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Crew Exploration Vehicle Destination for Human Lunar Exploration: The Lunar Surface

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As the first major element in Project Constellation, the Crew Exploration Vehicle (CEV) will serve as the cornerstone for the Nation's Vision for Space Exploration. As such, it is vitally important that it be developed so as to support sustainable and affordable exploration over the coming decades. The CEV will replace the Space Shuttle in ferrying crews to and from the space station and will enable human missions to the Moon and Mars. For lunar exploration, the CEV will, at minimum, be used to launch the crew and transport them to and from the lunar vicinity, and will be used as an Earth entry capsule at the conclusion of the mission. The Lunar Orbit Rendezvous (LOR) operational architecture chosen during the Apollo program offered significant mass benefits for the requirements and technology of the time. With present day technology and programmatic goals, however, our analysis indicates architectures in which the CEV travels to the lunar surface are mass competitive with Apollo-style LOR architectures. The additional architecture-classes of interest include Direct Return (DR) architectures, in which the CEV travels to the lunar surface and then returns directly to earth without rendezvous, and propulsion lunar orbit rendezvous architectures, in which the CEV transfers the crew to the lunar surface, and then performs a rendezvous with a propulsion stage in lunar orbit prior to returning the crew to Earth. Initial Mass in Low Earth Orbit (IMLEO) is typically used as a top-level screening criterion in architecture selection; however, as the IMLEO for these architectures is similar, additional factors must be taken into account in determining the preferred architecture. The two architectures with the CEV to lunar surface reduce development and operational costs by eliminating the need for the second crew compartment used for lunar surface access in LOR architectures. Direct Return architectures further reduce cost by eliminating one propulsion stage design and by enabling the use of a common lander for both human missions and the emplacement of large surface assets, by making use of the crew lander's increased payload capability. Our analysis also indicates that architectures with the CEV traveling to the lunar surface provide mission risk and crew safety benefits as the return crew compartment is accessible on the lunar surface for inspection and potential maintenance, which may be of particular importance during the long-duration missions currently envisioned. Direct Return architectures provide the additional benefits of having all assets accessible by the crew while on the surface and of not requiring a rendezvous on the critical path home. Due to the significant benefits provided by architectures in which the CEV travels to the lunar surface, we recommend developing the CEV with lunar surface capabilities. Our analysis indicates that the CEV can be developed in such a way as to limit the impact these additional capabilities would have on an initial version for ISS, while ultimately enabling the sustainable exploration of the Moon and Mars.

I. Introduction

THE Vision for Space Exploration calls for the sustainable and affordable exploration of the Moon, Mars, and beyond[1]. The first major element to be developed as part of the Vision is the Crew Exploration Vehicle, or CEV. The CEV will initially be used to replace the Space Shuttle in providing crew and perhaps also cargo transportation to the International Space Station, before serving as the primary crew launch and recovery system for

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lunar exploration. For lunar exploration, the CEV will be used in concert with other exploration elements of the overall architecture for transporting crew to and from the lunar surface. In order to properly develop the CEV such that it is well integrated with the other lunar architecture elements, it is essential that lunar architecture options be evaluated and their impacts upon the CEV functional allocation and requirements be assessed.

As part of a Draper Laboratory-led NASA Concept Exploration & Refinement study, MIT performed a comprehensive analysis of over 1000 operational architectures for both Moon and Mars missions in support of the Vision for Space Exploration. This analysis identified three major classes of lunar operational architectures of interest. This paper describes and compares these three architecture classes and draws conclusions regarding recommended architectures which the CEV should be developed to support.

II. Operational Architecture Analysis and Resulting Architectures

In order to comprehensively evaluate Moon and Mars transportation architectures, we have made use of an architecture generation tool developed in the Object Process Network (OPN) graphical programming language, to identify 1162 architectures for enabling missions to either Moon or Mars[2]. Each of these architectures is evaluated using parametric models for a set of technology options and screened using a series of proximate metrics related to cost and risk[3]. From this process, a series of candidate architectures emerges for further consideration. We employ the "Mars-back" approach of first looking at the requirements for Mars exploration systems and then projecting their capabilities towards lunar exploration systems. The goal is to develop a common set of systems to enable affordable and sustainable exploration of both locations. Using this approach, we are able to minimize the overall cost of the program while accelerating the timeline to initial Mars missions and allowing lunar missions to continue once Mars missions have commenced[4,5].

From this process, we have identified three lunar architectures that appear to be of particular interest as potential candidates for the lunar crew transportation system for short-duration lunar missions. The three architectures are described below. In terms of destination, two of them involve the CEV descending to the lunar surface and one has the CEV remain in a staging location in the lunar vicinity. The index number associated with the architectures are reference numbers to the architectures generated using the OPN tool.

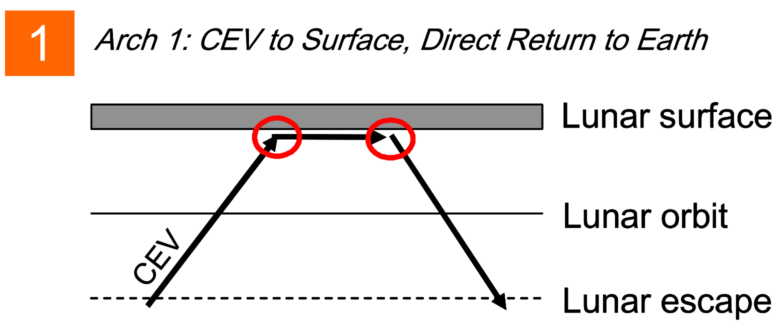


Figure 1. Lunar Direct Return Architecture.

1. Lunar Direct Return Architecture (Arch 1)

The first lunar architecture candidate also is the first architecture generated by the OPN tool due to its overall simplicity. In this architecture, the crew transfers to the surface of the moon in the CEV, makes use of the CEV for surface operations (potentially including a transfer to a surface habitat for long duration missions), and then returns directly to the Earth from the surface of the Moon in the CEV. Propulsion stages are used in series to accomplish the required maneuvers and remain with the CEV until expended.

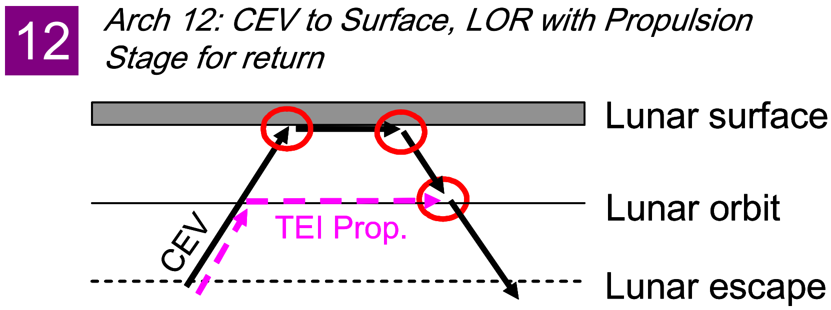


Figure 2. Architecture 12, Propulsion Lunar Orbit Rendezvous with Lunar Surface CEV.

2. *Propulsion Lunar Orbit Rendezvous with Lunar Surface CEV (Arch 12)*

The second architecture of note also has the CEV proceeding to the surface of the Moon. However, in this case the propulsive capability for Trans-Earth Injection (TEI) remains in lunar orbit as opposed to traveling to the surface. As in architecture 1, the CEV supports the crew while on the surface, possibly including transfer to a long duration surface habitat. After ascending from the surface, the CEV performs a rendezvous in lunar orbit with the TEI propulsion and then returns to Earth.

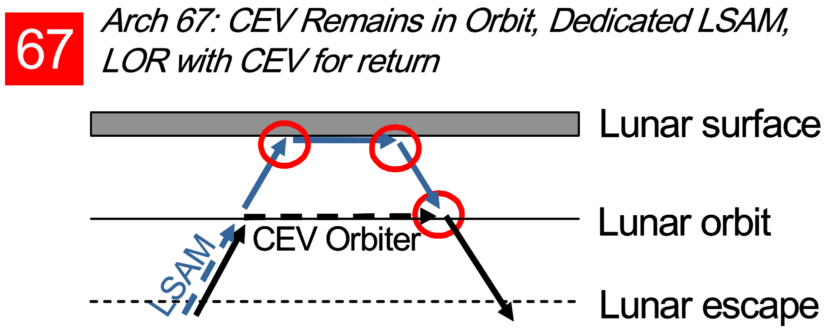


Figure 3. Architecture 67, Lunar Orbit Rendezvous with Separate LSAM.

3. *Arch 67 – LOR with separate Lunar Surface Access Module (Arch 67)*

The third architecture of interest is the classic Apollo-LOR architecture as used successfully during the late 1960s and early 1970s. In this architecture, the CEV proceeds only as far as lunar orbit. At that point the crew transfers to a separate Lunar Surface Access Module, descends to the lunar surface, and is supported by the LSAM on the surface for short periods of time or until the crew transfers to a long-duration surface habitat. Upon concluding the surface mission, the crew ascends to lunar orbit in the LSAM, rendezvous with and transfers to the CEV, and subsequently returns to Earth.

In examining these three architectures, it is interesting to note that there are two primary options which can be used to distinguish among them. These are whether the architecture has the CEV go to the surface or a separate LSAM is used and whether lunar orbit rendezvous is used or a direct return to earth is performed (Figure 4). The architecture grouping according to these two parameters is included in the following table and is provided for reference on each page of this report. It is interesting to note that while CEV to the surface is traditionally associated with the direct return architecture (Arch 1), CEV to the surface is also compatible with orbital rendezvous as in architecture 12.

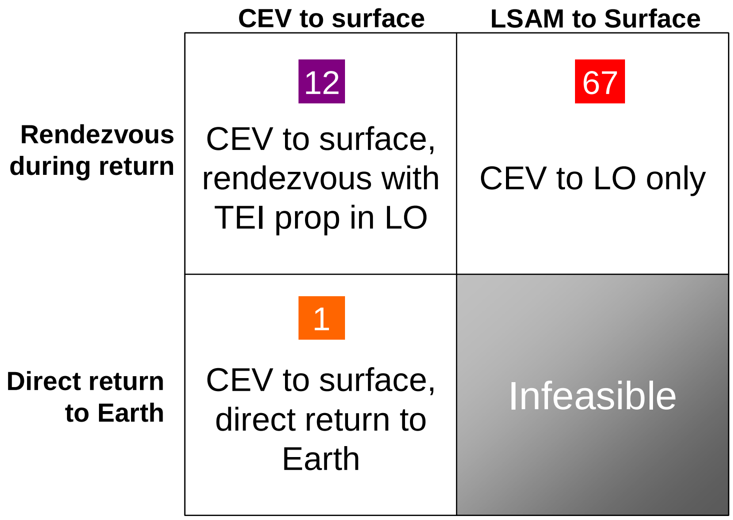


Figure 4. High-level lunar crew transportation architecture, CEV destination space.

Of these three architectures, Arch 67 has been frequently suggested for use in a variety of studies investigating lunar exploration architectures. Arch 1 has also been proposed, including as the original Apollo lunar architecture concept (either in single-launch NOVA mode or multi-launch Earth orbit rendezvous (EOR) mode) and was investigated in depth again as part of NASA's First Lunar Outpost study conducted in the early 1990s[6-8]. While some studies, such as NASA Langley's Broad Trade Study[9], have touched on Arch 12, it does not appear to have been previously studied in-depth.

III. Architecture Mass Comparison

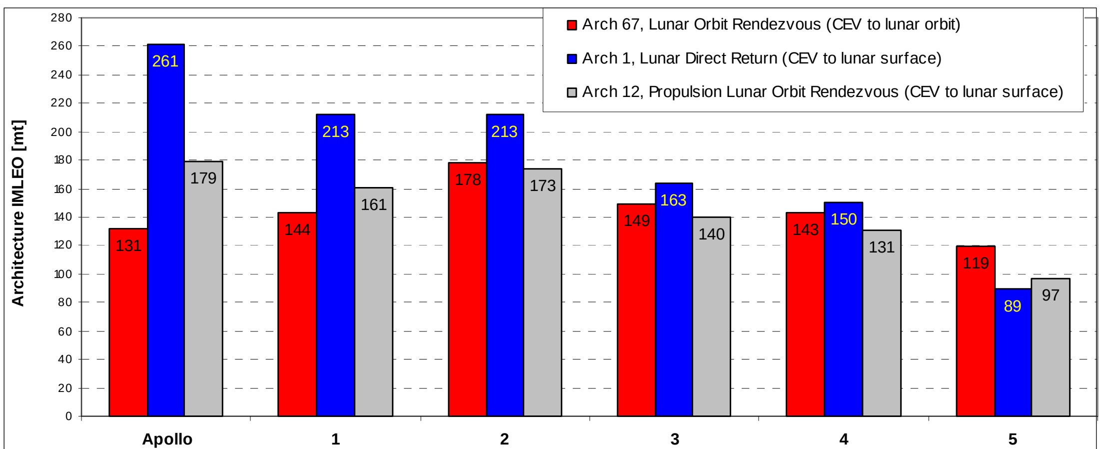
As the cumulative launch mass or Initial Mass in Low Earth Orbit (IMLEO) is a major driver for the overall cost of a given architecture, it is frequently used as a metric in architecture selection. A common belief exists that architectures in which the CEV goes to the lunar surface, in particular direct return architectures, incur a substantial mass penalty when compared to Lunar Orbit Rendezvous architectures. In order to determine the validity of this belief, the masses of the architectures of interest are presented for a series of mission requirements and technology assumptions.

In determining the IMLEO of human space exploration architectures, the mass of the crew compartments involved are of utmost importance as they are the primary payload and thus size the propulsion stages in use. For the purposes of this analysis, the CEV and LSAM were each modeled as single crew compartments with a fixed mass to meet their mission duration requirements. More detailed analysis strategies to reduce the mass can be employed such as by providing consumables storage and power provision equipment on elements which are jettisoned once they are no longer necessary. Modeling the crew compartments as fixed masses, however, will allow for a conservative mass estimate to be made consistent with the current level of analysis. The crew compartment masses are included in Table 1. They are based upon the Draper/MIT Concept Exploration and Refinement study CEV design and CEV and LSAM designs contained within the NASA Johnson Space Center Focused Trade Study Report[10]. A surface mission duration of 7 days and a crew size of 4 were assumed, which would enable both short-duration lunar scouting expeditions and long-duration stays at a pre-deployed lunar surface habitat. CEVs with lunar surface capability were modeled as incurring a 10% overhead relative to lunar-orbit-only CEVs in order to accommodate the design modifications required for surface missions and the additional consumables for the increase in crewed duration. The crew compartment also includes representative fixed masses for the Apollo spacecraft Command Module and Lunar Module Ascent Stage crew compartments, taking into consideration appropriate masses from the Service Module and Descent Stage for life support, thermal control, and power subsystems required to support the crew[11,12].

Table 1. Crew Compartment Masses Used in Architecture Analysis.

Crew Compartment	Mass [kg]
Apollo LM Habitat	2,700
Apollo CM	7,000
LSAM Habitat	6,800
Lunar Orbit CEV	9,150
Lunar Surface CEV	10,050

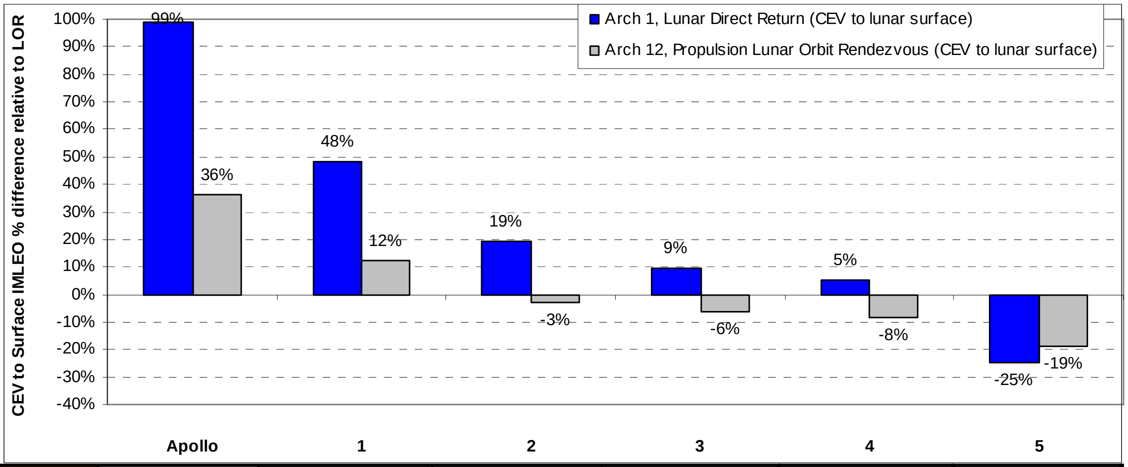
Using the parameters described in the Appendix, propulsion stages were sized based upon these crew compartments to determine the overall mass of the three architectures through a series of mission and technology options, the results of which are included in Figure 5. The baseline was an Apollo case utilizing the crew compartments, delta-v's, structural factors, and specific impulse (Isp) of the actual Apollo system applied to the three operational architectures under consideration (left-most case). The 1st variation from the baseline saw the introduction of the modern crew compartment masses, present day propulsion technology with the same propellant combinations as used during Apollo, and a modified operational sequence in which the high specific impulse (Isp) Earth Departure Stage (EDS) performs Lunar Orbit Insertion (LOI) of the stack prior to descent. The 2nd variation further modified the mission objectives by moving away from the short duration, near equatorial limits of the Apollo system to provide long duration, global access anywhere on the lunar surface. In order to maintain an anytime Earth return option, this required introducing a plane change capability into the orbital assets of architectures 12 and 67, using the plane change strategy described in the Appendix. (The Direct Return architecture inherently has global access with anytime return capability, so no modifications were required relative to the previous case.) For the 3rd variation, higher Isp methane-oxygen propellants were introduced for ascent, descent, and trans-Earth injection (TEI) replacing the hypergolic propellants used in the previous cases. The 362 s Isp of methane-oxygen is representative of pressure-fed specific impulse for this propellant combination; higher Isp is possible, particularly with pump-fed systems, as such this represents a conservative estimate of the benefit of introducing methane-oxygen propulsion. The 4th variation uses hydrogen-oxygen for descent propulsion. Methane-oxygen is maintained for ascent and TEI due to boil-off concerns related to the long term storage of hydrogen in a space environment. The 5th variation introduces the provision of in-situ produced lunar oxygen loaded into the ascent stage on the lunar surface. In the case of the Direct Return architecture the ascent stage performs both ascent and TEI, so the lunar oxygen is provided for both maneuvers.



Crew size	3 crew, 2 to surface	4 crew, 4 to surface	4 crew, 4 to surface	4 crew, 4 to surface	4 crew, 4 to surface	4 crew, 4 to surface
Propellant Type	H2/O2 430 s for EDS; Hypergolics 311s for Asc/Dsc, 314 s elsewhere	H2/O2 460 s for EDS; Hypergolics 316 s elsewhere	H2/O2 460 s for EDS; Hypergolics 316 s elsewhere	H2/O2 460 s for EDS; CH4/O2 362 s elsewhere	H2/O2 460 s for EDS; H2/O2 430 s for Descent; CH4/O2 362 s elsewhere	H2/O2 460 s for EDS; H2/O2 430 s for Descent; CH4/O2 362 s elsewhere
Operation Sequence	LOI with SM / descent stage	LOI with EDS	LOI with EDS	LOI with EDS	LOI with EDS	LOI with EDS, ISRU oxygen for ascent
Location	'Near equatorial'	'Near equatorial'	Global access	Global access	Global access	Global access

Figure 5. Lunar architecture IMLEO results across a series of mission and technology options.

It is interesting to note that in the Apollo case, the Lunar Orbit Rendezvous architecture, which was ultimately chosen, did indeed provide significant mass benefits when compared to the direct return architectures also under consideration at the time[6,7]. This is likely a large contributor to the belief that direct return architectures are unacceptably massive, although it becomes clear that with modern propulsion technology and current day mission objectives, this is no longer the case. Once methane-oxygen propulsion has been introduced, and particularly so with a hydrogen-oxygen lander, the IMLEO of the Lunar Direct Return architecture is comparable to that of a standard Lunar Orbit Rendezvous architecture. The Propulsion Lunar Orbit Rendezvous (Arch 12), in which the CEV travels to the lunar surface and performs a rendezvous with the Trans-Earth Injection propulsion prior to returning to Earth, also appears to offer a slight mass benefit relative to the LOR and LDR architectures once cryogenic ascent, descent, and TEI propulsion is introduced. In the case of In-Situ Resource Utilization (ISRU) for the provision of oxygen, a significant reduction IMLEO can be seen, particularly for the Direct Return architecture. It should be noted that the masses presented for ISRU make no account for the oxygen production equipment or potential feedstock required for the processing – the addition of these to an actual campaign would offset some of the benefit of ISRU relative to non-ISRU missions. As can be seen in Figure 6, while the percent difference between the architectures with lunar surface CEVs are comparable to a standard LOR architecture in the case of cryogenic propellants, once ISRU is introduced both of the CEV to surface architectures are significantly lower than LOR.



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Propellant Type	H2/O2 430 s for EDS; Hypergolics 311s for Asc/Dsc, 314 s elsewhere	H2/O2 460 s for EDS; Hypergolics 316 s elsewhere	H2/O2 460 s for EDS; Hypergolics 316 s elsewhere	H2/O2 460 s for EDS; CH4/O2 362 s elsewhere	H2/O2 460 s for EDS; H2/O2 430 s for Descent; CH4/O2 362 s elsewhere	H2/O2 460 s for EDS; H2/O2 430 s for Descent; CH4/O2 362 s elsewhere
Operation Sequence	LOI with SM / descent stage	LOI with EDS	LOI with EDS	LOI with EDS	LOI with EDS	LOI with EDS, ISRU oxygen for ascent
Location	'Near equatorial'	'Near equatorial'	Global access	Global access	Global access	Global access

Figure 6. Percent IMLEO difference of CEV to Surface architectures relative to Lunar Orbit Rendezvous with CEV in Orbit across a series of mission and technology options.

While ISRU provides significant benefit, it appears unlikely that this technology will be used on initial lunar missions. As such, we believe that the 3rd or 4th variations, meaning either methane-oxygen for descent, ascent, and TEI or hydrogen-oxygen for descent and methane-oxygen for ascent and TEI, are the most likely candidates for initial missions. ISRU may be introduced at a later date once the processes have been adequately demonstrated and a production facility emplaced. Doing so would either allow for reductions in the IMLEO of the crew transportation system or, perhaps more likely, the inclusion of significantly increased cargo capacity per flight. Focusing then on the cryogenic propellant cases without ISRU, the close proximity of the initial mass in low Earth orbit for each of the three architecture-classes indicates that other metrics should be the deciding factors in the selection of a lunar crew transportation architecture. As such, the remainder of this paper focuses on such factors, including mission risk, crew safety, and overall development and operational cost.

IV. Mission Risk and Crew Safety

The Direct Return, Propulsion Lunar Orbit Rendezvous, and standard Lunar Orbit Rendezvous architectures have notable differences regarding their operational sequences and the number and type of mission critical events and crew safety hazards. These translate to differences in risk and crew safety. Each of the following set of distinguishing features is analyzed further below:

- Crew safety for an Apollo 13-style emergency
- Rendezvous in lunar orbit
- Docking in lunar orbit
- Hardware accessibility in the lunar vicinity

A. Crew Safety for an Apollo 13-style Emergency

During the Apollo 13 mission, an explosion in the Service Module caused the power, life support, and propulsion subsystems of the Command and Service Module (CSM) to fail and rendered the CSM effectively useless beyond providing reentry capability. The crew survived by transferring to the lunar module (LM) and using its systems in a "lifeboat" fashion.

At first glance, architectures with two separate crew compartments (e.g., Arch 67) appear to provide higher crew safety because of the lifeboat option. This perception is, however, only partially true, because it is redundancy in the power, life support, and propulsion subsystems that saves the crew in an Apollo 13-style emergency, not specifically the second habitable volume.

The availability of a second pressurized volume could be beneficial in the case of a micrometeoroid or other debris strike. However, if the crew compartment with the heat shield is impacted, the second crew compartment will only provide a benefit if it is also equipped with a heat shield. Also, technologies are available for micrometeoroid and debris protection which can reduce the probable severity of an impact to acceptable levels.

A degree of subsystem redundancy comparable to Arch 67 could be achieved for Arch 1 and 12 by adding backup subsystem hardware for critical systems, with due consideration of the geometrical arrangement of the subsystems. A reference for this approach could be the modifications carried out for the Apollo 14 Service Module to achieve power and ECLSS subsystem redundancy. For lunar missions, the second pressurized volume therefore does not appear to substantially enhance crew safety. It is also important to note that in Arch 67, two pressurized volumes are available for only a relatively short portion of the mission.

B. Rendezvous in Lunar Orbit

Rendezvous in lunar orbit is essential for architectures 12 and 67. It is a source of risk because only a limited amount of lifetime and propellant is available to achieve the rendezvous. Also, due to the need to bring the entire crew to the lunar surface in long duration missions, only one of the two vehicles in the rendezvous is manned, as opposed to both in Apollo. As architecture 1 does not require any rendezvous in lunar orbit, it is clearly superior to Arch 12 and 67 regarding this factor.

C. Docking in Lunar Orbit

For nominal mission operations both Arch 12 and 67 require docking in lunar orbit in addition to rendezvous; Arch 1 requires neither. For Arch 67, however, an emergency transit in space suits from the Lander crew compartment to the orbiting CEV could be carried out if rendezvous can be achieved but docking is unavailable. This procedure was planned for Apollo contingencies and was partially tested during the Apollo 9 mission. For Arch 12, however, a stable structural connection between the CEV returning from the surface and the orbiting TEI propulsion stage has to be achieved to be able to successfully perform the TEI burn: a successful docking in lunar orbit thus is on the critical path for return to Earth. Arch 12 is therefore inferior to 67, which is in turn inferior to 1 regarding docking risk. This risk could potentially be mitigated through the provision of redundant docking adapters on the CEV and TEI stage in architecture 12.

D. Hardware Accessibility in the Lunar Vicinity

In Arch 12 and 67, hardware assets are left at a staging location in the lunar vicinity, either lunar orbit or a libration point (L1 / L2). These assets have to be controlled and maintained within operating parameters either by remote control from Earth or autonomously for periods several months (long-stay lunar surface missions). This adds considerable risk to architectures which require staging in the lunar vicinity. Such elements must also provide the delta-v required for station keeping in orbit, which is particularly high for assets in halo orbits around L1.

In Arch 1, however, all hardware is transported to the lunar surface, where it can be inspected and potentially repaired, if required. This hardware accessibility is also the reason for increased benefit of in-situ propellant production for Arch 1: propellant for both the ascent and TEI could be produced on the lunar surface. In summary, Arch 1 is superior to Arch 12 and 67 in terms of hardware accessibility.

E. Risk and Crew Safety Summary

A preliminary assessment of major mission risk was performed based on the four factors of risk and crew safety: an Apollo 13-style emergency, rendezvous and docking in the lunar vicinity, and hardware accessibility. If adequate power, ECLSS and propulsion subsystem redundancy is provided, Arch 1 dominates Arch 67, which in turn dominates Arch 12 in this preliminary risk assessment.

V. Development and Production Cost

A. Programmatic Drivers During the Apollo Program Relative to Today

In selecting the Lunar Orbit Rendezvous mode (with the Command Module, the CEV of the time, to lunar orbit) for the Apollo Program, one of the key drivers in the decision was the need to achieve the objectives of the endeavor rapidly with less regard to the particular funding level required to do so. By developing a separate "CEV" (CM) and "LSAM" (LM), the requirements on each could be focused on the individual mission phases the element needed to support and the development of each could be decoupled. This allowed each development to be managed independently to ensure completion on schedule.

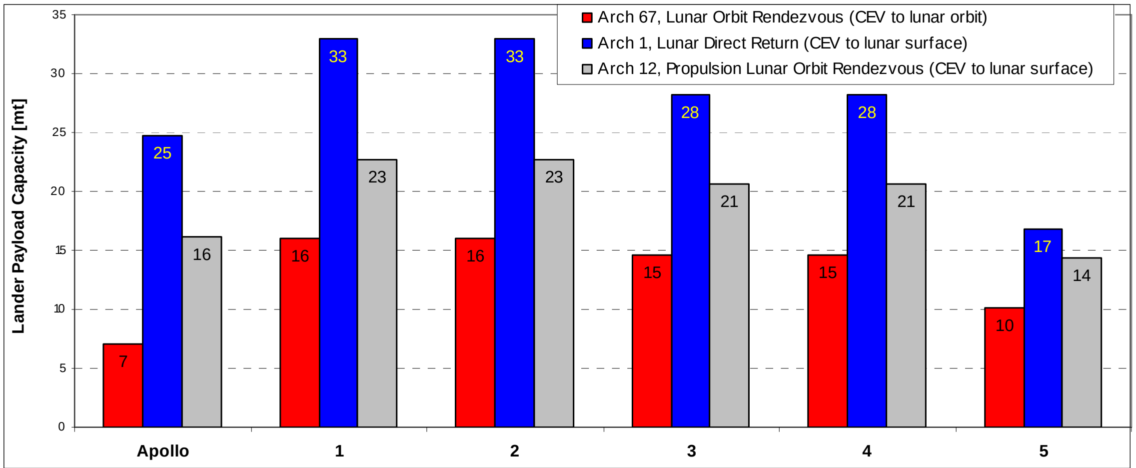
While this approach did allow for the achievement of the overall mission within the time allotted (i.e., by the end of the decade), by carrying out two parallel developments overall program costs were increased. While this suited the objectives of the Apollo Program, in many ways the objectives of today are opposite, with affordability now a key driver and the schedule being shifted to fit within a "go as you pay" style cost cap.

The present-day cost and schedule drivers point towards decreasing the cost by developing a lunar surface-capable CEV, as doing so would imply that only a single crew compartment needs to be developed for the Crew Transportation System, as opposed to two distinct crew habitats in the case of architectures with a separate LSAM. While the development cost of the CEV itself may be higher due to additional requirements, the overall development cost will be lowered by eliminating the additional LSAM crew compartment.

The affordability benefits of having a lunar surface-capable CEV go beyond development into production as well. Eliminating the LSAM crew compartment will eliminate the required production line and associated support infrastructure. A common saying goes "parts attract cost" – by lowering the overall number of elements, developing a lunar surface CEV will thus lower the overall cost of the program.

B. Lunar Crew and Cargo Descent Stage Commonality

While the primary analysis presented in this paper has focused on the crew transportation solution for lunar exploration, as the Vision for Space Exploration calls for the sustained and affordable exploration of the Moon, Mars, and beyond, it is important to also be cognizant of the other aspects of the overall exploration vision. To enable a long-duration stay on the lunar surface in order to both explore the Moon and prepare for the exploration of Mars, long duration habitats and other surface systems will need to be emplaced. Analysis we have conducted indicates that cargo delivery to the lunar surface on the order 25 to 30 metric tonnes may be required to meet such needs. As a system to deliver payloads of this magnitude will be required, it is important to consider whether the lander employed for the crew transportation system could be common with that for cargo delivery. Figure 7 presents the payload capacity of the descent stage employed in each of the three architectures for the mission and technology options analyzed in Section III. The lander capacity for the Lunar Direct Return architecture is consistently the highest of the architectures examined and, other than in the ISRU case, should be able to deliver cargos such as large surface habitats to enable a long duration presence on the lunar surface. While it may be possible to modify the hardware configuration or operational sequence of the landers in the other architectures, the direct applicability of the Direct Return architecture lander would be highly desirable as it could eliminate the need for the development of a second lander for habitats and large cargo elements.



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Operation Sequence	LOI with SM / descent stage	LOI with EDS	LOI with EDS	LOI with EDS	LOI with EDS	LOI with EDS, ISRU oxygen for ascent
Location	'Near equatorial'	'Near equatorial'	Global access	Global access	Global access	Global access

Figure 7. Descent stage payload capacity by architecture across mission and technology options.

C. Mars Applicability

Looking beyond the Moon, applicability of the lunar exploration elements to Mars exploration is also of great import. In our Mars architecture analysis, we have identified a number of leading candidates that include a CEV on the Martian surface, serving as either the ascent cabin of the Mars Ascent Vehicle or as the Earth entry capsule affixed to a surface-based Earth Return Vehicle. Developing the lunar CEV with surface capabilities will thus enable these leading architecture candidates, allowing full flexibility in Mars architecture selection, which is quite important at this early phase of development. We have also conducted extensive commonality analysis of the Lunar Direct Return architecture and have found that a high degree of element-level commonality is possible between lunar and Mars exploration systems when that lunar architecture is employed. While similar commonality approaches may be possible with the alternate architectures described in this paper, they have not yet been examined to the same level of detail. As such, the Lunar Direct Return architecture may offer additional benefits in terms of easing the use of common elements between Moon and Mars exploration, decreasing the overall lifecycle cost and accelerating the timeframe of missions to Mars.

D. Accelerated Development of CEV for ISS

While developing the CEV to directly support lunar surface missions appears to offer significant benefits, the need to provide a replacement U.S. crew transportation capability for the International Space Station (ISS) as promptly as practicable after the retirement of the Space Shuttle in 2010 will also be a driver on CEV development. To determine the impact of developing a CEV to support eventual lunar surface missions, we performed an option cost analysis to assess the additional upfront functionality that would need to be included in a LEO CEV to enable lunar surface missions in the future. In this analysis, we assumed that in the initial development of the LEO CEV, consideration will be included for its eventual Block 2 upgrade to support, at minimum, missions to Lunar Orbit as part of a Lunar Orbit Rendezvous surface exploration architecture. As such, the important aspect to determine currently is the impact on a Block 1 CEV of being able to be upgraded to a Block 2 Lunar Surface CEV relative to solely being able to be upgraded to a Block 2 Lunar Orbit CEV.

A functional analysis was performed to identify the differences between a Block 2 Lunar Surface CEV relative to a Block 2 Lunar Orbit CEV, and these functionality differences (or deltas) were then grouped according to their impact on the Block 1 CEV using the following categories:

Category 1: Provided in the initial CEV design for ISS missions.

Category 2: Scarred (interfaced) in the initial ISS design, and then fully integrated by way of a block-upgrade.

Category 3: Incorporated solely by way of a block upgrade prior to lunar missions.

The effect on development cost and time determines which of these three strategies is chosen for individual subsystem functions. The functional analysis extended one level below subsystem functions. Functions were arranged in three groups depending if the same, increased, or decreased functionality was needed for a lunar surface CEV compared to a lunar orbit CEV.

Out of a total of 74 functions analyzed, 17 functions showed significant deltas going from a lunar orbit capable CEV to a lunar surface capable CEV. Of these 17 functions, 5 would need to be incorporated directly into a Block 1 ISS CEV design (category 1), for another 5 appropriate interfaces or 'scars' would have to be provided in the Block 1 ISS CEV design (category 2) to ease the upgrade to a Block 2 Lunar Surface CEV, and the remaining 7 functions could be incorporated solely through the upgrade to a Block 2 Lunar Surface CEV. Figure 8 and Figure 9 provide an overview of the functions from categories 1 and 2:

Category 1: Need to be incorporated fully in Block 1

Block 2 LS CEV Functionality	Delta between Block 2 LO CEV and Block 2 LS CEV	Implications for Block 1 LEO CEV
Partial-g waste management	ECLS systems must function in partial-g environment (as opposed to just in a zero-g environment)	Design waste management system for partial-g and zero-g.
Provide crew accommodations for partial-g	Interior cabin layout more constrained in partial-g environment than in zero-g.	Ensure that consideration for eventual partial-g functionality is included in initial crew compartment geometry and crew accommodations design.
Provide crew support for EVA suit operation on the surface	Donning / doffing EVA suits in partial-g environment yields different requirements than in zero-g. Potentially need more volume or different cabin layout.	Provide sufficient volume and cabin layout for donning and doffing of EVA suits in partial-g environment (e.g., considerations for stowable seats to increase useable space).
Support crew ingress / egress for surface EVA	Must provide access to the surface from the LS CEV.	Include larger door, hand holds in LS CEV.
Provide cockpit for lunar descent & ascent ops	Must include appropriate human factors considerations for descent to Lunar surface	Ensure cock-pit is compatible or upgradeable for operation during descent to lunar surface [ascent will be similar to Earth launch]

Figure 8. CEV functions required for a lunar surface CEV but not for a lunar orbit CEV that need to be incorporated in the Block 1 ISS CEV design.

Category 2: Need to have appropriate interfaces ("hooks and scars") in Block 1 to enable Block 2 upgrade

Block 2 LS CEV functionality	Delta between Block 2 LO CEV and Block 2 LS CEV	Implications for Block 1 LEO CEV
Reject excess heat from system lunar surface	Interface to additional lunar surface radiator	Include as upgrade, leave hooks and scars for interface
Provide active thermal control on lunar surface (IPU/SM)	Provide additional cooling of LS CEV through IPU/SM.	Allow for increased thermal through-put in block upgrade
Provide thermal control on lunar surface (Capsule/CM)	In some cases, more strenuous thermal environment than in deep space, resulting in greater thermal load on Capsule/CM	May require increased cooling ability in Capsule/CM or option for increased insulation. (Most of this functionality will reside in other vehicles.)
Provide crew interface during lunar descent and ascent	Human factors (cockpit layout, crew orientation) for lunar descent and ascent may be different than for Earth launch and reentry	Consider human factors of crew interface with avionics during descent/ascent in Block 1. Add full capability in Block 2L.
Generate DC power on the lunar surface	Increased energy required for longer duration	Allow for additional energy storage capability in block upgrade (either internally or through external power)

Figure 9. CEV functions required for a lunar surface CEV but not for a lunar orbit CEV for which interfaces need to be incorporated in the Block 1 ISS CEV design.

While a difference of 10 functions out of 74 is not negligible, it should be taken into account that many of these functions solely involve providing partial-g functionality in addition to 0-g functionality; while this might result in a difference in design, it is not likely to increase development cost. Also, for crew egress and ingress on the lunar surface, a larger door has to be provided; as a door of some size is necessary in any case to allow crew ingress on Earth, providing a larger door is not likely to significantly increase the cost if accounted for up-front. In contrast, modifying an existing design to increase the door size would incur a substantial cost, which highlights the need for including these types of considerations up-front.

The two functions with the most significant impact on the Block 1 ISS CEV design are to provide a cockpit which can be upgraded for lunar descent and landing operations, and to provide CEV thermal control on the lunar surface (especially around lunar noon). The cockpit function could potentially be provided by the regular CEV cockpit with added instruments and displays showing real-time video from cameras on the vehicle outside. The additional thermal control could potentially be provided by an upgraded system in the CEV Service Module (SM) in concert with an additional surface thermal control module (plugged-in) alone; the thermal control system in the CEV capsule, however, would have to be sized to accommodate the additional heat-flux (such as by increasing the massflow in a cooling-loop). Again, this is not likely to cause an increase in development cost, because it merely warrants a modified design, not a more complex one.

It should be noted that the functional analysis presented in this section is preliminary; as a comparatively large number of functions on the sub-subsystem level was analyzed, however, the results give a good indication of the impact of preserving an option in Block 1 for an eventual Block 2 Lunar Surface CEV. Because of the significant life-cycle savings in development, test, fixed production and operations cost of a lunar direct return architecture along with the associated safety and mission risk benefits, it appears worthwhile to preserve the option of extending a Block 1 ISS CEV to a Block 2 lunar surface CEV for the time being and to study this issue in more depth. Initial analysis suggests the option cost is acceptable.

VI. Conclusions

From a mass perspective, given present day technology and mission objectives, architectures in which the CEV travels to the lunar surface are comparable to Lunar Orbit Rendezvous architectures. This contrasts with the results for the mission objectives and technologies employed during the Apollo program, which would have resulted in significantly higher masses for CEV to surface-style architectures. With the introduction of ISRU, CEV to surface architectures would offer significant mass savings relative to LOR.

From a risk perspective, the Lunar Direct Return architecture offers significant advantages as all mission assets are accessible by the crew for inspection and maintenance, no rendezvous or docking are required for return to Earth, and no assets must be operated autonomously in lunar orbit. These advantages will become particularly relevant in the long-duration lunar missions envisioned to prepare for expeditions to Mars.

From a cost perspective, having the CEV travel to the lunar surface offers significant savings by eliminating the development of second crew compartment with all of its life-cycle cost implications in development and operations, including design, testing, launch processing, flight control, software maintenance, and logistics. The Lunar Direct Return architecture offers the additional benefits of eliminating one propulsion stage and allowing the use of a common lander for both the crew transportation system and the emplacement of large surface assets, again with significant life-cycle benefits. Developing the CEV to support surface missions will allow it to support all of the preferred Mars architectures identified during our study.

For minimal upfront impact, the development of the CEV to support ISS missions can be done in such a manner as to provide the flexibility to select either a lunar surface CEV architecture or a lunar orbit CEV architecture up until lunar mission hardware development begins in earnest. This preserves the option to in the future select the Lunar Direct Return with all of its associated benefits, even if it is not selected as the initial baseline at the outset of CEV development.

Given the numerous benefits of the Lunar Direct Return architecture, we recommend selecting it as the baseline architecture for human lunar exploration. Whether or not the Direct Return architecture is initially the baseline, we recommend developing the ISS CEV such that it is extensible to lunar surface missions should such architectures be selected in the future.

Appendix

A. Architecture Mass Determination

To determine the architecture masses described in Section III and the crew lander descent stage capacity in Section V, each of the architectures were modeled as a series of stages with the crew compartments in Table 1 and the additional masses in Table 2 as the payload. The Lunar Direct Return architecture (Arch 1) was modeled as having 3 stages: an Earth Departure Stage, a Descent Stage, and an Ascent and Trans-Earth Injection (Ascent & TEI) stage. Both the standard and propulsion lunar orbit rendezvous architectures (Arch 67 and 12) were modeled as having 4 stages: an Earth Departure Stage, a Service Module stage which does not descend to the lunar surface, a Descent Stage, and an Ascent Stage. Each of the stages were modeled using the rocket equation based upon the stage's respective payload plus a structural factor, alpha, defined as the ratio of the dry stage mass to the propellant contained within the stage. The delta-v's for the Apollo case were obtained from the Apollo 11 press-kit. The delta-v's for the modern cases were obtained from the NASA Focused Trade Report[10] with the exception of the plane-change delta-v's, the derivation of which are described in Subsection B of the Appendix. The specific impulse (Isp) and structural factor (alpha) for the Apollo cases are representative of those actually used during the program[11,12]. The modern values are conservative estimates based primarily upon present day upper stages.

Table 2. Additional Payload Masses.

	Mass [kg]
Cargo	500
Samples	100
Per Crew	180

The parameters used for modeling the Earth Departure Stage are included in Table 3. As mentioned previously, the operational sequence was also modified such that the Earth Departure Stage performed both trans-lunar injection and lunar orbit insertion in the modern case, as opposed to solely trans-lunar injection in the Apollo case, so as to take advantage of its higher specific impulse.

Table 3. Earth Departure Stage parameters.

	Delta-V [m/s]	Isp [s]	Alpha [-]
Apollo, Trans-Lunar Injection	3,200	430	0.15
Modern, Trans-Lunar Injection & Lunar Orbit Insertion	4,082	462	0.11

The parameters used for modeling the Service Module in Arch 12 and 67 are included in Table 4. The specific impulse (Isp) and structural factor (alpha) for the Apollo case are representative of those for the Apollo Service Module when adjusted for the sub-systems assigned to the crew compartment. In the Apollo case, the first delta-v is for lunar orbit insertion, the second delta-v is for trans-Earth injection. The first delta-v is applied to the descent stage for the Lunar Direct Return architecture in the Apollo case as it does not have a specific Service Module stage. For the remainder of the cases, the first delta-v is representative of the lunar orbit plane change required to enable global access with anytime return as described in Subsection B (not required in the near-equatorial case). The second delta-v is once again used to perform trans-Earth injection.

Table 4. Service Module parameters.

	Delta-V 1 [m/s]	Delta-V 2 [m/s]	Isp [s]	Alpha [-]
Apollo Hypergolics; Near-Equatorial	950	950	314	0.27
Modern Hypergolics; Near-Equatorial	-	966	316	0.15
Modern Hypergolics; Global Access	1,500	966	316	0.15
Methane-Oxygen; Global Access	1,500	966	362	0.15

The parameters used for modeling the descent stage are included in Table 5. The decreased delta-v is attributable to higher thrust trajectories used during the modern case. The structural factor, alpha, was also increased in the case of a hydrogen-oxygen lander to take into account the lower density of hydrogen fuel.

Table 5. Descent Stage parameters.

	Delta-V [m/s]	Isp [s]	Alpha [-]
Apollo Hypergolics	2,083	311	0.20
Modern Hypergolics	1,881	316	0.20
Methane-Oxygen	1,881	362	0.20
Hydrogen-Oxygen	1,881	430	0.25

The parameters used for modeling the descent stage are included in Table 6. The Ascent & TEI delta-v was used in sizing the ascent and trans-earth injection stage for the Lunar Direct Return architecture. The Ascent Only delta-v was used in sizing the ascent stage for the standard and propulsion lunar orbit rendezvous architecture.

Table 6. Ascent Stage parameters.

	Delta-V [m/s]		Isp [s]	Alpha [-]
	Ascent & TEI	Ascent Only		
Apollo Hypergolics	2,821	1,871	311	0.15
Modern Hypergolics	2,681	1,834	316	0.15
Methane-Oxygen	2,681	1,834	362	0.15

B. Lunar Orbit Plane Changes for Global Access with Anytime Return

In developing a system to enable the lunar exploration objectives of the Vision for Space Exploration, it is highly desirable to provide access to lunar landing sites anywhere on the surface of the Moon. During their time on the lunar surface, innumerable events may occur which would lead to return the crew to Earth, thus in addition to global access, an anytime return capability is desired. A direct return architecture is unaffected by this combination of requirements, as it can return at anytime from anywhere on the lunar surface using its nominal propulsive and operational capabilities. For architectures with orbital assets, this is not the case however. Due to the Moon's rotation, the plane of the orbiting asset moves relative to the landing site, so that a direct ascent and rendezvous in the plane of the orbiting vehicle might no longer be possible. Also, in order to depart from the Moon to the Earth, the crew has to be in an orbital plane that contains the departure excess velocity vector.

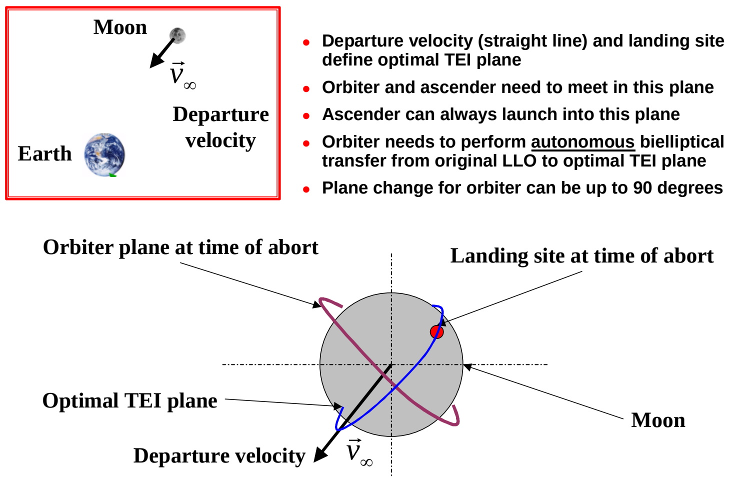

Figure 10. Lunar departure strategy with arbitrary landing location and departure time.

In order to accommodate this capability in architectures employing lunar orbit rendezvous, we make use of a plane change strategy suggested in the NASA Lunar Architecture Broad Trade Report[9]. The strategy involves an autonomous plane change of the orbital asset to bring it into an orbital plane containing both the landing site location and the desired departure velocity vector. The crew can thus ascend into this orbit, rendezvous with the orbital asset, and return to Earth. Figure 10 provides a visualization and description of this plane change strategy.

In order to accommodate lunar global access with anytime return using this strategy, a capability for up to a 90-degree plane change must be included in the orbital asset. As shown in Figure 11, this can be accomplished by providing a delta-v of 1,500 m/s and performing a three-impulse plane change with period of less than 15 hours. In a three-impulse plane change, the first impulse is used to raise the apocenter of the orbit, the second impulse is used at the new apocenter to rotate the plane, and the third impulse is used at pericenter to recircularize at the initial altitude. For plane changes in lunar orbit of approximately 50-degrees or higher, the three-impulse approach allows for a lower delta-v than typical single-impulse plane changes at constant altitude.

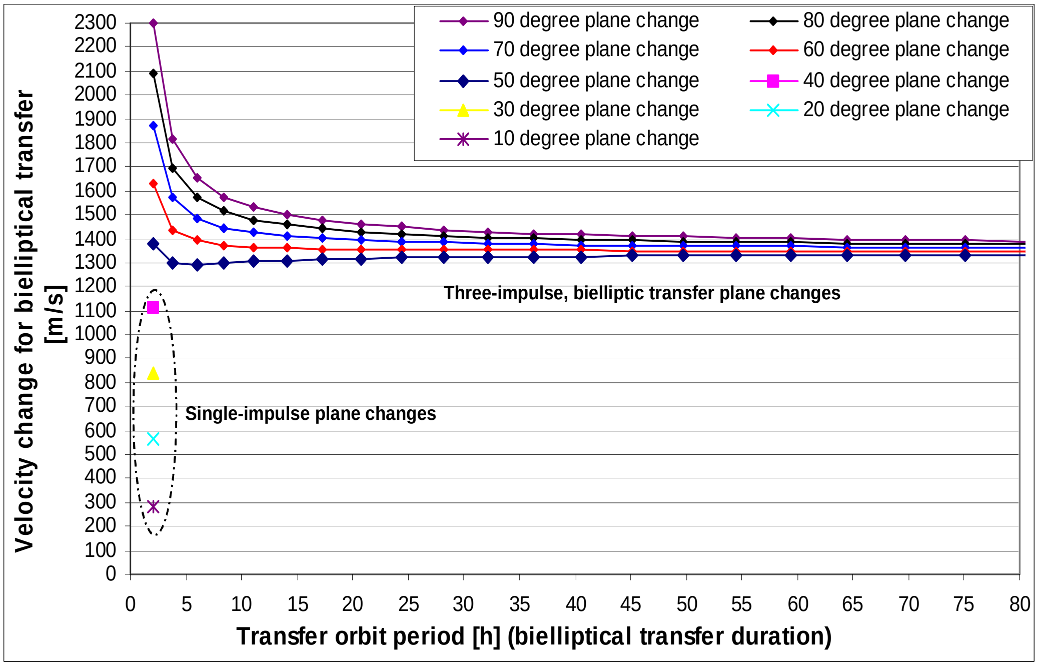

Figure 11. Plane change delta-v for single-impulse and three-impulse transfers over a range of plane change angles and transfer orbit periods.

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