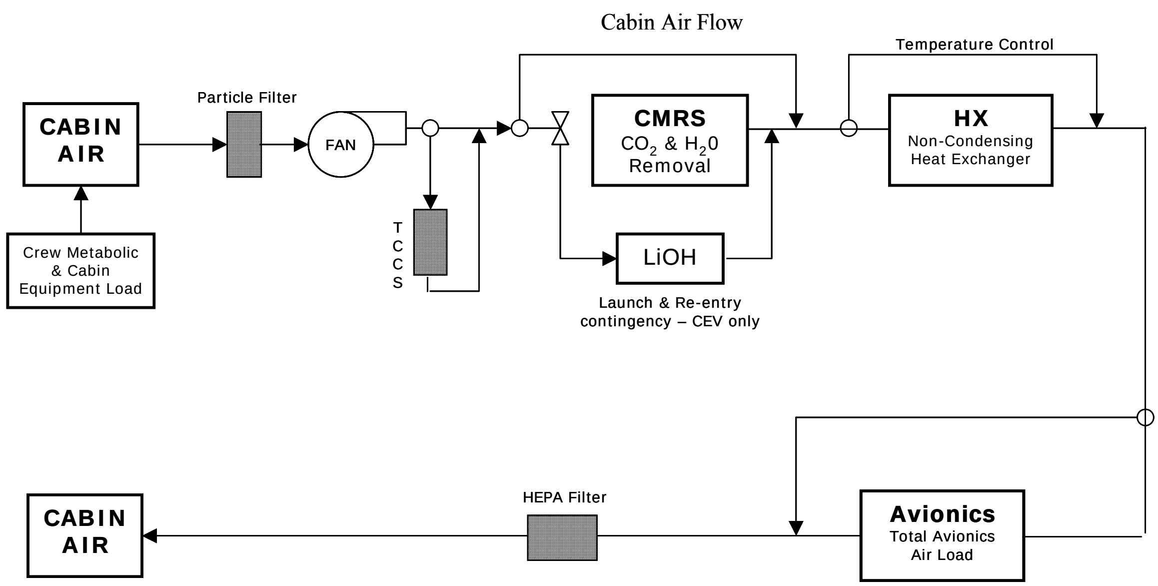

Figure 20.3.3.1-1: LDRM-2 CEV and Lander Atmosphere Revitalization Concept

The technology selected for the PCS will control CEV and Lander internal atmospheric pressures to a nominal 9.5-psia. Currently used (TRL 9) automatic solenoid valves were selected as the system's pressure valves. The CEV and Lander atmospheric composition is assumed to be 27-30% O2 and 70-73% nitrogen (N_2). The CEV and Lander provide and maintain crew respirable air to NASA-STD-3000 air quality standards for the mission duration. O2 is supplied by the propellant tanks on the service module (SM) and N_2 is stored in high-pressure tanks. Two full cabin repressurizations were assumed for the CEV and eight for the Lander. No new technology is baselined for this functional element for LDRM-2. An alternative technology to consider for the pressure valves could be motor-settable regulators. These valves could continue to regulate cabin atmosphere with a loss of power to the PCS. They have been used in the aircraft industry, but have not been tested in space to date and can be considered at a TRL of 5/6.

CO_2 removal and humidity control functions will both be provided by a regenerable combined CO_2 and Moisture Removal System (CMRS). A contingency lithium hydroxide (LiOH) canister is provided for potential use during launch and entry for the CEV only. Two CMRSs are assumed for redundancy. The CMRS is currently at a TRL of 5 and uses solid amine swing beds to remove CO_2 and moisture. Using the CMRS will allow the minimum control temperature for the cabin liquid cooling loop to be raised by about 10°F and this increase improves the heat rejection efficiency of the vehicle radiators and evaporators. Recent trades for other spacecraft designs have also shown that this technology can improve reliability since it is less complex and has fewer parts (eliminates rotary fans, gas/liquid separators and a condensing heat exchanger) than the current CO_2 and humidity removal system (LiOH and condensing cabin heat exchanger with humidity water separators) used on Shuttle. The internal cabin volume reduction is more than a 75% reduction vs. volume needed for LiOH cans stowage. This cabin volume reduction will result in smaller and lighter CEV and Lander crew cabins. The Shuttle has used a precursor version of this technology in the Regenerative CO_2 Removal System (RCRS). The CMRS uses a new amine formulation with significantly improved performance over the RCRS. A direct comparison in the lab of pure chemical activity shows that the new formulation has greater than 400% more activity than the RCRS chemical. The improved chemical performance allows for significant weight and volume savings. A single RCRS weighs approximately 240 lbs and occupies 10.7 cubic feet. A CMRS is estimated to weigh 120 lbs and occupy only 5 cubic feet. Primary and contingency systems for moisture and CO_2 removal will fit within the envelope for a single RCRS system.

Solid amine CO_2 and moisture removal has been demonstrated by small scale performance testing and in a full-scale test unit. The testing has verified the long chemical life of the tested amine. Additional testing is needed to minimize the airflow delta pressure drop across the bed and to determine the ideal cycle time for the beds. The bed sizing would be based on crew size and adjusting the airflow rate and cycle time will determine the CO_2 and humidity control levels. The risk is the delta pressure drop across the bed, which will drive the power level needed for the fan to achieve the desired flow rate and bed size. The expected fan power usage should trade nearly evenly with the power saved by eliminating the humidity water separator. NASA (Code U and Code M) is currently funding the following maturation activities for the CMRS:

1) Conduct full scale, parametric testing (Code M)
2) Establish vehicle interfaces (Code M)
3) Develop bed system with improved pressure drop performance (Code U)
4) Develop prototype system with new bed (Code U)
5) Conduct full-scale, long term, performance tests (unfunded future plan)

Assuming the final full-scale tests are funded, it is anticipated that the CMRS will be able to meet the LDRM–2 requirement of being at TRL 6 by 2009. This technology also has potential applicability to EVA systems. Demonstrating the use of the regenerable technology for lunar missions will advance the state for regenerable life support technologies for future Mars missions. Risks for this technology primarily center on NASA's ability to mature and test this technology within current schedule and funding profiles.

The LDRM-2 technology off-ramp for CO_2 and humidity removal would be the current Shuttle–based LiOH and condensing cabin heat exchanger with humidity water separators. There would be an increase to CEV and Lander mass and volume requirements as well as an increase in radiator size by reverting to this technology. Another CO_2 removal technology to consider could be the use of reactive plastic LiOH (TRL 4). Reactive plastic LiOH has been developed and tested that can pack 25+ % more LiOH into the same volume and also not reduce the LiOH CO_2 removal efficiency or capability. This reduction would save the volume needed to store LiOH and reduce the number of cans. A new technique was developed in forming LiOH in thin paper-like sheets that have better reaction surfaces and allow more LiOH to be packed in the same volume vs. granular LiOH material as currently use by Shuttle and the International Space Station (ISS). NASA has tested the paper LiOH reactive plastic and found that the process does not induce any toxins and the CO_2 reactive efficiency is improved. The plastic LiOH was made in disk layers with raised ribs to minimize airflow delta pressure drop. This stack met the performance requirements. Analysis indicates that 31% more LiOH could be packed in an existing Shuttle LiOH can. This would reduce the LiOH cans needed by 30% and reduce the cans' weight needed to support a human space flight. Risks for this technology primarily center on NASA's ability to mature and test this technology within current schedule and funding profiles. Using reactive plastic LiOH would still require another system for humidity control.

Another alternative CO_2 removal technology to be considered for future trades is the ISS CO2 Removal Assembly (CDRA) system (TRL 9). The CDRA includes two regenerative desiccant beds to remove water and two regenerative zeolite molecular sieve beds to remove CO_2. CO_2 is heated and vacuum-desorbed to space. Potential upgrades to the CDRA include recovering the CO_2 for application with a Sabatier reactor that would reduce the CO_2, and then convert CO_2 and hydrogen to methane and water. A CO_2 compressor would be needed to assist this functionality. The benefit from this upgrade is the recovery of water. However, for LDRM-2, fuels cells are planned for water production, and given the LDRM-2 mission lengths water recovery is not a driver.

The trace contaminant control system (TCCS) for the LDRM-2 CEV and Lander consists of activated charcoal, ambient temperature catalytic oxidizer and filters. These systems are currently used on Shuttle and the ISS and are considered to be at TRL 9. High Efficiency Particulate Air (HEPA) filters are considered for CEV and Lander given the potential of a higher particulate environment due to dust from the lunar surface. Alternative TCCS technology options for future trades are a charcoal bed with phosphoric acid coating (TRL 4) and an Advanced TCCS (TRL 3) that uses a regenerative sorbent bed.

CEV and Lander air distribution systems, avionic cooling support (with air) and air mixing systems consist primarily of ducting, fans and filters (TRL 9). Because the CEV and Lander will be operating with cabin pressures of 9.5 psia versus an Earth sea level standard atmospheric pressure of 14.7 psia, the ducting and fan speeds may need to be adjusted to accommodate required increases in air flow rate. This can translate into a slight power increase for the fans. Should the

Lander be used in a lifeboat scenario (ala Apollo 13) consideration should be given to manifest some ducting than could be used as a pass-through duct in an emergency situation. Additional avionics cooling support will come from the CEV and Lander ATCS as described in a separate LDRM-2 technology report.

LDRM-2 CEV and Lander emergency oxygen for the crew is provided with quick-disconnect masks and individual gas bottles or a common O_2 tank (to be traded). An alternative emergency O_2 system could include a four-person rebreather with individual masks that provides either mixed air or pure O_2. A rebreather is a device that captures and recirculates at least part of a person's exhalation, thereby allowing the person to rebreathe part of his previous breath.

Environmental monitoring systems for the LDRM-2 CEV and Lander provide the capability for in-flight monitoring of O_2 partial pressure, CO_2 concentration, dew point (humidity), cabin temperature and pressure. The CEV and Lander environmental monitoring and control systems should also provide in-flight verification that the air revitalization system is functioning and provide the capability to analyze combustion byproducts in the event of a smoke contingency. A key interface for this system is the vehicle IHM system. Current Shuttle and ISS sensors are baselined for LDRM-2 at this time. Advanced systems such as an Advanced Multi-gas Analysis System and a Low Maintenance Major Constituents Analyzer should be considered in future trades. The mass delta between current and advanced sensors would be negligible for LDRM-2.

Table 20.3.3.1-1 summarizes the technology options considered for the LDRM-2 ECLSS air system.

ECLSS Technologies	TRL	Pros for LDRM2	Cons for LDRM2	Comments
Air				
CO2 & Humidity Control				
CMRS	5	save mass, volume, complexity, & radiator size	Some maturation activities yet to be funded	regenerable combined CO2 & moisture removal system w/solid amine swing beds
LiOH & Condensing Heat Exchanger	9	Shuttle experience; off-ramp opton	more mass, volume, complexity, radiator size	Shuttle System
ISS CDRA	9	ISS experience; off-ramp opton	more mass, volume, complexity, power	regenerative zeolite molecular sieve beds
Reactive Plastic LiOH	4	saves volume shape contraints vs regular LiOH	Some maturation activities yet to be funded, not regenerative	25+ % more LiOH into the same volume while not reducing the LiOH CO2 removal efficiency or capability
CO2 Reduction				
Sabatier Reactor	6	helps produce water, if no fuel cells	more mass, volume, needs some maturation	produces water and methane
Pressure Control				
Automated Solenoid Valve	9	Shuttle & ISS experience		Used on Shuttle and ISS
Motor-Settable Regulators	5	Continue to regulate atmosphere w/a loss of power to PCS	not used in space yet	have been used in the aircraft industry
Air Distribution				
Standard Ducting, Fans, Filters	9	Shuttle & ISS experience		
Nanofilters	6		need development focus	
Emergency O2 Systems				
QD Masks	9	Shuttle & ISS experience		quick-disconnect face masks
Rebreathers	9	Shuttle & ISS experience		captures & recirculates at least part of a person's exhalation, allowing the person to rebreathe part of his previous breath
Trace Contaminant Control				
Activated Charcoal Adsorption	9	Shuttle experience		Shuttle System
Ambiant Temperature Catalytic Oxidizer	9	Shuttle experience		Shuttle System
ISS Baseline TCCS	9	ISS experience; off-ramp opton	high temp system in small volume	activated charcoal with high temp catalytic oxidizer

Table 20.3.3.1-1: ECLSS Air System Technologies

20.3.3.2 Water Management

ECLSS functional elements provided for CEV and Lander to support acceptable drinking and hygiene water for the crew include:

- Water generation (from fuel cells)
- Water stowage
- Water dumping

- Potability control (microbial control)
- Water dispensing (for drinking & mixing with foods; cold and hot water)
- Crew hygiene water (body wash, hygiene flush, etc.)
- Possible water transfer to Lander

Potable water will be provided for CEV and Lander crews per NASA-STD-3000 water quality standards for the mission duration (see Figure 20.3.3.2-1). For the LDRM-2 TRM, water is generated by the Service Module fuel cells for the CEV capsule and by fuel cells on the ascent stage for the Lander. If Polymer Electrolyte Membrane (PEM) fuels cells are chosen for lunar missions, consideration will need to be given to the removal of gas in the fuel cell-generated water prior to crew consumption. Approximately 1,261 kg of potable water will be produce by the CEV Service Module fuel cells for the mission duration. Additionally the CEV and Lander begin the mission with approximately 25 kg of potable water stored in tanks (metal bellows tanks or non-metallic lined tanks). Excess water is stored in wastewater tanks and dumped from the CEV when full, but no water dumping is planned to the lunar surface from the Lander. To control bacterial nutrients and initial bacteria contamination the water tanks will be processed similarly to ISS Water Processing Assembly (WPA) tanks for internal cleanliness. To control "in-situ" bacteria, water is dosed with iodine and a 0.2-micron filter is installed in the potable water delivery line. No new technology is baselined for this functional element for LDRM-2.

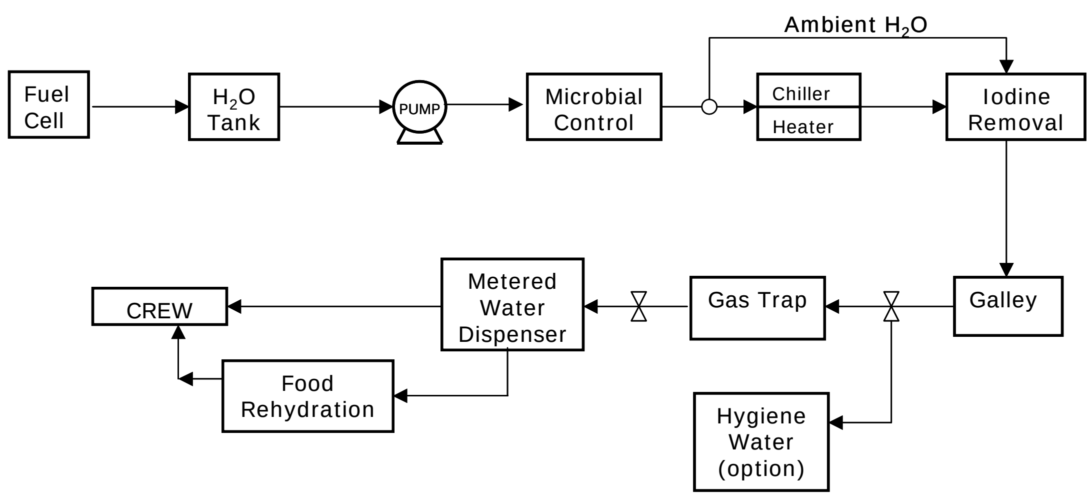

Figure 20.3.3.2-1: LDRM-2 CEV and Lander Water System Concept

No specific hygiene water system is planned for the LDRM-2 TRM. Crew can use drinking water to support hygiene wipes, if necessary. Assuming the waste management system includes only a passive waste collection system, no "flush water" is provided by the CEV or Lander water system. Future trades for the CEV and Lander water systems should consider the option of transferring CEV water to the Lander in the even that the Lander switches from fuel cells to solar arrays for power generation. If fuels cells are not provided on the CEV, stored potable water will need to be carried for the duration of the mission and its potability will need to be maintained. Consideration might then be given to improved biocides or other advanced microbial control systems for water.

Shuttle and ISS potable water systems use iodine for microbial control. Iodine needs to be reduced or totally removed before crew consumption because of the risk of inducing thyroid problems. The iodine level needs to be about 3 ppm to be effective and should be reduced to less than 0.15 ppm for crew water consumption. The equipment required to add iodine and subsequently to remove it makes it desirable to develop other microbial control methods. Other current biocides used in spacecraft are chorine and silver ions. Chorine is very corrosive. Silver ions will plate-out on metal surfaces, are less effective, and are difficult to replace. Ultraviolet light, hydrogen peroxide and electrolytic chlorine generation are other methods to investigate for exploration water systems/Mars evolvability.

Table 20.3.3.2-1 summarizes the technology options considered for the LDRM-2 ECLSS water management system.

ECLSS Technologies	TRL	Pros for LDRM2	Cons for LDRM2	Comments
Water				
Fuel Cell-provided Water	9	Shuttle experience		Shuttle system
PEM Fuel Cell-provided Water			need maturation; require removal of gas in water before crew consumption	will required removal of gas in water before crew consumption
Water Storage Tanks	9			metal bellows tanks or non-metallic lined tanks
Water Microbial Control				
Iodine	9	Shuttle experience	need to remove iodine before drinking water	needs to be about 3 ppm to be effective and should be reduced to less than 0.15 ppm for crew water consumption
Silver	9	Russian experience	Silver ions will plate-out on metal surfaces, are less effective, and difficult to replace	used in Russian systems
Ultraviolet (UV) light	4		need maturation for space applications	
Hydrogen Peroxide	4		need maturation for space applications	

Table 20.3.3.2-1: LDRM-2 ECLSS Water Management Technology Options

20.3.3.3 Waste Management

ECLSS functional elements provided for CEV and Lander to support waste management include:

- Trash stowage/disposal
- Liquid waste collection, storage, dumping
- Solid waste collection, storage, disposal
- Associated odor control

Managing waste and trash for the CEV and Lander will require internal crew cabin volume to accommodate collection activities and storage. The CEV and Lander will be volume-restricted spacecraft. Efforts should be made up front to minimize the amount of packaging materials for food and equipment in order to reduce trash during the mission. Wet and dry trash storage systems in the CEV must be safe (e.g., not allow for hazardous microbial contamination or gas generation) and provided for odor control (e.g., charcoal filters, venting, etc.). Wastewater can be collected and vented from the CEV. It is assumed that water will not be vented from the Lander while on the lunar surface but can be vented while the Lander is in space. It is also assumed that contained trash and used equipment from the surface mission can be left on the lunar surface prior to departure to minimize ascent stage lift-off weight and to allow for volume for return payload.

Crew wastes (feces, urine, wipes, etc.) are collected; urine is sent to the wastewater tank and vented when full along with other wastewater, and feces and associated wet trash is stored for the duration of the mission. Compaction of these wastes may be required if projected storage volumes cannot be accommodated in the CEV and Lander. Crew waste management interfaces are discussed in the Habitation Systems technology report.

Table 20.3.3.3-1 summarizes the technology options considered for the LDRM-2 ECLSS waste management system.

ECLSS Technologies	TRL	Pros for LDRM2	Cons for LDRM2	Comments
Waste Management				
Stored waste	9	Shuttle experience		
Sterilization & Stabilization	5 to 6		need maturation for space applications	
Compaction	5 to 6		need maturation for space applications	volume reduction for storage; can support dewatering
Odor Control				
Filters (charcoal, HEPA, etc.), Venting, Containment	9	Shuttle experience		
Filters (nano, etc.)	6		need maturation for space applications	

Table 20.3.3.3-1: LDRM-2 ECLSS Waste Management Technology Options

20.3.3.4 Fire Detection and Suppression

ECLSS functional elements provided for CEV and Lander to support fire detection and suppression include:

- Smoke Detection Systems
- Suppression/Extinguishing Systems (fixed and portable)
- Combustion Products analyzer
- Personal Breathing Systems

LDRM-2 CEV and Lander should incorporate features to prevent or detect and control fires that could result in loss of vehicle, crew, or injury to crew. Suppressants should not reach toxic concentrations and should also be non-corrosive. Fire suppressant by-products should be compatible with the CEV and Lander life support contamination control capabilities. For LDRM-2 it is assumed that photoelectric smoke detectors mounted in ventilation ducting and the crew cabin (TRL 9, used on ISS) will be included in the CEV and Lander. The suppression system will utilize Halon 1301 for fixed and portable extinguishers (TRL 9, Shuttle system). Depressurizing the CEV or Lander crew cabin is an alternative (contingency) suppression option. No new technology is baselined for LDRM-2 in this area.

Alternative fire detection systems could include ionization smoke sensors (Shuttle sensors) or advanced smoke cell sensors. Alternative fire suppression system could use CO2 (ISS suppressant), Inergen, or a water/foam spray as the suppressant. Inergen is composed of nitrogen, argon, and carbon dioxide. Fire suppression is performed by lowering the oxygen content of the protected area to a point sufficient to sustain human life but insufficient to support combustion.

Support for the fire suppression function includes the use of personal breathing systems during and post-event, as well as a combustion products analyzer for post-event air monitoring. These support functions are discussed in Section 20.3.3.1 – Atmosphere and Pressure Control.

Table 20.3.3.4-1 summarizes the technology options considered for the LDRM-2 ECLSS fire detection and suppression system.

ECLSS Technologies	TRL	Pros for LDRM2	Cons for LDRM2	Comments
Fire Detection &				
Detection Systems				
Photoelectric Smoke Detectors	9	ISS experience		ISS system
Ionization Smoke Sensors	9	Shuttle experience		Shuttle system
Advanced Smoke Cell Sensors	4		need maturation for space applications	
Suppression Systems				
Halon 1301	9	Shuttle experience	issues in small volumes	Shuttle fixed and portable suppressant
CO2	9	ISS experience	issues in small volumes	ISS fixed and portable suppressant
Inergen	5		need maturation for space applications	composed of N2, argon, & CO2; suppression performed by lowering O2 content of protected area to a point sufficient to sustain human life but insufficient to support combustion
Cabin Depressurization	9			
Water/Foam Spray	8		issues with avionics and 0-g	
FDS Support Systems				
Combustion Products Analyzer	9	ISS experience		for post-event air monitoring

Table 20.3.3.4-1: LDRM-2 ECLSS Fire Detection and Suppression Technology Options

20.4 Habitation System Technology Report

Susan Baggerman, NASA/JSC/SF - Habitability and Environmental Factors Office

20.4.1 Description of Habitation System Functions

All crewed spacecraft, including both transit vehicles and planetary habitats, must provide habitation accommodations for the crew. The Habitation System provides basic needs for the crew to live and work in space and includes:

- Maintenance and Repair System
- Emergency Support Equipment
- Waste and Hygiene Interface System
- Housekeeping System
- Crew System
- Galley and Food System

The Habitation System interfaces with the crew, the environmental control and life support system (ECLSS), the power system, and the structure of the vehicle.

Determining how these items may impact vehicle requirements is often considered late in program development. It is important to consider these important crew-related items early because they impact mass, volume, power, and crew performance.

20.4.2 Driving Habitation System Requirements for LDRM-2

Similar to the ECLSS System, LDRM-2 Habitation System considerations are largely tied to the mission durations and crew sizes for the CEV and Lander more specifically, the length of time the crew will occupy the CEV and Lander during any given mission phase. Current LDRM-2 operations concepts estimate the approximate crew residence times in the CEV and Lander as listed in Table 20.4.2-1. The crew size for LDRM-2 is four for both the CEV and the Lander. This dictates the amount of habitation accommodations and consumables needed per day per crew person.

Crew size, as well as duration of mission, will also impact crew cabin “habitable” volume requirements. Habitable volume is defined as the “free” volume provided to the crew within the pressurized volume for their human utilization to live and work. It does not include volume occupied by system hardware, consumables, or stowage. The habitable volume must be determined by assessing free volume that the crew can actually use. “Free space” located in small quantities between elements of system hardware should not be considered when calculating habitable volume. Therefore, habitable volume is not necessarily the equipment volume subtracted from the pressurized volume.

$$\textit{Habitable Volume} \neq \textit{Pressurized Volume} - \textit{Equipment Volume}$$

LDRM-2 Element/Phase	With Crew (days)
CEV Outbound	8.75
CEV Rendezvous w/Lander	1
CEV L1 Loiter	
CEV Rendezvous w/Lander	1.25
CEV Earth Return	4
CEV Totals	**15**
Lander (in space)	7.5
Lander (on surface)	7
Lander Totals	**14.5**

Table 20.4.2-1: Crew CEV and Lander Residence Times

Habitable volume recommendations are based on mission duration and crew size. For the CEV, the longest mission phase is the outbound trip and rendezvous with the lander (approximately 10 days). For a 10-day mission, the habitable volume should be no less than 3.8 m^3 per person[1] to ensure no degradation in crew performance. Given a crew of 4, this results in a habitable volume recommendation of 15.2 m^3. This data was derived from research done by T.M. Fraser, as shown in Figure 20.4.2-1.

[1] From T.M. Fraser, "The Effects of Confinement as a Factor in Manned Space Flight", p. 93, NASA CR-511, 1966. Lovelace Foundation for Medical Education and Research, Albuquerque, NM

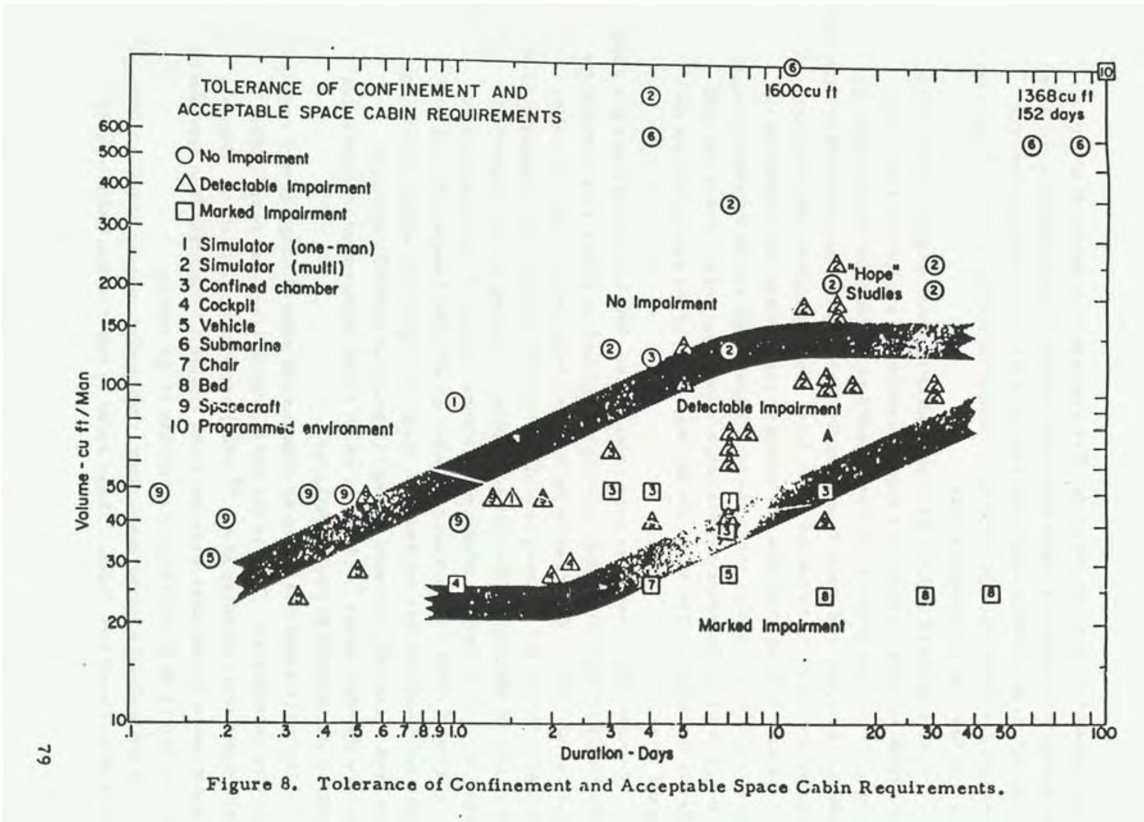

Figure 20.4.2-1: Habitable Volume Requirements

The current CEV design provides approximately 12.6 m^3 of habitable volume (Figure 20.4.2-2). To give this realistic perspective, this is approximately twice the "free" volume provided in the average minivan. While this is in the "acceptable" range, there will be predicted a degradation in crew performance over the duration of the mission given this volume. The current design, however, has not been optimized for mass or volume. Optimization of stowage, seating, suit options, subsystems sizing and habitation elements is critical. Current sizing of the Habitation System technologies is based on high TRL technologies that are available from Shuttle/ISS. However, more advanced technologies may need to be considered to reduce mass/volume and optimize sizing for the CEV and Lander.

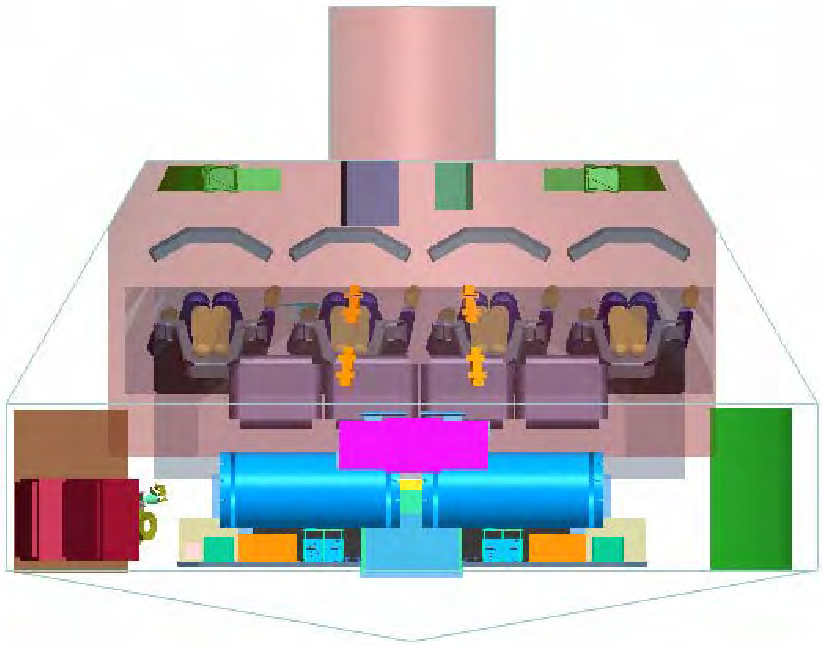

Figure 20.4.2-2: Estimated Habitable Volume provided by the CEV Crew Module

Assumptions: Pink area represents the estimated habitable volume obtained by removing and stowing the crew seats or lowering the seat approximately 16 inches (free space allowed for seat stroke). The estimated volume has been reduced to account for some small equipment that will be attached to the "ceiling" area. Note that the equipment dimensions and placement are not optimized at this time.

In addition to the Habitation System hardware components and habitable volume requirements, consideration needs to be given to the crew interface with all components of the vehicle. This includes, but is not limited to: anthropometry and biomechanics, human performance capabilities and limitations, the effects of natural and induced environments on the crew, crew safety, vehicle architecture and workstation design (including displays, controls, and labeling), human-computer interaction, design of hardware and equipment crew interfaces (e.g. fasteners, connectors, tools), and design of vehicle components for maintainability, facility management (e.g. inventory management, information control). Vehicle components also need to be designed following the principles of standardization and simplicity of design. Consideration must be given to determining what components of the CEV will be automated versus what components will require or allow human input.

The true figure of merit in user interfaces is operability--how fast an untrained person can learn the system, and how quickly and error-free a trained person can use it. Operability can be measured objectively using formal task analysis, where test subjects carry out flight-like work in a simulator, with times measured and errors counted. User interfaces can then be improved in response to task analysis, resulting in a significant increase in crew operability.

Significant lessons have been learned from previous spaceflight and military programs in designing hardware from a human-centered perspective. Incorporating human factors early in the design of a new vehicle in paramount to its success as a human-utilized space system.

20.4.3 Technology Selection and Assessments for CEV and Lander

Technologies considered for the LDRM-2 CEV and Lander are addressed below for each major Habitation System functional area.

20.4.3.1 Maintenance and Repair System

Maintenance and Repair Systems include items such as standard and specialized tool kits and supplies, spare parts and other consumables (filters, fluids, batteries, cables, etc.) for routine maintenance, and instruction manuals (hard copy or electronic).

20.4.3.2 Emergency Support Equipment

Emergency Support Equipment includes items such as survival gear for post landing support (CEV only). Crew launch and entry suit assumptions are discussed in the EVA System technology report.

20.4.3.3 Waste and Hygiene Interface System

The Waste and Hygiene Interface System includes items such as hand/face wash supplies, oral hygiene supplies, grooming/shaving supplies, soaps, shampoos, towels and other personal hygiene equipment. Water necessary for hygiene will be obtained from the CEV Galley water dispenser/rehydrator. Urine and fecal waste collection interface systems can range from urine collection devices (UCDs – "diapers", ala Shuttle), bags (ala Apollo), a passive "potty" system (ala Mir), an active "potty " system (ala Shuttle) and personal urine receptacles. For LDRM-2 a passive toilet system was assumed for both the CEV and Lander. This system can be likened to a "diaper genie" system where wastes are deposited in a bag-lined can with a suitable user interface. The bags can then be individually isolated and stored in an odor control container. While this system is similar to the system used on Mir, development will be required to adapt the system for CEV and Lander crew cabin volumes. A privacy curtain is also assumed equipment for the CEV and Lander. Additional hygiene systems to consider for longer duration exploration mission (Mars mission and longer duration lunar surface stays) include showers and a laundry system.

20.4.3.4 Housekeeping System

Housekeeping Systems include items such as dust abatement systems (this is discussed in more detail in the EVA Systems Technology Report), cleaning agents, disinfectants, microbial control systems, odor abatement systems, vacuum cleaners and trash collection systems and consumables (for other wet/dry trash, filters, used parts, packaging, etc.). The LDRM-2 CEV and Lander should provide the capability for limited in-flight microbial decontamination of surfaces and biological hazards (e.g., microbial wipes) as a contingency.

20.4.3.5 Crew System

Crew Systems include items such as medical support equipment and supplies, clothing, exercise equipment, stowage systems, inventory management systems, illumination systems, restraints and mobility aids, sleep accommodations (restraints), and crew relaxation and entertainment systems (DVDs, games, books, journals, photos, etc.). For LDRM-2, a simple medical kit—similar to those flown on Shuttle—was assumed. This provides all of the basic medical/first aid needs for the crew, but does not provide a defibrillator or other life-saving medical equipment. However, the specific medical equipment required for the CEV will need to be determined by a NASA Flight Surgeon. Due to limited volume, sleep accommodations will be completely stowable. For the CEV, the sleep accommodations will likely closely resemble Shuttle sleeping bags. For the Lander, convertible microgravity/partial-gravity sleep restraints will need to be developed. These will potentially be fashioned after a “hammock” concept.

20.4.3.6 Galley and Food System

Galley Systems include items such as food, food preparation equipment, food warmer, and drinking water systems. Because of the relatively short duration mission for the LDRM-2 CEV and Lander, a minimal galley approach was assumed (i.e., only potable water (hot and cold) and a small food warmer are provided). The current Shuttle/ISS food system, including rehydratable, thermostabilized, and limited fresh food, was assumed for both the CEV and Lander. EVA food and water is also assumed as a key consumable.

No new technology development was assumed for Habitation Systems for LDRM-2. However economies of scale must be realized due to limited crew cabin volumes. Additional engineering efforts to reduce equipment size and innovative packaging and storage solutions will be required. Habitation System technologies to consider for future trades with greater evolvability for Mars missions include improved low heat/low power lighting systems, trash reduction/compaction systems, and advanced medical support equipment, as well as a laundry system, shower system, and dishwashing system.

Table 20.4.3.6-1 summarizes the technology options considered for the LDRM-2 Habitation System.

Habitation Systems Technologies	TRL	Pros for LDRM2	Cons for LDRM2	Comments
Habitation Systems				
Maintenance and Repair Systems				
Standard & Specialized Tool Kits & Supplies	3 to 9	Shuttle and ISS experience	May need tech maturation for tools for new hardware	For basic CEV, can assume tool kit similar to Shuttle or ISS
Spare Parts & Other Consumables (filters, fluids, batteries, cables, etc.) for routine maintenance	9	Shuttle and ISS experience		
Instruction Manuals (hard copy or electronic)	9	Shuttle and ISS experience		
Multi-use Tools	3 to 9	Volume and mass efficiency; gain experience for Mars mission	Tech maturation required	
Advanced Electronic Support for Instructions & In-situ Training	3 to 9	Volume efficiency; improved crew performance/safety because of In-Situ Training; gain experiences for Mars mission	Tech maturation required	
Regenerable Filters &Nanofilter Systems				
Crew Emergency Support Equipment				
Survival Gear for Post Landing & Crew Escape Support	9	Shuttle experience		
Emergency Oxygen Systems (see ECLSS - Air)				
Pressure Suits (see EVA)				
Specialized Altitude/Environment Protection Suits & Masks	2 to 4	Improved crew safety	Tech maturation required	
Advanced Emergency Breathing Systems	2 to 4	Improved crew safety	Tech maturation required	
Improved Tracking & Communication Devices	2 to 4	Improved crew safety	Tech maturation required	

Waste & Hygiene Interface Systems				
Bags (for feces collection)	9		Duration makes crew interface undesirable	
Adult Diapers	9		Duration makes crew interface undesirable	
Passive Toilet Systems	9	Mir/ISS (Russian toilet) experience		Crew reviews of Mir and ISS Russian toilet have been very favorable
Active Toilet Systems	9	Shuttle experience	High mass/volume	May be overly complex for CEV
Active Toilet Systems that interface w/Waste Processing System	4	Gain experience for Mars mission	High mass/volume; tech maturation required	
0-g or Hypo-g Laundry System	1 to 2	Gain experience for Mars mission	Significant tech maturation required	Will be necessary for a Mars mission - may consider small unit on CEV as DTO
Improved Spacecraft-compatible and Crew-friendly Shampoos & Soaps	3	Gain experience for Mars mission	Tech maturation required	
User Interfaces to a Water Recovery System (e.g., Sink, Dispensers, etc.)	3 to 9	Gain experience for Mars mission; decrease volume of water needed on CEV	Tech maturation required	Would be very useful to optimize water usage on CEV and provide proof-of-technology for future Mars mission
0-g or Hypo-g Shower System	2 to 4	Gain experience for Mars mission	Significant tech maturation reqd	Previous attempts (Mir, Skylab) have not been well-received by the crew
Privacy Curtain	6	Low mass/volume provision of privacy	Minimal tech development required	Conceptual design has been developed for ISS but has not been built or flown
Housekeeping Systems				
Dust Abatement Systems				
Cleaning Agents				
Disinfectants				
Surface Microbial Control Systems				
Vacuum Cleaners	9	Shuttle and ISS experience		Potentially look at optimizing system to minimize mass/volume
Passive Trash Collection System - via trash bags	9	Shuttle and ISS experience		

Active Trash Collection System - via trash compactor	3 to 6	Gain experience for Mars mission	Tech maturation required	
Crew Systems				
Medical Support Equipment (see crew medicine assessment)	3 to 9	If only Shuttle med kit is reqd, then Shuttle experience	Additional equipment may be reqd because of health impacts of Lunar environment	This topic needs to be addressed with the NASA Flight Surgeons
Current Clothing	9	Shuttle and ISS experience		Duration of CEV doesn't necessitate a change--crew likes current system
Methods to Reduce Clothing Mass (longer wear, self cleaning, etc.)	2 to 4	Gain experience for Mars mission; reduce CEV mass/volume	Tech maturation required	
Maintainable, Repairable & Reconfigurable Exercise Equipment	2 to 4	Critical need for Mars mission	Tech maturation required	ISS experience has shown this to be a very significant issue for long-duration spaceflight
Innovative Storage Solutions	2 to 4	Critical need for Mars mission	Tech maturation required	ISS experience has shown this to be a very significant issue for long-duration spaceflight
Inventory Management Systems	3 to 9	Simple system needed for CEV; critical need for Mars mission	Tech maturation required for more complex system needed for Mars mission	Industry methods exist and should be addressed when designing the CEV
Current Illumination Systems	9	Shuttle and ISS experience	Power, reliability	
Low Heat/Power Illumination Systems	3 to 6	Efficient for CEV operations and future Mars mission	Tech maturation required	LED systems currently being developed should be considered for CEV
Restraints and Mobility Aids	3 to 9	For 0-g, Shuttle and ISS experience; for hypo-g, Apollo experience		Limited experience in hypo-g from Apollo--this will be the greatest area of development needed
Convertible Micro-g/Partial-g Sleep Restraints	3 to 9	Volume efficient for CEV; Shuttle and ISS experience for 0-g restraints	Tech/design maturation required	

Crew Relaxation & Entertainment Systems	3 to 9	Shuttle and ISS experience with current technology		Should consider more state-of-the-art systems (e.g. higher memory, etc.) for Lunar or Mars mission
Galley Systems				
Extruder	2 to 4	Gain experience for Mars mission		Uses shear force, increased temperature & increased pressure to convert plant material into edible food ingredients; increases available food texture & variety
Grain/Flour Mill	2 to 4	Gain experience for Mars mission		Convert various food crops to flour
Soy Milk Machine	2 to 4	Gain experience for Mars mission		Needed for processing soybeans to milk, tofu (used as meat substitute) etc.
Automatic Tofu System	2 to 4	Gain experience for Mars mission		
Food Processor	2 to 4	Gain experience for Mars mission		
Bread Machine	2 to 4	Gain experience for Mars mission		
Refrigerator	6	Based on concept created for ISS		Provides significantly increased capability for fresh food
Freezer - active	6	Based on concept created for ISS		Provides significantly increased capability for fresh food
Freezer - passive	6			
Dehydrator	2 to 4	Gain experience for Mars mission		
Press (oil extraction hydraulic)	2 to 4	Gain experience for Mars mission		
Food Warmer	9	ISS experience		Design used on ISS reheats thermostabilized packages
0-g or Hypo-g Dishwashing System	2 to 4	Gain experience for Mars mission	Significant tech maturation reqd	
Improved Food Packaging	3 to 6	Decrease stowage volume by increasing efficiency		Could help improve insufficient habitable volume issue
Food Storage/Preservation Systems	3 to 9			

Rehydratable & Thermostabilized Foods	9	Shuttle and ISS experience		
Longer Shelf-life Foods	3 to 6			Necessary for future Mars trip
Advanced Menu Development	3 to 6			

Table 20.4.3.6-1: LDRM-2 Habitation System Technology Options

20.5 Active Thermal Control System Technology Report

David Westheimer, NASA/JSC/EC – Crew and Thermal Systems Division

20.5.1 Subsystem Description

20.5.1.1 Primary Functions

The primary function of the Active Thermal Control System (ATCS) is to control the temperature of the cabin air and other key pieces of hardware. Thermal control system hardware acquires energy from the cabin air or heat producing hardware, transfers that energy within the vehicle, and then rejects the energy into space. The ATCS often provides humidity control for the cabin. However, humidity control for the baseline CEV will be performed by a regenerative combined CO2 and moisture removal system.

It was also assumed that this mission would not require refrigerators or freezers for preserving science samples.

20.5.1.2 Key Design Parameters

The primary design drivers for the ATCS include the total thermal load and the available environment for heat rejection. This study assumed a nominal total thermal load of 9 kW. This is the sum of the total vehicle power, waste heat from the fuel cells, and crew metabolic heat. The same heat load was assumed for sizing reentry cooling and for the Lander design.

The CEV will have to reject heat via radiation throughout most of the mission. This includes in the Low Earth Orbit (LEO) environment as well the Lunar transit mission phases. Radiation sink temperatures of 245 K and 100 K were assumed for LEO and Lunar Transit, respectively. Radiation sink temperature is defined with the following equation:

$$T_{\sin k} = \left(\frac{\frac{\alpha}{\varepsilon} q_{solar} + q_{IR}}{\sigma} \right)^{1/4}$$

where α is solar absorptivity, ε is infrared emissivity, q_{solar} is incident solar radiation flux, q_{IR} is incident infrared radiation flux, and σ is the Stefan Boltzman constant. Radiators were assumed to use 10 mil thick silver Teflon as the optical surface. The surface properties of silver Teflon can be approximated as $\alpha = 0.123$ and $\varepsilon = 0.868$.

For sizing purposes, the CEV was assumed to have radiators covering a cylindrically shaped Service Module with one panel, or one third of the radiator surface, rejecting the warm LEO radiation sink and two panels, or two thirds of the radiator surface, rejecting to the cold 100 K radiation sink. This could be represented by the following equation:

$$Q_{radiation} = \sigma\varepsilon\left(\frac{1}{3}A\right)\left(T_{radiator}^4 - T_{\sin k-hot}^4\right) + \sigma\varepsilon\left(\frac{2}{3}A\right)\left(T_{radiator}^4 - T_{\sin k-cold}^4\right)$$

Where A is the radiator surface area and $T_{radiator}$ is the average radiator surface temperature.

In addition to radiation sink environments, it was assumed that a water boiler was used for heat rejection during the initial stages of reentry. The 9 kW heat load would be rejected for 180 minutes by evaporating water and venting the vapor to space.

Additional requirements that impact ATCS design include the capability for contingency EVAs. Since the baseline reference design does not include an airlock, the cabin must be depressurized. This requirement impacts the ATCS because hardware that must operate during the cabin depress would have to be cooled by a coldplate.

Redundancy to achieve acceptable fault tolerance to failures is an additional design parameter. Active Thermal Control Systems typically consist of two redundant fluid loops, each with redundant pumps.

20.5.1.3 ATCS Resource Requirements

The primary vehicle accommodations required for the ATCS are locations to mount radiators. The baseline mission design shows that there is enough area for body mounted radiators on the outer surface of the SM and that it is possible to mount radiators on the Lander that require little or no deployment.

The ATCS will also require water for the water evaporators on the CEV and Lander. It is assumed that this will be provided by the fuel cells.

Resource conservation will be primarily a function of the heat load during different mission phases. If major portions of the vehicle can be powered down, the ATCS can also be partially powered down. Since redundant loops are the baseline design, one loop could easily be powered down during periods of reduced operation.

The primary ATCS concern for mission phases with reduced heat load is the fluid in the radiators can freeze. This can be problematic depending on the properties of the fluid and the design of the radiator fluid passages. Apollo addressed this issue by using a glycol based thermal control fluid that became increasingly viscous at low temperatures and by using a stagnation radiator design that allowed portions of the radiator to gracefully change from being stagnant to allowing fluid to freely flow.

20.5.1.4 Interfaces and Synergy with Other Vehicle Systems

The ATCS will primarily interact with the ECLSS, Power, Propulsion, and Avionics systems. In the current baseline design the ECLSS will perform the function of cabin humidity control allowing for decreased complexity in the cabin air heat exchanger. Fuel cells will be used to generate power, and water is an important by product of this system. Excess water can be used in a water evaporator to reject heat into space allowing for heat rejection from the CEV capsule following separation from the CEV Service Module up until re-entry. The water evaporator can also be used during peak periods on-orbit or on the lunar surface, thus allowing for a smaller radiator. The ATCS will interface with the propulsion system if a cryo-cooler is required in order to condition propellants. The ATCS will interface with the avionics via coldplates to collect waste energy even when the cabin is depressurized.

All of these interfaces place temperature requirements on the ATCS. Propulsion cryo-coolers and humidity control systems require low-temperature fluid loops. Avionics can typically be run at higher temperatures. These temperature requirements are important to the ATCS sizing because increasing the temperature of the radiators has a fourth order effect on the radiator size as mass, as shown in Equation 2.

20.5.2 Technology Options

Several technology options were considered for the baseline design. Table 20.5.2-1 summarizes several available technologies, their TRL level, benefits, and drawbacks.

Coldplates have been used in almost every human rated space vehicle. They are simple pieces of hardware that transfer heat, usually from an avionics component to a fluid loop. Integrating the structural support for the avionics and the cooling into a single piece of hardware can provide vehicle level mass savings. A lightweight composite coldplate shelf (TRL 3) has been developed at JSC that utilizes high strength and high thermal conductivity composite materials. This coldplate shelf was originally developed for an ISS payload rack application but the concept and materials could be applied to most coldplate applications.

The baseline design uses the CO2 removal system to also remove moisture from the air. Alternatives to this design include using a condensing heat exchanger and rotary separator (as is done on Shuttle and ISS) or a porous metal humidity control device as was used on Apollo. Both of these designs are TRL 9 technologies. Clogging, required maintenance, and a complex design are issues with the ISS and Shuttle technologies. The Apollo condensing heat exchanger had problems with maintaining the porous metal wet and rewetting the heat exchanger if it dried out. Currently, technology development for an advanced porous media condensing heat exchanger is being performed as a collaborative effort between JSC and Glenn Research Center. Hardware development is focusing on utilizing new porous materials and microbial growth techniques to develop a porous media heat exchanger that has improved reliability and life. This technology is currently at TRL 3.

Technology	**TRL**	**Pros**	**Cons**	**Comments**
Integral Coldplate Structures	9	Mass savings	Coldplates must be designed for specific applications	Currently done for high heat load hardware
Composite Coldplate Structure	3	Takes advantage of low mass composites	Current design is for a payload rack application	Payload rack application might not be appropriate for CEV
Advanced Porous Media Condensing Heat Exchanger	3	Improved reliability		May not be needed due to ECLSS CMRS
Condensing Heat Exchanger and Rotary Separator	9	Proven technology	Requires maintenance and is prone to clogging	May not be needed due to ECLSS CMRS
Fault Tolerant Heat Exchanger	5	Risk reduction for systems that use hazardous fluids		Is not needed if a single loop designed is used
Alternative Heat Transfer Fluid	5	Allows for a single loop design and greatly reduces ATCS risk and cost		Potential fluids include aqueous propylene glycol, Galden, HFE 7100, Fluorinert 72
Dual Loop Design	9	Shuttle and ISS use this configuration	External fluids are hazardous	Shuttle uses Freon 21 and ISS uses Ammonia for external loop, both use water for internal loop
Lightweight Radiators	5	Mass savings for radiators and for mounting structure		Several technologies currently in mid-TRL range
Structural Radiators	3	Mass savings		Integrates radiator into vehicle skin, could be load bearing
Shuttle Radiators	9	Simple, flight proven design	More massive than advanced technologies	Aluminum honeycomb technology
ISS Radiators	9	Radiator panels are simple and flight proven	Complex, massive deployment mechanism	Aluminum honeycomb technology
Flash Evaporator System (FES)	9	High heat flux evaporator	Complex device, has experienced on-orbit problems	Problems seem to have been resolved
Water Boiler	9	Several different flight proven designs exist		Used on Mercury, Apollo CM, Shuttle
Sublimator	9	Used on EMU, and Apollo LEM	Sensitive to contamination	
Multi-Fluid Evaporator	3	Can be used throughout reentry		Utilized fluids with different saturation pressures
Water Membrane Evaporator	5	Simple design, utilized membranes for fluid management	Massive compared to FES	
Advanced Sublimator	4	Less sensitive to contamination		Developed as part of X-38 program

Table 20.5.2-1: ATCS Technology Options

A vehicle can either use a single loop or dual loop ATCS design. Currently, both the Space Shuttle and ISS use a dual loop architecture utilizing water as the heat transport fluid internal to the cabin and a refrigerant (Freon 21 on the Shuttle and Ammonia on ISS) to flow through the radiators. If a low freezing temperature, non-toxic, and non-flammable fluid was available that could be certified for space flight, a single loop ATCS could be used. A single loop architecture eliminates an interface heat exchanger and additional pumps while reducing the risks associated with using a toxic fluid. This risk reduction not only applies to on-orbit operation, but will greatly reduce development cost for thermal control hardware. Aqueous propylene glycol mixtures have received serious consideration for ATCS use. This fluid has been used in advanced radiator testing and is commonly used in terrestrial applications. There are several other fluids that are commercially available that could be applicable for future vehicles. Galden, HFE 7100, and Fluorinert 72 are some other possibilities. These fluids all have low freezing temperatures, are non-flammable, and non-toxic.

If a dual loop ATCS is implemented, a Fault Tolerant Heat Exchanger would reduce the operational risk associated with using a hazardous refrigerant. This liquid-to-liquid heat exchanger provides two physical barriers between fluid loops. Current technology development through JSC has brought this concept to TRL 5.

Radiators are key targets for technology advances because they are typically the most massive component in the ATCS. Lightweight radiators have been a major focus of technology development at JSC for the past ten years. Several of these technologies were lightweight, flexible, and originally designed to be a body mounted radiators on Transhab. Prototypes of a Flexible Metal Fabric Radiator have been manufactured and tested in thermal-vacuum conditions. The prototype had a 3.4 kg/m^2 ratio of mass to surface area. This compares to 10.6 kg/m^2 for the Space Shuttle radiators. This technology is TRL 5. In addition to the Flexible Metal Fabric, several other lightweight radiators have been developed in recent years that have taken advantage of technologies such as laminates and lightweight carbon composites. These technologies range from TRL 3 to 5. The most recent developments in radiator technology have been to develop load bearing structural radiators. These radiators would be integrated into the structure of the vehicle, as opposed to being mounted on an existing structure, producing a mass savings. The Structural Radiator is currently at TRL 3.

Evaporative heat rejection devices will be required for the CEV and Lunar Lander. The Space Shuttle uses a Flash Evaporator System (FES). This is a flight proven technology (TRL 9) that sprays a water onto a heat transfer surface that is connected to the ATCS loop. When the fluid evaporates, it takes the latent heat of vaporization from the heat transfer surface. The energy is rejected with the vapor as it is vented into space. Several alternatives exist to the FES. The Apollo Command Module used a water boiler, the Apollo Lander and the current EMU use sublimators, and the Shuttle also uses an ammonia boiler once it enters the Earth's atmosphere. All of these are flight proven technologies, TRL 9.

Technology development has taken place for a Multi-Fluid Evaporator, a membrane based evaporator, and an advanced sublimator. The Multi-Fluid Evaporator could operate with different heat transfer fluids, enabling its operation in space, reentry, and post-landing environments. The membrane based evaporator utilized hydrophilic and hydrophobic membranes to distribute fluid over hot surfaces and allow vapor to vent to space. The X-38 program began development on an advanced sublimator. Sublimators operate by distributing water over a porous media. The water freezes and then sublimates, due to the relationship between the operating conditions and the triple point of the fluid. As the water sublimates, contaminants are left behind that can clog the pores in the porous media. The X-38 design used a technique to distribute the water over the porous media that reduced the impacts of contamination.

An alternative heat transfer fluid that enables a single loop design and radiators that provide mass savings provide the largest benefits to the CEV baseline design. Integral coldplate structures should be used as appropriate. However, this is more of a design challenge than a technology development task. Additional development work should be performed on evaporators and sublimators to improve on lessons learned from past programs.

20.5.3 Recommended LDRM-2 Subsystem Design Approach

20.5.3.1 Technology Selection

Flight proven technologies were chosen for fluid pumps, coldplates, heat exchangers, evaporators, and radiators. These technologies were primarily taken from the Space Shuttle and ISS. Sizing calculations were performed by scaling based on the Space Shuttle Freon Pump (TRL 9) and ISS Internal Thermal Control System (ITCS) coldplates (TRL 9). Metal bellows accumulators, which are used on both the Space Shuttle and ISS are a TRL 9 technology. The water evaporator was scaled from the Space Shuttle Flash Evaporator System (FES) (TRL 9). It will require additional analysis to determine if the FES is the appropriate technology for these missions. A variety of water boilers, sublimators, and evaporators have been flown on space vehicles and are at TRL 9. Modified versions of these evaporative heat rejection devices could easily be developed to TRL 6 in the desired timeframe.

The only advanced technologies chosen for the sizing calculations in this study were an advanced radiator and the use of an aqueous propylene glycol as the heat transfer fluid. Radiator designs are vehicle dependent and choosing a technology from Shuttle or ISS would not be realistic. Radiator sizing was based on an advanced Flexible Metal Fabric Radiator with a Silver Teflon coating. This radiator technology is currently at TRL 5. An alternative heat transfer fluid was selected, enabling a single loop architecture. Using an aqueous propylene glycol solution for the heat transfer fluid is also a TRL 5 technology, although it is commonly used in terrestrial applications.

A combined ATCS will be utilized for the CEV and SM. These vehicles will combine to manage the 9 kW heat load. The ATCS will be split between the two vehicles with the pumps, cabin heat exchanger, a water evaporator, and most of the coldplates in the CEV. Radiators and additional coldplates will be installed on the SM. Figure 20.5.3.1-1 is a schematic of the combined CEV and SM ATCS. No low altitude reentry or post-landing heat rejection hardware was specified for the baseline design. This could be performed either by utilizing the thermal capacitance of vehicle systems, like the water tanks, that are already in place or by selecting an alternative evaporative heat rejection device that uses a high saturation pressure fluid, such as a Freon.

The Lander is divided into Ascent and Descent stages. It is recommended that the entire ATCS be located on the Ascent stage because it is anticipated that a significant amount of heat must be managed as the Ascent stage returns to dock with the CEV. Figure 20.5.3.1-2 is a schematic of the Lander ATCS.

The baseline mission requires propulsion stages in addition to the CEV, SM, and Lander. These will also require some type of thermal control. Detailed analysis and design of a thermal control system was not pursued as part of the baseline design. However, passive thermal control methods, such as heat pipes and radiation will be used as much as possible.

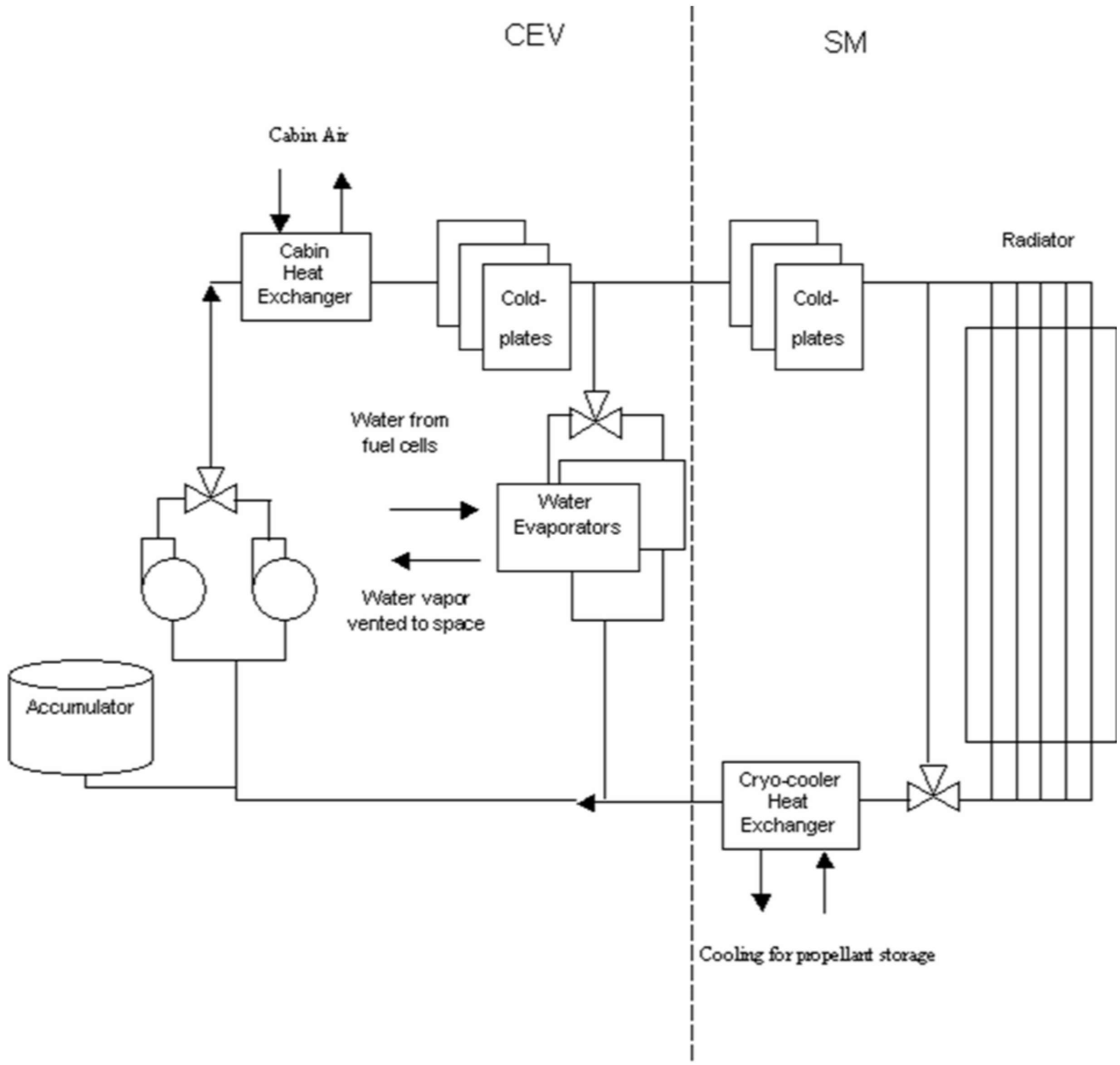

Figure 20.5.3.1-1: CEV Command Module/Service Module ATCS Schematic

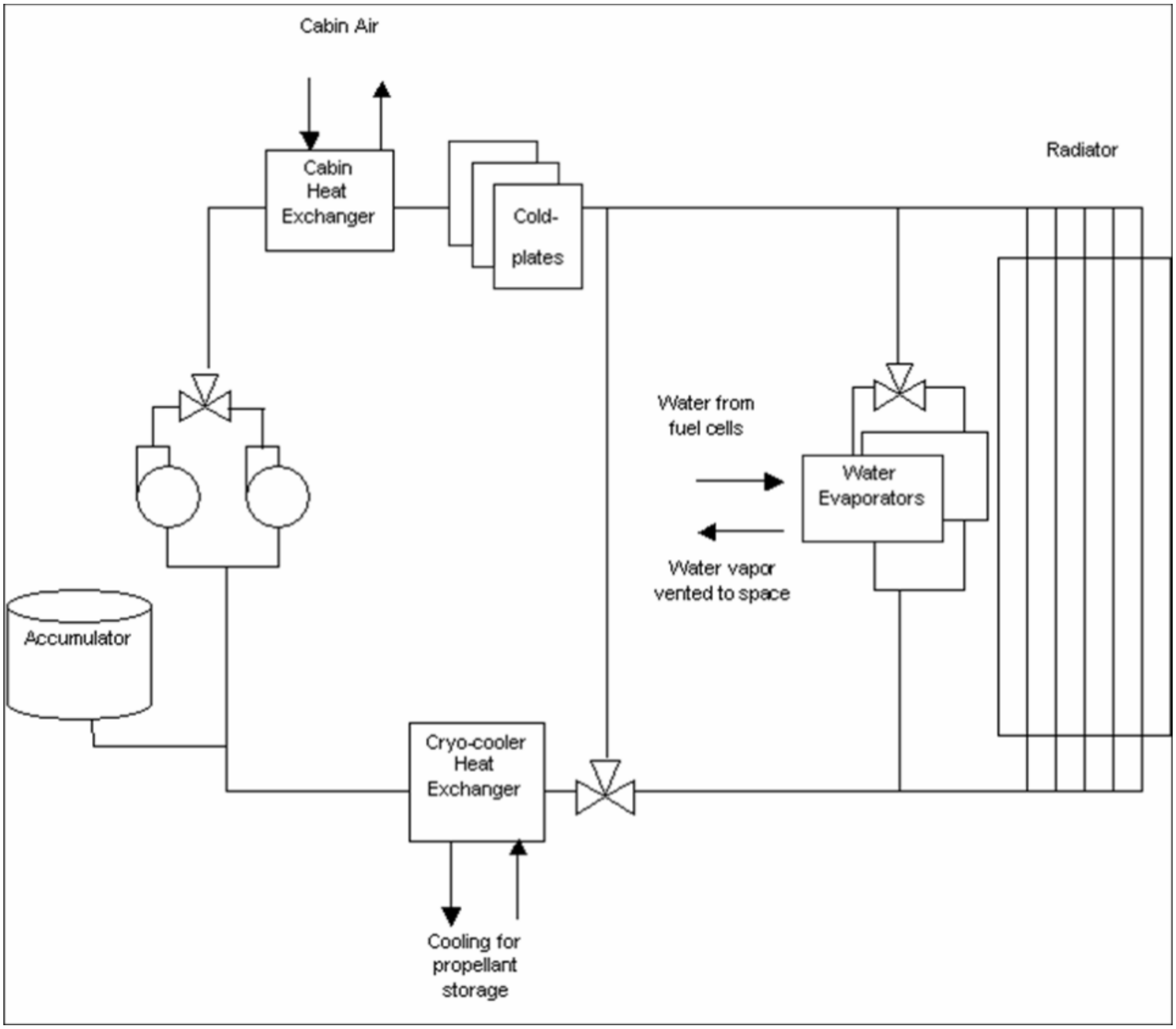

Figure 20.5.3.1-2: Lander ATCS Schematic

CEV CM/SM			
	Mass (kg)	**Power (W)**	**Volume (m³)**
Pumps & Accumulators	4.8	198	0.0188
Fluid	11.2	-	-
Heat Exchangers	17.3	-	0.0172
Coldplates	85.9	-	0.2094
Miscellaneous (Instrumentation, Lines, Filters, etc…)	85.2	-	0.0283
Evaporator	20	287	0.07
Radiator	109.7	-	0.392
Insulation (MLI)	83	-	0.41
Totals	**417.1**	**485**	**1.1457**

Lander			
	Mass (kg)	**Power (W)**	**Volume (m³)**
Pumps & Accumulators	4.3	176.5	0.0167
Fluid	10.0	-	-
Heat Exchangers	17.3	-	0.0172
Coldplates	85.9	-	0.2094
Miscellaneous (Instrumentation, Lines, Filters, etc…)	80.2	-	0.0274
Evaporator	20	51	0.07
Radiator	119.5	-	0.43
Insulation (MLI)	73	-	0.36
Totals	**410.2**	**227.5**	**1.1307**

Table 20.5.3.1-1: LDRM 2 Hardware Summary

20.5.3.2 Potential Design Breakpoints

The CEV and SM do not appear to have major break points that would impact the design of the vehicle. One possible design issue would be if the SM does not have enough available surface area for the radiators. This would lead to a deployable radiator design that would require additional mass, power, complexity, and risk. This scenario seems unlikely.

Design breakpoints for the Lander would include the availability of water for evaporative heat rejection, available locations to mount sky-facing radiators, and the dimensions available for radiators within the launch shroud. Two design variations exist if water is not available from fuel cells: additional water storage or additional radiator area. A deployable radiator will be required if mounting locations for sky facing radiators are not available or if the radiator is too large to fit within the launch shroud.

20.5.4 Mars Spiral Development

Several technologies that could be used in the Lunar missions are also applicable for Mars missions. Common hardware such as pumps, valves, accumulators, instrumentation, coldplates, and heat exchangers will be applicable to both Lunar and Mars missions.

The use of new heat transfer fluids, like aqueous propylene glycol solutions, could be used for Mars missions. The radiators developed for the CEV/SM and the Lander could also be used for Mars transit vehicles and Landers. Mars landers will have additional design drivers due to the Mars environment. They will be affected by the atmosphere, wind, and dust deposition. Evaporative heat rejection devices, such as sublimators and multi-fluid evaporators, could also be applicable to Mars mission. Evaporative devices require consumables, but evaporative heat rejection could be used during short periods in the mission or if a water surplus exists from fuel cells. Ascent and descent into an atmosphere, which is present on Earth and on Mars, are common mission phases that require the use of an evaporative heat rejection device. Reliability of these devices will be more critical on a Mars missions due to the longer duration and greater distance from home. This improved reliability also applies to the humidity control. The porous media condensing heat exchanger would be a prime candidate for a Mars mission. The condensing heat exchanger not only would provide improved reliability for a longer mission but would also provide for collection of humidity condensate that would then be recovered through the ECLSS. Water recovery on long duration missions is extremely critical. All of these technologies would make excellent candidates for flight testing during LRDM – 2.

20.6 Extravehicular Activity System (EVAS) Technology Report

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20.6.1 Description of EVAS Function

Spacecraft and planetary habitats will require an extravehicular activity (EVA) system (EVAS) to support crews for in-space and planetary surface EVA operations. Typically, an EVAS provides the crew the capability to conduct work outside the pressurized environment of the spacecraft. An EVA suit, the primary component of the EVAS, is a highly technical system - essentially a miniature spacecraft. Development of a new EVA suit, and the supporting systems and infrastructure to support the LDRM-2 requires technology advancements similar to those required in the development of a new space vehicle.

The EVAS includes:

- EVA suit consisting of a pressure garment and a portable life support system (PLSS)
- Airlock
- EVA tools and mobility aids
- Manned rovers
- Systems integration with vehicles
- EVA informatics

Key interfaces between the EVAS and the CEV and Lander include the Environmental Control and Life Support System (ECLSS), crew and human factors, power systems, airlocks, and operations and training. A primary dependence for EVAS is the selection of the cabin atmosphere pressure and constituents. Another LDRM-2 key dependency for EVAS is the oxygen (O_2) supply system for use in the EVA PLSS.

20.6.2 Driving Requirements Affecting EVAS

The EVAS requirements will be driven directly by the requirements for the new exploration missions beyond Low Earth Orbit (LEO), the Moon and Mars. A new EVAS will use revolutionary new technology, common components, human-robotic cooperation, and a flexible architecture for multi-destination operation with minimal system reconfiguration. Requirements for the EVAS will include:

– Lightweight, highly mobile suits and dexterous gloves to increase crew productivity, enable long-duration missions and high EVA use rates, mitigate crewmember injury, and fit a wide range of crewmember sizes

– Maintainable PLSS architecture that is easily reconfigurable to enable multiple destinations

- Integrated human-robotic interaction capability to increase safety, efficiency, & productivity
- State of the art communications and computing capability for multi-media crew-ground interaction (e.g., integrated communications, high tech information systems, and heads-up displays)
- Operating pressure regimes decrease EVA overhead by minimizing pre-breathe and exercise protocol
- Advanced thermal control to increase crew comfort, decrease consumables, and enable multiple destinations (e.g., aerogel insulation, active cooling and heating)
- Common hardware with other vehicle systems to increase vehicle safety & decrease mission mass through common sparing (e.g., power, communication, instrumentation, life support, thermal control)

LDRM-2 EVAS requirements are tied to specific mission requirements, durations, and crew sizes for the CEV and Lander; more specifically, the requirement to perform contingency EVAs from the CEV and nominal planetary surface EVAs from the Lander. Current LDRM-2 operations concepts estimate the approximate crew residence times in the CEV and Lander as listed in Table 20.6.2-1.

LDRM-2 Element/Phase	With Crew (days)	Without Crew (days)
CEV Outbound	10	
CEV Rendezvous w/Lander	1	
CEV L1 Loiter		13
CEV Rendezvous w/Lander	1	
CEV Earth Return	4	
CEV Totals	**16**	**13**
Lander (in space)	7.5	54
Lander (on surface)	7	
Lander Totals	**14.5**	**54**

Table 20.6.2-1: Crew CEV and Lander Residence Times

EVAS operations during the in-space phases for the CEV and Lander need to be considered only for contingency EVAs. EVAS operations during the planetary surface phase are part of nominal surface operations. The crew size for LDRM-2 is four for both the CEV and the Lander. This dictates the number of EVA suits that will be required and the quantity of EVAS consumables needed per EVA per crew person. EVA Systems requirements (including the airlock on the Lander) will also drive ECLSS consumables and subsystems sizing. The maturation of current EVA regenerative portable life support systems technologies could benefit CEV and Lander applications by reducing mass and volume and providing higher reliability.

Another driving requirement for CEV and Lander EVAS is the availability of consumables for the EVA suit power system, such as fuel cells. This requirement affects the EVAS power system technology selection. Lunar mission safety assumptions are also driving requirements for the CEV and Lander EVAS designs. Redundancy and reliability requirements will drive EVAS subsystem mass and volume requirements. For the CEV, four Crew Escape System (CES) suits that could be used to perform a contingency EVA would be available. For the Lander, four EVA suits would be stowed and be available for conducting planetary surface EVA operations by two crewmembers at a time using a Lander airlock. Contingency EVA operations from the CEV drive the requirement that the cabin be able to be fully depressurized.

Additional driving requirements for CEV and Lander EVAS include surface lighting conditions, dust mitigation, crew medical requirements, and the number of crew cabin EVAs and repressurizations assumed for the CEV and Lander (drives the quantity of gas consumables). Two full cabin repressurizations were assumed for the CEV and eight for the Lander.

20.6.3 Technology Selection and Assessments for CEV and Lander

EVA technologies considered for the LDRM-2 CEV and Lander are addressed below for each major EVAS functional element.

- Atmosphere Revitalization
  - Life support systems for suit mounted portability in a variety of gravity and pressurized environments
  - Components for high reliability, ease of maintenance, low mass, minimal volume, low power, long life and full regeneration without consumables
  - Efficient removal of CO2, humidity and trace contaminants
  - Compact and long lasting storage, supply and recharging of O2
  - Closed loop thermal heating and cooling devices which do not consume scarce resources and have improved means of heat transfer.
- Environmental Protection
  - Lightweight, highly mobile and robust EVA pressure suit
  - Low bulk thermal insulation for both vacuum and non-vacuum environments
  - Dust, chemical contamination and self sealing puncture resistant materials
  - Passive and/or active portable radiation protection
  - Lightweight and high strength structural materials integrated into the suit backpack, pressure garment, and bearings
- Human Integration
  - System level integration, modeling and prototyping to aid cost effective and efficient design of EVA crew interfaces

– Development of conceptual designs and mockups for airlock/hatch/controls should allow mobility testing of healthy and incapacitated crewmembers and feature minimal gas/power loss for frequent crew and equipment transfer

– Portable multi-sensory information displays and controls which work in harsh environments and are compatible with hands free input/output.

A top-level breakdown structure for the EVAS for LDRM-2 Lander is seen in Figure 20.6.3-1.

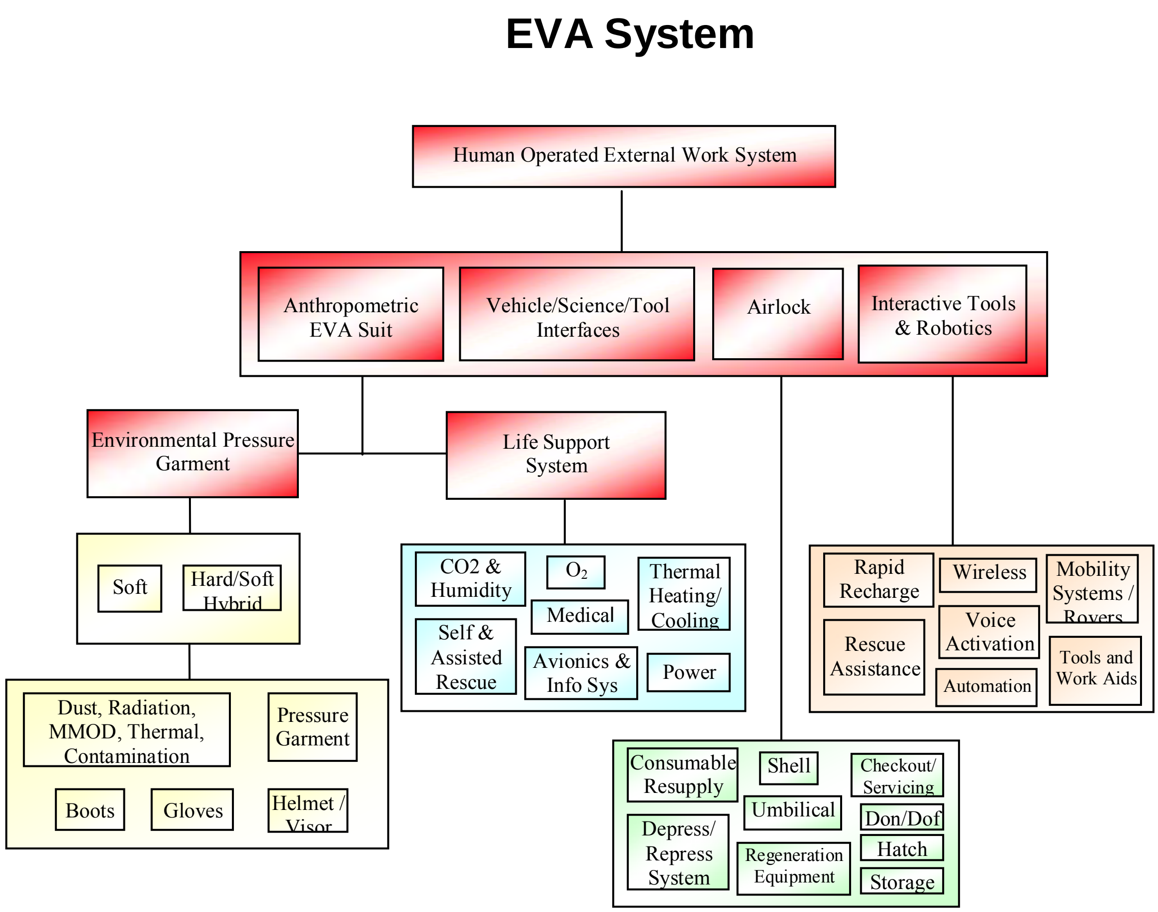

Figure 20.6.3-1: EVAS for LDRM-2 Lander

The CEV and Lander atmospheric composition is assumed to be 27-30% O_2 and 70-73% nitrogen (N_2). In order to perform two contingency EVAs, two full cabin repressurizations were assumed for the CEV. To meet the requirement for planetary surface EVAs, eight repressurizations were assumed for the Lander.

The EVAS consists of “core technologies” that apply to LEO, the Moon, and Mars and “applicable lunar technologies” that apply to the LDRM-2.

Core technologies include:

System architecture:	Flexible, lightweight, maintainable PLSS
	Lightweight structures
	Integral suit/PLSS interface
	Rapid recharge and checkout
CO2 Removal:	Rapid cycle amines
	Venting membranes
Thermal Control:	Freezable/gas gap radiators
	Micro refrigeration/heating system
	Auto cooling control
	Water membrane evaporator
	Phase change materials
	Conduction cooling garment
Interfaces:	Human-robotic work aids
	Airlock
	Crew Escape Systems
Power:	Batteries
	Fuel Cells
Suits:	Lightweight structures
	Mobility systems
	Gloves
	Visors
	Zero-prebreathe
	Aerogel thermal insulation
	Heads up display
	Integrated high capacity communications
	High reliability fans, pumps, actuators, sensors

Applicable lunar technologies include:

Environmental protection: Dust containment and removal

Field Recharge and In-the-Field Servicing: O2 connectors

Field serviceable packs

Interfaces:	Human-robotic work aids
	Human Rovers
Airlock:	Lightweight structures
	Reduced consumables
	Dust mitigation and removal systems

20.6.4 EVAS Overview, Technology Options, Trade studies, and Issues

An EVAS overview, technology options, trade studies, and issues are provided for consideration in the spiral development of the LDRM-2 CEV and the Lander.

20.6.4.1 EVA Systems Overview

Overview

The primary function provided by the EVA system is the ability for the crew to accomplish mission tasks in extreme space environments. Therefore, the EVA system will be required any time a human must leave the protective environment of the spacecraft, whether the task is to conduct planned scientific expeditions, assemble an in-space structure, perform nominal maintenance, or to intervene and solve problems outside of the vehicle that cannot be solved robotically or remotely. An EVA system consists of the hardware and software necessary to allow a crewperson to perform tasks outside of the primary vehicle. The central element of an EVA system, the space suit, is a one-person spacecraft that is “launched from” and “based out of” a primary vehicle. However, it is important to recognize that the complete EVA System, in addition to the suit, alsos include ancillary EVA tools and equipment, EVA translation and mobility aids, rover vehicles, robotic assistants, vehicle sub-systems interfaces, and an airlock. It truly requires System-of-Systems integration, with contributions and dependencies across many areas such as life support, power, communications, avionics, robotics, materials, pressure systems and thermal systems. Because of the complexity of the EVA system and the numerous interfaces with other systems, it is crucial to have a consistent, well-versed set of technical and programmatic experts leading the systems engineering effort. The EVA system should follow a spiral development in parallel with the CEV vehicles and should be based on a modular, flexible architecture that can support multi-destination operation with minimal system redesign and/or reconfiguration.

Background

Extravehicular Activity (EVA) is an integral part of the new exploration vision. Without EVA, we cannot accomplish the ultimate goal – human planetary exploration. But EVA is required for much more than just planetary exploration. It is required in all facets of the exploration vision. EVA will be required any time the human needs to leave the protective pressurized environment of a spacecraft, whether it be to conduct planned scientific expeditions, assemble an in-space structure, perform nominal maintenance, or to intervene and solve problems outside of the vehicle that can not be solved robotically or remotely. Mission success requires that humans be able operate outside the pressurized spacecraft environment through all phases of the journey.

The highly technical EVA suit is at the core of the EVA System. However, the EVA System spans much further than just the suit. It also consists of ancillary EVA tools and equipment, EVA translation and mobility aids, robotic assistants, and an airlock, which includes a dust mitigation system. Manned rovers (both un-pressurized and pressurized vehicles) should also be considered as part of an integrated EVA System. The two primary components of the space suit are a pressure garment and a portable life support system (PLSS). Ancillary EVA tools and equipment includes items that attach to the space suit, such as helmet lights and cameras, sensors, and tethers. EVA tools, such as power and hand tools, provide the capability for a space suited human to conduct assembly and repair operations. In a microgravity environment, EVA translation and mobility aids allow an EVA crewmember to translate, react forces and loads, and restrain themselves in order to perform useful work. EVA translation aids in microgravity also include self-rescue, zero-g translation units in case the crewmember becomes inadvertently detached from the spacecraft. An airlock is the system that permits an EVA crewmember to go from a pressurized space craft environment to an external space environment, be it a zero gravity hard vacuum or a low pressure partial gravity environment.

<u>Approach</u>

The EVA system should follow a spiral development in parallel with the CEV vehicles. It should be based on a modular, flexible architecture, which can support multi-destination operation with minimal system redesign and/or reconfiguration. A certain set of "core" technologies, such as light weight structures, power system, communication and data systems and CO_2 removal systems, must be developed regardless of the destination. These "core" technologies should be developed as part of the initial Spiral I design to support the 2014 human CEV mission. Lunar and Mars unique technologies can then be developed as part of future spirals.

The EVA Systems truly requires System-of-Systems integration. Development of the EVA suit subsystems includes interactions with, dependencies on, and contributions from many areas such as life support, power, communications, avionics, robotics, materials, pressure systems and thermal systems. Integration is required not only between the life support system and the pressure garment, but also between the suit, itself, and the vehicle, airlock, robotic assistants, and manned rovers.

20.6.4.2 EVAS Technology Options

<u>Summary</u>

A lightweight, highly reliable and integrated advanced Extravehicular Activity (EVA) System is a cross-cutting infrastructure which is fundamentally required to enable NASA's new vision for exploration. In order to enable a wide range of destinations and applications, an advanced EVA system is necessary. Current EVA technology will not support the new vision. Most advanced EVA technologies are at low Technology Readiness Levels (TRLs). The technology gaps in the main EVA elements that should be developed are the PLSS, the space suit pressure garment, instrumentation, information, power systems, robotic ergonomics, design, measurement, and analysis tools, vehicle requirements, and ground-based test facilities.

Issue

Most EVA advanced technologies are at low Technology Readiness Levels (TRLs). These technologies must be matured in order to support the exploration vision.

Background

A lightweight, highly reliable and integrated advanced Extravehicular Activity (EVA) System is a cross-cutting infrastructure which is fundamentally required to enable NASA's new vision for exploration. In order to enable a wide range of destinations and applications, an advanced EVA system is necessary. Current EVA technology will not support the new vision. Although the current state-of-the-art EVA system has undergone some upgrades, the basic technology is 1960's technology for the life support as well as the suit garment technology. It is adequate for LEO missions including ISS, and would even perform adequately for very short-term lunar missions, such as was conducted during Apollo. In the past, lunar missions only lasted three days on the surface and the life support equipment was not reusable: it was thrown overboard on the lunar surface prior to departure from the moon. However, for the Exploration Initiative, a more permanent and longer mission human presence is required (upwards of fourteen day lunar missions and even longer Mars missions) and current EVA technology is not adequate. In addition, there are EVA enhancements in various areas that are needed to accomplish the new mission's goals. These include a highly maneuverable space suit garment and a lighter weight EVA suit for increased crewmember productivity. Other requirements and improvements to the EVA system are detailed in the following sections.

To date, advanced EVA technology efforts have been limited to NASA NRA and SBIR grants. Most technologies are at low TRLs. Advanced EVA systems development is limited to component level research and testing. No integrated PLSS development and only limited pressure garment research efforts are in work. No significant glove work is being performed.

The technology gaps in the main EVA elements that should be developed are the PLSS, the space suit pressure garment, instrumentation, information, power systems, robotic ergonomics, design, measurement, and analysis tools, vehicle standards, and test facilities. The main EVA technology gaps are based on the need to develop EVA core technologies that will allow the development of a lightweight, modular, highly mobile, reliable and low consumables EVA system that can support multiple destinations and applications. These EVA core technologies need to be raised from very low Technology Readiness Levels (TRLs) to higher TRLs in order to get them into component and system level testing.

Approach/Recommendation

The following are critical EVA systems technology areas that require investment. It is recommended that these items be added to the list of technologies that need to be matured. Listed along with the technologies and systems requiring development are some of the limitations of the current systems. Table 20.6.4.2-1 shows only presents summaries of the technologies. More detailed discussions of these technologies and limitations are presented in subsequent sections.

Technology/System	**Current Limitations**	**Technologies to Investigate, Trade, or Develop**
System Architecture	-PLSS certified only 25 EVA's -PLSS wt is 160 lbs, need at least 50% reduction -Pressure garment designed for micro-g, need high mobility planetary suit.	Flexible, modular, lightweight, maintainable PLSS, a lightweight pressure garment, an integrated suit /PLSS interface, and rapid recharge and checkout
CO2 Removal	-Rechargeable CO2 system is heavy, has high power usage.	Cyclic CO_2 removal technology
Thermal Control	-Cooling method vents water to vacuum, need primary non-venting design -PLSS components heavy -Only manual thermal regulation control exists, need to free up astronaut with auto control option. -Current water evaporators will not work on Mars. -Cooling garment is heavy, over-designed.	Freeze-tolerant or freezable radiators, micro refrigeration and heating systems, auto cooling control, water membrane evaporators, phase change materials, and conduction cooling garments
Electronics and Data		Heads-up displays, integrated high capacity communications, smart systems monitoring, control, caution, and warning, and high reliability fans, pumps, actuators, and sensors
Power Systems		Including batteries and fuel cells
Space suit pressure garment and supporting systems (airlocks, dust control, etc.)		Including lightweight structures, mobility systems, gloves, visors, and zero pre-breathe capability, dust

		equipment, low volume airlocks.
Integrated EVA/robotic interfaces, rovers		Including robotic assistants, pressurized and un-pressurized rovers

Figure 20.6.4.2-1: Current Technologies and Technology Options

20.6.4.3 Impacts of Crew Anthropometric Sizing and Selection Criteria

Summary

For selection of future Lunar and Mars EVA exploration crew personnel, serious consideration should be given to the interrelationship between crew anthropometric size range and impacts to program cost and hardware logistics, particularly in regard to the EVA space suits. It is recommended that a trade study be performed to fully understand this interrelationship so that appropriate anthropometric sizing requirements can be levied on the EVA suit design and size range criteria can be determined for the crew selection process.

Issue

Anthropometric size range requirements can drastically drive the suit design options, hardware logistics and program cost.

Background

For selection of future Lunar and Mars EVA exploration crew personnel, serious consideration should be given to the interrelationship between crew anthropometric size range and impacts to program cost and hardware logistics, particularly in regard to the EVA space suits.

Anthropometric size range requirements drive hardware logistics. The wider the range of crewmember sizes, the more hardware needs to be flown to maximize redundancy capability. This will be particularly important on long duration missions, where bringing the suits home for servicing and repair is not an option. The ability to reuse space suit hardware components and sizing elements will be central to maintaining the operational capability, reasonable program costs and hardware logistics support over the life of future planetary surface programs.

Approach/Recommendation

It is recommended that a trade study be performed to fully understand the interrelationship between crew anthropometric size range and impacts to suit design, hardware logistics and program cost. Should the trade study conclude that it is appropriate to limit the anthropometric size range of crewmembers, the size range limitations must be clearly stated in the future Exploration Program crew selection criteria.

20.6.4.4 Interaction Between EVA System Design and Vehicle Design

Summary

The EVA System is integrally related to the space vehicle and habitat. As such, it is imperative that the EVA system requirements be developed in conjunction with the vehicle requirements. Of particular interest is the determination of operating pressure, which affects the suit and vehicle life support system design and drives the pre-breathe protocol, which has ramifications for both the operational scenarios (increased crew time and suit mobility limitations) and the physiological health and safety of the crew.

Issue

To maximize the efficiencies in both systems, the EVA suit requirements should be developed concurrently with the vehicle requirements.

Background

A lesson was learned from the Space Shuttle program. Initially, the EVA system was not integrated into the vehicle design. Concurrent development and integration of the vehicle and the EVA System did not occur for two reasons. First, EVA was not recognized as a required operational capability. Second, during the development of the Shuttle, resolution of failure mode and effects analysis (FMEA) issues lagged system design maturity. This led to safety critical design changes having to be incorporated during production instead of during the paper phase of design, which is significantly more expensive. The EVA System was a case in point for it addressed several criticality 1 FMEA issues for the Orbiter and the late integration of the EVA system was more expensive and more difficult than necessary. Therefore, it is strongly recommended that vehicle and EVA system requirements and designs be developed concurrently.

Additional rationale for concurrent development is that several of the program level design drivers for both systems are inter-related. Inter-related, high-level design considerations for the vehicle and the EVA system include the following:

i) vehicle and suit operating pressures,

ii) overall vehicle architecture considerations, such as mass and volume, and

iii) interfaces, sub-system and other physical interfaces.

The operating pressure trade has both design and operational implications. Operating pressure drives structural loads requirements in design and pre-breathe protocols for operations. The overall vehicle architecture benefits from design and cost efficiencies realized through a coordinated design effort.

Approach/Recommendation

It is strongly recommended that vehicle and EVA system requirements and designs be developed concurrently.

It is recommended that a trade study be performed to investigate the impacts of various suit and vehicle operating pressures. These parameters affect the suit and the vehicle life support system design. It also drives the prebreathe protocol, which has ramifications for both the operational scenarios and the physiological health and safety of the crew.

20.6.4.5 EVAS Existing Trade Studies

The results of these EVAS trade studies should be used as a starting point for determining future EVAS architectures. In addition, there are several trade studies that should be performed in order to define requirements.

EVA/Launch and Entry Suits: Separate versus Common

In the determination of combining both launch/re-entry operations and EVA system requirements into a single-suit configuration concept, one must consider and identify the specific mission requirements defining pressure suit support functions for each of these critical mission phases. This involves a complete understanding of what role the pressure suit plays and supports for intra-vehicular (IV) versus extra-vehicular (EV) functions.

Although it may seem that the concept of a “single-suit” is appealing, the identification of critical and unique sensitivity design drivers will show what significant factors influence configuration selection and correspondingly, whether a “single-suit” approach is functionally feasible. Selection of the appropriate suit configuration approach should be based on assessing the overall “value” to the various Exploration Program mission phases and program goals. Value in this case should be assessed in terms of “technology risk” and “operational performance capabilities” (i.e., mission success).

From a historical perspective, experiences gained during the Gemini and Apollo programs in which a single-suit configuration served in a dual-suit role capacity resulted in many design and operational compromises. It was found that the integration and mix of combined IV and EV critical functional requirements imposed design feature restrictions and subsequently significantly affected operational performance capabilities of the end-item suit configuration. In many cases, the functional requirements were diametrically opposite between IV needs and EV needs. An obvious operational advantage for one mission phase (long-term wear comfort; no sophisticated mobility features – launch/re-entry phase) poses a severe disadvantage for another phase (maximum use of sophisticated mobility features – EVA surface operations). This is only one example of imposing design and functional requirement differences. Also, technology selection feature opportunities to incorporate and maximize unique suit operational performance capabilities for either or both IV/EV are seriously restricted, or in many cases, totally eliminated.

As a result of this important Gemini/Apollo experienced gained and lessons learned baseline, the Shuttle Program adopted and successfully implemented an operational, dual-suit configuration approach which has provided for separate IV and EV operational capabilities for the past 30 years. Numerous single- versus dual-suit trade studies have been conducted in the past (i.e., Space Exploration Initiative program, First Lunar Outpost program, Shuttle Launch Initiative program) and the findings have resulted in continuing with a dual-suit approach. In addition, Russian experience has stated: "Only the soft type suit is acceptable as an IV suit, as its design is closer to usual clothes and the most comfortable to wear in the cabin. That is why, it was proposed to use two different types of suits for the Soyuz-Salyut and the Soyuz-TM-MIR space programs which provided for regular EVA's" ((Reference: "Space Suits: Concepts, Analysis, Perspectives -1990"; Professor Guy Severin, Zvezda, Chief Designer, USSR). For the future Exploration Program, it is recommended that the dual-suit approach be followed in the implementation of separate suit configurations for respective IV and EV applications.

Reusable vs. Limited-use Life Support Systems Components

Summary

The concept of limited use of Space Suit Life Support System components has its history in the Apollo Program where the Portable Life Support System had to be left on the lunar surface along with accumulated trash in order to provide additional launch capability for lunar sample return. As one examines the possible exploration mission architectures there are at least three levels of life support hardware ranging from the full up portable life support system to a minimum system that will just provide what is needed to cover the required EVA needs for each phase of the mission. Since missions have the possibility of having on the order of seven phases that need to be examined there are on the order of 20 cases for each architecture.

Embedded in the cases are various levels of limited use of the life support system as well as some cases of limited use of the entire space suit assembly. A bounding case is the complete reuse of all space suit hardware.

A trade study of a limited number of architectures needs to be accomplished to understand the Suit / Vehicle relationships. After this trade study is accomplished, the information needed to make trade studies concerning generic versus tailored space suit systems and the potential for use of the Launch and Entry Suit is available.

All of these trade studies are strongly influenced by cost making them inherently governmental.

Problem Statement

What is the right mix of reusable versus limited or one-time use items for exploration architectures for space suit components? Examine the trade space for space suit components and accompanying life support system technologies and space suit design. Is exploration architecture with limited mission requirements better served with generic suit and component sizing or sizes tailored to the crew member? Does this answer change with crew size? For life support components, what is the trade off between reusable systems in the suits and their support equipment in the exploration vehicles/habitation? Could a lunch and entry suit or its components be combined with an EVA suit or components? How do these issues fold into safety, reliability, maintainability and logistics concerns?

Space Suit System Characterization

Space suit systems are historically divided into two major subsystems, the space suit garments and the portable life support hardware. These two subsystems are driven by significantly different requirements, have different kinds of hardware and, traditionally, are supplied by significantly different kinds of industry.

Space suit garments are driven by human mobility, fit, protection, and interface requirements. Mobility is the first consideration and mobility is driven by the task the crew person is expected to accomplish as well as the location from which it is to be accomplished (i.e. zero-g vs. planetary). Human body anthropometric variation drives fit. Variations of a single person with time (the body grows as much as two inches in height when moved from a 1-g environment to a 0-g environment) and variations from person to person have to be addressed. Protection from thermal extremes, micrometeoroid impact and radiation all come into play in space suit garment design. Interfaces the suit garment must meet are with every vehicle from a standpoint of performing tasks for the vehicle (reach envelops, restraint locations, etc.). In use interfaces include not only physical interfaces (handholds, stairs, etc.) but visual (sight lines, lighting, etc.) and communications interfaces that are direct divers on the space suit garment.

The portable life support system is driven by human thermal and breathing physiology, equipment protection, and vehicle interfaces. Human thermal physiology sizes the thermal control system and human breathing physiology sizes the oxygen supply and the oxygen revitalization systems. Equipment protection for the life support equipment and communications equipment must be provided from the impact, thermal, micrometeoroid and radiation environments.

By far the largest driver on the portable life support system is the vehicle interfaces, specifically the interfaces to the supplies needed to recharge the portable life support system between uses. Those interfaces can exclude whole classes of portable life support technologies if they are not properly selected and can dictate far from optimum use of non-excluded technologies.

Mission Architecture and Space Suit Systems

The design of a space suit must include an examination of the trade space for space suit components and accompanying life support system technologies. The biggest impact that the mission architecture has on the Space Suit System is related to the vehicle from which the EVA occurs (CEV or Lander), the mission phase/environment (e.g., low earth orbit, in deep space, or on planetary surface), and the type of EVA that must be supported (e.g., contingency or significant planned). Table 20.6.4.5-1 describes the mission architecture impacts to the EVAS.

Space Suit Garment	Space Suit Life Support	EVA Hardware Flow	Vehicle Life Support	Vehicle Impacts
Full Up Mobile Suit	Full Up PLSS	Full up Suit moves all the way through the mission with crew	None Required for Contingency only cases. Full up recharge capability for planned EVA	Storage required for Full up EVA suits and Launch Escape Suits
Primary Working Envelope Mobility	Emergency Blow Down System	Full up system staged at libation point. Sized suit elements move with crew all the way through the mission	Umbilical Life Support with Comm, power, cooling, and ventilation.	Storage required for Umbilical. Life Support system equipment for each crewperson
	Mixed System with Emergency Blow Down System plus some part of PLSS	Full up system goes down to planet but only suit comes up	Reduced Umbilical Life Support.	Storage required for Umbilical. Life Support system equipment for each crewperson

Table 20.6.4.5-1: Mission Architecture Impacts to EVAS

The vehicle life support must be addressed for each vehicle in the architecture. As the trade moves toward minimum EVA Systems, more demands are made on the vehicle to supply life support capability to support those minimum EVA Systems. The full EVA Space Suits can cover any of the contingency cases without placing any burden on the vehicle life support system. But, the weight and volume of the full up EVA suits must be accommodated if they are used to address contingency cases. Options driven by the flow of the EVA hardware between these extremes exist and will also be included in the trade space. The flow of EVA Hardware options are where the issue of re-use versus one time use of space suits or parts of space suits becomes important. For example, in an architecture that contains a “line shack” vehicle, it may be possible to stage EVA Suits out of that vehicle and not have to bring them up for each crew. Another option is to leave the portable life support systems on the planetary surface as was done in Apollo. Other options exist and the logical ones would have to be examined for the different cases.

The different options of EVA system types could be examined to arrive at an architecture level optimum. The main output variables will, of course, be launch mass, down weight and volume, and mission cost since trades between vehicle impacts with its attendant launch cost and the cost of abandoned EVA hardware must be considered.

Suit / Vehicle Trade Recommendation

When the mission architecture options settle down to two or three is the time to perform a trade study of the impacts of different types of EVA support for those mission architectures. Because this trade study has significant monetary cost implications that should be optimized to the Government's benefit, it is an inherently government trade study and one the Government has the capability to accomplish.

Custom Suits versus Generic Suit Sizing

The trade between custom suits and modular suit systems is driven by two primary factors: 1) the number of crew members per mission and over the course of the program and 2) the frequency of human missions over the course of the program.

Additionally, flight-program experience has shown that significant cost-drivers, as well as, in some cases, significant reduction or restriction of mobility performance capabilities, are associated with implementing design modifications to accommodate unique individual crewmember anthropometries. Also, Shuttle EMU experience has shown that a modular approach to buildup of individual suit assemblies to accommodate a specific range of both male and female crewmembers works extremely well to provide functional mobility performance capabilities while maintaining a reasonable logistics inventory of component elements that can be re-used over the life of the program. This approach allows for near "custom" sizing of a suit assembly for a particular crewmember based on the selection and integration of the appropriate anthropometric suit modules from a general pool of hardware elements without requiring a unique, dedicated "custom" suit assembly that would fit only a single individual. This approach to suit sizing is especially critical based on the turnover and attrition rate of crewmembers over the life of the various program phases. Furthermore, the resulting logistics hardware from a "generic" component sizing approach versus a "custom" component system will enable a less costly and more effective use of spares with core maintenance and "ship-set" replacement elements over the life of the program. However, because of the critical nature of EVA gloves for future planetary surface operations and EVA boot accommodations for walking, custom sizing would be recommended for these specific suit components for individual EVA crewmembers.

20.6.4.6 Recommended Trade Studies

- Performance Requirements Definition
- Life Support System Performance Requirements Definition
- Vehicle versus Suit Operating Pressures
- Suit Commonality Study
- High Power Schematic Study
- Breathing Gas System Study
- Oxygen Pressure/Recharge System Study

- Nitrous Oxide Schematic Study
- PLSS Packaging Architecture
- Unpressurized Rover Architecture and Interfaces
- Pressurized Rover Architecture and Interfaces
- Crew Escape Systems
- Airlock Mock Up Evaluations, Transfer Vehicle
- Airlock Mock Up Evaluations, Lander Vehicle
- Airlock Mock Up Evaluations, Transfer to Pressurized Rover from the Lander Vehicle
- Suit Insulation Commonality and Approaches for Lunar and Mars Thermal Environments
- Active Thermal System Commonality and Approaches for Lunar and Mars Thermal Environments

20.6.4.7 Planetary Surface Airlock and Dust Lock Issue

Summary

Dust contamination will be a significant issue on the surface of both the Moon and Mars. Dust mitigation and control must be considered in the design of vehicle, habitat and EVA suit systems so that dust particles are not brought into the breathing volume. Several options exist for controlling dust on the planetary surface: 1) select space suit materials that repel dust particulates, 2) design the habitats and vehicles such that the dust can be removed prior to entering the structure and, 3) design a system such that the EVA suits and dust are left outside of the inhabited volume. It is recommended that 1) dust resistant space suit materials be included in the list of technologies that need to be matured, and 2) a trade study be performed on various concepts for airlock architectures.

Issue

Dust contamination will be a significant issue on the surface of both the Moon and Mars. Dust mitigation and control must be considered in the design of the vehicle, habitat and suit systems.

Background

An important lesson learned from the Apollo Program was the highly intrusive nature of the Lunar dust which will also be a similar experience encountered by crewmembers during future Mars surface extravehicular activities (EVA's). The dust material can pose hazards to mechanisms and may also pose a long-term breathing hazard for the crew. The properties of lunar dust are fairly well known, but the abrasiveness and potential breathing hazards of the Mars dust are yet to be determined. Representative simulants of both Lunar and Mars surface materials are currently available to initiate preliminary investigations into these areas of concern.

In both of these future locales, dust will inevitably be deposited on EVA systems and surface equipment and will be brought into the habitable spaces unless contained or controlled. A series or combination of "dust mitigation" techniques and methodologies must be applied ranging from requiring a selection of appropriate space suit materials (or removable outer garment) to defining a suitable airlock architecture (segmented airlock with "dust lock" feature, or a minimum volume airlock – "suit port") to aid in managing and controlling both Lunar and Mars dust intrusion problems.

One of the primary methods to be considered in a multilayered strategy for dust management is to cover those portions of the planetary surface space suit most likely to be in contact with dust with an easily removable outer layer of dust-resistant material. For example, the Apollo Program space suit utilized a pair of over-boots that were worn over the basic space suit pressure boots for thermal, dust, and abrasion protection. For future consideration, the planetary space suit outer layer garment and boots could be doffed after each EVA surface operation and retained in the un-pressurized "dust lock" portion of the airlock at all times. This would preclude transport and transfer of dust into the surface habitat area. This relatively simple technique or method should keep the majority of the dust outside. However, a number of attributes and issues of this method still need to be investigated: (1) the identification and selection of appropriate dust-resistant materials, (2) the ease of donning and doffing removable cover garments and boots while in a pressurized space suit, (3) what configuration should the outer worn garments be, and, (4) whether to have them disposable or reusable.

The next element or phase of operation in this representative dust management approach is a segmented airlock – one that has an un-pressurized ante-room or "dust lock" located adjacent to the main active pressurized airlock which in turn is mounted to the surface habitat. In this implementation, the first step in reentering the surface habitat would be to enter the outer unpressurized "dust lock" area where all loose dust and other particle material is physically brushed off by the returning EVA crewmembers. This "dust lock" area could be designed to be raised and have an open grated floor so that the loose material and particles removed from the suits would fall through the flooring back to the surface. After physically brushing the loose material, the space suit outer garment coverings and EVA boots could be removed by the crewmembers and stowed in the "dust lock" area.

The final element or phase of operation would then be for the EVA crewmembers to ingress the basic airlock for re-pressurization. The airlock re-pressurization system would be designed such that the air reentering the airlock would be directed by nozzles in a form of an "air shower", directed over the space suits and subsequently used to blow-off any remaining dust. A high filtration system in the airlock floor would collect the removed particles. Before complete re-pressurization of the airlock has taken place, another dust-removal feature that would be available to the EVA crewmembers would be a "vacuum" system umbilical that would allow the crewmembers to further vacuum off and remove as much of the remaining dust as possible. After the crewmembers connected their suits to the support stand, they could then doff and exit the suits and ingress the habitat. Although the approach is not necessarily finalized, comparable activities to those just described for a conventional dust lock/ airlock system will be necessary to maintain the appropriate level of separation between the outside environment and the habitat interior towards the solution of the dust management problem.

Other dust lock and airlock concepts also need to be considered for future planetary surface EVA operations. For mission durations and number of EVA sorties identified in the future vision of EVA, consumables or expendables (in addition with concerns for dust mitigation) can become a significant contribution to overall mass. The current Shuttle orbiter airlock (148 cu ft volume) is not pumped down for conducting EVA's. Instead, the atmosphere is bled off to vacuum and sacrificed. Although this procedure is satisfactory for Shuttle missions with a maximum of three EVA's per 10-day flight, this type of operation would be extremely prohibitive for extended planetary surface EVA operations. In comparison, the Crew Lock portion of the International Space Station (ISS) Joint Airlock is partially pumped down, but the remaining air (about 5 psi) is vented overboard through a valve on the outer hatch. The Crew Lock was built as small as possible to minimize the amount of air loss from the ISS to space during this venting process; the Crew Lock actually loses less air to space than the Shuttle airlock.

Even though airlocks can be pumped down to a small fraction of their original atmosphere, losses still occur and make-up gases are an item that must be tracked as a necessary expendable resource. Minimum volume airlock concepts have been proposed that can significantly reduce the lost breathing gas that occurs for each airlock cycle, as well as providing an environmental barrier to aid in the control and management of planetary dust problems. One concept, known as the "suit port", actually connects an individual EVA suit via a rear-entry hatch closure system to an outer wall of either a surface habitat or a pressurized surface rover vehicle. The crewmember enters the suit directly from the cabin through an airtight hatch in the back of the suit. This is actually a double hatch system – one part seals the EVA suit for detached operation while the other part seals the habitat or rover cabin volume from the outside surface environment. The nature of the suit port system permits the virtual elimination of pump-down in routine usage of the system for conducting EVA's. Only a small interstitial volume of space between the space suit portable life support system and hatch interface need to be pumped down. This concept represents a volume reduction of about 25.5 cubic meters (900 cubic feet) from a conceptual conventional dust lock/airlock design (as previously described) to about 0.03 cubic meters (one cubic foot) or less for the suit port. In fact, the remaining interstitial air volume as previously mentioned is so small that it could even be sacrificed rather than requiring power and weighty mechanisms for pump-down. Pump-down time, power and pump cooling are therefore eliminated in this mode of operation.

The suit port concept also offers the additional benefit of dust control. By sealing the suit to the outside of the shirtsleeve cabin environment, it is possible to isolate the dust (or other contaminants) from the crew. The crewmembers can don and doff their suits through the suit port without needing to decontaminate after each EVA sortie. The concept however is not without its own set of disadvantages. Since the suit port would isolate the majority of suit elements from direct access for routine checkout and servicing, some technique must be devised to allow for these operations. Also, if there are no provisions for protection for the exposed suits, the constant exposure to space and planetary surface environments will degrade the suits faster. The suit structure may have to be designed for a variable pressure regime since the habitat pressure may range from 9-10 psia whereas the EVA suit is currently planned for a 4.3 psia operational pressure. Since each suit would require an individual suit port and the complicated mechanisms for backing up, docking and latching the PLSS to the outer hatch of the suit port haven't been designed or developed in detail, the issue of whether or not any weight savings or operational advantage is gained by the suit port concept is still in question.

Approach/Recommendation

1) Include dust resistant space suit materials in the list of technologies that need to be matured.

2) Perform a trade study on various airlock architectures, including the segmented airlock with "dust lock" option and the minimum volume airlock, or "suit port" option.

20.7 Extravehicular Activity System (EVAS) Technology Report

Subdisciplines: ***Command & Data Handling***

Data Management System

Instrumentation

Display & Controls

Submitted by: ***NASA/JSC/EV - Avionic Systems Division***

Coy Kouba, David Jih, Helen Neighbors

20.7.1 Subsystem Description

20.7.1.1 Primary Functions

The Avionics Subsystem consists of the electronic hardware necessary to perform vehicle command, monitoring and data processing required to operate and control the spacecraft. It includes the primary flight computers, high-speed data bus, data acquisition units, instrumentation sensors, data recorders, and crew displays and controls. It also generally involves any other electronic device that performs digital or analog signal processing to support vehicle data and commanding functions, such as embedded processors or controllers.

The avionics subsystem may also include backup hardware/software that would be used if the primary system were to fail. This backup hardware may the same hardware as the primary system, or it may be completely dissimilar.

20.7.1.2 Key Design Parameters

For critical vehicle functions, the avionics architecture should be two fault-tolerant by using redundant components that work together. Being two fault-tolerant would allow for two permanent system faults to occur and still have enough resources to safely bring the crew home. The avionics architecture should also autonomously detect, isolate and recover from time-critical failures, without unnecessarily burdening the crew.

The method used to compare the health of the avionics system will drive the complexity of the system. This can be done with varying degrees of complexity, including bit-for-bit comparison with synchronized clocks (tightly coupled), or with a message-passing scheme at regular time intervals (loosely coupled). However implemented, the system must be able to determine the validity and correctness of vehicle inputs and outputs.

One of the biggest design drivers for avionic hardware is its susceptibility to the space radiation environment. Early during the design phase, candidate electronic parts must be thoroughly analyzed and tested to insure they will not fail on the spacecraft. A detailed parts management program must be in place to verify all components have passed the radiation assurance criteria. The environment in which the parts will operate must be fully understood, as well as the device's characteristics such as feature size, process technology, function, speed, duty cycle, and operating voltages. For critical spacecraft functions, radiation-hardened or radiation-tolerant parts will normally be required. These are much more expensive and the selection can be limited to a small subset of commercially available parts, so this can easily be a key design driver.

Another design factor to consider is how configurable is the avionics system. If components were to fail, it is desirable to be able to restring or reconfigure so that critical capabilities can be recovered. For time-critical functions, reconfiguring of failed components must take place autonomously or with little burden to the crew's time.

The more modular the hardware design, the easier it will be for installation, ground servicing, and in-flight maintenance. The avionics hardware should also be optimized for low power, volume and weight, while still providing sufficient performance margin for unexpected demands.

The avionics used do not necessarily need to be the fastest, most state-of-the-art electronics, but they need to be capable of executing the desired functions with plenty of margin or reserve (usually 25-50%). They must also be dependable, reliable, and rugged enough for the intended environment. Ease of maintenance, supportability, replacement, and a planned path for hardware upgrades should be implemented.

20.7.1.3 Typical Redundancy & Reliability Design Approach

The typical redundancy approach is to be two-fault tolerant for mission critical functions (i.e., spacecraft navigation or control), and single-fault tolerant for non-critical functions (i.e., data recorders). Spare hardware for critical items may need to be carried onboard the spacecraft, especially for long-duration missions. If a backup avionics system is used, it will normally be single-string with little extra redundancy in itself.

The key to the reliability design approach is to use a good, simulated, fault-injection testbed to qualify the system. Failure modes should be simulated over a wide range of test conditions (assuming both nominal and degraded hardware health) to verify that the system recovers as expected. A strong environmental test program must also be used to verify the hardware will operate in its intended environment, including radiation, thermal, outgassing, vibration, power quality, and electromagnetic interference testing.

20.7.1.4 Typical Vehicle Resource Requirements

There are three primary vehicle resources that avionics will require: input power, heat removal, and volume. Avionic hardware is typically supplied with 28 volts DC or less. Removing the heat that the hardware generates can be performed either convectively (using forced air), or conductively (either with an active or passive cold plate). If the avionics are placed inside the pressurized cabin, the environment will be more benign and better for the hardware, but the design trade-off with crew accommodations, noise, and ECLSS requirements must be made. Conversely, if avionics are placed in an external, unpressurized location, the environment is harsher on the hardware.

20.7.1.5 Potential For Resource Conservation During Coast Or Parking Orbit Mission Phases

It is expected that a fair amount of avionic hardware can be put in standby or even powered off during these times, thereby conserving power and reducing cooling requirements. The redundant set of flight computers could be reduced to a single fault-tolerant set, while still maintaining some level of fault-tolerance. At a minimum during these low activity periods, the avionics system would still need to provide vehicle health monitoring, external command reception and processing, and limited GN&C functions. Whenever the crew is present, a higher level of fault-tolerance would obviously be required.

20.7.1.6 Potential Vehicle Design Interactions Or Synergy

The avionics subsystem interfaces with almost every other spacecraft subsystem, especially the following: communications and tracking, vehicle health monitoring, power distribution, propulsion, guidance, navigation and control, environmental life support, crew escape, and crew interface.

20.7.2 Technology Options

20.7.2.1 Current and Advanced Technologies

The following table lists technologies that may be appropriate for the next exploration spacecraft.

Avionics Technology	*Description*	*Current TRL level*	*Notes*
Advanced modular computer units	Small, distributed processors performing specific tasks (versus larger centralized computers). Redundant and highly reliable. This would help eliminate having to design with hardware that carries extra resources not needed that are commonly found on big single-board-computers (i.e., integrated I/O ports, bus I/Fs that will never be used).	3-7	- Distributed, networked devices and computers may be more efficient than larger centralized computers. - Could be implemented in programmable logic (i.e., FPGAs) - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.

High-speed, fault-tolerant data bus	100 Mbps to over 1 Gbps bandwidth required, using either copper or optical core. Built-in fault-tolerance and low-power consumption are desired.	5-7	- IEEE 1394b and Fibre-Channel are good candidates. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
Wireless networked systems	Needed for crew cabin communication devices, laptops, PDAs, etc.	5-7	- Bluetooth variant could be used for non-secure cabin communication.
Advanced wireless instrumentation sensors	A network of MEMS/ nanotechnology based sensors used to monitor vehicle health (i.e., temp, strain, pressure). The RF module could be integrated onto the sensor, which could wirelessly communicate with the data acquisition device.	1-6	- Would eliminate a lot of vehicle wiring - Very small packaging would allow sensors to be easily located on vehicle. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
Radiation-hardened electronic technologies	New materials and fabrication processes may yield better radiation-hard/tolerant electronics. New materials may also provide better shielding solutions for electronics.	3-9	- Parts would be latchup immune, have very low SEE rates, & be total dose tolerant - Expensive option, but would greatly allow for stronger design performance, capability & flexibility - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.

Autonomous Reconfigurable computing	Programmable hardware that detects and isolates failures, and reconfigures itself to continue operating successfully (i.e., "self-healing" machines)	3-5	- Used to provide system fault-tolerance. - Designers must ensure inadvertent reconfiguration changes are not possible.
Embedded Avionics packaging	An embedded three-dimensional packaging technology with all EEE parts inside printed wiring boards which are bonded to a thermally conductive core to eliminate need for active cooling. The approach will also eliminate most solder connections, eliminate one third of the total electrical connections and metal housings to yield weight and volume savings	1-2	- New concept with great potential for spacecraft electronics. - Could lead to substantially smaller, lower power avionics that generate less heat. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
New display and control technologies	Useful for efficient crew-machine interfaces.	3-9	- Includes touch screens, heads-up displays, LCD flat panels - New technologies under commercial development include Field-Emission Displays and Organic Light Emitting Diodes displays
Speech recognition technology	Potential crew-machine interface with improved human factors.	2-4	- Uncertain of current technology and/or use in spacecraft applications.
Artificial Intelligence	Potential for great improvements in spacecraft operations, ranging from mundane crew tasks to, eventually, flight control.	1-4	- Uncertain of current technology and/or use in spacecraft applications. - Uncertain of development challenges.

Advanced error detection & correction schemes	Would allow for greater fault detection, isolation & recovery, thereby increasing the system's reliability.	2-4	- Uncertain of current technology or development challenges. - MOST PROMISING FOR FUTURE EXPLORATION MISSIONS.
Advanced data compression techniques	Would allow more vehicle data to be sent to the ground or to onboard data recorders.	2-4	- Uncertain of current technology or development challenges.
Advanced encryption/decryption codes	Would allow for more secure command uplink and telemetry downlink.	2-4	- Uncertain of current technology or development challenges.
Wire Integrity	Technology that will determine the condition of installed wiring and cable harnesses inside a spacecraft.	2-3	- Could be used for production verification on the ground, and possibly on-board in space for in-flight troubleshooting.

Table 20.7.2.1-1: Avionics Technologies

20.7.2.2 Current Research & Development Activities

NASA should closely partner with industry to develop these technologies for spaceflight applications. With the space market being so small compared to commercial industry, NASA will have a difficult time getting industry to implement all desired technologies. NASA should also closely form academic alliances with universities, since they have more resources to focus on technology research and development. Industry technical committees, such as the Institute for Electrical and Electronic Engineers, regularly draft standards for certain technologies (i.e., IEEE-1394b for high-speed data bus).

As for testing new technologies, the commercial market will have the largest customer base and can thus help discover production problems. Commercial aircraft are also good testbeds to develop and test new technologies. The Boeing-777 aircraft development led to substantially improved fly-by-wire and fault-tolerant flight control designs. The military is another good testing ground for new technologies, with the Joint Strike Fighter being one of the first applications to use IEEE-1394b for critical flight control.

20.7.3 Recommended LDRM-2 Subsystem Design Approach

20.7.3.1 Functional Description

Based on past experience, an appropriate avionics design for LDRM-2 would incorporate the following characteristics:

- Two-fault tolerant avionics architecture for critical systems, such that any two failures would not disrupt normal avionic functions.
- Three flight critical computers synched together, with voting on all inputs & outputs.
- Dual lock-step processors implemented inside each flight computer. This will facilitate the voting process and help determine which computer is faulty.
- Separation between flight critical and non-critical tasks. The primary flight critical computers should only process vehicle command & control functions. Smaller distributed computers could be used to process other functions, such as cabin lighting, temperature, etc.
- Wireless networked systems (for data acquisition and health monitoring)

20.7.3.2 Estimated Vehicle Resource Requirements

The following table lists the avionic components appropriate for the crew exploration spacecraft. Not included are the communication and tracking hardware, nor the guidance, navigation & control hardware (see those appropriate subsections for details).

Avionic Component	Qty	Unit Weight (lb)	Unit Avg Power (w)	Total Weight (lb)	Total Avg Power (w)
Command & Data Handling					
Flight Critical Computers	3	40	150	120	450
Data Acquisition Units	4	15	40	60	160
Data Recorders	2	10	30	20	60
Data Bus Hardware	2	30	30	60	60
Switch Panels	2	20	5	40	10
Communication					
<< see Comm & Track section >>					
Display & Controls					
Crew Displays	3	10	50	30	150
Hand Controller Sets (R&T)	2	10	10	20	20
D&C Processors	2	30	100	60	200
Crew Input Devices (e.g. key pads)	3	4	10	12	30
Caution & Warning Panel	3	10	15	30	45

Guidance and Navigation					
<< see GN&C section >>					
Integrated Vehicle Health Monitoring					
VHM components (processors)	2	30	50	60	100
Instrumentation					
Sensors	1000	-	-	150	10
Backup Flight System					
Flight Critical Computer	1	40	150	40	150
Wiring - avionics specific	lot	-	-	500	0
			TOTALS =	**1202 lbs**	**1445 watts**

Table 20.7.3.2-1: Avionics Components

The following assumptions were made:

1) Triple redundant Flight Critical Computer (FCC) system.
2) Dual lock-step processors in each FCC.
3) Single string Backup Flight System (BFS).
4) BFS hardware is same as FCC, but runs independent software
5) Each FCC has redundant cross-strapped power feeds
6) Data Acquisition Units receive distributed inputs; output data can be sent to any FCC
7) Same avionics hardware is used in both CEV and Lander
8) Hardware is radiation tolerant, with architectural fault-tolerance handling any failures

There would be similar hardware required for the lunar lander module, thus its avionics would also weigh approximately 1200 pounds and consume approximately 1400 watts. Since it would also be crewed, the lunar lander would need to have the same fault-tolerant philosophy as the CEV.

The un-crewed injection stage boosters and service modules would require a minimal subset of these avionics, primarily to provide guidance, navigation and control functions, health monitoring, and command/telemetry control with limited fault-tolerance. An estimate for these modules would weigh a few hundred pounds each and consume 300-500 watts of power each.

All together, the combined avionics on all spacecraft elements could weigh 3000 pounds and consume 3.6 kilowatts total (excluding communications and tracking, and GN&C hardware).

20.7.4 Mars Spiral Development

20.7.4.1 Promising/Enabling Technologies for Human Mars Mission

For longer crewed missions to Mars, the following technologies offer the most promising benefits for the avionics system:

- Autonomous operations for time-critical tasks and mundane operations
- Greater fault-tolerant, redundant, and reliable hardware
- Nanotechnology and MEMS based devices
- Wireless networked sensors and processing units
- Improved crew-machine interfaces

20.7.4.2 Potential for Development and Flight Testing of Mars Technologies

As new technologies are developed and used in spaceflight applications, they need to be proven in small controlled phases, such as short-duration flights in LEO, then longer missions to the moon, etc. Only after these technologies have been thoroughly proven in the space environment should they be used on a long-duration, self-sustained trip to Mars.

20.7.5 References

- Orbital Space Plane project
- X-38 project
- NGLT/SLI products
- Previous EV and EX-led exploration studies

20.8 Guidance, Navigation, and Control (GN&C) Technology Report

Submitted by: ***NASA/JSC/EG – Aeroscience & Flight Mechanics Division***

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20.8.1 GN&C Issues Summary

Much of the GN&C capabilities necessary to support the trade LDRM-2 architecture, and potentially any lunar exploration architecture under consideration, are sufficiently mature that they can be categorized at an advanced TRL (between 7 and 9). However, there are several key GN&C issues and technologies that require investment and TRL advancement to mitigate cost, schedule and mission/safety risks to enable the exploration architecture. Furthermore, developing the GN&C capabilities of a Moon mission to approach that of a Mars mission analog would likely entail expanding this set of GN&C issues and technologies. The final collection will ultimately be driven by overall exploration architecture and mission objectives, concept of operation, and top-level requirements, including human rating. Ideally, these will be iterated upon once more detail is available.

This section will summarize the top four GN&C issues identified that are expected to have the maximum impact on an exploration architecture that includes the LDRM-2 trade mission (see Table 20.8.1-1). These are further categorized as “Enabling”, meaning there are subsystem hardware or software components that require advances in technology in order to enable achieving the mission/architecture objectives, or “Enhancing”, meaning that there are subsystem hardware or software components that have the potential to significantly reduce risk and/or cost and require long lead time development. Note that these results were derived with limited, inferred top-level requirements and from available existing documentation. Specific technology recommendations should be viewed only as candidates, not final solutions. In order to properly assess the needs of the exploration architecture(s) a more rigorous analysis of the overall system needs should be combined with the requirements development and analysis phase as early in the process as possible. Results of upcoming ground/flight experiments should also be incorporated into the assessment.

Technology	Enabling	Enhancing	TRL
Automated Rendezvous and Docking	X		3 - 4
Automated and Precision Lunar (and Mars) Landing	X		3 - 4
Autonomous Flight Management	X	X	2 - 4
Deep Space Navigation		X	9*

* Reference Deep Space Navigation section below

Table 20.8.1-1: Top Level GN&C Technology Issues

Automated Rendezvous and Docking

Automated rendezvous and docking (including separation and departure) is in the critical path for both crewed and uncrewed mission elements for LDRM-2, and for virtually all exploration architectures under consideration. Successful design and implementation depends on both technology and process details. On the technology front, three key navigation sensors have been identified as providing the necessary rendezvous and proximity state information for relative position, velocity, attitude and attitude rate at appropriate ranges with the complement of equipment to meet safety, fault tolerance and where applicable, human-rating requirements. These are identified and described in Table 20.8.1-2, below. In addition, the accompanying GN&C algorithms and FDIR support, including human-in-the-loop monitoring and control are long lead technology development items to support the architecture. This is also listed in Table 20.8.1-2.

Sensor Candidate	Description	Current TRL	Notes
Radio Frequency (RF) Based Navigation (combined w/ communication)	Uses communication signal between spacecraft to provide long range relative state information Uses vehicle subsystem equipment (communication) that is required and present independent of relative navigation requirement	3 – 9**	State (position, velocity, bearing) information may be available to both halves of the interface (chaser and target) Potential for relative attitude measurement capability, so may be applicable through to docking **Proven in space; however, certain aspects have been developed/used by Russia only; h/w, s/w and detailed results are not accessible, so TRL is lower based on availability
LADAR ("Laser radar")	Laser based detection and ranging system that processes a scanned signal into three-dimensional relative state information	4	Flexible -- no need for any retro-reflectors or other devices on target vehicle Potential for use from long range (50 to 100 km) up to docking No lighting constraints Potential for dual use as landing sensor for altitude measurement and terrain mapping

NFIR (Natural Feature Image Recognition)	Processes video camera image into three-dimensional relative state information; software based solution	4	No need for any retro-reflectors or other devices on target vehicle Uses subsystem equipment (camera) that will likely be present to meet human-rating requirements May impose natural or artificial lighting requirements Useable range is a function of camera focal length (in practice max range ~1 km)
GN&C & FDIR Algorithms	Perform both nominal and contingency functions to ensure safe/successful docking for crewed and uncrewed elements.	2 - 4	Technology emphasis is on contingency capabilities, especially when crew is present Must be coordinated with AFM which will balance ground/onboard and human/computer responsibilities

Table 20.8.1-2: Candidate AR&D technologies and TRL Status

Lessons learned on other automated rendezvous systems, including those with a crewed element, have shown that this capability must be implemented as an overall spacecraft function rather than as a subsystem, otherwise cost, schedule and risk are all compromised significantly. A systems approach to the requirements, design, development, integration and test is critical to success and to achieving the risk mitigation goals. The appropriate level of automation must be evaluated and implemented along with the Failure Detection Isolation and Recovery (FDIR) functions. And finally, consideration of the crew role and the human-rating requirements must be made a priori to ensure proper integration with the GN&C system, to maintain safety, to achieve mission success goals, to exploit opportunities for dual use systems, and to perform arbitration and augmentation of the automated systems.

Automated & Precision Landing

Automated landing has been performed at the Moon and Mars (and elsewhere) with robotic systems; however, these have significantly different requirements than crewed systems. Future robotic precursors offer an excellent opportunity to test automated landing technologies and operations, and these technologies including guidance and navigation must be matured well in advance. Conveniently, two technologies identified as enabling for automated rendezvous and docking can also be enabling for landing: the LADAR and NFIR systems (see Table 20.8.1-2). They offer the ability to measure critical parameters such as altitude and descent rate, and to identify potential terrain hazards. Combined, these are sufficient to satisfy the navigation needs of the LDRM2 trade reference. However, complementary advanced guidance software for performing efficient, real time trajectory management and hazard avoidance must be also developed. The mass penalty in the overall architecture for excess propellant carried to the surface is significant. Algorithms and corresponding software that offer solutions are frequently overlooked as "technologies" that need to be matured and even tested in real environments. This enabling technology is estimated to be at a software analogous TRL of about 3.

Going beyond the basic LDRM-2 TRM, alternative lunar reference missions and most certainly Mars missions will require precision landing capabilities to support a base or to efficiently use previously emplaced surface infrastructure elements. Mars powered landing guidance may be required to remove large navigation-related range errors, state uncertainties, and wind effects (during for example, a parachute phase). Lunar landings may access navigation updates during powered landing if surface navigation aids, such as beacons, are baselined. Such surface aids would need to be developed, tested and deployed. These maneuvers, based on navigation updates, can require large amounts of propellant if not done as efficiently as possible. Control algorithms also need to be robust and correct for failures where appropriate. The combined GN&C TRL for precision landing has been assessed at 2.

Autonomous Flight Management

Autonomous Flight Management (AFM) is a key software technology that has the potential to reduce reliance on pre-launch mission design, enhance vehicle performance, and improve safety. It is intended to perform reliable decision-making during time critical situations and can provide a high level of spacecraft autonomy in all phases of flight by exploiting available computational capabilities. A modular design that allows for increasing capability can cost effectively balance ground/onboard and human/computer responsibility to meet the mission success and safety requirements for various architecture elements. The AFM system works in conjunction with GN&C and other vehicle management functions such as IVHM (Integrated Vehicle Health Management) and FDIR (Fault Detection Isolation and Recovery) to determine the best course of action. For the LDRMs being considered, AFM has the potential to significantly improve overall architecture implementation in cost, risk and safety. AFM is considered to be enabling for complex operations where no crew is present and time lags make ground operations impractical, such as rendezvous and docking, or precision landing at Mars. Some research and development activities with direct applicability have been performed in the Spacecraft Mission Assessment and Replanning Tool (SMART) project, and a TRL of 4 has been reached.

Deep Space Navigation

Execution for any mission beyond Low Earth Orbit (LEO) requires a Deep Space Navigation capability. The current Deep Space Network, which has long fulfilled this function for robotic spacecraft, is overburdened and the risk of having insufficient capabilities to fulfill the architecture objectives is significant. For example, DSN will be a driving element for the rendezvous and docking architecture at the Moon and Mars as it determines the range at which relative sensor acquisition must be performed to transition from inertial navigation. While the TRL for current robotic operations is clearly at 9, the limitations in availability, accuracy, and real-time utility of the current DSN system drive the true readiness level lower. The absence of detailed requirements or systematic analysis of DSN as applied to the LDRM and future Mars missions poses a further element of risk.

20.8.2 GN&C Subsystem Description

The primary functions for the GN&C system are fairly self-evident. The guidance function provides the vehicle trajectory and attitude planning and commanding based on navigation information and mission plan, and consists principally of software algorithms. The navigation function provides the vehicle position, velocity, acceleration, attitude and attitude rate measurement and estimation, and consists of both hardware sensors and accompanying software algorithms and filters. The navigation function may also include atmospheric measurements for the purpose of controlling planetary re-entry. The control function provides vehicle actuator commands based on the guidance function requests.

The GN&C subsystem depends heavily on the avionics subsystem, including computational resources, and many elements are integrated within the avionics purview. The GN&C subsystem also depends on many other vehicle subsystems, such as propulsion, docking mechanism capabilities, etc. A more detailed description of the specific functions is provided below for the principal flight regimes. For the purpose of this exercise, the GN&C subsystem encompasses the LDRM flight phases from initial LEO operations through to crew landing back on Earth. Note that dramatically varying levels of detail appear in the descriptions and discussions that follow. This was due to evolving information regarding the overall objectives, general accessibility and availability of data, and resource limitations.

20.8.2.1 Navigation

Navigation can be broken down into three main areas: attitude, inertial, and relative.

<u>Attitude Navigation</u>

Attitude navigation is required for pointing the spacecraft for power/thermal, communications, maneuvers, other navigation, viewing, etc. Attitude navigation includes determining the vehicle attitude either in an inertial frame, a local orbital frame, or a relative frame (such as attitude relative to another spacecraft) and maintaining that attitude knowledge when sensors cannot directly measure it. Sensors used for inertial and orbital attitude determination are very mature. Numerous sensors that are highly reliable are readily available at TRL 9. Sensors include star trackers, sun sensors and horizon sensors. Star trackers will work equally well in Earth orbit, Moon orbit, Mars orbit and L1. Star trackers are relatively light weight (< 5 kg) and low power consumption

(~10 W) but are typically expensive. As attitude determination is needed throughout the mission, a system that is at least two-fault tolerant is required. This fault tolerance can be accomplished with multiple star trackers or a combination of star trackers and other devices/methods for determining attitude. Attitude determination does not necessarily need to be done continuously. Depending on the accuracy of the gyros and the accuracy requirements for pointing, updates of attitude determination can be spaced quite a few hours apart. In fact, during proximity operations, it may sometimes be difficult to determine attitude because of interference with the other spacecraft. Star trackers can also be used for determining relative position at fairly large distances by identifying objects that don't fit into the expected star pattern. This technique has been used on the Space Shuttle and could be used on future vehicles. Star trackers can also be used in cis-Mars inertial navigation by identifying Mars and/or by identifying star occultation by Mars. If solar arrays that require pointing are included on the vehicle then sun sensors may also be included on the vehicle (allows sun pointing even if star trackers are not pointed for measurement).

Attitude navigation also includes attitude maintenance (keeping track of the attitude in the absence of direct measurements). This is typically done using gyros. Numerous sensors that are highly reliable are readily available at TRL 9. Gyros will work equally well in Earth orbit, Moon orbit, Mars orbit and L1. Gyros can be very lightweight and low power (a few pounds and a couple of watts) and are not that expensive. Highly accurate gyros can be much heavier, require more power, and be much more expensive. Accuracy requirements will dictate the selection. Note that gyros are often combined with accelerometers to form an inertial measurement unit (IMU) or inertial navigation system (INS). Because attitude maintenance is needed throughout a mission, a system that is at least two-fault tolerant is required. However, since the system is used continuously and is extremely critical to all phases of a mission, reliability requirements may indicate more than two-fault tolerance would be advisable.

The key design drivers for attitude navigation will be accuracy requirements for key maneuvers (such as orbital insertion or entry), fault tolerance (a critical subsystem for all phases), reliability (critical and used continuously), and interference/pointing issues (interference with other spacecraft or the need to point the vehicle for various operations).

Inertial Navigation

Inertial navigation is required to determine where the spacecraft is in order to calculate maneuvers, determine the coarse, relative location of other spacecraft or objects (Moon, Mars, Earth, L1, ground stations, lunar bases, etc.), determine pointing requirements, etc. Inertial navigation includes measurement of position and velocity and maintenance of that state when direct measurements cannot be taken. For spacecraft in Earth orbit the inertial navigation is typically accomplished through ground tracking, through TDRSS tracking or through GPS measurements. These methods are all fairly well established (TRL 9) and reliable (reliability of on-orbit GPS is still being evaluated but expected to be very reliable within the timeframe of CEV). With the use of software configurable radios it is likely that the same system may be used for communication as for TDRSS tracking or GPS measurements. This capability has been tested but the requirements will need to be integrated into the communications architecture for the Constellation program. This capability will be extremely useful in reducing the number of subsystems required to support a mission (lower mass, power, volume) and therefore should be pursued. For Moon orbit, Mars orbit, Earth/Moon L1, cis-Moon and cis-Mars operations, the primary method of inertial navigation is through the Deep Space Network (DSN). The DSN can use a series of techniques associated with the RF communication signal to determine position and velocity of the spacecraft. However, the navigation infrastructure for this task is extremely limited and primarily consists of the three Deep Space Network tracking facilities located in Goldstone, CA, Madrid, Spain, and Canberra, Australia. As they exist now, there is a significant risk that these tracking facilities and assets are insufficient for supporting future exploration needs because of limitations in availability, accuracy, and real-time utility. There are concepts for supporting navigation (along with support for communication) that need to be investigated to resolve issues with the limitations of the DSN. Additional tracking capability may be achieved from GPS signals "leakage" beyond the GPS shell and TDRSS ranging in the Earth-Moon vicinity to a limited degree but this capability has not been demonstrated. This technique may be very useful in removing errors from the initial maneuver to leave the Earth vicinity. There are also other techniques that use measurement of relative position of the Sun, Moon, Earth, Mars and occultation of stars by the Moon/Mars in support of inertial position determination. These can be helpful but are not likely to solve all of the issues related to deep space navigation.

Inertial navigation also includes inertial state maintenance (keeping track of the inertial state even when not directly measuring it). This is typically done using accelerometers. Numerous sensors that are highly reliable are readily available at TRL 9. Accelerometers will work equally well in Earth orbit, Moon orbit, Mars orbit and L1. Accelerometers can be very lightweight and low power (a few pounds and a couple of watts) and are not that expensive. Highly accurate accelerometers can be heavier, require more power, and be more expensive. Accuracy requirements will dictate which sensor is used. Note that as mentioned previously, accelerometers are often combined with gyros to form an inertial measurement unit (IMU) or inertial navigation system (INS). Because inertial state maintenance is needed throughout a mission, a system that is at least two-fault tolerant is required. However, since the system is used continuously (except possibly some "safe" modes) and is critical to all phases of the mission, reliability requirements may indicate more than two-fault tolerance would be advisable.

The key design drivers for inertial attitude navigation will be available resources for deep space navigation, and minimizing hardware requirements for LEO support

Relative Navigation

Relative navigation is required for rendezvous and docking and for landing. The rendezvous and docking relative navigation will be divided into far field rendezvous (before direct relative measurements can be taken – however, note that for some cases such as a low lunar rendezvous there may not be a far field rendezvous), near field rendezvous (from the beginning of direct relative state measurement), proximity operations (region where closed loop navigation/control occurs - as opposed to targeted burns), and departure.

Far Field Rendezvous Relative Navigation

During far-field rendezvous, the primary objective of the GN&C is to match orbits with the target vehicle. In order to do this position and velocity information is required, with velocity usually being the most critical element. Far field rendezvous navigation is typically based on the difference of inertial states (see Inertial State Navigation section for information on sensors). Errors in relative state can often be reduced by using the same sensors or same method of inertial state measurement on both vehicles (common errors cancel out). This will require a coordination of design across multiple vehicles/elements or the use of a common external resource (such as ground tracking or DSN). Even though the inertial state of each vehicle is maintained, measurement of the inertial states to support any rendezvous maneuvers will likely be required if the time between maneuvers is large (on the order of hours). One additional system on the vehicles required to support far field rendezvous relative navigation is the communication to allow the two inertial states to be collected in one place (either on the ground or on one of the vehicles) in order to difference them. For architecture phases where failure to rendezvous can lead to loss of crew (for example rendezvous after ascent from the lunar surface), all of the systems required will need a combined two-fault tolerance. This includes the inertial state information and communication on each vehicle. There are, however, some potential scenarios that will allow for estimated inertial state of one of the vehicles to be based solely on propagated state, and therefore loss of communication or inertial state measurement may not mean failure to rendezvous, thus potentially reducing fault tolerance requirements.

Near Field Rendezvous Relative Navigation

During near-field rendezvous the primary objective of the GN&C is to bring the two vehicles into close proximity. In order to accomplish this, position and velocity information is required. Near field rendezvous navigation is typically based on direct measurements between the two vehicles (although there are some exceptions such as relative GPS). The key design drivers in this region are range, fault tolerance, and FDIR. To fulfill these design needs for all current vehicles, multiple sensors (and/or sensors with multiple modes) are used. This impacts other design parameters such as power, weight, and avionics and software complexity. Relative navigation systems that cover large ranges (preferably from far field to dock) can help alleviate the demand on these design parameters. Sensors that are lightweight, require little power or on-board equipment, and that have the potential for dual-use as sensors (for example crew cameras, communication systems or landing sensors) can also help alleviate demand on these design parameters. Accuracy is also important, but high accuracy in this range is not a critical design driver. Multiple sensors (or multiple internal redundancies) will be required as the rendezvous will need to be two-fault tolerant for success. Dissimilar sensors have the advantage of more easily eliminating common mode failures than a single sensor type, but are not necessarily required

Proximity Operations Relative Navigation

For the purpose of this discussion, proximity operations relative navigation is needed in the region in which the attitude and dimensions of the target spacecraft become important. The objective is typically to get from close proximity to docking, and during departure, to get from a docked state to a region away from the target vehicle. There are some other contingencies to consider such as failed docking or break-out (abort maneuver), and some missions may include some level of visual or other inspection on the docking approach or after separation. The key design drivers include accuracy, fault tolerance, and FDIR (with fault tolerance and FDIR potentially driving power, weight, and avionics and software complexity). To support docking it is likely required that the relative navigation have full six degree of freedom measurement capability (relative position/velocity and relative attitude/attitude rate). The accuracy requirements on the navigation (and GN&C system overall) will be driven in large part by the capabilities of the docking mechanism combined with the mass of the vehicles involved. Using the same sensors as those used for longer range rendezvous can have a significant advantage but is not a requirement. Two-fault tolerance for success is required in this region. Dissimilar sensors have an advantage of being able to more easily eliminate common mode failures than a single sensor type, but are not necessarily required. To support crew operations (as required in the human rating requirements), relative navigation data to the crew and support for situational awareness will also be required.

Entry Navigation

Earth entry navigation requirements are minimal. The maneuver from L1 (for LDRM-2) and corrections along the path are based on inertial navigation, most likely using DSN, and this defines the entry interface target. Once at entry interface, temperature and/or pressure sensors in the form of an air data system can be used by the guidance system. The sensors, themselves, are at high TRLs and have low mass and low power requirements.

Landing Navigation

Lunar landing navigation to support LDRM-2 has two different additional needs: to reach the site from planetary orbit at some altitude, and to determine the terrain and potential hazards at the actual arrival point in order to land safely. The final set of landing navigation requirements will be driven by the mission objectives. Achieving detailed accuracy to support a base, for example, is a different problem than going to various sites with individual lander elements.

For missions with individual lander elements to unique sites, such as LDRM-2, altitude navigation combined with a local reference (using DSN) is sufficient to reach the site. Altitude sensors such as radar or laser altimeters provide sufficient resolution. This technology is at a high TRL (6 to 9) with laser altimeter units weighing several kg (or less) and consuming on the order of 10 to 100 W of power depending on, for example, range of operating altitude. Since the landing problem is conceptually similar to the rendezvous problem, opportunities for dual use exist with the rendezvous navigation sensors, such as LADAR, but this technology is currently at a lower TRL (4). If present as part of the existing complement of equipment, this can reduce overall mass, complexity and, possibly, resource consumption.

Real time terrain determination and navigation is also necessary for hazard detection and avoidance. Again, the potential for re-use exists with rendezvous sensors – namely, the LADAR and NFIR. These technologies are at a relatively low TRL (4), but could be advanced for the combined purpose of rendezvous and landing. Efficiently managing the overall vehicle path with this information could reduce the propellant requirements, which has an appreciable impact on LEO mass.

Earth landing can use the same altitude sensors as the lunar lander but because of the baselined water landing, there is no need for hazard detection/avoidance capability.

Ascent Navigation

This report does not address Earth ascent, but does include lunar ascent. For lunar ascent in LDRM-2, inertial deep space navigation is sufficient to target the L1 point and the complement of other general navigation sensors discussed previously provides attitude determination and maintenance.

20.8.2.2 Targeting and Guidance

This section does not address each flight phase individually, but rather addresses the needs for guidance and targeting overall. This consists of the targeting and guidance necessary to manage the overall trajectory as well as the attitude and rates of the spacecraft.

For phases of flight where updates are not time critical, ground based targeting is an option as either the prime or backup function. This levies requirements on the communication system (and other avionics functions) if it is included in the mission success/fault tolerance path. For all other flight phases, where communication time or capabilities are incompatible with guidance/targeting update requirements, on-board autonomy/automation combined with crew-in-the-loop functionality must be considered. In particular, for rendezvous & docking and lunar landing phases, the latter must be primary.

Because guidance and targeting consists of software algorithms, the need for technology development and analogous TRL assessment is often overlooked. However, significant advances in the realm of automation and autonomy are required to achieve the goals of LDRM-2 and the exploration architecture, in general. Operations at Mars, or the desire to more closely match Mars-like operations for lunar missions, would levy additional and more demanding automation/autonomy requirements, especially in elements where the crew is absent. Furthermore, rendezvous operations at a libration point have yet to be analyzed in detail. Algorithms for automated rendezvous and automated landing are estimated to be at a TRL of 2 to 3, and represent long lead-time development items in the software area. However, guidance and targeting algorithms for controlling attitude and other phases of flight for LDRM-2 are considered to be mature (TRL of 7 to 9).

20.8.2.3 Control

Spacecraft control requires the appropriate actuators and authority for accomplishing various translational and rotational vehicle maneuvering tasks. For the LDRM-2 TRM, propulsion systems are proposed to perform five distinctive classes of in-space maneuvers:

- large orbit maneuvers
- orbit correction maneuvers
- fine control for proximity operations and docking
- landing deceleration and ascent
- attitude control

In order to meet fault tolerance requirements, the propulsion system will be designed for minimum risk and/or include component level redundancy. It will include an orbital maneuvering system (OMS) and a smaller reaction control system (RCS). The capabilities (and requirements) may be distributed between vehicles when integrated free-flying or mated operations are considered.

In addition, for atmospheric flight on Earth re-entry (or for Mars arrival and departure), aerodynamic control may be considered. For the LDRM-2 TRM, the needs for Earth re-entry have not yet been fully explored.

When crew is present, an additional level of control is available via translational and rotational hand-controllers (THC & RHC), providing piloting authority (where AFM and contingency planning allow). The crew could have the capability to override autonomous functions when system degradation has progressed beyond automated recovery capability, or to recover from unanticipated situations. The level of control intervention will be determined by the AFM responsibility assessment and requirements. The various flight regimes and the propulsion control capability within each are provided for reference in Figure 20.8.2.3-1.

Although similar control functions on previous spacecraft have verified the control authority described herein, the TRLs for automatic and manual control for the LDRM-2 study have been placed in an interval from 6 to 9 due to the fact that the architecture will necessitate operations in regimes that have not been previously attempted.

Flight Phase	**Rotation Control**	**Translation Control**
Earth Orbit Vicinity		
Earth Orbit/De-orbit	RCS	OMS/RCS
Earth Orbit Rendezvous / Docking	RCS	RCS
Earth Orbit Arrival Departure	RCS	OMS
Earth Entry	RCS	
Earth Landing	N/A	N/A
Cis Lunar Space		
Mid-Course Corrections	RCS	OMS/RCS
L1 Vicinity		
L1 Arrival/Departure	RCS	OMS

L1 Station Keeping	RCS	RCS
L1 Rendezvous / Docking	RCS	OMS/RCS
Moon Vicinity		
Lunar Orbit	RCS	RCS/OMS
Lunar Ascent / Descent	RCS	OMS
Lunar Landing	RCS	RCS/OMS

Table 20.8.2.3-1: Propulsion Control for Differing Flight Regimes

20.8.2.4 Autonomous Mission and Flight Management

The CEV program will require autonomous capability for critical functions that were traditionally the responsibility of mission control or the on-board crew. Increased time delay and potential loss of signal in communications with Mars or Lunar spacecraft make the current mission control model ill-suited for the CEV program. The current model also uses Guidance, Navigation, and Control (GN&C) flight software that lacks self-reconfiguration capability and is heavily dependent on a pre-launch mission design process. Furthermore, certain critical functions need to be executed in the absence of on-board crew. As a result, the pre-launch flight products and limited onboard algorithms bound the real-time performance capability of the vehicle. These problems can be addressed by taking advantage of the computational advancements of the last 30 years. Functions such as real-time monitoring, option evaluation, and decision-making, can evolve from the ground-based mission control approach to a more balanced approach. The concept of an autonomous flight management (AFM) system is based on the approach of balancing ground/onboard and human/computer responsibility in a cost-effective manner.

<u>Autonomous Flight Management (AFM)</u>

Autonomy can be defined as “the ability of the vehicle, both crew and onboard systems, to operate independent of ground interaction.” Thus, AFM is defined, as a “system comprised of flight mechanics functions that execute the autonomous, reliable decision-making process and commands the vehicle GN&C system to perform the appropriate action during time critical situations.” AFM works in conjunction with the crew and other onboard computer-based systems that support the vehicle. This includes the Vehicle Management System (VMS), Integrated Vehicle Health Management (IVHM), and subsystem Fault Detection, Isolation, and Recovery (FDIR) systems, thereby reducing the reliance on pre-launch configuration and processing.

<u>AFM Background</u>

The idea of incorporating flight management functions onboard is certainly not new. However, consolidation of those functions into a system has not yet occurred in human-rated vehicles. The Space Shuttle program has tried over the last decade to improve performance and abort capability by implementing several modifications to the GN&C software, specifically in the guidance area. These changes enabled the Shuttle to utilize more of its available performance. However, other functions such as real-time abort monitoring and determination were not addressed until the creation of the Shuttle Cockpit Avionics (CAU) program in 2000. While not implementing complete autonomy, the Shuttle Abort Flight Management (SAFM) - developed to increase crew situational awareness and reduce their workload for abort determination and execution by providing the crew with dynamic information about the abort capability of the Shuttle during ascent and entry - was a significant step towards demonstrating AFM capability, having incorporated the key concepts of real-time monitoring, predication capability, and execution of GN&C commands.

JSC AFM Technology Project - SMART

Early investment in determining and maturing AFM capabilities is crucial to the exploration program. This includes a methodical determination of the appropriate level of autonomy in addition to advancing AFM technologies and GN&C algorithms. To this end, NASA-JSC has been developing an AFM concept and prototype called Spacecraft Mission Assessment and Replanning Tool (SMART). The goal of the SMART project is to develop a cost-effective AFM system that reduces the reliance on pre-launch mission design, increases vehicle performance and improves safety.

- The interfaces between SMART and GN&C components are kept clean and easily maintainable. Specific responsibilities are divided early in the design process to minimize impacts to GN&C design
- The SMART design can be easily combined with different GN&C systems without substantial redesign. This provides developers with the flexibility to utilize existing GN&C designs that can function in an AFM/GN&C system
- The GN&C system can function if the SMART software fails. GN&C will continue to execute the last plan and will not require the SMART to maintain that plan
- Advances in GN&C components can be incorporated with minimal impact to the SMART and vice versa. This allows for quick evaluation of advanced components within the GN&C or AFM domain

20.8.2.5 Fault Detection, Isolation, and Recovery

Lessons learned in the design and development of automated and crewed systems have shown that one of the most challenging areas of design, development and implementation is the GN&C Fault Detection Isolation and Recovery (FDIR), especially for time critical/safety critical fight regimes. Most technology assessments focus on the single string hardware (and sometimes software) required to perform the nominal function, but do not adequately address the detection of off-nominal behavior and the mitigation to safely continue operations or execute an optional contingency function. In the GN&C strategy for any exploration architecture including LDRM-2, the FDIR would work closely with the AFM function and the crew, when present, to provide information and options for decision-making. When integrated functionality is considered, such as in the rendezvous phase (before docking), or combined CEV/Injection Stage, Lunar Lander/Injection Stage or Lunar Lander/CEV operations, FDIR capabilities can be distributed between spacecraft elements to meet the overall requirements.

The GN&C FDIR (and AFM) functions must meet the overall fault tolerance and human-rating requirements. This collection of requirements will be a significant driver in the GN&C hardware and software selection and development. For example, dissimilar navigation sensors may be implemented to satisfy autonomy and human situational awareness while providing arbitration and/or augmentation capability. It is difficult to assess the FDIR TRL level in the absence of more detailed mission definition and requirements, but it is considered a long lead-time item, and should be considered in tandem with the AFM.

20.8.3 Technology Options

Figure 20.8.3-1 depicts a functional matrix containing current and advanced GN&C sensor technologies which includes sensor type, manufacturer, pertinent comments and weighted sensor scoring based on performance as well as mission criteria designating potential subsystem technologies deemed most likely to offer significant functional improvements in spacecraft design for exploration. Although relevant in establishing a purposeful side-by-side comparison of applicable sensor technologies, both the performance and mission criteria depicted for the weighted sensor scoring were incorporated from a previous study and, as such, have not been optimized for this LDRM-2 analysis.

<u>Sensor Scoring</u>

The score for each sensor was calculated by range of operation as a weighted average:

X = [5 * Accuracy + 3 * TRL + 2 * Cost + On-Orbit Lifetime + Risk + Cooperation Level] / 6

Note: Weights assigned by interpretation of mission objectives and criteria evaluation are qualitative

Sensor Weighted Performance Criteria

Accuracy (5) - Relative to applicable range:

Long-range [L], Mid-range [M] and Terminal-range (Docking) [D]

5: Exceeded

4: Met

3: Partially met

2: Not met

1: Unusable

Sensor Technology Readiness Level (TRL) Maturity (3)

5: TRL 8/9 With multiple missions in space

4: TRL 6/7 DTO's or integrated with vehicle

3: TRL 4/5 Tested in lab

2: TRL 2/3 Math concept proven

1: TRL 1 Concept formulated

Cost (2): Cost to advance TRL and purchase single flight unit

5: Less than $500k

4: $500k to less than $1M

3: $1M to less than $2M

2: $2M to less than $5M

1: More than $5M

Sensor Mission Criteria

On-Orbit Lifetime: Period in which the System can operate or remain dormant

5: Indefinite

4: Slight risk of corrosion/detaching

3: Moderate risk of corrosion/detaching

2: Radiation/Contamination problems

Risk / Redundancy required On-Orbit

5: No redundancy required - highly robust

4: Minimal sensor sub-element redundancy required

3: Dual Fault Tolerant for sensor to meet functional requirements

2: Single Fault Tolerant for sensor to meet functional requirements

1: Zero Fault Tolerant - multiple backups required

Cooperation Level

5: No cooperation

4: Needs stability or ground relayed communication

3: CEV & LL must maintain orientation or s/c-to-s/c communication

SENSOR NAME	DEFINITION	TYPE	PROGRAM/MISSION	MANUFACTURER	WEIGHTED RANGE SCORE	TRL / MATURITY	COST	ON-ORBIT LIFETIME	RISK / REDUNDANCY ON-ORBIT	ACCURACY	COOPERATION LEVEL	COMMENTS
ACVS	AutoTRAC Computer Vision System	Optical	STS-85 & STS-95	JSC	M8.5; D9.3	6 to 7	3	4	4	M4, D5	5	Uses single reflector, extremely accurate for docking, space demonstrated
ASVS	Advanced Space Vision System	Optical	Shuttle (STS-74) & ISS	CSA & MDR	M5.8	4 to 5	3	3	3	M2	4	
AVGS	Advanced Video Guidance Sensor	Laser	MSFC Engineering	OSC/MSFC	M8; D8	8 to 9	3	4	4	M3, D3	4	Lacks range and accuracy of Block II System, if target reflector is mounted on CEV & LL recommend upgrading to Block II
AVGS II	Advanced Video Guidance Sensor (next generation)	Laser	MSFC Engineering	OSC/MSFC	L8; M7.2; D8	4 to 5	1	4	3	L5, M4, D5	5	Extremely accurate at all ranges if development targets achieved, significant development cost, large complex reflector system
ETS-VII PXS	Proximity Sensor	Optical & CCD	ETS-VII	MELCO	M8.7; D8.7	8 to 9	3	4	3	M4, D4	4	Developed for specific engineering demonstration mission, not a production unit, may be hard to acquire from NASDA
ETS-VII RVR	Rendezvous Radar	Laser	ETS-VII	MELCO	M7.8	8 to 9	3	4	3	M3	4	Developed for specific engineering demonstration mission, not a production unit, may be hard to acquire from NASDA
GPS-DAbs	Global Potitioning System - Delta Absolute	Signal	N/A	N/A	L8	8 to 9	4	3	4	L3	3	GPS unit(s) on CEV & LL to difference inertial states, proven technology, on-orbit lifetime questions of receiver electronics
GPS-PR	GPS - precise relative	Signal	N/A	N/A	L8.8; M8	6 to 7	3	3	4	L5, M4	3	GPS unit(s) on CEV & LL to transmit measurements to chaser vehicle and process local and remote measurements to reduce common errors, demonstrated on-orbit, on-orbit lifetime questions of receiver electronics
Hand Held LIDAR	Manual laser used by Shuttle Crew	Laser	Shuttle	LTI	M7.5	8 to 9	1	5	3	M3	5	
IR Camera	Infra-red	Optical (IR)	N/A	N/A	M6.5	2 to 3	3	5	2	M3	5	
Ku band Radar	Shuttle Rendezvous Radar	Signal	Shuttle	Hughes	L5.5	8 to 9	1	5	1	L1	5	
KURS	Russian Transponder Radar	Signal	Soyuz, Progress	RSC-Energia	L6.3	8 to 9	3	2	2	L2	3	
LAMP	Low Attitude Mapping Photogrammetry	Optical	N/A	JPL	L5.8; M5.8; D5.8	4 to 5	2	5	3	L2, M2, D2	4	Development schedule in question (XSS11 dropped this sensor), good range but limited accuracy
LDRI	Laser Dynamic Range Imager	Laser with CCD	STS-97	Sandia	M8.7; D8.7	6 to 7	3	5	4	M4, D4	5	Scannerless LIDAR demonstrated on STS missions to ISS, possible range limitations beyond 50m, no reflectors required
Mini-Com Radar	Under Development by JPL	Signal	N/A	JPL	L5.7; M5.7; D5.7	2 to 3	3	5	2	L2, M2, D2	5	
MRR / MSTAR	Modulated Retro-Reflectors Moving & Stationary Target Acquisition Recognition	Optical	NRL Engineering	NRL	L8.2; M8.2; D8.2	4 to 5	3	3	3	L5, M5, D5	3	
NEAR Laser	Near Earth Astroid Rendezvous	Laser	NEAR	JPL/APL	L8; M8	8 to 9	2	5	4	L3, M3	5	Extreme long range sensor adds robustness to transition from inertial to relative navigation, demonstrated in deep space, could be modified to operate closer than 2km
NFIR	Natural Feature Image Resolution	Optical	JSC Engineering	NASA-JSC / ER	L7.2; M8; D8	4 to 5	3	5	3	L3, M4, D4	5	Natural feature recognition, optical navigation system can use features on CEV & LL now to perform precise nav, lighting conditions/control could be a risk factor
RELAVIS	Rendezvous Laser Vision System	Laser	N/A	OpTech	L8.7; M8.7; D8.7	6 to 7	3	5	4	L4, M4, D4	5	Relatively low cost non-cooperative sensor accurate at all but the last .5m of operations, would be the answer for many NASA needs if space demonstration on XSS11 successful
RF Tracking	Radar Frequency Tracking	Signal	JSC Engineering	NASA - JSC	L6.5; M6.5; D6.5	2 to 3	3	3	4	L3, M3, D3	5	
RVS/TGM	Rendezvous Sensor Telegoniometer	Scanning Laser	ATV & HTV	Sodern	L8; M8	8 to 9	3	4	4	L3, M3	4	ATV optical/reflector system, passable accuracy, limited to long and medium ranges
SLIR	Structured Light Image Recognition	Optical	N/A	NRL	D5.2	2 to 3	2	5	1	D2	5	
TCS	Trajectory Control Sensor	Scanning Laser	Shuttle	NASA-JSC / LM	L8; M8	8 to 9	3	4	4	L3, M3	4	Lacks range, operational flexibility of Block II, if reflectors are mounted on CEV & LL recommend upgrade to Block II
TCS II	Trajectory Control Sensor (next generation)	Laser	Shuttle	NASA-JSC / LM	L8; M7.2; D8	6 to 7	2	4	4	L4, M3, D4	4	Upgrade of current TCS technology to provide attitude information, operates in all ranges, benefit to Code T to realize upgrade
VDM	Videometer	Optical	ATV	Sodern	M8.2; D8.2	6 to 7	3	4	3	M4, D4	4	Still under development for the ATV, requires multiple targets for mid and near range operations
VGS	Video Guidance Sensor	Laser	STS 87 & STS 95	MSFC	L7.7; M6.8	8 to 9	3	3	3	L3, M2	4	Limited range and accuracy require additional sensors, if mounting reflectors on CEV & LL recommend AVGS II or TCS II
VisNav	Visual Navigation System - 3D	Optical ?	NSTL	Texas A&M	L5.3; M6.2; D5.3	4 to 5	2	3	2	L2, M3, D2	4	

Figure 20.8.3-1: GN&C Sensor Technology Options and Weighted Range Scoring

20.8.4 Candidate Subsystem Design Approach for LDRM-2

At the top level, the GN&C subsystem has been designed to be one-fault tolerant for mission success and, when crew is present, two-fault tolerant for safety. For flight phases where contingencies do not provide an alternative to preserve crew safety, two-fault tolerance for mission success is implemented. Multiple hardware strings were considered and software was assumed to reside on the three primary flight computers with the backup flight computer having only the GN&C algorithms required to maintain safety (or mission success for safety). Not all hardware is active in all flight phases. A first pass iteration was coordinated with the avionics subsystem team to reconcile mass and power budgets and to try to achieve a consistent baseline.

Maximum use was made of existing documentation and studies to assess TRL's and technology applicability. This was complemented by lessons learned of various team members. However, this meant that none of the studies were optimized for LDRM-2 or the exploration architecture. So, the candidate design equipment and software should not be construed as final recommendations. An attempt was made for practical consideration of dual use, especially for navigation equipment.

Figure 20.8.4-1 Sensor Redundancy vs. Range for Rendezvous/Docking (Pgs. 1 & 2) depicts the relevant system architectural element (i.e., CEV, Lunar Lander, Injection Stage) and the on-board sensor technologies redundancy criteria (Hot/Cold/Backup) versus navigation range for the Rendezvous and Docking phase of flight. This provides an example of the systematic analysis of the fault tolerance and redundancy for a first-cut verification of mission success and safety requirements for the candidate implementation.

Figure 20.8.4-2 Candidate GN&C Sensor Technology vs. Flight Phase (Pgs. 1 & 2) depicts a candidate set of GN&C equipment (and redundancy level) across each major flight regime for each LDRM-2 system architectural element. This yields the total complement of GN&C equipment and software needed on each of the elements while meeting performance, safety and human-rating requirements. The table also includes an estimate of TRL for the component hardware and software considered.

Notes for Figure 20.8.4-2:

1) Injection Stage: For disposal of IS (in all phases), attitude capability is required. Use of last, best state before separation is "good enough" for commanded disposal maneuver. However, we are not bookkeeping navigation capability to verify if it was "done right". This merits a trade study.

2) Natural Feature Image Recognition: The NFIR camera system serves as the camera for any on-board crew monitoring, as well as for video feed to Ground.

EV	EG	EV	EG																			
IS	IS	LL	LL			**Lunar Lander/Injection Stage 1 (LEO Dock, L1 Sep, 1 FT for mission success, 1 FT for collision, 1 FT for separation)**																
																						Bars on Nav Sensors Represent Usage for Relative Navigation Only
				Ground Tracking	C	C	C	C	C													Ground tracking not used unless problems arise (mission cost)
2*	2*	2	2*	TDRSS Tracking	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C											TDRSS Tracking and *GSP are S/W configurable RF - 2* RF system with 6 antennas
2*	2*	0	2*	GPS	2H	2H	2H	2H	2H	2H	2H											
1	2	2	3	Star Tracker		1H/1B/1C	1H/1B/1C	1H/1B/1C	1H/1B/1C													
0	0	2**	2	RF Rel Nav + Comm.	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***		2H***	2H***			NOTE 1
0	0	3	2	LADAR				2H	2H	2H	2H	2H	2H	2H	2H	2H		2H	2H	2H		
0	0	0	2	NFIR								1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		1H/1C	1H/1C	1H/1C		Puts lighting constraints on nominal mission
0	0	2	2	Camera								1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		1H/1C	1H/1C	1H/1C		
	2		0	Visual Targets/Marking																		Passive
0	0	0	2	Visual Target Lights										1H/1C	1H/1C							
0	0	0	1	Nav lights																		
0	0	3	3	Crew Monitors/Displays																		
0	0	3	3	Crew Input																		
0	0	2	2	Hand Controllers																		
	2		3	Docking Avionics										3H	3H	3H		3H				
2	2	4	4	IMU	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	
						Far Rndz		Mid Rndz		Near Rndz		Prox Ops		Final App.		Dock		Sep		Dep		
						> 50km		> 10 km		>1 km		> 200 m		<200 m		0		0		>10 m		
				NOTE 1:	Could have dual use RF nav but would prefer not to put antennas for "all the way to dock". **EV has 2 Space-to-Space Transcievers, 2 antennas, and 2 switch units - not full set for docking (unknown if ranging is included)																	

EV	EG	EV	EG																			
IS	IS	CEV	CEV			**CEV/Injection Stage 2 (LEO docking, L1 separation, 1FT for mission success, 2 FT for collision, separation, break-out from failed mating)**																
																						Bars on Nav Sensors Represent Usage for Relative Navigation Only
				Ground Tracking	C	C	C	C	C													Ground tracking not used unless problems arise (mission cost)
2*	2*	2	2*	TDRSS Tracking	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C											TDRSS Tracking and *GSP are S/W configurable RF - 2* RF system with 6 antennas
2*	2*	0	2*	GPS	2H	2H	2H	2H	2H	2H	2H											
1	2	2	3	Star Tracker		1H/1B/1C	1H/1B/1C	1H/1B/1C	1H/1B/1C													
0	0	2**	2	RF Rel Nav + Comm.	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***	2H***		2H***	2H***			NOTE 1
0	0	3	2	LADAR				2H	2H	2H	2H	2H	2H	2H	2H	2H		2H	2H	2H		
0	0	0	2	NFIR								1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		1H/1C	1H/1C	1H/1C		Puts lighting constraints on nominal mission
0	0	2	2	Camera					2H	2H	2H	2H	2H	2H	2H	2H		2H	2H	2H		
	2		0	Visual Targets/Marking																		Passive
0	0	0	2	Visual Target Lights										1H/1C	1H/1C							
0	1	0	1	Nav lights																		
0	0	3	3	Crew Monitors/Displays	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	
0	0	3	3	Crew Input	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	
0	0	2	2	Hand Controllers						1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		1H/1C	1H/1C	1H/1C		
	2		3	Docking Avionics										3H	3H	3H		3H				
2	2	4	4	IMU	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	
						Far Rndz		Mid Rndz		Near Rndz		Prox Ops		Final App.		Dock		Sep		Dep		
						> 50km		> 10 km		>1 km		> 200 m		<200 m		0		0		>10 m		
				NOTE 1:	Could have dual use RF nav but would prefer not to put antennas for "all the way to dock". **EV has 2 Space-to-Space Transcievers, 2 antennas, and 2 switch units - not full set for docking (unknown if ranging is included)																	

EV	EG	EV	EG																			
LL	LL	CEV	CEV			**CEV/Lunar Lander (1st L1 docking/sep, 1FT for mission success, 2 FT for collision, separation, break-out from failed matin**																
																						Bars on Nav Sensors Represent Usage for Relative Navigation Only
2	2	2	2	DSN Tracking	2H*/1C	2H*/1C	2H*/1C	2H*/1C	2H*/1C													Ground tracking not used unless problems arise (mission cost); Receive Hot & Trans Cold
2	3	2	3	Star Tracker		2B/1C	2B/1C	2B/1C	2B/1C													
2**	2	2**	2	RF Rel Nav + Comm.		2H	2H	2H	2H	2H	2H	2H	2H	2H	2H	2H		2H	2H	2H		NOTE 1
3	2	3	2	LADAR				2H	2H	2H	2H	2H	2H	2H	2H	2H		2C	2C	2C		
0	2	0	2	NFIR								1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		2C	2C	2C		Puts lighting constraints on nominal mission
2	2	2	2	Camera					2H	2H	2H	2H	2H	2H	2H	2H		2C	2C	2C		
	0		0	Visual Targets/Marking																		Passive
0	2	0	2	Visual Target Lights										1H/1C	1H/1C							
0	1	0	1	Nav lights														1H	1H	1H	1H	
3	3	3	3	Crew Monitors/Displays	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H					
3	3	3	3	Crew Input	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H					
2	2	2	2	Hand Controllers						1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C						
	3		3	Docking Avionics										3H	3H	3H		3H				
4	4	4	4	IMU	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	
						Far Rndz		Mid Rndz		Near Rndz		Prox Ops		Final App.		Dock		Sep		Dep		
						> 50km		> 10 km		>1 km		> 200 m		<200 m		0		0		>10 m		
				NOTE 1:	Could have dual use RF nav but would prefer not to put antennas for "all the way to dock". **EV has 2 Space-to-Space Transcievers, 2 antennas, and 2 switch units - not full set for docking (unknown if ranging is included)																	

Figure 20.8.4-1: Sensor Redundancy vs. Range for Rendezvous/Docking (Pg. 1/2)

H (Hot) = Active and being used for Nav and/or FDIR C (Cold) = Not powered but available for navigation B (Backup) = Powered but not being used for navigation (could be used if needed)

CEV/Lunar Lander (1st L1 docking/sep, 1FT for mission success, 2 FT for collision, separation, break-out from failed matin

EV	EG	EV	EG																			
LL	LL	CEV	CEV																			
																						Bars on Nav Sensors Represent Usage for Relative Navigation Only
2	2	2	2	DSN Tracking	2H*/1C	2H*/1C	2H*/1C	2H*/1C	2H*/1C													Ground tracking not used unless problems arise (mission cost); Receive Hot & Trans Cold
2	3	2	3	Star Tracker		2B/1C	2B/1C	2B/1C	2B/1C													
2**	2	2**	2	RF Rel Nav + Comm.		2H	2H	2H	2H	2H	2H	2H	2H	2H	2H	2H		2H	2H	2H		NOTE 1
3	2	3	2	LADAR				2C	2C	2C	2C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		1H/1C	1H/1C	1H/1C		
0	2	0	2	NFIR								2C	2C	2C	2C	2C		2H	2H	2H		Puts lighting constraints on nominal mission
2	2	2	2	Camera								2C	2C	2C	2C	2C		2H	2H	2H		
	0		0	Visual Targets/Marking																		Passive
0	2	0	2	Visual Target Lights										1H/1C	1H/1C							
0	1	0	1	Nav lights					1H	1H	1H	1H	1H									
3	3	3	3	Crew Monitors/Displays													3H	3H	3H	3H	3H	
3	3	3	3	Crew Input													3H	3H	3H	3H	3H	
2	2	2	2	Hand Controllers														1H/1C	1H/1C	1H/1C		
	3		3	Docking Avionics										3H	3H	3H		3H				
4	4	4	4	IMU	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	
						Far Rndz		Mid Rndz		Near Rndz		Prox Ops		Final App.		Dock		Sep		Dep		
						> 50km		> 10 km		>1 km		> 200 m		<200 m		0		0		>10 m		
				NOTE 1:	Could have dual use RF nav but would prefer not to put antennas for "all the way to dock". **EV has 2 Space-to-Space Transcievers, 2 antennas, and 2 switch units - not full set for docking (unknown if ranging is included)																	

CEV/Lunar Lander (2nd L1 docking/sep, 2FT for mission success, collision, separation, break-out from failed mating)

EV	EG	EV	EG																			
LL	LL	CEV	CEV																			
																						Bars on Nav Sensors Represent Usage for Relative Navigation Only
2	2	2	2	DSN Tracking	2H*/1C	2H*/1C	2H*/1C	2H*/1C	2H*/1C													Ground tracking not used unless problems arise (mission cost); Receive Hot & Trans Cold
2	3	2	3	Star Tracker		2B/1C	2B/1C	2B/1C	2B/1C													
2**	2	2**	2	RF Rel Nav + Comm.		2H	2H	2H	2H	2H	2H	2H	2H	2H	2H	2H		2H	2H	2H		NOTE 1
3	2	3	2	LADAR				2H	2H	2H	2H	2H	2H	2H	2H	2H		2C	2C	2C		
0	2	0	2	NFIR								1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		2C	2C	2C		Puts lighting constraints on nominal mission
2	2	2	2	Camera					2H	2H	2H	2H	2H	2H	2H	2H		2C	2C	2C		
	0		0	Visual Targets/Marking																		Passive
0	2	0	2	Visual Target Lights										2C	2C							
0	1	0	1	Nav lights														1H	1H	1H	1H	
3	3	3	3	Crew Monitors/Displays	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H					
3	3	3	3	Crew Input	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H	3H					
2	2	2	2	Hand Controllers						1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C						
	3		3	Docking Avionics										3H	3H	3H		3H				
4	4	4	4	IMU	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	
						Far Rndz		Mid Rndz		Near Rndz		Prox Ops		Final App.		Dock		Sep		Dep		
						> 50km		> 10 km		>1 km		> 200 m		<200 m		0		0		>10 m		
				NOTE 1:	Could have dual use RF nav but would prefer not to put antennas for "all the way to dock". **EV has 2 Space-to-Space Transcievers, 2 antennas, and 2 switch units - not full set for docking (unknown if ranging is included)																	

CEV/Lunar Lander (2nd L1 docking/sep, 2FT for mission success, collision, separation, break-out from failed mating)

EV	EG	EV	EG																			
LL	LL	CEV	CEV																			
																						Bars on Nav Sensors Represent Usage for Relative Navigation Only
2	2	2	2	DSN Tracking	2H*/1C	2H*/1C	2H*/1C	2H*/1C	2H*/1C													Ground tracking not used unless problems arise (mission cost); Receive Hot & Trans Cold
2	3	2	3	Star Tracker		2B/1C	2B/1C	2B/1C	2B/1C													
2**	2	2**	2	RF Rel Nav + Comm.		2H	2H	2H	2H	2H	2H	2H	2H	2H	2H	2H		2H	2H	2H		NOTE 1
3	2	3	2	LADAR				2C	2C	2C	2C	1H/1C	1H/1C	1H/1C	1H/1C	1H/1C		1H/1C	1H/1C	1H/1C		
0	2	0	2	NFIR								2C	2C	2C	2C	2C		2H	2H	2H		Puts lighting constraints on nominal mission
2	2	2	2	Camera								2C	2C	2C	2C	2C		2H	2H	2H		
	0		0	Visual Targets/Marking																		Passive
0	2	0	2	Visual Target Lights										1H/1C	1H/1C							
0	1	0	1	Nav lights					1H	1H	1H	1H	1H									
3	3	3	3	Crew Monitors/Displays													3H	3H	3H	3H	3H	
3	3	3	3	Crew Input													3H	3H	3H	3H	3H	
2	2	2	2	Hand Controllers														1H/1C	1H/1C	1H/1C		
	3		3	Docking Avionics										3H	3H	3H		3H				
4	4	4	4	IMU	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	4H	
						Far Rndz		Mid Rndz		Near Rndz		Prox Ops		Final App.		Dock		Sep		Dep		
						> 50km		> 10 km		>1 km		> 200 m		<200 m		0		0		>10 m		
				NOTE 1:	Could have dual use RF nav but would prefer not to put antennas for "all the way to dock". **EV has 2 Space-to-Space Transcievers, 2 antennas, and 2 switch units - not full set for docking (unknown if ranging is included)																	

Figure 20.8.4-1: Sensor Redundancy vs. Range for Rendezvous/Docking (Pg. 2/2)

H (Hot) = Active and being used for Nav and/or FDIR C (Cold) = Not powered but available for navigation B (Backup) = Powered but not being used for navigation (could be used if needed)

Function	Sub-function	H/W	TRL	Red	Red	Red	H/W	TRL	Red	Red	Red	H/W	TRL	Red	Red	Red	H/W	TRL	Red	Red	Red
NAVIGATION		Earth	Earth	CEV	LL	IS	Cis-Lunar	Cis-Lunar	CEV	LL	IS	L1	L1	CEV	LL	IS	Moon	Moon	CEV	LL	IS
Gen'l Navigation																					
	Determine attitude	Star Tracker	9	3	3	2	Star Tracker	9	3	3	2	Star Tracker	9	3	3	2	Star Tracker	9	N/A	3	N/A
	Determine attitude rate	Gyros	9	4	4	2	Gyros	9	4	4	2	Gyros	9	4	4	2	Gyros	9	N/A	4	N/A
	Maintain attitude knowledge	Gyros	9	4	4	2	Gyros	9	4	4	2	Gyros	9	4	4	2	Gyros	9	N/A	4	N/A
	Determine inertial state knowledge	GPS TDRSS	9* 9*	3 2	3 2	2 0	DSN	9	2	2	2	DSN	9	2	2	0	DSN (+Optical)	6 to 9	N/A	3	N/A
	Maintain inertial state knowledge	Accelerometers	9	3	3	2	Accelerometers	9	3	3	2	Accelerometers	9	3	3	2	Accelerometers	9	N/A	3	N/A
	Determine accelerations	Accelerometers	9	3	3	2	Accelerometers	9	3	3	2	Accelerometers	9	3	3	2	Accelerometers	9	N/A	3	N/A
	Maintain reference frames	S/W only	-	3**	3**	3	S/W only	9	3**	3**	3	S/W only	-	3**	3**	3	S/W only	9	N/A	3**	N/A
Relative Navigation																					
	Long Distance Relative State	GPS TDRSS	9* 9	2 2	3 2	0 0	DSN	6 to 9	2	2	N/A	DSN	6 to 9	2	2	0			N/A	N/A	N/A
	Far Field Relative State	RF Nav (combined w/ comm)	2 to 3	2	2	0			N/A	N/A	N/A	RF Nav (combined w/ comm)	2 to 3	2	2	0			N/A	N/A	N/A
	Near Field Relative State	LADAR RF Nav (combined w/ comm)	6 to 7 2 to 3	2 2	2 2	0			N/A	N/A	N/A	LADAR RF Nav (combined w/ comm)	6 to 7 2 to 3	2 2	2 2	0 0			N/A	N/A	N/A
	Close in Relative State	LADAR RF Nav (combined w/ comm) NFIR	6 to 7 2 to 3 4 to 5	2 2 2	2 2 2	0 0 0			N/A	N/A	N/A	LADAR RF Nav (combined w/ comm) NFIR	6 to 7 2 to 3 4 to 5	2 2 2	2 2 2	0 0 0			N/A	N/A	N/A
	Visual info for Crew + NFIR	Camera	7 to 9	2	2	0			N/A	N/A	N/A								N/A	N/A	N/A
	Beacon for crew	Navigation Lights	9	0	0	1			N/A	N/A	N/A	Navigation Lights	9	1	1	0			N/A	N/A	N/A
	Visual scene illumination	Visual Target Lighting	9	2	2	0			N/A	N/A	N/A	Visual Target Lighting	9	2	2	0			N/A	N/A	N/A
	Navigation Relative to Lunar Base													N/A	N/A	N/A	Beacon, Transponder		N/A	N/A	N/A
De-Orbit/Entry Navigation																					
	Altitude Determination	LADAR Radar	6 to 7 9	N/A 2	N/A N/A	N/A N/A			N/A	N/A	N/A			N/A	N/A	N/A	LADAR Radar	6 to 7 9	N/A	N/A 2	N/A
	Attitude Determination/Maintenance	Gyros	9	4	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
	Determine Inertial State	GPS	9	2	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
	Maintain Inertial State	Accelerometers	9	3	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
	Air Data System	Pressure, Temperature	9	3	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
																			N/A	N/A	N/A
Landing Navigation																					
	Altitude Determination	LADAR Radar	6 to 7 9	2 2	N/A N/A	N/A N/A			N/A	N/A	N/A			N/A	N/A	N/A	LADAR Radar	6 to 7 9	N/A	2 2	N/A
	Determine Inertial State	GPS	9	2	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	DSN		N/A	3	N/A
	Maintain inertial state	Accelerometers	9	3	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	Accelerometers		N/A	3	N/A
	Air Data System	Pressure, Temperature	9	0	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
	Landing site evaluation - hazard detection			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	LADAR NFIR	4 to 5 4 to 5	N/A	2 2	N/A
Lunar Ascent Navigation																					
	*covered by General Navigation at Moon			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	see General Navigation				

Figure 20.8.4-2: Candidate GN&C Sensor Technology vs. Flight Phase (Pg. 1/2)

**Assumes 3 Primary Flight Critical Computers are employed for mission success and Single-String Back-up Flight Computer is employed in contingency case(s).

Function		Sub-function	H/W	TRL	Red	Red	Red	H/W	TRL	Red	Red	Red	H/W	TRL	Red	Red	Red	H/W	TRL	Red	Red	Red
GUIDANCE/TARGETING			Earth	Earth	CEV	LL	IS	Cis-Lunar	Cis-Lunar	CEV	LL	IS	L1	L1	CEV	LL	IS	Moon	Moon	CEV	LL	IS
	Inertial Targeting																					
		Ground based targeting	Space-to-Ground Comm.	9	2	2	2	Space-to-Ground Comm.	9	2	2	2	Space-to-Ground Comm.	7 to 9	2	2	2	Space-to-Ground Comm.	9	N/A	2	N/A
		On-board targeting	S/W only	6 to 9	3	3	3	S/W only	6 to 9	3	3	3	S/W only	6 to 9	3	3	3	S/W only	6 to 9	N/A	3	N/A
	Inertial Guidance																					
		Attitude (pointing)	S/W only	9	3**	3**	3	S/W only	9	3**	3**	3	S/W only	6 to 9	3**	3**	3	S/W only	6 to 9	N/A	3**	N/A
		Translation	S/W only	6 to 9	3**	3**	3	S/W only	6 to 9	3**	3**	3	S/W only	6 to 9	3**	3**	3	S/W only	6 to 9	N/A	3**	N/A
	Rendezvous Targeting																					
		Ground based targeting	Space-to-Ground Comm.	9	2	2	N/A			N/A	N/A	N/A	Space-to-Ground Comm.	6 to 9	2	2	N/A	Space-to-Ground Comm.	9	N/A	N/A	N/A
		On-board targeting	S/W only	6 to 9	3**	3**	N/A			N/A	N/A	N/A	S/W only	4 to 9	3**	3**	N/A	S/W only	6 to 9	N/A	N/A	N/A
	Rendezvous Guidance																					
		Attitude (pointing)	S/W only	9	3**	3**	N/A			N/A	N/A	N/A	S/W only	6 to 9	3**	3**	N/A	S/W only	6 to 9	N/A	N/A	N/A
		Translation	S/W only	6 to 9	3**	3**	N/A			N/A	N/A	N/A	S/W only	4 to 9	3**	3**	N/A	S/W only	6 to 9	N/A	N/A	N/A
	Entry Targeting									N/A	N/A	N/A										
		Ground based targeting	Space-to-Ground Comm.	4 to 9	2	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
		On-board targeting	S/W only	4 to 9	3**	N/A	N/A								N/A	N/A	N/A			N/A	N/A	N/A
	Entry Guidance																					
		Attitude (pointing)	S/W only	3 to 9	3**	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
		Translation	S/W only	5 to 9	3**	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A
	Landing Targeting																					
		Ground based targeting	Space-to-Ground Comm.	6 to 9	N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	Space-to-Ground Comm.	9	N/A	2	N/A
		On-board targeting	S/W only	6 to 9	N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	S/W only	4 to 9	N/A	3**	N/A
	Landing Guidance																					
		Attitude (pointing)	S/W only	6 to 9	N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	S/W only	4 to 9	N/A	3**	N/A
		Translation	S/W only	6 to 9	N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	S/W only	4 to 9	N/A	3**	N/A
		Hazard avoidance	S/W only	3 to 6	N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	S/W only	3 to 6	N/A	3**	N/A
	Lunar Ascent Targeting																					
		Ground based targeting	Space-to-Ground Comm.	6 to 9	N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	Space-to-Ground Comm.	9	N/A	2	N/A
		On-board targeting	S/W only	6 to 9	N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	S/W only	4 to 9	N/A	3**	N/A
	Lunar Ascent Guidance																					
		Attitude (pointing)	S/W only		N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	S/W only	4 to 9	N/A	3**	N/A
		Translation	S/W only		N/A	N/A	N/A			N/A	N/A	N/A			N/A	N/A	N/A	S/W only	4 to 9	N/A	3**	N/A
CONTROL																						
	Control																					
		Attitude - Calculate maneuvers	S/W	6 to 9	3**	3**	3	S/W	6 to 9	3**	3**	3	S/W	6 to 9	3**	3**	3	S/W	6 to 9	N/A	3**	N/A
		Attitude - Execute maneuvers	Propulsion	9	3	3	3	Propulsion	9	3	3	3	Propulsion	9	3	3	3	Propulsion	9	N/A	3	N/A
		Attitude - Manual Takeover	Rotation Hand Controller	6 to 9	2	N/A	N/A	Rotation Hand Controller	6 to 9	2	N/A	N/A	Rotation Hand Controller	4 to 9	2	2	N/A	Rotation Hand Controller	6 to 9	N/A	2	N/A
		Translation - Calculate maneuvers	S/W	6 to 9	3**	3**	3	S/W	6 to 9	3**	3**	3	S/W	6 to 9	3**	3**	3	S/W	6 to 9	N/A	3**	N/A
		Translation - Execute maneuvers	Propulsion	9	3	3	3	Propulsion	9	3	3	3	Propulsion	9	3	3	3	Propulsion	9	N/A	3	N/A
		Translation - Manual Takeover	Trans. Hand Controller	6 to 9	2	N/A	N/A	Trans. Hand Controller	6 to 9	2	N/A	N/A	Trans. Hand Controller	4 to 9	2	2	N/A	Trans. Hand Controller	6 to 9	N/A	2	N/A
		FDI	Gyros Accelerometers Pressure transducers Temperature transducers Valve sensors										Gyros Accelerometers Pressure transducers Temperature transducers Valve sensors					Gyros Accelerometers Pressure transducers Temperature transducers Valve sensors				
Autonomous Flight Manager (AFM)			S/W	2 to 4	3**	3**	TBD	S/W	2 to 4	3**	3**	TBD	S/W	2 to 4	3**	3**	TBD	S/W	2 to 4	N/A	3**	N/A
	SMART																					

Figure 20.8.4-2: Candidate GN&C Sensor Technology vs. Flight Phase (Pg. 2/2)

**Assumes 3 Primary Flight Critical Computers are employed for mission success and Single-String Back-up Flight Computer is employed in contingency case(s).

20.8.5 Mars Spiral Development

Beyond the technologies required to support LDRM2 GN&C, several important items must be considered to enable a Mars mission. These include, but are not limited to:

- more advanced, autonomous flight management systems
- precision landing navigation infrastructure or strategy
- aerocapture and/or atmospheric entry GN&C
- surface mobility and transportation GN&C (depending on exploration requirements)

Additional technologies that have the potential to significantly enhance safety and mission success while reducing overall life cycle cost include:

- Deep Space Navigation
- Very long range relative navigation sensors
- Low thrust trajectory design and GN&C
- Multi-spacecraft rendezvous beyond LEO

20.9 Communications and Tracking Subsystem Technology Report

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20.9.1 Subsystem Description

20.9.1.1 Primary Functions

The communications and tracking subsystem consists of the equipment for the CEV, Lunar Lander, and Injection Stages to be able to provide communications and tracking between these elements and to the ground. Information on the communication links will include commands, telemetry, voice, video, and payload data.

The current communication infrastructure will not be able to support the communication and tracking needs of a human lunar mission. This is due to the availability and high data rates needed on both the forward and return links to support humans. The Space Network can not provide adequate coverage for human lunar missions. The Deep Space Network currently does not support high data rate forward links and it is also fully loaded with other deep space missions. The communication infrastructure which will support this mission has not been baselined. It is assumed that the infrastructure will provide continuous coverage to the spacecraft elements.

Figure 20.9.1.1-1 is a representation of the functional requirements of the communications required between the elements. This figure should not imply that there is a direct link between every element. For example, while EVA voice and data does need to get to the ground, it will likely be relayed through the CEV or lander instead of a direct link to the ground.

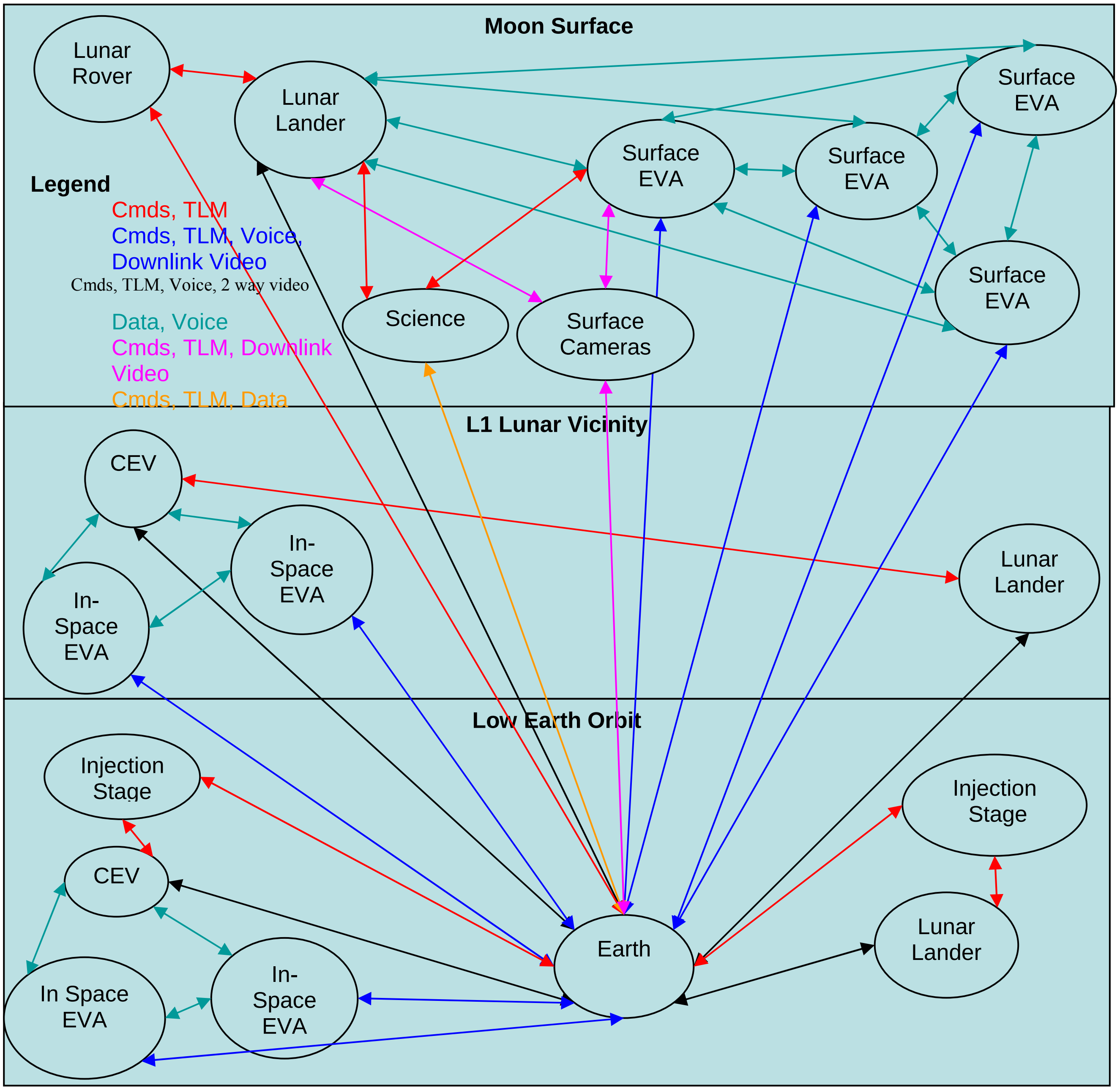

Figure 20.9.1.1-1: Global Lunar Access Architecture Illustration

20.9.1.2 Key design parameters (design drivers)

The Space-to-Ground communication systems for the CEV and Lunar Lander are dependent on the communications infrastructure that the vehicles will be communicating with. The size of the antenna and transmit power on the vehicle will depend on whether it is communicating directly to the Earth or using a relay satellite.

A comparison of the effect of different possible infrastructure architectures on the spacecraft will be assessed.

Two possible communication infrastructures will be investigated to compare the effects on the spacecraft communication system. The first architecture is direct communication between the lunar surface and a Earth ground station network. The second is through a low lunar orbiting relay communications relay network.

To compare possible communication infrastructures some assumptions were made on the minimum data requirements based on the communication requirements in Figure 20.9.1.1-1. It is assumed that the S-band omni data rate must be at least 20 kbps on forward link and 25 kbps on the return link. This will provide at least 1 full duplex voice channel and a minimum amount of commands and telemetry in order to recover the high data rate link. The lander power amplifier was sized to provide 25kbps on the direct to Earth link and then this amplifier power was used on all of the other S-band links. The Ka-band high data rate link will provide multiple channels of HDTV and science data in the assumed 300Mbps data rate. The power amplifier was again sized on the direct to Earth link.

Data Rates/Margins (dB)							
	Ground to Lander	**Lander to Ground**	**Ground to LOR to Lander**	**Lander to LOR to ground**	**Spacecraft Assumptions**	**Ground Station Assumptions**	**LOR Assumptions**
Vehicle-Comm Link	**FWD**	**RTN**	**FWD**	**RTN**			
Lander-Ka-band	25Mbps/3.0	300Mbps/3.0	25Mbps/3.0	375Mbps/3.0	1m antenna, 43W Amp	12m antenna, 5.5W Amp	LRO to Lunar:1m ant, 10W Amp, LRO to Ground: 1m ant, 46W Amp,Ground to LRO: 12m ant, 5.5W amp (Ka-band)
Lander-S-band HDR	1.0Mbps/9.8	7Mbps/3.0	1.0Mbps/9.8	42 Mbps/3.0	1m (24dB) ant, 110W Amp	12m antenna, 110W Amp	LRO to Lunar:1m ant, 12.4W Amp, LRO to Ground: 1m ant, 30W Amp,Ground to LRO: 12m ant, 4.5W amp (Ka-band)
Lander-S-band MDR	315kbps/3.0	400kbps/3.0	316kbps/3.0	2800kbps/3.0	12dB horn antenna, 110W Amp	12m antenna, 110W Amp	LRO to Lunar:1m ant, 12.4W Amp, LRO to Ground: 1m ant, 30W Amp,Ground to LRO: 12m ant, 4.5W amp (Ka-band)
Lander-S-band LDR	20kbps/3.0	25kbps/3.0	20Kbps/3.0	180kbps/3.0	0dB Omni ant, 110 W Amp	12m antenna, 110W Amp	LRO to Lunar:1m ant, 12.4W Amp, LRO to Ground: 1m ant, 30W Amp,Ground to LRO: 12m ant, 4.5W amp (Ka-band)

Table 20.9.1.2-1: Lunar Lander Data Rate Table

Data Rates/Margins (dB)						
	DSN to CEV	**CEV to DSN**	**Ground to CEV thru TDRSS**	**CEV to ground thru TDRS**	**Spacecraft Assumptions**	**DSN Ground Station Assumptions**
Vehicle-Comm Link	**FWD**	**RTN**	**FWD**	**RTN**		
CEV-Ka-band	25Mbps/3.0	265Mbps/3.0	25Mbps/3.0	150Mbps/3.0	1m antenna, 17W Amp	12m antenna, 4.2W Amp
CEV-S-band MDR	315kbps/3.0	400kbps/3.0	620kbps/3.0	1370kbps/3.0	12dB horn antenna, 35.5W Amp	12m antenna, 87W Amp
CEV-S-band LDR	20kbps/3.0	25kbps/3.0	20Kbps/3.0	88kbps/3.0	0dB Omni ant, 35.5 W Amp	12m antenna, 87W Amp

Table 20.9.1.2-2: CEV Data Rate Table

It can be seen from the results in the Table 20.9.1.2-1 that for the same lander antenna and power amplifier that the Lunar Orbiting Relay (LOR) satellite will provide higher data rates. The LOR will also provide coverage to the poles and the far side of moon. The amount of coverage depends on the number of satellites and their orbits. The Jet Propulsion Laboratory has been working on a three satellite constellation with a highly elliptical orbit that will provide excellent coverage to the south pole and some coverage to other areas. This constellation can also provide navigation information to elements on the lunar surface. While the direct to Earth links are easier to implement, they can only provide communication to the near side of the moon.

20.9.2 Technology Options

Technologies considered for the LDRM-2 CEV and Lander are addressed below for each major C&T functional element.

C&T Technologies	TRL	Pros for LDRM2	Cons for LDRM2	Comments
Communication				
Space-to-Ground				
Optical	2-7	High data rates possible	Pointing, need for low attitude uncertainty for initial acquisition and low platform jitter for tracking.	Reference: "Optical Communications Backbone Network to Support Exploration", Bernard Edwards, Michael Dennis, NASA/GSFC
Ka-band	4-5	High data rates possible with smaller antennas	Power amplifiers are not very efficient at this frequency.	
Software Defined Radio	4-5	Reduces size and weight of communication subsystems by having one box that can communicate with multiple networks (TDRSS, DSN, GPS, etc.)		Software defined radios have been tested in space but are not available yet for very high data rate communications.

Space-to-Space				
UWB	2	Can be used for tracking, Can operate at higher data rates than 802.11	Designed for short distances currently	Will need to look at using higher gain antennas and higher power amplifiers that are not allowed on Earth in order to increase range of system. May not be able to use near Earth because of interference.
802.11 Wireless	10	Available off the shelf but will need modification for space environment	No tracking capability, currently limited to 54Mbps	Concern about using it near Earth because of interference
Tracking				
GPS	10	Good for Low Earth Orbit	Coverage for Lunar Vicinity and Surface. No coverage for far side of Moon.	
Optical	4-5	Can be used for communication also.	Pointing	
DSN	10		No coverage for far side of Moon.	

Table 20.9.2-1: CEV Data Rate Table

Software Reconfigurable Radio

In order to avoid having a different communication system on the spacecraft for each network, a reconfigurable software defined radio should help reduce the size and weight of the communication system.

Project Constellation will have numerous elements that require communications with the ground and between elements. These elements maybe developed by different projects and will also communicate with infrastructure that is already in place or will need to be developed to support Exploration missions. The spacecraft communications systems may need to communicate with numerous networks as it goes from one mission phase such as ascent, low Earth orbit, lunar surface, and descent to another (Global Positioning System, Tracking and Data Relay Satellite System, Deep Space Network ground stations, new Lunar Relay Satellites, Search and Rescue Satellite, Air Traffic Control, and Space-to-Space links with other elements of the project). A technology that could be used to accomplish this is reconfigurable software defined radios. It is also important that the elements be able to communicate with each other to facilitate rendezvous and docking. Having a common space-to-space communication system and data protocols will make communicating between elements easier to integrate especially when new elements are developed.

Reconfigurable Software Defined Radios (SDR) are capable of operating with the wide variety of NASA communication systems that might be employed by various vehicles or subsystems having been developed at different times in an integrated surface or orbital work site. Human planetary surface exploration and deep space missions will deploy a wide variety of vehicles, tools and experiments that have unique requirements for modulation, data rate, etc. Reconfigurable SDR techniques are desired to minimize the development time and cost for providing such diverse communication systems, and to minimize the number of separate radios necessary on various spacecraft.

SDR components must be small and low power to broaden their application range. In the future, multiple functions may be implemented in large integrated devices that provide communication, video, processing, control and data acquisition. Target applications include such functions as integrating communications functions with remote battery or solar powered single-chip data acquisition functions. An ultra-simplified and efficient form of SDR is also desirable, for implementation in smaller Field Programmable Gate Arrays (FPGA's) or Application Specific Integrated Circuits (ASIC's).

The SDR should be able to accommodate various NASA and contractor developed communication systems, which support surface-to-surface, space-to-space and space-to-surface links, by reprogramming the SDR and with minimum RF front-end hardware changes. Consideration should be given for any CAD/EDA or other commercial tools or components used as to suitability, configuration control and maintainability for use in a long-duration mission-critical environment.

Lunar Surface Communications

Lunar surface communications and tracking will be another challenge due numerous elements that need communications, high data rates needed for video and science data, frequency spectrum allocations, distance between elements, and hills or craters that could reduce radio frequency coverage. A couple of technologies should be evaluated for possible application to a lunar surface communication and tracking network. One technology is Ultra Wide Band and the other is 802.11 wireless network.

Depending on the communications infrastructure, navigation information may be difficult to determine on the lunar surface. The Ultra Wideband (UWB) radios can be designed for passive position tracking. This system could be used to track the astronaut's positions while working on the lunar surface.

UWB radios do not transmit continuous radio waves like conventional radio. Only pulses are transmitted, and power is used only during the short duration of those pulses. A typical duty cycle is 1/200 and the pulse width is less than 1 nanosecond. The FCC (Federal Communications Commission) approved the deployment of UWB technology in commercial sector from 3.1 GHz to 10.6 GHz in February 2002 for terrestrial use. UWB technology is exploited to implement the video communications and tracking system due to its properties, such as high data rate, fine time resolution, and low power spectral density. The UWB system is robust to multipath interference, can co-exist with other radio systems used by the landing vehicle, and can provide precise tracking capability. The fine time resolution allows a corresponding tracking algorithm TDOA (Time Difference of Arrival) to be employed with which information can be extracted from the telemetry data. The feasibility of using UWB radios for position tracking has been demonstrated in the laboratory. A possible architecture for a lunar surface network is shown in the figure below.

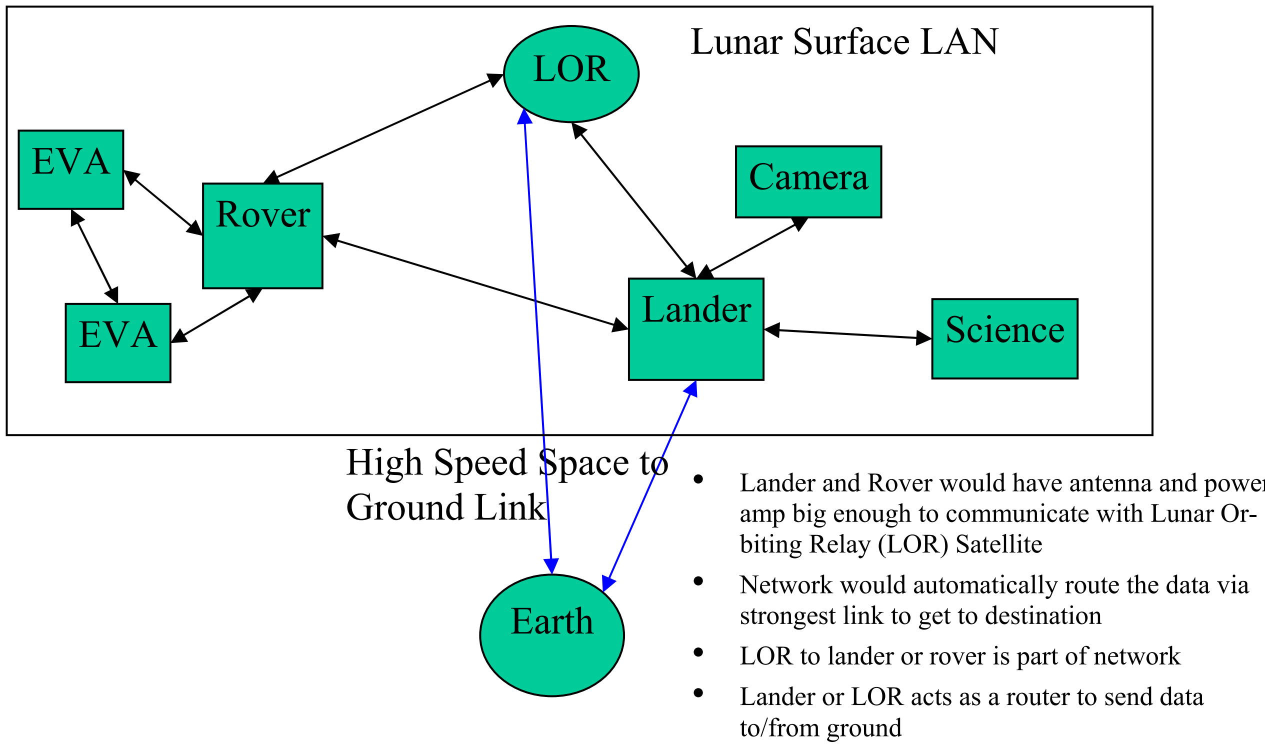

Figure 20.9.2-1: Lunar Surface Network

Surface LAN Requirements

To help size the Lunar Surface network, the following requirements were developed. No system currently available can meet all of these requirements, but it is probably possible by the 2009 timeframe to have one at TRL 6.

- Data rate: 100Mbps (estimate based on 2 HDTV channels (40Mbps), 4 voice channels (80kbps), commands (10kbps), telemetry (192kbps), science data (60 Mbps))
- System range: 10km diameter around lunar lander
- Link range: >1km from element to element

- Tracking accuracy: 10m (absolute), 1m (relative) DSN or other external nav can initialize the absolute
- Antenna: user antenna: omni similar to 802.11 access point or integrated into EVA suit, Access point antenna: <0.25 m
- Power: User: < 5W, Access Point: < 40W
- Ad-hoc network
- Standard network protocols for data routing
- Dynamic allocation of bandwidth
- Quality of service to ensure voice and other priority data is delivered in order, without packet loss, minimum delay
- Security: As required by National Security Agency to ensure security of LAN
- Radios: small, low power, reconfigurable, software defined
- Goal of using this LAN for all space to space communications for mission (CEV to In-Space EVAs, CEV to Lander during rendezvous)
- Spectrum Management: Ensure proper frequency allocation and Radio Frequency Interference does not degrade link

20.9.3 Recommended Subsystem Design Approach for LDRM-2

The following tables show the assumed equipment needed for the CEV, lander, and Earth Departure Stages.

CEV Communication	**Qty**	**Unit Weight (lb)**	**Unit Avg. Power (Watts)**	**Total Weight (lbs)**	**Total Avg. Power (Watts)**
Space-to-Ground Voice/Cmds/TLM System	2	30	275	60	275
Space-to-Ground Antennas	4	20	0	80	0
Space-to-Space Transceivers	2	20	25	40	25
Space-to-Space Antennas	2	10	0	20	0
Space-to-Space Switch Units	2	10	0	20	0
Internal TV Camera	2	10	20	20	40
External TV Camera	2	20	20	40	40
TV Compressor/Encryptor/Recorder	2	10	30	20	60
Video Switching system	2	10	15	20	30
Crew Intercom	4	10	10	40	40
ATC UHF system	2	10	75	20	150
Search and Rescue System	2	10	20	20	40
High Data Rate Space to Ground System	1	75	215	75	215

Table 20.9.3-1: CEV Communication Subsystem

Lander Communication	**Qty**	**Unit Weight (lb)**	**Unit Avg. Power (Watts)**	**Total Weight (lbs)**	**Total Avg. Power (Watts)**
Space-to-Ground Voice/Cmds/TLM System	2	30	275	60	275
Space-to-Ground Antennas	4	20	0	80	0
Space-to-Space Transceivers	2	20	25	40	25
Space-to-Space Antennas	2	10	0	20	0
Space-to-Space Switch Units	2	10	0	20	0
Space-to-Space Wireless Network	2	10	10	20	20
Internal TV Camera	2	10	20	20	40
External TV Camera	2	20	20	40	40
TV Compressor/Encryptor/Recorder	2	10	30	20	60
Video Switching system	2	10	15	20	30
Crew Intercom	4	10	10	40	40
High Data Rate Space to Ground System	1	75	215	75	215

Table 20.9.3-2: Lunar Lander Communication Subsystem

Assumptions for CEV and Lander C&T

1) CEV is not used as a L1 Comm Relay for the Lunar Lander. If it were needed as a relay then the High Data Rate Comm System on the CEV would have to increased to a higher data rate and another High Data Rate Comm System would have to be added to the CEV for CEV to Lunar Lander comm.

2) Space-to-Space Comm System will be used to communicate with EVAs, Lunar Lander during rendezvous, and CEV Earth Departure stage during rendezvous.

3) ATC UHF system is used for backup voice communications during ascent and entry.

4) Lunar Lander High Data Rate Space to Ground System will be used for 2 way video conferences, inspection video, docking video and payload data. Assume 25Mbps Fwd and 300Mbps Return. This system can also be a backup to Cmd/TLM system. Assume 1m Ka-band dish and RF power of 43Watts. Assume efficiency of power amp to be 20%. System weighs about 75lbs (amp:15lbs, ant:25lbs, transceiver:20lbs, misc: 15lbs.)

5) CEV Space-to-Ground Cmd/Tlm system will be used for commands, telemetry, 2 way voice, and low rate video. The assumptions are: 110W RF power, 12dB ant gain for medium data rate communications and an omni antenna for low data rate emergency communications. The low data rate communications will be at least 20kbps on the forward link and 25kbps on the return link. Lunar Lander would have a similar system but will not be able to communicate at these rates on lunar surface unless there is a Low Lunar Orbiting Relay Satellite.

6) Lunar Lander will have a Space-to-Space Wireless Network to communicate with surface elements such as EVAs, cameras, rovers, science payloads. This network will provide at least 54Mbps and carry voice, video, commands and telemetry.

7) General assumptions are that EVA suit communications system allocations are carried with the EVA suit and science, surface cameras and EVA Enhanced Mobility communication system allocations are included in the science payload allocation.

8) TV cameras for CEV and Lunar Lander are 2 external video cameras for docking ops and 2 internal video cameras for astronaut video conferences.

Earth Departure Stage Communication	**Qty**	**Unit Weight (lb)**	**Unit Avg. Power (Watts)**	**Total Weight (lbs)**	**Total Avg. Power (Watts)**
Space to Ground CMD/TLM transceiver & GPS	2	10	30	20	60
S-band and L-band antennas	4	10	0	40	0
Space to Space CMD/TLM system	2	10	10	20	20

Table 20.9.3-3: Earth Departure Stage Communication Subsystem

Assumptions for Earth Departure Stage

1) Earth Departure Stages will need Space-to-Ground cmd/tlm system for communicating with ground. Estimate is based on Low Power Transceiver from GSFC. This transceiver also includes GPS. Earth Departure Stage will also need Space-to-Space communication system to provide cmds/tlm to CEV or Lunar Lander to which it is docking.

20.9.4 Mars Spiral Development

The biggest concerns for communication systems for the Mars missions are time delays and the increase in power or antenna size or reduction in data rate capability due to the greater distance.

The round trip time delay between the Moon and Earth is less than three seconds which, in most cases, does not greatly impair real time communications. The round trip time delay between Mars and the Earth is between 6 and 44 minutes. These delays will make autonomous operations critical for Mars exploration.

While optical communications for Lunar operations is probably not necessary, it may be critical for high data rate links between Mars and the Earth.

20.10 Structures Technology Report

Gregg Edeen, NASA/JSC/ES – Structural Engineering Division

20.10.1 Subsystem Description

The structures subsystem provides the primary load bearing structure designed to withstand all mission structural loading environments as well as providing secondary structure for the support of vehicle systems. For most rigid structures, currently available material and fabrication technology will be used. The structural mass will be determined by the design of the structure and the environments, like loads and thermal. A wide variety of current structural fabrication and materials technologies can be used to support this program without further development; however, some materials will require more work be done.

Composite materials use requires a properties development program be developed for the particular application and lay-up, but the basic materials themselves are fairly well understood. Each new lay-up of fibers and resins has unique characteristics that need to be understood. The characteristics include fabrication characteristics of the part as well as basic material properties.

Aluminum-Lithium materials show promise for decreasing density and increasing stiffness, but more work is needed to characterize the materials, particularly for reusable structures. New materials like AL 2099, 2097, and 2195 are candidate materials at varying levels of technology readiness. AL 2195 is currently used in the Shuttle External Tank, but it will need additional materials characterization for a reusable vehicle. The TRL levels can be summarized as follows:

Aluminum-Lithium	TRL	Density (lb/in^3)	Strength (Ksi)	Module (Msi)
AL 2097	5	0.096	60	11.0
AL 2099	5	0.0945	65	11.5
AL 2195	9	0.0975	73	11.0

Inflatable structures may be used in a number of components of the lunar mission. Inflatable crew modules require significant development, but they offer large potential mass savings if the module is large enough and it is suitable for deployment in orbit or on the Moon's surface. The TRL for space inflatable crew modules is 4 or less. Inflatable modules have two primary advantages. They don't have to endure launch loads in the fully deployed configuration and they can be packaged into a smaller volume for launch and re-entry. The airbag recovery system suggested for the crew module must also be developed and demonstrated for this application. Airbags have been successfully used before, but this combination or size, environment, and application is new. TRL for this could be considered 7 or less.

20.10.2 Technology Options

A wide variety of technology options are available. Currently available metallic materials have significantly higher strengths compared to the same types of materials from 20 or 30 years ago. Aluminum-Lithium alloys show promise but need some additional development for reusable applications. Composite materials are better understood but still require development for the specific application. Inflatable structures may be used, but significant development is needed.

20.10.3 Recommended LDRM-2 Subsystem Design Approach

Weight will be one of the most important factors to the success of the program, so the development of lightweight structures will be critical. Structures should use designs and fabrication methods that are weight efficient. In some cases weight efficiency may have to be traded against reusability as an example.

Initial design should use materials that have been characterized, or will be characterized before fabrication begins. Since the loading environments effect weight just as much as material properties and design do, the mission should be designed to reduce loading if possible (for instance, lower cabin pressure, lower entry gees, lower abort loading).

During preliminary design sensitivity, studies should be performed to evaluate design, material, and fabrication choices made. A periodic weight review should be held to determine if the systems would reach their weight targets.

Technology development issues such as inflatable modules and landing airbags should have precursor development programs laid out to ensure success. Alternate approaches should be suggested in case the technology is not ready.

20.10.4 Mars Spiral Development

The environment and exposure duration for a Mars mission will be different than a Moon mission; however, many of the elements and mission phases are the same. Components like the crew Earth entry vehicle and surface habitats might be the same or similar so that they can just be an extension of the existing design. Other elements like the transit crew module might be significantly larger so that it is a new design. The technology for the new design could be developed by a smaller precursor designed for the Moon mission. With the exception of inflatables fabrication, methods and materials will be similar to those used on the Moon mission that just precedes the Mars mission.

20.11 Passive Thermal Control System Technology Report

Steve Rickman, NASA/JSC/ES – Structural Engineering Division

20.11.1 Subsystem Description

The Passive Thermal Control System will be designed to maintain all components, subsystems, and systems within their specified temperature ranges. The PTCS will protect the spacecraft from environmental temperature extremes in all expected environments.

The key parameters for consideration in the PTCS design are: the expected operating environments, component temperature limits, overall heat rejection, power requirements, and weight considerations. A properly designed PTCS will meet the needs within the available power, weight, and volume resource allocations.

The PTCS for the proposed DRM will consist of judiciously selected thermo-optical coatings tailored to meet the component thermal requirements as dictated by operational and non-operational temperature limits within the context of the design thermal environment. Heat dissipation will be accomplished using low solar absorptance (α), high infrared emittance (ε) coatings on radiator surfaces. This provides not only an optimum surface for heat rejection, it also desensitizes the radiator surface from direct sunlight as well as solar energy reflected from the Earth's surface and the lunar surface (albedo flux). Coatings to be used on the lunar surface will be selected to ensure they may be easily cleaned of the lunar surface dust – a contaminant that will degrade (and warm-bias) the component through degradation of α.

The PTCS will also utilize one or more varieties of multi-layer insulation (MLI) tailored for the specific application. The insulation is light weight and very efficient and will shield components from the temperature extremes expected during the various mission phases. The exterior surface of the MLI will be selected for robustness in the solar ultraviolet environment, material temperature limits, and suitability and stability of the thermo-optical properties.

Overall, an orbiting spacecraft will experience a variety of environments. Careful placement of components during the design phase, with considerable input from thermal analysis, will result in a better integrated design. Such a configuration can effectively utilize waste heat from one component to make up for an energy deficit in another. The ultimate goal is to minimize additional heater power needs. However, it must be noted that past spacecraft design philosophy suggests that it is more desirable to cold-bias a configuration as it is easier to heat the configuration (using thermostatically controlled heaters) than it is to cool it (requiring, perhaps, pumps, working fluids, heat pipes, radiators, etc.). The overall thermal design will be one that considers all of these factors. It is not to be assumed that all thermal control will be accomplished through passive means. Rather, the split between what can be achieved passively and what requires an active system will be defined as the configuration matures.

Prolonged stays on the Moon will require thermal control technologies to protect the lander from extreme daytime temperatures as well as the extreme cold of lunar night. Daytime temperatures may be mitigated with careful placement of radiating surfaces such that the view to the lunar surface is minimized. Additionally, low α, high ε coatings on surfaces will reduce the solar input.

However, care must be taken in coating property selection because a surface with a high ε will admit infrared energy from the lunar surface if there is a high view factor to the surface.

MLI, in concert with a tightly integrated thermal design, will allow the hardware remaining on the lunar surface to survive and/or function during the nearly two-week-long lunar night. Special care must be given to the insulation design and installation so as to minimize heat leaks. Given that the cold environment lasts for such an extended period, a small heat leak with integrate into a significant energy requirement. One possible means of reducing power needs is to consider the use of radioisotope heater units (RHU). Each RHU generates one Watt of heat as a byproduct of radioactive decay. Another possible means of lunar night survival may be the use of insulated thermal compartments. These compartments, used successfully on the Surveyor missions, provide an isolated environment in which components may survive the lunar night using only limited power. The configuration consists of a box, insulated with MLI on all but on side. The remaining side is a radiator and becomes physically isolated from the internal components during cold periods via the opening of a thermal switch – a mechanical switch driven by a bi-metallic that changes shape based on temperature. Use of such a compartment significantly reduces the need for make-up energy (i.e., heater power).

20.11.2 Technology Options

Many currently available passive thermal control technologies will satisfy the needs for future lunar missions however there is potential for improvement in the area of thermo-optical coatings. Some research is being performed in the area of nano-structured additives to tailor the performance of these coatings.

Additional benefits may be derived from improved analytical capabilities for electronics components and integrated, multi-discipline spacecraft analysis.

20.11.3 Recommended LDRM-2 Subsystem Design Approach

Estimation of vehicle resource requirements (component total mass and power requirements) cannot be performed until a configuration is defined. Given the distributed nature of passive thermal control hardware solutions, specification may be made with increasing fidelity as the vehicle design matures. Power requirements will be driven by component temperature limits, operating environment, configuration, attitudes, and the degree of integration in the thermal design.

The recommended subsystem design approach is detailed in the following flow:

1. Identify the design reference mission and the overall system requirements;
2. From the design reference mission, define the expected flight thermal environments including ascent, Earth-orbit assembly/loiter, trans-lunar, L1 staging/loiter/assembly, lunar descent and landing, lunar surface operations, lunar ascent and rendezvous, trans-Earth coast, and atmospheric entry and landing;
3. Perform preliminary thermal analysis on candidate design concepts identifying gross PTCS characteristics (preliminary coating selection, radiator sizing, assessments of

component locations, identification of potential thermal control problems associated with isolation and/or make-up energy requirements). These efforts are focused at establishing a baseline system in support of a preliminary design review (PDR);

4. Progressively refine and enhance the detail of thermal mathematical models as the design matures with the objective of finalizing coating selections, required MLI locations and blanket performance as the critical design review (CDR) approaches. Development and refinement of thermal mathematical models will likely involve the performance of a number of component and/or subsystem level thermal-vacuum tests to determine critical parameters not easily obtained from analysis (such as interface conductances, insulation effective emittance, etc.). PTCS development will greatly benefit from system level thermal-vacuum testing by obtaining integrated thermal performance and overall system heat balance data. These data are critical to model correlation and overall system verification;
5. The correlated thermal models are analyzed within the context for the variety of expected thermal environments. The analysis will ensure that all components will remain within their specified temperature limits with acceptable margins.

20.11.4 Mars Spiral Development

Systems and subsystems developed for human lunar exploration will have been designed to withstand a variety of space environments. Additional design and analysis will likely be required to accommodate these subsystems in a configuration for human exploration of Mars. Surface systems designed for the vacuum environment of the Moon will likely not suffice for Mars surface operations as the environments are vastly different. Martian temperature extremes are less severe than those experienced on the Moon and the diurnal temperature variation occurs on a much shorter scale time. These will tend to make the environmental extremes less severe than on the Moon. However, the presence of the martian atmosphere, even at reduced pressures, results in a considerable convective heat transfer component that will require a different thermal control scheme than that employed on the Moon. Mars surface equipment will likely require bulk silica fiber insulations or aerogels. Multilayer insulation will not be effective and thermal control coatings will have only limited effectiveness.

Spacecraft not subjected to the martian atmosphere, however, will benefit from robust thermo-optical coating development and any technological developments performed to improve MLI technology. Additionally, improved thermal analysis tools and optimization employed for spacecraft integration will be of great benefit for Mars-bound spacecraft.

Since the mission duration for a Mars mission is significantly greater than that planned for a lunar mission, system reliability will be a key factor in ensuring success. High reliability heater systems and thermostats can be tested during lunar missions and should reduce the risk of failure on prolonged Mars missions.

20.12 Thermal Protection System Technology Report

Chris Madden, NASA/JSC/ES – Structural Engineering Division

20.12.1 Subsystem Description

The primary function of the thermal protection system (TPS) is to protect the spacecraft from the aerothermodynamic heating during reentry. Secondary functions and considerations include micrometeoroid and orbital debris (MMOD) shielding, radiation protection, and thermal control for on-orbit operations. The most significant design driver is the reentry heating rate transient and spatial distribution on the external surface. The heating rate is proportional to the reentry velocity. The determination of accurate heating rates is critical so that the TPS thickness can be accurately determined with the appropriate level of conservatism. The thickness of the TPS must be controlled in order to reduce the mass fraction. Typically redundancy is not provided in a thermal protection system although recent efforts and design concepts have considered a backup layer of TPS in the event of a heat shield penetration. These types of concepts should be carefully weighed against the mass and reliability trades. Design/material options such as hardening the surface for impact resistance is highly recommended. Typically the TPS does utilize any vehicle resources other than engineering flight instrumentation to be used for post-flight assessments. There are no real-time data from the instrumentation system used for flight control. During mission phases such as Earth orbit the pointing of the spacecraft should consider the orientation of the heat shield for protection from MMOD. Synergy of the TPS system with other systems can be realized by utilizing advanced technology to incorporate cosmic radiation within the TPS itself and construction of durable TPS as discussed previously. Although the TPS thickness is usually driven by the reentry heating rates, the design should also consider on-orbit thermal environments so that the thermal control system is appropriately designed.

20.12.2 Technology Options

For inter-planetary and lunar return missions, the reentry velocities result in very high heating rates which necessitate the use of ablative TPS in most cases. The original Apollo ablator material AVCOAT-5061 is no longer available. A human-rated material does not exist today which is suitable for lunar returns alhtough significant progress in the development of a replacement material has been made by Applied Research Associates (ARA). ARA has developed a family of ablative materials which should be able to meet missions requirements. Further testing is required to human-rate the material system.

20.12.3 Recommended LDRM-2 Subsystem Design Approach

The TPS design approach should follow a similar approach to previous spacecraft including Space Shuttle and X-38. After outer mold line (OML) shape selection, the TPS is sized based on a design trajectory and conservative predictions for the heating rates. Then the inner mold line (IML) is determined from the TPS sizing analysis. At this point, the spacecraft heatshield can be designed. Successive design/analysis cycles will provide verification of the sizing and provide further resolution of heat distribution for the thermal control system design. In parallel, details of the TPS design may be incorporated including penetrations and interfaces to mechanical subsystems.

Rationale for the material selection will draw from the arc-jet test and thermal performance as well as the compatibility with the environmental requirements such as toxicity, atomic oxygen resistance, humidity, vacuum, etc. As total vehicle mass will be of utmost concern, the TPS material selection and system design will be heavily driven by mass concerns. Mass estimates for the ablator heat shield is estimated to be 2,700 lbm for an Apollo shape 132% of Apollo scale for direct entry.

The TPS system should induce no significant restraints on mission duration or crew-days.

20.13 Advanced Mating System Technology Report

James Lewis, NASA/JSC/ES – Structural Engineering Division

20.13.1 Introduction

The U.S. Space Program has a newly defined vision that includes long-term planning built on a spiral development philosophy. The scope of human exploration taken in incremental steps becomes feasible and manageable, tempered only by fiscal limits. As typical with any Program formulation phase, early requirements development should consider common architectural elements that offer modularity and reusability to minimize unique interfaces, operations, hardware, and development. Proper architectural planning offers substantial long-term savings by reducing cost and schedule risk. One credible need that offers tangible, long-term benefits and savings is the application of a standard, multipurpose space vehicle mating interface (docking mechanism) and operations to be used throughout the program's mission architecture for vehicle rendezvous and in-space assembly. Early recognition that elevating mating attachment hardware and operations to the architectural element level allows timely mating requirements development into high-level program requirements and vehicle and mission development and planning.

When space flight hardware (including a new next-generation mating element) can be standardized and used in multiple mission profiles, overall development, certification, and life cycle costs are generally reduced. By controlling the standard hardware interface, NASA can more effectively control many of the larger mission architecture parameters even if different contractors and foreign partners provide the modules and vehicles. Such proof is shown by the successful use of the Common Berthing Mechanism as a segment-to-segment attachment device for the ISS. By developing and certifying a common interface and using it as a standard between modules made by different contractors and different international partners, the ISS reduced integration complexity and risk by simplifying interface requirements. Continued NASA development and use of a standardized mating element as a key component in the exploration infrastructure is recommended. And while both NASA and the Russia Space Agency have a long history of involvement in mating system development and use, for future development and leadership this technology paper considers U.S. interests only.

NASA has been involved in space mating systems development and use throughout U.S. space flight; e.g., Gemini, Apollo, Apollo-Soyuz, ISS assembly. Additional experience has been gained in support of the procurement, modification, and re-certification of the APAS for shuttle-Mir and shuttle-ISS. Recent experience includes detailed development of LIDS for the X-38 Crew Return Vehicle (CRV) program, the operational integration of ISS visiting vehicles (automated transfer vehicle and the H-II transfer vehicle), and a host of unique ISS logistics berthing mechanisms. Except for the Russian APAS, NASA personnel have always maintained a lead role in the development of human-transfer rated mating interfaces for its programs. NASA civil servant experience has great depth and is inclusive of every aspect of design, development, certification, testing, and use. NASA has the necessary experience and capability for long-lead development of a new mating system and should take the lead in establishing the new mating system baseline for the Exploration Program.

20.13.2 Description of Mating System Functions

Spacecraft vehicles, modules and structures will require a mating attachment device to support rendezvous and in-space assembly. Typically, mating systems provide the mechanical, structural, and electrical systems to:

- Provide soft-capture and dynamic attenuation of the two mating interfaces to prevent over specification of vehicle propulsion and control authority.
- Provide hard-capture of the two mating interfaces to support mated dynamics and pressurization for crew and cargo transfer.
- While mated provide the utilities required transferring power, data, and fluids across the interface.
- Provide the necessary functions to support nominal and expedited separation of mated interfaces.
- For crewed vehicles provide the necessary pressure vessel close out, i.e. hatch, to allow for crew compartment volume to pressurized and maintained during operations.

Key mating system interfaces include the crew and human factors; the passive thermal control system; power and data systems; primary and secondary structure; integrated vehicle control, command logic, and system health management; pressurization, depressurization and air sampling; maintenance and checkout; operations and training.

20.13.3 Driving Requirements Affecting Mating Systems

Mating system considerations are largely tied to the ops concepts, to the structural loads during capture and while mated, to the need for crew transfer, and to the size of cargo and/or the type of utility to be transferred.

Current LDRM-2 operations concepts include two LEO rendezvous missions to assemble an injection stage to the lunar Lander and an injection stage to the crew transfer vehicle. This is followed by a rendezvous of the crew transfer vehicle to the Lander at L1 or some other point close to the Moon prior to the moon landing and again after lunar Lander departure from the moon.

In an effort to minimize the number of unique mechanisms required for development and qualification and to support future Exploration program mission flexibility, it is offered that any future mating system built provide a fully androgynous interface, i.e. two identical copies of the mating interface can mate to each other. This would provide a common set of hardware, operations, and training and support a standard set of mating system functions that are the same throughout the Program whether in LOE orbit or around the Moon or Mars. This also supports an anytime and anywhere mating functionality beneficial to crew rescue operations and system level of redundancy since any two mating vehicles would have the necessary systems to support the capture and mating of each other.

Thirty years of using 'mechanically passive' docking systems have indicated a need for designing a mating system that requires little or no force for capture and alignment during mating. Current mating systems are mechanically complex and require significant force (i.e. velocity) to engage latches and comply mechanisms. Both Russian docking systems are mechanically complex and contain single-point failures and zero-fault-tolerant functions. In combination with required dynamic operational aspects, this creates critical operations with lower reliability and very limited flexibility. Additionally, the use of these high force systems drives other design and operational aspects for vehicles such as primary and secondary structure, and positional accuracy and control authority for guidance, navigation, and propulsion. A low impact capture capability would dually support berthing dynamic in-space assembly of segments, modules, or structures.

Another driving requirement would be time critical two fault tolerant separation. Existing systems do not meet this requirement and utilize interface-scarring pyrotechnics to meet 1-fault tolerance. The Shuttle Program goes as far as to accept the use of a 4-hour EVA for removal of 96 bolts at an interface plane on the Orbiter Docking System to be 2-fault tolerant. The ISS Common Berthing Mechanism is also 1-fault tolerant by use of a pyrotechnic on the power bolts it uses for hard capture.

Another driving requirement will be the size of items to be transferred across the mating interface. Existing docking systems are sized for crew transfer only and have openings approximately 32 inches in diameter. This limits the overall size of the items transferred to approximately shoulder width. One factor that is related to consider is the use of the mating interface port as the egress path for EVA. If EVA suited crewmembers and their EVA payload become too large then a separate opening would be required while using current mating systems or force cargo to be divide further into smaller pieces adding assembly time large items. A new mating interface and its transfer diameter would be designed to meet the requirements for in-space assembly and EVA for the Exploration Program. A larger diameter is likely needed to support the mated loads of larger diameter modules during injection burns. These loads will likely be larger than loads seen on existing systems used on the ISS. Lastly, no existing mating system offers automatic mating of fluid transfer umbilicals that are likely to be required for future unmanned Exploration missions.

20.13.4 Technology Selection and Assessments

20.13.4.1 Existing Technology

Currently available for use are the three mating systems on the International Space Station (ISS). The two Russian docking mechanisms are complex and have performance limitations that create highly dynamic, critical operations, increasing risk for missions, vehicles, and crews and they do not support berthing operations. The criticality and hazardous nature of docking was demonstrated in 1997 when a Russian Progress re-supply vessel missed the docking port and collided with the Mir Space Station while performing a high-velocity docking during a piloted training exercise. The collision, which could have resulted in loss of life, was a direct result of hardware limitations and human failures coupled with the highly dynamic operational nature required for mating using an existing mechanism. The third mating system currently available is the ISS berthing mechanism, which requires a robotic arm to deliver and to align interfaces and will not support the docking dynamics associated with autonomous mating operations. All three existing systems failed to meet the fault tolerance requirements identified during the X-38, SLI, and OSP programs. This, coupled with other limitations, indicates a real need for development of a new mating system.

APAS

The Russian APAS uses an extensible capture ring with three inward-facing guide petals with capture latches mounted midway down the outside of the petals (see Figure 20.13.4.1-1). The capture ring and soft-capture latches require force created by the vehicle closing velocity to align and engage, respectively. When the passive mating ring is captive, the active ring extends to equalize ball screw actuator length and to align the vehicles; it then retracts to engage structural latches mounted on the periphery of the APAS base structure. The latches provide the structural clamping force (preload) necessary to compress the seals for pressurization and to withstand the dynamic forces applied at the interface while mated. The latching system uses a complex, interconnected cable-driven structural hook system with a pyrotechnic as backup. During capture, the passively mechanical, complex APAS extension/retraction system requires a separate dynamic load attenuation system in series with the capture ring system to dampen out post capture dynamics. Because of the significant contact forces needed for alignment and soft capture, the APAS cannot be used for berthing.

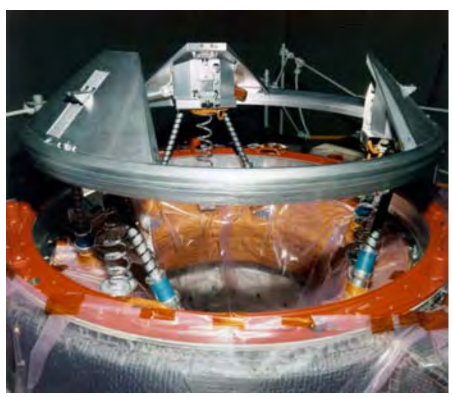

Figure 20.13.4.1-1: Russian APAS

Probe-Drogue

The Russian probe-drogue system (probe cone or pin cone) is functionally the same as the APAS, but differs in the embodiment of the soft-capture system (see Figure 20.13.4.1-2). The design is geared toward axial or centerline mating of small vehicles. Instead of a capture ring, an active pin mates to a passive cone. The cone provides the gross-to-fine alignment for the pin during operations. The pin contains trip latches that are actuated when bottomed out in the cone for soft capture and are driven back during docking to attenuate contact loads. After capture, the pin is fully retracted to allow for structural mating and sealing for pressurization. In most other aspects, the pin and cone system is identical to the APAS, including its design as a docking system only; i.e., it is not credible for berthing. After structural mating, the pin and cone are removed to allow for transfer of crew and cargo, but they have to be reinstalled prior to undocking to ready the docking interface for use. Because of its central placement and limited size, the pin is susceptible to damage from bending if post-capture dynamic excursions are not limited.

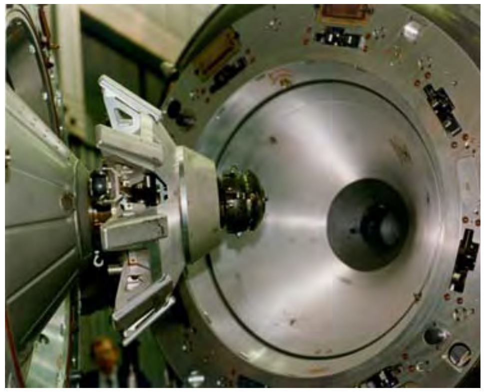

Figure 20.13.4.1-2: Russian Probe Drogue

Both docking systems are mechanically complex and contain single-point failures and zero-fault-tolerant functions. In combination with required dynamic operational aspects, this creates critical operations with lower reliability and very limited flexibility. Additionally, the use of these systems drives other design and operational aspects for vehicles; e.g., a complicated contact thrusting (additional jet thruster firing at the moment of mating system contact) is used on the space shuttle to ensure alignment and capture because of the complex mechanism and unreliable contact sensors in APAS design. Contact thrusting is used even with the modified version of the shuttle APAS, a version modified to the limits possible to reduce the capture forces required. The operation is training intensive and requires the crew to monitor a centerline camera and look out the hatch window for timing cues.

These docking systems support only small cargo or crew transfer because of the 32-in.-diameter hatch; they do not provide a fluid umbilical and have only a limited electrical power and data umbilical. Such limitations preclude the use of these docking systems for a U.S. autonomous rendezvous capability.

<u>CBM</u>

The U.S. CBM (see Figure 20.13.4.1-3)—designed for use on the International Space Station (ISS) for the large-diameter module-to-module attachment—is a berthing mechanism and was not designed to accommodate the larger misalignments, forces, and dynamics of docking. To use the CBM, a robotic arm places the two halves of the CBM interface, one active and one passive, within a capture envelope. Small latches (arms) reach around and capture the passive interface, prevent it from leaving the petal envelope, and when retracted force alignment of the guide petals. When fully mated, powered bolts are engaged to provide seal preload and to support the mated dynamics. After pressurization the hatch is opened, and the CBM electronics are manually removed from the CBM passageway. The large square opening supports large cargo logistics transfer.

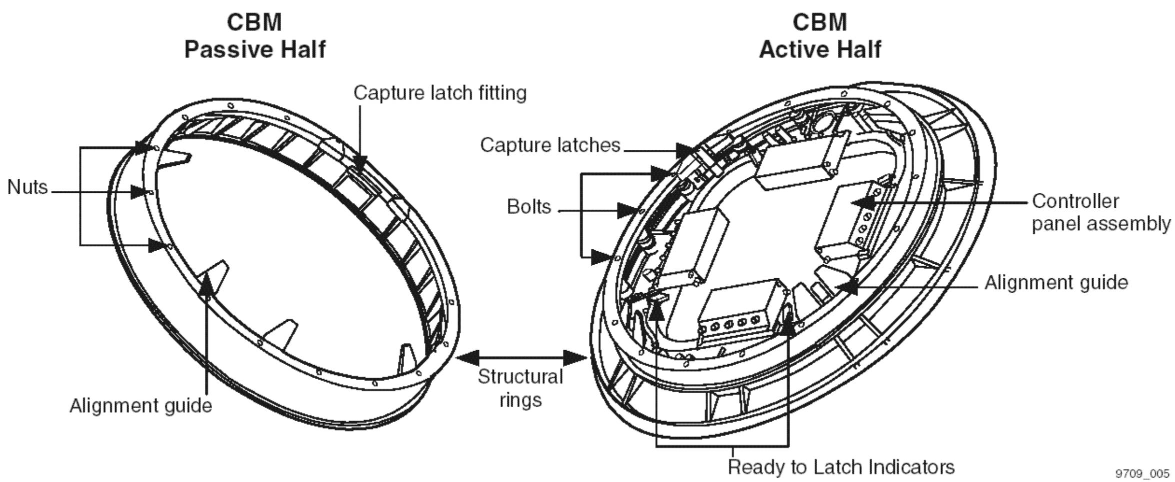

Figure 20.13.4.1-3: U.S. CBM

None of the three existing systems meet dual-fault tolerance requirements for critical operations; e.g., time-critical release, which is very important for an emergency or expedited separation. While both docking mechanisms provide nominal hook release and a pyrotechnic backup, the Space Shuttle Program accepts the use of a 96-bolt APAS release via a 4-hour extravehicular activity to satisfy dual-fault tolerance requirements. CBM powered bolts do not operate fast enough to support expedited release because of the threaded bolt and nut design, and they are operated in groups of four to prevent binding and galling during unthreading. The CBM uses a pyrotechnic to provide 1 fault tolerance for release. All three systems contain uniquely passive and active (male and female) interfaces that limit mission mating flexibility, and each has a specific operational range or performance for use. These facts indicate that the development of a modular, generic mating infrastructure is one of the key elements needed for the success of future NASA missions and programs.

20.13.4.2 New Technology

Since the early 1990's, NASA has had a team continuously working on the development of a modern, next generation mating system designed to simplify operations and reduce the risks associated with mating spacecraft. This effort, first called Smart Docking and later the Low Impact Docking System (LIDS), has focused on developing and testing a force-feedback controlled soft capture system built from modern electromechanical technology and the design of a modern hard capture system that provides reliable and redundant time critical release. This advanced docking mechanism is software re-configurable to support different mission applications but always with a low kinetic energy mating with very low contact forces. Since the beginning a major goal was to design a fully androgynous interface that is capable of mating to a copy of itself. This additional flexibility provides complete system level redundancy, which has never before been available to NASA and the U.S. Space effort. Additionally, the use of a software controlled active capture system provides a mechanism which can tuned to the unique performance requirements for all types of mating operations to support docking or berthing, autonomous or piloted rendezvous, and in-space assembly of vehicles, modules and structures.

The LIDS design team has worked to incorporate the lessons learned and experiences from the development and use of all previous and existing systems. The engineering organization developing LIDS for NASA has been involved in space mating systems development and use throughout U.S. space flight; e.g., Gemini, Apollo, Apollo-Soyuz and ISS assembly. Additional experience has been gained in support of the procurement, modification and re-certification of the Russian APAS for the Shuttle-Mir and Shuttle-ISS missions. More recent experience includes the development of LIDS for the X-38 crew return vehicle (CRV) program. Except for the Russian APAS, NASA personnel have always maintained a lead role in the development of human-transfer-rated mating mechanisms for its programs.

It has been established through previous prototype hardware development and testing that a fully androgynous mating interface built around low-impact characteristics is feasible and capable of meeting docking and berthing requirements. Dynamic simulation testing demonstrated the ability of a low impact type docking system to dock CRV-size vehicles (20,000 lbs) and also Orbiter-scale vehicles (200,000 lbs) moving at low velocities and resulting in contact forces less than 50 lbf. These earlier efforts resulted in a TRL-4 maturity of a low impact docking system. Benefits of this design include safer mating operations, more effective and flexible mission implementation, and greater system level redundancy, which can lead to a mission architecture built around a common mating system.

The primary goal of the effort described above is the continued development of an advanced mating system to TRL-6 maturity to be available future exploration missions.

Development objectives include:

- Elimination of the need for high velocity docking
- Robust and safe operation for deep space missions
- Androgynous, modular design that is re-configurable for multiple operations and applications

This goal and these objectives are best accomplished by starting the development based on the existing TRL-4 design of the Low Impact Docking System and evolving the system design to meet new Exploration Program requirements.

LIDS Design

The LIDS design is functionally analogous to the APAS with a central soft-capture ring but differs by using modern electromechanical systems with redundant mechanical, electronics, and software elements in lieu of the traditional mechanically interconnected gears, clutches, linkages, and electronic relays.

The LIDS soft-capture system incorporates an active load-sensing system to realign the soft-capture ring automatically rather than requiring force to realign it. As a result of this force-feedback control, the soft-capture system can be used as an active damper to absorb any residual energy or motion left from mating the two vehicles. A LIDS does not require the separate complex and heavy load attenuation system like those used in existing docking systems.

The LIDS design replaces traditional trip latches with electromagnets. When energized, an electromagnet requires only striker plate contact for retention and eliminates the need for forces to open and close latches. The elimination of the two force requirements is key to realizing all of the benefits offered by a low-impact docking system. This design alleviates the requirement to ram-mate vehicles together with large closing velocities. The change in docking technology from the APAS to the LIDS is analogous to the change in technology from linkage and cable-driven aerosurfaces in older aircraft to fly-by-wire technology in modern aircraft.

20.13.5 Future Mission Application and Development

It has been generally accepted that the development of a next generation mating system is an enabling technology for future space exploration programs. The recent X-38, SLI, and OSP programs each concluded that the development a new mating system was a priority. Each Program funded or was in the process of funding continued mating system development when it was cancelled.

While these programs were focused on development for LEO applications, the efforts described above to develop a new mating system included accounting for the functionality, reliability, and redundancy that would be needed for beyond Earth orbit applications. So it is credible to expect that any advanced mating system developed will be applicable to LEO and beyond since the mating requirements are nearly identical when applying an androgynous, low-impact, reconfigurable mating system to the array of mating operations.

The technology development path required to provide an advanced mating system for use in a human exploration program has been estimated several times in recent years. It is anticipated that it will take 3-4 years to reach TRL 6 and then around 2-3 years to complete the development to space flight hardware levels. An estimate for hardware development and phasing is included in Figure 20.13.5-1. This schedule is an estimate and the final schedule would depend upon the requirements, decisions and priorities levied by the Program.

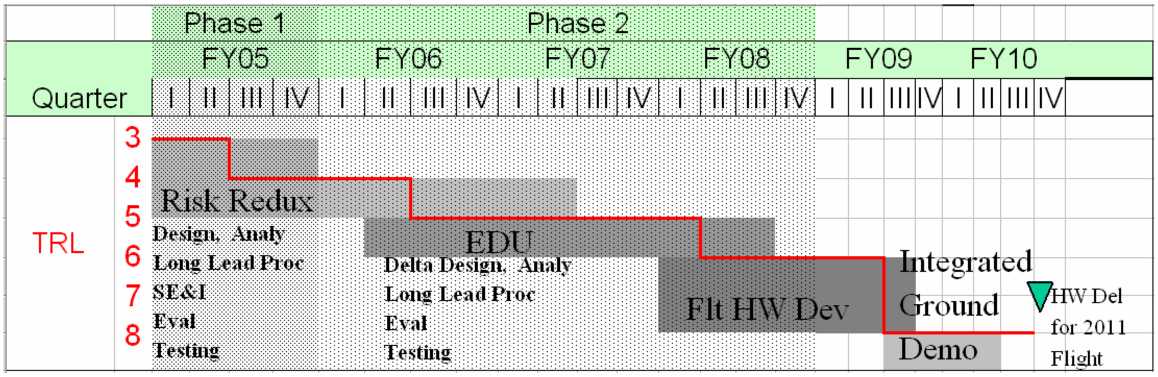

Figure 20.13.5-1: Advanced Mating System Technology Development

The timing of Phase I development would allow the designs to be available for selection and adaptation for use on other NASA missions. Extensive discussions with the Hubble Program have concluded that the low impact soft capture system could be used in conjunction with the existing Hubble capture mechanism and provide increased flexibility and chances for mission success. This opportunity, if pursued, would provide an early precursor for in-space assembly to validate future deep space operations and an opportunity for early on-orbit maturation for a major subassembly of the advanced mating system design. Any decision to apply the advanced mating system technology to the tentative 2007 Hubble mission would have to be made very early in FY05 to support long lead flight hardware procurement decisions and to allow time to develop an integrated schedule and resource plan that minimizes the affects to the content in this proposal. It is also worth noting that no impact would occur during early phases since the soft capture system risk reduction development would directly support the Hubble mission as currently planned due to the flexible, software reconfigurable nature of the soft capture system design in meeting Hubble mating requirements.

After completion of the earlier integrated risk reduction development and testing phase in FY07, the operational benefits from the application of a low impact mating system will be readily apparent. At this point in time, approximately midway through the engineering development phase, decision gates for the final mating system flight hardware development will likely be chosen that dictate the implementation path for the Constellation exploration program. It is possible in the period following Phase II that mating system flight hardware be developed in support of potential flight demonstration(s) adding early space flight experience and maturation that may be required before the actual usage by crewed missions.

Additionally, it is anticipated that the advanced mating system work and maturation contained in this proposal would provide new options for solving the post-Shuttle Program, ISS logistics problems by supporting a US-based, ISS autonomous rendezvous capability. The recognition of this may ultimately influence the actual implementation of the new mating system into the U.S. Space Program.

20.14 Thermal Environment for a Lunar Mission

Steve Rickman, NASA/JSC/ES – Structural Engineering Division

20.14.1 Introduction

Spacecraft that are part of a manned lunar mission will experience a variety of thermal environments. Each phase of the mission presents various thermal control system (TCS) design challenges.

This section discusses the environments expected and identifies some of the more challenging aspects of TCS design.

20.14.2 Earth Orbit And Near-Earth Environment

Scenarios for a human lunar mission include prolonged loiter and assembly of two or more elements in low-Earth orbit (LEO). Spacecraft and components experience environmental heating from direct solar flux, reflected solar (albedo) flux, and planetary infrared flux.

Natural environmental constants have been derived for numerous programs. These parameters experience variation due to Earth's distance from the sun, local and seasonal variations in the Earth's reflectance, and cloud cover. Hence, the solar, albedo, and planetary infrared heating components exhibit variation. The proposed natural thermal environmental parameters are presented in Tables 20.14.2-1 and 20.14.2-2 and were derived from International Space Station natural thermal environmental parameters in SSP 30425, Rev. B.

Additionally, the orbital parameters directly affect the spacecraft thermal environment. The orbit inclination and right ascension of the ascending node along with the solar declination affect the orbit beta angle -- the angle between the solar vector and its projection onto the orbit plane. For an orbit inclination between 0 and 90 degrees, the beta angle extremes experienced by an spacecraft are bounded by the orbit inclination plus the obliquity of the ecliptic plane, currently 23.45 degrees. A low, circular orbit inclined 28.5 degrees with respect to the equator (attainable from a launch due east from the Cape) will result in an orbit bounded by +/-52 degrees. The orbit altitude (in conjunction with the inclination) determines the rate at which the beta angle changes. Beta angle and altitude have a number of effects on the on-orbit thermal environment. First, they affect the fraction of the orbit spent in sunlight. In LEO, at say, 220 nm altitude, a spacecraft in a beta = 0 degree orbit will spend approximately 65 percent of the orbit in sunlight. The percentage of time spent in sunlight does not change considerably for betas up to about 30 degrees but increases to about 75 percent at beta = 52 degrees. The variation is presented in Figure 20.14.2-1.

Condition[2]	Orbit Time					
	0 to 0.25 hr		0.25 to 0.4 hr		After 0.4 hr	
	Albedo	OLR (Btu/hr ft^2)	Albedo	OLR (Btu/hr ft^2)	Albedo	OLR (Btu/hr ft^2)
Cold A	(3)	65.6	(3)	56.1	0.27	68.7
Cold B	(3)	65.6	(3)	56.1	0.22	76.4
Mean					0.27	76.4
Hot A	0.21	91.0	0.20	97.3	0.27	86.5
Hot B	0.36	76.4	0.40	76.4	0.35	76.4
Solar Constant (Btu/hr ft^2)						
Cold	418.7					
Mean	434.5					
Hot	451.0					

Notes:

1. Values in this table are expected to be exceeded no more than 0.5% of the time.
   Albedo and OLR are adjusted to the top of the atmosphere at 18.65 miles (30 km) altitude.
2. Both Set A and Set B are design requirements.
3. No Albedo value, extreme cold case occurs in eclipse.

Table 20.14.2-1: Hot and Cold Natural Thermal Environments[1]

Condition[2]	Orbit Time					
	0 to 0.25 hr		0.25 to 0.4 hr		After 0.4 hr	
	Albedo	OLR (Btu/hr ft^2)	Albedo	OLR (Btu/hr ft^2)	Albedo	OLR (Btu/hr ft^2)
Cold A	(3)	60.5	(3)	48.5	0.27	65.3
Cold B	(3)	60.5	(3)	48.5	0.20	76.4
Hot A	0.25	102.5	0.25	110.6	0.30	90.7
Hot B	0.45	76.4	0.25	76.4	0.40	76.4

Notes:

1. Values in this table are expected to occur no more than 0.05% of the time.
   Albedo and OLR are adjusted to the top of the atmosphere at 18.65 miles (30 km) altitude.
2. Both Set A and Set B are design requirements.
3. No Albedo value, extreme cold case occurs in eclipse.

Table 20.14.2-2: Extreme Hot and Cold Natural Thermal Environments[1]

For a constant local-vertical, local-horizontal attitude, the variations in beta will cause a variation to the environment experienced by various sides of the spacecraft. For example, a port-facing surface, orbiting at beta = 52 degrees at an altitude of 220 nm (408 km) with surface optical properties of $\alpha/\varepsilon = 1/1$ will experience an orbital average absorbed heating flux approximately ten times that experienced by the same surface at beta = -52 degrees. Beta will, in turn vary as a function of orbit inclination and altitude. A significant change in the orbital environment can occur over the span of days -- an orbiting spacecraft must be designed to withstand the expected range of orbital environments.

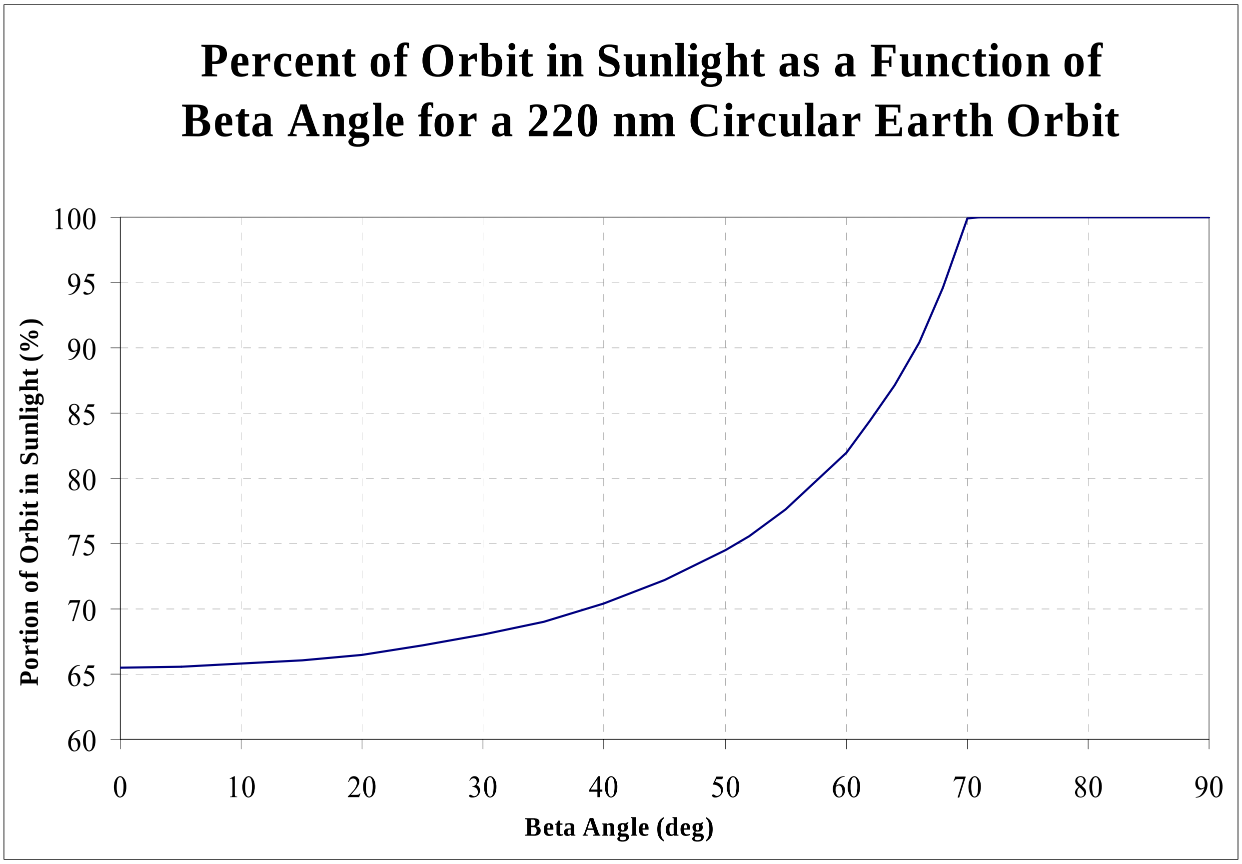

Figure 20.14.2-1: Portion of Orbit in Sunlight as a Function of Beta Angle

20.14.3 Cis-Lunar Environment

At distances that are sufficiently far from, both, Earth and the Moon, the cis-lunar environment is dominated by incoming solar flux. The Earth-Moon distance from the sun dictates the magnitude of the incoming solar flux. Earth reaches perihelion resulting in the maximum solar constant in early January of each year while aphelion is reached six month later in early July, resulting in the minimum value.

Spacecraft in this environment experience solar illumination on one side while a zero flux environment exists on the other. During the Apollo missions, spreading the heat evenly was accomplished by placing the Apollo Command-Service Module in a passive thermal control rotation (four revolutions per hour). In such a mode, the incoming solar flux is scaled by $1/\pi$ times its magnitude -- or approximately one third. This scheme not only reduces the effective magnitude of the incoming flux, it also provides a benign thermal environment for the spacecraft.

20.14.4 Lunar Orbit and Near-Lunar Environment

The environment in low lunar orbit is driven by the incoming solar flux, the lunar surface reflectance (albedo), and the infrared flux emanating from the lunar surface.

While the solar constant experiences the same variation as is specified for Earth-orbiting spacecraft, the lunar albedo is considerably lower than Earth's -- on the order of 7 percent.

Because the Moon is tidally locked to Earth, its rotation rate matches its orbit period. The system's motion about the sun along with the Moon's orbit period, gives rise to an average synodic period of 29.531 days. Unlike Earth, the Moon does not have an atmosphere capable of globally moderating temperature extremes. These factors give rise to extreme variations in the planetary infrared emission experienced by an orbiting spacecraft. Daytime infrared flux values occurring near lunar noon in the equatorial region approach one-sun equivalent flux. On the dark side, flux values vary as the temperature drops throughout the long lunar night. At temperatures close to –250 deg F, the infrared flux emitted is only about 3 Btu/hr ft^2.

20.14.5 Lunar Orbit and Near-Lunar Environment

The lunar surface environment is driven by the incoming solar flux, the surface reflectance (albedo), the surface emissivity, and the specific heat and thermal conductivity of the lunar regolith. Direct solar radiation rains down on the surface. Approximately 93 percent is absorbed and raises the local ground temperature. As the surface temperature increases, the infrared flux radiated goes up. The peak temperature reached is governed by the soil's α/ε ratio, the sun angle, and the diffusivity of the soil. The lunar surface is highly insulative with low thermal diffusivity. The α/ε ratio is very nearly unity. This results in daytime peak temperatures close to 250 degrees F providing some moderation in the temperature drop on the night side. The diurnal surface temperature variation is a function of the location on the surface. The peak temperature at a given latitude will be a function of the solar declination and the latitude. A simplified thermal math model of the lunar surface was developed to show general temperature trends. The results of this simplified analysis are presented in Figure 20.14.5-1.

Additional analysis will be required to determine environments induced by the presence of local terrain features such as hills, valleys, and craters. A lander situated in a crater will have a greater view factor to the lunar surface (in the form of crater walls). This not only reduces the view factor to space (a radiative heat sink), it also increases the infrared heat load on the lander and provides a surface to reflect additional sunlight onto the lander surfaces. Hills can not only increase the view to infrared sources and provide a solar reflection surface, they can also cold-bias the environment by inhibiting direct solar energy from reaching the lander during periods after sun rise and prior to sun set. At extreme latitudes (i.e., near the poles), the presence of these features may have a significant affect on the local surface illumination. Since these features can have a significant effect the thermal environment and, hence, the overall thermal design, it is recommended that local terrain features as they pertain to lander requirements be defined early in the program.

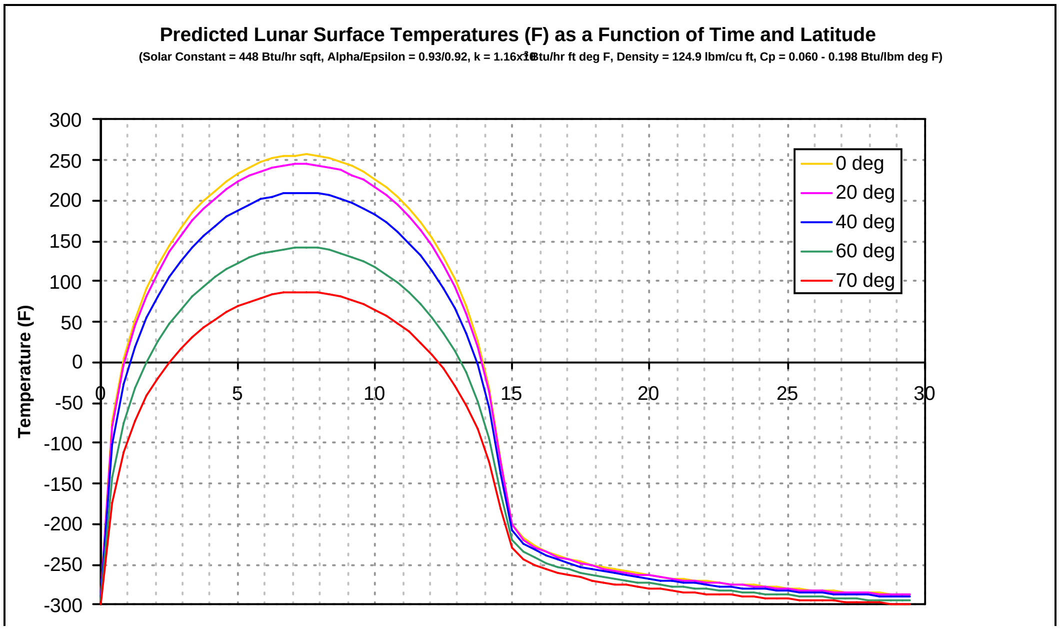

Figure 20.14.5-1: Predicted Diurnal Lunar Surface Temperatures as a Function of Latitude

20.15 Space Radiation Protection

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20.15.1 Radiation Risks

The Bioastronautics Critical Path Roadmap outlines the risks of concern for radiation missions. These include mortality or morbidity risks from:

1. Carcinogenesis,
2. Central nervous system damage,
3. Late degenerative risks including cardiac risks and cataracts
4. Acute radiation syndromes
5. Hereditary and Fertility Risks

Figure 20.15.1-1 shows a schematic diagram of the Radiation Risk and their concerns.

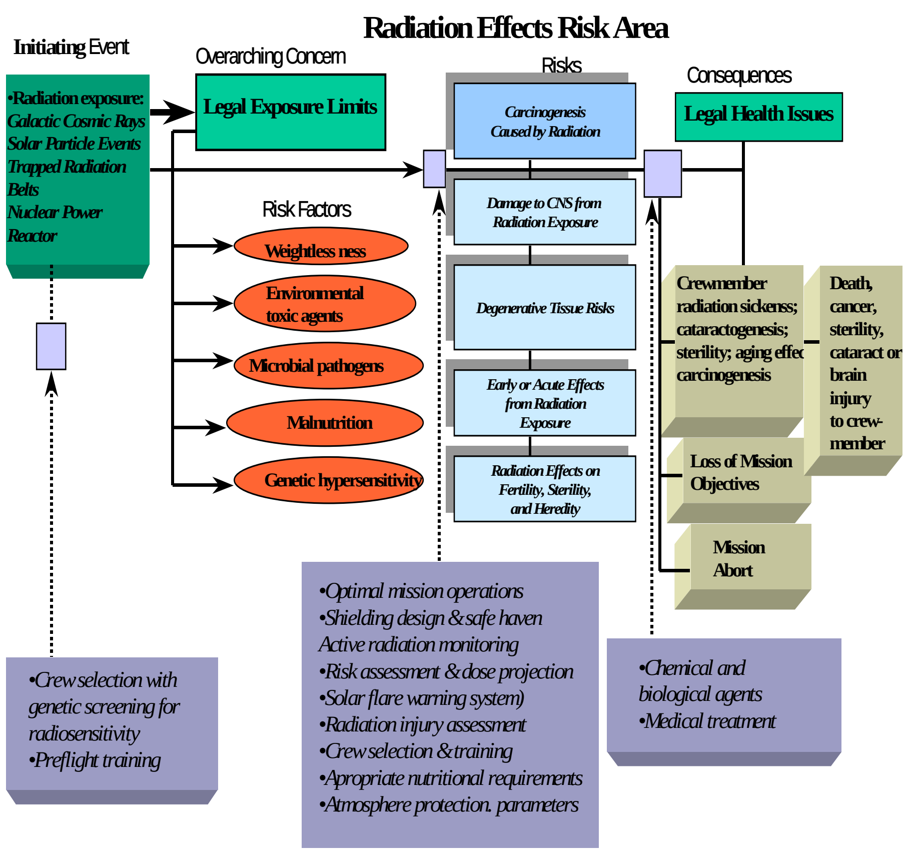


Figure 20.15.1-1: Schematic of Radiation Critical Path Roadmap concerns

A 5x5 matrix for radiation risks can be designed based on the following scoring criteria:

"(1) Expected Occurrence of Risk

Score the expected occurrence of the risk under the two following conditions: (a) when no countermeasures or risk mitigation strategies are present; and (b) when current countermeasures or risk mitigation strategies are applied

(a) no countermeasures or risk mitigation strategies present

1= 0.001-0.01 %

2= 0.01-0.1%

3= 0.1-1%

4= 1-10%

5= 10-100%

(b) current countermeasures are applied

1= 0.001-0.01 %

2= 0.01-0.1%

3= 0.1-1%

4= 1-10%

5= 10-100%

(2) Expected Impact on Crew: If the risk were to occur what would its expected impact be on crew health and safety?

Score the expected impact to the crew according to a 1 - 5 scale, where "1"= no impact or impairment, and "5" = significant, irreversible, long-term impairment, or death.

1 = *no impact to crew*

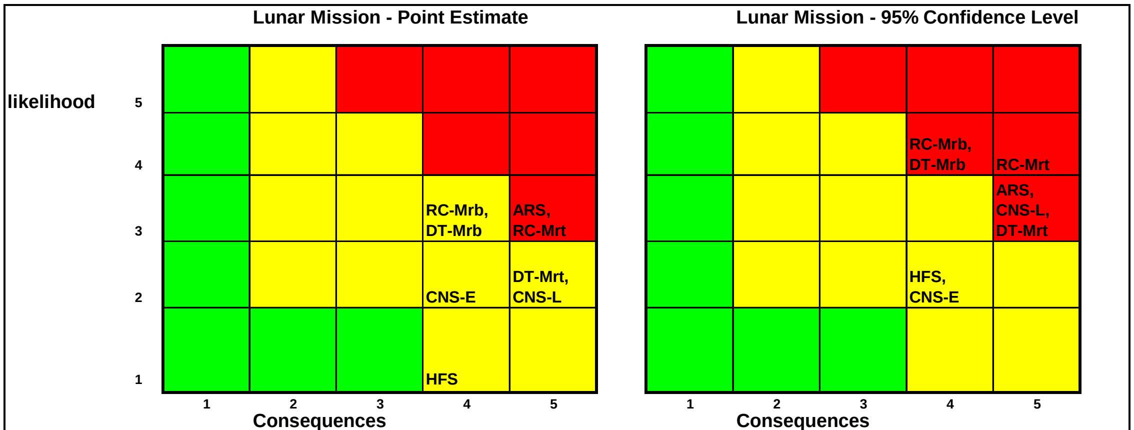
2 = *short-term, minor injury, illness, incapacitation, or impairment to crewmember*

3 = *serious injury, illness, incapacitation or impairment but not long term*

4 = *significant and long term impairment, but not permanent*

5 = *irreversible, catastrophic impairment, or death*

A Draft 5x5 matrix for Lunar and Mars missions was estimated as shown in Figure 20.15.1-2a-b. Because the current ability to predict the level of radiation risk is highly uncertain [1-3] due to the lack of radiobiology knowledge for heavy ions in space, the 5x5 matrix is also presented at the upper level of a 95% Confidence Interval. Of note is that there are several radiation risks of importance- not just carcinogenesis. Very distinct concerns are apparent for Lunar and Mars missions. For short-duration Lunar Missions (<60 days), risk of late effects are small, except in the circumstances of a large solar particle event (SPE). Protection from solar particle events, especially during EVA's, are the highest concern for lunar missions. For Mars explorations a host of late mortality and morbidity risks dominate the concerns, however SPE risks are also of importance. The lack of epidemiology data and the distinct modes of biological injury make risk projection for heavy ions in space highly uncertain, however the mission risks are just as likely to be marginal as excessive – because of the uncertainties it is not known.



Radiation CPR Risks	Abbreviation
1) Radiation Carcinogenesis	RC
Morbidity	RC-Mrb
Mortality	RC-Mrt
2) Degenerative Tissue Damage	DT
Morbidity	DT-Mrb
Mortality	DT-Mrt
3) Central Nervous System	CNS
CNS-early	CNS-E
CNS-late	CNS-L
4) Acute Radiation Syndromes	ARS
5) Hereditary, Fertility, Sterility	HFS

Figure 20.15.1-2a: 5x5 Matrix for Radiation Risks for Lunar Missions

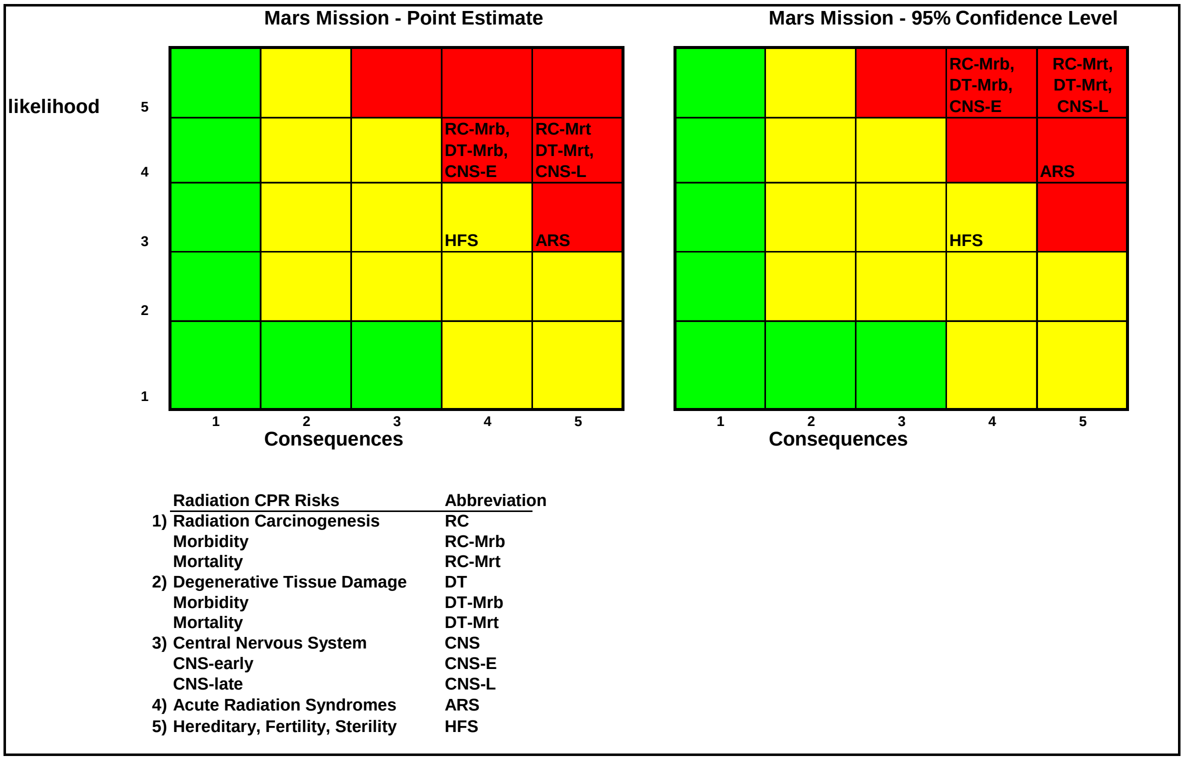


Figure 20.15.1-2b: 5x5 Matrix for Radiation Risks for Mars Missions

20.15.2 Radiation Protection: Dose Limits and ALARA

Radiation protection requirements are based on the following principles:

1. Risk Justification
2. Risk Limitation
3. ALARA

Risk justification requires ethical considerations on the benefits to society from the risk. For lunar or Mars missions the implementation of each of these Principles must be re-considered relative to practices for workers on ISS or on the ground. Risk justification for low-Earth Orbit missions, is based on a maximum increased risk of 3% cancer mortality, and avoidance of any clinically significant deterministic risks. These levels are similar to those for ground-based radiation worker's, however dose limits corresponding to these risks are adjusted for the distinct time-age factors in exposures. For exploration missions higher risk levels could be justified (NAS, 1970).

The current trade study has identified the need for Preliminary Dose Limits to be provided in support of Lunar Design Studies. The JSC Radiation Health Officer (RHO) made this recommendation to the NASA Medical Policy Board (MPB) on April 30, 2004, and it is likely that a Tiger team will be formed to issue Draft requirements by the end of Fiscal year 2004. Concurrently, NASA will task the National Council on Radiation Protection and Measurements (NCRP) to provide recommendations of lunar dose limits. This report would take 2-3 years to be published. Figure 20.15.2-1 shows a schematic for the spiral approach to lunar and Mars dose limits that is based on new knowledge of the radiobiological effects of space radiation becoming available as NASA's Space Radiation Health Research Program. This program has begun to accelerate the acquirement of new knowledge since the Brookhaven National Laboratory NASA Space Radiation Laboratory (NSRL) opened on July 7, 2003.

For lunar missions, we recommend dose limits similar to those used on ISS, however augmented with a probabilistic assessment for the 30-day limits (Appendix A). For example, the 30-day limit for blood forming organ doses of 25 cGy-Eq. would have to be guaranteed with a >99% probability. This will require a SPE design model to be developed and approved by an expert committee. We also recommend a stronger requirement of 10 cGy-Eq. to be satisfied at a >95% probability. This is reasonable achievable and would provide protection against the possibility of two large SPE's during a single mission. Other requirements for EVA protection will need to be addressed. ***A dedicated cost-benefit analysis program is needed for lunar mission shielding design***. The need for use of lunar regolith shielding is expected to be minimal if poly and other shielding materials are used in habitat design. ***A standard SPE design model is needed that will consider the frequency, size, energy spectra, and duration characteristics of such events in a probabilistic manner***.

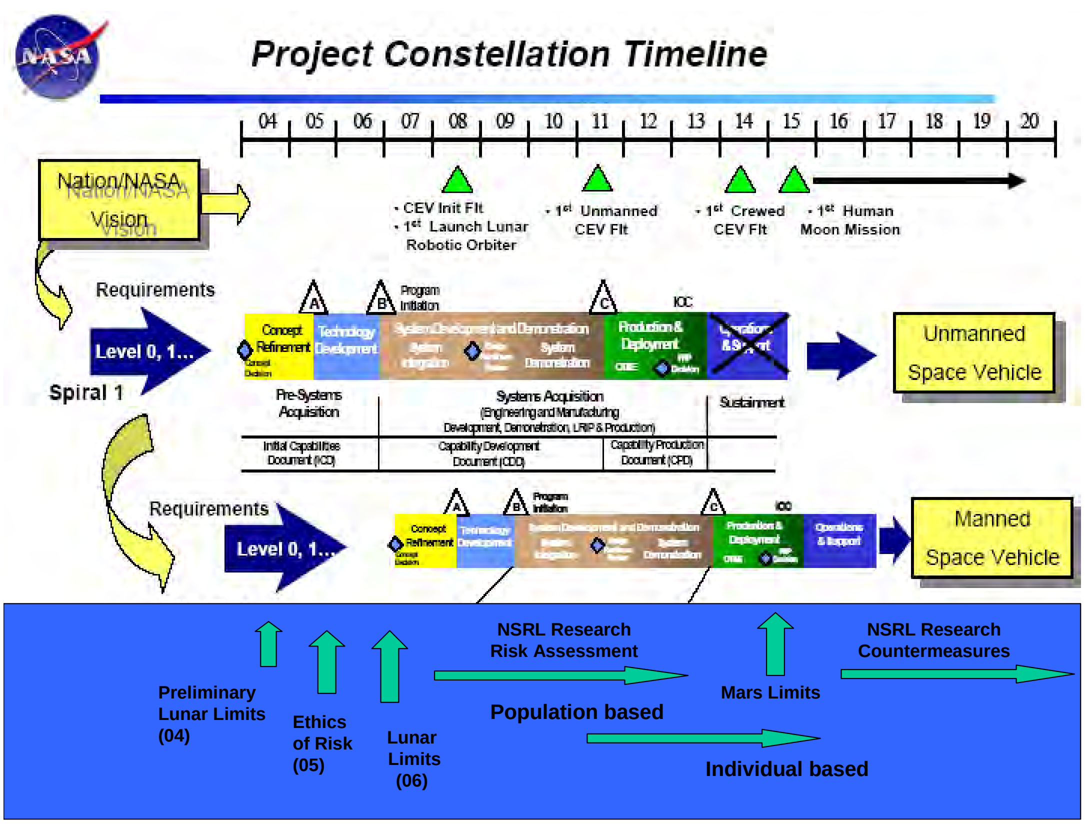


Figure 20.15.2-1: Spiral approach to lunar-Mars dose limits

20.15.3 Shielding Requirements

After the release of Preliminary Lunar Dose Limits approved by the NASA Medical Policy Board, shielding designs for the CEV, LTV, and any lunar habitat can begin in earnest. Doses from large SPE's can cause acute radiation sickness, or appreciable risk of late effects if adequate shielding is not provided. Figure 20.15.3-1 shows example dose-rates and cumulative doses for the recent October, 2003 event.

It is already established that materials with lower atomic-weight constituents, especially hydrogen, provide the most protection against high-energy nuclei due to [5,6]:

1. Optimal slowing down of ions per unit mass.
2. Optimal projectile fragmentation
3. Reduced target fragment production including neutrons

To support the spiral approach to exploration, new material developments including testing will have to be supported in the near term. Aluminum-based structures may be sufficient for lunar missions, however there use could prevent any spiraling of technologies for Mars missions to occur. At the minimum a storm-shelter made of aluminum should be incorporated into the CEV, LTV, etc. Figures 20.15.3-2 show the value of polyethylene shielding within aluminum structures for dose reduction for the August of 1972 SPE, one of the largest ever observed. Each SPE will have distinct temporal and spectral characteristics.

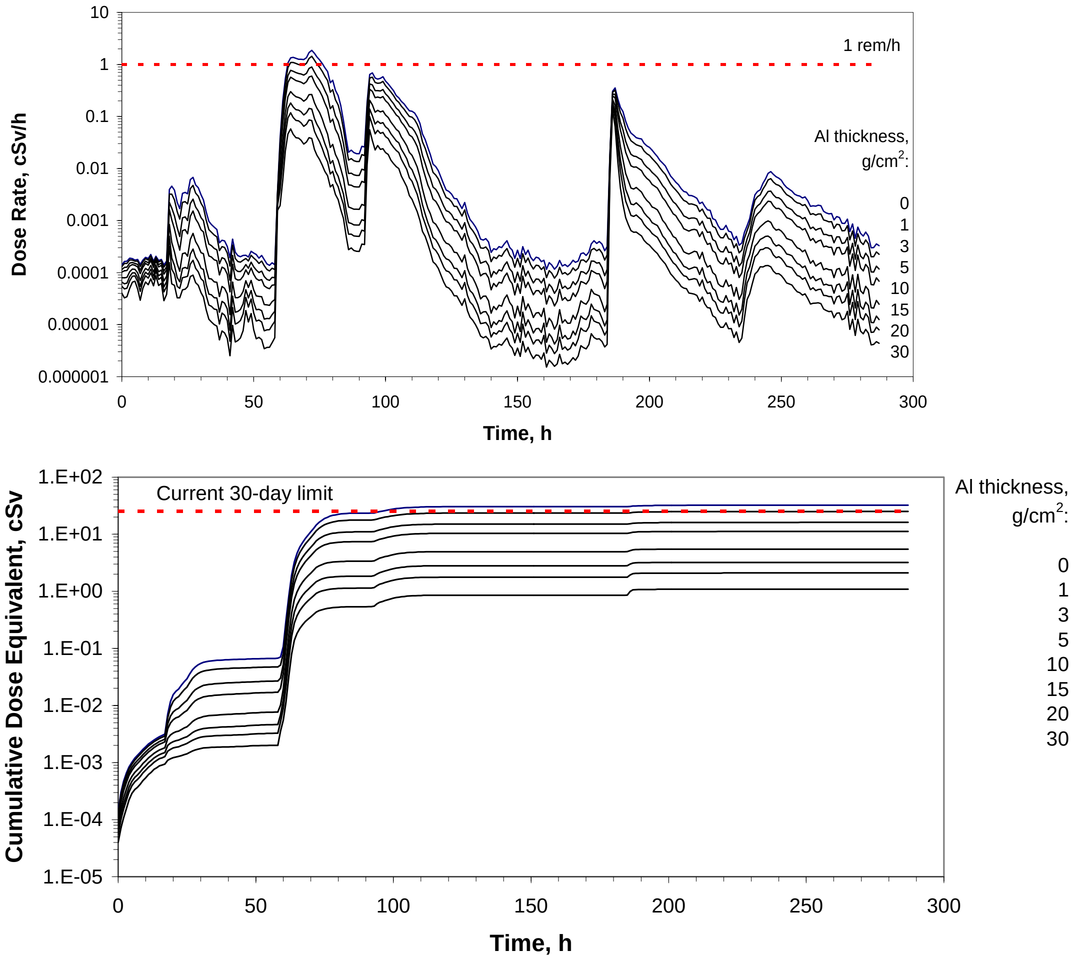


Figure 20.15.3-1: Dose-rates and doses to the blood forming organs (BFO) from the Halloween event of 2003 calculated with NASA BRYNTRN code

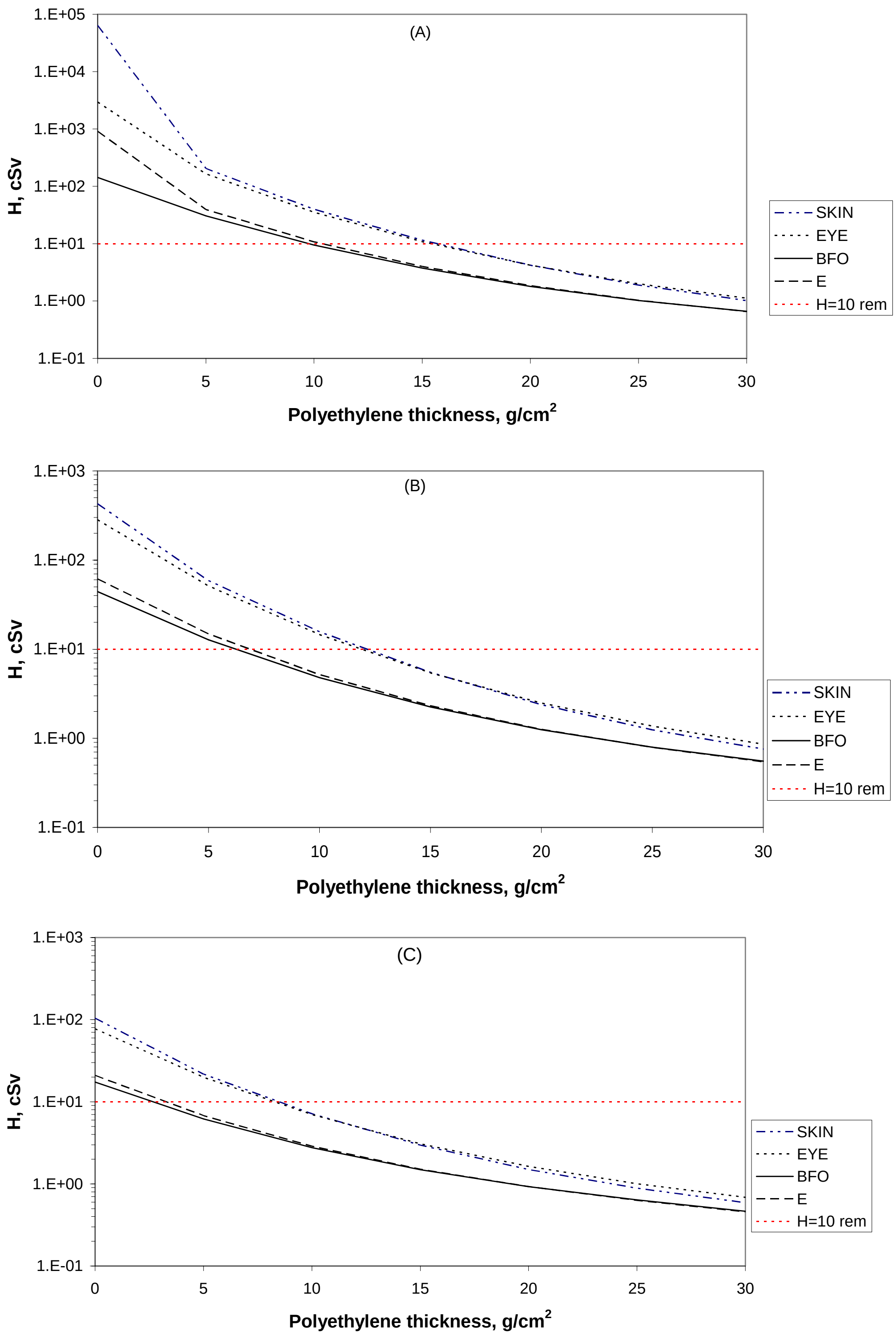


Figure 20.15.3-2: Dose equivalent at sensitive sites from August 1972 SPE with various polyethylene shielding inside (A) 0, (B) 5, and (C) 10 g/cm2 of aluminum

The number of events per year can be estimated as shown in Figure 20.15.3-3, however a prediction on the actual size or date of occurrence can not be made at this time. Observations of active regions of the sun are useful information, but do not provide an accurate predictive capability.

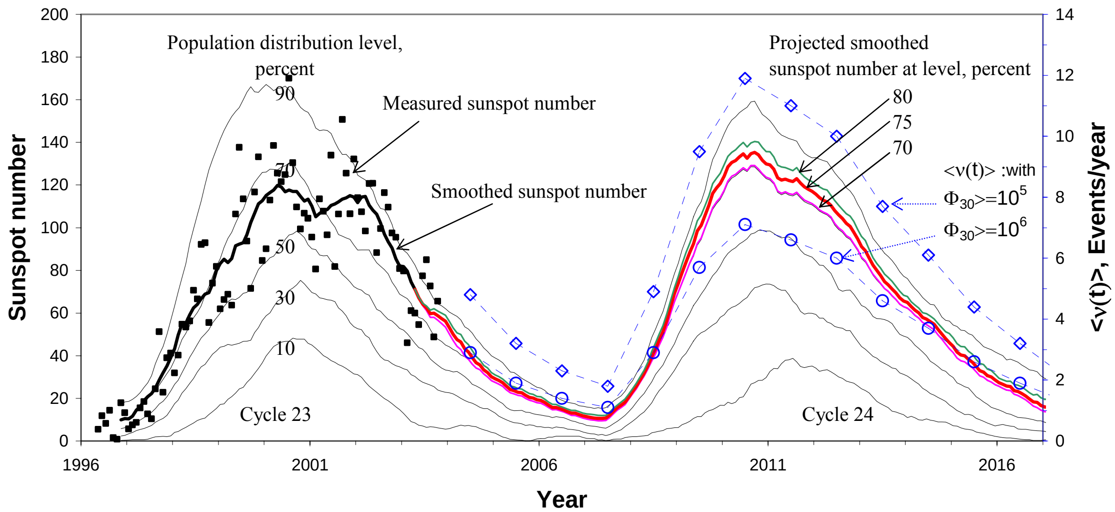


Figure 20.15.3-3: Sunspot sampling distribution and projections of solar cycles and mean occurrence frequency of SPE

Since typical spacecraft have a minimum of 3-4 g/cm^2 aluminum shielding and a maximum of 10-15 g/cm^2 aluminum equivalent shielding storm shelters of 5-10 g/cm^2 polyethylene may be sufficient. New materials with higher hydrogen content would provide more protection at less mass. Retrofit designs may fail because of volume constraints, and a storm-shelter should be designed from the early states of mission design. The totality of radiation sources for the mission will need to be considered (GCR, SPE, Van allen belts, Nuclear sources, etc.)

MONITORING AND DOSIMETRY

Draft Monitoring requirements adapted from the ISS requirements are attached as Appendix B. More attention to early forecasting of SPE's is needed compared to ISS. An assessment of future possibilities for SPE forecasting for the 2015-2020 time periods is needed.

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GLOSSARY

BCPR	**=**	**Bioastronautics Critical Path Roadmap**
BP (UB)	**=**	**Bioastronautics Program**
CEV	**=**	**Crew Exchange Vehicle**
CNS	**=**	**Central Nervous System**
CQ	**=**	**Critical Question**
GCR	**=**	**Galactic Cosmic Ray**
HZE	**=**	**High Charge and Energy Ion**
IOM	**=**	**Institute of Medicine**
LET	**=**	**Linear Energy Transfer**
LTV	**=**	**Lunar Transfer Vehicle**
MPB	**=**	**Medical Policy Board**
NCRP	**=**	**National Council on Radiation Protection and Measurements**
NRC	**=**	**National Research Council**
OBPR	**=**	**Office of Biological and Physical Research**
RHO	**=**	**Radiation Health Officer**
SPE	**=**	**Solar Particle Event**
SRI	**=**	**Space Radiation Initiative**
SRHP	**=**	**Space Radiation Health Project**

APPENDIX A: Proposed Draft Astronaut Supplementary Radiation Standards for Lunar Missions

I. Executive Summary

Career Risk Limits: Career exposure to radiation is limited to not exceed 3% increased fatal cancer risk from occupational radiation exposures. NASA will assure that this risk limit is not exceeded at a 95% confidence level using a statistical assessment of the uncertainties in the risk projection calculations to limit the cumulative effective dose (in units of Sievert) received by an astronaut throughout his or her career. For solar particle event protection, statistical probabilities for mission design are levied.

Career Radiation Limits: These are determined using estimates made by the National Council on Radiation Protection and Measurements (NCRP) of age and gender dependent radiation dose that corresponds to the Career Radiation Limit. Table I lists examples of Career radiation limits for 1-year missions. Dose limits for other career lengths are evaluated as described below.

Table-I. Example career effective dose limits recommended by NCRP (2000).

	Effective Dose for 3% increase fatal risk (mSv)	
Age, yr	Male	Female
25	600	350
35	850	500
45	1150	700
55	2000	1200

Dose Limits for Deterministic Effects: Short-term dose limits (30-day and 1-year) for the skin, eye, and blood forming organs (BFO), and career limits for the skin and eye are imposed to prevent the occurrence of clinically significant deterministic health effects. For mission design these requirements must be met at a <99% probability against the NASA Standard Solar Particle Event Design Model.

The Principle of As Low as Reasonably Achievable (ALARA): The ALARA principle is a legal requirement intended to ensure astronaut safety. An important function of ALARA is to ensure that astronauts do not approach dose limits and that such limits are not considered as "tolerance values". ALARA is especially important for space missions in view of the large uncertainties in cancer and other risk projection models. Manned-mission programs and terrestrial occupational procedures resulting in radiation exposures to astronauts are required to find cost-effective approaches to implement ALARA.

II. Method of Calculation- Career Limits

Radiation Doses and Risk Limits: Cancer risk is not measured directly, but is calculated using radiation dosimetry methods. The **absorbed dose D** (in units of Gray) is calculated using measurements of radiation levels provided by **dosimeters** (e.g., film badges, thermoluminescent dosimeters (TLDs), spectrometers such as the TEPC, area radiation monitors, biodosimetry or biological markers) and corrections for instrument limitations. The limiting risk is calculated using the **Effective Dose, *E***, (in units of mSv) and risk conversion factors provided by the NCRP in Report 132, relating the limiting cancer fatality risk to *E*.

For the purpose of determining radiation exposure limits at NASA, the probability of fatal cancer is calculated as follows:

1. The body is divided into a set of sensitive tissues, and each tissue *T* is assigned a weight w_T according to its estimated contribution to cancer risk, as shown in **Table-II**.
2. The absorbed dose, D_T, delivered to each tissue is determined from measured dosimetry. Different types of radiation have different biological effectiveness, dependent on the ionization density left behind locally (e.g., in a cell or a cell nucleus) by their passage through matter. For the purpose of estimating radiation risk to an organ, the quantity characterizing this ionization density is the **Linear Energy Transfer (LET)** (in units of keV/µm).
3. For a given interval of LET, between L and ΔL, the **dose equivalent** risk (units of Sievert, where 1 Sv = 100 rem) to a tissue *T*, $H_T(L)$ is calculated as:

   $$H_T(L) = Q(L)D_T(L), \quad (1)$$

   where the **quality factor**, *Q(L),* is obtained according to the ICRP prescription shown in **Table-III** This way of calculating $H_T(L)$ differs from the method used by ICRP, where a tabulated set of weighting factors is given instead of the quality factor. The method used here is considered to yield a better approximation by using the quality factor as the weight most representative of cancer risk, while the ICRP method may over-estimate the risk, especially for high-energy protons. Neutron contributions are evaluated by their contribution to $D_T(L)$.

Table-II Tissue Weighting Factors

Tissue or Organ	*Tissue Weighting Factor, w_T*
Gonads	0.20
Bone Marrow (red)	0.12
Colon	0.12
Lung	0.12
Stomach	0.12
Bladder	0.05
Breast	0.05
Liver	0.05
Esophagus	0.05
Thyroid	0.05
Skin	0.01
Bone Surface	0.01
Remainder*	0.05

*For purpose of calculation, the remainder is composed of the following additional tissues and organs: adrenals, brain, upper intestine, small intestine, kidney, muscle, pancreas, spleen, thymus, and uterus.

4. The average risk to a tissue T, due to all types of radiation contributing to the dose, is given by:

$$H_T = \int D_T(L)Q(L)dL, \qquad (2)$$

or, since $D_T(L) = LF_T(L)$, where $F_T(L)$ is the **fluence** of particles with LET=L, traversing the organ,

$$H_T = \int dLQ(L)F_T(L)L. \qquad (3)$$

Table III Quality Factor – LET relationship according (ICRP,1991)

Unrestricted LET, keV/μm in Water	*Q(LET)*
<10	1
10 to 100	0.32 LET – 2.2
>100	300/ Sqrt(LET)

5. The effective dose is used as a summation over radiation type and tissue using the tissue weighting factors, w_T,

$$E = \sum_T w_T H_T. \qquad (4)$$

6. Risk coefficients R_0*(age, gender)* per unit effective dose have been evaluated by the NCRP as excess absolute risk (EAR) per year (The NCRP report also provides separate accounting for solid tumors and leukemia, and computes life shortening due to radiation). The risk factors used by NCRP are shown in **Table-IV** for an effective dose of 100 mSv delivered in less than 1-yr.

7. For a mission of duration t, the effective dose will be a function of time, $E(t)$, and the effective dose for mission i will be:

$$E_i = \int E(t)dt \qquad (5)$$

and in applying the associated risk factor $R_0(age_i, gender)$, age_i is the average age during the mission.

Evaluation of Cumulative Radiation Risks: The cumulative cancer fatality risk to an astronaut for N, occupational radiation exposures, is found by summing over the tissue-weighted effective dose, E_i, as:

$$Risk = \sum_{i=1}^{N} E_i R_0(age_i, gender). \qquad (6)$$

The effective dose limits given in the **Table-I** are based on **Table-IV** and illustrate the effective dose that corresponds to a 3% increase in lifetime cancer fatality risk. **Table-I** assumes equal radiation doses per year occur over a time period beginning with the age at exposure listed in the left column, and lasting for 10

Table-IV Risk Coefficients corresponding to the (%)-Probability of Excess Fatal Cancer Risk normalized to an effective dose of 100 mSv*.

	Probability of Excess Fatal Cancer (%)	
Age at Exposure, Yr	Males	Females
25	0.502	0.860
35	0.361	0.610
45	0.258	0.430
55	0.147	0.249

*Assumes low dose-rate exposures in less than one-year; values for other ages found by interpolation.

years subsequently. Dose limits for other career lengths and radiation exposure patterns is evaluated using eq.(6) and the risk coefficients listed in **Table-IV**.

III. Method of Evaluation for Deterministic Dose Limits

Deterministic dose limits are intended to prevent the occurrence of clinically significant deterministic health effects from radiation. The method used for evaluating the equivalent dose of space radiation for deterministic effects uses the "Gy-Equivalent" to distinguish effective doses based on relative biological effectiveness factors (RBE) from deterministic effects from those based on Q-values to be used for late effects. **Table-V** shows the values for deterministic dose limits.

Table-V. Dose limits (in Gy-Eq.) for preventing deterministic radiation effects.

Organ	*30 day limit*	*1 Year Limit*	*Career*
Eye	1000 mGy-Eq.	2000 mGy-Eq.	6000 mGy-Eq
Skin	1500	3000	4000
BFO	250	500	Not applicable

Because RBE's for deterministic effects will depend on dose, the RBE used for specifying the Gy-Equivalent are the values determined at the threshold dose for deterministic effect being considered (risk to skin, eye, or BFO). NCRP recommendations for RBE values for deterministic effects are listed in **Table-VI** and are generally smaller than the Q-values.

Table-VI. NCRP Recommendations on RBE values for deterministic radiation effects[a].

Radiation Type	*Recommended RBE*[b]	*Range*
1 to 5 MeV neutrons	6.0	(4-8)
5 to 50 MeV neutrons	3.5	(2-5)
Heavy ions	2.5[c]	(1-4)
Proton > 2 MeV	1.5	-

[a]RBE values for late deterministic effects are higher than for early effects in some tissues and are influenced by the doses used to determine the RBE.

[b]There are not sufficient data on which to base RBE values for early or late effects by neutrons of energies <1 MeV or greater than about 25 MeV.

[c]There are few data for the tissue effects of ions with a Z>18 but the RBE values for iron ions (Z=26) are comparable to those of argon (Z=18). One possible exception is cataract of the lens of the eye because high RBE values for cataracts in mice have been reported.

IV. Confidence Level Evaluation For Career Limits

95% confidence levels for career cancer risks are evaluated using the methods specified by the NCRP in their Report No. 126 modified to account for the uncertainty in quality factors and space dosimetry. The uncertainties considered in the evaluation of the 95% confidence levels are:

1. The uncertainties in human epidemiology data including uncertainties in
   a. statistics limitations of epidemiology data
   b. dosimetry of exposed cohorts
   c. bias including misclassification of cancer deaths
   d. the transfer of risk across populations
2. The uncertainties in the dose- and dose-rate reduction (DDREF) factor used to scale acute radiation exposure data to low dose and dose-rate radiation exposures.
3. The uncertainties in the radiation quality factor (Q) as a function of LET.
4. The uncertainties in space dosimetry.

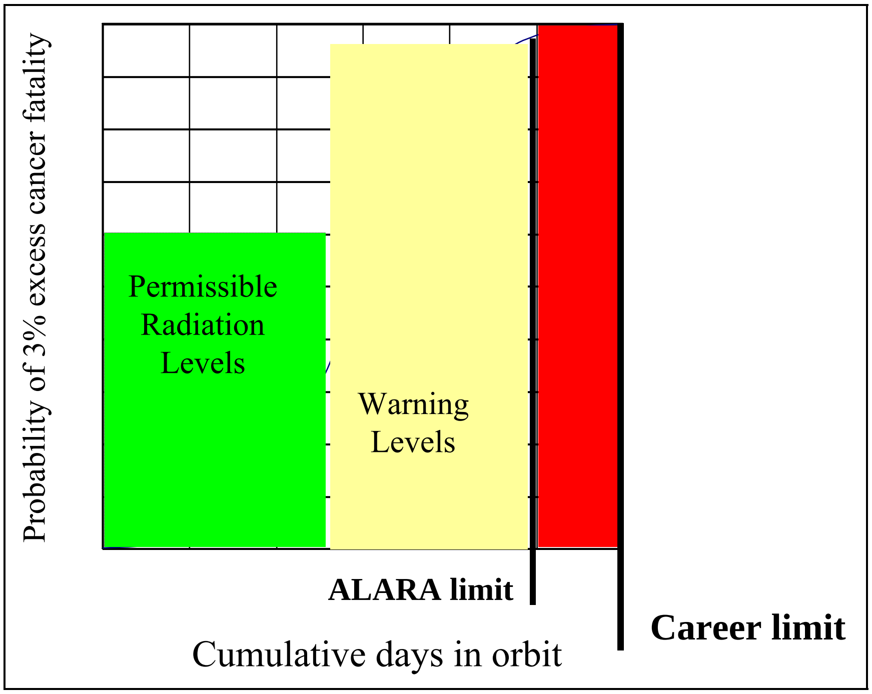

Figure A1: Schematic of method used to implement ALARA

The probability of reaching the limiting risk of 3% excess cancer fatality in any given mission is illustrated in **Fig. A1**.The statistical distribution for the estimated probability of fatal cancer is evaluated in order to project the most likely values and the lower and upper 95% confidence intervals (C.I) reported within brackets. For example, for the average adult exposed to 100 mSv (10 rem) of gamma-rays, the estimated cancer risk is 0.4 % and the 95% C.I.'s estimated by the NCRP are written as [0.11%, 0.82%] where 0.11% is the lower 95% level and 0.82% is the upper 95% confidence level. In order to assure that the career risk limit is not exceeded with a safety margin corresponding to a 95% confidence level, the upper confidence level (worse-case) is considered in the mission selection process.

V. Probabilistic Assessment for Solar Particle Event Protection.

Lunar mission designs must ensure with a 99% probability sufficient protection to stay below the 30-day and 1-year dose limits. This requirement must be tested against the NASA Standard Solar Particle Event Model using Radiation Transport Evaluation Tools approved by NASA RHO.

In addition a requirement to be below a 100 mGy-Eq. dose to the BFO with a probability > 95% must be met by lunar mission designs.

DRAFT- Lunar Radiation Monitoring Requirements

The ionizing radiation environment is monitored by passive and active (powered) instruments and evaluated in order to document crew exposures and to provide data for dose management. The following requirements ensure adequate environmental monitoring of crewmembers' exposure to cosmic rays and onboard-radiation exposure sources of various biological effectiveness. For Lunar EVA's the time and distance from shelter lead to definitions of required monitoring and dosimetry.

Area monitors are passive and active detectors placed throughout the vehicle to provide additional information about the temporal behavior, biological effectiveness ("radiation quality"), and inhomogeneity of the ambient radiation field. External radiation detection instruments are necessary to provide near real-time information about the dynamic radiation environment experienced by crewmembers during EVA and for verification of models used to evaluate exposures internal and external to vehicle.

Personal Dosimetry

Each crewmember must be provided with a personal radiation dosimeter for continuous use during a mission. The personal dosimeter serves as the dosimeter of record. The continuous use of the dosimeter of records fulfills a legal requirement for radiation workers when working in a radiation area (vehicle or lunar surface). When combined with environmental monitoring and analytical calculations, the dosimeter results provide the individual crewmember's exposure record that is used to track against defined exposure limits.

Passive Radiation Area Dosimetry

Passive dosimetry, capable of measuring time-integrated absorbed dose and estimating average quality factor, must be deployed at designated fixed locations within each pressurized module. The exposure rates change with rack and stowage reconfigurations and throughout the vehicle assembly. Knowledge of the spatial distribution of exposure rate is necessary to identify areas that have a relatively high exposure rate (i.e. avoidance areas) and to reconstruct a crewmember's exposure in the event of a lost or otherwise unrecordable personal dosimeter. Passive dosimeters collect data even during situations when power is lost to other instruments.

Active Radiation Area Monitoring

Active radiation area monitoring is necessary to provide continuous information to ground controllers and to the crewmembers for the purpose of maintaining crew exposures ALARA and to serve as an alarm for increased radiation environments. Active monitoring throughout the habitable areas identifies high dose rate areas to be avoided by the crew, reduces uncertainty in final calculated crew risk assessments, and supports ALARA practices through verification of numerical vehicle shielding model.

Internal Time-resolved Charged-particle Monitoring

The time-resolved energy- and direction-dependent distribution of charge-identified particles inside the vehicle or surface habitat must be monitored. Measured charged-particle energy spectra are necessary for validating analytical models of the radiation flux environment. These data contain information to re-construct the crew equivalent dose and dynamic risk assessment models. All other physical quantities (such as LET spectra and absorbed dose) are not singular, and therefore result in ambiguity and hence increased uncertainty in estimates of crew health risk.

Time-resolved LET or y Spectrum Monitoring

Instrumentation to monitor the time-resolved LET spectrum, or as a surrogate, the lineal energy (y) spectrum. LET is a radiation parameter used to interpret the biological significance of absorbed dose from energetic ions and is used to derive the regulatory quantities equivalent dose and effective dose. The LET spectrum varies with position in orbit and with local solar weather conditions. Active, or powered instruments are required to report time-resolved LET or the surrogate lineal energy (y) spectral distributions.

Neutron Monitoring

Radiation monitoring instruments should provide the capability to characterize the neutron contribution to crew exposures. Results from scientific research demonstrate that secondary neutrons may contribute 10-30% of the total radiation effective dose received by astronauts inside a space vehicle. Since neutrons represent an important fraction of the crew's effective dose, it is necessary that this contribution be monitored for accurate reporting (as required by agency regulations) and accurate risk assessment determination. Neutrons can be monitored directly through neutron spectroscopy. However, because of the technical difficulties inherent in performing such measurements in the mixed neutron-charged particle environment behind spacecraft shielding, measurements designed to accurately measure the contribution of neutrons to the dose and dose equivalent can be used as a surrogate for direct neutron spectroscopy.

External Radiation Area Monitoring

External active radiation area monitoring should monitor the time-resolved direction- and energy-dependent charged-particle spectra immediately exterior to the vehicle. Measurements of the external direction- and energy-dependent charged particle spectra are used with radiation transport codes and models of the vehicle's mass distribution to calculate the radiation environment inside the vehicle as part of the crew health risk assessment process. In addition, instruments inside the vehicle cannot monitor a significant portion of the external radiation environment that is important to EVA crew exposures.

Radiation Contingency Monitoring

High range, high rate dosimeters should be present on board in order to measure high dose-rate contingency events. Extreme space radiation environmental conditions are possible that greatly exceed levels that can be accurately measured by LET or charged particle spectrometers. High rate dosimeters that can be read by the crew are specifically designed to accurately measure under such extreme conditions.

Charged-particle Survey Coverage

Time-resolved measurements of the energy-and direction-dependent distribution of charge-identified particles should be made in each habitable module. Instrumentation should be capable of surveying the majority of each module. Charged-particle energy spectra are necessary for validating models of the radiation environment inside a vehicle. These data contain sufficient information to estimate crew organ exposures and resulting risk.

Internal Charged-particle Data Down-link

Detailed data from time-resolved energy- and direction-dependent charged-particle detector should be down-linked on a time scale that precludes loss of data or to support contingency evaluation for real-time flight support. Due to the volume of detailed particle data which will be acquired and the finite quantity of instrument data storage onboard, it is necessary to frequently download the charged particle data to the ground to ensure data will not be lost. The detailed particle data will also be used periodically to update the estimated crew cumulative exposure risk.

Internal Charged-particle Dose Rate Down-link

Dose rate from charged-particle monitoring equipment should be continuously transferred to the ground for operational evaluation and real-time flight support. The requirement provides flight control personnel with an accurate insight into the radiation environment experienced by the crew, especially during periods of enhanced space environment conditions. Dose rate data transferred to the ground serves as the basis for implementation of immediate dose management actions. Although this requirement is not the primary purpose for charged-particle monitoring equipment, it provides a measure of redundancy for dose rate monitoring.

LET or y Spectrum Data Downlink

Time-resolved data from at least one LET monitoring instrument should be transferred to the ground as required for operational evaluation. The requirement provides flight control personnel with an accurate insight into the radiation environment experienced by the crew, especially during periods of enhanced space environment conditions.

External Time-resolved Charged-particle Data Down-link

Detailed time-resolved particle spectra should be down-linked on a timescale that precludes loss of data. Due to the volume of detailed particle data that will be acquired, and the finite quantity of instrument data storage, it is necessary to frequently down-link or download the charged particle data to the ground to ensure data will not be lost. The detailed particle data will also be used for periodically updating the estimated crew cumulative exposure risk.

External Dose Rate Data Down-link

Dose rate data characterizing the local radiation environment outside the vehicle should be continuously transferred to the ground for operational evaluation and real-time flight support. The requirement provides flight control personnel with an accurate insight into the status of the external radiation environment, especially during enhanced periods associated with space weather activity. External dose rate data transferred to the ground serves as part of the information used to make EVA go/no-go recommendations.

Alarm Capability

At least one onboard active instrument should have the ability to alert the crew when exposure rates exceed a set threshold. An onboard radiation alarm/warning system enables the crew to implement immediate countermeasures for transient high-radiation events. Without an alarm, the crew will not be able to apply immediate countermeasures. Reliance on crew alerts via ground-based monitoring or model predictions requires continuous communication coverage, which is not always available.

Biodosimetry Requirements

TBD

Lunar EVA Requirements

TBD

20.16 Micrometeoroid and Orbital Debris (MMOD) Technology Assessment

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20.16.1 MMOD Protection Implications from Columbia Accident Investigation Board

Impact damage from micrometeoroids and orbital debris (MMOD) represent a threat to crew safety and mission success for exploration spacecraft. Past MMOD damage to Shuttle vehicles has required costly repair or replacement of parts of the vehicle, and in a few cases the damage represented a "close-call" in that more serious consequences (mission abort or loss of vehicle) were narrowly averted.

NASA has raised the bar on flight safety. The MMOD protection system for Crew Exploration Vehicle (CEV) and other future exploration spacecraft must meet and exceed the level of safety and reliability achieved by past spacecraft. Apollo was designed to meet a protection requirement of no more then 1 in 250 risk of loss of vehicle per mission due to meteoroid impact (orbital debris was not a factor in Apollo design). CAIB recommendations for Shuttle MMOD protection indicate that no more then 1 in 1000 risk of loss of crew (LOC) and loss of vehicle (LOV) are desired per mission for the remaining life of the program. These and other factors imply that conventional MMOD shielding approaches used for Apollo are inadequate for CEV. Simply adding more MMOD protection is not desirable due to cost, launcher up mass and volume constraints. Development of low-weight, high-performance MMOD shielding techniques are recommended to meet protection requirements for crew safety and mission success of exploration spacecraft. Advanced MMOD shielding technologies are currently being researched by NASA Johnson Space Center, and offer promise in application to exploration missions for MMOD protection. These technologies include: (1) stronger shielding using a new class of materials, (2) smarter shielding using integrated sensors to detect and locate impact damage, (3) self-healing shielding materials that would seal holes in pressure shells and bladders, and (4) combined MMOD and radiation shielding. Meeting high MMOD protection requirements are more easily accomplished given adequate distance between outer mold-line (OML) and inner mold-line (IML) of the vehicle, which governs the overall thickness of the MMOD protection system. Vehicle OML and IML is a fundamental vehicle design variable that is established early in the design process. Given the close relationship between fundamental vehicle architecture parameters and crew safety/mission success from MMOD impact, it is recommended that MMOD assessments be factored into early design trades, and that clear and specific requirements for MMOD protection be included in the design specifications for exploration vehicles.

20.16.2 MMOD Risk Assessment Process

NASA applies a standard methodology to assess MMOD risks to ISS, Shuttle, EVA suits (EMU and Orlan), and other satellites/spacecraft such as Hubble Space Telescope and the Gamma-ray Large-Area Space Telescope (GLAST). The methodology, shown in the schematic below, is based on sound engineering test and analysis principles [Ref.1-3]. Hypervelocity impact tests are conducted on representative samples of the spacecraft structure to provide data for ballistic limit equations, which are semi-empirical equations defining MMOD particle sizes on the failure threshold limit of the spacecraft structure as function of impact speed, particle density, and shape. Results from numerical simulations (hydrocodes) are also used in developing the ballistic limit equations. Failure criteria are established for each region of the spacecraft based on engineering test/analysis of impact-damaged structures and their residual capabilities to perform the mission (i.e., failure criteria definitions are spacecraft specific and region/location specific). MMOD risks are assessed using the BUMPER code, using a finite element model defining the geometry of the spacecraft as an input, and the required ballistic limit equations and the standard NASA MMOD models (SSP 30425, Rev.B for meteoroids; and NASA TP-2002-210780 for orbital debris) which are contained within the BUMPER code. Assessed MMOD risks are compared to protection requirements, with iteration of the risk assessment process necessary until the design is complete, impact tests/analysis complete, and requirements are met. The risk assessment process is useful in identifying the risk drivers for a particular spacecraft (i.e., not everything is broken). Greater emphasis is placed on reducing risks for the drivers, through shielding design modification or operational changes, as they have the greatest effect on reducing overall spacecraft risks.

Figure 20.16.2-1: NASA JSC Two Stage Light-Gas Gun

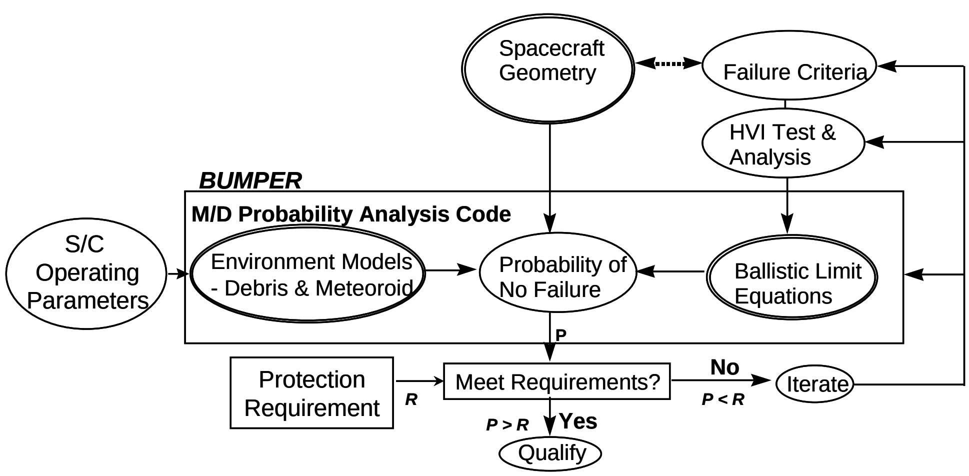


Figure 20.16.2-2: BUMPER Model Schematic

20.16.3 Observed MMOD Impact Damage

MMOD is a real threat with observable consequences in terms of post-flight inspection and repair activities. On average, 30 MMOD impacts are found after each Shuttle mission during inspections of Orbiter vehicle radiators, windows and wing leading edge (~10% of vehicle) [Ref.4-6]. Over 2000 impacts have been recorded to the Shuttle orbiter vehicle since STS-50, and dozens of MMOD impacts have been found on the ISS mini-pressurized logistics module post-flight. On average, 1 window per Shuttle flight is replaced due to MMOD impact damage. In a few cases, the MMOD damage approaches a "close-call" from the sense that a more serious consequence was narrowly averted. For instance, after STS-82, a crater was found on a Orbiter radiator interconnect line that had detached spall for the inside of the crater (pieces of the interconnect line came off into the coolant stream). There was 0.4mm of metal left in the interconnect line. If the impacting MMOD particle occurred at a slightly higher speed, or was slightly larger, the line could have been punctured. A line puncture would have resulted in loss of coolant from 1 of the 2 coolant loops on the vehicle and an early mission abort.

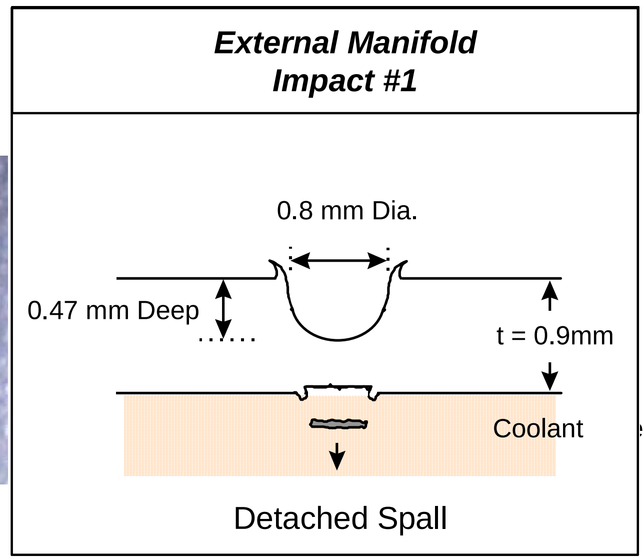


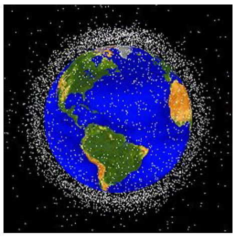
Figure 20.16.3-1: MMOD Impact Damage

20.16.4 MMOD Environments

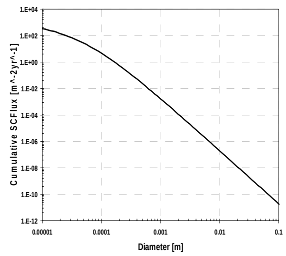
Lunar mission spacecraft will be exposed to three types of hypervelocity micrometeoroid/orbital debris (MM/OD) environments, with variable impact velocity and particle density:

- Micrometeoroids (11-72 km/s, 0.5 to 2.0 g/cm^3): CEV, service module, transfer stages, Lander, Extravehicular Mobility Unit (EMU), surface hardware
- Orbital debris in Earth orbit (1-17 km/s, 2.8 to 7.8 g/cm^3): CEV, service module, transfer stages, Lander, EMU (for EVAs in Earth Orbit)
- Secondary ejecta on lunar surface (1-3 km/s, 2.5 to 3.5 g/cm^3): Lander, EMU, surface hardware

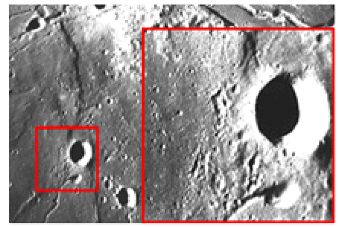
Secondaries on the lunar surface were the risk driver for the Apollo lander (i.e., the risk from secondaries was assessed as a greater risk than from micrometeoroids). MMOD protection systems for Crew Exploration Vehicle (CEV) and other components of lunar missions must be designed to protect from all three environments.



Orbital Debris in Earth Orbit



Grün Meteoroid Model (cumulative flux #/m^2-yr vs. particle diameter)



Aristarchus crater (40km dia.) and secondary craters

Figure 20.16.4-1: MMOD Environments

20.16.5 MMOD Requirements

MMOD protection system for Crew Exploration Vehicle (CEV) and other future exploration spacecraft must meet and exceed the level of safety and reliability achieved by past spacecraft. Apollo was designed to meet a protection requirement of no more then 1 in 250 risk of loss of vehicle per 8.3 day mission due to meteoroid impact (orbital debris was not a factor in Apollo design). CAIB recommendations for Shuttle MMOD protection indicate that no more than 1 in 1000 risk of loss of crew/loss of vehicle (LOC/LOV) are desired per mission (10.8 day average duration) for the remaining life of the program. The exploration missions will be exposed to a greater risk of MMOD impact than past missions. The CEV and other exploration spacecraft (service module, transfer stages, lander) will be larger then Apollo equivalents to accommodate a larger crew complement. In addition, the CEV will be exposed to the growing threat of orbital debris in Earth orbit, and exploration mission durations will be longer than Apollo (14 to 28 day missions). These factors imply that the conventional MMOD shielding approaches used for Apollo are inadequate for the CEV. Simply adding more MMOD protection is not desirable due to cost, launcher up mass and volume constraints. The cumulative risk from MMOD should be considered for the exploration missions when determining what level of protection is acceptable on a per mission basis. The graph below illustrates the difference in cumulative MMOD risks after 10 years of lunar exploration missions (assuming 1 exploration mission per year for 10 years) for two levels of overall mission risk corresponding to (1) Apollo risk of 1 in 250 per mission, and (2) Shuttle CAIB recommended risk of 1 in 1000 per mission. As illustrated, the cumulative risk for an Apollo type MMOD requirement is 4% (1 in 25) risk of failure, which is 4X higher than the 1% (1 in 100) failure risk resulting from a Shuttle CAIB recommended MMOD requirement. Clearly it is desirable to lower MMOD risk on a per-mission basis in order to achieve lower risk to crew for the long-term life of the exploration Program, if technology is readily available to accomplish the risk reduction.

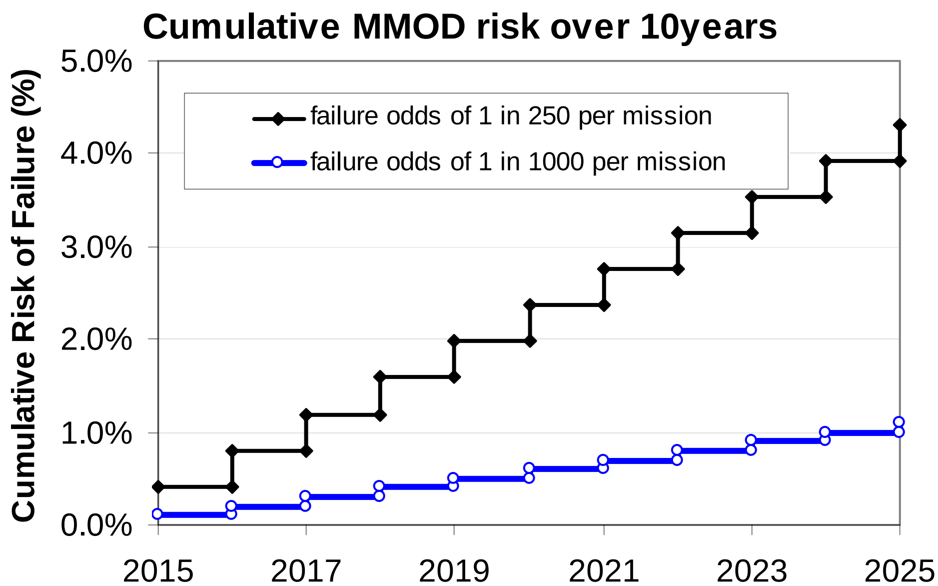


Figure 20.16.5-1: Cumulative MMOD Risk

Recommendation for Top-Level MMOD requirements

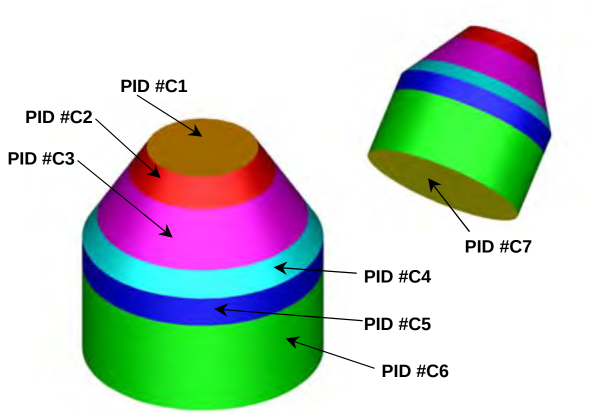
To provide adequate MMOD protection to ensure crew safety and mission success, it is essential from a design standpoint to provide clear and specific MMOD protection requirements in top-level design specifications for exploration vehicles. Meeting high MMOD protection requirements is most easily accomplished when the requirements are written into top-level specifications. Clear and specific requirements for MMOD protection in terms of crew safety (probability of no MMOD failure leading to loss of crew or loss of vehicle) and mission success (probability of no MMOD failure leading to mission loss or early abort) are desired. The top-level MMOD requirement focuses Program attention (civil servant and contractor alike) on the need to provide adequate MMOD protection. An alternative approach that does not specify an MMOD requirement, but instead specifies that MMOD would be calculated as part of an overall probabilistic risk assessment (PRA) for the spacecraft, is not suitable to provide incorporation of adequate MMOD protection early in the design process when it is cheaper and more feasible to attain high safety standards. PRA's are very complex, costly and difficult undertakings. PRAs for Shuttle and ISS have been attempted, but they are very late-stage products that are only developed after the design is complete and data obtained on failure rates for components of the design. For MMOD portions of the ISS and Shuttle PRAs, BUMPER output for MMOD risks is used in the PRAs. However, because they only use a single attitude/mission MMOD risk, the PRAs are very limited in terms of the type of MMOD risk calculations they can perform and are, therefore, unsuited for readily assessing MMOD shielding changes or modifications to vehicle operations/flight attitudes to meet protection requirements. The best approach specifies MMOD requirement early in the design and development cycle, well before majority of the vehicle design is fixed, so that providing robust MMOD protection levels will not involve difficult and expensive late changes to the spacecraft. With ISS, top-level MMOD requirement specifications were used in the design process to allocate risks to elements, so that very early in the design process engineering considerations were given by each element design team to necessary changes to meet MMOD requirements. Costs are relatively low if MMOD is considered early in the design process. It is much more difficult and costly to incorporate MMOD changes late in the design process and, ultimately, not as successful in lowering risk. Thus, top-level, specific MMOD requirements are absolutely essential in the Exploration Program to ensure crew safety and mission success. Any other alternative invites serious consequences.

MMOD Failure Modes and Requirements Allocation

Separate MMOD requirements for crew safety and mission success are necessary. MMOD impacts that can lead to loss of vehicle/loss of crew can differ considerably in terms of damage modes and damage to vehicle subsystems from impact damage that can lead to a mission abort. For instance, impact damage to the thermal protection system (TPS) of the CEV could lead (if undetected/unrepaired) to loss of vehicle and crew during reentry, and would, therefore, be a factor to consider in assessing compliance to crew safety requirements. But CEV TPS damage would not necessarily be involved in considerations for a mission abort. MMOD damage that leads to loss or unacceptable degradation of the Lander and/or lander transfer stage function on the non-crewed outboard portion of the mission would be a mission abort/mission success issue, but not crew safety. Specific and clear definitions of failure criteria are part of the MMOD risk assessment process. MMOD requirements will need to be allocated to major vehicle elements at an early point in the design process in order for risk assessments to be completed and optimum design solutions to be identified. Such allocations can be considered on an area-time basis for each vehicle element.

MMOD Risk Assessment Tools

BUMPER code is a MMOD risk assessment tool that combines environment definition models as well as results from hypervelocity impact tests and simulations to determine risks, optimize design, and verify requirements compliance. MMOD risks for ISS, Shuttle, EVA and other NASA spacecraft have been assessed using BUMPER. Risks for MMOD damage causing loss of vehicle/loss of crew of an Apollo type capsule have been assessed based on a simplified model of the geometry and spacecraft structure shown below.



PID #	Location	General Description
C1	Docking	AFRSI and Crew Cabin
C2	Forward Crew Cabin	AFRSI and Crew Cabin
C3	Central Crew Cabin	AFRSI and Crew Cabin
C4	Aft Crew Cabin	AFRSI and Crew Cabin
C5	Service Module and Aft TPS	Aft heat shield protected by Service Module
C6	Service Module	Pressure Vessels
C7	Service Module, Rear Wall	Pressure Vessels
C8	ISS	Shadowing

Figure 20.16.5-2: Results from BUMPER Tool

20.16.6 MMOD Technology Development

It is essential to advance the state-of-the-art in shielding protection and apply them to MMOD protection for exploration spacecraft, to ensure crew safety and mission success with the least-possible shielding mass and cost. Shielding research leads to new technologies and capabilities that allow Programs to set requirements on contractors for hardware that utilizes the new technologies (as indicated in schematic below).

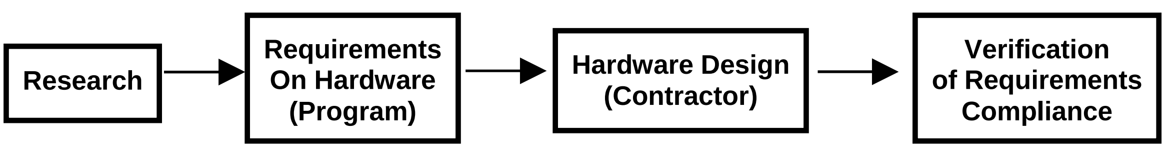
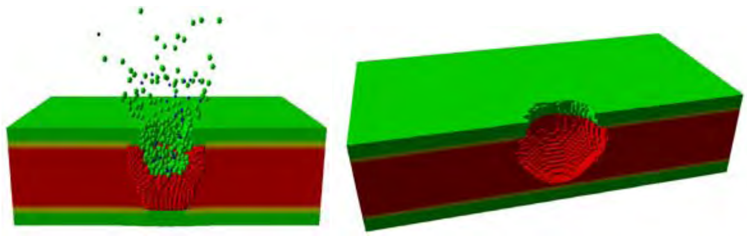


Figure 20.16.6-1: Schematic of MMOD Technology Development

<u>Advanced MMOD shielding</u>

Near-term advancements (2-3 years) in shielding technology being researched by the NASA Johnson Space Center [Ref.7] offer promise in meeting MMOD requirements for exploration vehicle crew safety and mission success. Advanced MMOD shielding technologies include: (1) stronger shielding using a new class of materials, such as metal foams, ceramic foams, high-strength fabrics, and ceramic fabrics; (2) smarter shielding using integrated sensors to detect and locate impact damage such as acoustic emission sensors, piezoelectric thin-film materials, fiber optic sensors; and (3) self-healing shielding materials that would seal holes in pressure shells and bladders. Through the proper selection of materials, advanced MMOD shielding can provide enhanced radiation protection, as well. In addition, another desirable characteristic of the new MMOD shielding under development is a reduction in the generation of secondary debris fragments when impacted, which translates into lower growth of orbital debris.



Hypervelocity impact and simulation results indicate that 1" of low-density SiC-foam behind the RCC panel or on WLE spar reduces meteoroid/debris penetration risk by factor of 20 from current single wall RCC design.

Figure 20.16.6-2: Hypervelocity Impact Simulation Results

20.16.7 References

1. NASA Technical Publication, TP-2003-210788, Meteoroid/Debris Shielding, August 2003.
2. NASA Johnson Space Center, Hypervelocity Impact Technology Facility (HITF) web site: http://hitf.jsc.nasa.gov/hitfpub/main/index.html, report archive: http://hitf-archive.jsc.nasa.gov
3. NASA JSC Report No. JSC-29581, Space Shuttle Meteoroid & Orbital Debris Threat Assessment Handbook, Using the BUMPER-II Code for Shuttle Analysis, 2001.
4. NASA JSC Report No. JSC-28033, Orbiter Meteoroid/Orbital Debris Impacts: STS-50 (6/92) through STS-86 (10/97), 1998.
5. E.L.Christiansen, et al., Space Shuttle debris & meteoroid impacts, Advances in Space Research, COSPAR publication, JASR 6525, DTD 4.3.1/SPS-N, 2004.
6. NASA JSC Shuttle MMOD web site: http://hitf.jsc.nasa.gov/hitfpub/shuttle/pword_admin.cfm
7. NASA technology inventory database, FY04 task #9476, Advanced meteoroid and debris shielding, http://inventory.gsfc.nasa.gov

20.17 Risks and Hazards Assessments

Submitted by: ***NASA/JSC/NX – Advanced Programs and Analysis Division, Safety & Mission Assurance Directorate***

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20.17.1 Preliminary Hazards Analysis

As part of the Safety and Mission Assurance (S&MA) support to the LDRM-2 study, a list of preliminary hazards for the CEV, CEV Injection Stage, and Lunar Lander was generated. Preliminary hazards that were identified in previous exploration studies and the 2nd Generation Reusable Launch Vehicle (2GRLV) Space Launch Initiative (SLI) were compiled into one document. Once compiled, it was decided to propagate three separate Preliminary Hazard lists for the CEV, CEV Injection Stage, and Lunar Lander elements. Preliminary hazards were not generated for the Lunar Lander Injection Stage since the crew could not be harmed if there were any type of failure. Ground processing and pre-launch hazards were out of scope for this study.

Once the preliminary hazards were categorized by element, they were reviewed for applicability and content. Several of the preliminary hazards were very detailed and were modified to be more general. This was done to keep the level of detail consistent between the LDRM-2 study, the preliminary hazards, and risks. There were a total of thirteen hazardous conditions identified with an associated one hundred twenty-nine causes for the CEV. A total of eleven hazardous conditions with an associated one hundred seven causes were identified for the Lunar Lander. The CEV Injection Stage had a total of four hazardous conditions with an associated seventeen causes identified. The CEV Injection Stage had fewer hazardous conditions than the CEV and Lunar Lander due to having a potential impact on crew safety for a short period of time (TRM-25 thru TRM-28). In addition to identifying the hazardous conditions and associated causes for the CEV, CEV Injection Stage, and Lunar Lander, the TRM critical events were mapped backed to each cause showing where in the mission profile they could occur. All three preliminary hazard analyses are listed in the tables below.

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-01-01	**Contamination in habitable volume**	Lunar Dust entering the pressurized volume	Respiratory irritation; Subsystem degradation	EVA post-ops procedures; Vacuum system; ECLSS	TRM-48 thru TRM-55
CEV-01-02		Non-containment of Payloads/Science/ Lunar samples	Respiratory, Mucous membrane, skin irritation	Adequate crew procedures and equipment for isolation and containment of samples; adequate monitors	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-03		Leakage from Power batteries/fuel cells	Leakage from power storage batteries can damage hardware or injure crew members	Containment of electrolytic battery/fuel cell media to reduce the possibility of cabin atmosphere exposure	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-04		Failure to remove Smoke/Fire by-products	Injury or loss of crew member	FDSS; Emergency O2 supply; Material selection; Redundant cabin venting system.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-05		Loss of CO2 removal capability	Loss of Crew/Vehicle	Redundant CO2 Removal System during launch and re-entry; EVA back-up capability	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-06		Toxic Environment in the CEV pressurized volume	Injury or death to crewmember	Materials Selection; Emergency O2 supply; Redundant cabin venting system with a manual override; Atmospheric monitoring	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-01-07		Leakage of fuel from Propulsion/RCS system into the CEV pressurized volume	1) Toxic effects to crew 2) Increased likelihood of explosion	1) All propulsion lines for forward RCS will be outside habitable volume to eliminate the possibility of Methane contamination 2) Trace Contaminant Control system for detection of propulsion fuel and procedures to isolate leak	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-08		Leakage of Human Byproducts from the WCS	Possible injury to crew	Adequate waste containment system with redundancy; Adequate crew procedures and equipment for isolation and containment	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-09		Inadequate protection from shattering or containment of shatterable material allows release of debris in habitable environment	Crew exposed to particulate contamination	All shatterable material is provided with positive protection to prevent fragments from entering the habitable environment.	TRM-20 thru TRM-33;TRM-48 thru TRM-55
CEV-01-10		Tool/Science Equipment Battery Leakage	Respiratory, skin, eye irritation; Mucous membrane irritation	Battery design; Adequate leakage containment	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-11		Battery leakage into the CEV pressurized volume	Respiratory, skin, eye irritation; Mucous membrane irritation	Battery design; Adequate storage of batteries; Adequate detection of leakage	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-01-12		Leakage of TCS media into the CEV pressurized volume	Respiratory irritation; Mucous membrane irritation	ECLSS; Adequate detection of leakage; Adequate storage of spares/ replaced media	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-01-13		Post landing venting/ingestion failures	Injury or death to crewmember	Adequate monitoring to land in a safe environment; Redundant cabin venting system	TRM-54 thru TRM-55
CEV-02-01	**Electrical Shock**	Inadequate grounding of surfaces accessible to the crew	Injury or death to crewmember	Design; Testing; Redundancy; Proper procedures	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-02-02		CEV Static Discharge	Injury or death to crewmember	Adequate measures for controlling potential static discharges; Proper insulation	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-02-03		CEV System/Payload Short Circuit	Injury or death to crewmember	Circuit breakers; Adequate grounding; Sufficient insulation; Design & Testing	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-02-04		Improper Circuit/Equipment Design	Injury or death to crewmember	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design limitations	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-03-01	**Environmental Hazards**	Excessive Thermal Conditions inside the CEV pressurized volume	Exceed lower or upper thermal limit of crew/vehicle components within the CEV pressurized volume	Adequate Passive Thermal Control System (PTCS); Active TCS (ATCS)	TRM-20 thru TRM-33;TRM-48 thru TRM-55
CEV-03-02		Excessive External Thermal Conditions post landing	Exceed lower thermal limit of the crew when exposed to extreme temperatures	Adequate survival equipment. Stand-alone suit cooling.	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-03-03		Excessive Noise within the CEV pressurized volume	Physiological and psychological effects on crew	Incorporate a passive acoustic abatement system; Hearing protection used in areas of high noise generation; System will be tested to ensure the vehicle meets NASA-STD-3000 requirements.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-03-04		Excessive radiation exposure to the crew	Long-term Crew Health; Carcinoma	Accepted Risk, minimum radiation protection by design; Adequate monitoring of solar activity	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-03-05		Inadequate/ inappropriate lighting in habitable volume	Physiological and psychological effects on crew	Permanent general lighting and portable task lighting installed on the vehicle; Emergency lighting	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-03-06		Sharp Edges/Pinch Points within the CEV pressurized volume	Possible Injury to crewmember	Hardware designed where they will not pinch or snag the crew or their clothing. Exposed surfaces should be smooth and free of burrs	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-03-07		Water leak in the pressurized cabin	Possible Injury to crew; Physiological and psychological effects on crew	Redundant valves and piping to control water leaks; fracture controls for water tank (leak before burst)	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-03-08		Landing in adverse weather conditions	Possible Injury to crew; Physiological and psychological effects on crew	Motion sickness medicine; TCS; Ventilation system;	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-03-09		Landing in a hazardous or toxic environment	Loss of Crew/Vehicle	Adequate design for pressurized volume; Post-landing ventilation system	TRM-54 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-03-10		Repressurizing cabin during descent/landing allows hazardous gas ingestion	Loss of Crew/Vehicle	Repressurize with CEV supplied consumables; Use non-toxic OMS/RCS propellants for descent.	TRM-20 thru TRM-33;TRM-48 thru TRM-55
CEV-04-01	**Fire or Explosion**	Improper Circuit Design causing a fire inside the CEV pressurized volume	Loss of Crew/Vehicle	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design, FDSS	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-04-02		Improper power connector design that does not preclude improper mismate / demate.	Loss of Crew/Vehicle	All power connectors are designed such that they cannot be mismated or cross-connected.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-04-03		Inadvertant OMS/ACS liquid propellant explosion	Loss of Crew/Vehicle	Pressure relief valves; Leak before burst design; Adequate shielding	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-04-04		Use of Flammable Materials within the CEV pressurized volume	Loss of Crew/Vehicle	Design in accordance with manned space flight Material Selection Requirements, Fire Detection & Suppression System (FDSS)	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-04-05		High Pressure Vessel rupture on the CEV Service Module (SM)	Loss of Crew/Vehicle	High pressure vessels will be designed to leak before bursting by material selection/properties; Positive Pressure Relief Valve (PPRV) on pressure vessel and Vehicle Cabin; FDSS	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-04-06		Ignition Source malfunction(s)	Loss of Crew/Vehicle	Preclude ignition sources by design; Material selection; FDSS; Safeing and arming circuitry	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-04-07		High concentration of Oxygen within the CEV pressurized volume	Increased flammability of materials	Redundant O2 Partial Pressure sensing and control, Material selection; FDSS	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-04-08		Inadvertent CEV Escape Rocket solid propellant ignition	Loss of Crew/Vehicle	Adequate safing and arming circuitry; Design the pod with enough structural integrity to withstand the blast wave from a pusher rocket explosion	TRM-20 thru TRM-22
CEV-05-01	**Impact/Collision**	Collision with CEV Injection Stage	Loss of Crew/Vehicle	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture. 2) Procedures for safe proximity operations will be maintained to minimize potential for collision. 3) Crew has option of manual control	TRM-25 thru TRM-28
CEV-05-02		Loss of CEV attitude control	Loss of Crew/Vehicle	Two-fault tolerant RCS; RCS shall be designed in accordance with the Human Rating Requirements	TRM-24 thru TRM-33; TRM-48 thru TRM-55
CEV-05-03		Impact of Rotating or moving equipment within the CEV pressurized volume	Injury or death to crewmember	Proper equipment design and adequate crew procedures.	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-05-04		Inadequate positive backout prevention for safety critical fasteners results in structural damage.	Injury or death to crewmember	All system safety critical fasteners will be designed to prevent backout.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-05-05		Loss of vehicle control during proximity operations with Injection Stage	Potential loss of crew, vehicle, or Injection Stage	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture. 2) Procedures for safe proximity operations will be maintained to minimize potential for collision. 3) Crew has option of manual control	TRM-25 thru TRM-26; TRM-28
CEV-05-06		Loss of vehicle control during proximity operations with Lunar Lander	Potential loss of crew, vehicle, or Lunar Lander	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture.2) Procedures for safe proximity operations will be maintained to minimize potential for collision.3) Crew has option of manual control	TRM-31 thru TRM-33;TRM-48 thru TRM-49
CEV-05-07		Collision with Lunar Lander	Loss of Crew/Vehicle	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture. 2) Procedures for safe proximity operations will be maintained to minimize potential for collision. 3) Crew has option of manual control	TRM-31 thru TRM-33; TRM-48 thru TRM-49

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-05-08		Impact with MMOD	Loss of Crew/Vehicle	Accepted risk or MMOD protection designed to shield CEV or at least the critical systems	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-05-09		Inadequately restrained equipment in Habitable Volume	Loss of Crew/Vehicle	1) Adequate design of restraints 2) Adequate crew procedures for stowage of items	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-05-10		Use of non-conforming fasteners results in release of hardware components	Injury to crewmember	All fasteners conform to an approved fastener integrity program.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-05-11		Inadequate hardware design results in structural damage	Crew exposed to debris/shrapnel or other hazardous condition as result of structural failure of hardware	System design to provide positive margins of safety under all loading conditions including crew handling, on-orbit vibration with respect to the required safety factors.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-05-12		Landing in an unfavorable terrain environment	Injury or death to crewmember	Adequate crew restraints; Adequate structural integrity	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-05-13		CEV SM contact with the CEV CM during an abort scenario	Injury or death to crewmember	Adequate attitude control and solid rocket motor thrust	TRM-20 thru TRM-24;TRM-52 thru TRM-54
CEV-05-14		CEV contact with the LV during an abort scenario	Injury or death to crewmember	Adequate attitude control and solid rocket motor thrust (solid motor thrust was for the worst-case Q-bar)	TRM-20 thru TRM-23

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-05-15		Airbags fail to deploy prior to the CEV landing	Injury or death to crewmember	Redundant airbag deployment mechanisms; Pod is designed to float in the water without the use of airbags.	TRM-54 thru TRM-55
CEV-05-16		Landing loads cause damage to the CEV egress hatch mechanisms	Blocked egress path; Possible injury or death of crewmember	Redundant egress path; Emergency pyros for the hatch	TRM-54 thru TRM-55
CEV-05-17		Crew restraint system fails	Injury or death to crewmember	Adequate and redundant crew restraints as defined in NASA-STD-3000, 5.3	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-05-18		The CEV landing mechanism fails to deploy	Injury or death to crewmember	Redundant landing mechanism	TRM-54 thru TRM-55
CEV-05-19		EVA Crewmember or equipment impact with CEV	Loss of Crew/Vehicle	All EVA crewmembers and equipment tethered; EVA translation paths on CEV clearly defined as in NASA-STD-3000; Robust CEV TPS system	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-06-01	**Loss of Habitable Environment**	Loss of O2 Supply in the CEV pressurized volume	Loss of Crew/Vehicle	Redundant O2 Partial Pressure Supply, Sensing and Control with backup procedures to use EVA suit.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-02		Loss of TCS within the CEV pressurized volume	Loss of Crew/Vehicle	Redundant loop TCS system	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-06-03		Toxic Environment in the CEV pressurized volume	Injury or death to crewmember	Materials Selection; Emergency O2 supply; EVA crew member procedures to assure decontamination prior entering CEV habitable volume; Experiments/Payloads meet standard safety requirements; TCCS	TRM-20 thru TRM-33;TRM-48 thru TRM-55
CEV-06-04		CEV sustains a Loss of Power	Loss of Crew/Vehicle	Two fault-tolerant power system	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-05		Inadequate/ inappropriate lighting in habitable volume	Physiological and psychological effects on crew	Acceptable/adequate lighting design on the CEV and Lunar Lander; Emergency lighting	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-06		Leakage of fuel from Propulsion/RCS system into the CEV pressurized volume	1) Toxic effects to crew 2) Increased likelihood of explosion	All propulsion lines for forward RCS will be outside habitable volume to eliminate the possibility of fuel contamination; TCCS for detection of propulsion fuel and procedures to isolate leak; Emergency O2 supply; Prop/RCS relief valves	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-07		Inability to adequately vent the CEV pressurized volume post landing	Injury or death to crewmember	Redundant electro-mechanically actuated vent and fan; Manual override	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-08		Loss of CO2 removal capability	Loss of Crew/Vehicle	Redundant CO2 Removal capability with back up procedures to use EVA suit.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-09		High concentration of Nitrogen within the CEV pressurized volume	Injury or Loss of Crew	Redundant N2 Partial Pressure sensing and control	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-06-10		ACS propellant leakage into the CEV pressurized volume	Injury or death to crewmember	ACS relief valves; Propellant leakage sensors; Cabin venting; Emergency O2 supply; ACS is outside the habitable volume	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-11		Loss of CEV SM TPS	Injury or death to crewmember	Adequate amount of shielding; Redundant TPS	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-12		Loss of CEV CM TPS	Injury or death to crewmember	Adequate amount of shielding; Redundant TPS	TRM-20 thru TRM-33;TRM-48 thru TRM-55
CEV-06-13		Helium contamination of Oxygen supply	1) Crew suffocation with high concentrations of Helium 2) No toxic effects to crew with low concentrations of Helium	Helium tanks used for pressurizing the propulsion tanks will be outside of the habitable volume.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-06-14		CEV compartment depressurization	Loss of Crew/Vehicle	1) Adequate MMOD protection through design. 2) Adequate resources for cabin pressurization in the event of a critical leak. 3) Adequate structural design to prevent excessive leakage in habitable environment. 4) Docking vestibule pressure checked prior to opening CEV hatch.	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-07-01	**Physiological/ Psychological**	Acceleration, shock, impact & Vibration during descent and landing	Injury or death to crewmember	Adequate design of crew/equipment restraints; Adequate crew procedures for stowage of items	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-02		Effects of Pressure Changes on Crew	Possible Injury to crewmember	Adequate crew safety procedures for EVA pre-breath	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-03		Illness/Incapacitation of Crew Member during transit to L1	Injury or death to crewmember	Crew Health equipment and procedures	TRM-27 thru TRM-30; TRM-49 thru TRM-52
CEV-07-04		Excessive Noise inside the CEV pressurized volume	Possible Injury to crewmember	Incorporate a passive acoustic abatement system; Hearing protection used in areas of high noise generation; System will be tested to ensure the vehicle meets NASA-STD-3000 requirements.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-05		Sharp Edges/Pinch Points within the CEV pressurized volume	Possible Injury to crewmember	Hardware designed where they will not pinch or snag the crew or their clothing; Exposed surfaces should be smooth and free of burrs; Adhere to NASA-STD-3000 requirements.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-06		EVA Workloads & Fatigue	Possible Injury to crewmember	Crew procedures established to minimize crew fatigue	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-07-07		Interference with Translation Paths. Hardware impinges into translation paths.	Possible Injury to crewmember	Hardware designed to comply with traffic flow and translation paths; Adequate volume provided for a suit egress as stated in NASA-STD-3000.	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-07-08		Appendage Entrapment in Holes or Latches	Possible Injury to crewmember	Holes and latches meet NASA-STD-3000 design requirements designed to prevent entrapment of crew member's appendage.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-09		Loss of H2O supply	Possible injury or death to crewmember	Redundant water supply	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-10		Excessive Thermal Conditions inside the CEV pressurized volume	Possible injury or death to crewmember	Adequate passive thermal control system which will not allow an excessive amount of heat in at anytime	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-11		Crew experiences High G-Loads during reentry and descent	Possible Injury to crew; Physiological and psychological effects on crew	Full pressure suits; Recumbent seats; Trajectory & propulsion system to operate within human limits	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-12		Inadequate size of the CEV's habitable volume	Physiological and psychological effects on crew	Design for adequate amount of habitable volume for crew accommodation guidelines	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-13		Inadequate/ inappropriate lighting in habitable volume	Physiological and psychological effects on crew	Acceptable/adequate lighting design. LED's are a proposed solution for lighting.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-07-14		Excessive External Thermal Conditions post landing	Possible injury or death to crewmember	Survival equipment will be tethered onto suit. Stand-alone suit cooling.	TRM-54 thru TRM-55
CEV-08-01	**Loss of Vehicle Control**	Loss of CEV Guidance, Navigation, & Control	Loss of Crew and Vehicle	Redundant NAV systems	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-08-02		Loss of CEV Attitude Sensing	Loss of Crew and Vehicle	Redundant attitude sensing system	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-08-03		Loss of CEV Attitude Control	Loss of Crew and Vehicle	Redundant attitude control systems	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-08-04		Loss of CEV Central Command and Control Capability	Loss of Crew and Vehicle	Redundant General Purpose Computers to perform all vehicle functions.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-08-05		Loss of Delta-V Capability	Loss of Crew and Vehicle	1) 2 fault-tolerant propulsion system 2) The system contains an engine out capability for all injection/maneuvers 3) Redundant fuel and oxidizer supply to each engine 4) Redundant ignition at each engine	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-08-06		Parachutes fail to deploy	Loss of Crew and Vehicle	Deployment mechanisms for the main chutes consist of a barometer altimeter, a reef timer, and a pyro system to deploy the main chutes. Has three main chutes, of which two are needed for a safe landing.	TRM-54 thru TRM-55
CEV-08-07		CEV crew becomes incapacitated during the mission profile	Loss of Crew and Vehicle	Autonomous crew escape system and mechanisms; Autonomous crew rendezvous and capture system; Autonomous descent approach and landing systems	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-08-08		CEV health monitoring system(s) fails	Loss of Crew and Vehicle	Redundant system health monitoring	TRM-20 thru TRM-33; TRM-48 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-08-09		Drogue parachute fails to deploy	Loss of Crew and Vehicle	Redundant and reliable deployment mechanism for the drogue chute with a pyro system to cut the drogue chute away	TRM-54 thru TRM-55
CEV-08-10		Electro-magnetic actuators (EMAs) fail	Loss of Crew and Vehicle	Adequate redundancy for EMA's; Redundant power source for the EMA's	TRM-54 thru TRM-55
CEV-09-01	**Radiation**	Solar Flare occurs during CEV transit to/from L1	Injury or death to crewmember	Accepted Risk, Adequate monitoring of solar activity, maximum radiation protection	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-09-02		Crew and/or hardware is exposed to Non-Ionizing Radiation	Injury or death to crewmember	Minimize radiation emittance and maximize protection of components sensitive to EMI.	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-09-03		Crew and/or hardware is exposed to Ionizing Radiation	Injury or death to crewmember	Minimize sources of Ionizing radiation to be used in the CEV design	TRM-20 thru TRM-33; TRM-48 thru TRM-55
CEV-10-01	**Contingency EVA Operations**	Inability to return to Crew Habitat	Injury or death to crewmember	Crew EVA rescue procedures in place to secure injured crew member back to habitat	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-02		Crew Injury during a contingency EVA	Injury or death to crewmember	Crew CHeCs medical equipment	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-03		Contamination of EVA crewmember from leaking RCS/Engine Thruster	Possible Injury to crewmember	Isolation valves for upstream manifold and tanks.	TRM-25 thru TRM-33; TRM-48 thru TRM-51

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-10-04		EVA crewmember exposed to Sharp Edges/Corners and Pinch Points	Possible Injury to crewmember	Hardware designed where they will not pinch or snag the EVA suit. Exposed surfaces are smooth and free of burrs	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-05		EVA crewmember exposed to Excessive Non-Ionizing Radiation	Possible Loss of EVA Crew Member	Redundant Inhibits to ensure power is isolated from RF Amps and Electro-Magnets	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-06		Solar Flare/Ionizing cosmic Radiation exposure during a contingency EVA	Injury or death to crewmember	Safe Haven for radiation protection by design; Adequate monitoring of solar activity	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-07		Crew member collision with other elements attached to the CEV/Lunar Lander.	Possible Loss of EVA Crew Member	1) The worksite is clearly defined to minimize the possibility of an EVA crew member being struck by an object. 2) Adequate tools, equipment, and lighting for the safe performance of planned tasks.	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-08		EVA crewmember exposed to Ionizing Radiation	Possible Long-term Injury to EVA Crew Member	No sources of ionizing radiation will be used in the CEV design	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-09		Excessive EVA touch temperatures during contingency operations	Possible Loss of EVA Crew Member	Adequate Design to meet Touch Temperature Requirements	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-10		Inadequate design for EMU to handle deep space environment (temps, radiation, MM, solar dust).	Possible Loss of EVA Crew Member	Adequate design for EMU to handle deep space environment.	TRM-25 thru TRM-33; TRM-48 thru TRM-51

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-10-11		EVA crewmember receives Electric Shock	Possible Loss of EVA Crew Member	Adequate Circuit Design and Grounding	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-12		EVA crewmember is exposed to Static Discharge	Injury or death to crewmember	Adequate measures for controlling potential	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-13		Inadequate Restraints for EVA crewmember	Possible Loss of EVA Crew Member	Establish EVA Worksites, Pathways, Handholds, and Tether Attachment Points	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-14		EVA crewmember is Inadequately grounded	Injury or death to EVA crewmember	Design; Testing; Redundancy	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-15		EVA crewmember is exposed to Improper Circuit Design	Injury or death to EVA crewmember	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-16		EMU encumbrances	Inability to Maintain Payload	Adequate Design for EVA Maintenance or Servicing	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-17		EVA crewmember suffers MMOD impact	Possible Loss of EVA Crew Member	MMOD protection designed to shield EVA suit	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-10-18		Inability to re-pressurize CEV/internal airlock after contingency EVA is completed	Loss of Mission	Redundant Cabin pressure control system, Sensing and Control with backup procedures to use Lunar Lander vehicle for crew return to CEV.	TRM-25 thru TRM-33; TRM-48 thru TRM-51
CEV-11-01	**Inability to Dock/Transfer Crew /Undock**	Inability for CEV to Dock with Injection Stage	Loss of Crew/Vehicle	1) Redundant Docking systems to achieve docking 2) Backup procedures to depressurize CEV and perform contingency EVA	TRM-25 thru TRM-26

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-11-02		Inability for CEV to Dock with Lunar Lander	Loss of Crew/Vehicle	1) Redundant Docking systems to achieve docking 2) Backup procedures to depressurize CEV and perform contingency EVA	TRM-31 thru TRM-32
CEV-11-03		Inability to equalize pressure between the CEV & Lunar Lander	Loss of Crew/Vehicle	Hatches incorporate redundant pressure equalization valves.	TRM-32 thru TRM-34
CEV-11-04		Inability for CEV to undock from Lunar Lander	Loss of Crew/Vehicle	1) Docking system is two fault tolerant for undocking 2) Procedures for contingency EVA to undock vehicles	TRM-34
CEV-11-05		Structural failure of docking mechanism	Loss of Crew/Vehicle	Mechanisms designed to structural margin	TRM-26; TRM-32; TRM-47
CEV-11-06		Inability for CEV to undock from CEV Injection Stage	Loss of Crew/Vehicle	1) Docking system is two fault tolerant for undocking 2) Procedures for contingency EVA to undock vehicles	TRM-28
CEV-12-01	**Inability to egress vehicle after contingency Earth return**	Blocked primary pathway to egress vehicle post landing	Potential Loss of crew	Secondary egress escape route that is on a different plane than the primary route and is accessible from outside the vehicle is being incorporated into the CEV	TRM-54 thru TRM-55
CEV-12-02		Landing in adverse weather conditions at sea	Potential Loss of crew	Adequate built-in vehicle buoyancy; Vehicle floatation devices; Crew survival equipment; Procedures for emergency crew recovery	TRM-54 thru TRM-55

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-13-01	**Loss of Entry Capabilities**	Damage to the CEV's Thermal Protection System (TPS)	Loss of Crew and Vehicle	Provide heat shield inspection	TRM-53
CEV-13-02		CEV loses Electrical Power	Loss of Crew and Vehicle	Contain two-fault tolerance for the Electrical Power System	TRM-53
CEV-13-03		Loss of CEV Guidance, Navigation, & Control	Loss of Crew and Vehicle	1) Onboard Navigation consists of two-fault tolerant INS systems of which any one could perform the Navigation function. 2) DNS and redundant Communicating system	TRM-53
CEV-13-04		Loss of CEV Attitude Sensing	Loss of Crew and Vehicle	Two Star Trackers are provided with back-up capability to use optical equipment	TRM-53
CEV-13-05		Loss of CEV Attitude Control	Loss of Crew and Vehicle	CEV CM Attitude Control is through a monopropellant Tridyne Reaction Control System. Fail-Safe redundancy is incorporated for all vehicle translations and rotations.	TRM-53
CEV-13-06		Loss of Vehicle Central Command and Control Capability	Loss of Crew and Vehicle	Vehicle Contains two-fault tolerance for the General Purpose Computers which perform all vehicle functions.	TRM-53

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CEV-13-07		Loss of CEV Delta-V Capability	Loss of Crew and Vehicle	1) 2 fault-tolerant propulsion system 2) The system contains an engine out capability for all injection/maneuvers 3) Redundant fuel and oxidizer supply to each engine 4) Redundant ignition at each engine	TRM-53

Table 20.17.1-1: CEV Preliminary Hazard Analysis

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CIS-01-01	**Fire or Explosion**	Improper Circuit Design	Loss of Crew/Vehicle	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design, FDSS	TRM-26 thru TRM-28
CIS-01-02		Improper power connector design that does not preclude improper mismate / demate.	Loss of Crew/Vehicle	All power connectors are designed such that they cannot be mismated or cross-connected.	TRM-26 thru TRM-28
CIS-01-03		Inadvertant OMS/ACS liquid propellant explosion	Loss of Crew/Vehicle	Pressure relief valves; Leak before burst design; Adequate shielding	TRM-26 thru TRM-28
CIS-01-04		High Pressure Vessel rupture	Loss of Crew/Vehicle	High pressure vessels will be designed to leak before bursting by material selection/properties; Positive Pressure Relief Valve (PPRV) on pressure vessel and Vehicle Cabin; FDSS	TRM-26 thru TRM-28
CIS-01-05		Ignition Sources	Loss of Crew/Vehicle	Preclude ignition sources by design; Material selection; FDSS; Safeing and arming circuitry	TRM-26 thru TRM-28
CIS-01-06		Inadvertent CEV Injection Stage propellant explosion	Loss of Crew/Vehicle	Design the CEV with enough structural integrity to withstand the blast wave from a pusher rocket explosion	TRM-26 thru TRM-28

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CIS-02-01	**Impact/Collision**	Loss of CEV Injection Stage attitude control	Loss of Crew/Vehicle	Two-fault tolerant RCS; RCS shall be designed in accordance with the Human Rating Requirements	TRM-26 thru TRM-28
CIS-02-02		Loss of CEV Injection Stage control during proximity operations with the CEV	Potential loss of crew, vehicle, or Injection Stage	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture. 2) Procedures for safe proximity operations will be maintained to minimize potential for collision.	TRM-26 thru TRM-28
CIS-02-03		CEV Injection Stage Collision with CEV	Loss of Crew/Vehicle	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture. 2) Procedures for safe proximity operations will be maintained to minimize potential for collision.	TRM-26 thru TRM-28
CIS-02-04		CEV Injection Stage Impact with MMOD	Loss of Crew/Vehicle	Accepted risk or MMOD protection designed to shield CEV/Lunar Lander/Injection Stages or at least the critical systems	TRM-26 thru TRM-28
CIS-03-01	**Loss of Vehicle Control**	Loss of CEV Injection Stage Guidance, Navigation & Control	Loss of Crew and Vehicle	Redundant NAV systems	TRM-26 thru TRM-28

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CIS-03-02		Loss of CEV Injection Stage Attitude Sensing	Loss of Crew and Vehicle	Redundant attitude sensing system	TRM-26 thru TRM-28
CIS-03-03		Loss of CEV Injection Stage Attitude Control	Loss of Crew and Vehicle	Redundant attitude control systems	TRM-26 thru TRM-28
CIS-03-04		Loss of CEV Injection Stage Central Command and Control Capability	Loss of Crew and Vehicle	Redundant General Purpose Computers to perform all vehicle functions.	TRM-26 thru TRM-28
CIS-03-05		Loss of CEV Injection Stage Delta-V Capability	Loss of Crew and Vehicle	1) 2 fault-tolerant propulsion system 2) The system contains an engine out capability for all injection/maneuvers 3) Redundant fuel and oxidizer supply to each engine 4) Redundant ignition at each engine	TRM-26 thru TRM-28
CIS-03-06		Vehicle health monitoring system(s) fails	Loss of Crew and Vehicle	Redundant system health monitoring	TRM-26 thru TRM-28
CIS-04-01	**Inability to Dock/Undock with CEV**	Inability for CEV to Dock with Injection Stage	Loss of Crew/Vehicle	1) Redundant Docking systems to achieve docking 2) Backup procedures to de-pressure CEV and perform contingency EVA	TRM-26 thru TRM-28

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
CIS-04-02		Inability for CEV Injection Stage to undock from the CEV	Loss of Crew/Vehicle	1) Docking system is two fault tolerant for undocking 2) Procedures for contingency EVA to undock vehicles	TRM-26 thru TRM-28
CIS-04-03		Structural failure of docking mechanism(s)	Loss of Crew/Vehicle	Mechanisms designed to structural margin	TRM-26 thru TRM-28

Table 20.17.1-2: CEV Injection Stage Preliminary Hazard Analysis

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-01-01	**Contamination in habitable volume**	Lunar Dust entering the LL pressurized volume	Respiratory irritation; Subsystem degradation	EVA post-ops procedures; Vacuum system; ECLSS	TRM-40
LL-01-02		Non-containment of Payloads/Science/ Lunar samples	Respiratory, Mucous membrane, skin irritation	Adequate crew procedures and equipment for isolation and containment of samples; adequate monitors	TRM-34 thru TRM-47
LL-01-03		Leakage from Power batteries/fuel cells	Leakage from power storage batteries can damage hardware or injure crew members	Containment of electrolytic battery/fuel cell media to reduce the possibility of cabin atmosphere exposure	TRM-34 thru TRM-47
LL-01-04		Failure to remove Smoke/Fire by-products	Injury or loss of crew member	FDSS; Emergency O2 supply; Material selection; Redundant cabin venting system.	TRM-34 thru TRM-47
LL-01-05		Loss of CO2 removal capability	Loss of Crew/Vehicle	Redundant CO2 Removal System during launch and re-entry; EVA back-up capability	TRM-34 thru TRM-47
LL-01-06		Toxic Environment in the LL pressurized volume	Injury or death to crewmember	Materials Selection; Emergency O2 supply; Redundant cabin venting system with a manual override; Atmospheric monitoring	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-01-07		Leakage of fuel from Propulsion/RCS system into the LL pressurized volume	1) Toxic effects to crew 2) Increased likelihood of explosion	1) All propulsion lines for forward RCS will be outside habitable volume to eliminate the possibility of fuel contamination 2) Trace Contaminant Control system for detection of propulsion fuel and procedures to isolate leak	TRM-34 thru TRM-47
LL-01-08		Leakage of Human Byproducts from the WCS	Possible injury to crew	Adequate waste containment system with redundancy; Adequate crew procedures and equipment for isolation and containment	TRM-34 thru TRM-47
LL-01-09		Inadequate protection from shattering or containment of shatterable material allows release of debris in habitable environment	Crew exposed to particulate contamination	All shatterable material is provided with positive protection to prevent fragments from entering the habitable environment.	TRM-34 thru TRM-47
LL-01-10		Tool/Science/ Equipment Battery Leakage	Respiratory, skin, eye irritation; Mucous membrane irritation	Battery design; Adequate leakage containment	TRM-34 thru TRM-47
LL-01-11		Leakage of TCS media into the LL pressurized volume	Respiratory irritation; Mucous membrane irritation	ECLSS; Adequate detection of leakage; Adequate storage of spares/ replaced media	TRM-34 thru TRM-47
LL-01-12		Post landing venting/ingestion	Injury or death to crewmember	Adequate monitoring to arrive safe environment	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-02-01	**Electrical Shock**	Inadequate grounding of surfaces accessible to the crew	Injury or death to crewmember	Design; Testing; Redundancy; Proper procedures	TRM-34 thru TRM-47
LL-02-02		Lunar Lander Static Discharge	Injury or death to crewmember	Adequate measures for controlling potential static discharges; Proper insulation	TRM-34 thru TRM-47
LL-02-03		Lunar Lander Short Circuit	Injury or death to crewmember	Circuit breakers; Adequate grounding; Sufficient insulation; Design	TRM-34 thru TRM-47
LL-02-04		Improper Circuit/Equipment Design on the Lunar Lander	Injury or death to crewmember	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design	TRM-34 thru TRM-47
LL-03-01	**Environmental Hazards**	Treacherous Lunar Surface	Injury or death to crewmember	Accepted Risk; adequate crew training	TRM-40
LL-03-02		Excessive Thermal Conditions inside the LL pressurized volume	Exceed lower or upper thermal limit of crew/vehicle components	Adequate Passive Thermal Control System (PTCS); Active TCS (ATCS)	TRM-34 thru TRM-47
LL-03-03		LL exposed to Excessive External Thermal Conditions while on the Lunar surface	Exceed lower and upper thermal limits of the crew when exposed to extreme temperatures	Adequate survival equipment. Stand-alone suit cooling.	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-03-04		Excessive Noise within the LL pressurized volume	Physiological and psychological effects on crew	Incorporate a passive acoustic abatement system; Hearing protection used in areas of high noise generation; System will be tested to ensure the vehicle meets NASA-STD-3000 requirements.	TRM-34 thru TRM-47
LL-03-05		Excessive radiation exposure to the crew	Long-term Crew Health; Carcinoma	Accepted Risk, minimum radiation protection by design; Adequate monitoring of solar activity	TRM-34 thru TRM-47
LL-03-06		Inadequate/ inappropriate lighting in habitable volume	Physiological and psychological effects on crew	Permanent general lighting and portable task lighting installed on the vehicle; Emergency lighting	TRM-34 thru TRM-47
LL-03-07		Sharp Edges/Pinch Points inside and outside the LL	Possible Injury to crewmember	Hardware designed where they will not pinch or snag the crew or their clothing. Exposed surfaces should be smooth and free of burrs	TRM-34 thru TRM-47
LL-03-08		Water leak in the LL pressurized cabin	Possible Injury to crew; Physiological and psychological effects on crew	Redundant valves and piping to control water leaks; fracture controls for water tank (leak before burst)	TRM-34 thru TRM-47
LL-03-09		Repressurizing the LL airlock post EVA allows hazardous gas ingestion	Loss of Crew/Vehicle	Repressurize with LL supplied consumables; Use non-toxic OMS/RCS propellants for descent.	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-04-01	**Fire or Explosion**	Improper Circuit Design causing a fire inside the LL	Loss of Crew/Vehicle	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design, FDSS	TRM-34 thru TRM-47
LL-04-02		Improper power connector design that does not preclude improper mismate / demate.	Loss of Crew/Vehicle	All power connectors are designed such that they cannot be mismated or cross-connected.	TRM-34 thru TRM-47
LL-04-03		Inadvertant OMS/ACS liquid propellant explosion	Loss of Crew/Vehicle	Pressure relief valves; Leak before burst design; Adequate shielding	TRM-34 thru TRM-47
LL-04-04		Use of Flammable Materials within the LL pressurized volume	Loss of Crew/Vehicle	Design in accordance with manned space flight Material Selection Requirements, Fire Detection System (FDS)	TRM-34 thru TRM-47
LL-04-05		High Pressure Vessel rupture on the LL descent and/or ascent stage(s)	Loss of Crew/Vehicle	High pressure vessels will be designed to leak before bursting by material selection/properties; Positive Pressure Relief Valve (PPRV) on pressure vessel and Vehicle Cabin; FDSS	TRM-34 thru TRM-47
LL-04-06		Ignition Source malfunction(s)	Loss of Crew/Vehicle	Preclude ignition sources by design; Material selection; FDSS; Safeing and arming circuitry	TRM-34 thru TRM-47
LL-04-07		High concentration of Oxygen within the LL pressurized volume	Increased flammability of materials	Redundant O2 Partial Pressure sensing and control, Material selection; FDSS	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-05-01	**Impact/Collision**	LL Collision with Lunar Surface	Loss of Crew/Vehicle	Design for abort during descent; redundant altimeter and surface scanner	TRM-40
LL-05-02		Loss of Lunar Lander attitude control	Loss of Crew/Vehicle	Two-fault tolerant RCS; RCS shall be designed in accordance with the Human Rating Requirements	TRM-34 thru TRM-47
LL-05-03		Impact of Rotating or moving equipment within the LL pressurized volume	Injury or death to crewmember	Design and crew procedures.	TRM-34 thru TRM-47
LL-05-04		Inadequate positive backout prevention for safety critical fasteners results in structural damage.	Injury or death to crewmember	All system safety critical fasteners will be designed to prevent backout.	TRM-34 thru TRM-47
LL-05-05		Loss of LL control during proximity operations with CEV post Lunar surface mission	Potential loss of crew, vehicle, or Injection Stage	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture. 2) Procedures for safe proximity operations will be maintained to minimize potential for collision. 3) Crew has option of manual control	TRM-46 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-05-06		Lunar Lander Collision with CEV	Loss of Crew/Vehicle	1) Redundant vehicle systems controlling attitude, translation, monitoring of range, range rate, capture. 2) Procedures for safe proximity operations will be maintained to minimize potential for collision. 3) Crew has option of manual control	TRM-46 thru TRM-47; TRM-49
LL-05-07		Lunar Lander Impact with MMOD	Loss of Crew/Vehicle	Accepted risk or MMOD protection designed to shield CEV/Lunar Lander/Injection Stages or at least the critical systems	TRM-34 thru TRM-47
LL-05-08		Inadequately restrained equipment in Habitable Volume	Loss of Crew/Vehicle	1) Adequate design of restraints 2) Adequate crew procedures for stowage of items	TRM-34 thru TRM-47
LL-05-09		Use of non-conforming fasteners results in release of hardware components	Injury to crewmember	All fasteners conform to an approved fastener integrity program.	TRM-34 thru TRM-47
LL-05-10		Inadequate hardware design results in structural damage	Crew exposed to debris/shrapnel or other hazardous condition as result of structural failure of hardware	System design to provide positive margins of safety under all loading conditions including crew handling, on-orbit vibration with respect to the required safety factors.	TRM-34 thru TRM-47
LL-05-11		Landing in an unfriendly terrain environment	Injury or death to crewmember	Adequate crew restraints; Adequate structural integrity	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-05-12		Landing loads cause damage to the LL egress hatch mechanisms	Blocked egress path; Possible injury or death of crewmember	Redundant egress path; Emergency pyros for the hatch	TRM-34 thru TRM-47
LL-05-13		Crew restraint system fails	Injury or death to crewmember	Adequate and redundant crew restraints as defined in NASA-STD-3000, 5.3	TRM-34 thru TRM-47
LL-05-14		The LL landing mechanism fails to deploy	Injury or death to crewmember	Redundant landing mechanism	TRM-34 thru TRM-47
LL-05-15		EVA Crewmember or equipment impact with LL	Loss of Crew/Vehicle	All EVA crewmembers and equipment tethered; EVA translation paths on LL clearly defined as in NASA-STD-3000; Robust LL TPS system	TRM-34 thru TRM-47
LL-06-01	**Loss of Habitable Environment**	Loss of O2 Supply in the LL pressurized volume	Loss of Crew/Vehicle	Redundant O2 Partial Pressure Supply, Sensing and Control with backup procedures to use EVA suit.	TRM-34 thru TRM-47
LL-06-02		Loss of TCS within the LL pressurized volume	Loss of Crew/Vehicle	Redundant loop TCS system	TRM-34 thru TRM-47
LL-06-03		Toxic Environment in the LL pressurized volume	Injury or death to crewmember	Materials Selection; Emergency O2 supply; EVA crew member procedures to assure decontamination prior entering RLL habitable volume; Experiments/Payloads meet standard safety requirements; TCCS	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-06-04		LL sustains a Loss of Power	Loss of Crew/Vehicle	Two fault-tolerant power system	TRM-34 thru TRM-47
LL-06-05		Inadequate/ inappropriate lighting in habitable volume	Physiological and psychological effects on crew	Acceptable/adequate lighting design on the CEV and Lunar Lander; Emergency lighting	TRM-34 thru TRM-47
LL-06-06		Leakage of fuel from Propulsion/RCS system into the LL pressurized volume	1) Toxic effects to crew 2) Increased likelihood of explosion	All propulsion lines for forward RCS will be outside habitable volume to eliminate the possibility of fuel contamination; TCCS for detection of propulsion fuel and procedures to isolate leak; Emergency O2 supply; Prop/RCS relief valves	TRM-34 thru TRM-47
LL-06-07		Inability to adequately vent pressurized volume	Injury or death to crewmember	Redundant electro-mechanically actuated vent and fan; Manual override	TRM-34 thru TRM-47
LL-06-08		Loss of CO2 removal capability	Loss of Crew/Vehicle	Redundant CO2 Removal capability with back up procedures to use EVA suit.	TRM-34 thru TRM-47
LL-06-09		High concentration of Nitrogen within the LL pressurized volume	Injury or Loss of Crew	Redundant N2 Partial Pressure sensing and control	TRM-34 thru TRM-47
LL-06-10		ACS propellant leakage into the LL pressurized volume	Injury or death to crewmember	ACS relief valves; Propellant leakage sensors; Cabin venting; Emergency O2 supply; ACS is outside the habitable volume	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-06-11		Helium contamination of Oxygen supply	1) Crew suffocation with high concentrations of Helium 2) No toxic effects to crew with low concentrations of Helium	Helium tanks used for pressurizing the H2O2 will be outside of the habitable volume.	TRM-34 thru TRM-47
LL-06-12		Lunar Lander Compartment depressurization	Loss of Crew/Vehicle	1) Adequate MMOD protection through design. 2) Adequate resources for cabin pressurization in the event of a critical leak. 3) Adequate structural design to prevent excessive leakage in habitable environment. 4) Docking vestibule pressure checked prior to opening CEV hatch. 5) Backup procedures to use EVA suit.	TRM-34 thru TRM-47
LL-07-01	**Physiological/ Psychological**	Acceleration, shock, impact & Vibration	Injury or death to crewmember	Adequate design of crew/equipment restraints; Adequate crew procedures for stowage of items	TRM-34 thru TRM-47
LL-07-02		Effects of Pressure Changes on Crew	Possible Injury to crewmember	Adequate crew safety procedures for EVA pre-breath	TRM-34 thru TRM-47
LL-07-03		Illness/Incapacitation of Crew Member during Lunar surface operations	Injury or death to crewmember	Crew Health equipment and procedures	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-07-04		Excessive Noise inside the LL pressurized volume	Possible Injury to crewmember	Incorporate a passive acoustic abatement system; Hearing protection used in areas of high noise generation; System will be tested to ensure the vehicle meets NASA-STD-3000 requirements.	TRM-34 thru TRM-47
LL-07-05		Sharp Edges/Pinch Points inside and outside the LL	Possible Injury to crewmember	Hardware designed where they will not pinch or snag the crew or their clothing; Exposed surfaces should be smooth and free of burrs; Adhere to NASA-STD-3000 requirements.	TRM-34 thru TRM-47
LL-07-06		EVA Workloads & Fatigue	Possible Injury to crewmember	Crew procedures established to minimize crew fatigue	TRM-34 thru TRM-47
LL-07-07		Interference with Translation Paths. Hardware impinges into translation paths.	Possible Injury to crewmember	Hardware designed to comply with traffic flow and translation paths; Adequate volume provided for a suit egress as stated in NASA-STD-3000.	TRM-34 thru TRM-47
LL-07-08		Appendage Entrapment in Holes or Latches	Possible Injury to crewmember	Holes and latches meet NASA-STD-3000 design requirements designed to prevent entrapment of crew member's appendage.	TRM-34 thru TRM-47
LL-07-09		Loss of H2O supply	Possible injury or death to crewmember	Redundant water supply	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-07-10		Excessive Internal Thermal conditions within the LL	Possible injury or death to crewmember	Adequate passive thermal control system which will not allow an excessive amount of heat in at anytime	TRM-34 thru TRM-47
LL-07-11		Crew sustains High G-Loads while on Lunar Lander	Possible Injury to crew; Physiological and psychological effects on crew	Full pressure suits; Recumbent seats; Trajectory & propulsion system to operate within human limits	TRM-34 thru TRM-47
LL-07-12		Inadequate size of the LL's habitable volume	Physiological and psychological effects on crew	Design for adequate amount of habitable volume for crew accommodation guidelines	TRM-34 thru TRM-47
LL-07-13		Inadequate/ inappropriate lighting in the LL habitable volume	Physiological and psychological effects on crew	Acceptable/adequate lighting design. LED's are a proposed solution for lighting.	TRM-34 thru TRM-47
LL-07-14		Excessive Lunar Thermal Conditions post crew egress	Possible injury or death to crewmember	Survival equipment will be tethered onto suit. Stand-alone suit cooling.	TRM-34 thru TRM-47
LL-08-01	**Loss of Vehicle Control**	Loss of LL Guidance, Navigation, & Control	Loss of Crew and Vehicle	Redundant NAV systems	TRM-34 thru TRM-47
LL-08-02		Loss of LL Attitude Sensing	Loss of Crew and Vehicle	Redundant attitude sensing system	TRM-34 thru TRM-47
LL-08-03		Loss of LL Attitude Control	Loss of Crew and Vehicle	Redundant attitude control systems	TRM-34 thru TRM-47
LL-08-04		Loss of LL Central Command and Control Capability	Loss of Crew and Vehicle	Redundant General Purpose Computers to perform all vehicle functions.	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-08-05		Loss of LL Delta-V Capability	Loss of Crew and Vehicle	1) 2 fault-tolerant propulsion system 2) The system contains an engine out capability for all injection/maneuvers 3) Redundant fuel and oxidizer supply to each engine 4) Redundant ignition at each engine	TRM-34 thru TRM-47
LL-08-06		LL Landing mechanism(s) fail to deploy	Loss of Crew and Vehicle	Redundant landing mechanism	TRM-40
LL-08-07		LL crew becomes Incapacitated during the mission profile	Loss of Crew and Vehicle	Autonomous crew escape system and mechanisms; Autonomous crew rendezvous and capture system; Autonomous descent approach and landing systems	TRM-34 thru TRM-47
LL-08-08		LL health monitoring system(s) fails	Loss of Crew and Vehicle	Redundant system health monitoring	TRM-34 thru TRM-47
LL-09-01	**Radiation**	Crew is exposed to Solar Flare while on the Lunar Lander	Injury or death to crewmember	Accepted Risk, Adequate monitoring of solar activity, maximum radiation protection	TRM-34 thru TRM-47
LL-09-02		LL Crew and/or hardware is exposed to Non-Ionizing Radiation	Injury or death to crewmember	Minimize radiation emittance and maximize protection of components sensitive to EMI.	TRM-34 thru TRM-47
LL-09-03		LL crew and/or hardware is exposed to Ionizing Radiation	Injury or death to crewmember	Minimize sources of ionizing radiation to be used in the LL design	TRM-34 thru TRM-47

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-10-01	**Lunar Surface EVA Operations**	Inability to return to Crew Habitat	Injury or death to crewmember	Crew EVA rescue procedures in place to secure injured crew member back to habitat	TRM-40
LL-10-02		Crew Injury during nominal surface EVA	Injury or death to crewmember	Crew CHeCs medical equipment	TRM-40
LL-10-03		Contamination of EVA crewmember from leaking LL RCS/Engine Thruster(s)	Possible Injury to crewmember	Isolation valves for upstream manifold and tanks.	TRM-40
LL-10-04		EVA crewmember exposed to Sharp Edges/Pinch Points inside the airlock and outside the LL	Possible Injury to crewmember	Hardware designed where they will not pinch or snag the EVA suit. Exposed surfaces are smooth and free of burrs	TRM-40
LL-10-05		EVA crewmember exposed to Excessive Non-Ionizing Radiation	Possible Loss of EVA Crew Member	Redundant inhibits to ensure power is isolated from RF Amps and Electro-Magnets	TRM-40
LL-10-06		Solar Flare/Ionizing cosmic Radiation during a nominal surface ops EVA	Injury or death to crewmember	Safe Haven for radiation protection by design; Adequate monitoring of solar activity	TRM-40
LL-10-07		Crew member collision with other elements attached to the Lunar Lander.	Possible Loss of EVA Crew Member	1) The worksite is clearly defined to minimize the possibility of an EVA crew member being struck by an object. 2) Adequate tools, equipment, and lighting for the safe performance of planned tasks.	TRM-40

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-10-08		EVA crewmember exposed to Ionizing Radiation	Possible Long-term Injury to EVA Crew Member	No sources of ionizing radiation will be used in the CEV design	TRM-40
LL-10-09		Excessive EVA touch temperatures during contingency operations	Possible Loss of EVA Crew Member	Adequate design to meet touch temperature requirements	TRM-40
LL-10-10		Inadequate design for EMU to handle deep space environment (temps, radiation, MM, solar dust).	Possible Loss of EVA Crew Member	Adequate design for EMU to handle deep space environment.	TRM-40
LL-10-11		EVA crewmember receives Electrical Shock	Possible Loss of EVA Crew Member	Adequate Circuit Design and Grounding	TRM-40
LL-10-12		EVA crewmember is exposed to Static Discharge	Injury or death to crewmember	Adequate measures for controlling potential	TRM-40
LL-10-13		Inadequate Restraints for EVA crewmember	Possible Loss of EVA Crew Member	Establish EVA Worksites, Pathways, Handholds, and Tether Attachment Points	TRM-40
LL-10-14		Insufficient working volume	Inability to Maintain Payload	Adequate Design for EVA Maintenance or Servicing	TRM-40
LL-10-15		EVA crewmember is Inadequately grounded	Injury or death to EVA crewmember	Design; Testing; Redundancy	TRM-40
LL-10-16		EVA crewmember is exposed to Improper Circuit Design	Injury or death to EVA crewmember	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design	TRM-40
LL-10-17		EMU encumbrances	Inability to Maintain Payload	Adequate Design for EVA Maintenance or Servicing	TRM-40

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS	Associated Critical Event ID #'s
LL-10-18		EVA crewmember suffers MMOD impact	Possible Loss of EVA Crew Member	MMOD protection designed to shield EVA suit	TRM-40
LL-11-01	**Inability to Dock/Transfer Crew /Undock**	Inability for LL to Dock with CEV	Loss of Crew/Vehicle	1) Redundant Docking systems to achieve docking 2) Backup procedures to depressurize LL and perform contingency EVA	TRM-32; TRM-47
LL-11-02		Inability to equalize pressure between the CEV & Lunar Lander post surface mission	Loss of Crew/Vehicle	Hatches incorporate redundant pressure equalization valves.	TRM-32 thru TRM-33; TRM-47 thru TRM-48
LL-11-03		Inability for CEV to undock from Lunar Lander post Lunar surface mission	Loss of Crew/Vehicle	1) Docking system is two fault tolerant for undocking 2) Procedures for contingency EVA to undock vehicles	TRM-47
LL-11-04		Structural failure of docking mechanism	Loss of Crew/Vehicle	Mechanisms designed to structural margin	TRM-32; TRM-47

Table 20.17.1-3: Lunar Lander Preliminary Hazard Analysis

20.17.2 Programmatic and Subsystem Risks

In addition to the preliminary hazards, programmatic and subsystem risks were documented. Example subsystems included, but are not limited to, Avionics, Propulsion, Extravehicular Activity (EVA), and Environmental Control and Life Support System (ECLSS). As with the preliminary hazards, risks that were documented in previous exploration studies and the 2nd Generation Reusable Launch Vehicle (2GRLV) Space Launch Initiative (SLI) were compiled into a single document. Due to some risks being hazards, S&MA attempted to not have any overlap between the two. To ensure there was no overlap, it was decided that if the crew could be injured or lost the risk would become a hazard. If the risk did not result in injuring or losing a crewmember, it would documented as either a programmatic or subsystem risk.

Once all the risks were combined into a single document, they were thoroughly reviewed for content and applicability. Like the hazards, the TRM critical events were mapped back to each risk showing when in the mission profile they would need to be mitigated. The elements (CEV, CEV Injection Stage, Lunar Lander, and Lunar Lander Injection Stage) that were affected by each risk were identified as well. The risks are listed below and divided out by subsystem.

Avionics and Communication		
Affected Elements	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	Given the current maturity level of avionics and communication technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-56
All	Given a loss of avionics and communications system capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-01 thru TRM-56
CEV; CEV Injection Stage	Given the short amount of time required to activate an abort system during a catastrophic failure event; there is the possibility the crew escape capability will not have an adequate fault detection and warning capability to activate required systems to assure crew survival.	TRM-01 thru TRM-56
CEV; CEV Injection Stage	Given a loss of controlled flight of the primary vehicle; the crew may not be able to initiate a mission abort.	TRM-01 thru TRM-56
CEV; Lunar Lander; CEV Injection Stage	Given that there is a loss of instrumentation, there is a possibility of not detecting a system failure to initiate the mission abort.	TRM-01 thru TRM-56
All	Radiation environments may damage equipment/software; an inability to properly activate the Crew Exploration Vehicle may occur.	TRM-01 thru TRM-56
All	If there is electrostatic discharge causing failure to hardware; there may be a loss of avionics and control.	TRM-01 thru TRM-56
CEV	If the avionic architecture concept uses INS/GPS, the contractor must perform major modifications since the integrated box was originally designed for terrestrial applications. The INS/GPS blended solution may limit access to other GPS vendors as well. There are some risks in accepting the claim that the INS/GPS integrated solution it is a true tightly coupled self-calibrating system that requires no lab or pre-launch calibration.	TRM-01 thru TRM-56
CEV; Lunar Lander	If there is electrostatic discharge causing failure to hardware; then there may be a loss of communication.	TRM-01 thru TRM-56

Avionics and Communication		
Affected Elements	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	Given the current maturity level of avionics, communications technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-56
All	Given a loss of avionics and communications capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-01 thru TRM-56
All	If there is electrostatic discharge causing failure to hardware; then there maybe loss of communication and control.	TRM-01 thru TRM-56
Lunar Lander	Loss of critical avionics systems could result in a hard landing or a landing involving an environmental hazard; damage to landing legs and other external structure may result. Landing leg damage could resulting damage to engines, loss of egress ability, loss of EVA equipment, damage to pressurized cabin, or injury to the crew.	TRM-41
Lunar Lander	A lack of avionics imagery sensing components may cause Lunar Lander damage for one out of TBD Lunar landings. Typically landing radars do not produce imagery of the landing location. If typical radar or laser altimeters are used to sense altitude only, their major terrain features, such as boulders, could be missed and cause landing damage.	TRM-41
Lunar Lander	Use of the Lunar Lander LADAR system while crewmembers are on the Crew Exploration Vehicle (while docked) may cause permanent eye-damage or blindness to the crew.	TRM-41

Table 20.17.2-1: Avionics and Communication Risks

Environmental Control Life Support System		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given the unknown oxygen supply for CEV; there is a possibility the system will not be sized correctly to supply adequate O2 leading to injury and loss of crew	TRM-21 thru TRM-56
CEV	Given a failure results in the loss of breathable atmosphere in the CEV; there is possibility of injury or loss of crew (crew cabin; and flight suit).	TRM-21 thru TRM-56
CEV	Given a failure that leads to a rupture or explosion of an internal pressure vessel; there is a possibility of injury or loss of crew.	TRM-21 thru TRM-56
CEV	Given a loss of crew H_2O; there is a possibility the crew may become sick due to dehydration.	TRM-21 thru TRM-56
CEV	Given that there are overboard venting failures; there is the possibility of the crew suffocating.	TRM-21 thru TRM-56
CEV	Given that there is a loss of pressurization for the habitable volume due to an ECLSS system failure; there is a possibility of losing the crew.	TRM-21 thru TRM-56
CEV	Given the presence of a toxic environment (internal or external) due to the malfunction or failure of a CEV subsystem; there is possibility for injury or loss of crew.	TRM-21 thru TRM-56
CEV	Given the crew may have to remain in the CEV after an abort landing with an inadequate waste collection system could lead to crew exposure to human waste (fecal, vomit, and urine).	TRM-56
CEV	Given the CEV is exposed to a mission induced toxic environment post-landing once the cabin has been vented; there is a possibility of injury or loss of crew due to toxic exposure.	TRM-56
CEV	Given a failure in the flight suit's ability to provide breathable oxygen during pre-launch and H2O landing scenarios; the crew may not have breathable air resulting in injury or loss of the crew.	TRM-56
CEV	Given the crew may have to remain in the CEV after landing contamination of the crew food and water supplies could lead to crew dehydration and weakness	TRM-56
CEV	Given a failure results in the loss of breathable atmosphere in the CES; there is possibility of LOC.	TRM-21 thru TRM-56

Environmental Control Life Support System		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given a failure resulting in a loss of CES thermal control; the possibility exists the crew will be exposed to excessive temperatures leading to injury or loss of crew	TRM-21 thru TRM-56
CEV	Given a failure that leads to a rupture or explosion of an internal pressure vessel; there is a possibility of injury or LOC.	TRM-21 thru TRM-56
CEV	Given a loss of crew H2O; there is a possibility the crew may become sick due to dehydration.	TRM-21 thru TRM-56
CEV	A failure of the overboard venting system to remain closed during flight; may result in crew suffocation.	TRM-21 thru TRM-56
CEV	Given that there is a loss of pressurization from the ECLSS system; there is a possibility of losing the crew.	TRM-21 thru TRM-56
CEV	Given a failure in the flight suit's ability to provide breathable oxygen during pre-launch and H2O landing scenarios; the crew may not have breathable air resulting in injury or loss of the crew.	TRM-21 thru TRM-56
CEV; Lunar Lander	Given the current maturity level of ECLSS system technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-21 thru TRM-56
CEV; Lunar Lander	Given a loss of ECLSS system capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-21 thru TRM-56
CEV	Given the presence of a toxic environment (internal or external); there is possibility for injury or loss of crew.	TRM-21 thru TRM-56
CEV	Given a failure leading to a fire in the crew compartment of the CM; there is the possibility for injury or LOC.	TRM-21 thru TRM-56
CEV	Given the crew may have to remain in the CM after landing inadequate waste collection system could lead to crew exposure to human waste (fecal, vomit, and urine).	TRM-56
CEV	Given the crew may have to remain in the CM after landing contamination of the crew food and water supplies could lead to crew dehydration and weakness.	TRM-56

Environmental Control Life Support System		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given the CM is exposed to a mission induced toxic environment post-landing once the cabin has been vented; there is a possibility of injury or loss of crew due to toxic exposure.	TRM-56
Lunar Lander	A micrometeoroid or orbital debris impact may penetrate the pressure vessel leading to cabin depressurization, system failure, or crew injury/death.	TRM-34 thru TRM-48
Lunar Lander	Given the possibility of transferring lunar dust into the lunar lander; adverse effect on the Landers critical life support systems could be affected.	TRM-41
Lunar Lander	If lunar dust contaminates the habitable volumes of the Lunar Lander; then health and crew performance could be negatively impacted.	TRM-41
CEV, Lunar Lander	The mission may operate cabin pressure at less than atmospheric; there could be an enhanced risk of fire, outgassing, an less time of consciousness if decompression occurs.	TRM-34 thru TRM-48

Table 20.17.2-2: ECLSS Risks

EVA / LES		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV; Lunar Lander	Given the current maturity level of EVA system technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-41
CEV; Lunar Lander	Given a loss of EVA system capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-41
CEV	Given the minimum volume required for four spacesuits, tools, equipment, and ORM's; there is a possibility that there may not be sufficient space inside the CEV and in the payload bay.	TRM-21 thru TRM-56
Lunar Lander	If lunar dust contaminates the habitable volumes of the RLL; then health and crew performance could be negatively impacted. Lunar dust has sharp edges and can be inhaled or get in the crewmembers' eyes causing discomfort or damage to the lungs and eyes.	TRM-41
Lunar Lander	If the translation path from the crew cabin to the lunar surface is blocked due to improper landing; then the EVA operations timeline could be affected and objectives may not be met. A poor translation path is hazardous to the EVA crewmember and could be a problem for transporting equipment, tools, and samples during egress and ingress.	TRM-41
CEV; Lunar Lander	Given a loss of crew H2O; there is a possibility the crew may become sick due to dehydration.	TRM-20 thru TRM-56
CEV	Given a failure in the flight suit's ability to provide breathable oxygen during pre-launch and H2O landing scenarios; the crew may not have breathable air resulting in injury or loss of the crew.	TRM-20, TRM-55, TRM-56
CEV	A lack of communication with the ground/SAR; may delay the crew rescue operations. After separation before landing	TRM-56
CEV	Given a failure of the crew H2O flotation or survival gear, the possibility exists that the crewmember will not be able to survive after egress from the CEV.	TRM-56

EVA / LES		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV; Lunar Lander	Given the unknown external crew environments there is a possibility the crew will not be protected from environmental extremes	TRM-21 thru TRM-56
CEV; Lunar Lander	Given a loss of lighting in the cabin; there is the possibility the crew will be injured	TRM-21 thru TRM-56

Table 20.17.2-3: EVA/LES Risks

Power		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	Given the current maturity level of power system technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-56
All	Given a loss of power system capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-21 thru TRM-56
All	Given the power profile that the electrical power subsystem is designed to; there is a possibility that many electrical loads were underestimated or omitted which require a redesign	TRM-01 thru TRM-56
CEV, Lander	Given a loss of lighting in the cabin; there is the possibility the crew will be injured	TRM-21 thru TRM-56
All	If there is electrostatic discharge causing failure to hardware; then there maybe loss of communication and control.	TRM-21 thru TRM-56
All	Given the total loss of electrical power; all subsystems that rely on electrical power will experience an immediate total or partial failure, dependent on their needs.	TRM-21 thru TRM-56
CEV	The CEV electrical power systems electrical energy is utilized by most of the other CEV subsystems and is used from the first CEV activation sequences to the end of the mission	TRM-21 thru TRM-56
All	If electrical power is lost during any portion of the CEV mission; loss of mission and crew could result.	TRM-21 thru TRM-56
All	The power system may be undersized because of insufficient knowledge of actual power demands.	TRM-21 thru TRM-56
CEV, Lunar Lander	Swelling and rupture of the battery cells may lead to a small voltage drop, reduction in capacity, and eventually an open circuit over time.	TRM-21 thru TRM-56
CEV, Lunar Lander	An unimproved water separator for the PEM Fuel Cells on the O2 side could lead to a lower overall working efficiency. A flooding condition covering the reacting surfaces would be created. This is irreversible and would cause a loss of battery.	TRM-21 thru TRM-56

Table 20.17.2-4: Power Risks

Propulsion		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	Given the current maturity level of propulsion system technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-56
All	Given a loss of propulsion system capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-21 thru TRM-56
All	If the mass properties are not checked often; the quantity of fuel and hardware could drive the mass of the vehicle beyond the limitations of the design. Exceeding the mass limitations of the vehicle could result in a delay in schedule.	TRM-01 thru TRM-56
CEV; CEV Injection Stage	Given the CEV payloads are TBD: there is a possibility that the vehicle design will not be able to identify and integrate requirements to satisfy delivery of certain payloads to a given target	TRM-21 thru TRM-56
All	Frozen propellant in lines or around valves could cause propellant lines to rupture, propellant to be spilled and possibly even explosive conditions in closed compartments. Decomposition and thermal run away are also a concern, which could cause catastrophic failures of the system. Space vehicles in fixed orientations on orbit will experience environments which, in all likelihood, will require active thermal controls for propulsion subsystems. If temperatures in areas that remain shadowed drop below propellant freezing levels, tank lines and valves could experience propellant icing. Iced up lines could burst or eject frozen elements from rocket nozzles. Heaters could prevent such occurrences, if properly placed and operated. Beside low temperature conditions, depending on the propellant combination, decomposition and thermal runaway can also high pressures, line or pressure vessel failures.	TRM-01 thru TRM-56
CEV	Given a failure that prevents the CM from separating from the CEV; there is a possibility for injury or LOC. Failure to mechanically separate from the CEV	TRM-21 thru TRM-56

Propulsion

Affected Element(s)	Risk Statement	Associated TRM Critical Event ID #'s
CEV	Given a loss of capability for the CEV to release/ activate escape mechanism; there is the possibility the CM will not separate from the SM leading to injury or LOC. Pyro-tech; escape rocket; mechanism; flotation bags, etc.	TRM-21 thru TRM-56
CEV	If the CEV tractor rocket fails to operate at full potential; the CM may not be able to reach a safe distance from the blast wave.	TRM-19, TRM-20, TRM-21
All	Given a failure that leads to a rupture or explosion of an internal pressure vessel; there is a possibility of injury or LOC.	TRM-19 thru TRM-49
All	Given that there are no fuel leak prevention and clean-up provisions; the type and quantity of fuel could adversely affect the environment.	TRM-01, TRM-05, TRM-19
All	If the propulsion system is packaged near a potentially high energy system; there could be a loss of vehicle due to explosion. The propulsion system is subjected to loads and forces which could trigger an explosion.	TRM-01 thru TRM-56
All	Absence of an effective hazardous gas purge system could result in buildup of an explosive gas combination behind a bulkhead which could seriously damage the vehicle in flight. In the case of the Shuttle Orbiter, there is a need to purge compartments of trapped gases such as hydrogen and oxygen. With a crew transfer vehicle separate from the LV booster vehicles, there should not be much linkage to the main propulsion system's), but hazardous gases could still build up. Examples are leaking oxidizer and fuel for the orbital maneuver and attitude control thrusters, propellants that could power auxiliary power units (e.g., hydrogen peroxide, hydrazine). Oxygen would be subject to leaking or boil-off and monopropellants could decompose and vent. In closed compartments, there must be measures to prevent buildup of explosive combinations.	TRM-01, TRM-05, TRM-19

Propulsion		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV; Lunar Lander	Thrusters sized simply for performing the maximum magnitude required burn could result in damage to the station or other rendezvous objectives. In the case of existing visiting vehicles to the International Space Station, there is a need to protect the station from unnecessary pluming with attitude control thruster firings. For example, the Orbiter achieves low Z thruster maneuvers by firing a number of indirectly oriented thrusters that cancel x and y components, but result in a small net z thrust. In effect they are canted away from station. But, beside the portions of portion in the direct path of a visiting vehicle, there are objects such as solar panels and radiators off to the sides which should be protected as well. Means to such end could be indirectly pointed thrusters, thrusters firing at low levels or cold gas thrusters.	N/A
CEV	Given that the Booster fails to shut down/ fly the predicted trajectory; the vehicle can make re-contact with the CEV. Separation went good, but the vehicle comes back	TRM-21, TRM-22
CEV	If CES propulsion system is not adequately designed; then the crew may not be able to reach a safe distance.	TRM-21 thru TRM-56
CEV	Given an inadvertent initiation of the CES; there is the possibility of loss of vehicle and mission. Pre-launch and Post-landing	TRM-21 thru TRM-56
All	Given the current maturity level of propulsion system technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-56
All	Given a loss of propulsion system capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-21 thru TRM-56
CEV	Given that the systems required to separate the CEV from the LV have not been defined; there is a possibility the CES would be damaged during separation.	TRM-21 thru TRM-56
CEV	Separation of the CES below a survivable altitude; may result in a loss of crew. Inability to deploy chutes	TRM-21 thru TRM-56

Propulsion		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given the condition that a new propulsion system could fail shortly after liftoff; the CEV vehicle could loose control and crash shortly after liftoff.	TRM-21, TRM-22
CEV; CEV Injection Stage; Lunar Lander	Given a failure that leads to a rupture or explosion of an internal pressure vessel; there is a possibility of injury or LOC.	TRM-21 thru TRM-56
Lunar Lander	Lunar dust getting into critical engine components during and following lander touchdown, may cause engine failures leaving the crew stranded on the lunar surface.	TRM-40, TRM-41
Lunar Lander	Lack of definition from the customer on the fluid transfer interface may lead to incompatibilities with the Lunar Lander	TRM-35 thru TRM-48
Lunar Lander	Given the LL main engines will face the lunar surface when landing; dust and debris may limit the reusability of the engines.	TRM-41
All	Insufficient analysis of the interplanetary and lunar surface thermal environment may mean the thermal control and propulsion systems are improperly designed.	TRM-35 thru TRM-48

Table 20.17.2-5: Propulsion Risks

Structures & Mechanisms		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	Given the current maturity level of structural and Mechanical technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-56
All	Given a loss of structural and mechanical system; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-01 thru TRM-56
CEV	If the CEV is top-mounted; then there is a possibility that it may not meet weight and frequency constraints of the launch vehicle.	TRM-21 thru TRM-33
CEV	Given that the blast environment is not defined; there is a possibility that the structure will not provide adequate crew protection.	TRM-21, TRM-22
CEV	Given that the load environments for the CEV are not defined; there is a possibility that the structure will be inadequately sized.	TRM-21, TRM-22
All	If flaws exist in materials, there is a possibility that they may fail prematurely	TRM-01 thru TRM-56
All	If fastener locking features fail, there is a possibility of structural failure	TRM-01 thru TRM-56
CEV	Given that corrosion damage may occur and be undetected; there is a possibility that the CEV structure will fail.	TRM-21 thru TRM-33
CEV	Given that fatigue damage may occur and be undetected; there is a possibility that the CEV structure will fail.	TRM-21 thru TRM-33
All	If access panels are inadequate in size or number, there is a possibility that all necessary routine maintenance/inspection will not be possible or be very costly.	TRM-21 thru TRM-33
CEV; Lunar Lander	Given that actuators and mechanisms can bind, seize, or jam; there is a possibility of loss of vehicle and LOC on launch and entry and loss of mission throughout all flight phases.	TRM-21 thru TRM-33
CEV; Lunar Lander	Given that there are excessive landing loads; there is the possibility of losing the crew.	TRM-55

Structures & Mechanisms		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV; Lunar Lander	Given that there may be high landing velocities; there is the possibility of structural failure leading to LOC and vehicle.	TRM-55, TRM-56
All	Given that there is a bird Strike; there is the possibility of having a Structure failure.	TRM-01, TRM-06, TRM-16, TRM-21
CEV	Given that the LV blast wave environment has not been fully defined; there is a possibility the CEV structure will not be designed adequately to protect the crew during a catastrophic explosion.	TRM-21, TRM-22
Lunar Lander	Because of the height of the cabin relative to the lunar surface, an injured crewmember may be unable to ingress into the lander.	TRM-41
Lunar Lander	Unknown/mischaracterized loads on the lander may lead to a structure design that is insufficient for actual flight.	TRM-35 thru TRM-48
CEV; Lunar Lander	Not fully understanding what the radiation environment is at lunar L1 may result in insufficient radiation protection for the crew and vehicle. Radiation exposure could lead to loss of crew or a loss of a critical subsystem.	TRM-31 thru TRM-51
Lunar Lander	All possible conditions at lander touchdown may not have been considered, possibly causing the crew to be stranded on the lunar surface. The vehicle may land on lunar terrain causing the lander to tip over.	TRM-41
Lunar Lander	A lack of avionics imagery sensing components may cause LL damage for one out of TBD lunar landings. Typically landing radars do not produce imagery of the landing location. If typical radar or laser altimeters are used to sense altitude only, their major terrain features, such as boulders, could be missed and cause landing damage.	TRM-41
CEV; Lunar Lander	Not fully understanding what the MMOD environment is at lunar L1 may result in insufficient MMOD protection for the vehicle.	TRM-31 thru TRM-50

Structures & Mechanisms		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV; Lunar Lander	If the translation path from the crew cabin to the lunar surface is blocked due to improper landing; then the EVA operations timeline could be affected and objectives may not be met. A poor translation path is hazardous to the EVA crewmember and could be a problem for transporting equipment, tools, and samples during egress and ingress.	TRM-41
Lunar Lander	Given the failure of LL landing gear deployment; an on-orbit EVA will be necessary to correct the landing gear. In order for the LL vehicle to be able to be used for a lunar mission, an EVA will need to occur to correct the deployment.	TRM-41
CEV; Lunar Lander	Not fully understanding what the thermal environment is at lunar L1 may result in insufficient insulation protection for the crew, vehicle, and components.	TRM-41, TRM-42

Table 20.17.2-6: Structures & Mechanisms Risks

TCS / TPS		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	Given the current maturity level of TCS and TPS technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-52
All	Given a loss of TCS and TPS system capability; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-01 thru TRM-56
All	Given a heat pump may be required to minimize radiator area and optimize the vehicle heat rejection capability; new development technology has not been proven in this application, failures in certification testing could result in a delay in alternate system design	TRM-01 thru TRM-56
All	Given the uncertainties involved in thermal analysis; heaters can be undersized, and insulation can be sized improperly.	TRM-01 thru TRM-56
All	If element requirements change, the thermal control system may be sized inadequately to satisfy these new requirements. Heaters, insulation, and other thermal control hardware will be sized for a particular design reference mission. However, these design requirements may change through the design process (e.g., extended missions, rescue vehicle, etc.).	TRM-01 thru TRM-56
All	If attitude selection is constrained, subsystem hardware may violate temperature limits, and subsystems may increase power requirements. On-orbit vehicle may require periodic attitude adjustment for thermal conditioning. Attitudes may be constrained by mission objectives or hardware failure.	TRM-01 thru TRM-56
All	If heaters within heater zones are mis-located or installed with incorrect power concentration, protected equipment may violate temperature limits. Heater configurations are designed so that all temperatures within a heater zone remain within temperature limits. In addition, the placement of heaters will not influence the performance of adjacent heater zones.	TRM-01 thru TRM-56

TCS / TPS		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	If a thermostat is incorrectly located, protected equipment may violate temperature limits. Thermostats are located so that the coldest location within a heater zone remains above lower temperature limits.	TRM-01 thru TRM-56
All	If subsystem components requiring thermal conductive relief are installed incorrectly, subsystem components may violate temperature limits. Subsystem components may be dependent on thermal conduction paths to transfer heat to or from other areas in order to protect against temperature limit violations.	TRM-01 thru TRM-56
All	If blankets are improperly installed or damaged during installation, heat transfer between the subsystem hardware and the surrounding environment may exceed design specification. The performances of blankets are dependent on the efficient coverage of surface area, minimizing thermal shorts, and the prevention of compression.	TRM-01 thru TRM-56
All	If a blanket or blankets are lost during any mission phase, heat transfer between the subsystem hardware and the surrounding environment may exceed design specification. Dynamic loads may dislodge blankets from the protected area.	TRM-01 thru TRM-56
All	If blankets are damaged by water absorption, heat transfer between the subsystem hardware and the surrounding environment may exceed design specification. Exposure to on-pad weather environments may lead to absorption of water by blankets. Exposure to on-pad weather environments may lead to absorption of water by blankets.	TRM-01 thru TRM-56
All	If a power string is lost, reliability of all heater systems decreases. Heaters rely on power being delivered by the electrical power system	TRM-01 thru TRM-56

TCS / TPS		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	If downloaded data is interrupted, vehicle thermal health may be compromised by insufficient response time from ground monitors. Vehicle thermal health depends on monitoring critical systems and ground response to anomalies.	TRM-01 thru TRM-56
All	If heaters within heater zones are designed improperly, duty cycles of adjacent heater zones may be reduced to the point that protected equipment may violate temperature limits. Heater configurations are designed so that all temperatures within a heater zone remain within temperature limits. In addition, the placement of heaters will not influence the performance of adjacent heater zones.	TRM-01 thru TRM-56
All	If additional attitude constraints are implemented due to off-nominal performance of a subsystem, other subsystem hardware thermal control may be adversely affected. When subsystem components do not perform as specified, additional attitude constraints may be required to protect them thermally or to reduce their wear.	TRM-01 thru TRM-56
All	If sustaining engineering becomes understaffed, anomaly resolution may become slow and incorrect. CEV may require the on-going task of sustaining engineering. In the event of layoffs (budget) or weather, sustaining engineering may become understaffed.	TRM-01 thru TRM-56
All	If computer resources are lost for any reason, anomaly resolution may become slow and incorrect. Weather conditions, viruses, and power loss may lead to computer resources being lost.	TRM-01 thru TRM-56
All	If a thermostat is subjected to localized heating, the heater zone associated with that thermostat may be inhibited and would therefore violate temperature limits. Localized heating of a heater zone thermostat may occur due to attitude or by warm fluid flow. This may inhibit the performance of that heater zone.	TRM-01 thru TRM-56

TCS / TPS		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	If damaged blankets, heaters, and instrumentation can not be accessed, identification and replacement can not be performed leading to inadequate thermal control. Maintenance of blankets, heaters, and instrumentation depends on access during the normal ground operations between flights. In addition, subsystem hardware may block access to the blankets, heaters, and instrumentation.	TRM-01 thru TRM-56
All	If an attitude timeline modification can not be thermally evaluated, temperature limit violations may occur. During a mission, operations may require attitude maneuver before a thermal evaluation can be performed.	TRM-01 thru TRM-56
All	If thermostats fail closed or open, protected equipment may violate temperature limits. Thermostat reliability decreases due to cycling and age. In addition, thermostats may fail due to manufacturing defects.	TRM-01 thru TRM-56
CEV; Lunar Lander	Given the unknown external crew environments there is a possibility the crew will not be protected from environmental extremes	TRM-01 thru TRM-56
All	Insufficient analysis of the interplanetary and lunar surface thermal environment may mean the thermal control and propulsion systems are improperly designed.	TRM-01 thru TRM-56
Lunar Lander	Given the unknown effects of lunar dust on the radiator optical properties; there is a possibility the TCS may not operate at within the design parameters.	TRM-41
CEV; Lunar Lander	A lack of knowledge of the thermal environment at L1 and on the moon could lead to an insufficient radiator design which allows freezing to occur.	TRM15 thru TRM-50

TCS / TPS		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given that some CEV concepts require drastic improvements in TPS materials, there is a possibility that the TPS improvement may not occur and the CEV concept will become unachievable. Some vehicle concepts are highly dependent on advancements in TPS. Small radius noses and leading edges, which provide extremely high cross ranges, also result in very high heat rates that will produce TPS surface temperatures approaching 4000 deg F. Current reusable materials, such as RCC and C/SiC, are capable of surface temperatures of only 3250 deg F.	TRM-21 thru TRM-55
CEV	If vehicle weight is not accurately predicted and controlled; then growth weight may occur, leading to increased re-entry heating and inadequate thermal protection.	TRM-51 thru TRM-53
CEV	If the aero thermal environment is not accurately defined early in the design cycle; the thermal protection system may not be adequate to protect the vehicle during re-entry.	TRM-51 thru TRM-53
CEV	Given the CEV may encounter extreme thermal conditions during separation, there is the possibility the CEV thermal environment will exceed safe limits resulting in injury or loss of crew.	TRM-21 thru TRM-34
CEV	Given touch temperatures of the vehicle may exceed safe limits; there is the possibility the crew may sustain burns during post landing emergency egress of the CM	TRM-21 thru TRM-34
CEV	Given the unknown external crew environments there is a possibility the crew will not be protected from environmental extremes	TRM-21 thru TRM-56

Table 20.17.2-7: TCS/TPS Risks

Program		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given a robust CEV cross range capability is beneficial in supporting the safe return of the crew after an abort event; a high L/D CEV could have an undesirable impact on the LV during launch dynamics.	TRM-21 thru TRM-23
CEV	Given the LV/CEV configuration has not been baselined; there is the possibility the CES design study will develop a design concept that will not adequately interface with the LV/CEV systems.	TRM-21 thru TRM-56
Lunar Lander	A massive solar particle event/coronal mass ejection while the crew is on the lunar surface may subject the crew to a critical radiation dose, resulting in crew illness or death and other long term health problems	TRM-36 thru TRM-42
All	Given the need to incorporate advanced technologies to produce a feasible lunar concept; there is the possibility that the technologies will not be developed to sufficient maturity in time for the first planned mission date.	TRM-01 thru TRM-56
Lunar Lander	The abrasiveness of lunar regolith may cause failures in external lander hardware, leading to a loss of mission.	TRM-41, TRM-42
Lunar Lander	Incomplete mission abort analyses may mean the current lander concept is unable to successfully perform all potential abort modes.	TRM-38, TRM-39
All	A "worse than expected" lunar L1 environment (radiation, thermal, MMOD) may lead to system component failures jeopardizing mission success.	TRM-15 thru TRM-47
All	There may be an inability to achieve sizing goals for some of the systems and attaining a TRL 6 by 2010; this could result in not meeting the mission objectives.	N/A
All	Assuming a two-fault tolerance just for redundancy sake may result in unavoidable increase in mass; may result in the lack of innovative functional redundancy measures (unlike redundancy).	TRM-01 thru TRM-56

Program		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	There is major technology and schedule risk in maturing Integrated Vehicle Health Monitoring system and fault management technologies to TRL= 6 by 2009. Major investments and partnerships with government agencies and industry are required for successful development.	N/A
CEV	There is a major technology and schedule risk in developing Ka-band transponder and ground infrastructure technologies to support Ka-band compatible TDRSS-II by 2006.	N/A
All	If the program does not plan for and develop an integration and verification test facility; avionic software development costs cannot be reduced. System integration should begin early in the program and not be an afterthought when the system is developed.	TRM-01 thru TRM-56
Lunar Lander	Given the improved RCRS does not reach a TRL 10 by 2010; a less efficient CO2 removal system may be used. The RCRS has minimum mass, volume, and power requirements. Using another CO2 removal system will penalize mass, volume, and power.	N/A
All	Given a lack of funding for new technology needs; low TRL items may not be developed within the schedule needs.	N/A
CEV	If the technology selected for CEV EPS systems has a technology readiness less than 9; there is a possibility that it could not be developed in time to meet program schedules	N/A

Table 20.17.2-8: Program Risks

Trajectory		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
All	Given the current maturity level of vehicle trajectory technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-01 thru TRM-56
CEV	Given the uncertainties of multi-body flow field interactions, there is the possibility a high dynamic pressure contingency separation could impose prohibitive propulsive and structural impacts to the CEV design. The uncertainties associated with this highly dynamic environment will require large margins.	TRM-21 thru TRM-25
CEV; Lunar Lander	Given the limited stability and control analysis of early configurations, there is the possibility that trajectories being considered for various abort scenarios may not be consistent with the final flight envelope. Abort capabilities may not be as robust once stability and control envelopes are defined.	TRM-21 thru TRM-25
All	Given the preliminary nature of mass property data and likelihood of mass growth, there is the possibility that trajectories defined for TPS and structural analysis may not provide representative heat rates and dynamic pressures. The TPS and structure may not provide the intended operational margins or capability.	TRM-01 thru TRM-56
CEV	Given the location of many of the landing sites being considered for contingency landings, there is the possibility the CEV will need to operate in unfavorable weather conditions to provide an intact abort capability. Operating in high winds and/or low visibility implies the need for robust landing gear, landing aids, and energy management capability, e.g., speedbrakes, L/D.	TRM-21 thru TRM-25
CEV	Given that the Booster fails to shut down/ fly the predicted trajectory; the vehicle can make re-contact with the CEV. Separation went good, but the vehicle comes back	TRM-21 thru TRM-25

Trajectory		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	If CES propulsion system is not adequately designed; then the crew may not be able to reach a safe distance.	TRM-21 thru TRM-25
CEV	Given that there is a loss of controlled flight of the CEV; there is a possibility of a hazardous landing (power line, tree, rock).	TRM-55
CEV	Given the unknown effects of high level winds; there is the possibility of the CM getting blown back into the booster system during a pad abort.	TRM-21 thru TRM-22
All	If Lunar Orbit Insertion (LOI) burn is not executed on time and/or with full Delta V requirement and/or with proper guided thrusting, resulting trajectory may compromise both mission success and crew safety. After libration point departure burn, following coast to low lunar altitude (~100km), a successful LOI maneuver is critical to successful capture of LL into low lunar orbit.	TRM-41

Table 20.17.2-9: Trajectory Risks

Operations

Affected Element(s)	Risk Statement	Associated TRM Critical Event ID #'s
CEV	A lack of communication with the ground/SAR may delay crew rescue operations.	TRM-56
CEV	Given a malfunction in the crew egress hatch; there is the possibility the crew will not be able to clear the vehicle after landing	TRM-55, TRM-56
CEV	Given night time and severe weather conditions; there is the possibility the crew will not be rescued safely from the CEV leading to injury or LOC.	TRM-55, TRM-56
CEV	Given the unknown operational environment of the CEV there is a possibility for CEV re-contact with the Launch Escape System	TRM-21 thru TRM-24
CEV	Given the presence of a toxic environment (internal or external) due to the malfunction or failure of a CEV subsystem; there is possibility for injury or loss of crew.	TRM-21 thru TRM-56
CEV	Given the unknown CEV parachute requirements; there is a possibility chute design will be inadequate resulting in injury to crew.	TRM-55
CEV	A lightning strike to the CEV; could disable the CES.	TRM-21, TRM-22, TRM-23
All	If there is electrostatic discharge causing failure to hardware; then there may be loss of communication and control.	TRM-21 thru TRM-56
CEV	Given that the Booster fails to shut down/ fly the predicted trajectory; the vehicle can re-contact the CEV. Separation went good, but the vehicle comes back	TRM-21 thru TRM-24
CEV	Given a failure of the crew H2O flotation or survival gear, the possibility exists that the crewmember will not be able to survive after egress from the CEV.	TRM-55, TRM-56
CEV	Given the unknown effects of high winds; there is the possibility of the CEV getting blown back into the booster system during an escape/abort.	TRM-21 thru TRM-24

Operations

Affected Element(s)	Risk Statement	Associated TRM Critical Event ID #'s
CEV	Given the presence of hazardous weather conditions at or around the launch facility; there is possibility of CES will not operate sufficiently to ensure crew safety.	TRM-21 thru TRM-24
CEV	Given a loss of the CEV ability to stabilize (keep upright) in the H2O; there is the possibility the crew will not be able to operate the hatch and make safe egress. System can also be affected; operations	TRM-56
CEV	Given the potential for CEV landing in an adverse or hazardous environment; the rescue support teams will be unable to reach the crew in a timely manner leading to injury or LOC.	TRM-56
CEV; Lunar Lander	Given the unknown external crew environments there is a possibility the crew will not be protected from environmental extremes	TRM-21 thru TRM-56
CEV	Given an inadvertent initiation of the CES; there is the possibility of loss of vehicle and mission.	TRM-21, TRM-55, TRM-56
CEV	Given the CEV will encounter thermal soak back; there is the possibility the internal temp of the cabin will exceed safe limits resulting in injury or LOC	TRM-55, TRM-56
CEV	Given that there is a loss of controlled flight of the CEV; there is a possibility of a hazardous landing (power line, tree, rock).	TRM-55, TRM-56
CEV	Given the CEV is exposed to a mission induced toxic environment post-landing once the cabin has been vented; there is a possibility of injury or loss of crew due to toxic exposure.	TRM-55, TRM-56
CEV	An inability to find the crew if they leave the vehicle; could result in a loss of crew	TRM-56
CEV	Given that the SAR beacon could be damaged during CES initiation; there is possibility that the SAR team may be hampered in their effort to locate the crew.	TRM-56

Operations

Affected Element(s)	Risk Statement	Associated TRM Critical Event ID #'s
CEV	Given the effects on the vehicle in a low Q-bar environment there is the possibility of chute deployment failure	TRM-55, TRM-56
CEV	Given the effects on the vehicle in a high Q-bar environment there is the possibility of chute deployment failure	TRM-55, TRM-56
CEV	Given that at post ignition, pre-liftoff vehicle may collide with tower; may result in a failure of the CES system.	TRM-21
CEV	Separation of the CES below a survivable altitude; may result in a loss of crew.	TRM-21, TRM-23
CEV	Given the short amount of time required to activate CES systems during a catastrophic failure event; there is the possibility the CES will not have an adequate fault detection and warning capability to activate required systems to assure crew survival.	TRM-21
CEV	Given a loss of capability for the CES to release/ activate escape mechanism; there is the possibility the CES will not separate from the CEV leading to injury or LOC.	TRM-20, TRM-55
CEV	Given a loss or absence of CEV maneuvering capabilities during final decent; there is possibility of the CEV being forced to land on obstacles (trees, etc).	TRM-51, TRM-55
CEV	Given a malfunction in the crew egress hatch; there is the possibility the crew will not be able to clear the vehicle and operate pad escape system prior to launch	N/A
CEV	A lack of communication with the ground/SAR; may delay the crew rescue operations.	TRM-56
CEV	Given a malfunction in the crew egress hatch; there is the possibility the crew will not be able to clear the vehicle after landing	TRM-56
CEV	Given night time and severe weather conditions; there is the possibility the crew will not be rescued safely from the CES leading to injury or LOC.	TRM-56
All	If there is electrostatic discharge causing failure to hardware; then there maybe loss of communication and control.	TRM-01 thru TRM-56

Operations		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given that the SAR beacon could be damaged during CES initiation; there is possibility that the SAR team may be hampered in their effort to locate the crew. For the pod beacon	TRM-56
All	Using a single launch pad for the mission may result in a delayed launch of three of the four elements should the launch vehicles encounter rollout delays? Also, a catastrophic failure on one launch pad will eliminate the entire launch infrastructure.	N/A
All	There may not be adequate margins for launch delays in the mission elements; this could result in a failed mission simply because the elements could not be launched on time.	TRM-01, TRM-06, TRM-16, TRM-21
CEV Injection Stage; Lunar Lander Injection Stage	There is a risk of boil-off of the element propellants before the following launches can take place; this could result in a failed mission simply because the elements could not be launched on time.	TRM-01 thru TRM-56
Lunar Lander	Should a night landing be planned, additional infrastructure and night vision will be required for the crew for landing; could result in hazard avoidance issues.	TRM-21
Lunar Lander; CEV Injection Stage; Lunar Lander Injection Stage	The lack of a disposal plan for injection stages and Lunar Landers could result in future orbital debris and potential menace to navigation.	N/A

Table 20.17.2-10: Operations Risks

Human Factors

Affected Element(s)	Risk Statement	Associated Critical Event ID #'s
CEV	Given the effects of high G-loads on the CEV are not defined; there is possibility the CEV design may expose the crew to excessive G loads	TRM-21 thru TRM-56
CEV	Given a failure resulting in a loss of CEV thermal control; the possibility exists the crew will be exposed to excessive temperatures leading to injury or loss of crew	TRM-21 thru TRM-56
CEV	Given the crew is in a deconditioned state (physical and physiological); there is a possibility of loss of crew. An emergency egress from the CEV may not be possible.	TRM-21 thru TRM-56
CEV; Lunar Lander	Given the presence of a toxic environment (internal or external) there is possibility for injury or loss of crew.	TRM-21 thru TRM-56
CEV	Given a malfunction in the crew egress hatch; there is the possibility the crew will not be able to clear the vehicle after landing	TRM-53
CEV	Given a loss of the CEV ability to stabilize (keep upright) in the H_2O; there is the possibility the crew will not be able to operate the hatch and make safe egress	TRM-52, TRM-53
CEV	Given the CEV will encounter thermal soak back; there is the possibility the internal temp of the cabin will exceed safe limits resulting in injury or loss of crew	TRM-21 thru TRM-56
CEV	Given the crew may have to rely on the CEV for exercise; a possibility exists that the crew may experience bone loss and muscle atrophy from limited exercise capabilities	TRM-21 thru TRM-56
CEV	Given a failure in the flight suit's ability to provide breathable oxygen during pre-launch and H_2O landing scenarios; the crew may not have breathable air resulting in injury or loss of the crew.	TRM-38
CEV	Radiation environment may damage equipment/software; an inability to properly activate the CEV may occur.	TRM-21 thru TRM-56
CEV	Given a malfunction in the crew egress hatch; there is the possibility the crew will not be able to clear the vehicle and operate pad escape system prior to launch	TRM-20

Human Factors		
Affected Element(s)	**Risk Statement**	**Associated Critical Event ID #'s**
CEV	Given a malfunction in the crew egress hatch; there is the possibility the crew will not be able to clear the vehicle after landing	TRM-52, TRM-53
CEV	Given touch temperatures of the vehicle may exceed safe limits; there is the possibility the crew may sustain burns during post landing emergency egress of the CES	TRM-51, TRM-52
CEV	Given that CES crew restraints fail or are inadequate due to high accelerations allowing crew flailing; the possibility exists for injury or LOC.	TRM-20 thru TRM-53
CEV; Lunar Lander	Given that there are excessive landing loads; there is the possibility of losing the crew.	TRM-52
CEV	Given a failure leading to a fire in the crew compartment of the CES; there is the possibility for injury or LOC	TRM-21 thru TRM-56
CEV; Lunar Lander	Given a loss of crew H_2O; there is a possibility the crew may become sick due to dehydration.	TRM-21 thru TRM-56
CEV	Given a failure in the flight suit's ability to provide breathable oxygen during pre-launch and H_2O landing scenarios; the crew may not have breathable air resulting in injury or loss of the crew.	TRM-21 thru TRM-56
CEV	Radiation environment may damage equipment/software; an inability to properly activate the CES may occur.	TRM-25 thru TRM-50
CEV	Given a malfunction in the crew egress hatch; there is the possibility the crew will not be able to clear the vehicle and operate pad escape system prior to launch	TRM-20
CEV	Given touch temperatures of the vehicle may exceed safe limits; there is the possibility the crew may sustain burns during post landing emergency egress of the CES	TRM-53
CEV	Given that CEV crew restraints fail or are inadequate due to high accelerations allowing crew flailing; the possibility exists for injury or LOC.	TRM-21 thru TRM-55

Human Factors		
Affected Element(s)	**Risk Statement**	**Associated Critical Event ID #'s**
CEV; Lunar Lander	Given the absence of an adequate medical kit and or medical expertise to care for an injured crewmember; there is a possibility injuries requiring immediate attention will not be addressed.	TRM-21 thru TRM-56
CEV; Lunar Lander	Given the unknown external crew environments there is a possibility the crew will not be protected from environmental extremes	TRM-21 thru TRM-56
CEV	Given the crew may have to remain in the CM after landing, contamination of the crew food and water supplies could lead to crew dehydration and weakness	TRM-56
CEV	Given the crew may have to remain in the CM after landing inadequate waste collection system could lead to crew exposure to human waste (fecal and urine)	TRM-56
CEV	Given the CM is exposed to a mission induced toxic environment post-landing once the cabin has been vented; there is a possibility of injury or loss of crew due to toxic exposure.	TRM-53, TRM-54
CEV; Lunar Lander	Given a loss of lighting in the cabin; there is the possibility the crew will be injured	TRM-21 thru TRM-55
CEV; Lunar Lander	Given a medical emergency occurs at a point when the crew return is not possible for a length of time; there is the possibility needed medical attention may not be given in time resulting in possible LOC	TRM-28 thru TRM-50
CEV	Given a failure resulting in a loss of CEV thermal control; the possibility exists the crew will be exposed to excessive temperatures leading to injury or loss of crew	TRM-21 thru TRM-56
Lunar Lander	Because of the height of the cabin relative to the lunar surface, an injured crewmember may be unable to ingress into the lander.	TRM-41

Human Factors		
Affected Element(s)	**Risk Statement**	**Associated Critical Event ID #'s**
CEV; Lunar Lander	Not fully understanding what the radiation environment is at lunar L1 may result in insufficient radiation protection for the crew and vehicle. Radiation exposure could lead to loss of crew or a loss of a critical subsystem.	TRM-31 thru TRM-51
Lunar Lander	If lunar dust contaminates the habitable volumes of the LL; then health and crew performance could be negatively impacted. Lunar dust has sharp edges and can be inhaled or get in the crewmembers' eyes causing discomfort or damage to the lungs and eyes.	TRM-41

Table 20.17.2-11: Human Factors Risks

Crew Escape		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given the current maturity level of crew escape system technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-21 thru TRM-55
CEV	Given a loss of crew escape system capability to provide powered flight; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Vehicle, Loss of Mission and/or Loss of Crew.	TRM-21 thru TRM-55
CEV	A lack of communication with the ground and Search and Rescue after separation and before landing, may delay the crew rescue operations.	TRM-56
CEV	Given a loss of voice communication coverage; there is the possibility the crew will not be able to transmit data regarding an injured crewmember requiring immediate attention post Earth landing.	TRM-56
CEV	Given the short amount of time required to activate abort systems during a catastrophic failure event; there is the possibility the crew escape will not have an adequate fault detection and warning capability to activate required systems to assure crew survival.	TRM-21 thru TRM-55
CEV	Given that there is loss of controlled flight of the primary vehicle; the crew may not be able to initiate the Crew Escape System.	TRM-21 thru TRM-55
CEV	Given that there is a loss of instrumentation, there is a possibility of not detecting a system failure to initiate the Crew Escape System.	TRM-21 thru TRM-55
CEV	Radiation environment may damage equipment/software; an inability to properly activate an end-of-mission abort may occur.	TRM-28 thru TRM-53

Crew Escape		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	A lack of communication with the ground/SAR; may delay the crew rescue operations. Putting you in a orientation that the Crew Escape System is out safe projection. After separation before landing	TRM-56
CEV	Given a loss of crew to crew communication capabilities; there is the possibility crewmembers will not be able to status the condition of other crewmembers during Crew Escape System operations	TRM-21 thru TRM-56
CEV	Given a loss of voice communication coverage; there is the possibility the crew will not be able to transmit data regarding an injured crewmember requiring immediate attention upon landing.	TRM-21 thru TRM-56
CEV	Given that there is loss of controlled flight of the primary vehicle; the crew may not be able to initiate the CES.	TRM-21 thru TRM-56
CEV	Given a failure resulting in a short in the CES power system; there is a possibility he crew will be without power systems during CES operations. Technician mishandle	TRM-21 thru TRM-56
CEV	Given that there is a loss of instrumentation, there is a possibility of not detecting a system failure to initiate the CES.	TRM-21 thru TRM-56
CEV	Given an inadvertent initiation of the CES; there is the possibility of loss of vehicle and mission.	TRM-21, TRM-55
CEV	Given the LV blast environment has not been fully defined; there is a possibility the CES systems will not be sized correctly to protect the crew during a catastrophic blast wave failure.	TRM-21, TRM-22, TRM-23

Table 20.17.2-12: Crew Escape Risks

Search and Rescue		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given the current maturity level of crew escape system technology; there is a possibility the required capabilities will not be developed and tested in sufficient time to meet a TRL 6 by 2009.	TRM-56
CEV	Given a loss of crew escape system capability; there is a possibility that some or all critical functions required for mission success will be lost resulting in Loss of Crew.	TRM-56
CEV	Given that the CEV may be required to protect the crew in adverse sea conditions; there is the possibility that the CES systems will not be designed adequately to protect the crew during (post-landing) survival and rescue (SAR) operations.	TRM-56
CEV	A lack of communication with the ground/SAR; may delay the crew rescue operations. After separation before landing	TRM-56
CEV	Given a failure of the crew H2O flotation or survival gear, the possibility exists that the crewmember will not be able to survive after egress from the CEV.	TRM-56
CEV	Given the potential for CEV landing in an adverse or hazardous environment; the rescue support teams will be unable to reach the crew in a timely manner leading to injury or LOC.	TRM-56
CEV	Given that there is a loss of controlled flight of the CEV; there is a possibility of a hazardous landing (power line, tree, rock).	TRM-56
CEV	Given that the SAR beacon could be damaged during CES initiation; there is possibility that the SAR team may not be hampered in their effort to locate the crew.	TRM-56

Search and Rescue		
Affected Element(s)	**Risk Statement**	**Associated TRM Critical Event ID #'s**
CEV	Given that the CEV may be required to protect the crew in adverse sea conditions; there is the possibility that the CES systems will not be designed adequately to protect the crew during (post-landing) survival and rescue (SAR) operations.	TRM-56
CEV	A lack of communication with the ground/SAR; may delay the crew rescue operations. After separation before landing	TRM-56
CEV	Given a failure of the crew H_2O flotation or survival gear, the possibility exists that the crewmember will not be able to survive after egress from the CES.	TRM-56
CEV	Given the potential for CES landing in an adverse or hazardous environment; the rescue support teams will be unable to reach the crew in a timely manner leading to injury or LOC.	TRM-56
CEV	Given that there is a loss of controlled flight of the CEV; there is a possibility of a hazardous landing (power line, tree, rock).	TRM-56
CEV	Given that the SAR beacon could be damaged during CES initiation; there is possibility that the SAR team may not be hampered in their effort to locate the crew. For the pod beacon	TRM-56

Table 20.17.2-13: Search and Rescue Risks

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1.0 Introduction

The Lunar Design Reference Mission 2 (LDRM-2) study was initiated by NASA Code T, later designated the Exploration Systems Mission Directorate, in April of 2004 to perform a series of focused lunar mission trade studies intended to provide a better understanding of the relative benefits of differing mission approaches, as well as to determine mission sensitivities to key system design parameters.

The intent of the planned lunar missions is to support a wide range of scientific investigations, technology and systems development, and integrated testing to reduce the risks of future human exploration of Mars. Three LDRM studies were originally outlined to bracket a range of potential lunar mission scenarios and associated flight element functionality. LDRM-1 consists of a seven-day surface stay in the equatorial region of the moon. LDRM-2 is also based on a seven-day lunar surface stay, but includes global lunar access with the capability to initiate an Earth return at any time. LDRM-3 provides the capability for a long-duration lunar surface stay in the range of thirty to ninety days with multiple missions to a single polar landing site outfitted with additional surface elements.

The LDRM-2 exploration objectives and requirements were selected as the starting point for the lunar architecture study. The Phase 1 deliverables from the LDRM-2 study document the results of the architecture analyses associated with short duration lunar missions with global access capability. The LDRM-2 study was subsequently expanded with a Phase 2 effort focused on the LDRM-3 exploration objectives. The LDRM-2 Phase 2 deliverables document the results of the architecture analyses associated with long duration lunar missions with a restricted range of surface access.

The results of the LDRM studies will support the development of Level 1 requirements for the Crew Exploration Vehicle (CEV) by quantifying the sensitivities of flight elements to key system design parameters and subsystem technologies in the context of an end-to-end lunar mission. The definition and sizing of a complete lunar mission also provides valuable insight into the launch vehicle characteristics and infrastructure that are needed to support the delivery of flight elements to low Earth orbit (LEO).

2.0 Study Scope

2.1 LDRM-2 Background

There are three basic architectures for executing a human lunar exploration mission - direct return (also referred to as lunar surface rendezvous or LSR), libration point rendezvous (LPR) and lunar orbit rendezvous (LOR) - defined by the method that is employed to return the crew to the Earth after the conclusion of lunar surface operations. Due to orbital mechanics considerations, each of these architectures offers distinct advantages and disadvantages with respect to a given set of mission objectives and requirements. The purpose of the LDRM-2 task is to provide focused "down-and-in" assessments of specific lunar exploration mission designs. The LDRM-2 study results are intended to complement the data from parallel NASA exploration studies. Some of these studies are focused on the launch and lunar surface system segments required for a lunar exploration mission. Others are targeted to a broad, higher-level assessment of lunar architecture alternatives including advanced subsystem technologies.

The LDRM-2 study evolved into two separate phases, each focused on a specific lunar exploration architecture. The Phase 1 mission leverages the cislunar Earth-moon libration point known as "L1" as an orbital staging point to enable global lunar access with anytime return capability to Earth. The Phase 2 mission is based on a variant of the lunar orbit rendezvous approach employed successfully during the Apollo Program. Unlike Apollo, however, the Phase 2 mission is targeted to long duration, near-polar lunar surface missions. The Phase 1 and Phase 2 results are documented in separate volumes of the LDRM-2 Final Report.

Mass is a primary driver for the Earth-to-orbit launch infrastructure and is often also used as the basis for cost estimation. Therefore, the flight element mass estimates developed using the Envision parametric sizing tool are key products of the LDRM-2 study. Element sizing is based upon a top-level nominal mission timeline, functional requirements, critical spacecraft dimensions, mission environments and subsystem component and propellant selections. A common set of mission environments and subsystem technology options, which were identified early in the execution of the LDRM-2 Phase 1 task, were applied to the sizing for both the Phase 1 and Phase 2 missions. The associated technology and environment reports provided in Section 20.0 of the Phase 1 report were also submitted to the NASA Headquarters Exploration Systems Mission Directorate to support the development of an overall exploration technology development and testing plan.

Independent studies of launch vehicle capabilities and lunar surface infrastructure are being pursued in parallel to the LDRM-2 task. Initially the flight elements defined in the LDRM-2 study may not fit within the payload mass and volume envelopes deemed feasible for the next generation of launch vehicles. Similarly, the cargo delivery capability of the LDRM-2 flight elements has not yet been linked with the infrastructure requirements currently being defined for long duration lunar surface missions. In subsequent design cycles, however, the requirements and constraints associated with the ground, flight and lunar surface segments will be blended to establish comprehensive and integrated lunar mission architectures.

2.2 Phase 2 Mission Definition

The LDRM-2 Phase 2 study expands the scope of the lunar orbit rendezvous mission analysis performed during the Phase 1 study. The Phase 1 lunar orbit rendezvous variant to the L1 TRM supports short duration surface exploration missions with global lunar access and anytime Earth return capability. The Phase 2 study maintains the anytime Earth return capability, but constrains the range of landing sites and increases the maximum duration of the surface mission.

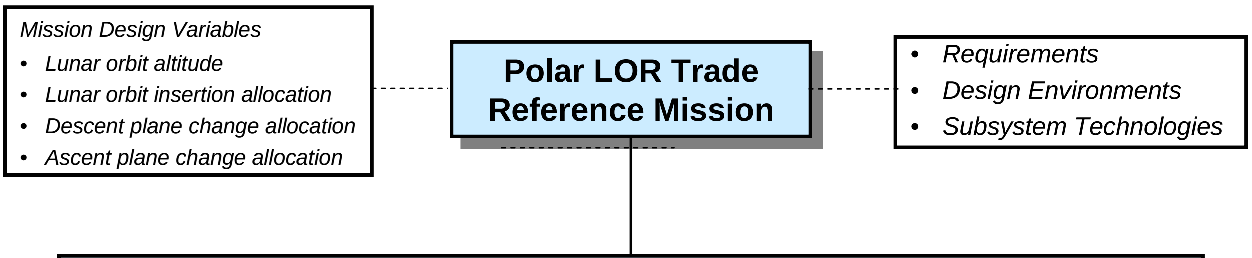
The core of the Phase 2 study is the polar LOR trade reference mission which employs the lunar orbit rendezvous architecture for long duration surface missions to near-polar landing sites with anytime Earth return capability. The polar LOR TRM uses a polar lunar parking orbit as the staging point for accessing landing sites between seventy and ninety degrees north or south latitude. Short duration exploration missions to the lunar surface can be conducted out of the lander, but are limited in scope by the on-board power generation, thermal control and life support resources. Long duration lunar surface missions in the range of thirty to ninety days are supported using additional surface elements.

2.3 Phase 2 Study Approach

Similar to the approach used for the Phase 1 study, the Phase 2 study employs a trade reference mission as a point of departure for evaluating mission design sensitivities. The Phase 2 polar LOR TRM builds upon the results of the LOR variant to the L1 TRM developed in Phase 1. As shown in the upper, right-hand area of Figure 2.3-1, two of the basic inputs to the polar LOR TRM are the design environments and subsystem technologies defined in the Phase 1 study. The third input is the revised set of assumptions defined in the Phase 2 task statement to support the new mission scenario for near-polar landing sites. The major changes to the assumptions for the polar LOR TRM are as follows:

- Low lunar orbit is used as the lunar vicinity rendezvous point to support near-polar surface access.
- Surface mission durations range from 7 to 90 days
- Near-polar landing site locations range from seventy to ninety degrees north or south latitude
- The lander will provide a minimum independent active surface operating capability of 4 days to allow for transition to/from surface assets
- The lunar lander will be transported with the crew from low Earth orbit to low lunar orbit.

The design flexibility of the lunar orbit rendezvous architecture increases the complexity of the mission definition and analysis process. Unlike in the L1 TRM architecture, however, the CEV parking orbit for a lunar orbit rendezvous architecture can be optimized to reduce the total mass of the flight elements by matching the lunar parking orbit altitude and inclination to a specific set of mission objectives. The magnitude of the required descent and ascent plane changes and lunar orbit departure maneuver are closely coupled to the altitude and orientation of the CEV lunar parking orbit. The functionality of a LOR flight element can be expanded to envelope multiple mission profiles by increasing its maximum propellant capacity, offloading propellant, or reallocating or sharing responsibility for major maneuvers, as appropriate.



Alternate Mission Capture

Mission Option	Crew Size	Staging Location	Time at Location
1. Global Access	4	Lunar Orbit	TBD days
2. L1 CEV Dormant	4	L1 Libration Point	15-90 days dormant*
3. L1 CEV Active	4	L1 Libration Point	TBD days active
4. Mars Mission Staging	6	High Earth Orbit (800 km x 70764 km)	2 days
5. Mars Mission Return	6	Direct Entry at 13 km/s	800 days dormant 2 days active

* Assume that additional resources necessary for crew and vehicle support functions while at the L_1 location are provided by separate elements.

Figure 2.3-1: LDRM-2 Phase 2 Study Approach

Rather than focusing on parametric mission sensitivities, the Phase 2 study utilizes the polar LOR TRM to evaluate the extensibility of the LOR architecture to five alternate missions. Alternate mission #1 explores the ability of the flight elements defined for the polar LOR TRM to perform short duration missions to a range of surface sites. Alternate missions #2 and #3 assess the capability of the CEV to support human missions to the L1 libration point in both dormant and active loiter modes. Alternate missions #4 and #5 examine the applicability of the CEV to support crew transfer and return for Mars missions for a crew size of six. Alternate mission #5 also adds the requirement for the Mars return spacecraft to support a long duration dormancy phase.

Alternate missions #4 and #5 can be approached in two ways. In the coupled approach, the spacecraft that delivers the crew to the Mars transit vehicle in alternate mission #4 remains attached to the Mars transit vehicle and serves as the Earth return spacecraft for alternate mission #5. Another approach is to de-mate and return the alternate mission #4 spacecraft to Earth after completing the crew transfer. In this scenario the recovery of the Mars crew in alternate mission #5 is accomplished using a pristine spacecraft, perhaps a long duration variant, that is attached during the orbital assembly of the Mars transit vehicle.

3.0 Executive Summary

The Phase 1 and Phase 2 studies show that the lunar orbit rendezvous architecture can result in a lower initial mass in low Earth orbit than an L1 rendezvous architecture for a wide range of mission designs, even if anytime Earth return capability from the lunar surface is required. In the Phase 1 study the lunar orbit rendezvous architecture provided a 31t (13.5%) lower IMLEO than the L1 rendezvous architecture for global lunar access and a seven-day surface mission. In the Phase 2 study the lunar orbit rendezvous architecture provided a 27 to 33t (12.8 to 15.6%) reduction in IMLEO relative to the L1 rendezvous option for near-polar landing sites and long duration surface missions. The primary factor in the relative mass-efficiency of the lunar orbit rendezvous architecture for these missions is the reduction in ΔV allocated to the lander element, even though the total mission ΔV is roughly equal to that of the L1 rendezvous approach. This result is consistent with the characteristics of the rocket equation, as is demonstrated by the lander descent stage and ascent stage gear ratio data for the two architectures.

The lunar orbit rendezvous architecture also offers the potential to reduce IMLEO by substituting loiter time on the lunar surface and/or in lunar orbit for some or all of the propulsive plane change capability required for anytime Earth return. For the Phase 2 polar lunar orbit rendezvous mission, a maximum loiter implementation (minimum ΔV for the lunar ascent and lunar orbit departure maneuvers) can save an additional 20t (11.2%) in IMLEO relative to a mission design with anytime Earth return. The cumulative IMLEO savings of polar lunar orbit rendezvous combined with a maximum loiter implementation is 53t (25.1%) compared to the L1 rendezvous approach. However, the mass advantages of loiter time over anytime Earth return must be weighed in relation to its effects on crew safety. Loiter time is not necessarily a practical alternative to anytime Earth return for a time-critical medical condition or hardware problem. Long loiter times will also increase the risk of crew exposure to hazardous conditions such as solar particle events.

In order to optimize IMLEO for the lunar orbit rendezvous architecture, it is important to select a CEV lunar parking orbit (altitude and orientation) that efficiently supports the desired ranges of lunar surface access and surface mission duration in combination with the selected loiter/anytime Earth return implementation. In addition, the IMLEO for any lunar architecture is likely to vary significantly with changes to the number of propulsive stages, ΔV distribution among the stages, or the propulsive efficiency of those stages. High propulsion efficiency is most effective for stages that must transport a large payload through a substantial ΔV, such as the Earth Departure Stage.

A few of the primary considerations in defining the number of Earth-to-orbit launches for a lunar mission are the maximum payload capacities of the cargo and human-rated launch vehicles, natural breakpoints in the design of the flight elements, and operational considerations such as crew mission duration, rendezvous and docking maneuvers, and the frequency of Earth orbit departure windows necessary to support the mission objectives and constraints. A tandem Earth orbit departure can help to reduce the maximum launch vehicle payload requirement by enabling the use of equal-mass Earth Departure Stages, but also requires more on-orbit assembly than a split mission (pre-deployment or convoy) for a given number of launches. In a split mission scenario, the mission duration for the crew can be minimized by the delivery of the CEV and Earth Departure Stage in one launch of a multiple launch strategy.

The subsystem technologies for the CEV and lander that were defined in Phase 1 do not appear to be greatly affected by the shift to a long duration, near-polar lunar mission in Phase 2. The only significant high-level change to the CEV is the use of solar arrays rather than fuel cells to support the extended CEV loiter time in lunar orbit. The scope of the CEV and lander design environments will also be affected by the Phase 2 mission design. The decision to eliminate the lander airlock was driven by the Phase 2 focus on long duration exploration using surface assets plus the reduction in independent active lander operation time to four days.

The results of alternate mission #1 show that the CEV and Lunar Lander that were sized to support the polar LOR TRM have sufficient propulsive capability to perform short duration missions to any landing site on the lunar surface while the total mass of the two Earth Departure Stages would have to be increased by 3 t each. The primary limitations inherited from the polar LOR TRM are the four days of consumables on the lander and the potential thermal limitations of a lander designed for a near-polar environment. The CEV also requires one additional day of crew consumables to accommodate a 24-hour sequence of maneuvers during lunar orbit arrival. But this additional CEV mass is small and is not considered to be significant issue. The lunar orbit arrival ΔV must also be increased to provide full support for the short duration global access mission. The options are to increase the size of the EDS and the payload capacity of the cargo launch vehicle, or to restrict the regions of the Moon that are accessible for short duration exploration missions.

The analysis results for alternate missions #2 and #3 show that the CEV and EDS elements sized to support the polar LOR TRM both exceed the propulsive capability required for a mission to L1. The CEV defined for the polar LOR TRM supports the extended dormancy period at L1 for alternate mission #2, and can also provide up to three days of active, crewed operation at L1 in support of alternate mission #3. The active mission duration and functional capabilities of the CEV for near-Earth missions can be enhanced, if desired, through the addition of a resource module. The mass of the resource module could be covered within the launch vehicle capability defined for the polar LOR TRM by offloading excess propellant from the EDS.

The analysis results for alternate mission #4 show that the CEV and EDS elements sized to support the polar LOR TRM both exceed the propulsive capability required for the Mars Mission Staging scenario. The CEV internal layout developed for the lunar missions would have to be altered to support six crew and their associated equipment, but the CEV consumables are sufficient due to the shorter mission duration, even with the increase in crew size. Alternate mission #4 involves the launch of the CEV and EDS elements to a high inclination low Earth orbit, thus raising the issues of launch performance and launch aborts. If the scope of the Constellation Program includes high-inclination, near-Earth missions, then the design requirements for the CEV and the human-rated and cargo launch vehicles should reflect the need for that functionality.

Since the alternate mission #5 also involves a crew size of six, it involves the same internal modification of the CEV crew module as described for the Mars Mission Staging scenario. The subsystem technologies identified for the lunar exploration missions are viable for both the long period of dormancy during the Earth-Mars transits and Mars surface exploration, as well as the short active phase following separation from the Mars transit vehicle. The subsystem components would have to be initially designed with a consideration for eventual long-duration dormancy, though. Crew module thermal protection system impacts associated with the high Earth entry velocities for Mars return are expected to be modest since an ablative base heat shield is already employed on the lunar CEV capsule. The primary spacecraft design issues associated with the Mars Return mission are the revisions to the internal layout of the crew module to support six crewmembers, and the repackaging of the power generation, active thermal control and life support resources that are supplied by the Service Module for the lunar exploration missions.

4.0 Task Statement

4.1 Background

Human missions to the moon will be conducted in preparation for future human missions to Mars. Three lunar design reference missions (LDRMs) have been developed to bracket a range of lunar mission scenarios to determine required functionality of the system elements:

1) 7-day surface stay in the equatorial region of the moon
2) 7-day surface stay, with global access of the lunar surface enabled via multiple missions
3) 30 to 90 day surface stay with multiple missions to a single polar site using additional surface elements

In the Phase 2 portion of the LDRM-2 study, a polar lunar orbit rendezvous architecture is developed to address the third mission scenario. The polar LOR trade reference mission is then used to evaluate the potential for "capturing" five alternate missions.

4.2 Phase 2 Task Description

Perform a system and mission assessment of a polar lunar landing scenario. The emphasis of this analysis is to understand the system and mission design impacts of both the CEV and lunar lander for near-polar landing sites. Analysis of the surface systems necessary for the extended duration mission is not the focus of this study and should not be conducted as a part of this task. Analysis of the lander element is necessary in order to determine the proper mission design tradeoffs between it and the CEV for anytime crew return capabilities.

4.2.1 Analysis Products

The following analysis products are required for the Polar LOR Trade Reference Mission:

- Determine the effect of landing site location on the mission and system designs for both the CEV and lander
- Determine the effect of surface mission duration on both the CEV and lander designs with emphasis placed on breakpoints for technology or system design selections
- Determine the effect of launch strategy as conducted in the Phase 1 effort (2 launch, 4 launch, and 25 t launch solutions)

4.2.2 Revised Trade Reference Mission Assumptions

The following revisions to the Phase 1 assumptions are necessary to support the mission scenario for the Polar LOR Trade Reference Mission:

- Low lunar orbit is used as the lunar vicinity rendezvous point to support near-polar surface access.
- Surface mission durations range from 7 to 90 days

- Near-polar landing site locations range from 70-90 degrees north or south latitude
- The lander will provide a minimum independent active surface operating capability of 4 days to allow for transition to/from surface assets
- The lunar lander will be transported with the crew from low Earth orbit to low lunar orbit.

4.2.3 Alternate Mission Capture

Following the definition of the Polar LOR TRM, perform system and mission analysis on a range of alternate mission concepts with an emphasis placed on determining the key CEV system drivers and operational approaches. The range of mission concepts should, at a minimum, include the following:

Option	*Crew Size*	*Staging Location*	*Landing Site*	*Time at Location*
1 – LOR Global Access	4	Low Lunar Orbit	Global	TBD days
2 – L1 Mission Dormant	4	L1 Libration Point	-	15-90 days dormant
3 – L1 Mission Active	4	L1 Libration Point	-	TBD days active
4 – Mars Mission Staging	6	High Earth Orbit (800 x 70764 km)	-	2 days
5 – Mars Mission Return	6	Direct Entry @ 13 km/s	-	800 days dormant 2 days active

4.2.4 Ground Rules

The ground rules for LDRM-2 Phase 1 and Phase 2 are identical, but are repeated for clarity.

- Subsystem technology freeze at (TRL 6) six years before IOC (use TRL 6 by 2009 as your reference for design). Freeze time increases to 9 years for “major architectural” drivers (e.g., in-flight refueling).
- First lunar mission 2015-2020
- Exploration missions are expected to be mass and volume limited, thus placing a premium on design efficiency.
- The primary focus of the study is to provide Code T with the information needed to develop effective CEV Level 1 requirements.

4.2.5 Trade Reference Mission Assumptions

With the exception of the revisions noted in Section 4.2.2, the trade reference mission assumptions for LDRM-2 Phase 2 are unchanged from those identified for Phase 1. For clarity, however, the complete set of TRM assumptions are provided below with the Phase 2 revisions/additions highlighted in ***bold italics***.

1. One human lunar mission per year

2. Return mass from the moon is 100 kg. Return samples may require conditioning (consider biological and planetary materials samples, TBD)
3. Payload to lunar surface (science and enhanced EVA mobility) is 500kg
4. All mission elements placed in LEO (28.5 deg 407km circular)
5. DRM analysis should determine and baseline minimum launch capability required for a 4-launch solution.
6. Consider the lunar mission elements to be "cargo" in terms of delivery to the LEO parking orbit. The propulsive capabilities of the lunar mission elements will not be employed for orbit insertion, but may be required for orbit maintenance.
7. Automated rendezvous and docking shall be used to assemble the elements (identify required interfaces, resources across the interfaces, and contingency operations)
8. Assume 2 weeks between launches (identify any sensitivities/major architectural implications).
9. Crew must be launched on a human rated launch system
10. A dedicated lunar lander element with a separate crew module will be used to transfer the crew from the lunar vicinity to the lunar surface and back to lunar vicinity.
11. ***Surface mission durations range from 7 to 90 days***
12. 4 crew with all crew going to the lunar surface
13. Daily EVAs will be conducted on the surface of the Moon from the lunar lander.
14. ***The lander will provide a minimum independent active surface operating capability of 4 days to allow for transition to/from surface assets.***
15. The CEV and lunar lander are not required to be reusable and will not be explicitly designed for reusability.
16. The lunar lander will not be designed to provide functionality beyond that required for the planned lunar surface stay time.
17. ***The reference lunar surface design environment is defined by the lighting and thermal conditions for a near-polar lunar mission.***
18. A Crew Exploration Vehicle (CEV) element will provide the crew habitation function from the earth's surface to lunar vicinity and back to the earth's surface.
19. The nominal Earth return for the CEV is a direct entry with a water landing.
20. The CEV design will incorporate functionality for land landing as a contingency for an ascent abort.
21. CEV shall include the capability for contingency EVA's
22. Radiation shielding shall be incorporated into the design of the CEV and lunar lander crew modules to provide a core level of biological protection for the crew during transit and on the lunar surface (Code T to give guidance).

23. ***Low lunar orbit is used as the lunar vicinity rendezvous point to support near-polar surface access.***
24. ***Near-polar landing site locations range from 70-90 degrees north or south latitude.***
25. Communications and tracking systems will be emplaced to support global human and robotic surface operations.
26. ***The lunar lander will be transported with the crew from low Earth orbit to low lunar orbit.***
27. Assume LH2/LO2 propellants for the L1 transfer stage(s).
28. Assume CH4/LO2 propellants for all other propulsive stages.

4.3 Figures of Merit

The figures of merit for LDRM-2 Phase 1 and Phase 2 are identical, but are repeated for clarity.

Figures of Merit (FOMs) are provided for guidance in helping the analysis team develop the baseline DRM within the constraints listed above. Data from trades and analysis should support an independent FOM assessment. Note some FOM data has been identified as not required for this study.

4.3.1 Safety/Reliability

To what degree does an architecture ensure safety and productivity for all mission phases?

- Reliability estimates (Not required in this assessment)
- Design redundancy (For this study, only an assessment of functional redundancy between elements is required)
- Abort options for all mission phases
- Time required to return the crew to Earth at various key points in the mission in the event of a contingency.
- Identification of mission risks and system hazards
- Launch risks (Not required in this assessment)

4.3.2 Effectiveness and Evolvability

To what degree does an architecture provide flexibility to meet current mission and future mission needs?

- Applicability and evolvability of technologies, systems (life support, in-space propulsion, power), elements (CEV, landers, habitat, EVA suit, surface power, etc.), and operations of a lunar architecture to future Mars missions, and Mars mission risks that are retired.
- Assessment of degree to which the architecture allows for simple interfaces between elements.
- Assessment of architecture mission complexity (e.g. number of elements, docking and assembly requirements, total mission duration, launch and return opportunities, etc.).

- Assessment of capability to satisfy science objectives (not required for this assessment).

4.3.3 Development Risk and Schedule

To what degree does an architecture reduce development and schedule risks?

- New technologies used
- Benefits of the new technologies (either to lunar missions or as a development step to support Mars missions)
- Current TRL of new technologies, and assessment of effort required to bring technology to TRL 6 by 5 years prior to initial ops capability date
- Assessment of technologies used versus IOC date
- Assessment of ability to develop required architecture elements within integrated schedule (not required for this study)

4.3.4 Affordability

To what degree is an architecture expected to provide lower initial and total life cycle costs?

- New technologies identified
- Program flight elements, mass
- Program facility needs
- Identification of Program elements that will have fixed operating costs (e.g. sustaining engineering hardware production, ground and mission ops, etc.).
- Identification of Program elements that will have recurring cost for each mission (e.g. sustaining engineering hardware production, ground and mission ops, etc.)
- Identification of investments in Lunar missions that directly support future Mars missions (technologies, systems, elements)
- Total mass required to be delivered to LEO to support initial mission (includes pre-deployed/infrastructure, if any) and for each subsequent mission.

5.0 LDRM-2 Study Participants

5.1 Roles and Responsibilities by Organization

The LDRM-2 Phase 2 study leveraged the extensive discipline and subsystem design data generated by the entire LDRM-2 team during the Phase 1 effort. The following contributors provided direct inputs into the Phase 2 study.

Organization	*Function*	*Name*
HQ/ESMD	Task Lead	Bret Drake
EX	Study Lead	Ed Robertson
EX	Deputy Study Lead / Sizing	Jim Geffre
EX	Systems Integration Lead	Jon Lenius
DM	Mission Operations Consultation	Doug Rask
DM	Mission Operations Consultation	Gurpartap Sandhoo
EC	ECLSS Consultation	Kathy Daues
EC	ATCS Consultation	David Westheimer
EG	Mission Analysis Team Lead	Jerry Condon
EG	Orbital Phasing and Rendezvous	Robert Merriam
EG	Mission Design and Orbital Mechanics	Sam Wilson (retired consultant)
EG	Trajectory Analysis and Visualization	Carlos Westhelle
EG	GN&C Consultation	Tom Moody
EG – UT	Trajectory Design and Rendezvous	Dr. Juan Senent
EP	Power Consultation	Karla Bradley
EP	Propulsion Consultation	Eric Hurlbert
ES	TPS/PTCS Consultation	Steve Rickman
EV	Communications & Tracking Consultation	Laura Hood
EV	Avionics/DMS Consultation	Coy Kouba
EX	Systems Integration & Mass Properties	Wayne Peterson
EX	CEV Crew Module Layout	Ann Bufkin
EX	CEV Crew Module Layout	Liana Rodriggs
EX	Co-op Student, Functionality Matrix	Paul Dum

EX – LM	Information Management	Demetria Lee
SK	Radiation Analysis Team Lead	Frank Cucinotta
SD	Radiation Analysis	Mark Shavers
SF	Radiation Analysis	Neal Zapp
SX	MMOD Consultation	Eric Christiansen

5.2 Final Report Documentation

The following individuals authored or provided material contributions to sections of the LDRM-2 Phase 2 Final Report.

Section	*Description*	*Authors*
Section 1	Introduction	Ed Robertson
Section 2	Study Scope	Ed Robertson
Section 3	Executive Summary	Ed Robertson
Section 4	Task Description	Bret Drake
Section 5	LDRM-2 Study Participants	Ed Robertson
Section 6	Polar LOR Mission Design Considerations	Jerry Condon Sam Wilson (retired)
Section 7	Polar LOR Trade Reference Mission (TRM)	Jim Geffre
Section 8	Alternate Mission Capture #1 – LOR Global Access	Jim Geffre
Section 9	Alternate Mission Capture #2 – L1 CEV Dormant	Jon Lenius
Section 10	Alternate Mission Capture #3 – L1 CEV Active	Jon Lenius
Section 11	Alternate Mission Capture #4 – Mars Mission Staging	Jon Lenius
Section 12	Alternate Mission Capture #5 – Mars Mission Return	Ed Robertson

6.0 Polar LOR Mission Design Considerations

6.1 Introduction

Recent interest in the Moon as a stepping-stone for future robotic and human mission targets at Mars and beyond has revitalized the evaluation of concepts for establishing a sustained human presence on the Moon. An underlying assumed constraint for this mission profile is summed up in the phrase “anytime abort from the lunar surface.” Specifically it is taken to mean that a flight crew faced with a life-support system failure or a medical emergency at the landing site should not have to wait longer than three times the period of the lunar phasing or rendezvous orbit to initiate a lunar orbit departure (LOD) maneuver that will return them to Earth atmospheric entry and landing.

The Phase 2 study focuses on the performance (ΔV) cost of long duration, near-polar missions, and the applicability of such a mission design to envelope short duration excursions to other lunar sites. If a mission design allows for a long duration surface stay with anytime abort from the surface, at latitudes of 70° or higher (i.e., up to a polar landing site), then this performance capability also accommodates near equatorial long duration missions to landing site latitudes in the range of 0° to about 30°. This performance comparison applies to both the northern and southern lunar hemispheres. The performance cost for landing site latitude ranges of 70° to 90° and 0° to 30° also applies to -70° to -90° and 0° to -30°, respectively.

6.2 LOR Mission Design and Performance Overview

In support of the Phase 2 study, the performance costs of eight landing site latitude ranges for long duration (90-day) lunar surface stays with anytime surface abort and Earth return capability (i.e., no required lunar loiter) were analyzed. For the purposes of mission design and performance, any surface stay duration above 28-days, the duration of one lunar rotation around the Earth, will result in no additional performance cost as the full range of possible performance-impacting maneuver geometries is merely repeated. For missions with lunar landing site latitudes less than or equal to 50°, the missions include an equatorial (180°) lunar parking orbit, while missions with landing latitudes greater than 50° employ a polar (90°) lunar parking orbit. This approach helps minimize on-orbit plane change for a reduced overall ΔV cost.

Table 6.2-1 shows the mission constraints for a range of lunar missions. The eight LOR missions defined in the table individually enable access to portions of the lunar globe and, when considered as a whole, cover global lunar landing site access. Missions 1 through 4 employ an equatorial (180°) lunar parking orbit and missions 5 through 8 employ a polar (90°) lunar parking orbit. Missions 9 and 10 represent the L1 TRM and the LSR (direct return) missions developed for the Phase 1 study, and serve as a performance reference for the LOR missions. All of these missions accommodate an anytime lunar surface abort and immediate Earth return. All of these missions also accommodate a long duration (90-day) surface stay, although overall mission performance does not change for surface stays greater than approximately 28 days due to repeating of the orbital geometry after each 27.3 day lunar orbit period.

Missions 6 through 8 represent the primary near-polar missions for Phase 2 with mission 6 targeted to a lunar landing site of 70° latitude, mission 7 to 80° latitude, and mission 8 to a polar (90° latitude) landing site. Missions 1, 2 and 3 can be performed for the same approximate total ΔV that is required for the near-polar missions, and are designed for lunar landing site latitudes of 0°, 15°, and 30°, respectively. Mission 4 performance covers all landing site latitudes from equatorial through 50°. Mission 5 covers all landing sites with latitudes greater than 50°.

	MISSION	1	2	3	4	5	6	7	8	9	10
CONSTRAINTS	**Mode**	LOR	LOR	LOR	LOR	LOR	LOR	LOR	LOR	**LPR**	**LSR**
	Launch	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit
	Lander	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable
	Landing Site Latitude	0°	**15°**	**30°**	**≤50°**	**≥50°**	**70°**	**80°**	**90°**	Global Access	Global Access
	Surface Stay Time	90d	90d	90d	90d	90d	90d	90d	90d	90d	90d
	Comment	Lunar parking orbit inclination of 180°. No loiter required in lunar orbit.				Lunar parking orbit inclination of 90°. No loiter required in lunar orbit.				Phase 1 TRM	Min. ΔV - No Lunar Loiter

Table 6.2-1: Mission Constraints for Ten Selected Missions

6.3 Detailed LOR Mission Design

Mission performance sensitivities to variations in the definition of the CEV lunar parking orbit and changes in the allocations of major maneuvers among the flight elements were assessed for the lunar orbit rendezvous architecture. The eight LOR missions that are described in this section were defined based on minimum total mission ΔV, which is primarily driven by the altitude and orientation of the CEV lunar parking orbit and the associated plane change required to provide access to a specified range of latitudes on the lunar surface. Initially, based on minimum total ΔV considerations, a 3000 km lunar parking orbit was selected for missions 3 and 6 described in this section. It is important to note, however, that minimum initial mass in low Earth orbit (IMLEO) is not always achieved by minimizing the total mission ΔV. Mass and ΔV data for a range of mission designs is provided in Section 7.0, and the mass results for polar LOR missions with surface access up to 70° north or south latitude were found to favor the use of a 100 km rather than a 3000 km lunar parking orbit. As a result, missions 3 and 6 for the vehicle sizing were designed around the use of a 100 km orbit.

The designs for missions 1 through 10 contain worst-case parameter values to provide robust mission capabilities while enveloping the full range of possible orbital alignments. Furthermore, the EOD maneuvers for all ten missions are coplanar, which is close to optimal and considered to be acceptable for an architecture trade study. Table 6.3-1 provides a quick look at the mission profile characteristics for missions 1 through 10 as designed for minimum overall ΔV. The performance costs for missions 1 through 8 are designed to support a robust mission with anytime lunar surface departure and anytime Earth return. The performance available for polar and near-polar missions can also accommodate equatorial and near-equatorial lunar landing site missions. For this second phase of the study, long duration global lunar landing site access is achieved via a dual mission design approach. For missions to landing site latitudes (with an absolute value of) of 50° or higher, a 90° LPO design minimizes the overall ΔV cost. For landing site latitudes from the equator to (an absolute value of) 50°, an equatorial LPO provides a minimum the equator to (an absolute value of) 50°, an equatorial LPO provides a minimum ΔV cost. This dual mission design approach provides an overall reduced ΔV cost for long duration global lunar landing site access in comparison to the equatorial LPO design approach described in the Phase 1 mission design results. This approach provides insight into the required ΔV costs to achieve a particular landing site latitude (or range of latitudes), which can be used in vehicle sizing algorithms to assess the relative size of the spacecraft associated with the various missions.

Mission	1	2	3	4	5	6	7	8	9	10
EPO Altitude	407 km									200 km
EM (EL) Transfer Orbit Inclination w.r.t. Earth Equator	28.7°									28.5°
	Optimum ERO Rendezvous								***	72° Launch Azimuth
EM (EL) Transfer Orbit Inclination w.r.t. EM Plane	57.28°									22°
	Moon @ Orbit Node with Maximum Inclination									Moon @ Orbit Apex With Minimum Inclination
EOD ΔV	3074 m/s								3057 m/s	3125 m/s
LPA ΔV	***								889 m/s	***
LPD ΔV	***								244 m/s	***
LOI V_{inf}	986 m/s								***	893 m/s
	4.0 Day Xfer Time, Moon @ Perigee								3.5 Day Xfer - EPO to L1, 2.3 Day Xfer - L1 to LPO	4.0 Day Xfer Time, Moon @ Perigee
LPO Altitude	100 km	100 km	3000 km	3000 km	3000 km	3000 km	100 km	100 km	100 km	100 km
LPO Inclination	180°				90°				Select	Select
	Designed for Minimum LOA and LOD ΔV and Minimum Descent Plane Change Cost				Designed for Minimum Descent Plane Change Cost				Selectable LPO Incl. for Min. LOI and LOD ΔV	Tailored to Latitude of Lunar Landing Site
Relative Delcination of Arrival V_{inf}	19°				0°				***	0°
	Worst Case Arrival Plane Change				Minimum Arrival Plane Change				***	***
LOI ΔV (# Impulses)	978 m/s (3-impulse)	978 m/s (3-impulse)	879 m/s (1-impulse)	879 m/s (1-impulse)	727 m/s (1-impulse)	727 m/s (1-impulse)	878 m/s (1-impulse)	878 m/s (1-impulse)	631 m/s	843 m/s (1-impulse)
Descent Plane Change 1881 m/s Coplanar	0°	15°	30°	50°	40°	20°	10°	0°	0°	0°
	Plane Change ΔV = 0 m/s	Plane Change ΔV = 428 m/s (Worst Case)	Plane/Alt. Change ΔV = 850 m/s (Worst Case)	Plane/Alt. Change ΔV = 1113 m/s (Worst Case)	Plane/Alt. Change ΔV = 981 m/s (Worst Case)	Plane/Alt. Change ΔV = 726 m/s (Worst Case)	Plane Change ΔV = 285 m/s (Worst Case)	Plane Change ΔV = 0 m/s	Descent to 100 km Phasing Orbit w/ 0° Plane Change	Xfer Direct to 100 km Phasing Orbit w/ 0° Plane Change
Ascent Plane Change 1834 m/s Coplanar	0°	15°	30°	50°	40°	20°	10°	0°	0°	0°
	Plane Change ΔV = 0 m/s	Plane Change ΔV = 428 m/s (Worst Case)	Plane/Alt. Change ΔV = 850 m/s (Worst Case)	Plane/Alt. Change ΔV = 1113 m/s (Worst Case)	Plane/Alt. Change ΔV = 981 m/s (Worst Case)	Plane/Alt. Change ΔV = 726 m/s (Worst Case)	Plane Change ΔV = 285 m/s (Worst Case)	Plane Change ΔV = 0 m/s	Plane Change ΔV = 0 m/s	Plane Change ΔV = 0 m/s
EVP Altitude	38 km (Apollo 17)									
ME (LE) Transfer Orbit Inclination w.r.t. Earth Equator	40° - For Favorable Landing Latitude									
ME Transfer Orbit Inclination w.r.t. EM Plane	36.22° - Moon @ Orbit Apex w/ Minimum Inclination In Lunar Cycle (18.6 years)									
LOD V_{inf}	952 m/s								***	952 m/s
	3.5 Day Xfer Time for Earth Landing Longitude Control								2.3 Day Xfer - LOD to L1, 3.5 Day Xfer - L1 to Earth Entry	3.5 Day Xfer Time for Earth Landing Lon. Ctrl.
LOD2 ΔV	***				***				631 m/s	***
LPA2 ΔV	***				***				241 m/s	***
LPD2 ΔV	***				***				800 m/s	***
Relative Declination of Departure V_{inf}	19°				90°				***	0°
	Worst Case Departure Plane Change From Lunar Parking Orbit with Inclination = 180°				Worst Case Departure Plane Change From Lunar Parking Orbit with Inclination = 90°				***	Preferred Launch Orientation
LOD ΔV (# Impulses)	966 m/s (3-impulse)	966 m/s (3-impulse)	864 m/s (1-impulse)	864 m/s (1-impulse)	1202 m/s (3-impulse)	1202 m/s (3-impulse)	1410 m/s (3-impulse)	1410 m/s (3-impulse)	***	865 m/s (1-impulse)
Comment	Landing Site Latitude Range				Non-Minimum ΔV - No Lunar Loiter				Phase 1 TRM	Min. ΔV - No Lunar Loiter

Table 6.3-1: Mission Profile Characteristics for Ten Selected Missions

6.3.1 Missions 1, 2, 7, and 8 with Overall Characteristics of Missions 1 through 8

The constraints for missions 1, 2, 7, and 8, with respective lunar landing site latitude targets of 0°, 15°, 80°, and 90°, are listed in Table 6.3-1, with a "bat chart" of the mission profile in shown in Figure 6.3.1-1. The grouping of these missions is based on the common 100 km circular altitude employed for the lunar parking orbit. All ten missions shown in Table 6.3-1 employ a long duration 90-day surface stay with enough performance to accommodate anytime lunar surface abort and anytime Earth return.

All of the LOR missions (1 through 8) begin from a circular 407 km, 28.7° inclination Earth construction parking orbit. For these missions, the construction orbit supports a 4-launch on-orbit assembly targeted to provide a nominal CEV Earth departure. As in the original study, the multiple launch sequence design targets a departure orbit with a favorable right ascension of the ascending node (RAN) for the EOD maneuver. The 28.7° inclination provides an Earth launch window of reasonable duration from a near minimum cost (i.e., near due east) Cape Canaveral launch latitude of 28.5°.

All LOR missions are designed to provide a maximum 57.3° geocentric plane change at lunar arrival. This worst-case plane change requirement arises from a lunar arrival coincident with a transfer orbit nodal crossing at an orientation that results in a required geocentric plane change of the sum of the transfer orbit inclination (28.7°) and the lunar inclination with respect to the Earth equator (28.6°). In general, a ground launch-based sequence to a LEO construction orbit provides mission designers with some ability to select a (future) LEO angular momentum vector that minimizes the geocentric transfer angle between the outbound transfer orbit and the lunar orbit about the Earth. This minimum lunar arrival plane change can be about 10° for a 28.7° LEO inclination departure to a lunar arrival at nodal crossing and with a lunar inclination about the Earth's equator of about 18.3° (the minimum lunar inclination in the 18.6-year lunar cycle). When the Moon is at its maximum inclination in the lunar cycle (i.e., 28.6°), the geocentric lunar arrival plane change can be close to zero degrees. Unforeseen delays in on-orbit assembly of the flight elements could result in subsequent EOD opportunities with increased geocentric lunar arrival transfer angles. In any event, designing for the maximum plane change of 57.3° ensures that the spacecraft, once assembled in LEO, will possess sufficient performance to handle all possible lunar arrival plane changes for the Cape Canaveral launches.

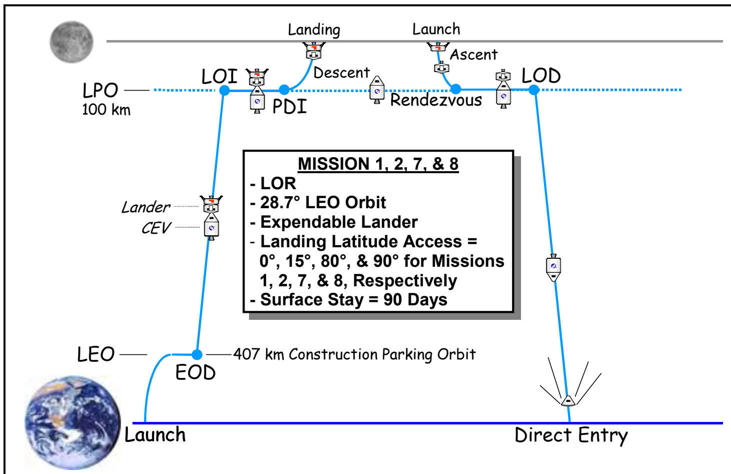


Figure 6.3.1-1. LOR Missions 1, 2, 7 and 8

As with the original study, all LOR missions employ a 4-day Earth-to-Moon transfer designed for a minimum ΔV transfer to the Moon with lunar arrival coincident with the Moon at the perigee of its orbit about the Earth. Arrival with the Moon at its perigee provides a worst-case performance requirement for vehicle design purposes. Each mission carries an EOD ΔV cost of 3074 m/s.

The 90° target LPO inclination for missions 7 and 8 supports polar and near-polar landing site latitudes (80°, and 90°) while an equatorial (180°) LPO inclination serves the equatorial and near equatorial landing site latitudes (0° and 15°).

All LOR missions carry a LOA V_∞ vector magnitude of 986 m/s. For missions 1, 2, 7, and 8, the selection of a 1-impulse versus a 3-impulse LOA maneuver sequence depends upon which approach provides the lower ΔV cost. A 3-impulse LOA maneuver for missions 1 and 2 inserts the spacecraft into an equatorial (180°) LPO at a ΔV cost of 978 m/s. The 19° relative declination of the arrival V_∞ vector represents a worst-case insertion into an equatorial 100 km altitude LPO. This worst-case relative declination stems from the sum of approximately 10° due to a worst-case angle (perpendicular) between the Moon's and the spacecraft's geocentric velocities at lunar arrival plus approximately 7° to account for lunar libration in addition to 2° to account for second order variations. For missions 7 and 8, a 0° relative declination of the V_∞ vector allows for a coplanar 1-impulse LOA ΔV of 878 m/s.

Following LOA, the lander spacecraft performs deorbit from a 100 km circular altitude LPO and, subsequently, powered descent to the lunar surface. For mission 1 with an equatorial LPO and an equatorial landing site, a coplanar descent maneuver sequence (i.e., deorbit and powered descent) to the surface carries a ΔV cost of 1881 m/s. For a 15° landing latitude in mission 2, a maximum 15° "fail-safe" plane change brings the total ΔV cost of the landing sequence to 2309 m/s. A fail-safe descent maneuver is designed to perform the plane change during the deorbit maneuver such that the transfer orbit periapse always remains positive. This approach ensures the safety of the crew in the event of an engine failure during the plane change maneuver. For mission 8 with a polar (+90° latitude) landing site, a 90° LPO provides an 1881 m/s coplanar descent maneuver sequence. For the 80° landing site latitude specified for mission 7, the maximum 10° fail-safe descent sequence results in a ΔV of 2166 m/s.

After the surface stay, the maximum plane changes (if any) applied to the descent sequence also apply to the ascent return to a 100 km circular LPO altitude. For mission 1, a coplanar ascent from an equatorial latitude back to an equatorial LPO costs a ΔV of 1834 m/s. This coplanar cost also applies to mission 8 with a polar (+90°) launch site to a polar LPO. For mission 2, a 15° maximum plane change requirement for a launch from a 15° latitude site to an equatorial orbit carries a 428 m/s additional plane ΔV cost for a total ascent cost of 2262 m/s. Similarly for mission 7, a 10° plane change from an 80° launch site to a polar LPO costs an additional ΔV of 285m/s for a total ascent cost of 2119 m/s. For all LOR missions, a 100 m/s lunar orbit departure rendezvous provides maneuvering capability to bring the Lander and the CEV together for docking and crew transfer.

For all LOR missions, a 3.5-day transfer from the Moon, at its orbit perigee, to Earth carries a lunar departure V_∞ vector magnitude of 952 m/s. The reduced flight time provides up to 180° Earth landing longitude control. For missions 1 and 2, a 3-impulse departure with a worst-case 19° V_∞ vector relative declination results in a LOD ΔV cost of 966 m/s. The 19° relative declination represents a worst-case departure plane change from lunar parking orbit with an equatorial inclination (180°). For missions 7 and 8, a worst-case orientation of the polar parking orbit results in a 90° declination of the V_∞ vector, resulting in a 3-impulse LOD ΔV cost of 1410 m/s. The 90° relative declination represents the worst-case departure plane change from a lunar parking orbit with an inclination of 90°. The LOD maneuver for all missions targets a 38 km (Apollo 17-based) vacuum periapse altitude at Earth.

All LOR missions are designed around a 40° ME transfer orbit inclination (w.r.t. the Earth equator). This results in a maximum 36.22° ME transfer orbit (w.r.t. the EM plane) for a departure from the Moon at its orbit apex (maximum latitude w.r.t. Earth) with the Moon at its minimum inclination in its 18.6-year lunar cycle. These maximum values ensure that the spacecraft has the capability to return to Earth in the +40° latitude range, allowing it to accommodate all possible Earth landing latitudes. In addition, this capability allows for lower latitude approaches, avoiding potentially dangerous approaches over the Earth's poles.

6.3.2 Missions 3, 4, 5, and 6

The constraints for missions 3, 4, 5, and 6, with respective lunar landing site latitude targets (or target ranges) of 30°, 50°, 50°, and 70°, are listed in Table 6.3-1, with a "bat chart" of the mission profile in shown in Figure 6.3.2-1. While missions 1, 2, 7, and 8 employ a 100 km circular orbit altitude LPO, missions 3, 4, 5, and 6 are based on a common 3000 km circular altitude LPO. Transfers between the lunar surface and this 3000 km altitude LPO employ a 100 km circular phasing orbit.

The 50° latitude landing site target serves as an approximate performance crossover point for the two LPO inclination mission options. For a 50° landing site latitude target, the total mission ΔV for mission 4, which employs an equatorial LPO, approximately equals that of mission 5, employing a polar LPO. Note that overall mission capability is increased for mission designs targeting mid-latitude landing sites. For example, mission 7 provides the capability to access landing sites at 80° to 90° latitudes whereas mission 8 can only support a polar (90° latitude) landing site. Similarly, mission 6 possesses enough performance to access landing sites at latitudes of 70° or greater.

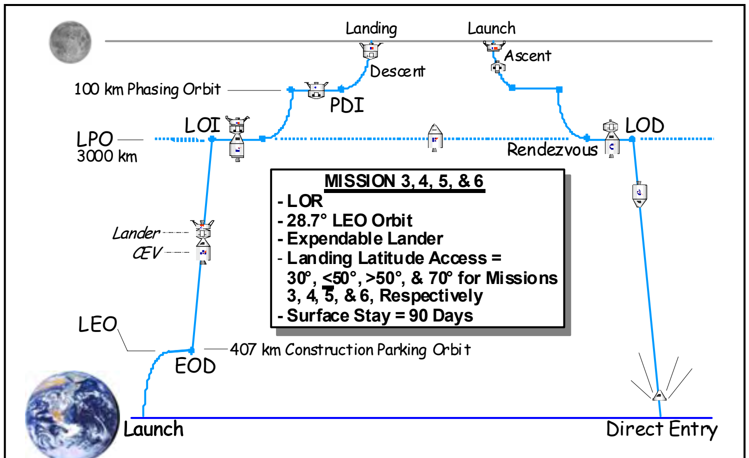


Figure 6.3.2-1. LOR Missions 3, 4, 5 and 6

Mission 5 possesses enough performance to support landing latitudes of 50° or greater using a polar LPO inclination. Similarly, mission 4 possesses enough performance for missions to landing site latitudes of 50° or less while employing an equatorial LPO.

All LOD missions carry a LOA V_∞ vector magnitude of 986 m/s. For all cases, the selection of a 1-impulse versus a 3-impulse LOA maneuver sequence depends upon which approach provides the lower ΔV cost. Missions 3, 4, 5, and 6 all employ a 1-impulse LOA maneuver to insert into a 3000 km circular altitude LPO. The LOA maneuver for missions 3 and 4, with a 19° relative declination of the V_∞ vector, inserts the spacecraft into an equatorial (180°) LPO at a ΔV cost of 879 m/s. For missions 5 and 6, a 0° relative declination of the V_∞ vector allows for a coplanar 1-impulse LOA ΔV of 727 m/s.

Following LOA, the lander spacecraft performs a transfer and plane change from a 3000 km circular altitude LPO to the 100 km phasing orbit and, subsequently, powered descent to the lunar surface. For mission 3, with an equatorial LPO and a 30° landing latitude constraint, an 850 m/s ΔV requirement enables transfer from the 3000 km circular LPO to the 100 km phasing orbit including a 30° fail-safe plane change. Once in the 100 km phasing orbit, the spacecraft performs a coplanar descent maneuver sequence (i.e., deorbit and powered descent) to the surface at a ΔV cost of 1881 m/s, for a total descent ΔV cost of 2731 m/s. For mission 4, the transfer cost from the 3000 km LPO to a 100 km phasing orbit with a 50° fail-safe plane change costs a ΔV of 1113 m/s. The addition of the 1881 m/s descent maneuver sequence brings the total descent ΔV cost for mission 4 to 2994 m/s. For mission 6, the transfer from the 3000 km polar LPO to the 100 km phasing orbit along with a 20° fail-safe plane change (i.e., plane difference between the 70° latitude constraint and the 90° inclination LPO) carries a ΔV cost of 726 m/s. Combining the 1881 m/s cost of final coplanar descent to the lunar surface with this transfer ΔV brings the total descent ΔV cost for mission 6 to 2607 m/s. Mission 5 carries a 981 m/s ΔV cost for the orbit to orbit transfer cost with a 40° degree fail-safe plane change. Adding the 1881 m/s coplanar cost for phasing orbit to surface landing to the transfer and plane change ΔV results in a total descent ΔV of 2862 m/s for mission 5. The total ΔV cost for missions 4 and 5 are within about 5% of each other, making the 50° latitude landing site a reasonable transition for the equatorial and polar LPO mission modes.

After the surface stay, the ascent plane change and transfer ΔV costs between the 100 km phasing orbit and the 3000 km LPO are the same as those for the descent phase. The coplanar ascent from the lunar surface to the circular 100 km altitude phasing orbit carries a ΔV cost of 1834 m/s (1815 m/s from the surface to a 100x18.5 km transfer orbit with a 19 m/s circulation maneuver). Adding the transfer and plane change costs to the 1834 m/s coplanar surface ascent ΔV produces a total ascent ΔV of 2684 m/s, 2947 m/s, 2815 m/s, and 2560 m/s for missions 3, 4, 5, and 6, respectively. Once in the 3000 km LPO an additional 100 m/s is allocated for rendezvous between the Lander and the CEV.

For the Earth return, missions 3 and 4 employ a 1-impulse departure to a 952 m/s V_∞ vector magnitude with a worst-case 19° V_∞ vector relative declination resulting in a LOD ΔV cost of 864 m/s. The 19° relative declination represents a worst-case departure plane change from lunar parking orbit with an equatorial inclination (180°). For missions 3 and 4, a worst-case orientation of the polar parking orbit results in a 90° declination of the V_∞ vector, resulting in a 3-impulse LOD ΔV cost of 1202 m/s. The 90° relative declination represents the worst-case departure plane change from a lunar parking orbit with an inclination of 90°. The LOD maneuver for all missions targets a 38 km (Apollo 17-based) vacuum periapse altitude at Earth.

6.3.3 Missions 9 and 10

The constraints, mission characteristics and performance data for missions 9 and 10 from the Phase 1 study are included for comparison purposes with the eight LOR missions and are listed in Table 6.3-1. Details of the mission designs for missions 9 and 10 are provided in section 8.0 of the Phase 1 report.

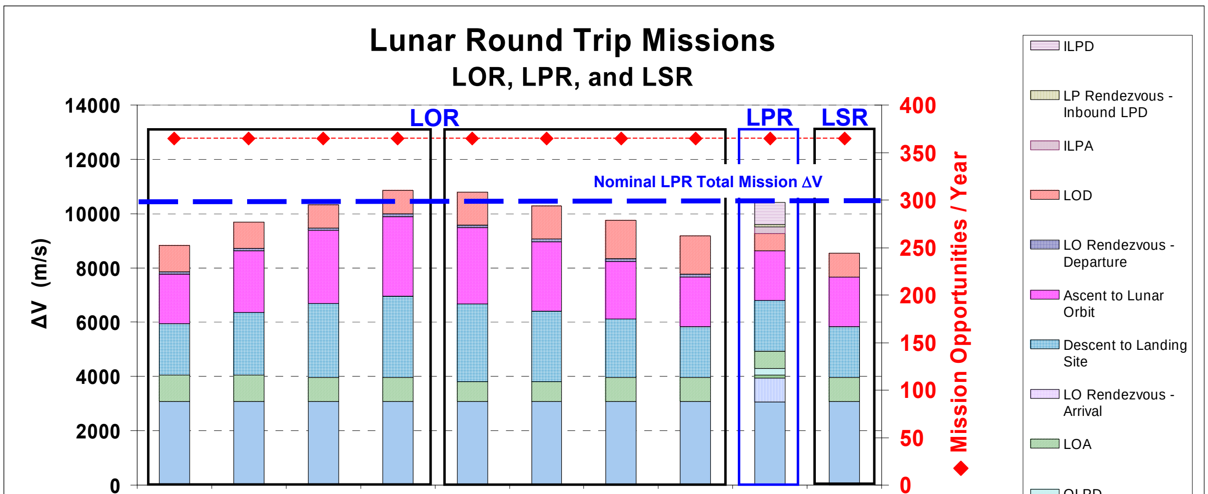
6.4 Mission Performance

The primary goal of the Phase 2 study was to assess the performance cost for a long duration LOR trade reference mission to near-polar lunar landing sites (within 20° latitude of the pole), and then assess the ability of the TRM performance requirement to also cover a short-stay mission with global lunar landing site access. For the sake of completeness, the LOR mission design portion of the Phase 2 study was expanded to assess long duration access to global lunar landing sites. The ΔV performance results are shown in graphical and tabular form in Figure 6.4-1 and Table 6.4-1, respectively. Note that the ΔV performance provides an initial guide to preferable mission design. The resulting vehicle mass sizing based on this ΔV data provides a better metric to evaluate the impact of different mission capabilities on the required initial mass in low Earth orbit (IMLEO).

The ΔV performance data represents an attempt to blend the desire for a minimum performance cost mission with one that can accommodate minimum on-orbit time and anytime surface launch for off-nominal situations (such as aborts). All of the ΔV data in Figure 6.4-1 and Table 6.4-1 represent a mission that possesses the capability to perform anytime abort off the lunar surface followed by a lunar orbit rendezvous with the CEV and an immediate Earth return. The results of this analysis show that in addition to enveloping the performance for a 4-day, global access mission, the near-polar TRM also envelopes the performance required for a 7-day, global access mission. From Table 6.4-1, a near-polar long duration mission capable of lunar landing site latitude magnitudes of 70° or higher comes at a total ΔV cost of 10270 m/s. This mission requires both the lunar descent and ascent stages to accommodate 20° plane changes. The Phase 1 study showed that the total ΔV cost for a short duration (i.e., 7-day surface stay) mission with global lunar landing site access and anytime surface and Earth return capability carries a total mission ΔV cost of 9906 m/s. Therefore, from the perspective of total mission ΔV, the near-polar LOR TRM also accommodates a 7-day surface stay mission to anywhere on the lunar surface. Note that the 20° descent and ascent plane change required by the near-polar mission exceeds the requirement for the short stay, global access mission (i.e., coplanar descent with a 6.7° ascent plane change).

For long duration global access, the maximum mission ΔV cost of 10858 occurs for the mid-latitude landing sites (around 50° latitude), which is only about 4% greater than the cost of the Phase 1 L1 TRM. The dual mission mode employs an equatorial (180° inclination) LPO for landing site latitudes with magnitudes between 0° and 50º and a polar (90º) inclination LPO for latitudes between 50º and 90º. The lowest overall mission ΔV cost occurs for the equatorial mission with a total ΔV of 8833 m/s.

For all long duration LOR missions, the lunar orbiting vehicle (CEV) remains in either a 100 km or a 3000 km circular altitude lunar orbit for the entire 90-day mission. At the lower altitude, periodic orbit maintenance may be required. This cost is not reflected in the ΔV summaries. The orbit lifetime data provided in Section 6.6 indicate a reduced orbital maintenance requirement (if any) for at the higher, more stable, 3000 km altitude.



MISSION	1	2	3	4	5	6	7	8	9	10
Mode	LOR	LOR	LOR	LOR	LOR	LOR	LOR	LOR	**LPR**	**LSR**
Launch	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit
Lander	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable
Landing Site Latitude	0°	**15°**	**30°**	**≤50°**	**≥50°**	**70°**	**80°**	**90°**	Global Access	Global Access
Surface Stay Time	90d	90d	90d	90d	90d	90d	90d	90d	90d	90d
Comment	Lunar parking orbit inclination of 180°. No loiter required in lunar orbit.				Lunar parking orbit inclination of 90°. No loiter required in lunar orbit.				LDRM-2 BRM	Min. ΔV - No Lunar Loiter
Available Lunar Launch Days / Available LOD Days / Days Spent in Lunar Vicinity	90 / 90 / 92	90 / 90 / 92	90 / 90 / 90	90 / 90 / 90	90 / 90 / 91	90 / 90 / 91	90 / 90 / 91	90 / 90 / 91	90 / 90/ 90	90 / 90 / 90

(CONSTRAINTS: Mode, Launch, Lander, Landing Site Latitude, Surface Stay Time)

Figure 6.4-1: ΔV cost, constraints, and availability for ten selected round-trip lunar mission scenarios for the Phase 2 study

During the 90-day surface stay, the ascent vehicle can lift off at any time, as indicated by the 90 days of "Available Lunar Launch Days", shown in Figure 6.4-1 and Table 6.4-1. After lunar orbit rendezvous, the crew transfers to the CEV, which can perform a lunar departure at anytime during the mission (as indicated by the 90 days of "Available LOD Days"). As previously mentioned, the CEV remains in orbit during the entire 90-day mission. The number of "Days Spent in Lunar Vicinity" for the CEV ranges from 90 to 92 days. The differences stem from the use of either a 1-impulse or a 3-impulse LOA and/or LOD. The 24-hour period of the transfer orbit used in the 3-impulse maneuver sequences adds corresponding days to the time spent in lunar vicinity.

All of the missions possess 365 launch opportunities per year. This daily launch opportunity stems from a pseudo-ground launch which places the initial hardware in a 4-launch sequence into the Earth rendezvous and construction parking orbit that is poised for favorable EOD following the launch and rendezvous of the fourth flight element (i.e., the CEV with crew). This multi-launch sequence depends upon the timeliness and completion of all launches in the sequence, but particularly the fourth launch. While the performance values, indicated in Figure 6.4-1 and Table 6.4-1 encompass a Moon-targeted EOD at any departure opportunity, they may not be sufficient to accommodate operations-related performance impacts due to deviation from the nominal maneuver sequence. For example, desirable lunar landing lighting conditions (at a particular landing site) associated with an on-time EOD may be lost due to delay and subsequent recycle to the next Earth departure opportunity. The performance cost to re-establish favorable lighting conditions at the same landing site can be high and is not included in this study. One alternative, however, would be to achieve the same lighting conditions by accepting a different landing site longitude.

	ΔV Requirement for Selected Lunar Missions (m/s)									
	Lunar Orbit Rendezvous								L1 Rendezvous	Lunar Surface Rendezvous
Mission Features / Flight Phase	28.7 Deg ERO Lch Expendable Lander Landing Lat = 0° Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Landing Lat = 15°** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Landing Lat = 30°** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Landing Lat ≤ 50°** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Landing Lat ≥ 50°** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Landing Lat = 70°** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Landing Lat = 80°** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Landing Lat = 90°** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander **Global Access** Surface Stay = 90 d	28.7 Deg ERO Lch Expendable Lander Global Access Surface Stay = 90 d
ILPD	0	0	0	0	0	0	0	0	800	0
LP Rendezvous - Inbound LPD	0	0	0	0	0	0	0	0	100	0
ILPA	0	0	0	0	0	0	0	0	241	0
LOD	966	966	864	864	1202	1202	1410	1410	631	865
LO Rendezvous - Departure	100	100	100	100	100	100	100	100	0	0
Ascent to Lunar Orbit	1834	2262	2684	2947	2815	2560	2119	1834	1834	1834
Descent to Landing Site	1881	2309	2731	2994	2862	2607	2166	1881	1881	1881
LO Rendezvous - Arrival	0	0	0	0	0	0	0	0	0	0
LOA	978	978	879	879	727	727	878	878	631	878
OLPD	0	0	0	0	0	0	0	0	244	0
LP Rendezvous - Arrival	0	0	0	0	0	0	0	0	100	0
OLPA	0	0	0	0	0	0	0	0	889	0
EOD	3074	3074	3074	3074	3074	3074	3074	3074	3057	3074
TOTAL	**8833**	**9689**	**10332**	**10858**	**10780**	**10270**	**9747**	**9177**	**10408**	**8532**
	EARTH DEPARTURE WINDOWS / YEAR									
	365	**365**	**365**	**365**	**365**	**365**	**365**	**365**	**365**	**365**
	Available Lunar Launch Days / Available LOD Days / Days Spent in Lunar Vicinity									
	90 / 90 / 92	**90 / 90 / 92**	**90 / 90 / 90**	**90 / 90 / 90**	**90 / 90 / 91**	**90 / 90 / 91**	**90 / 90 / 91**	**90 / 90 / 91**	**90 / 90/ 90**	**90 / 90 / 90**

	MISSION	**1**	**2**	**3**	**4**	**5**	**6**	**7**	**8**	**9**	**10**
CONSTRAINTS	**Mode**	LOR	LOR	LOR	LOR	LOR	LOR	LOR	LOR	**LPR**	**LSR**
	Launch	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit	28.7° Orbit
	Lander	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable	Expendable
	Landing Site Latitude	0°	**15°**	**30°**	**≤50°**	**≥50°**	**70°**	**80°**	**90°**	Global Access	Global Access
	Surface Stay Time	90d	90d	90d	90d	90d	90d	90d	90d	90d	90d
	Comment	Lunar parking orbit inclination of 180°. No loiter required in lunar orbit.				Lunar parking orbit inclination of 90°. No loiter required in lunar orbit.				LDRM-2 BRM	Min. ΔV - No Lunar Loiter

Table 6.4-1: ΔV cost, constraints, and availability for ten selected round-trip lunar mission scenarios for the Phase 2 study

As in the Phase 1 study, the LSR mission requires the minimum total ΔV for all missions shown on Figure 6.4-1 and Table 6.4-1 at a cost of 8532 m/s. Note that vehicle sizing work done in the Phase 1 study indicates that, while the LSR may have the lowest overall mission ΔV, it carries the highest IMLEO. The prime driver of this mass growth is the lander/ascent vehicle which must perform LOA, descent, ascent, and LOD. The need to transport the ascent and return propulsion system and Earth recovery systems to the lunar surface and back to orbit carries a substantial mass penalty. However, with the additional mass penalty come benefits such as reduced number of critical space maneuvers (including a rendezvous), reduced time (possibly on the order of minutes) for the crew to get to their backup habitat, and a general simplification of the mission.

6.5 Final Thoughts on LOR Mission Design

Given the objectives and operational constraints it was designed to meet, the Apollo mission profile would be hard to improve. However, major modifications are required before it can satisfy the objectives defined for this study. The Apollo landing sites were all situated within 30° of the lunar equator, and the lunar surface stay times were all shorter than a week. In the case at hand, the mission profile must provide the ability to land at any arbitrary site on the lunar surface, and to stay there a week or perhaps much longer while preserving the Apollo capability for anytime abort from the lunar surface (i.e., to a nearby backup habitat and thence to Earth).

Initial mass in low Earth orbit (IMLEO) is often used as a proxy for the monetary expense of a space mission in preliminary studies such as this one. IMLEO minimization usually is achieved by separating, sometime before final descent to the lunar surface, the assets needed by the flight crew while on the surface from those needed to transport them between the lunar vicinity and Earth. Before the mission can be accomplished and the flight crew returned to Earth, there must be a rendezvous – in a chosen locale near the Moon – between the separated assets.

No matter what locale is chosen for rendezvous, a selenocentric phasing orbit is required for economical access to an arbitrary landing site. The reason is essentially the same as that which applies to the launch of a lunar or interplanetary spacecraft from a site on the Earth surface. Although on-orbit plane-change penalties can be eliminated easily for such a launch (by choosing a launch azimuth and time of day such that the plane of the predeparture orbit will contain the required departure velocity vector), in the general case a coasting arc in a phasing orbit is required to avoid the penalty associated with non-optimal flight path angles during injection into the departure trajectory. Absent ΔV penalties associated with orbit plane or flight path angle, the total propulsive velocity increment is minimized by setting the altitude of the phasing orbit as low as possible – consistent with atmospheric drag effects in the case of the Earth, and terrain clearance in the case of the Moon. Because it is sometimes advantageous to let the phasing orbit serve also as the rendezvous orbit, estimation of terrain clearance must account for the long-term effect of large perturbations arising at low altitude from scattered concentrations of lunar mass.

For this study, the altitude of the selenocentric phasing orbit was chosen to be 100 km. The propulsive ΔV required for an in-plane round trip from that altitude to the lunar surface is a little more than 3700 m/s, which is greater than the EOD velocity increment by about 20%. Consequently, the IMLEO required for a lunar round trip can sometimes be reduced by leaving all assets not needed on the lunar surface – before descending to it – at the rendezvous locale, where they can be retrieved/reoccupied by the landing crew after ascending from the surface. The LOR and LPR trajectory profiles are designed to utilize such a scheme.

6.5.1 Lunar Descent/Ascent Plane Changes Associated with LOR

For stays shorter than about eleven days, the sum of descent/ascent plane-change angles – required for descent to the chosen landing time and for ascent at the most inopportune time(s) during the nominal surface stay period – can be minimized by orienting the rendezvous orbit so that its plane contains the landing site at landing time, and its apex (maximum latitude) in the landing-site hemisphere is a few degrees closer to the nearest pole than the landing site itself.

For longer stays, the minimum sum of plane-change angles is realized by choosing either an equatorial or a polar plane for the rendezvous orbit, depending on whether the landing site lies closer to the equator or to one of the poles. For either choice, landing exactly at latitude 45° (north or south) requires a plane change of 45° during transfers in both directions between rendezvous orbit and landing site.

Descent/ascent plane changes of this magnitude benefit from establishment of two separate phasing orbits (one for descent and the other for ascent), each having an altitude of 100 km and oriented independently so as to contain the lunar site at landing and at liftoff time, respectively. The rendezvous orbit (equatorial or polar) is established at a considerably higher altitude so that the major part of the plane change can be made where the orbital speed is lower. This allows any given plane change angle to be achieved with a smaller velocity increment, and facilitates further economy by allowing a change of orbital energy and of orbital plane to be accomplished simultaneously with a single impulse. The down side of this stratagem is that it adds 4 major maneuvers to the round-trip profile, together with a moderate increase in the time required for the Earth-Moon-Earth round trip.

The altitude chosen for the elevated rendezvous orbit is 3000 km, where the circular orbit period is approximately 8 hours. A higher altitude would provide a further reduction in the required descent/ascent propulsive velocity increments, but would increase the associated flight times, decrease the frequency of opportunities for transfers to and from the surface, and increase the susceptibility of the rendezvous orbit to earth and solar perturbations.

6.5.2 LOD Plane Changes Associated with LOR

The plane of a selenocentric orbit is stationary with respect to inertial space if it is polar, and is nearly stationary for any other orientation. Since the inertial rotation rate of the moon itself is about 13° per day, the plane of the LRO regresses at that rate in the selenographic frame, which is fixed with respect to surface features rather than inertial space.

Conservation of geocentric angular momentum in the Moon-Earth transfer orbit dictates that the selenographic declination of the departure V_∞ vector for any return-to-earth trajectory lies within the range of ±19°, after taking lunar libration into account. For reasonable flight times (on the order of 2.5 to 5.0 days) – no matter when lunar departure occurs – the selenographic longitude of such a vector lies within the approximate range of 30°-95°.

Said another way, the gist of the two preceding paragraphs is this: At LOD time the V_∞ vector to be achieved will always be confined within a quasi-rectangle that is bounded by the 19th parallels of north and south latitude and by the 30th and 95th meridians of longitude on the surface of the selenographic reference sphere. The LOD plane-change penalty will be moderate if the track of the LRO at that time passes through aforesaid rectangle. Otherwise it will be more severe, depending on the minimum angular distance between the orbit track and the perimeter of the rectangle.

If the rendezvous orbit plane coincides with the equator, its track will always pass through the center of the rectangle, the relative declination of the LOD V_∞ vector (i.e., the angle it makes with the orbit plane) will never exceed 19°, and the LOD plane-change penalty will be minimal. Otherwise, the relative declination of the LOD V_∞ vector at the most inopportune departure time (during a nominal surface stay longer than about 14 days) will be either 90° or the sum of 19° and the selenographic inclination of the rendezvous orbit (or the supplement of the inclination if the orbit is retrograde), whichever is smaller. (A retrograde rendezvous orbit is usually preferred when a choice is available, because it yields a smaller abort velocity increment for a nonstop return to Earth if, for some reason, the LOA maneuver cannot be executed at the planned time).

If the landing site lies more than 71° from the lunar equator, the relative declination of the LOD V_∞ vector that must be achieved for immediate departure will be equal to or very near 90° at one or more times during the longer stays required in this study. The attendant ΔV penalty is severe (with a 3-impulse maneuver sequence, on the order of 350 m/s for departure from a 3000 km rendezvous orbit, or 450 m/s for departure from a 100 km orbit), and appears to be unavoidable if LOR is used to satisfy the operational requirements previously described.

6.5.3 LOA Plane Changes Associated with LOR

In contrast to lunar orbit departure, the lunar orbit insertion plane-change penalty associated with the lunar orbit rendezvous trajectory profile is minimal if the rendezvous orbit plane is equatorial, or nil if it is polar.

The ΔV requirements determined in this study for transferring between landing sites and a polar rendezvous orbit are based on the assumption, in each case, that the ascending node of the rendezvous orbit on the lunar equator lies at the worst possible location it could have for landing at or launching from the site under consideration. This means the node location can always be chosen so that the V_∞ vector, at LOA time, will lie in the rendezvous orbit plane and therefore there will be no plane-change penalty at all associated with LOI.

No comparable node can be defined for an equatorial orbit, but (as pointed out earlier in the discussion of the LOD maneuver) the declination of the LOA V_∞ vector relative to an equatorial orbit can never exceed 19°. The associated ΔV penalty is typically on the order of 350 m/s for a 1-impulse LOA maneuver if the rendezvous orbit altitude is 100 km, but only about 150 m/s if the orbit altitude is 3000 km, or 100 m/s if a 3-impulse maneuver sequence with an intermediate 24-hour ellipse is used for insertion into the 100 km orbit.

6.6 Lunar Rendezvous Orbit Stability

The Apollo missions employed a low lunar orbit rendezvous altitude on the order of 111 km (60 n. mi.). This approach worked well for a short-duration stand-alone mission to a low-latitude landing site. However, future missions almost certainly will require longer stay times at arbitrary lunar surface sites.

In the case of LOR missions requiring global access and a long stay time, an altitude of 3000 km was chosen for the circular rendezvous orbit. Selection of this particular altitude reflects a compromise among competing mission parameters including the magnitude of plane-change ΔV penalties, orbit stability and lifetime, phasing opportunities, and efficient division of ΔV requirements between the LTV and the LLV.

A major benefit of the higher altitude is that it provides a significant reduction of the total ΔV penalty for orbital plane changes that are required in both nominal and emergency (abort) maneuvers. This is offset to some extent by the longer orbit period, which reduces the frequency of phasing opportunities for transfers to and from the lunar surface. Also, there is a practical limit to the altitude of the rendezvous orbit, which arises from the performance burden that it shifts from the LTV to the LLV. The burden imposed by the 3000 km orbit is a marginally tolerable compromise, and the resulting 8.1 hour orbital period provides a reasonable frequency of phasing opportunities. The second major benefit of the higher altitude lies in the increased stability of the rendezvous orbit. The more stable orbit provides a lower maintenance ΔV requirement and a certain element of added safety to a crew in the event of a propulsion system failure. For a low altitude (e.g., 100 km) the complex seleno-potential model can cause instability in the orbit and, without timely performance of orbit maintenance, surface impact.

A lunar parking orbit lifetime analysis provided insight to the stability of potential lunar parking orbits for selected ranges of initial altitude, inclination, and right ascension of the ascending node (RAAN), over a 365-day propagation. This study employed a Goddard Lunar Gravity Model 2 (GLGM2) of degree and order 35. The analysis used a 9th order Runge-Kutta-Verner integration, with 8th order error control, to propagate the trajectories. No lunar atmosphere or solar radiation models were used. Other point mass gravity fields used in the propagations included the Earth and the Sun. Analysis results provided in Tables 6.6-1 through 6.6-4 confirm instability and possible surface impact for a 100 km parking orbit in as little as 83 days, for certain inclination and RAAN combinations. In the mid-range of altitudes, between 500 and 5000 km, no surface impacts occur during the 1-year propagation. For the cases examined, the 3000 km circular orbit altitude revealed a relatively stable maximum altitude variation of +9.1% to –5.4%. For the higher orbit altitudes (10,000 and 20,000 km), some surface impacts again occurred during the 365-day propagation with the addition of cases where the orbit migrated back to a geocentric orbit. The stability of these higher orbits appears to be much more significantly affected by the gravitational effects of the Earth and Sun. A detailed data set follows.

With regard to LPR Missions, the inherently unstable nature of the cislunar libration point, L1, has been demonstrated in past studies.2,3 Depending on the duration of uncontrolled flight and without active station-keeping maneuvers, a spacecraft will drift away from L1 in a relatively short time (possibly a few months). However, past studies show station-keeping ΔV costs at the libration point to be small (on the order of 1 m/s/y).[1,2]

The lunar parking orbit lifetime analysis examines the result of 365-day propagations of circular lunar parking orbits of selected initial altitude, inclination, and RAAN. Specifically, a sample space of 336 propagations consisted of parking orbits with the following ranges of initial conditions:

1. Circular orbit altitude = 100, 500, 1000, 3000, 5000, 10000, 20000 km
2. Inclination = 0, 30, 60, 90 degrees

3. Right ascension of the ascending node = 0, 30, 60, 90, 120, 150, 180, 210, 240, 270, 300, 330 degrees

The propagation study employed a Goddard Lunar Gravity Model 2 (GLGM2) of degree and order 35 combined with a 9th order Runge-Kutta-Verner integration, with 8th order error control. While no lunar atmosphere or solar radiation models were used, point mass Earth and Sun models supplemented the gravity fields used in the propagations. Data tables are provided for the final epoch, maximum altitude, and minimum altitude. The final epoch reflects the time of propagation before impact on the lunar surface. For cases with no impacts during the entire propagation, the epoch is 365 days. Shaded cells reflect and impact occurring during the propagation. The maximum and minimum altitude tables show the altitude extremes that occurred during the propagation. Extremely large maximum altitudes (on the order of 6x105 km) indicate that the trajectory has departed lunar space and entered geocentric space.

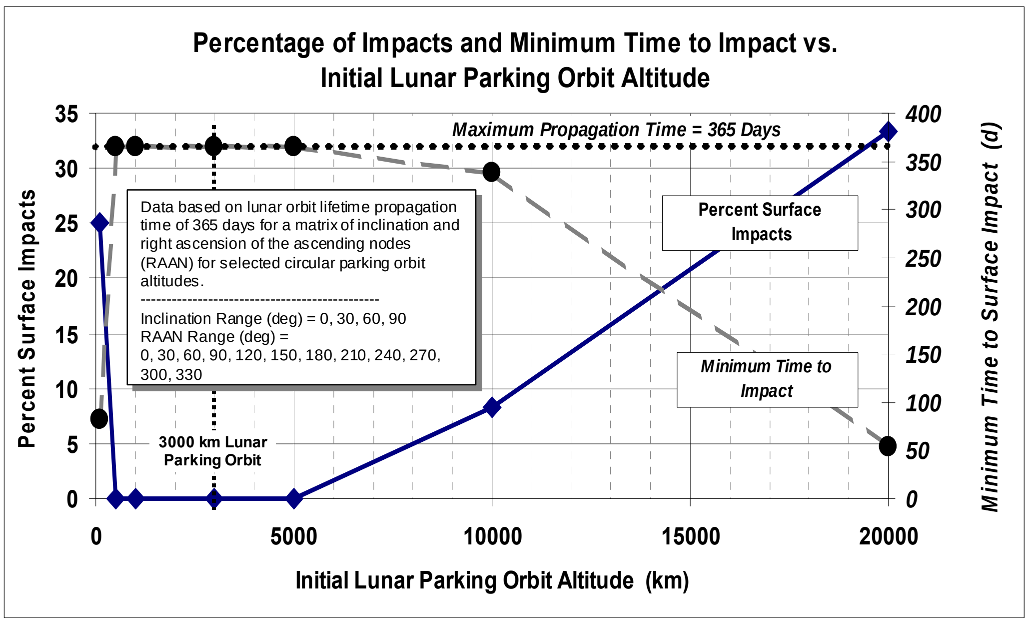


Table 6.6-1: Lunar circular parking orbit stability over a 1-year period for selected orbit altitudes

		FINAL EPOCH (d) -- MAXIMUM PROPAGATION TIME = 1 YR											
		RAAN (deg)											
Init. Alt. (km)	**Inc(deg)**	**0**	**30**	**60**	**90**	**120**	**150**	**180**	**210**	**240**	**270**	**300**	**330**
20000	**0**	365	365	365	365	365	365	365	365	365	365	365	365
20000	**30**	365	365	341	365	365	54	365	365	365	204	365	245
20000	**60**	365	365	365	365	365	365	365	365	365	365	365	365
20000	**90**	97	138	98	102	106	101	107	113	111	117	96	94
10000	**0**	365	365	365	365	365	365	365	365	365	365	365	365
10000	**30**	365	365	365	365	365	365	365	365	365	365	365	365
10000	**60**	365	365	365	365	365	365	365	365	365	365	365	365
10000	**90**	342	365	365	350	365	339	365	365	365	365	365	365
5000	**0**	365	365	365	365	365	365	365	365	365	365	365	365
5000	**30**	365	365	365	365	365	365	365	365	365	365	365	365
5000	**60**	365	365	365	365	365	365	365	365	365	365	365	365
5000	**90**	365	365	365	365	365	365	365	365	365	365	365	365
3000	**0**	365	365	365	365	365	365	365	365	365	365	365	365
3000	**30**	365	365	365	365	365	365	365	365	365	365	365	365
3000	**60**	365	365	365	365	365	365	365	365	365	365	365	365
3000	**90**	365	365	365	365	365	365	365	365	365	365	365	365
1000	**0**	365	365	365	365	365	365	365	365	365	365	365	365
1000	**30**	365	365	365	365	365	365	365	365	365	365	365	365
1000	**60**	365	365	365	365	365	365	365	365	365	365	365	365
1000	**90**	365	365	365	365	365	365	365	365	365	365	365	365
500	**0**	365	365	365	365	365	365	365	365	365	365	365	365
500	**30**	365	365	365	365	365	365	365	365	365	365	365	365
500	**60**	365	365	365	365	365	365	365	365	365	365	365	365
500	**90**	365	365	365	365	365	365	365	365	365	365	365	365
100	**0**	365	365	365	365	365	365	365	365	365	365	365	365
100	**30**	365	365	365	365	365	365	365	365	365	365	365	365
100	**60**	103	105	103	83	84	87	89	90	90	93	97	101
100	**90**	365	365	365	365	365	365	365	365	365	365	365	365

Table 6.6-2: Lunar Circular Orbit Lifetime – Final Epoch Data

		MAXIMUM ALTITUDE (km) -- MAXIMUM PROPAGATION TIME = 1 YR											
		RAAN (deg)											
Init. Alt. (km)	**Inc(deg)**	**0**	**30**	**60**	**90**	**120**	**150**	**180**	**210**	**240**	**270**	**300**	**330**
20000	**0**	666965	666965	666965	666965	666965	666965	666965	666965	666965	666965	666965	666965
20000	**30**	35075	676734	693896	36944008	675780	54839	667537	661332	26408433	662156	49794695	676691
20000	**60**	39406	40292	41794	42088	36813	30549	28853	37772	44173	45080	45348	41773
20000	**90**	40918	41159	41748	42591	40801	41294	40387	39798	40832	41597	40412	40357
10000	**0**	12615	12615	12615	12615	12615	12615	12615	12615	12615	12615	12615	12615
10000	**30**	12827	12874	13067	13043	12699	12368	12295	12605	13076	13421	13387	13064
10000	**60**	20757	20440	19856	18921	17908	17223	17661	18963	20223	20941	21145	21007
10000	**90**	20017	19758	19758	20143	16147	19953	19440	13518	10991	20059	19535	19538
5000	**0**	5603	5603	5603	5603	5603	5603	5603	5603	5603	5603	5603	5603
5000	**30**	5735	5697	5671	5641	5602	5567	5564	5608	5690	5776	5819	5790
5000	**60**	6802	6765	6645	6419	6183	6046	6069	6271	6526	6693	6773	6801
5000	**90**	5025	5021	5024	5043	5013	5017	5005	5017	5011	5032	5019	5011
3000	**0**	3246	3246	3246	3246	3246	3246	3246	3246	3246	3246	3246	3246
3000	**30**	3266	3260	3253	3243	3232	3222	3222	3232	3251	3267	3274	3271
3000	**60**	3275	3279	3273	3256	3235	3219	3217	3229	3245	3256	3263	3269
3000	**90**	3008	3009	3013	3015	3013	3011	3010	3008	3007	3006	3008	3008
1000	**0**	1073	1073	1073	1073	1073	1073	1073	1073	1073	1073	1073	1073
1000	**30**	1075	1074	1072	1070	1070	1071	1074	1076	1078	1078	1077	1076
1000	**60**	1040	1041	1044	1047	1048	1050	1051	1050	1049	1046	1043	1040
1000	**90**	1025	1027	1027	1027	1027	1028	1023	1019	1018	1020	1022	1022
500	**0**	569	569	569	569	569	569	569	569	569	569	569	569
500	**30**	577	574	572	568	567	569	572	578	584	586	582	579
500	**60**	577	579	588	596	599	600	599	600	601	596	586	577
500	**90**	517	520	520	520	520	523	518	513	512	514	516	515
100	**0**	191	191	191	191	191	191	191	191	191	191	191	191
100	**30**	195	190	196	187	192	188	180	194	213	214	206	200
100	**60**	209	210	211	210	209	209	209	208	210	210	210	209
100	**90**	140	134	132	131	135	130	138	147	149	143	142	150

Table 6.6-3: Lunar Circular Orbit Lifetime – Maximum Altitude Data

		MINIMUM ALTITUDE (km) -- MAXIMUM PROPAGATION TIME = 1 YR (Impact Tolerance Alt. +/- 1km)											
		RAAN (deg)											
Init. Alt. (km)	**Inc(deg)**	**0**	**30**	**60**	**90**	**120**	**150**	**180**	**210**	**240**	**270**	**300**	**330**
20000	**0**	7816	7816	7816	7816	7816	7816	7816	7816	7816	7816	7816	7816
20000	**30**	15112	8983	0	8287	1431	0	11428	6935	8440	0	1703	0
20000	**60**	6064	6160	5560	5794	8652	13703	15582	8091	4782	2829	2729	4135
20000	**90**	0	0	0	0	0	0	0	0	0	0	0	0
10000	**0**	9819	9819	9819	9819	9819	9819	9819	9819	9819	9819	9819	9819
10000	**30**	9244	9250	9144	9184	9483	9744	9784	9502	9108	8784	8776	9043
10000	**60**	428	788	1437	2357	3310	3950	3502	2226	1060	378	131	218
10000	**90**	0	180	233	0	4018	0	707	6456	9000	0	711	370
5000	**0**	4982	4982	4982	4982	4982	4982	4982	4982	4982	4982	4982	4982
5000	**30**	4767	4810	4845	4878	4912	4939	4939	4899	4824	4741	4696	4714
5000	**60**	3480	3522	3646	3874	4104	4250	4228	4015	3765	3599	3524	3502
5000	**90**	4971	4978	4977	4961	4990	4980	4993	4982	4992	4976	4985	4988
3000	**0**	2993	2993	2993	2993	2993	2993	2993	2993	2993	2993	2993	2993
3000	**30**	2940	2947	2957	2967	2977	2985	2985	2975	2958	2943	2936	2937
3000	**60**	2843	2839	2846	2863	2884	2900	2902	2889	2874	2863	2856	2850
3000	**90**	2990	2989	2987	2986	2987	2988	2989	2990	2994	2995	2992	2990
1000	**0**	987	987	987	987	987	987	987	987	987	987	987	987
1000	**30**	977	978	980	982	982	980	978	975	974	974	975	976
1000	**60**	990	989	986	983	982	980	979	979	981	984	987	990
1000	**90**	975	973	974	974	973	972	976	981	982	980	979	978
500	**0**	467	467	467	467	467	467	467	467	467	467	467	467
500	**30**	453	456	460	463	464	462	459	452	447	446	449	452
500	**60**	441	439	430	422	418	418	418	417	417	422	432	441
500	**90**	483	480	480	480	479	477	482	487	488	486	484	485
100	**0**	29	29	29	29	29	29	29	29	29	29	29	29
100	**30**	23	29	23	31	26	30	38	23	5	4	13	18
100	**60**	1	1	1	1	1	1	1	1	1	1	1	1
100	**90**	59	65	68	68	64	69	61	51	50	56	58	49

Table 6.6-4: Lunar Circular Orbit Lifetime – Minimum Altitude Data

REFERENCES

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3. Condon, G. L., Wilson, S. W., Ranieri, C. L., "Performance Requirements and Kick-stage Disposal Options for a Cislunar Gateway Station Transfer Vehicle" Proceedings of the International Conference on Libration Point Missions and Applications, World Scientific Publishing (ISBN 981-238-363-8), Aug 2003.

7.0 Polar Lunar Orbit Rendezvous Trade Reference Mission

Phase 2 of the LDRM-2 requirements formulation study called for the establishment of a new trade reference mission (TRM) to enable long-duration expeditions to the Moon to prepare for the eventual exploration of Mars and beyond. The Phase 1 TRM used the cislunar libration point as a rendezvous point to enable global lunar access and anytime return for missions up to 7 days on the surface. However, for the eventual missions needed to demonstrate technologies and operations applicable to Mars, crews will need to live and work on the surface for several months at a time and will revisit pre-emplaced surface assets such as a habitat over the course of multiple missions. Preliminary analysis indicates that the polar regions of the Moon may be an ideal location for such missions to be based. Therefore, with less emphasis placed on the global access capability offered by staging through libration points, a new TRM is selected for Phase 2 studies, one that uses lunar orbit rendezvous and polar parking orbits for missions to the lunar polar regions lasting up to 90 days per mission. The TRM provides a point of departure from which architecture trades can be performed. This section describes the new trade reference mission, referred to as the polar LOR TRM, and outlines the key assumptions made in the establishment of the TRM. Architecture trades performed to select a specific trade reference architecture and mission timeline, including considerations of various allocations of major maneuvers among elements, are also described.

Critical events for the polar lunar orbit rendezvous TRM are also outlined in this section along with their ranking by criticality, mission abort options are identified, and vehicle mass properties are described. This section investigates mission launch strategies of two, three, and four launches per mission, as well as an architecture where all elements are limited to launching on a vehicle capable of delivering 25 t to orbit. Finally, the cost of anytime return from the surface of the Moon is determined, and a different Earth orbit departure strategy from the TRM is explored.

7.1 Major Assumptions/Clarifications

This section outlines the major architecture assumptions made in formulation of the Phase 2 trade reference mission. The study's NASA HQ customer levied these assumptions, listed in the Phase 2 Requirements Formulation Task RFT 0001.04, on the LDRM-2 study team. Assumptions added or changed from the LDRM-2 Phase 1 TRM are listed in *italics*.

- One human lunar mission per year
- Return mass from the moon is 100 kg
- Payload to lunar surface is 500kg
- All mission elements placed in LEO (28.5^{o} 407 km circular)
- 4-launch solution
- Consider the lunar mission elements to be "cargo" in terms of delivery to the LEO parking orbit
- Automated rendezvous and docking shall be used to assemble the elements

- Assume 2 weeks between launches
- Crew must be launched on a human-rated launch system
- A dedicated lunar lander element with a separate crew module will be used to transfer the crew from the lunar vicinity to the lunar surface and back to lunar vicinity
- *Lunar surface stay time up to 90 days*: The Phase 2 task was developed to examine potential crew transportation architectures for missions up to 3 months in duration on the lunar surface. This mission class, called "Spiral 3" in the Project Constellation program plan, will establish the capability to conduct routine human missions on the surface of the Moon to test out technologies and operational techniques for expanding human presence to Mars and beyond. The Phase 1 effort focused on architectures for "Spiral 2" missions, which are relatively short-stay (7 days) missions to the lunar surface.
- 4 crew with all crew going to the lunar surface
- Daily EVAs will be conducted on the surface of the Moon from the Lunar Lander
- The CEV and Lunar Lander are not required to be reusable and will not be explicitly designed for reusability
- *The Lunar Lander will be designed to provide 4 days of independent operating capability on the lunar surface*: Spiral 3 missions envision the use of pre-emplaced habitation elements to enable long-stay human expeditions on the Moon, and the Lander will be used to deliver crews to and from those elements. A 4-day operating capability is sufficient for the crew to transfer from their landing site to the habitat and at the end of the mission, return to the Lander and depart from the Moon.
- *Lunar orbit is used as the lunar vicinity rendezvous point to enable near-polar landing site access between 70 and 90 degrees latitude*: The Phase 1 L1 TRM used the L1 libration point to provide global surface access for short-stay exploration missions. Based on the results of that study and other previous lunar exploration study efforts, though, lunar orbit rendezvous can provide architecture mass savings over L1 rendezvous for limited-access long-stay missions. Therefore, the preferred Phase 2 mission approach is lunar orbit rendezvous with a polar lunar parking orbit.

  Near-polar missions between 70 and 90 degrees latitude were selected because the lunar poles offer high scientific interest, specifically because of the potential for the presence of ice and the existence of deepest known exposed crater near the South Pole (South Pole-Aitken Basin). Polar landing locations also provide a surface environment which is more similar, though not identical, to the surface of Mars, and more benign than equatorial lunar locations. Polar missions can therefore provide an analog for system, technology, and operational testing for future Mars missions. In addition, polar landing locations, when combined with polar staging and rendezvous, provide a balanced approach for enabling anytime return of the crew.
- *The reference lunar surface environment is a lunar polar thermal and lighting condition*: The vehicles will be designed for the thermal and lighting conditions provided by limited-access polar exploration missions. Future Lunar Lander design efforts should exam-

ine the added cost of developing a vehicle capable of operating anywhere at any time on the lunar surface.

- The CEV will provide the crew habitation function from Earth's surface to lunar vicinity and back to Earth's surface
- The nominal Earth return for the CEV is a direct entry with a water landing
- The CEV design will incorporate functionality for land landing as a contingency for an ascent abort
- CEV shall include the capability for contingency EVAs
- Radiation shielding shall be incorporated into the design of the CEV and Lunar Lander crew modules to provide a core level of biological protection for the crew during transit and on the lunar surface
- Communications and tracking systems will be emplaced to support global human and robotic surface operations
- *The Lunar Lander will mate with the CEV in LEO prior to departing Earth orbit*: As in the Phase 1 lunar orbit rendezvous architecture variation, the CEV and Lunar Lander assemble in low Earth orbit and transit to the Moon as a single combined element stack.

7.2 Architecture Description

Based on the assumptions made above, the polar LOR trade reference mission is driven primarily by the following architecture decisions:

1. Four launches per mission
2. Separate in-space transportation (CEV) and landing (Lunar Lander) vehicles
3. Up to 90 days on the lunar surface
4. Four crew with all crew going to the surface
5. Latitude-restricted lunar surface landing access
6. Direct Earth entry
7. CEV/Lunar Lander assembly in LEO
8. CEV/Lunar Lander rendezvous in lunar orbit

Operating within this framework, the Phase 2 LDRM-2 task effort developed architecture mass estimates for a number of architecture sub-trades including different lunar parking orbit (LPO) altitudes, landing site latitudes, and delta-V allocation between elements. Possible orbit altitudes for the circular polar orbit included 100, 500, and 3,000 km above the lunar surface, where 100 km represents the approximate altitude used in the Apollo missions. For each of these points, vehicle estimates were generated for architectures limited to polar (90^{o}) landings, 80^{o} – 90^{o} latitude landings, and 70^{o} – 90^{o} landings. Near-equatorial missions using a 100 km altitude equatorial parking orbit were also considered. This section describes the polar LOR baseline and trades performed in support of its formulation.

7.2.1 Polar LOR Trades and Results

The Phase 2 task uses the same complement of flight elements developed for the Phase 1 analyses. This includes a CEV (comprised of a Command and Service Module) to deliver crew to lunar vicinity and return them to Earth, a Lunar Lander to transport the crew from lunar vicinity to the surface and back again, and two Earth Departure Stages (EDS1 and EDS2) to depart low Earth orbit. The Earth Departure Stages have equal total masses to minimize the required launch vehicle payload delivery capability. As with the Phase 1 LOR variant, the Kick Stage from the L1 TRM is omitted. The trades performed in support of the polar LOR TRM examined how certain maneuvers in the mission could be allocated among this suite of vehicles. The first major maneuver, Earth orbit departure, is allocated in all trade options to EDS1 and EDS2. For the first delta-V allocation trade, lunar orbit insertion, options include (1) EDS2 and (2) the CEV (ala Apollo CSM). Next, to access off-polar landing sites (70^o – 90^o latitude) from the polar parking orbit without waiting for the landing site to pass under the orbit plane, a descent plane change up to 20^o may be required. The Lander Descent Stage performs this maneuver in all trade options. It was not efficient for the CEV to do the plane change for the Lander because the propellant necessary to change the orbital plane for both the CEV and the Lander would be much greater than required for just the Lander. In addition, if the CEV was used for the descent plane change, it would be left to loiter in a 70^o LPO, and to support an anytime ascent later in the mission, a worst-case 40^o plane change could be required for the Lander to rendezvous with the CEV. Leaving the CEV in a polar LPO only requires a worst-case 20^o plane change. It may be possible to use EDS2 to perform the descent plane change, thereby reducing the Lander size, however that option was left for further trade studies.

In-plane powered descent and ascent is allocated to the Lunar Lander, but like the descent plane change, a similar 20^o ascent plane change may be required. That burn is traded between (1) the Lander, (2) the CEV, and for the 100 km LPO, (3) both elements capable of executing the maneuver. Finally, the CEV performs lunar orbit departure.

All Delta-V's in m/s

	100 km LPO						**500 km LPO**			**3,000 km LPO**		
latitude	**0°**	**<15°**	**<30°**	**>70°**	**>80°**	**90°**	**>70°**	**>80°**	**90°**	**>70°**	**>80°**	**90°**
Earth Orbit Departure	3,104	3,104	3,104	3,104	3,104	3,104	3,104	3,104	3,104	3,104	3,104	3,104
Lunar Orbit Insertion	978	978	978	878	878	878	834	834	834	727	727	727
Descent Plane Change	0	428	855	567	285	0	432	187	0	142	177	0
Descent	1,881	1,881	1,881	1,881	1,881	1,881	2,034	2,034	2,034	2,465	2,465	2,465
Ascent	1,834	1,834	1,834	1,834	1,834	1,834	1,987	1,987	1,987	2,418	2,418	2,418
Ascent Plane Change	0	428	855	567	285	0	432 / 514 *	187 / 258 *	0	142 / 353 *	42 / 177 *	0
Lunar Orbit Departure	966	966	966	1,410	1,410	1,410	1,340	1,340	1,340	1,202	1,202	1,202
Total for Mission (m/s)	**8,763**	**9,619**	**10,473**	**10,241**	**9,677**	**9,107**	**10,163 / 10,245**	**9,673 / 9,744**	**9,299**	**10,200 / 10,411**	**10,000 / 10,135**	**9,916**

Table 7.2.1-1: Assumed Delta-V's for Polar LOR Trade Options

Table 7.2.1-1 lists the maneuver delta-V's (in m/s) assumed for the altitude/landing latitude/delta-V allocation trades. As expected, the lowest delta-V's are seen with missions that are restricted to land either exactly at the poles or the equator. As the landing site moves toward mid-latitudes, descent and ascent plane changes become significant, particularly for low-altitude parking orbits. It should also be noted that two values are listed for the ascent plane change with both the 500 km and 3,000 km LPO. The first value refers to the mission cost if the Ascent Stage performs the ascent plane change, and the second value is the cost if the CEV performs the burn. The Ascent Stage has a lower required delta-V because it can combine the plane change burn with a burn to circularize its orbit. The vector summation of the two maneuvers performed simultaneously is less than the sum of the two when performed separately. The CEV, on the other hand, is already in a circular orbit at the parking altitude, therefore if it performs the ascent plane change, it will be a pure plane change maneuver and hence have a higher delta-V cost. The ascent plane change cost for the CEV and Ascent Stage with a 100 km LPO is assumed to be identical as the cost of orbit circularization relative to the plane change is minor (19 m/s vs. 567 m/s for a 70° landing site), so a combined maneuver here provides little delta-V savings.

Given these delta-V's and the vehicle configurations from the Phase 1 L1 TRM, the Envision sizing tool was used to evaluate total architecture mass for the different trade options. The results of these trades are in Table 7.2.1-2. It is immediately evident that allocating the lunar orbit insertion (LOI) maneuver to the CEV instead of the Earth Departure Stage results in an 8-10% increase in total architecture mass. Coupled with the strong desire to minimize the size of the CEV's human-rated launch vehicle, the mass savings possible with using EDS2 for LOI warrants the selection of that option. Using the EDS instead of the CEV also decouples the mass of the CEV from the mass of the Lander and any possible future mass growth of that vehicle. That is particularly important for a serial vehicle development program where the CEV would be designed and built long before the Lunar Lander is completed. *For the polar LOR trade reference mission, the Earth Departure Stage will perform lunar orbit insertion.*

Next, Table 7.2.1-2 shows that for missions constrained to 70^o – 90^o landing sites as dictated by the LDRM-2 Phase 2 task statement, using a 100 km lunar parking orbit requires a total architecture mass equivalent to or lower than a 500 km or 3,000 km orbit. The total architecture mass increases from 184 t to 184 t to 189 t with increasing orbit altitude when the Lander performs the ascent plane change, and from 178 t to 180 t to 189 t when the CEV performs it (EDS2 performs LOI). The 100 km LPO also enables the lowest Lunar Lander mass (and therefore size). This is particularly desirable because the Lander will drive launch shroud diameter for the cargo launch vehicle and launch vehicles with smaller shrouds have better performance, controllability, and cost. Landing stability considerations for landing on the Moon also favor a smaller Lander. However, very low altitude lunar parking orbits such as this will require station-keeping during the 3-month stay on the surface to prevent orbit degradation and eventual impact. The ideal parking orbit altitude is likely somewhere between 100 km and 500 km. *For the polar LOR trade reference mission, a 100 km altitude orbit will be used as the lunar parking orbit.*

Finally, the trade for allocation of the ascent plane change shows that some mass savings are possible by using the CEV instead of the Ascent Stage to perform that maneuver – 178 t vs. 184 t (EDS2 performs LOI, 100 km orbit). This also provides a smaller Lander size and its associated benefits. However, operational considerations might favor the ascent plane change being allocated to the Lander Ascent Stage as that vehicle, not the CEV, has people on-board when the crew is lifting off the lunar surface. If the uncrewed CEV is unable to perform the plane change when needed, the crew could be stranded in lunar orbit. This scenario only applies, though, when the ascent plane change is needed, which would be after an emergency ascent. A nominal mission scenario would plan for an in-plane ascent. The ideal delta-V allocation for the ascent plane change would be for both the CEV and the Ascent Stage to be capable of the maneuver. The architecture mass cost for both vehicles to have this capability for a 100 km LPO is approximately 10% or 19 t (178 t increases to 197 t). Launch vehicle limitations will determine if this is feasible and cost-effective, and until future studies can determine this, the CEV is used. The cost of allocating the ascent plane change to both vehicles was not assessed for the 500 km and 3,000 km orbits because a plausible scenario here would likely require the CEV to not only perform a plane change but also go into a lower orbit to meet the Ascent Stage. *For the polar LOR trade reference mission, the ascent plane change is allocated to the CEV.*

Note: All Masses in 1000's of Kilograms

			100 km LPO						**500 km LPO**			**3,000 km LPO**			**L1**	
	Delta-V Allocation		**0°**	**<15°**	**<30°**	**>70°**	**>80°**	**90°**	**>70°**	**>80°**	**90°**	**>70°**	**>80°**	**90°**	**0°**	**90°**
MASS	**LOI/LPA -> EDS Asc Pln Chg -> Lander**	**Arch Mass**	144	171	207	184	165	149	184	167	155	189	180	177	211	211
		CEV Mass	18	18	18	20	20	20	20	20	20	19	19	19	18	18
		Lander Mass	28	37	50	40	34	28	42	35	31	46	43	42	51	51
		EDS Mass	49	58	70	62	55	50	61	56	52	62	59	58	71	71
	LOI/LPA -> EDS Asc Pln Chg -> CEV	**Arch Mass**	144	165	190	178	163	149	180	166	155	189	182	177	211	211
		CEV Mass	18	20	24	25	23	20	24	22	20	21	20	19	18	18
		Lander Mass	28	32	38	34	31	28	37	33	31	44	43	42	51	51
		EDS Mass	49	56	64	60	55	50	60	56	52	62	60	58	71	71
	LOI/LPA -> EDS Asc Pln Chg -> Both	**Arch Mass**	144	178	224	197	171	149	TBD	TBD	TBD	TBD	TBD	TBD	N/A	N/A
		CEV Mass	18	20	24	25	23	20								
		Lander Mass	28	37	50	40	34	28								
		EDS Mass	49	61	75	66	57	50								
	LOI/LPA -> CEV Asc Pln Chg -> Lander	**Arch Mass**	159	187	227	203	182	164	202	183	170	204	195	192	226	226
		CEV Mass	38	42	47	45	43	41	44	41	40	40	39	39	46	46
		Lander Mass	28	37	50	40	34	28	42	35	31	46	43	42	51	51
		EDS Mass	46	54	65	59	53	48	58	53	50	59	56	55	65	65
	LOI/LPA -> CEV Asc Pln Chg -> CEV	**Arch Mass**	159	184	214	200	181	164	200	183	170	207	198	192	226	226
		CEV Mass	38	45	53	51	45	41	48	44	40	43	41	39	46	46
		Lander Mass	28	32	38	34	31	28	37	33	31	44	43	42	51	51
		EDS Mass	46	53	61	58	52	48	58	53	50	60	57	55	65	65

Note: All Delta-V's in m/s

			100 km LPO						**500 km LPO**			**3,000 km LPO**			**L1**	
	Delta-V Allocation		**0°**	**<15°**	**<30°**	**>70°**	**>80°**	**90°**	**>70°**	**>80°**	**90°**	**>70°**	**>80°**	**90°**	**0°**	**90°**
DELTA-V	**LOI/LPA -> EDS Asc Pln Chg -> Lander**	**Arch Delta-V**	9.0	9.8	10.7	10.5	9.9	9.3	10.4	9.9	9.5	10.4	10.2	10.1	10.5	10.5
		CEV Delta-V	1.1	1.1	1.1	1.6	1.6	1.6	1.5	1.5	1.5	1.4	1.4	1.4	1.0	1.0
		Lander Delta-V	3.7	4.6	5.5	4.9	4.3	3.7	4.9	4.4	4.1	5.2	5.0	4.9	5.5	5.5
		EDS Delta-V	4.1	4.1	4.1	4.0	4.0	4.0	4.0	4.0	4.0	3.9	3.9	3.9	4.1	4.1
	LOI/LPA -> EDS Asc Pln Chg -> CEV	**Arch Delta-V**	9.0	9.8	10.7	10.5	9.9	9.3	10.5	10.0	9.5	10.6	10.4	10.1	10.5	10.5
		CEV Delta-V	1.1	1.6	2.0	2.1	1.9	1.6	2.0	1.8	1.5	1.7	1.5	1.4	1.0	1.0
		Lander Delta-V	3.7	4.2	4.6	4.3	4.0	3.7	4.5	4.2	4.1	5.1	5.0	4.9	5.5	5.5
		EDS Delta-V	4.1	4.1	4.1	4.0	4.0	4.0	4.0	4.0	4.0	3.9	3.9	3.9	4.1	4.1
	LOI/LPA -> CEV Asc Pln Chg -> Lander	**Arch Delta-V**	9.0	9.8	10.7	10.5	9.9	9.3	10.4	9.9	9.5	10.4	10.2	10.1	10.5	10.5
		CEV Delta-V	2.1	2.1	2.1	2.4	2.4	2.4	2.3	2.3	2.3	2.1	2.1	2.1	1.9	1.9
		Lander Delta-V	3.7	4.6	5.5	4.9	4.3	3.7	4.9	4.4	4.1	5.2	5.0	4.9	5.5	5.5
		EDS Delta-V	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1
	LOI/LPA -> CEV Asc Pln Chg -> CEV	**Arch Delta-V**	9.0	9.8	10.7	10.5	9.9	9.3	10.5	10.0	9.5	10.6	10.4	10.1	10.5	10.5
		CEV Delta-V	2.1	2.5	3.0	3.0	2.7	2.4	2.8	2.6	2.3	2.4	2.3	2.1	1.9	1.9
		Lander Delta-V	3.7	4.2	4.6	4.3	4.0	3.7	4.5	4.2	4.1	5.1	5.0	4.9	5.5	5.5
		EDS Delta-V	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1	3.1

Table 7.2.1-2: Vehicle Sizing Results for Polar LOR Trade Options

Figure 7.2.1-1 graphically illustrates the vehicle sizing results for the polar LOR TRM trade options. Total architecture mass is plotted as a function of orbit altitude, landing site latitude, and ascent plane change allocation. For all options, EDS2 performs lunar orbit insertion. While architecture mass is smallest with low altitude orbits for near-equatorial or near-polar landing sites, the *rate* of architecture mass increase moving away from the poles actually decreases for higher-altitude orbits (the curves "flatten"). This is entirely due to the higher plane change cost with lower altitude orbits. In a sense, L1 rendezvous architectures, which cost approximately the same for any landing site latitude, can be considered as merely very high altitude parking orbits. Following the architecture mass trends to mid-latitude sites, it appears that libration point rendezvous is more competitive with LOR for long-stay missions to the Moon.

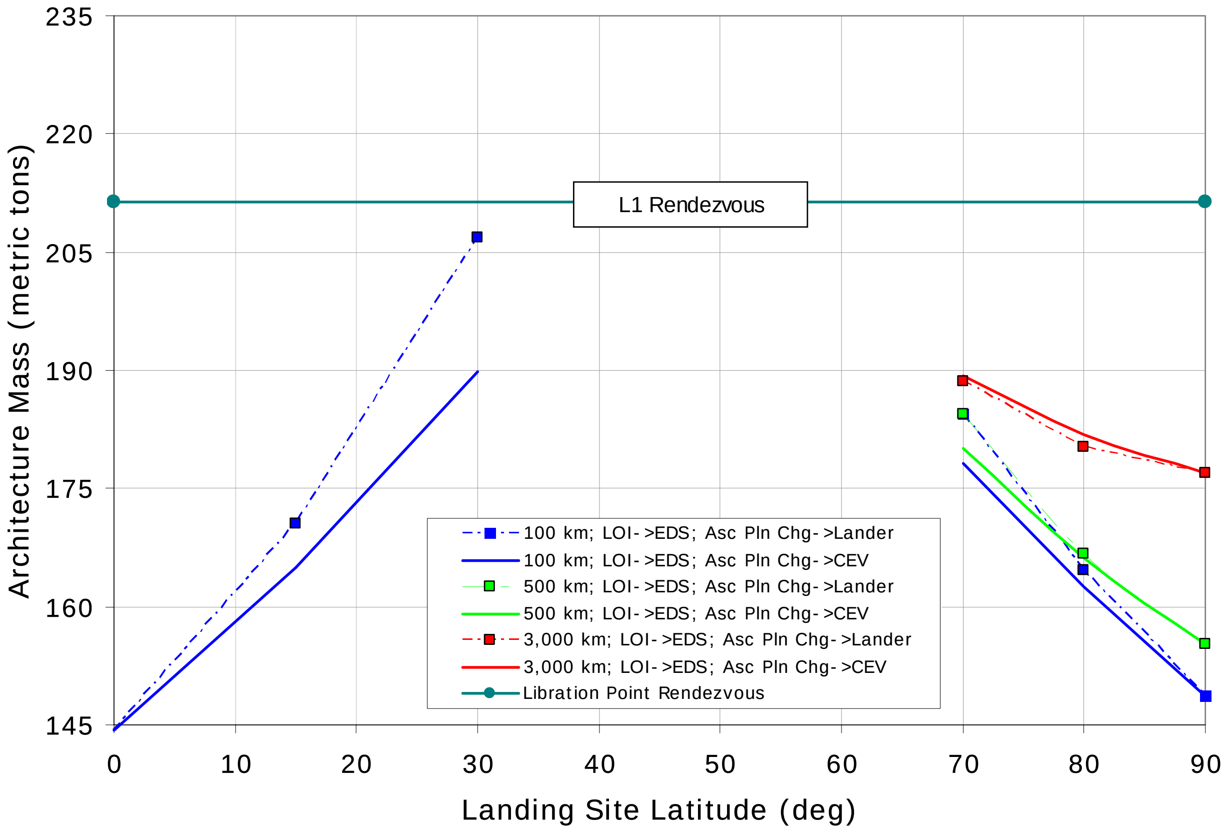

Figure 7.2.1-1: Vehicle Sizing Results for Polar LOR Trade Options

7.2.2 Trade Reference Mission Description

The LOR mission begins with two consecutive launches of equal-mass Earth Departure Stages with two-week spacing between launches. The assumed cargo launch vehicle for the architecture delivers the elements to the LEO parking orbit previously assumed (28.5° 407 km), where they loiter for assembly. Two weeks after the second launch, the Lunar Lander is delivered to LEO by a third cargo launch vehicle. The Lander performs a variable-length double coelliptic rendezvous maneuver profile to rendezvous and dock with the assembled Earth Departure Stages (the target vehicle) within 50 hr after launch. Recall that for the Phase 1 L1 TRM, a Kick Stage was included and it launched with the Lunar Lander. The Kick Stage performed the libration point arrival, libration point departure, and lunar orbit insertion maneuvers for the Lander. With LOR, there are no libration point-related maneuvers and the EDS, the CEV, or the Lander can perform lunar orbit insertion. Therefore, this variant does not need a Kick Stage.

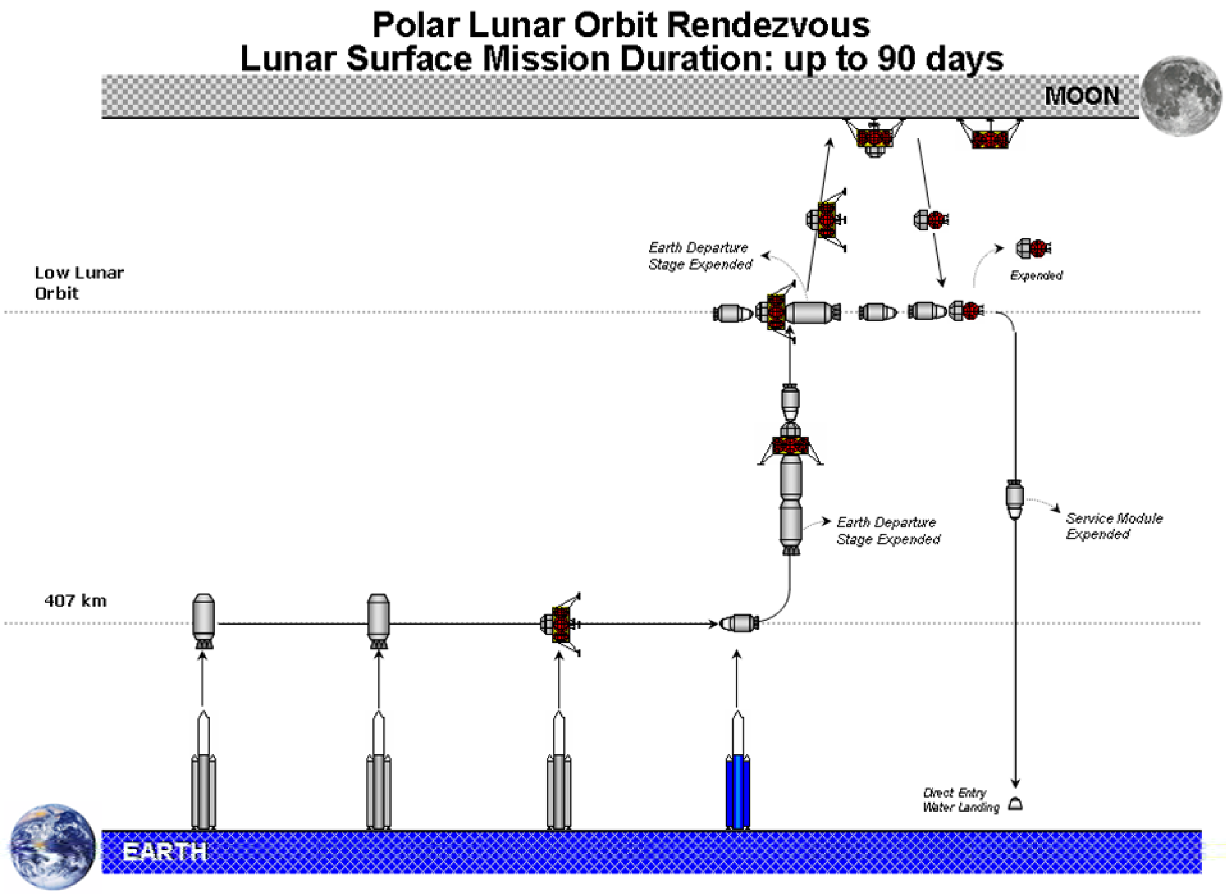

Figure 7.2.2-1: Polar LOR Trade Reference Mission Architecture Illustration

Finally, two weeks after the Lander, the crew launch in the CEV (the 4th launch of a four launch per mission architecture) on a separate, human-rated launch vehicle. The CEV, as the chaser vehicle, performs a stable orbit rendezvous maneuver profile to rendezvous and dock with the Lander and EDS's within 50 hr after orbit insertion. Once the CEV mates with the assembled stack in LEO, the crew and mission control check out the vehicles and the first EDS performs part of the Earth orbit departure maneuver (48%) at the opening of the window. That stage separates, disposes itself, and the second EDS ignites to complete the burn. The selected Earth-to-Moon trajectory is a near-minimum delta-V transfer with a flight time of 96 hr. A 24-hr minimum delta-V injection window has been included in the sizing of the Earth Departure Stages so flight time to lunar orbit may vary between 108 hr for injection at the opening of the window to 84 hr for injection at window closing. At perilune, the Earth Departure Stage will insert the CEV and Lunar Lander into a polar 100 x 100 km lunar parking orbit.

After successful insertion into the lunar parking orbit, the Earth Departure Stage separates from the Lunar Lander and CEV and disposes itself via lunar impact. The crew then transfers to the Lander, checks out the vehicle, and undocks from the CEV. Lander descent plane change (up to 567 m/s), deorbit, and powered descent (1,881 m/s) follows undocking at the first available opportunity. The lunar surface exploration strategy for the Phase 2 TRM is different from L1 TRM in that the Lander is used here to deliver the crew to the vicinity of pre-emplaced habitation elements. After landing, the crew prepares to transfer from the Lander over to these assets and "safes" the vehicle for 3 months of unoccupied loiter time on the lunar surface. The Lander is only designed for 4 days of independent operation capability on the Moon, so any power needed to support the vehicle for the other 86 days on the surface is assumed to be provided by a "to be determined" surface asset. As the surface mission is expiring, the crew will return to the Lander from the habitat and prepare the Ascent Stage for return to lunar orbit. The Ascent Stage separates from the Descent Stage on the lunar surface and ascends (1,834 m/s) to a 100 x 100 km parking orbit. Nominally, ascent will occur when the landing site passes under the CEV's orbit plane, but the CEV also includes an extra 567 m/s of delta-V for a 20^{o} plane change with an emergency ascent. Arriving in lunar orbit, the Ascent Stage performs a series of rendezvous maneuvers to re-dock with the CEV within 6 hr.

After docking, the crew transfers back over to the CEV to start up and check out the vehicle, transfers over any cargo returned to Earth, and undocks from the Lander Ascent Stage. The CEV then executes a 3-impulse, 24-hr sequence of maneuvers to return to Earth independent of parking orbit alignment. After 90 days on the surface, the lunar parking orbit may not be aligned for minimum-energy Earth return and may require a plane change to meet the anytime return requirement. In a worst-case, the plane may be up to 90^{o} out of alignment. Here, the first impulse of the sequence raises the apolune altitude to create a 24-hr period orbit, the second performs the plane change at apolune where it is most efficient, and the third burn departs the Moon and targets the CEV for Earth atmospheric entry 96 hr later. The end of the CEV mission – separation of the Command and Service Modules followed by direct entry – is the same as in the Phase 1 L1 trade reference mission. The Ascent Stage, left unoccupied in low lunar orbit, is disposed on the lunar surface.

Figure 7.2.2-2 and Tables 7.2.2-1 – 7.2.2-2 outline the assumed timelines and delta-V's for the polar LOR trade reference mission as described above.

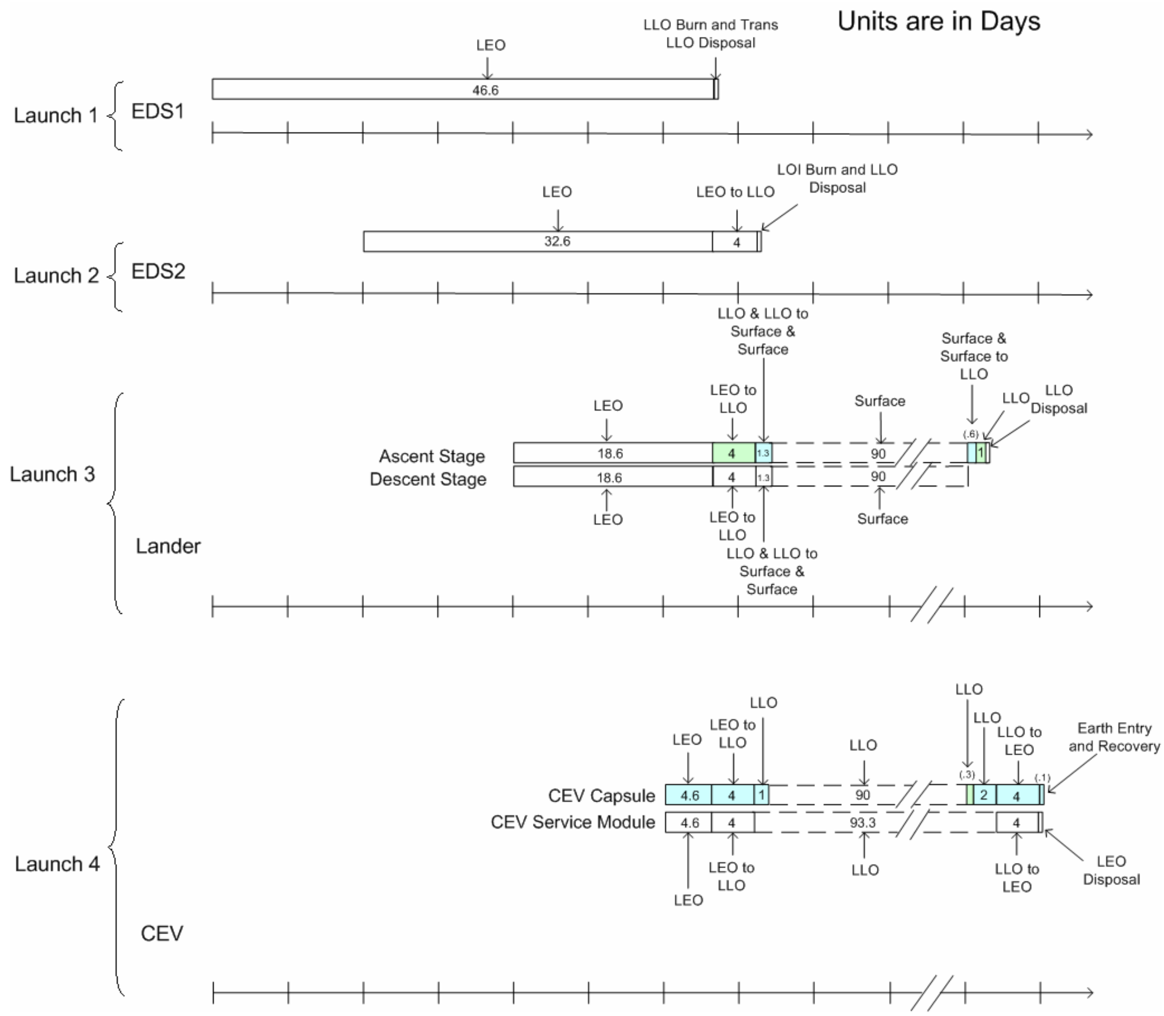

Figure 7.2.2-2: Nominal Timeline for Polar LOR Trade Reference Mission

Vehicle	Phase Name	Phase Length (hr)	Mission Elapsed Time: Overall MET (hr)	Overall MET (days)	EDS1 (hr)	EDS2 (hr)	Lander (hr)	CEV (hr)
EDS1	Launch from Earth/Loiter	2	2	0.1	2			
EDS1	Loiter in LEO	334	336	14.0	336			
EDS2	Launch from Earth/Loiter	2	338	14.1	338	2		
EDS2	Rendezvous & Dock w/ EDS1	50	388	16.2	388	52		
EDS2	Loiter in LEO	284	672	28.0	672	336		
Lander	Launch from Earth/Loiter	2	674	28.1	674	338	2	
Lander	Rendezvous & Dock w/ EDS's	50	724	30.2	724	388	52	
EDS/Lander	Vehicle Checkout	12	736	30.7	736	400	64	
EDS/Lander	Loiter in LEO	272	1008	42.0	1008	672	336	
EDS/Lander	Missed EOD Opportunity	240	1248	52.0	1248	912	576	
CEV	Launch Weather Delay	48	1296	54.0	1296	960	624	48
CEV	Launch from Earth/Loiter	2	1298	54.1	1298	962	626	50
CEV	Rendezvous & Dock w/ Stack	50	1348	56.2	1348	1012	676	100
EDS/Lander/CEV	Vehicle Checkout	12	1360	56.7	1360	1024	688	112
EDS	Earth Orbit Departure	0	1360	56.7	1360	1024	688	112
EDS/Lander/CEV	Coast	48	1408	58.7		1072	736	160
EDS	MCC & EDS Disposal	0	1408	58.7			736	160
EDS/Lander/CEV	Coast	48	1456	60.7			784	208
EDS2	Lunar Orbit Insertion	0	1456	60.7			784	208
Lander/CEV	Crew Transfer & Checkout	24	1480	61.7			808	232
Lander	Undock from CEV	0	1480	61.7			808	232
Lander	Powered Descent	0	1480	61.7			808	232
Lander	Surface Mission	2160	3640	151.7			2968	2392
Lander	Ascent	0	3640	151.7			2968	2392
Lander	Rendezvous & Dock w/ CEV	6	3646	151.9			2974	2398
Lander/CEV	Crew Transfer & Checkout	24	3670	152.9			2998	2422
CEV	Undock from Lander	0	3670	152.9			2998	2422
Lander	Ascent Stage Disposal	0	3670	152.9			2998	2422
CEV	3-Impulse Plane Change	24	3694	153.9				2446
CEV	Lunar Orbit Departure	0	3694	153.9				2446
CEV	Coast	93	3787	157.8				2539
CEV	Dispose Service Module	0	3787	157.8				2539
CEV	Coast & Entry	3	3790	157.9				2542
CEV	Recovery	1	3791	158.0				2543

Table 7.2.2-1: TRM Mission Phase Description

Maneuver Name	Element	ΔV (m/s)	Comments
Earth Orbit Departure	EDS1 & EDS2	3,104	Co-planar departure from LEO assembly orbit (407 km, 28.5^o) w/ 24-hr injection window. Nominal flt time to lunar orbit = 96 hr. Moon @ perigee.
Lunar Orbit Insertion	EDS2	878	Insertion into 100x100 km polar parking orbit (V_∞ = 986 m/s, 0^o relative declination angle).
Descent Plane Change	Descent Stage	567	20^o plane change from polar insertion orbit to access any landing site between 70^o – 90^o latitude.
In-Plane Powered Descent	Descent Stage	1,881	Fuel-optimal powered descent design for in-plane descent from 100x100 km orbit (ref. First Lunar Outpost study)
In-Plane Ascent	Ascent Stage	1,834	Fuel-optimal powered ascent design for in-plane ascent to 100x100 km orbit.
Ascent Plane Change	CEV	567	20^o plane change from polar parking orbit to rendezvous with ascent stage returning from 70^o landing site latitude (required for anytime ascent).
Lunar Orbit Departure	CEV	1,410	Departure from 100x100 km polar orbit (V_∞ = 952 m/s). Includes 3-impulse departure maneuver w/ 24-hr intermediate orbit for 90^o worst-case relative declination angle. Nominal flt time to Earth = 96 hr.

Table 7.2.2-2: Summary of Major Maneuvers for the Polar LOR TRM

7.2.3 Polar TRM Safety and Mission Success

Fifty critical events were identified for the polar LOR trade reference mission. Out of the events identified, nineteen occurred during uncrewed portions of the mission while the remaining thirty-one occurred during the crewed portions of the mission. Each TRM critical event identified was assigned an identification number. As the TRM critical events were identified, they were arranged in sequential order and reviewed with LDRM-2 team members. Once the sequence ordering and terminology of critical events were reviewed and approved by the participating team members, the TRM critical events were assigned a rank describing their importance. The critical event ranking methodology used is described below.

Due to the TRM critical event descriptions being very general, it was decided to keep the critical event ranking criteria at a high-level for consistency purposes. A simplistic approach was used for determining the critical event ranking methodology. The TRM critical events were assigned a ranking of 1, 2, or 3, with 3 representing the most critical of mission events. The ranking definitions are defined below:

Rank of 1: *Failures during mission critical events that could lead to a Loss of Mission (LOM) but not a Loss of Crew (LOC).*

Rank of 2: *Failures during mission critical events that could lead to a LOC but would have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

Rank of 3: *Failures during mission critical events that would not have a mission abort or emergency procedure mitigation option available to prevent a LOC.*

Of the fifty total critical events identified for the Lunar Orbit Rendezvous approach, seven received a rank of 3, nineteen received a rank of 2, and the remaining twenty-four received a rank of 1. The complete set of identified and ranked critical events for the Lunar Orbit Rendezvous approach is listed in the table below.

	ID #	TRM with Lunar Orbit Rendezvous Critical Events	TRM with Lunar Orbit Rendezvous Critical Event Rank
Uncrewed Critical Events	VAR-04-01	EDS-1 Launch	1
	VAR-04-02	EDS-1 Ascent	1
	VAR-04-03	EDS-1 Launch Shroud Separation	1
	VAR-04-04	EDS-1 Separation from Booster	1
	VAR-04-05	EDS-1 Orbital Maneuvering	1
	VAR-04-06	EDS-2 Launch	1
	VAR-04-07	EDS-2 Ascent	1
	VAR-04-08	EDS-2 Launch Shroud Separation	1
	VAR-04-09	EDS-2 Separation from Booster	1
	VAR-04-10	EDS-2 Orbital Maneuvering	1
	VAR-04-11	EDS-1 & EDS-2 Dock	1
	VAR-04-12	EDS-1 & EDS-2 Orbital Maneuvering	1
	VAR-04-13	LL Launch	1
	VAR-04-14	LL Ascent	1
	VAR-04-15	LL Launch Shroud Separation	1
	VAR-04-16	LL Separation from Booster	1
	VAR-04-17	LL Orbital Maneuvering	1
	VAR-04-18	LL Docks to EDS-1 & EDS-2	1
	VAR-04-19	EDS-1, EDS-2, & LL Orbital Maneuvering	1
Crewed Critical Events	VAR-04-20	CEV (CM+SM) Launch	2
	VAR-04-21	CEV Ascent	2
	VAR-04-22	LES Separation	2
	VAR-04-23	CEV Launch Shroud Separation	2

	ID #	TRM with Lunar Orbit Rendezvous Critical Events	TRM with Lunar Orbit Rendezvous Critical Event Rank
	VAR-04-24	CEV Separation from Booster	2
	VAR-04-25	CEV Orbital Maneuvering	2
	VAR-04-26	CEV Docks to EDS-1, EDS-2, & LL	2
	VAR-04-27	EDS-1, EDS-2, LL, & CEV Burn for Lunar Orbit	2
	VAR-04-28	EDS-1 Separates from EDS-2, LL, & CEV	2
	VAR-04-29	EDS-2, LL, & CEV Mid-course Correction Burn	1
	VAR-04-30	EDS-2 Separates from LL & CEV	2
	VAR-04-31	LL & CEV Mid-course Correction Burn	1
	VAR-04-32	LL & CEV Lunar Orbit Insertion (LOI)	2
	VAR-04-33	LL & CEV Orbital Maneuvering	2
	VAR-04-35	Crew Transfers from the CEV to LL	1
	VAR-04-36	LL Separates from CEV	2
	VAR-04-37	LL Powered Descent & Landing to the Moon	3
	VAR-04-38	LL Ascent Stage Separation & Ascent	3
	VAR-04-39	LL Ascent Stage Orbital Maneuvering	3
	VAR-04-40	LL Ascent Stage Lunar Orbit Departure	3
	VAR-04-41	LL Ascent Stage Mid-course Correction Burn	3
	VAR-04-42	LL Ascent Stage Docks with CEV	2
	VAR-04-43	Crew Transfers from the LL to CEV	2
	VAR-04-44	CEV Separates from LL Ascent Stage	2
	VAR-04-45	CEV Burn for Earth	3
	VAR-04-46	CEV Mid-course Correction Burn	1
	VAR-04-47	CM Separates & Maneuvers away from SM	2
	VAR-04-48	CM Entry	3
	VAR-04-49	CM Landing	2
	VAR-04-50	Crew Recovery	2

Table 7.2.3-1: Polar LOR TRM Critical Events and Ranking

7.2.4 Mission Aborts

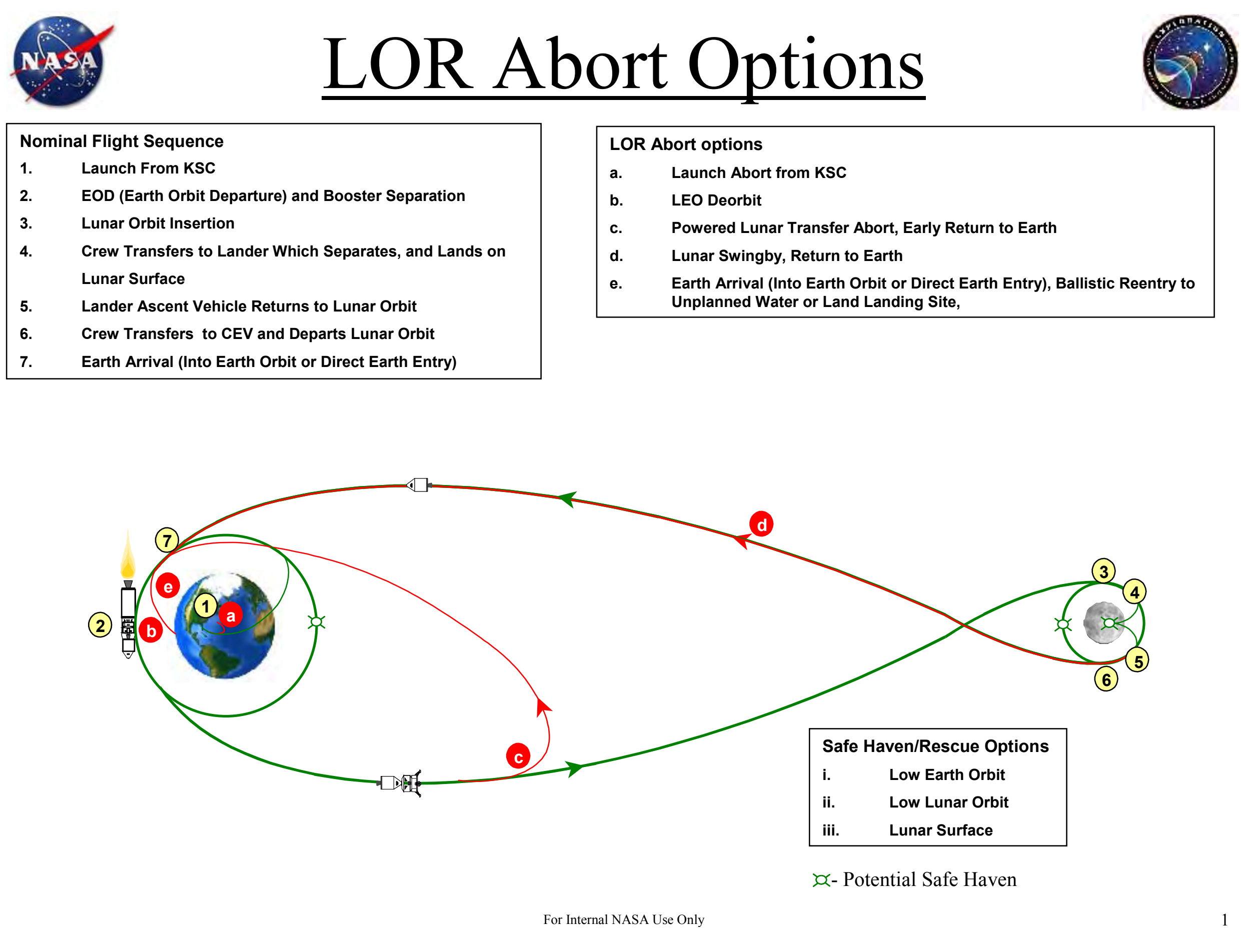

Figure 7.2.4-1: Polar LOR Abort Options

The chart above depicts the crew survival options for the lunar orbit rendezvous mission architecture. The aborts selected for this LOR Trade Reference Mission (TRM) addresses those aborts occurring after CEV launch which result from an inability to complete a critical event required by the LOR mission architecture. Other system failures or problems with the crew may lead to a decision to abort the mission but those aborts can be readily accomplished by moving forward into the next mission phase or bypassing certain mission phases when necessary and completing a safe return to Earth transfer. The following aborts are described for each flight regime of the LOR architecture.

1. Launch and Ascent to Low Earth Orbit (LEO)

This mission phase begins with the launch from Earth surface and ends after the Crew Exploration Vehicle (CEV) is established in the desired LEO.

a. Booster or major CEV system failure

i. CEV emergency separates and returns to Earth

During the CEV launch and ascent to LEO should the Expendable Launch Vehicle (ELV) booster or the CEV suffer catastrophic failure the CEV or ground control can initiate the Launch Abort System, triggering an emergency separation from the ELV and return to Earth using the CEV descent and touchdown systems.

2. LEO Orbit And Rendezvous Operations

This mission phase begins after the CEV is in LEO and ends after the completion of any LEO rendezvous and mating of the Earth Departure Stages, CEV and Lunar Lander.

a. CEV systems failure or failure to mate to Lunar Lander and Earth Departure Stage (EDS)

i. CEV de-orbit and return to Earth

Once the CEV has reached LEO, should the CEV suffer a significant system failure prior to initiating the EDS L1 transfer burn the CEV must perform a standard de-orbit maneuver, re-enter the Earth's atmosphere and successfully touchdown on land or water. If the abort takes place after the CEV mates to the EDS the CEV must separate from the Lunar Lander and EDS prior to re-entry. If CEV propulsion system failures preclude performing a de-orbit maneuver, the Lander or the EDS could be used for that deorbit maneuver. Otherwise the CEV is stranded in LEO and an Earth based CEV rescue mission is required to prevent a loss of crew (LOC) event from occurring. The CEV would need the appropriate resources to provide this safe haven for the crew until that rescue mission is performed (TBD-weeks).

3. LEO to Lunar Transfer

This phase begins at the LEO to LLO departure burn and ends just prior to the lunar orbit insertion burn.

a. Early EDS shutdown and high elliptical orbit

i. CEV maneuver to desired orbit

ii. De-orbit burn and re-entry to touchdown

Should the EDS fail to fully complete the lunar departure burn the CEV/Lander can separate, perform any required transfer orbit adjustments within the limits of available CEV or Lander propulsion constraints, establish a return to Earth trajectory and perform a CEV de-orbit and re-entry to touchdown. After completion of the lunar transfer burn the CEV can also abort by eliminating the Lunar Orbit Insertion (LOI) burn, completing a lunar swingby and returning to Earth on a return transfer orbit. The CEV can adjust this orbit within CEV or Lunar Lander propulsion constraints (if still mated) to ensure a safe Earth re-entry and touchdown.

4. Lunar Orbit Insertion

This phase begins at the start of the LOI burn and ends with the circularization of the lunar orbit.

a. No LOI burn

i. CEV/Lander swingby and return to Earth

If the combined CEV/Lander is not successful in completing the LOI burn then the vehicle must be capable of performing a lunar swingby maneuver and returning to Earth. This can be accomplished by using either the Lander or CEV propulsion systems.

b. Partial LOI burn

i. Ascent stage Delta-V maneuver and return to Earth

If the CEV/Lander partially completes the LLO insertion burn the Lander propulsion stages could be used to complete the insertion burn and then perform the LLO to Earth transfer burn if within the Lander descent or ascent stage propellant budgets. The CEV could also be used to perform the return to Earth maneuver. If CEV failures are known that would preclude the safe execution of a return to earth transfer maneuver, the Lander must be used to adjust the lunar trajectory to perform a lunar swingby and establish the CEV on a safe return to Earth transfer.

5. Lunar Orbit Operations

This phase begins with crew transfer from the CEV to the Lunar Lander and ends continues through CEV/Lander demate and separation from the CEV.

a. Inability to transfer crew from CEV to Lander

i. The CEV separates from the Lander and performs a nominal return to Earth burn. The Lander could also perform the return to earth burn with CEV/Lander separation occurring sometime before CEV reentry.

b. Inability to demate CEV and Lander

i. CEV returns to Earth

The crew returns to the CEV, performs the return to Earth burn, and performs and emergency CEV/Lander separation. Either the CEV or the Lander can perform the return to Earth burn.

c. Inability to perform Lander separation maneuver

i. CEV re-rendezvous and mate with Lander

The CEV, as the active vehicle, will re-rendezvous and mate with the Lander. The crew transfers back to the CEV and performs a nominal return to Earth burn.

6. LLO to Powered Descent Initiation

This phase begins at the start of the lunar descent transfer orbit maneuver and ends just prior to the Powered Descent Initiation burn.

a. No lunar descent transfer orbit burn

i. Lander returns to CEV

During the Lander de-orbit and descent to the lunar surface if any non-propulsion related failure causes an abort, the Lander descent stage will be used to return to LLO and rejoin the CEV. The CEV can then perform the nominal LLO to Earth transfer burn. If the Lander cannot complete the de-orbit to the powered descent initiation point then the Lander can abort using the remainder of the descent stage or the ascent stage to return to LLO and rejoin the CEV.

b. Partial lunar descent transfer orbit burn

i. Lander ascent return to LLO

If a partial de-orbit burn is performed, the Lander ascent stage will return to LLO and rejoin the CEV.

7. Powered Descent Initiation to Lunar Surface

This phase begins at the start of the powered descent initiation burn and ends at lunar surface touchdown.

a. No powered descent

i. Lander ascent stage return to LLO

If the powered descent maneuver is not initiated then the Lander can use either the remainder of the descent stage or the ascent stage to return to LLO and rejoin the CEV.

b. Descent abort

i. Lander ascent stage return to lunar descent transfer orbit

If the need to abort the lunar touchdown occurs late in the powered descent phase, the Lander ascent stage will be used to return to the lunar descent transfer orbit and rejoin the CEV.

8. Lunar Surface Operations

This phase begins just after touchdown, encompasses all lunar surface activities and ends just prior to lunar ascent.

a. EVA suit failures

i. Emergency ingress from EVA

During Lunar surface operations the crew must have the ability to rapidly ingress the Lander from a lunar surface EVA to protect against EVA suit failures. This requires the ability to rapidly transit from any EVA site back to the Lander and re-enter the Lander pressurized volume without extensive stays in any airlock. For long distance EVA sites a pressurized rover with rapid ingress capability may be required to provide a habitable environment in the event of EVA suit failure. In addition, the Lander must be capable of supporting crew medical emergencies resulting from lunar surface operations including the ability to ingress, treat and transport injured crewmembers.

9. Lunar Ascent to LLO

This phase begins at lunar ascent initiation and ends when the Lander has achieved the desired Lunar Orbit.

a. No lunar liftoff

i. Long duration safe haven until Earth based rescue mission arrives (TBD weeks) or LOC

ii. Predeploy extended stay safe haven resources near touchdown site

If the Lander ascent stage fails to ignite then the crew is stranded on the lunar surface and must wait for an Earth based rescue mission. To prevent a LOC event requires the ability for a long duration (TBD weeks) safe haven on the lunar surface, which will require predeployment of safe haven resources near the touchdown site.

b. Failure to reach LLO

i. No functional failure allowed; the Lander ascent stage must reach safe lunar orbit or LOC will occur, physical and functional redundancy is required

After lift off from the lunar surface, the Lander ascent stage must reach a safe LLO or a LOC event will occur. Physical or functional redundancy in the Lander ascent stage is required to ensure that the lunar ascent to LLO is successfully completed.

10. LLO Orbit and Rendezvous Operations

This mission phase begins after the vehicle is in lunar orbit and ends after the completion of any rendezvous and mating of the Lunar Ascent stage with the CEV.

a. Inability to maneuver onorbit (Lander)

i. Passive and Active vehicle exchange roles

If either the Lander ascent stage is unable to maneuver then it becomes the passive vehicle and the CEV becomes the active maneuvering vehicle.

b. Failure to mate Lander and CEV

i. EVA crew transfer to CEV

If the Lander ascent stage and CEV are unable to mate there must be a way to allow the crew to perform an EVA transfer to the CEV for the re-turn to Earth transfer. Otherwise a LOC event will occur.

11. LLO to Earth Transfer

This phase begins with the CEV lunar orbit departure burn and ends just prior to Earth atmospheric re-entry.

a. No LLO departure burn

i. LLO safe haven operations until Earth based rescue

Upon reaching LLO if the CEV is unable to perform the LLO to Earth transfer burn then the crew is stranded in LLO until an Earth based rescue mission arrives or a LOC event occurs. The CEV will require enough re-sources to accommodate the long duration safe haven (TBD weeks) for the crew.

12. Earth Re-entry to Touchdown (direct entry)

This phase begins with the direct re-entry into Earth atmosphere and ends with CEV touchdown on the Earth surface.

a. Re-entry flight control failures

i. Ballistic re-entry (no lift vector control)

The only abort addressed for the Earth re-entry to touchdown phase is the possibility of performing a passive (zero lift) re-entry. This abort will be possible only if the Earth return trajectory allows the re-entry g-levels to remain below the human tolerance limits during the passive re-entry. Otherwise a lift vector controlled trajectory would be required to lower the g loads on the crew and if the CEV lost all control during re-entry a LOC event might occur if the human limits are exceeded.

b. Entry targeting failures

i. Water or land touchdown

CEV equipped with appropriate crew survival and search and rescue equipment for touchdown site

The CEV will be designed to support either land or water touchdown allowing for entry targeting or control system failures that may force the CEV to miss the desired touchdown site. The LOR architecture study is using 3 hr as the time required to find and rescue the crew from the CEV after touchdown on the Earth.

- OR -

13. Earth Aerocapture to LEO

This phase begins with CEV re-entry into Earth atmosphere, encompasses CEV aerobraking into the desired LEO operations and ends just prior to the CEV final de-orbit burn.

a. Failure to aerocapture and circular burn (elliptical orbit)

i. Delta-V maneuver to appropriate orbit with physical or functional redundancy

ii. Safe haven until Earth based rescue or natural orbital decay

iii. Passive control/ballistic re-entry

For missions designed to use aerobraking to LEO instead of a direct entry, a failure to successfully complete the aerocapture leads to the following aborts. If the aerocapture fails to produce the desired LEO, available CEV propulsion can be used to provide the desired orbit. In addition, the CEV may be designed to allow for a passively controlled ballistic re-entry using the aerobrake heat shield in addition to the CEV. Once in LEO the CEV could provide a safe haven for TBD weeks until an Earth based rescue could be performed.

b. Failure to aerocapture (escape trajectory)

i. LOC

If aerocapture failure results in an atmospheric skip out and corresponding Earth escape trajectory, a LOC event will occur. Physical or functional redundancy must be provided to ensure that the CEV is safely captured into LEO.

14. De-orbit to Touchdown

This phase begins with the CEV de-orbit burn and ends with CEV touchdown on the Earth's surface.

a. No de-orbit

i. Safe haven until rescue, orbital decay, or LOC

After reaching a safe LEO if the CEV fails to perform the de-orbit maneuver there is a LOC unless the CEV can provide a safe haven for TBD weeks until an Earth based rescue can be performed.

b. Re-entry flight control failures

i. Passive re-entry (no lift vector control

After a successful de-orbit burn the CEV will have the capability to perform a ballistic re-entry in the event a nominal re-entry is not possible.

c. Entry targeting and control failures

i. Water or land touchdown

CEV with appropriate crew survival and search and rescue equipment for touchdown site.

The CEV will be designed to support either land or water touchdown allowing for entry targeting failures to force the CEV to miss the desired touchdown site. The LOR architecture is using 3 hr as the time required to find and rescue the crew from the CEV after touchdown.

7.3 Element Overview & Mass Properties

This section describes any changes made in sizing the polar LOR TRM mission elements compared to the element configurations in the Phase 1 LOR variant. The total architecture mass for the Phase 2 polar lunar orbit rendezvous architecture variant is estimated at 178 t.

7.3.1 Crew Exploration Vehicle

The primary subsystem change made to the Phase 2 CEV is that it uses solar arrays to generate power instead of fuel cells. Based on results from the Phase 1 alternate power sources parametric variation, a solar array-based CEV in a 90-day surface mission architecture has a total mass 26% lower than a CEV with fuel cells. All other subsystem technologies remain unchanged, though some are scaled up or down based on different delta-V and mission time requirements for Phase 2.

7.3.2 Lunar Lander

Compared to Phase 1, the only change made in subsystem technologies with the Lunar Lander is the removal of the inflatable airlock. In that architecture variant, the role of the Lander was to allow the crew to operate on the lunar surface for up to 7 days, conducting daily EVAs from the vehicle. The Lander's role here is to merely transfer the crew to and from pre-emplaced surface habitation elements, not necessarily to independently operate for long periods on the Moon. After landing, the entire crew will egress the vehicle simultaneously to transfer over to the habitat. Therefore, the value of a dedicated airlock would be very limited, so it was eliminated from vehicle configuration. Like the CEV, the Lander subsystems are scaled based on new delta-V's and mission lifetimes.

7.3.3 Earth Departure Stages

The Earth Departure Stage subsystem technology selections are identical to the Phase 1 LOR variant. The stages have been resized for a lower total delta-V and payload requirement.

7.3.4 Vehicle Mass Properties for Trade Reference Mission

Polar TRM vehicle mass properties as generated by the Envision parametric sizing tool are listed in Table 7.3.4-1. Subsystem components are categorized according the mass properties reporting standard outlined in JSC-23303 Design Mass Properties: Guidelines and Formats for Aerospace Vehicles. All estimates include 20% margin applied to categories one through eight of the vehicle mass properties for dry mass growth.

	CEV CM	CEV SM	Ascent Stage	Descent Stage	Earth Dep. Stage #1	Earth Dep. Stage #2
1.0 Structure	1,523	1,445	788	549	1,366	1,366
2.0 Protection	816	0	73	50	0	0
3.0 Propulsion	117	1,338	1,189	1,373	3,130	3,130
4.0 Power	482	384	737	137	190	190
5.0 Control	0	0	0	0	0	0
6.0 Avionics	737	0	738	0	175	175
7.0 Environment	709	73	786	0	105	105
8.0 Other	833	100	455	514	455	455
9.0 Growth	1,043	668	953	525	1,084	1,084
DRY MASS	6,260 kg	4,008 kg	5,718 kg	3,148 kg	6,506 kg	6,506 kg
10.0 Non-Cargo	884	305	1,202	450	2,377	2,377
11.0 Cargo	1,478	0	227	500	0	0
INERT MASS	8,622 kg	4,313 kg	7,147 kg	4,098 kg	8,883 kg	8,883 kg
12.0 Non-Propellant	329	0	409	0	0	0
13.0 Propellant	64	11,518	5,322	17,046	50,772	50,772
GROSS MASS	9,015 kg	15,831 kg	12,878 kg	21,144 kg	59,655 kg	59,655 kg

Table 7.3.4-1: Polar LOR TRM Vehicle Mass Properties

The two largest single elements to be launched are the equal-mass Earth Departure Stages for the Lunar Lander and CEV with an initial mass in low Earth orbit (IMLEO) of 59.7 t. These two stages are the first elements launched in the architecture, and they execute the Earth orbit departure and lunar orbit insertion burns for the Lander and CEV. The launch of these elements will drive the payload delivery capabilities of the cargo launch vehicle. The Lunar Lander is launched next with a launch mass of 34.0 t. Finally, the CEV is launched with the crew on a human-rated launch vehicle capable of delivering 24.8 t to LEO. The combined architecture elements of the polar LOR trade reference mission have a total IMLEO of 178 t.

Polar TRM Mass Properties (kg)	**CEV CM**	**CEV SM**	**Ascent Stage**	**Descent Stage**	**EDS1**	**EDS2**
1.0 Structure	**1,523**	**1,445**	**788**	**549**	**1,366**	**1,366**
Primary Structure	1522.1	0.0	737.1	400.0	0.0	0.0
Stowage Equipment	0.0	0.0	0.0	0.0	0.0	0.0
Chemical Propulsion Stage Structure	1.0	1445.3	50.7	148.8	1365.6	1365.6
2.0 Protection	**816**	**0**	**73**	**50**	**0**	**0**
Thermal Protection System	732.8	0.0	0.0	0.0	0.0	0.0
Insulation	82.7	0.0	72.6	50.0	0.0	0.0
3.0 Propulsion	**117**	**1,338**	**1,189**	**1,373**	**3,130**	**3,130**
OMS Engines & Installation	0.0	208.6	474.4	0.0	688.7	688.7
RCS Engines & Installation	63.0	163.0	81.5	108.7	153.1	153.1
OMS Fuel Tanks & Feed/Fill/Drain System	0.5	422.0	271.5	561.6	1237.5	1237.5
OMS Oxidizer Tanks & Feed/Fill/Drain System	0.5	379.8	253.9	505.4	661.8	661.8
RCS Fuel Tanks & Feed/Fill/Drain System	53.0	11.8	6.7	8.4	117.4	117.4
RCS Oxidizer Tanks & Feed/Fill/Drain System	0.0	13.2	7.5	9.4	145.8	145.8
Pressurization System	0.0	139.2	93.2	179.8	126.0	126.0
4.0 Power	**482**	**384**	**737**	**137**	**190**	**190**
Fuel Cell	0.0	0.0	210.1	0.0	0.0	0.0
Regenerative Fuel Cells	0.0	0.0	0.0	0.0	0.0	0.0
Hydrogen PRSD Tanks	0.0	0.0	131.1	0.0	0.0	0.0
Oxygen PRSD Tanks	0.0	0.0	91.9	0.0	0.0	0.0
Photovoltaic Arrays	0.0	264.4	0.0	47.7	84.0	84.0
Battery Type #1	171.1	0.0	0.0	12.2	6.1	6.1
Battery Type #2	0.0	0.0	0.0	0.0	0.0	0.0
Power Management & Distribution	311.3	120.0	304.3	77.5	100.0	100.0
Nuclear Reactor	0.0	0.0	0.0	0.0	0.0	0.0
5.0 Control	**0**	**0**	**0**	**0**	**0**	**0**
Flight Control Surface Actuation	0.0	0.0	0.0	0.0	0.0	0.0
6.0 Avionics	**737**	**0**	**738**	**0**	**175**	**175**
Command, Control, and Data Handling	161.5	0.0	161.5	0.0	38.9	38.9
Guidance & Navigation	145.1	0.0	145.1	0.0	40.7	40.7
Communications	79.5	0.0	117.7	0.0	36.0	36.0
Vehicle Health Management	0.0	0.0	0.0	0.0	0.0	0.0
Cabling and Instrumentation	351.2	0.0	313.3	0.0	59.8	59.8
7.0 Environment	**709**	**73**	**786**	**0**	**105**	**105**
Environmental Control & Life Support System						
Nitrogen Storage	30.7	0.0	50.0	0.0	0.0	0.0
Oxygen Storage	18.4	0.0	10.3	0.0	0.0	0.0
Atmosphere Supply Reg, Dist, and Control	57.0	0.0	60.1	0.0	0.0	0.0
Atmosphere Contaminant Control	122.3	0.0	94.2	0.0	0.0	0.0
Fire Detection and Suppression	20.3	0.0	20.3	0.0	0.0	0.0
Venting and Thermal Conditioning	47.5	0.0	54.9	0.0	0.0	0.0
Water Management	96.7	0.0	28.8	0.0	0.0	0.0
Airlock/EVA Support	0.0	0.0	0.0	0.0	0.0	0.0

Airlock	0.0	0.0	0.0	0.0	0.0	0.0
Umbilicals and Support	8.9	0.0	71.4	0.0	0.0	0.0
Thermal Control System						
ETCS	56.2	0.0	52.1	0.0	21.6	21.6
ITCS	108.6	0.0	124.7	0.0	59.1	59.1
Radiator	0.0	73.1	95.6	0.0	24.4	24.4
Fluid Evaporator System	20.2	0.0	20.0	0.0	0.0	0.0
Phase Change Heat Rejection	0.0	0.0	0.0	0.0	0.0	0.0
Heat Pump	0.0	0.0	0.0	0.0	0.0	0.0
Crew Accommodations						
Galley	39.1	0.0	36.2	0.0	0.0	0.0
Waste Collection System	29.6	0.0	26.2	0.0	0.0	0.0
Seats & Tables	53.3	0.0	41.0	0.0	0.0	0.0
8.0 Other	**833**	**100**	**455**	**514**	**455**	**455**
Parafoil Assembly	0.0	0.0	0.0	0.0	0.0	0.0
Main Parachutes	194.9	0.0	0.0	0.0	0.0	0.0
Drogue Parachutes	47.9	0.0	0.0	0.0	0.0	0.0
Landing, Flotation, & Misc Chutes	90.8	0.0	0.0	414.0	0.0	0.0
Shell Heaters	0.0	0.0	0.0	0.0	0.0	0.0
Doors, Hatches, Pyros, and Docking Adapters	499.0	100.0	455.2	100.0	455.2	455.2
9.0 Growth	**1,043**	**668**	**953**	**525**	**1,084**	**1,084**
10.0 Non-Cargo	**884**	**305**	**1,202**	**450**	**2,377**	**2,377**
Personnel Provisions						
Recreational Equipment	20.0	0.0	20.0	0.0	0.0	0.0
Crew Health Care	54.9	0.0	54.9	0.0	0.0	0.0
Personal Hygiene	11.9	0.0	8.5	0.0	0.0	0.0
Clothing	28.9	0.0	7.8	0.0	0.0	0.0
Housekeeping Supplies	25.6	0.0	16.4	0.0	0.0	0.0
Operational Supplies	72.7	0.0	52.0	0.0	0.0	0.0
Maintenance Equipment	25.0	0.0	25.0	0.0	0.0	0.0
Photography Supplies	45.0	0.0	45.0	0.0	0.0	0.0
Sleep Accommodations	36.0	0.0	9.2	0.0	0.0	0.0
EVA Suits and Spares	0.0	0.0	381.8	0.0	0.0	0.0
EVA Tools	17.7	0.0	0.0	0.0	0.0	0.0
Food	144.7	0.0	39.1	0.0	0.0	0.0
Crew	400.0	0.0	400.0	0.0	0.0	0.0
Reserve, Residual Fluids, and Gases						
Pressurant	0.0	74.5	35.8	108.6	169.2	169.2
Unused Fuel	0.3	48.0	22.2	71.0	991.4	991.4
Unused Oxidizer	1.0	182.4	84.3	269.9	1216.3	1216.3
11.0 Cargo	**1,478**	**0**	**227**	**500**	**0**	**0**
Ballast & Other Misc. Mass	100.0	0.0	0.0	0.0	0.0	0.0
Radiation Protection	1378.0	0.0	227.0	0.0	0.0	0.0
Payload	0.0	0.0	0.0	500.0	0.0	0.0
12.0 Non-Propellant	**329**	**0**	**409**	**0**	**0**	**0**
Fuel Cell Oxygen	0.0	0.0	287.7	0.0	0.0	0.0

Fuel Cell Hydrogen	0.0	0.0	31.6	0.0	0.0	0.0
Oxygen (Life Support)	66.3	0.0	0.0	0.0	0.0	0.0
Nitrogen (Life Support)	29.8	0.0	48.5	0.0	0.0	0.0
Fluid Evaporator System Water	0.0	0.0	0.0	0.0	0.0	0.0
Potable Water	232.9	0.0	41.6	0.0	0.0	0.0
13.0 Propellant	**64**	**11,518**	**5,322**	**17,046**	**50,772**	**50,772**
Usable OMS Fuel (None)	0.0	2325.8	1103.4	3528.4	7148.8	7148.8
Usable OMS Oxidizer (None)	0.0	8838.1	4193.1	13407.8	42892.6	42892.6
Usable RCS Fuel (None)	13.3	73.8	5.2	22.9	104.4	104.4
Usable RCS Oxidizer (None)	50.7	280.3	19.8	86.9	626.4	626.4
Dry Mass	**6,260 kg**	**4,008 kg**	**5,718 kg**	**3,148 kg**	**6,506 kg**	**6,506 kg**
Inert Mass	**8,622 kg**	**4,313 kg**	**7,147 kg**	**4,098 kg**	**8,883 kg**	**8,883 kg**
Total Vehicle	**9,015 kg**	**15,831 kg**	**12,878 kg**	**21,144 kg**	**59,655 kg**	**59,655 kg**

Table 7.3.4-2: Polar LOR TRM Detailed Vehicle Mass Properties

7.4 Alternate Launch Solutions

As in LDRM-2 Phase 1, several TRM architecture options have been examined in which the number of launches per mission is varied. The polar LOR baseline assumes a 4-launch solution consisting of separate launches for the EDS1, EDS2, Lunar Lander, and CEV. This section describes analyses of architectures with a 2-launch solution, a 3-launch solution, and a 25 t launch limit.

The first variant, the 2-launch solution, retains Lander/CEV assembly in low Earth orbit as in the polar TRM. However, in the 2-launch solution the Lander and EDS1 launch as one single combined element and the CEV and EDS2 launch two weeks later as a second combined element. The Lander and CEV then mate in LEO, giving rise to a different stack configuration than with the 4-launch polar TRM. As Figure 7.4-1 illustrates, the configuration for the 4-launch mission arranges the vehicles in sequential order EDS1-EDS2-Lander-CEV. The 2-launch solution instead has a configuration of EDS1-Lander-CEV-EDS2. While this is beneficial in that an on-orbit mating interface between EDS1 and EDS2 is eliminated, and the CEV/EDS2 and Lander/EDS1 interfaces can be made on the ground instead of in space, it introduces a severe complication during the Earth orbit departure maneuver. After EDS1's propellant supply is exhausted, the entire stack must rotate 180^{o} for EDS2 to be in proper position to complete the burn. The time lag that is required to rotate the stack and assure that EDS2 is properly aligned will introduce delta-V performance penalties. One alternative 2-launch scenario to avoid this might be to resize EDS1 such that it is capable of executing the entire Earth orbit departure burn and launch that element individually. The second launch would then consist of the EDS2, CEV, and Lunar Lander, or in another option the CEV Service Module could take on the performance requirements of EDS2 and that element would be eliminated.

As will be discussed in a later section, the 2-launch solution lends itself better to an architecture where the Lander and CEV depart low Earth orbit separately. This strategy, called a 'convoy departure', can be used to either pre-deploy the Lander to lunar orbit or to have the CEV and Lander depart LEO nearly simultaneously, traveling to the Moon like a convoy.

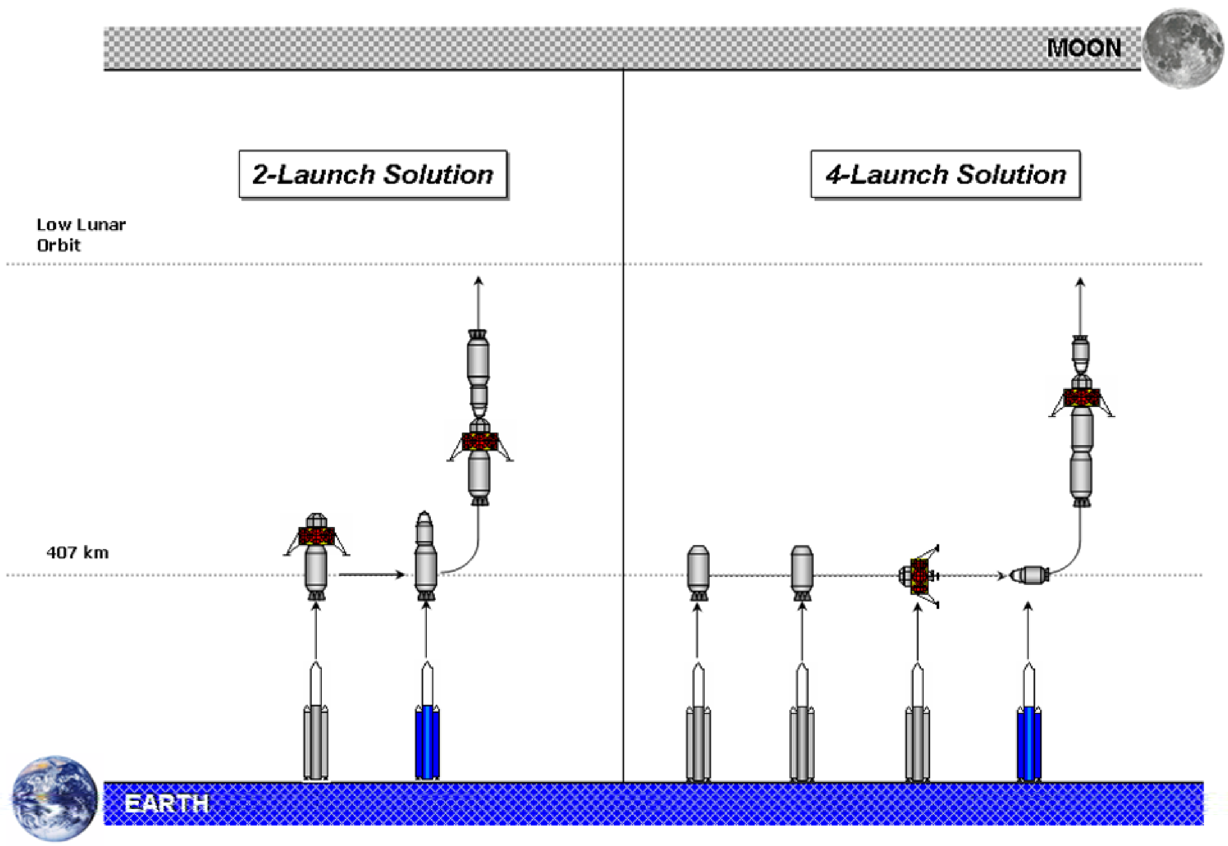

Figure 7.4-1: Earth Orbit Departure Stack Configuration for 2-Launch Solution

Two potential launch options exist for the 3-launch solution. The first option launches EDS2 and the Lander first as a combined element, followed by a separate launch for EDS1. EDS1 then docks with EDS2, and two weeks later, the crew launches in the CEV on the same human-rated launch vehicle as in the 4-launch baseline. A second option launches EDS1 and EDS2 separately and combines the launch of the Lander and CEV on a third launch vehicle. These two options have their own merits relative to the TRM and to each other. Option 1 is advantageous because besides only requiring three launches per mission, it eliminates an interface between the Lander and EDS2. However, as will be seen, it results in an uneven split in payload between the different launches by combining the 60 t EDS and 34 t Lander into one launch and thus requires a high payload mass delivery capability. Option 2 provides a better split in mass, but it does not eliminate an on-orbit interface between elements and in order to use the Lander's volume during the outbound transit to the Moon, the CEV must undock from the Lander once on orbit and then re-dock using the pressurized docking interface. In addition, if all other factors are considered equal, larger human-rated launchers such as required with this option are less desirable than a smaller launcher as in the 4-launch solution. Figure 7.4-2 shows the various launch configuration options for the 3-launch solution.

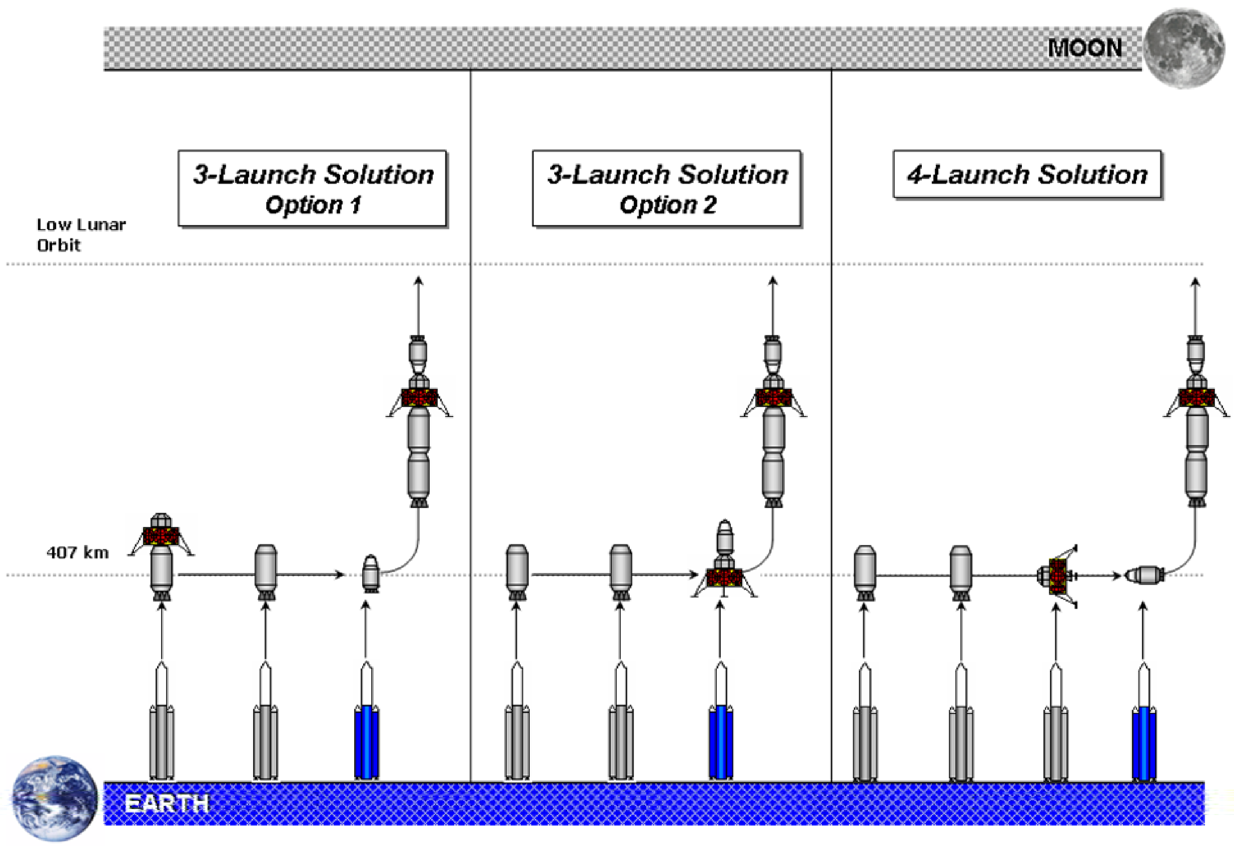

Figure 7.4-2: Launch Configurations for 3-Launch Solution

The final launch option investigated in LDRM-2 Phase 2 was a 25 t limit per launch architecture. Retaining Lander/CEV assembly in LEO here means that the functionality previously provided by two 60 t Earth Departure Stages must now be provided by five 25 t stages. Those five stages launch first in the architecture and assemble themselves into a configuration (Figure 7.4-3) such that three stages simultaneously burn to perform the first part of the Earth orbit departure maneuver and then separate from the stack. The remaining two Earth Departure Stages complete the maneuver. The 25 t launch limit also forces the Lander to be launched as a separate Ascent Stage and Descent Stage and then be assembled on orbit, as was the case with the Phase 1 25 t architecture variation. Finally, the CEV and crew launch in the 8th launch of the architecture. The same issues raised in Phase 1 with this variant are still applicable here, most notably the large number of dockings and complex docking interfaces required, and the challenges of successfully launching eight critical architecture elements in a short period just to perform one lunar mission. More work is necessary to formulate a credible scenario for automatically assembling such a complex stack configuration.

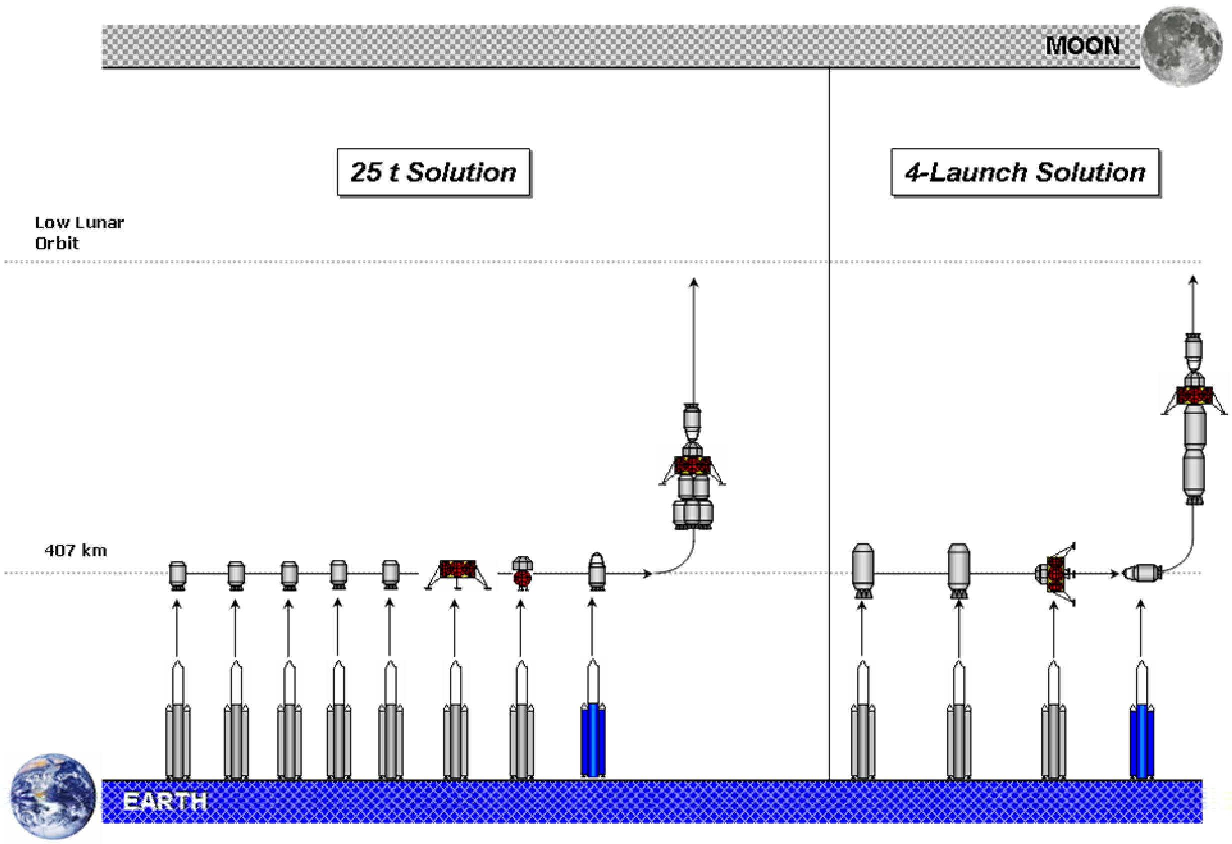

Figure 7.4-3: Earth Orbit Departure Configuration for 25 t Launch Solution

Mass per launch requirements for the alternate launch solution architectures are depicted in Figure 7.4-3. The 4-launch polar trade reference mission requires two 60 t launches for EDS1 and EDS2, a 34 t launch for the Lunar Lander, and a 25 t human-rated launcher for the CEV.

The selected 2-launch solution architecture requires 91 t launched for the Lander and EDS1, and a 81 t human-rated launch vehicle for the CEV and EDS2.

Two options were analyzed for the 3-launch solution variant. The first option requires one 94 t launch for EDS1 and the Lander, a 60 t launch for EDS2, and a 25 t human-rated launch vehicle for the CEV. The second option separates EDS1 and EDS2 onto two 60 t cargo launchers and combines the Lander and CEV onto a 59 t human-rated launch vehicle.

Finally, the 25 t launch limit architecture uses eight 25 t launchers to perform the mission. The separate elements here are EDS1-5, the Descent Stage, the Ascent Stage, and the CEV.

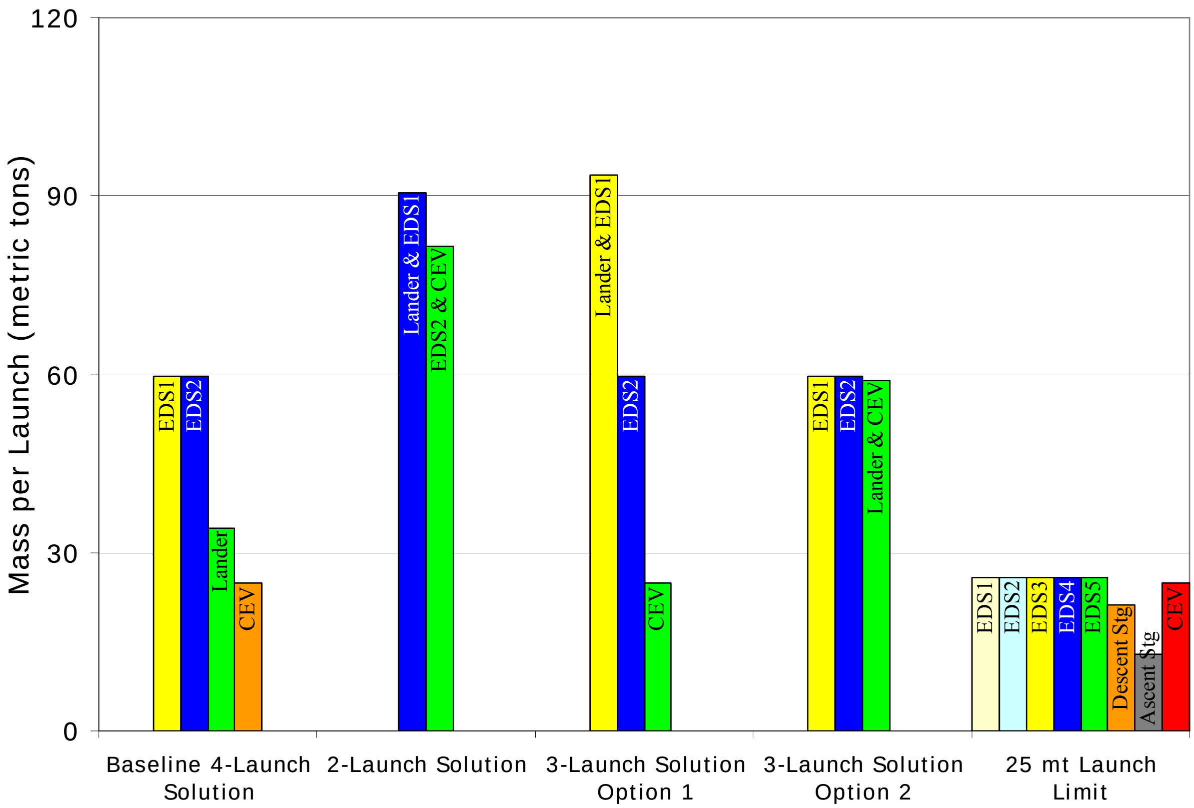

Figure 7.4-4: Launch Requirements for Polar TRM Architecture Variants

7.5 Anytime Return vs. Loitering for Planar Alignment

One of the key study groundrules for LDRM-2 Phase 1 and Phase 2 architectures has been the requirement to return the crew to Earth at any time from the lunar surface, independent of orbital plane alignment. For the long-stay missions at 70^o latitude landing sites, this means in a worst-case scenario that a 20^o plane change may be required on ascent and a 90^o plane change for lunar orbit departure. This section determines the cost of the anytime return requirement for the polar TRM in terms of vehicle and total architecture mass.

The 20^o ascent plane change can be avoided by waiting in the surface habitat until the landing site passes under the plane of the CEV's lunar parking orbit. For an inertially-fixed polar orbit around the Moon, these opportunities arise at non-polar sites once every 13.6 days. Therefore, if an in-plane (minimum delta-V) ascent opportunity is just missed, the crew must wait on the surface for 13.6 days until the next such opportunity is available. If the CEV, on the other hand, does not carry the 20^o plane change delta-V (567 m/s) but only half that (10^o), the worst-case wait-time can be reduced to 6.8 days. Figure 7.5-1 illustrates how CEV mass changes with the amount of ascent plane change capability. Recall that for the polar LOR TRM the CEV performs the ascent plane change, not the Lander. With the full plane change capability on board, the CEV has a total mass of 24.8 t and the crew can ascend from the lunar surface to rendezvous with the CEV at any point in the mission. At the other extreme, if the CEV does not carry any plane change capability for anytime ascent, the CEV total mass drops by 4.4 t, but in that case, the crew may be required to wait for 13.6 days on the surface. The mass of personnel and provisions is constant here because the CEV is unoccupied during the loiter period. For all data shown in the figure, the CEV carries a full 90° of plane change for anytime Earth return.

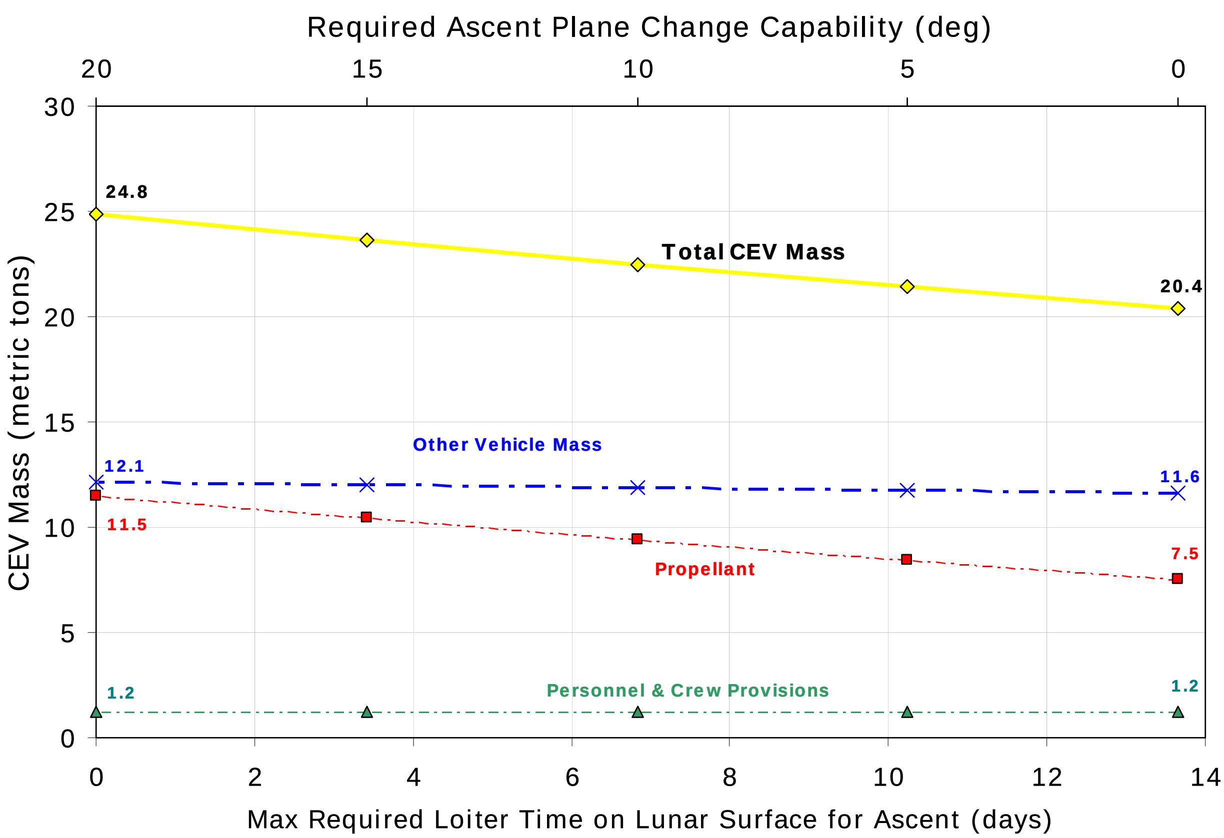

Figure 7.5-1: CEV Mass Cost of Anytime Ascent

In addition to a 20° plane change capability for ascent, the CEV carries enough delta-V (1,410 m/s) to depart lunar orbit at any time. The CEV and total architecture mass can be reduced by limiting the amount of plane change delta-V included for anytime Earth return and instead waiting for more favorable planar alignments. With an inertially-fixed polar lunar parking orbit, minimum energy lunar departure opportunities arise once every 13.6 days. Note that the Earth return opportunity frequency is identical to the in-plane ascent opportunity frequency because the Moon is tidally locked – the Moon performs one complete rotation about its axis in the same time it takes to perform one complete revolution about Earth. Thus, if the CEV is only capable of a minimum delta-V (865 m/s) return to Earth, and the first return opportunity is just missed, the crew must loiter in the CEV up to 13.6 days for the next opportunity. In that case, the total CEV mass is 21.5 t, a 3.3 t decrease from the TRM CEV. Even though the delta-V cost is comparable – 567 m/s for anytime ascent vs. 545 m/s for anytime Earth return – loitering for in-plane ascent provides more CEV mass savings (4.4 t vs. 3.3 t) than loitering for in-plane return. This is because in the latter case, the crew may be waiting in the CEV for up to 13.6 days, and the propellant mass savings are partly offset by higher crew provision costs. Figure 7.5-2 shows how CEV mass varies with plane change capability or length of loiter time in lunar orbit that will be accepted. For all data shown in the figure, the CEV carries a full 20^{o} of plane change for anytime ascent.

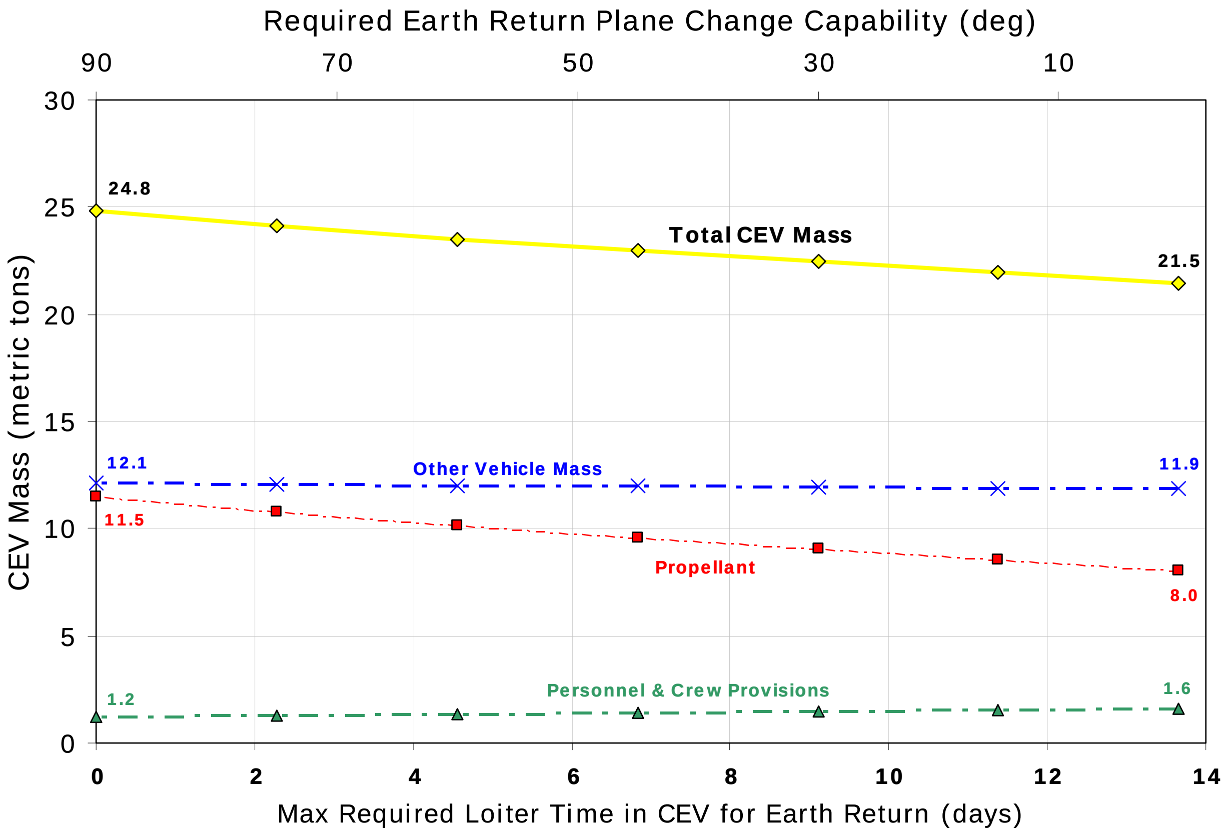

Figure 7.5-2: CEV Mass Cost of Anytime Earth Return

Figure 7.5-3 illustrates the cost of anytime ascent and anytime Earth return on the overall architecture. For the polar TRM, which includes the full plane change delta-V for anytime return and is capable of landing at latitudes as low as 70^{o}, the total architecture mass is 178 t. For the same landing access but instead loitering in the habitat for in-plane ascent and in the CEV for in-plane Earth return, the total mass decreases by 20 t to 158 t. The CEV mass in such a scenario is reduced by 7 t as expected based on the results shown above. Similar data is also provided for architectures that are restricted to landing sites between 80^{o} – 90^{o} and sites exactly on the poles.

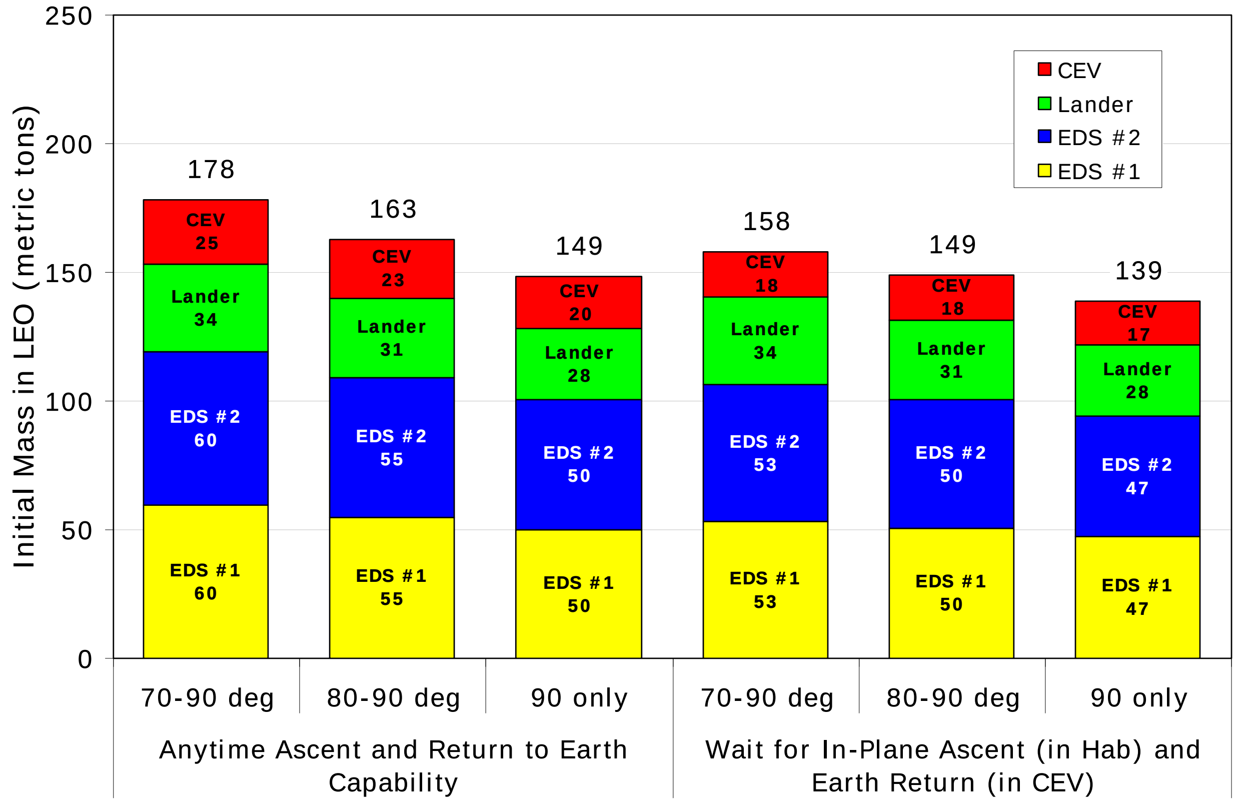

Figure 7.5-3: Architecture Mass Cost of Anytime Ascent and Earth Return

7.6 Alternate Earth Orbit Departure Strategy

The polar LOR TRM uses an Earth orbit departure where the CEV and Lander mate in low Earth orbit and travel to the Moon as a single combined element. This method, henceforth referred to as a ‘tandem’ departure, was selected primarily because it may allow the crew to use the capabilities of the Lander for an abort during the outbound transfer and it approximately doubles the amount of habitable volume available to the crew during that phase of the mission. However, the tandem departure strategy also means that a complex assembly sequence is required in LEO with a four launch per mission architecture. The second Earth Departure Stage requires a mating interface to both EDS1 and the Lunar Lander, and the Lunar Lander has a mating interface to EDS2 and the CEV. Tandem departure also requires a staging event during the Earth orbit departure maneuver.

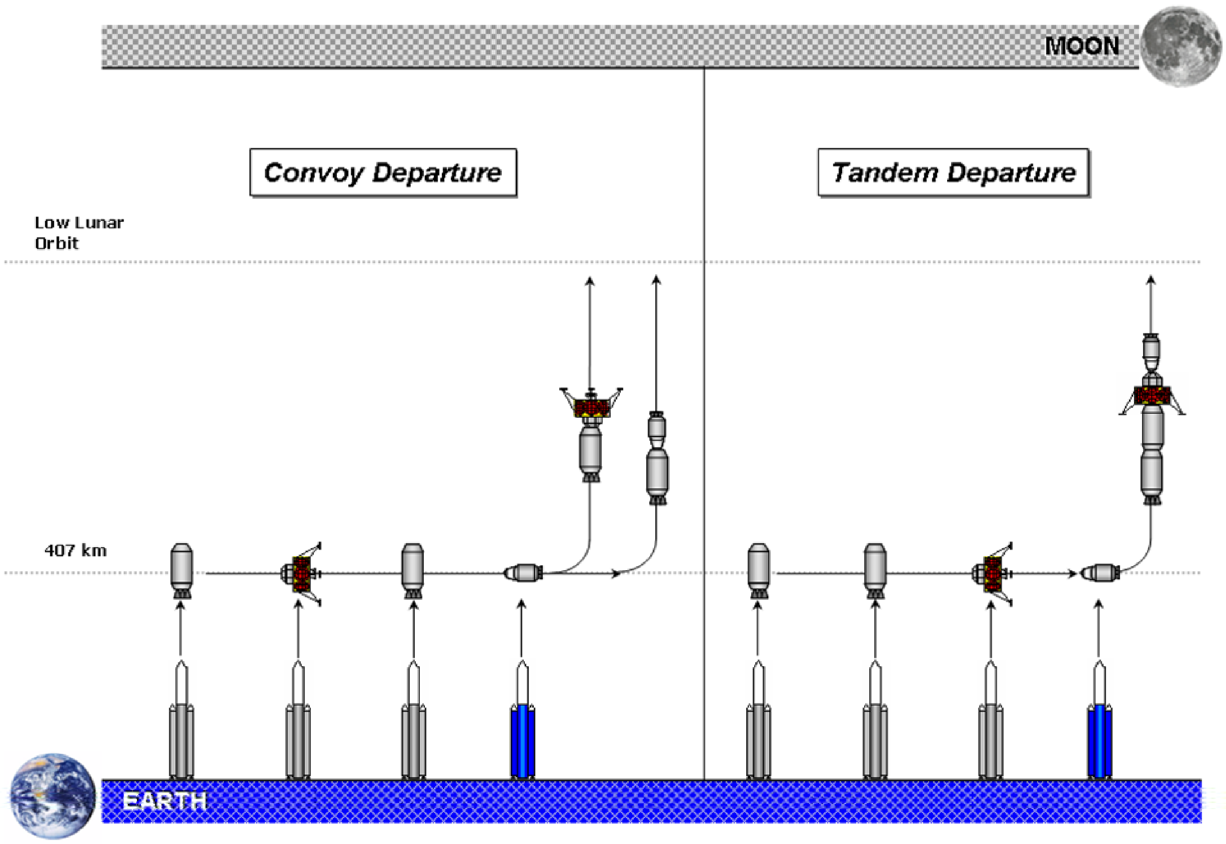

Figure 7.6-1: Convoy Earth Departure Strategy

An alternate Earth departure strategy, discussed in Section 7.4 for the 2-launch solution, is to provide one EDS for the Lander, one for the CEV, and have those elements depart LEO and travel to the Moon separately. Two potential options exist for delivering the Lander to low lunar orbit. In the first option, the Lander can depart LEO and arrive at the Moon before the CEV is launched (pre-deployment). Or, the Lander and CEV can depart LEO nearly simultaneously and travel to the Moon together. This strategy is called a 'convoy' departure and is seen in Figure 7.6-1. For this analysis, the Lander EDS nominally performs Earth orbit departure 90 minutes (one orbit revolution) before the CEV, but the CEV EDS provides enough delta-V for the burn to allow the CEV to depart LEO up to 24 hours later yet still arrive in low lunar orbit at the same time as the Lander (Figure 7.6-2). Arriving in lunar vicinity, the CEV is inserted into a 500 km temporary phasing orbit for rendezvous and the Lander is inserted into the nominal 100 km circular orbit. Both vehicles arrive in the same orbital plane. No later than twelve hours after insertion, the CEV docks with the Lander and the rest of the mission is identical to the polar TRM. The rendezvous and docking cost in low lunar orbit to cover a 360^{o} phase angle is ~160 m/s.

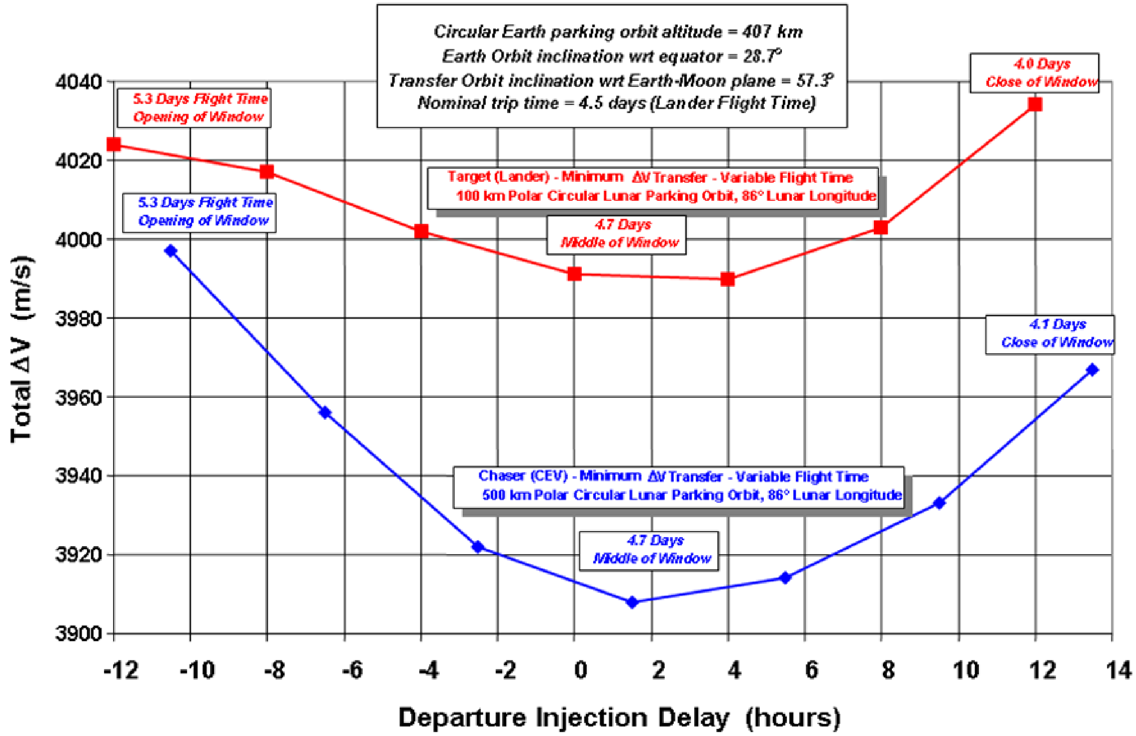

Figure 7.6-2: Earth Departure Stage Delta-V for Convoy Earth Departure

The primary benefit of the convoy departure strategy is that it greatly simplifies element interfaces. The Lander and CEV can use their pressurized crew transfer interfaces to mate with their respective Earth Departure Stages for the outbound transfer, and there is no EDS-to-EDS interface required. The convoy departure also eliminates staging during the Earth orbit departure maneuver. However, it does add two additional events – a second Earth orbit departure and lunar orbit insertion – and transfer one rendezvous and docking sequence from LEO to low lunar orbit. Most importantly, though, in the convoy strategy the Lander is no longer attached to the CEV on the way to the Moon. Therefore, its volume is not available to the crew and its capabilities cannot be used for aborts.

While the nominal convoy architecture is still a 4-launch solution, 2- and 3- launch architectures are provided here for information. With the 4-launch option, the Lander EDS requires a 73 t cargo launch vehicle in order to deliver the Lander to lunar orbit. The CEV EDS also requires a 73 t launch vehicle. Since the CEV mass is less than the Lander's mass, the CEV EDS actually requires less propellant than the Lander EDS to deliver its payload to the Moon, but it is assumed to have the same size as that stage for launch. The unused propellant can be used for outbound aborts. The Lunar Lander is the same as in the TRM (34 t), and the CEV requires a 26 t human-rated launch vehicle.

For the 2-launch solution, the Lander and its EDS are combined into one launch (103 t launcher), and the CEV and its EDS are combined on a second launch. This requires an 80 t human-rated launcher. Unlike the 4-launch convoy baseline, the 2-launch solution does not assume that the

CEV EDS carries any unused propellant in order to minimize the size of that launch vehicle. If the same launcher were used for both launches, the human-rated vehicle could potentially provide engine-out capability for the crew if the payload it was carrying was less than the cargo variant.

The 3-launch convoy variant also combines the Lander and its EDS into one 103 t launch, however, it minimizes the size of the human-rated launch vehicle by splitting the CEV and its EDS into separate launches. The CEV EDS must carry additional provisions for docking with the CEV in LEO. Figure 7.6-3 shows the launch requirements for the 2-,3-, and 4-launch convoy architectures.

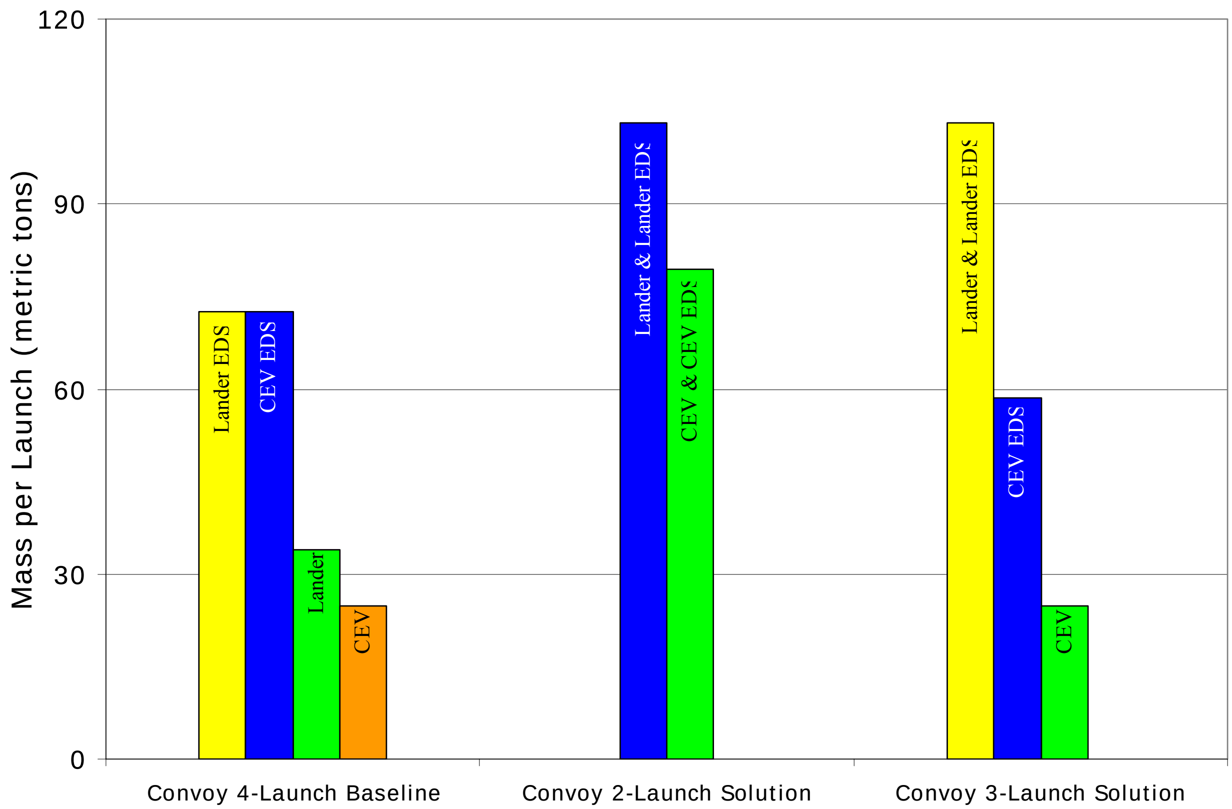

Figure 7.6-3: Launch Variants for Convoy Earth Departure

As the elements are currently sized, the convoy departure architecture provides another benefit over the tandem baseline. The polar TRM (tandem) used equal-mass Earth Departure Stages to minimize the size of the cargo launch vehicle. That strategy results in a cargo launcher required to deliver only 60 t to orbit as compared to 73 t for the convoy departure. However, it also means that a single Earth Departure Stage and Lander Descent Stage for the tandem architecture are capable of landing less cargo on the lunar surface in a scenario where the Descent Stage is used as a cargo delivery vehicle rather than a human transportation system. The Descent Stage for cargo delivery would be able to land the full mass of the Ascent Stage on the Moon, but the single 60 t EDS would not be able to deliver it to lunar orbit. A larger 73 t EDS would be required for that, which is the size of the EDS for the convoy architecture. As the lunar exploration program will likely require a robust cargo delivery for habitats, rovers, ISRU/power plants, logistics resupply, or other TBD assets, its needs should be carefully considered when selecting a crew transportation architecture. The benefits that an equal-mass optimized tandem departure strategy may not be sufficient for cargo transportation. If the tandem architecture is preferred, unequal-mass stages may be necessary which would drive up the required size of the cargo launch vehicle.

8.0 Alternate Mission Capture #1 – Global Lunar Access

The polar LOR trade reference mission described in the previous section was formulated to enable long-duration surface expeditions to the polar regions of the Moon. However, there is also strong interest in using Project Constellation elements to explore other high-value scientific targets with focused, short-stay missions to other locations on the lunar surface. Such missions are henceforth identified as alternate mission #1 – global lunar access. This section investigates the ability of the TRM elements to capture these other potential missions without affecting their design for the trade mission. Particular emphasis is placed on determining whether any additional CEV capabilities that may be required to support these missions. Areas in which other TRM elements would have to be modified to capture global lunar access are also identified.

8.1 Major Assumptions/Clarifications

The "global lunar access" alternate mission capture affects the following polar TRM assumptions from the LDRM-2 Phase 2 task request statement. Assumptions from Section 7.0 not explicitly listed here are still applicable to the architecture.

- Lunar surface stay time up to 90 days: The global mission access variant will determine how long the TRM elements are capable of operating on the Moon.

  Lunar orbit is used as the lunar vicinity rendezvous point to enable near-polar landing site access between 70 and 90 degrees latitude: This alternate mission will also incorporate lunar orbit rendezvous with a low lunar orbit though it is not restricted in landing site latitude.

8.2 Architecture Description

This section describes the global lunar access alternate mission architecture, its safety and mission success aspects, and potential mission abort options.

8.2.1 Global Lunar Access Architecture

The global lunar access architecture operates identically to the lunar orbit rendezvous variant described in the Phase 1 section of this report. Like that architecture, this variant inserts the elements into a lunar parking orbit tailored to the specific latitude and longitude of the mission's landing site. This is done to minimize the cost of anytime ascent during a short-stay mission to some of the more challenging locations to access on the Moon (such as mid-latitude sites). At lunar orbit insertion, a series of three impulsive maneuvers are performed over a 24-hr period to select a particular inclination and ascending node for the parking orbit. In contrast, the polar TRM inserted the CEV and Lander into a polar parking orbit in anticipation of a long-stay mission at the Moon's polar regions. Like the polar TRM, though, the short-stay LOR may also require a 3-impulse, 24-hr sequence to properly align the parking orbit for Earth return.

The only difference between the Phase 1 and Phase 2 short-stay LOR architectures is that the Phase 1 elements were sized for 7 days on the lunar surface while here the mission is restricted to 4 days. As the Lunar Lander was designed for the polar TRM to have 4 days of independent operating capability on the Moon, that will limit how long the Lander without modification can operate when used in this alternate mission mode. As will be shown later, though, the "Global Access" Lander requires less propellant than the "Polar TRM" Lander, so it may be possible to trade propellant for additional crew consumables to enable longer surface missions.

See Section 14.0 of the Phase 1 study for additional details on the short-stay LOR architecture variant. Figure 8.2.1-1 illustrates the global lunar access architecture.

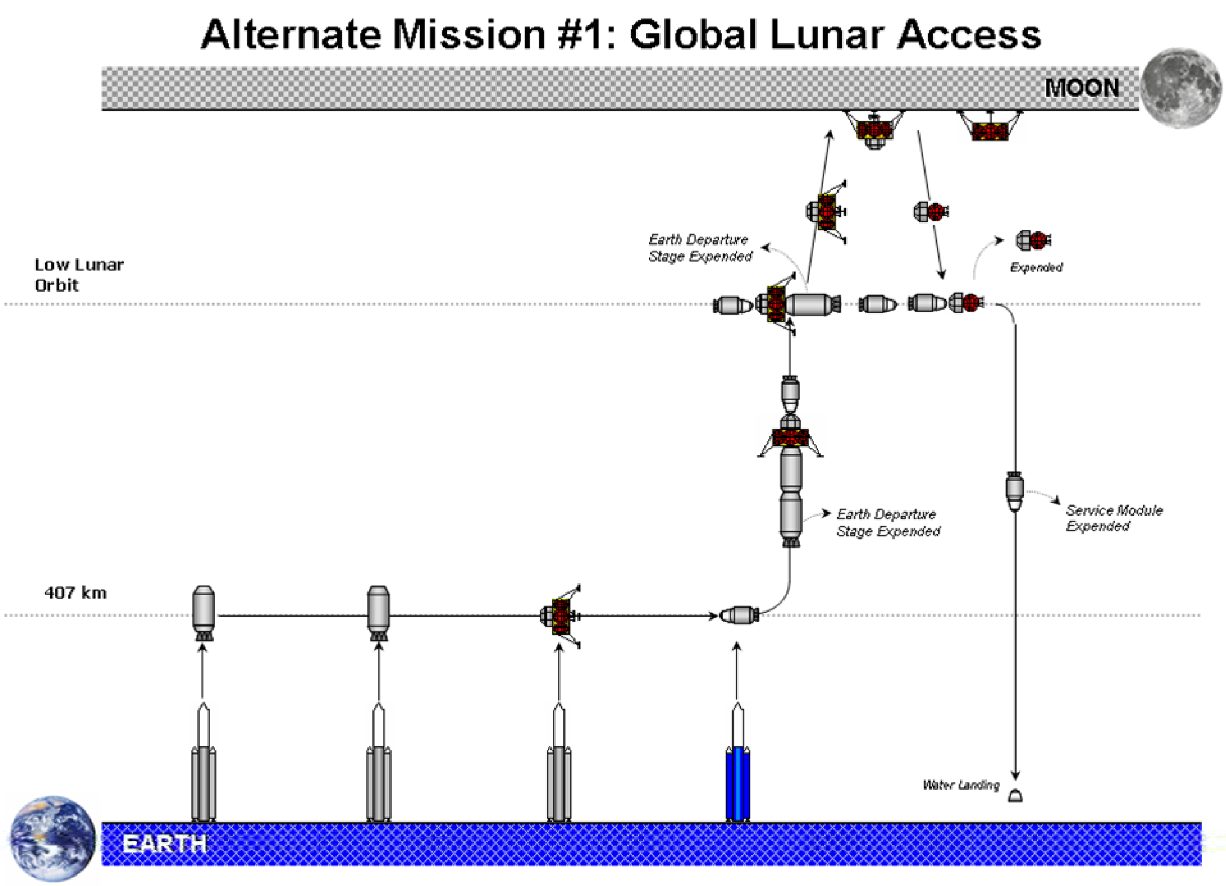

Figure 8.2.1-1: Global Lunar Access Architecture Illustration

Figure 8.2.1-2 and Tables 8.2.1-1 – 8.2.1-2 outline the assumed timelines and delta-V's for the global access LOR alternate mission as described above. The only mission phase added to polar TRM timeline is the 3-impulse plane change sequence part of lunar orbit insertion.

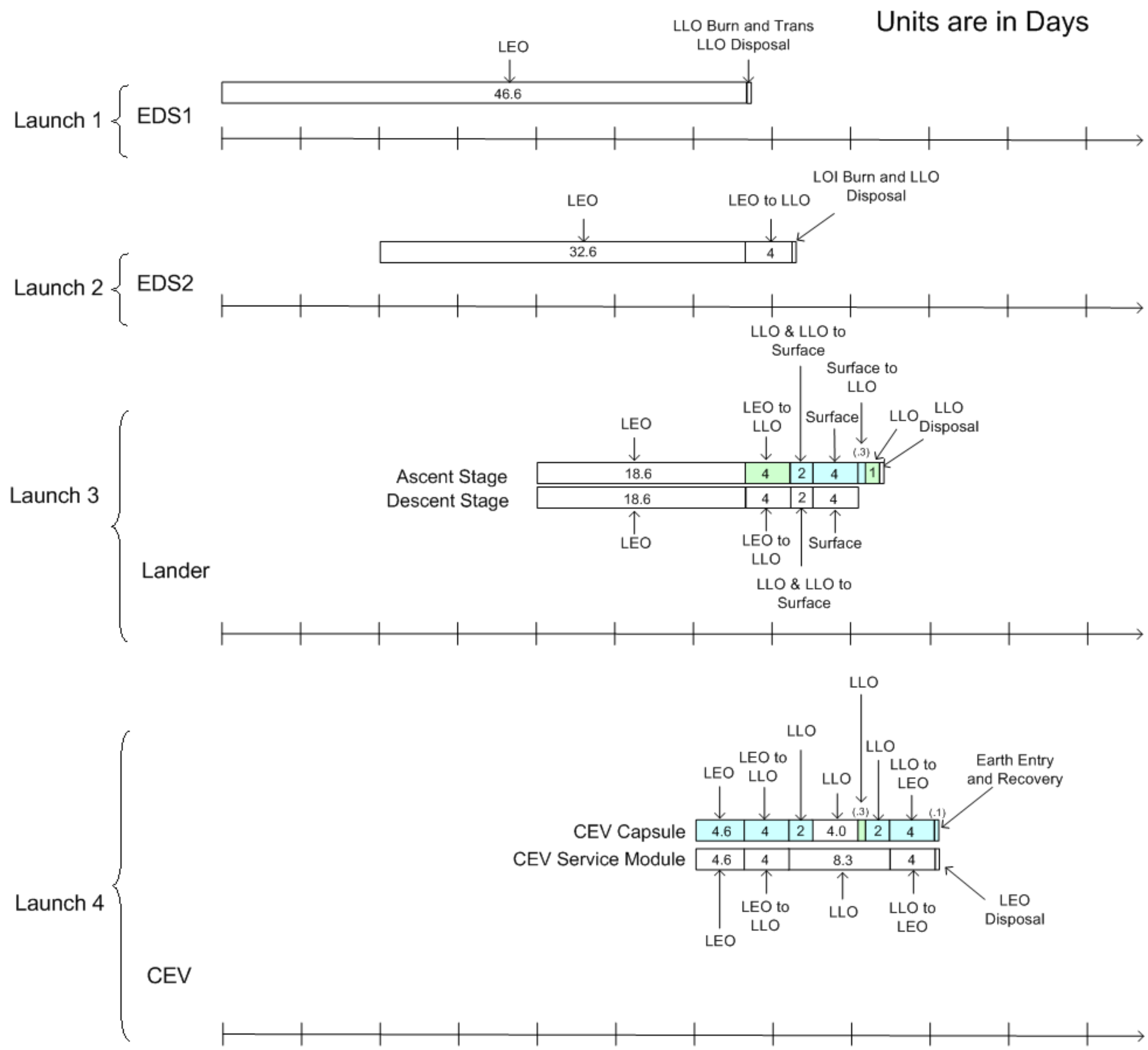

Figure 8.2.1-2: Nominal Timeline for Short-Stay LOR Mission

Vehicle	Phase Name	Phase Length (hr)	Mission Elapsed Time: Overall MET (hr)	Overall MET (days)	EDS1 (hr)	EDS2 (hr)	Lander (hr)	CEV (hr)
EDS1	Launch from Earth/Loiter	2	2	0.1	2			
EDS1	Loiter in LEO	334	336	14.0	336			
EDS2	Launch from Earth/Loiter	2	338	14.1	338	2		
EDS2	Rendezvous & Dock w/ EDS1	50	388	16.2	388	52		
EDS2	Loiter in LEO	284	672	28.0	672	336		
Lander	Launch from Earth/Loiter	2	674	28.1	674	338	2	
Lander	Rendezvous & Dock w/ EDS's	50	724	30.2	724	388	52	
EDS/Lander	Vehicle Checkout	12	736	30.7	736	400	64	
EDS/Lander	Loiter in LEO	272	1008	42.0	1008	672	336	
EDS/Lander	Missed EOD Opportunity	240	1248	52.0	1248	912	576	
CEV	Launch Weather Delay	48	1296	54.0	1296	960	624	48
CEV	Launch from Earth/Loiter	2	1298	54.1	1298	962	626	50
CEV	Rendezvous & Dock w/ Stack	50	1348	56.2	1348	1012	676	100
EDS/Lander/CEV	Vehicle Checkout	12	1360	56.7	1360	1024	688	112
EDS	Earth Orbit Departure	0	1360	56.7	1360	1024	688	112
EDS/Lander/CEV	Coast	48	1408	58.7		1072	736	160
EDS	MCC & EDS Disposal	0	1408	58.7			736	160
EDS/Lander/CEV	Coast	48	1456	60.7			784	208
EDS2	Lunar Orbit Insertion	0	1456	60.7			784	208
CEV	3-Impulse Plane Change	24	1480	61.7			808	232
Lander/CEV	Crew Transfer & Checkout	24	1504	62.7			832	256
Lander	Undock from CEV	0	1504	62.7			832	256
Lander	Powered Descent	0	1504	62.7			832	256
Lander	Surface Mission	2160	1672	69.7			1000	424
Lander	Ascent	0	1672	69.7			1000	424
Lander	Rendezvous & Dock w/ CEV	6	1678	69.9			1006	430
Lander/CEV	Crew Transfer & Checkout	24	1702	70.9			1030	454
CEV	Undock from Lander	0	1702	70.9			1030	454
Lander	Ascent Stage Disposal	0	1702	70.9			1030	454
CEV	3-Impulse Plane Change	24	1726	71.9				478
CEV	Lunar Orbit Departure	0	1726	71.9				478
CEV	Coast	93	1819	75.8				571
CEV	Dispose Service Module	0	1819	75.8				571
CEV	Coast & Entry	3	1822	75.9				574
CEV	Recovery	1	1823	76.0				575

Table 8.2.1-1: Short-Stay LOR Mission Phase Description

Maneuver Name	Element	ΔV (m/s)	Comments
Earth Orbit Departure	EDS1 & EDS2	3,104	Co-planar departure from LEO assembly orbit (407 km, 28.5^o) w/ 24-hr injection window. Nominal flt time to lunar orbit = 96 hr. Moon @ perigee.
Lunar Orbit Insertion	EDS2	1,416	Insertion into 100x100 km orbit tailored to landing site (V_∞ = 986 m/s). Includes 3-impulse insertion maneuver w/ 24-hr intermediate orbit for 90^o worst-case relative declination angle.
In-Plane Powered Descent	Descent Stage	1,881	Fuel-optimal powered descent design for in-plane descent from 100x100 km orbit (ref. First Lunar Outpost study)
In-Plane Ascent	Ascent Stage	2,025	Fuel-optimal powered ascent design for ascent to 100x100 km orbit. Includes 191 m/s for 6.7^o on-orbit plane change (anytime ascent for up to 7-day surface stay, only 4-day surface stay possible without Lander modification)
Lunar Orbit Departure	CEV	1,410	Departure from 100x100 km orbit tailored to landing site (V_∞ = 952 m/s). Includes 3-impulse departure maneuver w/ 24-hr intermediate orbit for 90^o worst-case relative declination angle. Nominal flt time to Earth = 96 hr.

Table 8.2.1-2: Summary of Major Maneuvers for the Short-Stay LOR Mission

8.2.2 Global Lunar Access Safety and Mission Success

The global lunar access alternate mission adds one crewed critical event to the list generated for the polar LOR trade reference mission. This mission requires a 3-impulse plane change sequence upon lunar orbit insertion to tailor the parking orbit inclination and ascending node, while the polar TRM only required a 1-impulse event to insert the CEV and Lunar Lander into a polar parking orbit.

8.2.3 Mission Aborts

Mission abort options for Alternate Mission #1- Global Lunar Access are identical to the polar lunar orbit rendezvous TRM.

8.3 Element Overview & Mass Properties

This section describes any changes made in sizing the global lunar access mission elements compared to the element configurations in the polar LOR baseline. Preliminary analysis indicates that the CEV and Lander as designed for the TRM are capable of performing a 4-day mission to anywhere on the lunar surface, however the mass of EDS1 and EDS2 must be increased by ~3 t each to enable global coverage.

Note: All Masses in 1000's of Kilograms

MASS

Delta-V Allocation		100 km LPO >70°	100 km LPO >80°	100 km LPO 90°	Alt Msn #1 Global 4-day
LOI/LPA -> EDS Asc Pln Chg -> Lander	Arch Mass	184	165	149	176
	CEV Mass	20	20	20	20
	Lander Mass	40	34	28	29
	EDS Mass	62	55	50	63
LOI/LPA -> EDS Asc Pln Chg -> CEV	Arch Mass	178	163	149	175
	CEV Mass	25	23	20	22
	Lander Mass	34	31	28	28
	EDS Mass	60	55	50	63
LOI/LPA -> EDS Asc Pln Chg -> Both	Arch Mass	197	171	149	TBD
	CEV Mass	25	23	20	
	Lander Mass	40	34	28	
	EDS Mass	66	57	50	
LOI/LPA -> CEV Asc Pln Chg -> Lander	Arch Mass	203	182	164	204
	CEV Mass	45	43	41	57
	Lander Mass	40	34	28	29
	EDS Mass	59	53	48	59
LOI/LPA -> CEV Asc Pln Chg -> CEV	Arch Mass	200	181	164	205
	CEV Mass	51	45	41	59
	Lander Mass	34	31	28	28
	EDS Mass	58	52	48	59

Note: All Delta-V's in m/s

DELTA-V

Delta-V Allocation		100 km LPO >70°	100 km LPO >80°	100 km LPO 90°	Alt Msn #1 Global
LOI/LPA -> EDS Asc Pln Chg -> Lander	Arch Delta-V	10.5	9.9	9.3	10.1
	CEV Delta-V	1.6	1.6	1.6	1.6
	Lander Delta-V	4.9	4.3	3.7	3.9
	EDS Delta-V	4.0	4.0	4.0	4.6
LOI/LPA -> EDS Asc Pln Chg -> CEV	Arch Delta-V	10.5	9.9	9.3	10.1
	CEV Delta-V	2.1	1.9	1.6	1.8
	Lander Delta-V	4.3	4.0	3.7	3.7
	EDS Delta-V	4.0	4.0	4.0	4.6
LOI/LPA -> CEV Asc Pln Chg -> Lander	Arch Delta-V	10.5	9.9	9.3	10.1
	CEV Delta-V	2.4	2.4	2.4	3.0
	Lander Delta-V	4.9	4.3	3.7	3.9
	EDS Delta-V	3.1	3.1	3.1	3.1
LOI/LPA -> CEV Asc Pln Chg -> CEV	Arch Delta-V	10.5	9.9	9.3	10.1
	CEV Delta-V	3.0	2.7	2.4	3.2
	Lander Delta-V	4.3	4.0	3.7	3.7
	EDS Delta-V	3.1	3.1	3.1	3.1

Table 8.3-1: Comparison of the Polar TRM and Global Lunar Access Missions

Table 8.3-1 shows the required vehicle and total architecture masses and delta-V's for different allocations of the lunar orbit insertion and ascent plane change maneuvers. The polar TRM, highlighted in blue, has a CEV mass of 25 t, a Lander mass of 34 t, and EDS1/EDS2 mass of 60 t. To the right of the polar TRM is the global lunar access mission. The CEV and Lander for that mission have masses 3 t and 6 t less than the TRM, respectively. EDS1 and EDS2 require a mass of 63 t, though, which exceeds the 60 t sizing of the TRM. This is due to the greater delta-V needed for lunar orbit insertion in the global short-stay mission (1,416 m/s) as compared to the baseline (878 m/s). The extra delta-V offsets the decrease in stage payload mass (the Lander and CEV). The table above also shows that if the polar TRM were designed for access between 80^o – 90^o rather than for landing site latitudes above 70^o, the CEV and Lander would still be capable of performing global 4-day missions. The EDS mass would have to be increased by 8 t, though, instead of 3 t.

The following sections describe any differences in vehicle sizing for the CEV, Lander, and Earth Departure Stages.

8.3.1 Crew Exploration Vehicle

The primary changes to the Crew Exploration Vehicle for the global lunar access missions are made in total mission duration and delta-V. The CEV here has a much lower mission duration owing to the 4-day surface mission instead of 90 days in the TRM, however it does require one extra day of crew consumables for the four crewmembers. This extra day of consumables comes with the 3-impulse 24-hr sequence added for lunar orbit insertion. Table 8.3.1-1 outlines the change in CEV Command Module mass for the short-stay mission. The CM mass is only increased by 39 kg.

CEV Crew Module's System Mass Changes				
System	**Polar TRM**	**Global 4-Day**	**Mass Change (kg)**	**% Change**
Structure	1523	1523	No Change	0.0
Protection	816	818	2	0.2
Propulsion	117	117	No Change	0.0
Power	482	482	No Change	0.0
Control	0	0	No Change	0.0
Avionics	737	737	No Change	0.0
Environment	709	716	7	1.0
Other	833	833	No Change	0.0
Growth	1043	1045	2	0.2
Non-Cargo	884	896	12	1.4
Cargo	1478	1478	No Change	0.0
Non-Propellant	329	345	16	4.9
Propellant	64	64	No Change	0.0
Total	9015	9054	39	0.4

Table 8.3.1-1: Variation in CEV CM Mass with Short-Stay Global LOR

As Table 8.3.1-2 shows, the CEV Service Module has a significantly lower mass than in the polar TRM despite the slight increase in CM mass. This is because of a lower required delta-V for ascent plane changes. The polar TRM carries a 20^{o} plane change capability to meet the anytime ascent requirement, which translates into a 567 m/s delta-V penalty. With an orbit tailored to a specific landing site, though, the ascent plane change penalty for a short-stay mission can be limited to only a few degrees. The vehicle sizing given below assumes a required plane change capability of 6.7^{o} for a 7-day mission (191 m/s), the value used for the Phase 1 LOR variant. That delta-V savings provides 2.7 t of decreased CEV SM mass. Some additional savings are possible if the ascent plane change for a 4-day mission were used for the sizing. The plane change cost there would be 2.1^{o} or 60 m/s.

CEV Service Module's System Mass Changes				
System	**Polar TRM**	**Global 4-Day**	**Mass Change (kg)**	**% Change**
Structure	1445	1424	(21)	(1.5)
Protection	0	0	No Change	0.0
Propulsion	1338	1338	No Change	0.0
Power	384	384	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	73	73	No Change	0.0
Other	100	100	No Change	0.0
Growth	668	664	(4)	(0.6)
Non-Cargo	305	237	(68)	(22.3)
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	11518	8959	(2559)	(22.2)
Total	15831	13179	(2652)	(16.8)

Table 8.3.1-2: Variation in CEV SM Mass with Short-Stay Global LOR

8.3.2 Lunar Lander

The Lander Ascent Stage has the same mass as in the polar TRM since both vehicles provide the same mission duration and delta-V. The vehicles are limited to the 4-day stay on the Moon and only perform in-plane ascent to a 100 km circular orbit.

The Lander Descent Stage mass is 6.2 t lower here, though, because of a lower delta-V requirement. The Descent Stage for the polar TRM is required to perform a 20^{o} plane change on descent in order to access any landing site longitude in the 70^{o}-90^{o} latitude zone. This adds 567 m/s of delta-V to the 1,881 m/s it carries for in-plane descent from a 100 km parking orbit. The "Global Access" Lander only performs in-plane descent since the Earth Departure Stage puts it into an orbit tailored to the mission landing site, and this lower delta-V results in the mass savings seen in Table 8.3.2-2.

No attempt has been made to determine whether a Lander vehicle designed to land and operate at a near-polar landing site is capable of withstanding the thermal and lighting conditions that may be seen in a 4-day mission at a non-polar site. That is a subject for future analysis.

Lander's Ascent Stage's System Mass Changes				
System	**Polar TRM**	**Global 4-Day**	**Mass Change (kg)**	**% Change**
Structure	788	788	No Change	0.0
Protection	73	73	No Change	0.0
Propulsion	1189	1189	No Change	0.0
Power	737	737	No Change	0.0
Control	0	0	No Change	0.0
Avionics	738	738	No Change	0.0
Environment	786	786	No Change	0.0
Other	455	455	No Change	0.0
Growth	953	953	No Change	0.0
Non-Cargo	1202	1202	No Change	0.0
Cargo	227	227	No Change	0.0
Non-Propellant	409	409	No Change	0.0
Propellant	5322	5322	No Change	0.0
Total	12878	12878	No Change	0.0

Table 8.3.2-1: Variation in Lander Ascent Stage Mass with Short-Stay Global LOR

Lander's Descent Stage's System Mass Changes				
System	**Polar TRM**	**Global 4-Day**	**Mass Change (kg)**	**% Change**
Structure	549	501	(48)	(8.7)
Protection	50	50	No Change	0.0
Propulsion	1373	1008	(365)	(26.6)
Power	137	137	No Change	0.0
Control	0	0	No Change	0.0
Avionics	0	0	No Change	0.0
Environment	0	0	No Change	0.0
Other	514	498	(16)	(3.1)
Growth	525	439	(86)	(16.4)
Non-Cargo	450	304	(146)	(32.4)
Cargo	500	500	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	17046	11535	(5511)	(32.3)
Total	21144	14973	(6171)	(29.2)

Table 8.3.2-2: Variation in Lander Descent Stage Mass with Short-Stay Global LOR

8.3.3 Earth Departure Stages

The only change made to the Earth Departure Stages is a higher total delta-V for lunar orbit insertion. As discussed previously, EDS2 performs a minimum delta-V maneuver in the polar TRM to insert the CEV and Lander into a polar parking orbit. This maneuver requires a delta-V of 878 m/s. For the global access alternate mission, the lunar parking orbit is tailored in inclination and ascending node to the mission's landing site, which may require a 90^{o} plane change at insertion. This extra 538 m/s of delta-V results in an Earth Departure Stage design mass that is 3.6 t higher than in the polar TRM despite the combined CEV and Lander mass being 8,784 kg less. No other vehicle design changes were made.

EDS1 and EDS2 System Mass Changes				
System	**Polar TRM**	**Global 4-Day**	**Mass Change (kg)**	**% Change**
Structure	1366	1441	75	5.5
Protection	0	0	No Change	0.0
Propulsion	3130	3232	102	3.3
Power	190	190	No Change	0.0
Control	0	0	No Change	0.0
Avionics	175	175	No Change	0.0
Environment	105	105	No Change	0.0
Other	455	455	No Change	0.0
Growth	1084	1120	36	3.3
Non-Cargo	2377	2502	125	5.3
Cargo	0	0	No Change	0.0
Non-Propellant	0	0	No Change	0.0
Propellant	50772	54016	3244	6.4
Total	59655	63236	3581	6.0

Table 8.3.3-1: Variation in EDS Mass with Short-Stay Global LOR

8.3.4 Vehicle Mass Properties for Trade Reference Mission

Global lunar access vehicle mass properties as generated by the Envision parametric sizing tool are listed in Table 8.3.4-1. Subsystem components are categorized according the mass properties reporting standard outlined in JSC-23303 Design Mass Properties: Guidelines and Formats for Aerospace Vehicles. All estimates include 20% margin applied to categories one through eight of the vehicle mass properties for dry mass growth.

	CEV CM	CEV SM	Ascent Stage	Descent Stage	Earth Dep. Stage #1	Earth Dep. Stage #2
1.0 Structure	1,523	1,424	788	501	1,441	1,441
2.0 Protection	818	0	73	50	0	0
3.0 Propulsion	117	1,338	1,189	1,008	3,232	3,232
4.0 Power	482	384	737	137	190	190
5.0 Control	0	0	0	0	0	0
6.0 Avionics	737	0	738	0	175	175
7.0 Environment	716	73	786	0	105	105
8.0 Other	833	100	455	498	455	455
9.0 Growth	1,045	664	953	439	1,120	1,120
DRY MASS	**6,272 kg**	**3,983 kg**	**5,718 kg**	**2,634 kg**	**6,718 kg**	**6,718 kg**
10.0 Non-Cargo	896	237	1,202	304	2,502	2,502
11.0 Cargo	1,478	0	227	500	0	0
INERT MASS	**8,645 kg**	**4,221 kg**	**7,147 kg**	**3,438 kg**	**9,220 kg**	**9,220 kg**
12.0 Non-Propellant	345	0	409	0	0	0
13.0 Propellant	64	8,959	5,322	11,535	54,016	54,016
GROSS MASS	**9,054 kg**	**13,179 kg**	**12,878 kg**	**14,973 kg**	**63,236 kg**	**63,236 kg**

Table 8.3.4-1: Short-Stay LOR (Global Access) Vehicle Mass Properties

The two largest single elements to be launched are the equal-mass Earth Departure Stages for the Lunar Lander and CEV with an initial mass in low Earth orbit (IMLEO) of 63.2 t. These two stages are the first elements launched in the architecture, and they execute the Earth orbit departure and lunar orbit insertion burns for the Lander and CEV. The launch of these elements will drive the payload delivery capabilities of the cargo launch vehicle. The Lunar Lander is launched next with a launch mass of 27.9 t. Finally, the CEV is launched with the crew on a human-rated launch vehicle capable of delivering 22.2 t to LEO. The combined architecture elements of Alternate Mission Capture #1 – Global Lunar Access have a total IMLEO of 175 t which compares to 178 t for the polar TRM.

8.4 Summary

Preliminary architecture element sizing indicates that the CEV and Lunar Lander, as designed for the near-polar LOR trade reference mission, have sufficient mass to perform a 4-day mission to any landing site on lunar surface. The CEV will require one additional day of crew consumables to accommodate a 24-hr sequence of maneuvers on lunar orbit insertion, but this added mass can be offset with propellant mass savings in the Service Module. The Lunar Lander for the polar TRM is already designed for a 4-day operational capability on the surface, so no change is needed for that element. However, the Earth Departure Stages that would be required for a short-stay global access mission each have a total mass that is 3 t more than the TRM design. If global short-stay missions will be a required capability for Project Constellation vehicles, the EDS and cargo launch vehicle mass will have to be increased to meet those needs. Otherwise, the regions of the Moon that are accessible for short-stay missions may be restricted.

If global access missions longer than 4 days are desired, it may be possible to use the Lander Descent Stage propellant mass savings achieved with this variant to carry additional power and life support consumables without exceeding the design of the "Polar TRM" Lander. This would further increase the required size of the Earth Departure Stage, though. Finally, it should be noted while the Lander mass from the polar TRM is sufficient to support this mission, a "Global Access" Lander may require design changes to handle the different thermal and lighting conditions of a non-polar landing site. A Lander designed to operate with crew on-board for longer than 4 days may also require some modifications. Both of these considerations require more analysis.

9.0 Alternate Mission Capture #2 – L1 CEV Dormant

This study examined the ability of the Phase 2 LOR TRM elements to perform the operations associated with a mission in which the crew is transported to a pre-emplaced asset at L1, performs a mission out of that pre-emplaced asset, and then the crew is returned to Earth.

9.1 Major Assumptions/Clarifications

This section outlines the major architecture assumptions made in formulation of this alternate mission. These assumptions were levied by the study's NASA HQ customer on the LDRM-2 study team to be used as an initial point of reference.

One human mission per year: This is a programmatic assumption dictated in the LDRM-2 task statement. Mission rate has no impact on the analyses performed in this study.

All mission elements placed in LEO (28.5 deg 407 km circular): L1 missions will require the mating of elements in Earth orbit prior to departure for L1. Launches into 28.5^{o} inclination orbits allow the maximum payload to orbit from the Eastern Test Range. Additionally, this inclination affords large planar launch windows required for rendezvous. The assembly altitude of 407 km is specified to minimize the effects of atmospheric drag on orbital lifetime while minimizing payload deployment altitude required on the launch vehicle upper stage. Future trades between the launch vehicle and orbital elements will be required to determine the optimum staging altitude.

Consider the mission elements to be "cargo" in terms of delivery to the LEO parking orbit: The launch vehicle will be responsible for delivering architecture elements to a 28.5^{o} 407 km circular orbit. This assumption puts the entire burden of cargo delivery on the launch vehicle, which helps to determine maximum launch vehicle capabilities. For this study, the propulsive capabilities of the mission elements will not be employed for orbit insertion, but will likely be required for orbit maintenance. Future trades can be performed to optimize the allocation of the orbit insertion function between the launch vehicle and orbital elements.

Automated rendezvous and docking shall be used to assemble the elements: Mission elements from the lunar missions will already be designed with automated rendezvous and docking (AR&D) capabilities, due to their functionality during those missions. Those same AR&D capabilities will be used during these L1 missions.

Assume 2 weeks between launches: This assumption is a balance between a desire to minimize total mission duration and vehicle lifetime while not severely impacting launch vehicle production, processing, and launch facilities for a four launch per mission baseline. A launch vehicle processing trade study will be required to determine the feasibility of meeting this assumption.

Mission at L1 pre-emplaced asset will be 90 days: This is the duration of missions that are envisioned for L1, such as telescope deployment or satellite repair. This duration was dictated by NASA HQ. The purpose of this study is to assess whether the TRM elements encapsulate the capabilities required to perform a 90-day L1 mission.

Crew must be launched on a human-rated launch system: This is dictated by the NASA human rating requirements document NPR 8705.2.

The architecture will support 4 crew: This assumption was dictated by NASA HQ and was used as the crew size for sizing the architecture.

The CEV is not required to be reusable and will not be explicitly designed for reusability: Previous spaceflight experience has taught that reusability should not be dictated a priori, rather the decision to reuse vs. build new should be made based on cost and schedule trades for a given flight rate and total program duration.

The CEV will provide the crew habitation function from Earth's surface to the vicinity of L1 and back to Earth's surface: The crew will use the CEV for transportation between the surface of the Earth and L1 and back, but will not use the CEV as a living quarters at L1.

The CEV will be placed in a dormant mode during the mission at L1 and a pre-emplaced element located at L1 will serve as the primary living quarters and workplace for the crew during their mission: The CEV will not serve as a living quarters or workspace at L1; therefore, it will be place in a dormant mode.

The nominal Earth return for the CEV is direct entry with a water landing: Direct entry followed by a water landing is a proven and reliable way of returning crews from Earth-Moon transfer orbits. Additionally, some aborts during ascent from Earth may result in water landings. A water landing and recovery will therefore be a capability required by the vehicle. Providing a capability to nominally perform both water and land landings will increase CEV and architecture mass, and CEV complexity (deployment of airbags, firing of retrorockets, etc.).

The CEV design will incorporate functionality for land landing as a contingency for an ascent abort: Some ascent aborts may result in the CEV landing on land. The ability to meet this requirement will be for crew survival only – i.e. the vehicle may be damaged beyond repair/reuse as long as the crew survives.

Radiation shielding shall be incorporated into the design of the CEV crew module to provide a core level of biological protection for the crew during transit: Radiation shielding is required to meet crew safety requirements during solar particle events (SPEs). Short-term and cumulative crew dose limits for exploration missions have not yet been defined.

9.2 Architecture Description

9.2.1 Architecture Description/Operations for L1 CEV Dormant Mission

As with all the alternate missions in this study, the CEV Dormant Mission's architecture was formulated around the use of the LDRM-2 Phase 2 elements. Unlike the LDRM-2 Phase 2 lunar mission, the Lunar Lander element is not needed in this architecture. After a quick assessment of the EDS's capabilities, it was determined that only one EDS would be needed in order to deliver the CEV to L1.

This architecture begins with the launch of an EDS. This is followed two weeks later by the launch of the CEV. The timing of the launch of the EDS is tied to the 24-hour Earth Orbit Departure (EOD) window, which will enable the CEV to depart Earth orbit for L1. Per the guidance of NASA HQ, each launch is separated be two weeks. Additionally, in order to protect the EOD window, which occurs on an average of once every ten days, the CEV will be launched such that it has time to perform rendezvous and docking, protect against weather delays, and allow for the checkout of the systems prior to the EOD burn. Therefore, the launch of the EDS was scheduled for 18.6 days prior to the beginning of the 24-hour EOD window.

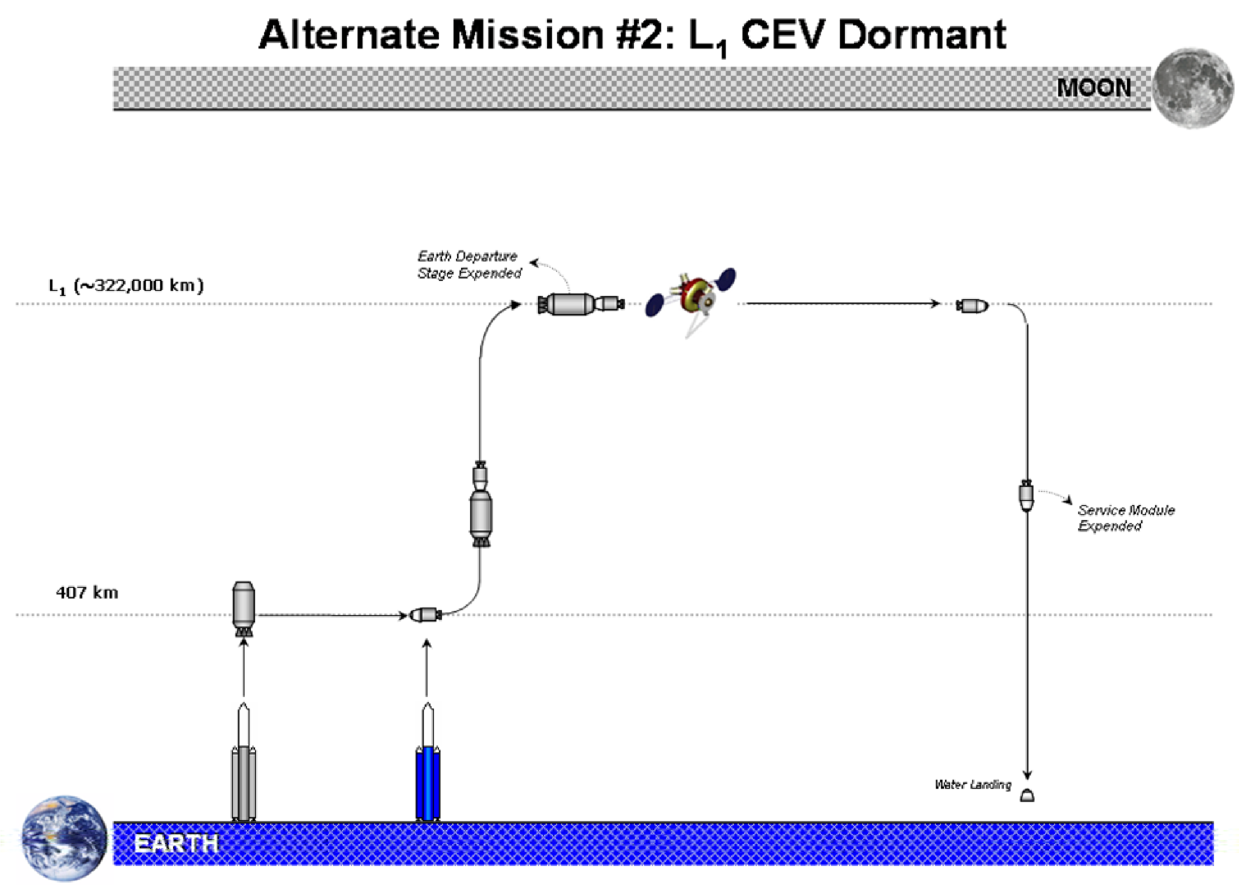

Figure 9.2.1-1: L1 CEV Dormant Architecture Illustration

Once the CEV and EDS are docked in LEO, the crew will have 12 to 36 hours (depending on any launch delays) to perform a checkout of their systems prior to the EOD burn. Once the EOD window opens, the crew will initiate the EOD burn and begin a 94 hour trip to L1. After the EDS has performed the EOD maneuver and any needed mid-course corrections, the CEV and EDS will separate. At this point, the EDS will dispose of itself into Earth's atmosphere and the CEV will continue on its journey to L1.

Upon arrival at L1, the CEV will dock to a pre-emplaced asset. Once docked, the CEV has been sized to provide a 24-hour period for the crew to transfer themselves and any cargo over to the pre-emplaced asset. The CEV will be placed in a dormant mode once the crew has finished the transfer operations. An assumption was also made that some level of data transfer would take place between the CEV and the pre-emplaced asset. There may be multiple benefits of transferring data, but the main thought was that it would allow the crew to have some level of oversight of the CEV while residing within the pre-emplaced asset. At the level of detail of this study, it was not possible to assess whether it would be better to have the CEV to rely upon support from the pre-emplaced asset for other functions, such as cooling or power. Therefore, at this point in time the CEV has been designed to be completely self-sufficient, barring any hindrances from the pre-emplaced asset.

At the end of the 90-day L1 mission, the crew will power-up the CEV and transfer themselves and cargo back into the CEV. The CEV has been sized to accommodate a 24-hour crew/cargo-transfer period. Once the crew has transferred over to the CEV, they will initiate separation. At this point, the crew will time their L1 departure burn and adjust its magnitude in order to achieve the desired landing target on Earth. Once the burn has been initiated, the crew will coast on a trajectory towards Earth of approximately 94 hours. Three hours prior to entry interface, the Crew Module and Service Module will separate. The Service Module will be targeted for atmospheric re-entry where it will break up and land up-range of the Crew Module's targeted landing area. At this point, the Crew Module will re-orient itself for atmospheric re-entry and return the crew to the targeted landing area where they will be rescued and their spacecraft will be recovered.

Figure 9.2.1-2 shows the nominal timeline for this mission.

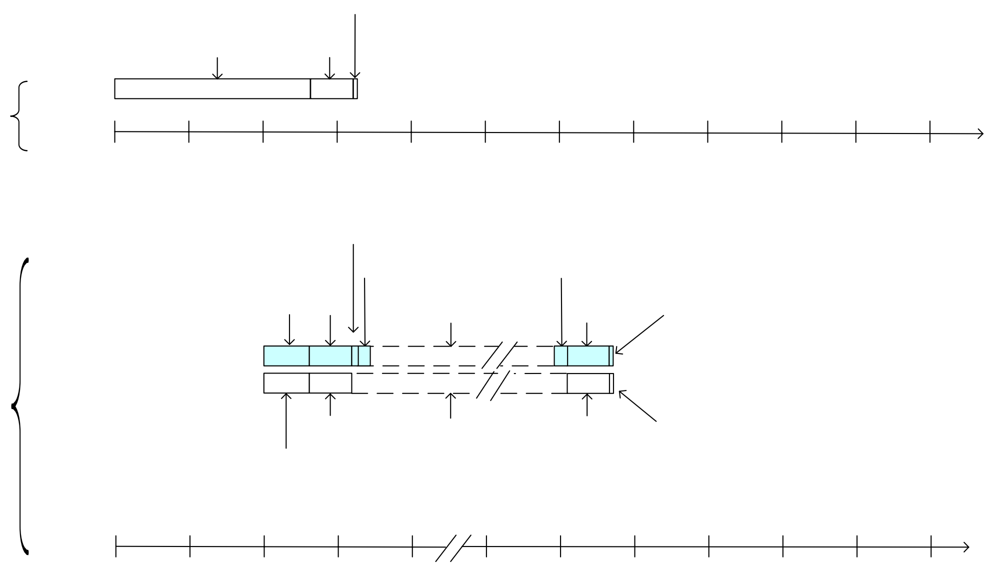

Figure 9.2.1-2: L1 CEV Dormant Architecture Nominal Timeline

9.2.2 Mission Abort Options

Nominal Flight Sequence

1. Launch From KSC
2. EOD (Earth Orbit Departure) and Booster Separation
3. L1 Arrival
4. L1 Operations, Transfer to L1 Asset
5. L1 to Earth Transfer
6. Earth Arrival and Re-entry to Touchdown

L1 Abort options

a. Launch Abort from KSC
b. LEO Deorbit
c. Powered L1 Transfer Abort, Early Return to Earth
d. L1 Swingby, Return to Earth
e. Crew Transfers to CEV and Departs L1, Shorten Transfer Time
f. Earth Arrival, Ballistic Reentry to Unplanned Water or Land Landing Site

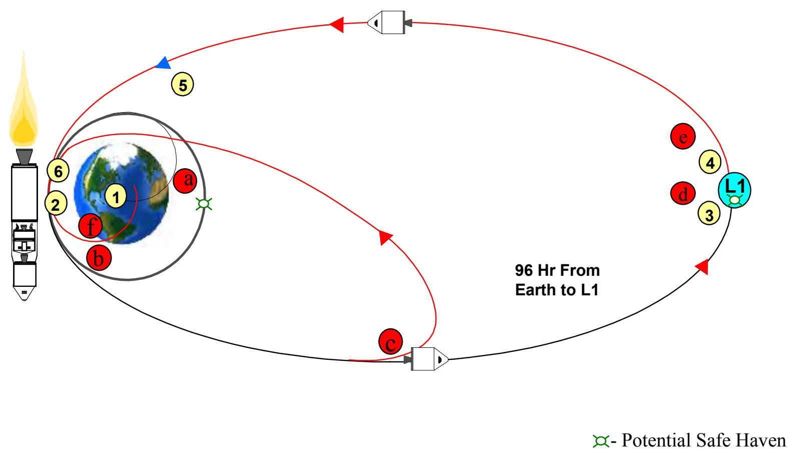

Figure 9.2.2-1: Mission Abort Options for L1 CEV Dormant Architecture

The mission abort options for this architecture are very similar to the Earth-L1-Earth segments described in the Lunar Architecture Focused Trade Study Final Report Volume 1 report in section 10.4.3. Therefore, the discussion of those applicable mission phases has been repeated (and adapted) for this section.

Aborts were developed and assessed for each mission phase from low Earth orbit to L1 and the return to Earth's surface. Figure 9.2.2-1 illustrates the nominal mission flight phases and their corresponding mission abort opportunities.

1. Launch and Ascent to Low Earth Orbit (LEO)

This mission phase begins with the launch from the surface of the Earth and ends after the vehicle is established in the desired LEO.

a. Booster or major CEV system failure

i. CEV emergency separates and returns to Earth

During the CEV launch and ascent to LEO should the Human-Rated Launch Vehicle (HRLV) or the CEV suffer catastrophic failure the CEV can initiate the Launch Abort System, triggering an emergency separation from the HRLV and return to Earth using the CEV descent and touchdown systems.

2. LEO Orbit and Rendezvous Operations

This mission phase begins after the vehicle is in LEO and ends after the completion of LEO rendezvous and mating of the Earth Departure Stage and the CEV.

a. CEV systems failure or failure to mate to Earth Departure Stage (EDS)

i. CEV de-orbit and return to Earth

Once the CEV has reached LEO, should the CEV suffer a significant system failure prior to initiating the EDS L1 transfer burn, the CEV must perform a standard de-orbit maneuver, reenter and touchdown on land or water. If the abort takes place after the CEV mates to the EDS, the CEV must separate from the EDS prior to re-entry. If CEV propulsion system failures preclude performing a de-orbit maneuver, the EDS could be used for that maneuver. Otherwise the CEV is stranded in LEO and an Earth based CEV rescue mission is required to prevent a loss of crew (LOC) event from occurring. The CEV would need the appropriate resources to provide this safe haven for the crew until that rescue mission is performed (x-weeks).

3. LEO to L1 Transfer

This phase begins at the start of the (L1) transfer burn and ends just before the start of the L1 arrival burn.

a. Early EDS shutdown and high elliptical orbit

i. CEV maneuver to desired orbit

ii. De-orbit burn and re-entry to touchdown

Should the EDS fail to fully complete the L1 departure burn the CEV can separate, perform any required transfer orbit adjustments within the limits of available CEV propulsion constraints, establish a return to Earth trajectory and perform a de-orbit and re-entry to touchdown. After completion of the L1 transfer burn the CEV can also abort by eliminating the L1 arrival burn and returning to Earth on the elliptical transfer orbit. The CEV can adjust this orbit within CEV propulsion constraints to ensure a safe Earth re-entry and touchdown.

4. L1 Operations

This phase begins at the start of the L1 arrival burn and includes all rendezvous and mating operations. This phase ends after pre-emplaced asset/CEV separation just prior to the CEV departing for Earth.

a. No CEV L1 arrival burn by EDS

i. CEV swing-by at L1 and return to Earth

If the CEV's EDS can not perform the L1 arrival burn, the CEV can abort by continuing on the current elliptical transfer orbit and use the Service Module to perform any maneuvers necessary to establish a safe return-to-Earth trajectory for a direct re-entry.

b. L1 Rendezvous and mating with the pre-emplaced asset

i. CEV return to Earth

c. Crew transfer failure

i. CEV return to Earth

d. Event worthy of shortening L1 mission

i. CEV times early L1 departure burn and/or magnitude of burn to reach appropriate Earth longitude for early Earth return.

5. L1 to Earth Transfer

This phase begins with the CEV L1 departure burn and ends just prior to Earth atmospheric re-entry.

a. No L1 burn post CEV/pre-emplaced asset separation by Service Module

i. Use CEV RCS or pre-emplaced asset's propulsion to mate the CEV and pre-emplaced asset.

Crew transfers to pre-emplaced asset and waits for rescue mission.

ii. Rescue mission staged from Earth

The CEV can support the crew for up to 5 days plus any margin. Depending on the capabilities of the Earth-based assets and the timing of the EOD opportunities, it may be possible to stage a rescue mission.

6. Earth Arrival and Re-entry to Touchdown

This phase begins with the direct re-entry into Earth atmosphere and ends with CEV touchdown on the Earth's surface.

a. Re-entry flight control failures

i. Ballistic re-entry (no lift vector control)

b. Entry targeting failures

i. Water or land touchdown

CEV equipped with appropriate crew survival and search and rescue gear for touchdown site

The CEV will be designed to support either land or water touchdown allowing for entry targeting failures to force the CEV to miss the desired landing site. The LDRM architecture is using 3 hours as the time required to find and recover the crew from the CEV after touchdown.

9.3 Mass Property Differences

The purpose of this study was to assess the ability of the LDRM-2 Phase 2 TRM elements to accomplish this mission. This study focused on understanding the deltas between what the Phase 2 TRM elements were sized for and what was needed in order to accomplish the objectives of this alternate mission. Therefore, none of the mission element designs were changed. Refer to sections 7.3.1 and 7.3.3 for the mass properties of the Phase 2 TRM EDS and CEV.

The right-hand graph in Figure 9.3-1 shows the differences in delta V that the CEV and EDS are required to deliver in the two different architectures. The Phase 2 TRM CEV is required to perform a maneuver of 159 m/s in order to rendezvous in LEO and then perform another maneuver of 1,977 m/s in order to depart lunar orbit for Earth. The CEV for this alternate mission is required to perform a maneuver of 159 m/s in order to rendezvous in LEO, but performs a maneuver of only 798 m/s in order to depart L1 for Earth. This results in a total delta V of 2,136 m/s for the TRM CEV and 957 m/s for this alternate mission's CEV, as shown in Figure 9.3-1.

The Phase 2 TRM EDS1 is required to perform a maneuver of 1,489 m/s in order to perform its portion of the EOD burn. A mid-course correction of 50 m/s was also book kept. The EDS for this alternate mission required a total delta V of 4,058 m/s for the EOD burn and L1 arrival burn, combined. A mid-course correction of 10 m/s was also book kept. This results in a total delta V of 1,539 m/s for the TRM EDS1 and 4,068 m/s for this alternate mission's EDS, as shown in Figure 9.3-1.

Using these delta V's, the CEV and EDS masses were calculated. The left-hand graph in Figure 9.3-1 shows the mass delta between the Phase 2 TRM CEV and EDS elements versus what is needed in order to accomplish this alternate mission. The delta in the CEV's mass occurs due to propellants that were off-loaded in the Service Module. The TRM Service Module required a total of 11,518 kg of propellant, which led to a total IMLEO of 15,831 kg for this element. Due to the decreased propulsive requirements of this alternate mission, only 4,317 kg of propellant was needed by the Service Module, which led to a total IMLEO of 8,370 kg for this element. Therefore, the IMLEO of the CEV is able to be reduced by 7,461 kg for this alternate architecture.

The difference in the EDS masses occur for the same propellant off-loading reasons. Each TRM EDS required a total of approximately 51,000 kg, which led to a total IMLEO of approximately 60,000 kg. Although the delta V requirements for the EDS in this alternate mission are much greater than those of the TRM, the EDS's payload is much less in the alternate mission. Not only is the CEV mass much less in this alternate architecture, but the TRM EDS1 was required to perform a portion of the EOD burn while pushing EDS2 and the Lunar Lander in addition to the CEV. Therefore, due to the decreased payload and resultant decreased propulsive requirements of this alternate mission, only 35,250 kg of propellant was needed by the alternate mission's

EDS, which led to a total IMLEO of 42,288 kg for this element. Therefore, the IMLEO of the EDS is able to be reduced by approximately 17,700 kg for this alternate architecture.

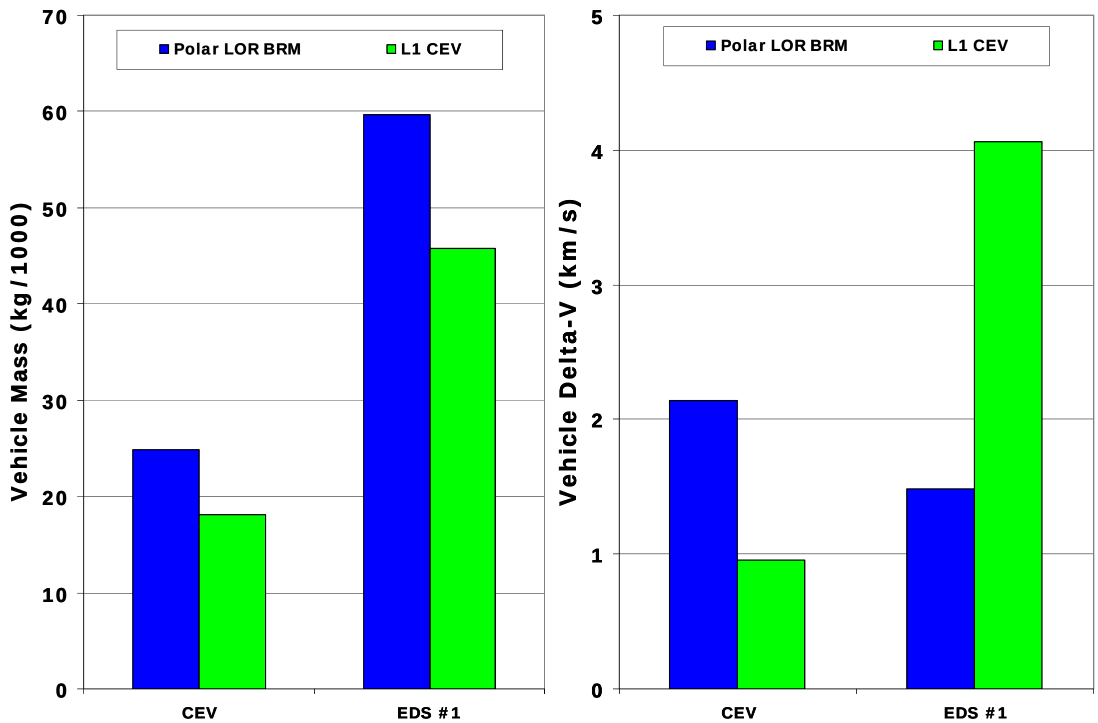

Figure 9.3-1: CEV/EDS Mass and Delta V Comparison Between the LDRM-2 Phase 2 TRM and L1 CEV Dormant Missions

9.4 Summary

The LDRM-2 Phase 2 TRM elements (CEV and EDS) easily encapsulate the capabilities required to perform the L1 CEV Dormant alternate mission. This alternate mission could be performed with the exact same CEV and EDS required for the TRM, with off-loaded propellants. Therefore, it would allow mission planners flexibility in carrying extra propellant (which could possibly be used for enhanced abort options or extra mission objectives) or it could allow the CEV and EDS to have lower IMLEO's (which may have cost or processing benefits).

At the level of detail of this study, the CEV was designed to be self-sufficient for its dormant period of operation at L1. Therefore, the mission duration might only be limited by the Limited Life Items (LLI's) that are used on the CEV and the ability of the pre-emplaced asset to support the crew and CEV.

However, it was not possible to perform an exhaustive study of the CEV/L1 asset's interaction without knowing more about the pre-emplaced asset. Depending on the pre-emplaced asset's design and required orientation, it may be necessary to provide the means for thermal control fluid exchange, cabin air gas exchange, power exchange, or some other form of interaction between the two elements. Some level of data exchange between the two elements would probably be highly desirable as well, which would allow the crew to monitor the status of the CEV while residing within the pre-emplaced asset.

Each phase of this mission allowed the crew to either abort the mission and return to Earth or reach a safe-haven, with one possible exception. Since both the TRM Service Module and EDS are oversized for this mission, the crew is assured adequate delta V in order to return from wherever they are during the course of the mission. The one exception to the crew's ability to return to Earth is in the case of a Service Module propulsion system malfunction at the time of L1 departure. In this situation, the crew could try one of two options. First, they could try to re-mate with the L1 asset using the CEV RCS thrusters or the propulsion system of the L1 asset. Second, if they cannot return to the L1 asset, they would be forced to reside in the CEV while a rescue mission is staged. This is probably the portion of the mission during which this failure has the highest risk. In this situation, the crew could reside within the CEV for a period of up to nearly 5 days plus any margin. Depending on the capabilities of the Earth-based assets and the L1 EOD opportunities, it may be possible to launch an EDS and CEV in time to stage a rescue mission. However, since the nominal lifetime of the CEV is only 5 days at this point of the mission, it may be best to put resources into assuring that the Service Module's systems are highly reliable.

10.0 Alternate Mission Capture #3 – L1 CEV Active

This study examined the ability of the Phase 2 LOR TRM elements to perform the operations associated with a mission in which the crew travels to L1, performs a mission while residing in the CEV, and then returns to Earth.

10.1 Major Assumptions/Clarifications

This section outlines the major architecture assumptions made in formulation of this alternate mission. These assumptions were levied by the study's NASA HQ customer on the LDRM-2 study team to be used as an initial point of reference.

<u>One human mission per year</u>: This is a programmatic assumption dictated in the LDRM-2 task statement. Mission rate has no impact on the analyses performed in this study.

<u>All mission elements placed in LEO (28.5 deg 407 km circular)</u>: L1 missions will require the mating of elements in Earth orbit prior to departure for L1. Launches into 28.5^{o} inclination orbits allow the maximum payload to orbit from the Eastern Test Range. Additionally, this inclination affords large planar launch windows required for rendezvous. The assembly altitude of 407 km is specified to minimize the effects of atmospheric drag on orbital lifetime while minimizing payload deployment altitude required on the launch vehicle upper stage. Future trades between the launch vehicle and orbital elements will be required to determine the optimum staging altitude.

<u>Consider the mission elements to be "cargo" in terms of delivery to the LEO parking orbit</u>: The launch vehicle will be responsible for delivering architecture elements to a 28.5^{o} 407 km circular orbit. This assumption puts the entire burden of cargo delivery on the launch vehicle, which helps to determine maximum launch vehicle capabilities. For this study, the propulsive capabilities of the mission elements will not be employed for orbit insertion, but will likely be required for orbit maintenance. Future trades can be performed to optimize the allocation of the orbit insertion function between the launch vehicle and orbital elements.

<u>Automated rendezvous and docking shall be used to assemble the elements</u>: Mission elements from the lunar missions will already be designed with automated rendezvous and docking (AR&D) capabilities, due to their functionality during those missions. Those same AR&D capabilities will be used during these L1 missions.

<u>Assume 2 weeks between launches</u>: This assumption is a balance between a desire to minimize total mission duration and vehicle lifetime while not severely impacting launch vehicle production, processing, and launch facilities for a four launch per mission baseline. A launch vehicle processing trade study will be required to determine the feasibility of meeting this assumption.

<u>Crew must be launched on a human-rated launch system</u>: This is dictated by the NASA human rating requirements document NPR 8705.2.

<u>The architecture will support 4 crew</u>: This assumption was dictated by NASA HQ and was used as the crew size for sizing the architecture.

The CEV is not required to be reusable and will not be explicitly designed for reusability: Previous spaceflight experience has taught that reusability should not be dictated a priori, rather the decision to reuse vs. build new should be made based on cost and schedule trades for a given flight rate and total program duration.

The CEV will provide the crew habitation function throughout the entire mission: The purpose of the "active mission" assessment is to determine the ability of the TRM elements to support the crew while operating out of the CEV. This will require the CEV to be in an active, operational state throughout the course of the mission.

The nominal Earth return for the CEV is direct entry with a water landing: Direct entry followed by a water landing is a proven and reliable way of returning crews from Earth-Moon transfer orbits. Additionally, some aborts during ascent from Earth may result in water landings. A water landing and recovery will therefore be a capability required by the vehicle. Providing a capability to nominally perform both water and land landings will increase CEV and architecture mass, and CEV complexity (deployment of airbags, firing of retrorockets, etc.).

The CEV design will incorporate functionality for land landing as a contingency for an ascent abort: Some ascent aborts may result in the CEV landing on land. The ability to meet this requirement will be for crew survival only – i.e. the vehicle may be damaged beyond repair/reuse as long as the crew survives.

Radiation shielding shall be incorporated into the design of the CEV crew module to provide a core level of biological protection for the crew during transit: Radiation shielding is required to meet crew safety requirements during solar particle events (SPEs). Short-term and cumulative crew dose limits for exploration missions have not yet been defined.

10.2 Architecture Description

10.2.1 Architecture Description/Operations for L1 CEV Active Mission

As with all the alternate missions in this study, the L1 CEV Active Mission's architecture was formulated around the use of the LDRM-2 Phase 2 elements. Unlike the LDRM-2 Phase 2 lunar mission, the Lunar Lander element is not needed in this architecture. After a quick assessment of the EDS's capabilities, it was determined that only one EDS would be needed in order to deliver the CEV to L1.

This architecture begins with the launch of an EDS. This is followed two weeks later by the launch of the CEV. The timing of the launch of the EDS is tied to the 24-hour Earth Orbit Departure (EOD) window, which will enable the CEV to depart Earth orbit for L1. Per the guidance of NASA HQ, each launch is separated be two weeks. Additionally, in order to protect the EOD window, which occurs on an average of once every ten days, the CEV will be launched such that it has time to perform rendezvous and docking, protect against weather delays, and allow for the checkout of the systems prior to the EOD burn. Therefore, the launch of the EDS was scheduled for 18.6 days prior to the beginning of the 24-hour EOD window.

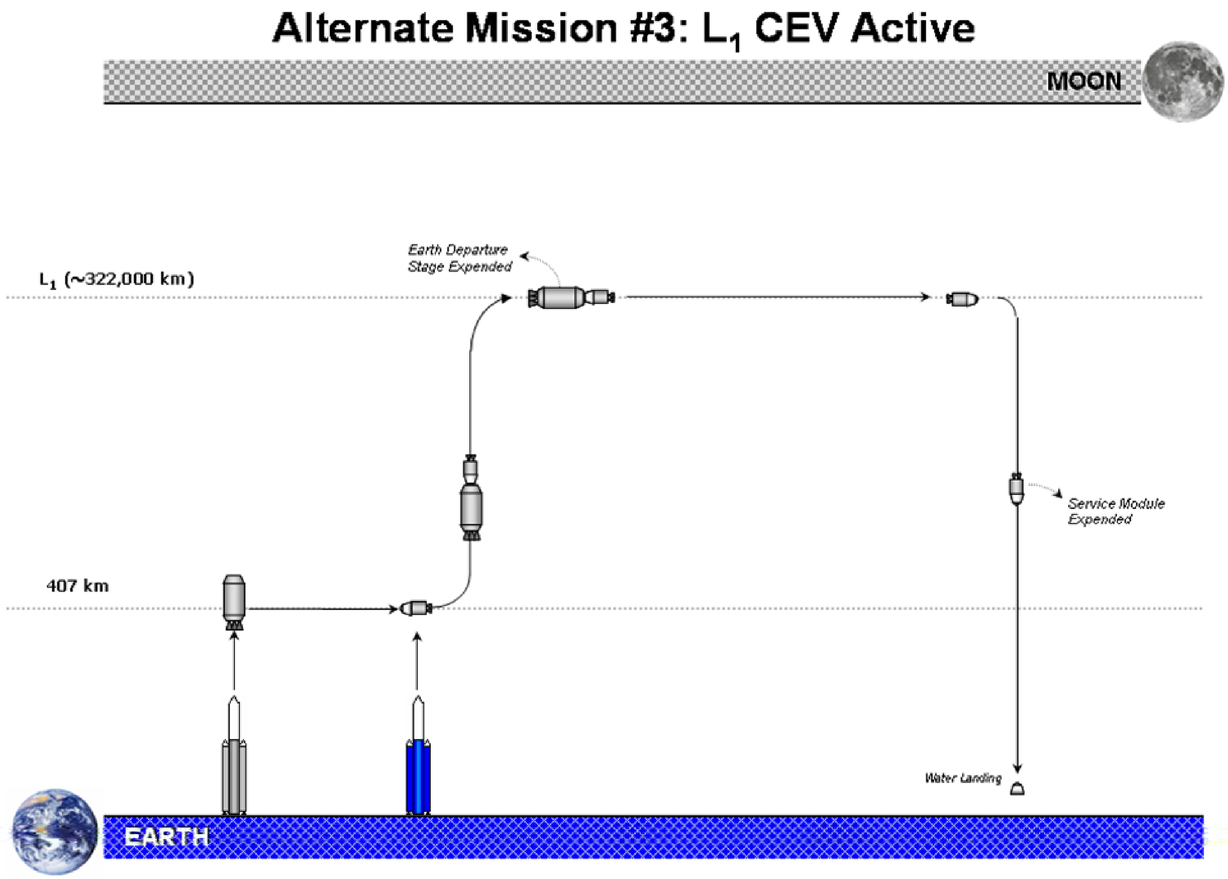

Figure 10.2.1-1: L1 CEV Active Architecture Illustration

Once the CEV and EDS are docked in LEO, the crew will have 12 to 36 hours (depending on any launch delays) to perform a checkout of their systems prior to the EOD burn. Once the EOD window opens, the crew will initiate the EOD burn and begin a 94 hour trip to L1. After the EDS has performed the EOD maneuver and any needed mid-course corrections, the CEV and EDS will separate. At this point, the EDS will dispose of itself into Earth's atmosphere and the CEV will continue on its journey to L1.

Upon arrival at L1, the crew will perform their mission. NASA HQ did not specify the objectives and tasks for this L1 mission. Therefore, no extra capabilities were designed into the CEV. Currently, the CEV would be fairly limited in its ability to perform a mission at L1 without the aid of extra resources or facilities. The calculated duration that the CEV could remain at L1 is 3.2 days. Additionally, the crew would be limited to the volume of the crew cabin, since no other elements were specified for this mission and EVA is only built into the CEV for contingency purposes. There is a possibility that the mission duration and the scope of the activities could be expanded through the use of a resource module, but this was outside the scope of the study.

Near the end of the 3.2-day L1 mission, the crew will time their L1 departure burn and adjust its magnitude in order to achieve the desired landing target on Earth. Once the burn has been initiated, the crew will coast on a trajectory towards Earth of approximately 94 hours. Three hours prior to entry interface, the Crew Module and Service Module will separate. The Service Module will be targeted for atmospheric re-entry where it will break up and land up-range of the Crew Module's targeted landing area. At this point, the Crew Module will re-orient itself for atmospheric re-entry and return the crew to the targeted landing area where they will be rescued and their spacecraft will be recovered.

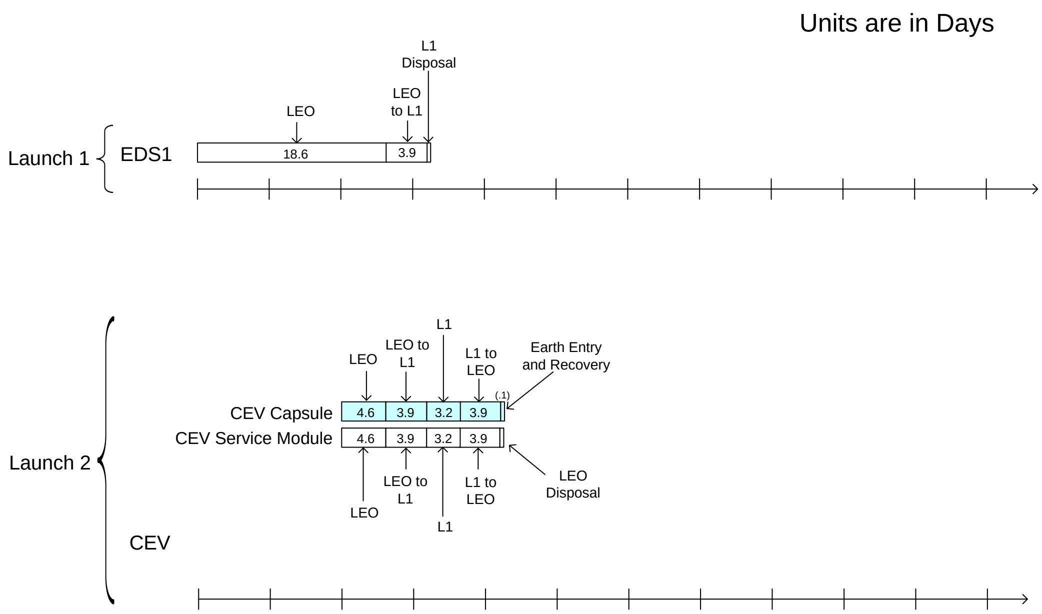

Figure 10.2.1-2: L1 CEV Active Architecture Nominal Timeline

10.2.2 Mission Abort Options

Nominal Flight Sequence

1. Launch From KSC
2. EOD (Earth Orbit Departure) and Booster Separation
3. L1 Arrival
4. L1 Operations
5. L1 to Earth Transfer
6. Earth Arrival and Re-entry to Touchdown

L1 Abort options

a. Launch Abort from KSC
b. LEO Deorbit
c. Powered L1 Transfer Abort, Early Return to Earth
d. L1 Swingby, Return to Earth
e. Depart L1, Shorten Transfer Time
f. Earth Arrival, Ballistic Reentry to Unplanned Water or Land Landing Site

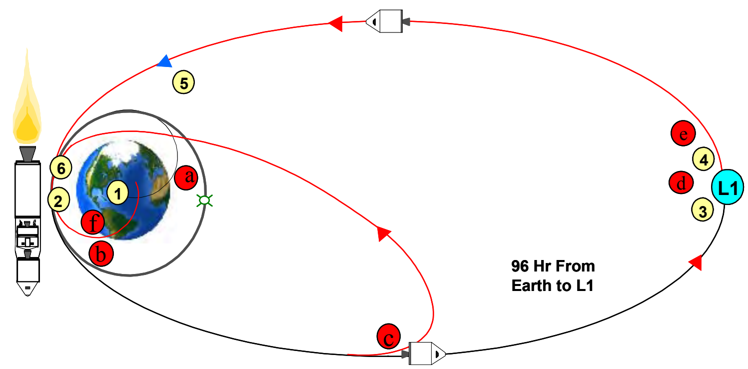

Figure 10.2.2-1: Mission Abort Options for L1 CEV Active Architecture

The mission abort options for this architecture are very similar to the Earth-L1-Earth segments described in the Lunar Architecture Focused Trade Study Final Report Volume 1 report in section 10.4.3. Therefore, the discussion of those applicable mission phases has been repeated (and adapted) for this section.

Aborts were developed and assessed for each mission phase from low Earth orbit to L1 and the return to Earth's surface. Figure 10.2.2-1 illustrates the nominal mission flight phases and their corresponding mission abort opportunities.

1. Launch and Ascent to Low Earth Orbit (LEO)

This mission phase begins with the launch from the surface of the Earth and ends after the vehicle is established in the desired LEO.

a. Booster or major CEV system failure

i. CEV emergency separates and returns to Earth

During the CEV launch and ascent to LEO should the Human-Rated Launch Vehicle (HRLV) or the CEV suffer catastrophic failure the CEV can initiate the Launch Abort System, triggering an emergency separation from the HRLV and return to Earth using the CEV descent and touchdown systems.

2. LEO Orbit and Rendezvous Operations

This mission phase begins after the vehicle is in LEO and ends after the completion of LEO rendezvous and mating of the Earth Departure Stage and the CEV.

a. CEV systems failure or failure to mate to the Earth Departure Stage (EDS)

i. CEV de-orbit and return to Earth

Once the CEV has reached LEO, should the CEV suffer a significant system failure prior to initiating the EDS L1 transfer burn, the CEV must perform a standard de-orbit maneuver, reenter and touchdown on land or water. If the abort takes place after the CEV mates to the EDS, the CEV must separate from the EDS prior to re-entry. If CEV propulsion system failures preclude performing a de-orbit maneuver, the EDS could be used for that maneuver. Otherwise the CEV is stranded in LEO and an Earth based CEV rescue mission is required to prevent a loss of crew (LOC) event from occurring. The CEV would need the appropriate resources to provide this safe haven for the crew until that rescue mission is performed (x-weeks).

3. LEO to L1 Transfer

This phase begins at the start of the (L1) transfer burn and ends just before the start of the L1 arrival burn.

a. Early EDS shutdown and high elliptical orbit

i. CEV maneuver to desired orbit

ii. De-orbit burn and re-entry to touchdown

Should the EDS fail to fully complete the L1 departure burn the CEV can separate, perform any required transfer orbit adjustments within the limits of available CEV propulsion constraints, establish a return to Earth trajectory and perform a de-orbit and re-entry to touchdown. After completion of the L1 transfer burn the CEV can also abort by eliminating the L1 arrival burn and returning to Earth on the elliptical transfer orbit. The CEV can adjust this orbit within CEV propulsion constraints to ensure a safe Earth re-entry and touchdown.

4. L1 Operations

This phase begins at the start of the L1 arrival burn and includes all L1 operations. This phase ends just prior to the CEV L1 departure burn.

a. No CEV L1 arrival burn by EDS

i. CEV swing-by at L1 and return to Earth

If the CEV's EDS can not perform the L1 arrival burn, the CEV can abort by continuing on the current elliptical transfer orbit and use the Service Module to perform any maneuvers necessary to establish a safe return-to-Earth trajectory for a direct re-entry.

b. Event worthy of shortening L1 mission

i. CEV times early L1 departure burn and/or magnitude of burn to reach appropriate Earth longitude for early Earth return.

5. L1 to Earth Transfer

This phase begins with the CEV L1 departure burn and ends just prior to Earth atmospheric re-entry.

a. No L1 burn by Service Module

i. Rescue mission staged from Earth

The CEV can support the crew for up to 4 days plus any margin. Depending on the capabilities of the Earth-based assets and the timing of the EOD opportunities, it may be possible to stage a rescue mission.

6. Earth Arrival and Re-entry to Touchdown

This phase begins with the direct re-entry into Earth atmosphere and ends with CEV touchdown on the Earth's surface.

a. Re-entry flight control failures

i. Ballistic re-entry (no lift vector control)

b. Entry targeting failures

i. Water or land touchdown

CEV equipped with appropriate crew survival and search and rescue gear for touchdown site

The CEV will be designed to support either land or water touchdown allowing for entry targeting failures to force the CEV to miss the desired landing site. The LDRM architecture is using 3 hours as the time required to find and recover the crew from the CEV after touchdown.

10.3 Mass Property Differences

The purpose of this study was to assess the ability of the LDRM-2 Phase 2 TRM elements to accomplish this mission. This study focused on understanding the deltas between what the Phase 2 TRM elements were sized for and what was needed in order to accomplish the objectives of this alternate mission. Therefore, none of the mission element designs were changed. Refer to sections 7.3.1 and 7.3.3 for the mass properties of the Phase 2 TRM EDS and CEV.

The right-hand graph in Figure 10.3-1 shows the differences in delta V that the CEV and EDS are required to deliver in the two different architectures. The Phase 2 TRM CEV is required to perform a maneuver of 159 m/s in order to rendezvous in LEO and then perform another maneuver of 1977 m/s in order to depart lunar orbit for Earth. The CEV for this alternate mission is required to perform a maneuver of 159 m/s in order to rendezvous in LEO, but performs a maneuver of only 798 m/s in order to depart L1 for Earth. This results in a total delta V of 2,136 m/s for the TRM CEV and 957 m/s for this alternate mission's CEV, as shown in Figure 10.3-1.

The Phase 2 TRM EDS1 is required to perform a maneuver of 1,489 m/s in order to perform its portion of the EOD burn. A mid-course correction of 50 m/s was also book kept. The EDS for this alternate mission required a total delta V of 4,058 m/s for the EOD burn and L1 arrival burn, combined. A mid-course correction of 10 m/s was also book kept. This results in a total delta V of 1,539 m/s for the TRM EDS1 and 4,068 m/s for this alternate mission's EDS, as shown in Figure 10.3-1.

Using these delta V's, the CEV and EDS masses were calculated. The left-hand graph in Figure 10.3-1 shows the mass delta between the Phase 2 TRM CEV and EDS elements versus what is needed in order to accomplish this alternate mission. The delta in the CEV's mass occurs due to propellants that were off-loaded in the Service Module. The TRM Service Module required a total of 11,518 kg of propellant, which led to a total IMLEO of 15,831 kg for this element. Due to the decreased propulsive requirements of this alternate mission, only 4,513 kg of propellant were needed by the Service Module, which led to a total IMLEO of 8,572 kg for this element. Therefore, the IMLEO of the CEV is able to be reduced by 7,259 kg for this alternate architecture.

The difference in the EDS masses occur for the same propellant off-loading reasons. Each TRM EDS required a total of approximately 51,000 kg, which led to a total IMLEO of approximately 60,000 kg. Although the delta V requirements for the EDS in this alternate mission are much greater than those of the TRM, the EDS's payload is much less in the alternate mission. Not only is the CEV mass much less in this alternate architecture, but the TRM EDS1 was required to perform a portion of the EOD burn while pushing EDS2 and the Lunar Lander in addition to the CEV. Therefore, due to the decreased payload and resultant decreased propulsive requirements of this alternate mission, only 38,109 kg of propellant were needed by the alternate mission's EDS, which led to a total IMLEO of 45,845 kg for this element. Therefore, the IMLEO of the EDS is able to be reduced by approximately 14,150 kg for this alternate architecture.

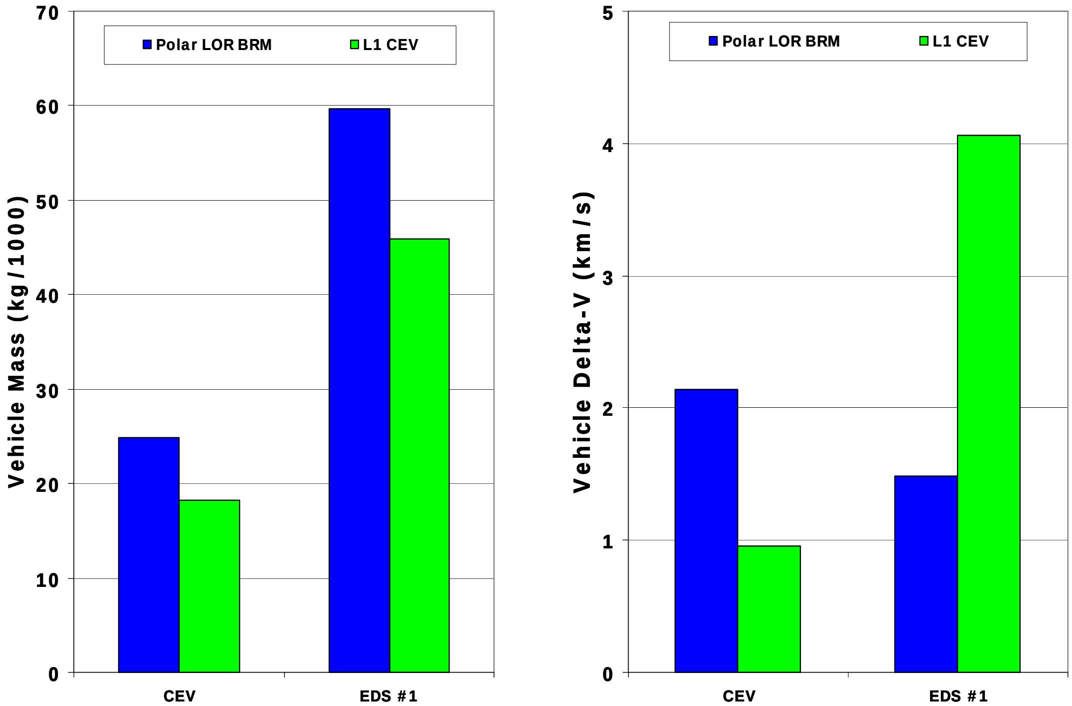

Figure 10.3-1: CEV/EDS Mass and Delta V Comparison Between the LDRM-2 Phase 2 TRM and L1 CEV Active Missions

10.4 Summary

The LDRM-2 Phase 2 TRM elements (CEV and EDS) easily encapsulate the capabilities required to perform the L1 CEV Active alternate mission. This alternate mission could be performed with the exact same CEV and EDS required for the TRM, with off-loaded propellants. Therefore, it would allow mission planners flexibility in carrying extra propellant (which could possibly be used for enhanced abort options or extra mission objectives) or it could allow the CEV and EDS to have lower IMLEOs (which may have cost or processing benefits).

The 3.2-day mission at L1 will be extremely limiting as far as the scope of tasks that can be undertaken during this mission. However, it would most likely be possible to extend the mission duration and possibly expand the functionality of the CEV through the use of a resource module tailored for the specific mission.

Each phase of this mission allowed the crew to either abort the mission and return to Earth, with one possible exception. Since both the TRM Service Module and EDS are oversized for this mission, the crew is assured adequate delta V in order to return from wherever they are during the course of the mission. The one exception to the crew's ability to return to Earth is in the case of a Service Module propulsion system malfunction at the time of L1 departure. In this situation, the crew's options would be extremely limited. If this were to happen, they would be forced to reside in the CEV while a rescue mission is staged. This is probably the portion of the mission during which this failure has the highest risk. In this situation, the crew could reside within the CEV for a period of up to nearly 4 days, plus any margin that was built into the vehicle. Depending on the capabilities of the Earth-based assets and the L1 EOD opportunities, it may be possible to launch an EDS and CEV in time to stage a rescue mission. However, since the nominal lifetime of the CEV is only 4 days at this point of the mission, it may be best to put resources into assuring that the Service Module's systems are highly reliable.

11.0 Alternate Mission Capture #4 – Mars Mission Staging

This study examined the ability of the Phase 2 LOR TRM elements to perform the operations associated with a mission in which the crew is delivered to a Mars transfer vehicle staged in High Earth Orbit (HEO).

11.1 Major Assumptions/Clarifications

This section outlines the major architecture assumptions made in formulation of this alternate mission. These assumptions were levied by the study's NASA HQ customer on the LDRM-2 study team to be used as an initial point of reference.

<u>All mission elements placed in LEO (28.5 deg 474 km circular)</u>: This is a slight variation from the rest of the architectures. After analysis of the altitudes in other missions, it was determined that a 474 km orbit was preferable because this altitude resulted in a phase repeating orbit for an inclination of 28.5 degrees. This guarantees the ability to perform Flight-Day-1 rendezvous between assets. Launches into 28.5^{o} inclination orbits allow the maximum payload to orbit from the Eastern Test Range. Future trades between the launch vehicle and orbital elements will be required to determine the optimum staging altitude, once launch vehicle sensitivities are known.

<u>Consider the mission elements to be "cargo" in terms of delivery to the LEO parking orbit</u>: The launch vehicle will be responsible for delivering architecture elements to a 28.5^{o}, 474 km circular orbit. This assumption puts the entire burden of cargo delivery on the launch vehicle, which helps to determine maximum launch vehicle capabilities. For this study, the propulsive capabilities of the mission elements will not be employed for orbit insertion, but will likely be required for orbit maintenance. Future trades can be performed to optimize the allocation of the orbit insertion function between the launch vehicle and orbital elements.

<u>Automated rendezvous and docking shall be used to assemble the elements</u>: Mission elements from the lunar missions will already be designed with automated rendezvous and docking (AR&D) capabilities, due to their functionality during those missions. Those same AR&D capabilities will be used during this mission.

<u>Assume 2 weeks between launches</u>: This assumption is a balance between a desire to minimize total mission duration and vehicle lifetime while not severely impacting launch vehicle production, processing, and launch facilities for a four launch per mission baseline. A launch vehicle processing trade study will be required to determine the feasibility of meeting this assumption.

<u>Crew must be launched on a human-rated launch system</u>: This is dictated by the NASA human rating requirements document NPR 8705.2.

<u>The architecture will support 6 crew</u>: This assumption was dictated by NASA HQ and was used as the crew size for sizing the architecture.

<u>The CEV is not required to be reusable and will not be explicitly designed for reusability</u>: Previous spaceflight experience has taught that reusability should not be dictated a priori, rather the

decision to reuse vs. build new should be made based on cost and schedule trades for a given flight rate and total program duration.

The CEV will provide the crew habitation function during the delivery of the crew to the Mars Transit Vehicle: The purpose of the "active mission" assessment is to determine the ability of the TRM elements to support the crew while operating out of the CEV. This will require the CEV to be in an active, operational state throughout the course of the mission.

The CEV design will incorporate functionality for land landing as a contingency for an ascent abort: Some ascent aborts may result in the CEV landing on land. The ability to meet this requirement will be for crew survival only – i.e. the vehicle may be damaged beyond repair/reuse as long as the crew survives.

Radiation shielding shall be incorporated into the design of the CEV crew module to provide a core level of biological protection for the crew during transit: Radiation shielding is required to meet crew safety requirements during solar particle events (SPEs). Short-term and cumulative crew dose limits for exploration missions have not yet been defined.

11.2 Architecture Description

11.2.1 Architecture Description/Operations for Mars Mission Staging

As with all the alternate missions in this study, the Mars Mission Staging architecture was formulated around the use of the LDRM-2 Phase 2 elements. This mission focused on delivering a crew of 6 to a Mars Transit Vehicle (MTV). NASA HQ did not state a mission objective beyond the delivery of the crew; therefore, no analysis was performed on the portion of the mission past this point.

This architecture begins with the launch of an EDS to a 28.5°, 474 km orbit. This is followed two weeks later by the launch of the CEV to the same orbit. The choice to use a 474 km orbit was a deviation from the other missions that were analyzed during the LDRM-2 study, all of which used a 407 km staging orbit. During the course of this study, it was found that a 474 km orbit offered certain benefits due to the fact that it is a phase repeating orbit (28.5° inclination remained constant). This characteristic allows the mission planners to have control over the position of the on-orbit assets with respect to the launch facilities at the time of the launch of the assets and provides a daily launch opportunity to rendezvous with the assets. It also guarantees a consistent ability to perform Flight-Day-1 rendezvous. Preliminary estimates show that this rendezvous procedure between the CEV and EDS could take place on the order of 10 hours versus the 50 hour rendezvous that was book kept for the other missions that were analyzed during the LDRM-2 study.

Once the CEV and EDS are docked in LEO, the crew will have 12 hours to perform a checkout of their systems prior to beginning their rendezvous with the MTV. For the purposes of this study, the LDRM-2 team had to assume a staging orbit for the Mars mission. This orbit would become the target orbit for the CEV to rendezvous with the MTV. Therefore, the results from a previous Mars mission study were used as a starting point. In 1999 the Aeroscience and Flight Mechanics Division at NASA/JSC performed a Mars mission analysis in which a 70,778 km x

800 km orbit was used as a Mars mission staging orbit for a MTV that used a high-power electric propulsion system. The 70,778 km x 800 km orbit was chosen since it was an approximately 24-hour, phase repeating orbit. As part of the LDRM-2 study, the Aeroscience and Flight Mechanics Division performed an analysis on the rendezvous procedure between the CEV (staged at an altitude of 474 km) and the MTV (staged at an altitude of 70,778 km x 800 km), both at an inclination of 28.5°. The estimates from this analysis showed that the rendezvous procedure would take on the order of 2 days once the CEV initiates the rendezvous sequence and that opportunities for rendezvous between the two vehicles arise on a daily basis.

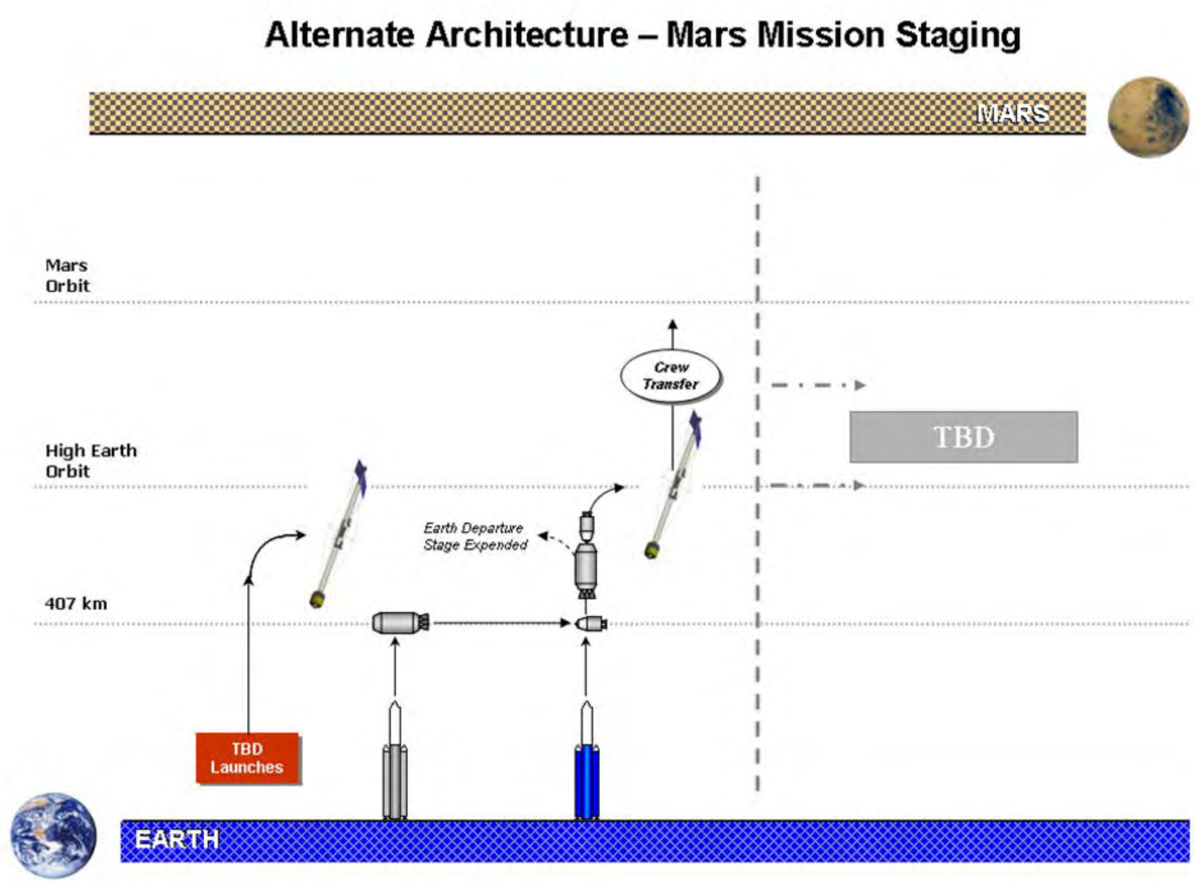

Figure 11.2.1-1: Mars Mission Staging Architecture Illustration

Once docked, the CEV has been sized to provide 24 hours for crew and cargo transfer. At this point there are two main options for the CEV, neither of which were analyzed because they fall out of the scope of this study. One option would be to de-orbit the CEV either to a targeted landing site from which it could be recovered or target it for atmospheric break-up. The second option would be to leave the CEV attached to the MTV and use it as a re-entry vehicle after the Mars mission is complete. Both of these options would carry certain requirements for the CEV, but the long on-orbit dormancy of the second option is likely to have the most impact to the design of the CEV.

Two contingencies worthy of note were planned into CEV's lifetime, which do not show up in the other architectures assessed during the LDRM-2 study. First, a 24-hour LEO loitering period was added to the CEV's lifetime in order to accommodate a missed rendezvous opportunity with the MTV. As stated previously, the CEV has daily opportunities for initiation of the rendezvous procedure. If the first one is missed, a second one has been afforded in the sizing. In reality, if the CEV was sized to accomplish the Phase 2 TRM, more days could be afforded. However, in order to assess the deltas between what was needed for this mission and what was required to accomplish the TRM, a quantity of missed opportunities had to be chosen. One missed opportunity was felt as adequate for the time-being, which allows a total of 36 hours to solve any on-orbit anomalies (12-hour checkout period plus 24-hour period in the event of a missed opportunity).

The second contingency that was planned into the lifetime of the CEV was a failed or aborted rendezvous with the MTV while the CEV is in the 70,778 km x 800 km orbit. In this situation, the CEV would perform a perigee-lowering maneuver at the apogee of the 70,778 km x 800 km orbit in order to perform re-entry. The maximum duration to return to Earth from the point in time marking the beginning of the rendezvous attempt through re-entry is 24 hours. Therefore, an extra 24-hour period has been added to the lifetime of the CEV.

Figure 11.2.1-2 shows the nominal timeline of this mission without the contingencies described above.

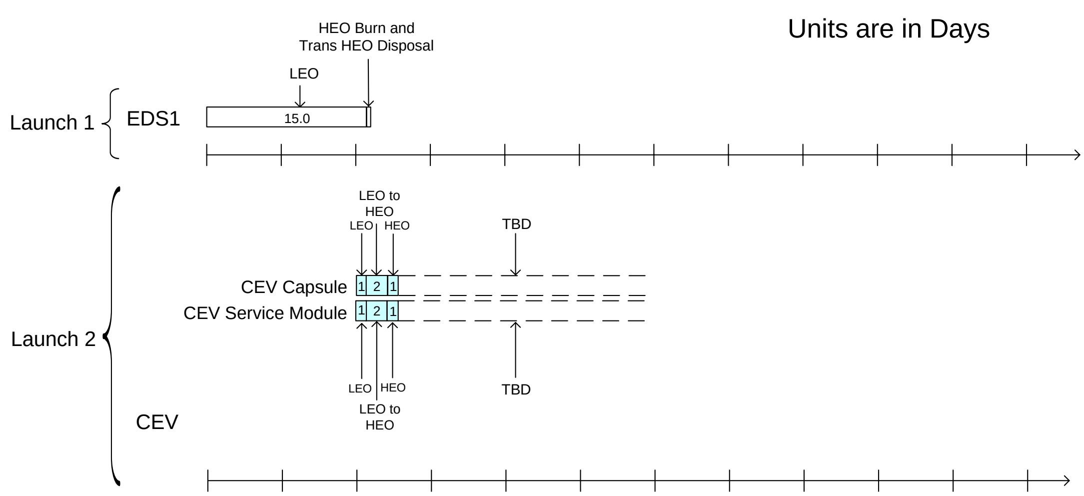

Figure 11.2.1-2: Mars Mission Staging Architecture Nominal Timeline

11.2.2 Mission Abort Options

Nominal Flight Sequence	
1.	Launch From KSC
2.	EOD (Earth Orbit Departure) and Booster Separation
3.	Terminal Phase Initiation (TPI)
4.	Terminal Phase Final (TPF)

L1 Abort options	
a.	Launch Abort from KSC
b.	LEO Deorbit
c.	MTV Rendezvous Abort, Early Return to Earth
d.	Failed or Aborted MTV Rendezvous, Early Earth Return
e.	Failed or Aborted MTV Rendezvous, Early Earth Return, Minimum Energy

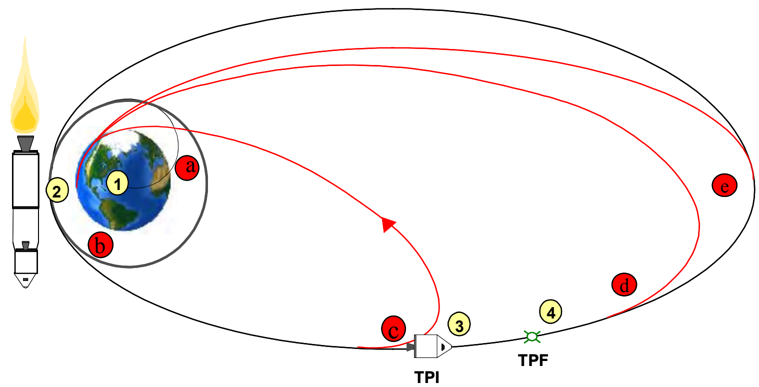

Figure 11.2.2-1: Mission Abort Options for Mars Mission Staging Architecture

Aborts were developed for each mission phase from low Earth orbit to rendezvous with the MTV. Figure 11.2.2-1 illustrates the nominal mission flight phases and their corresponding mission abort opportunities.

1. Launch and Ascent to Low Earth Orbit (LEO)

This mission phase begins with the launch from the surface of the Earth and ends after the vehicle is established in the desired LEO.

a. Booster or major CEV system failure

i. CEV emergency separates and returns to Earth

During the CEV launch and ascent to LEO should the Human-Rated Launch Vehicle (HRLV) or the CEV suffer catastrophic failure the CEV can initiate the Launch Abort System, triggering an emergency separation from the HRLV and return to Earth using the CEV descent and touchdown systems.

2. LEO Orbit and Rendezvous Operations with EDS

This mission phase begins after the vehicle is in LEO and ends after the completion of LEO rendezvous and mating of the Earth Departure Stage and the CEV.

a. CEV systems failure or failure to mate to the Earth Departure Stage (EDS)

i. CEV de-orbit and return to Earth

Once the CEV has reached LEO, should the CEV suffer a significant system failure prior to initiating the MTV rendezvous sequence, the CEV must perform a standard de-orbit maneuver, reenter and touchdown on land or water. If the abort takes place after the CEV mates to the EDS, the CEV must separate from the EDS prior to re-entry. If CEV propulsion system failures preclude performing a de-orbit maneuver, the EDS could be used for that maneuver. Otherwise the CEV is stranded in LEO and an Earth-based CEV rescue mission is required to prevent a loss of crew (LOC) event from occurring. The CEV would need the appropriate resources to provide this safe haven for the crew until that rescue mission is performed (x-weeks).

3. LEO to TPI

This phase begins at the start of the LEO departure burn and ends just before TPI.

a. Early EDS shutdown and high elliptical orbit

i. CEV maneuver to desired orbit

Should the EDS fail to fully complete the LEO departure burn, the Service Module could be used to maneuver to the desired orbit depending on whether the propellant tanks were filled with extra propellant (see sections 12.3 and 12.4 for a discussion on carrying extra propellant).

ii. De-orbit burn and re-entry to touchdown

Should the EDS fail to fully complete the LEO departure burn the CEV can separate, perform any required transfer orbit adjustments within the limits of available CEV propulsion constraints, establish a return to Earth trajectory and perform a de-orbit and re-entry to touchdown. After completion of the L1 transfer burn the CEV can also abort by eliminating the L1 arrival burn and returning to Earth on the elliptical transfer orbit. The CEV can adjust this orbit within CEV propulsion constraints to ensure a safe Earth re-entry and touchdown.

4. TPI to TPF

This phase begins at the start of TPI and ends with TPF.

a. Failed Service Module

i. Finish docking procedure with CEV RCS

ii. Remain on trajectory until perigee is reached (800 km) and perform CEV de-orbit maneuvers with CEV RCS

b. Failure in rendezvous and docking system

i. De-orbit burn and re-entry to touchdown

If a failure should occur that prohibits the docking of the CEV and MTV, the crew would be forced to return to Earth. Depending on the amount of propellants carried by the Service Module, they could choose to either expedite their return trip to Earth or they could wait until they reach apogee and perform the burn to lower their perigee to that of a re-entry profile (minimum energy maneuver).

11.3 Mass Property Differences

The purpose of this study was to assess the ability of the LDRM-2 Phase 2 TRM elements to accomplish this mission. This study focused on understanding the deltas between what the Phase 2 TRM elements were sized for and what was needed in order to accomplish the objectives of this alternate mission. Therefore, none of the mission element designs were changed. Refer to sections 7.3.1 and 7.3.3 for the mass properties of the Phase 2 TRM EDS and CEV.

The right-hand graph in Figure 11.3-1 shows the differences in delta V that the CEV and EDS are required to deliver in the two different architectures. The Phase 2 TRM CEV is required to perform a maneuver of 159 m/s in order to rendezvous in LEO and then perform another maneuver of 1977 m/s in order to depart lunar orbit for Earth. The CEV for this alternate mission is required to perform a maneuver of 159 m/s in order to rendezvous in LEO, but performs a maneuver of only 20 m/s in order to perform final rendezvous maneuvers with the MTV in HEO. An additional 49 m/s was also added to the CEV in order to return to Earth in the event of a failed docking. This results in a total delta V of 2,136 m/s for the TRM CEV and 228 m/s for this alternate mission's CEV, as shown in Figure 11.3-1.

The Phase 2 TRM EDS1 is required to perform a maneuver of 1,489 m/s in order to perform its portion of the EOD burn. A mid-course correction of 50 m/s was also book kept. The EDS for this alternate mission required a total delta V of 2,825 m/s for the LEO departure burn. A mid-course correction of 10 m/s was also book kept. This results in a total delta V of 1,539 m/s for the TRM EDS1 and 2,835 m/s for this alternate mission's EDS, as shown in Figure 11.3-1.

Using these delta V's, the CEV and EDS masses were calculated. The left-hand graph in Figure 11.3-1 shows the mass delta between the Phase 2 TRM CEV and EDS elements versus what is needed in order to accomplish this alternate mission. The delta in the CEV's mass occurs for two reasons. The reason for the reduction in mass is attributable to the amount of propellants that can be off-loaded in the Service Module. The TRM Service Module required a total of 11,518 kg of propellant, which led to a total IMLEO of 15,831 kg for this element and an IMLEO of 24,969 kg for the CEV (Service Module plus Crew Module). Due to the decreased propulsive requirements of this alternate mission, only 1,113 kg of propellant were needed by the Service Module. However, certain modifications were made to the CEV Crew Module to provide accommodations for 6 people for the Mars Staging mission as opposed to 4 people in the TRM. For example, the Atmosphere Contaminant Control system had to be scaled upwards in order to accommodate the higher rate of CO2 production. Additionally, two additional launch/entry suits, two additional seats, and additional crew accommodation/hygiene items were added to the Crew Module. With these modifications, the net IMLEO of the CEV was 14,374 kg. Therefore, the IMLEO of the CEV is able to be reduced by 10,595 kg for this alternate architecture.

The difference in the EDS masses occur for the same propellant off-loading reasons that were observed in the Service Module. Each TRM EDS required a total of approximately 51,000 kg, which led to a total IMLEO of approximately 60,000 kg. Although the delta V requirements for the EDS in this alternate mission are much greater than those of EDS1 in the TRM, the EDS's payload is much less in the alternate mission. Not only is the CEV's mass much less in this alternate architecture, but the TRM EDS1 was required to perform a portion of the EOD burn while pushing EDS2 and the Lunar Lander in addition to the CEV. Therefore, due to the decreased payload and resultant decreased propulsive requirements of this alternate mission, only 16,710 kg of propellant were needed by the alternate mission's EDS, which led to a total IMLEO of 21,354 kg for this element. Therefore, the IMLEO of the EDS is able to be reduced by approximately 38,646 kg for this alternate architecture.

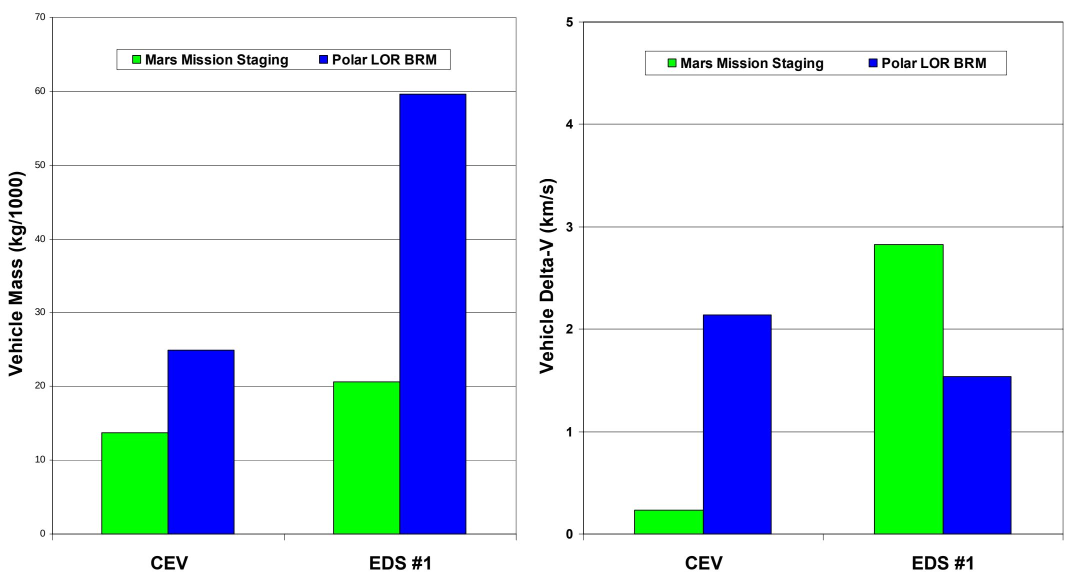

Figure 11.3-1: CEV/EDS Mass and Delta V Comparison Between the LDRM-2 Phase 2 TRM and Mars Mission Staging Missions

11.4 Summary

The LDRM-2 Phase 2 TRM elements (CEV and EDS) easily encapsulate the capabilities required to perform the Mars Mission Staging alternate mission. This alternate mission could be performed with the exact same Service Module and EDS required for the TRM, with off-loaded propellants. Therefore, it would allow mission planners flexibility in carrying extra propellant (which could possibly be used for enhanced abort options or extra mission objectives) or it could allow the CEV and EDS to have lower IMLEO's (which may have cost or processing benefits). However, the Crew Module would need to be outfitted for 6 people rather than 4 people for the TRM, as specified in the ground rules for this alternate mission. At this point, the Crew Module has not been analyzed with the amount of rigor needed in order to recommend structural changes due to the increased crew size, but the extra accommodations that were obvious (seats, launch/entry suits, personal hygiene/crew accommodations, life support systems, etc.) were scaled appropriately.

The mission of the CEV beyond the point of crew delivery was not defined, but will most likely carry requirements beyond what is needed in order to perform the crew delivery function. For example, if the CEV is to separate from the MTV and dispose of itself or return to Earth to be recovered, requirements for autonomous operations would be necessary. If the CEV is to remain attached throughout the duration of the Mars mission and then serve as the re-entry vehicle for the crew at the end of the mission, the systems would need to accommodate a long period of on-orbit dormancy.

Each phase of this mission allowed the crew to abort the mission and return to Earth. Since both the TRM Service Module and EDS are oversized for this mission, the crew is assured adequate delta V in order to return from wherever they are during the course of the mission. An additional benefit of this architecture is that the target orbit for the CEV is a highly elliptical HEO. This means that the CEV, regardless of its ability to dock with the MTV, will be on a trajectory that will bring it back in the vicinity of Earth (800 km). Therefore, for a small amount of delta V at apogee, the CEV can be placed on a re-entry trajectory that will bring them back to Earth.

12.0 Alternate Mission Capture #5 – Mars Return

Alternate mission #5 explores the ability of the polar LOR TRM elements to support the safe recovery of six crew at Earth for a Mars exploration mission. A primary objective for alternate mission #5 is the identification of CEV crew module technology impacts resulting from a long period of dormancy during the Earth-Mars transits and Mars surface exploration. Crew module design impacts associated with the high-speed Earth entry and the support of six crew over two days are also of interest.

Although alternate mission #5 is, in many ways, complementary to the Mars Mission Staging discussed in Section 11.0, it is primarily addressed as an independent mission in terms of spacecraft design and technology implications.

12.1 Major Assumptions/Clarifications

The Mars Return alternate mission affects the following polar LOR TRM assumptions from the LDRM-2 Phase 2 task statement. These assumptions were developed by NASA HQ as an initial point of reference for the lunar mission design.

<u>All mission elements placed in LEO (28.5 deg 474 km circular)</u>: The manner in which the Mars Return spacecraft is launched and mated with the Mars transit vehicle is not addressed in alternate mission #5. The Mars Return mission is loosely defined in terms of a dormant phase that begins when it is attached to the Mars transit vehicle and an active phase after it separates from the Mars transit vehicle to return the crew to Earth.

<u>Consider the mission elements to be “cargo” in terms of delivery to the LEO parking orbit</u>: The manner in which the Mars Return spacecraft is launched and mated with the Mars transit vehicle is not addressed in alternate mission #5.

<u>Automated rendezvous and docking shall be used to assemble the elements</u>: Mission elements from the lunar missions will already be designed with automated rendezvous and docking (AR&D) capabilities, due to their functionality during those missions. It is assumed that those same AR&D capabilities will be used during alternate mission #5.

<u>Assume 2 weeks between launches</u>: The manner in which the Mars Return spacecraft is launched and mated with the Mars transit vehicle is not addressed in alternate mission #5.

<u>Crew must be launched on a human-rated launch system</u>: The manner in which the Mars Return spacecraft is launched and mated with the Mars transit vehicle is not addressed in alternate mission #5.

<u>The architecture will support 6 crew</u>: This assumption was dictated by NASA HQ and was used as the crew size for alternate missions #4 and #5.

<u>The CEV is not required to be reusable and will not be explicitly designed for reusability</u>: Previous spaceflight experience has taught that reusability should not be dictated a priori, rather the decision to reuse versus build new should be made based on cost and schedule trades for a given

flight rate and total program duration. The potential for reusability of a CEV-type spacecraft following the completion of a Mars mission is unlikely to be a significant factor.

<u>The CEV will provide the crew habitation function during the return of the crew to Earth at the end of a Mars exploration mission</u>: The purpose of alternate mission #5 is to assess the ability of the CEV to support six crew for up to several days after separating from the Mars transit vehicle. This will require the CEV to be in an active, operational state throughout the course of the mission.

<u>The CEV design will be capable of being efficiently stored for long periods (800 or more days) prior to being used in an active mode</u>: The subsystem design impacts of the long period of dormancy during the Earth-Mars transits and the Mars surface activity are of primary interest.

<u>Radiation shielding shall be incorporated into the design of the CEV crew module to provide a core level of biological protection for the crew during transit</u>: The Mars transit vehicle is expected to provide radiation protection for the crew during the long duration Earth-Mars transits. The crew will only be exposed to ionizing radiation for approximately two days while occupying the CEV.

12.2 Architecture Description

This section describes the Mars Return alternate mission including its safety and mission success aspects, and potential mission abort options.

12.2.1 Mission Profile

The operational sequence for the Mars Return mission begins with a lengthy period of spacecraft dormancy followed by a short coast phase and a high-speed direct entry at Earth. As shown in Figure 12.2.1-1, the mission sequence for the Mars Return mission is quite simple because the launch and assembly phases for the Mars Return spacecraft are not defined. One possibility is that the same vehicle that delivers the crew to the Mars transit vehicle (e.g., Mars Mission Staging) remains attached to the Mars transit vehicle, and is used to return the crew to Earth at the end of the mission. One drawback to this approach is that the crew module systems would be activated for several days prior to being shut down for the lengthy Mars transit phase. Another drawback is that the CEV Service Module propulsion systems are not needed for the Mars Return phase and would increase the propellant required for the Earth-Mars transits. An alternate option is to use a pristine spacecraft that is maintained in a predominantly dormant state until the final stages of the Mars mission.

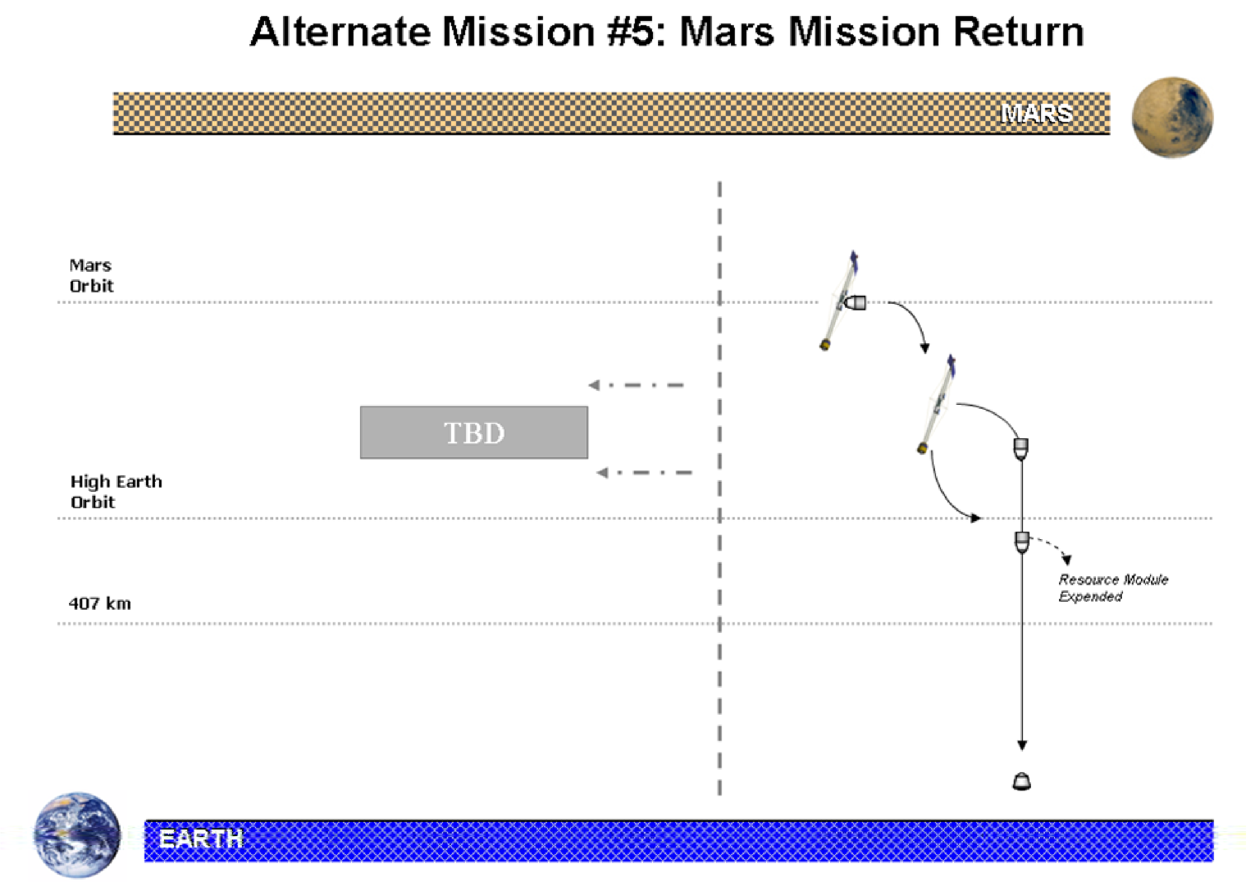

Figure 12.2.1-1: Mars Return Mission Sequence

12.2.2 Nominal Timeline

The nominal timeline for the Mars Return mission is provided in Figure 12.2.2-1. Following the 800-day dormant period defined in the mission assumptions, the spacecraft is activated and checked out by the crew prior to separation from the Mars transit vehicle. Lacking sufficient design detail to develop a good estimate, a spacecraft checkout interval of one day was assumed. If the resources necessary to support the attached spacecraft are supplied by the Mars transit vehicle, then the duration of the checkout interval may not be an important design issue. The active phase of the Mars Return mission starts with the separation of the spacecraft from the Mars transit vehicle. The two-day duration of the active phase is defined in the task statement definition for alternate mission #5. During this time the spacecraft must independently supply the necessary electrical power, life support and active thermal control resources for the six crewmembers. The nominal timeline also includes several hours at the end of the mission for atmospheric entry and crew recovery.

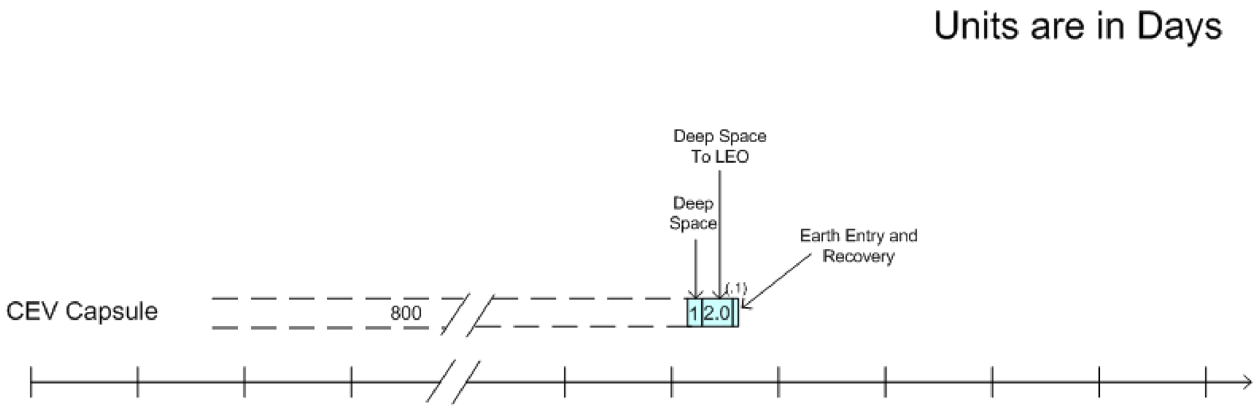

Figure 12.2.2-1: Nominal Timeline for Mars Return Mission

12.2.3 Safety and Mission Success

The list of crewed critical events for the Mars Return mission is very short due to the omission of the launch, assembly and transit phases:

- Spacecraft separation from the Mars transit vehicle
- Trajectory adjustments to accurately target the Earth entry interface point
- Separation of a Service Module or Resource Module, if applicable
- CM entry
- CM landing
- Crew recovery

The crew is completely dependent on the reliability and redundancy of the spacecraft systems from the time of separation from the Mars transit vehicle through Earth entry and landing.

12.2.4 Mission Aborts

The abort options in the event of a spacecraft failure or other factor that precludes the safe return of the crew to Earth are extremely limited. If the spacecraft is able to aerocapture into a low Earth orbit, but not re-enter and land, then an Earth-based rescue mission may be feasible. In the absence of atmospheric deceleration at Earth, the relative velocity of the spacecraft is likely to be in the range of 13 km/s – far too fast for an intercept mission. The only potential alternative in that case is to return to the Mars transit vehicle, assuming that it is not targeted for disposal at Earth, to utilize its power and life support resources. The prospects for crew recovery from the Mars transit vehicle are likely to be poor, however.

12.3 Spacecraft and Subsystem Design Impacts

The CEV subsystems defined for the polar LOR BRM are believed to be generally applicable to the Mars Return mission, including the 800-day dormant phase and the two to three days of active support for six crewmembers. As discussed for the Mars Staging Mission, the CEV crew cabin would need to be redesigned to support two additional crewmembers including seats, pressure suits and the associated emergency crew recovery equipment. Such a crew module redesign appears feasible considering the dimensions of the crew module defined for the lunar exploration mission and the significantly shorter duration of the Mars Return mission.

The Earth entry velocity of a Mars Return mission may exceed 13 km/s in comparison to the 11 km/s that is typical of a lunar return trajectory. The higher heating rates associated with a Mars Return mission are expected to impact the design of the crew module thermal protection system to some extent. Because ablative base heat shield materials were selected for the lunar missions, however, the design impact may be primarily an increase in ablator thickness rather than an entirely new design.

Since the propulsive capability of the CEV Service Module is not needed for the Mars Return mission, the packaging of the external subsystems needed to support the crew module must also be considered. The Service Module, as defined in the polar LOR BRM, supplies electrical power and life support gases to the crew module, and also provides the radiator loops for active thermal control. There are several possible approaches for deriving a Mars Return spacecraft using the flight elements defined for the lunar exploration mission:

1) Redesigned lunar Crew Module modified for 6 crewmembers and sized to incorporate the necessary Service Module equipment
2) Modified lunar Crew Module for 6 crewmembers, supplemented with a resource module or modified Service Module
3) Modified lunar Crew Module for 6 crewmembers with a standard Service Module

The second option appears to offer a mass and cost efficient approach for developing the Mars Return spacecraft. The resource module could utilize the same structural, fluid and electrical interfaces as the Service Module without the mass penalty of the Service Module propulsion systems. Introducing a resource module to the architecture would increase the total number of elements, but would offer the opportunity to have an element streamlined to accomplish only the functions that are needed for this unique mission. An alternative to introducing a resource module would be to modify the Service Module. The modified Service Module would have the benefit of using the existing LOR TRM Service Module hardware and configuration, minus the propulsion system hardware.

12.4 Summary

The CEV crew module defined for the lunar missions appears to offer reasonable potential to support a Mars Return mission for six crew. The subsystem technologies selected for the lunar exploration missions are viable for both the long period of dormancy during the Earth-Mars transits and Mars surface exploration as well as the short active phase following separation from the

Mars transit vehicle. Crew module thermal protection system impacts due to the high Earth entry velocities associated with a Mars return are expected to be modest since an ablative base heat shield is already employed on the lunar CEV capsule. The primary spacecraft design issues associated with the Mars Return mission are the revisions to the internal layout of the crew module to support six crewmembers, and the repackaging of the power generation, active thermal control and life support resources that are supplied by the Service Module for the lunar exploration missions.

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