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CEV TRAJECTORY DESIGN CONSIDERATIONS FOR LUNAR MISSIONS

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The CEV translational maneuver ΔV budget must support both the successful completion of a nominal lunar mission and an "anytime" emergency crew return with the potential for much more demanding orbital maneuvers. This translational ΔV budget accounts for Earth-based LEO rendezvous with the LSAM/EDS stack, orbit maintenance during the lunar surface stay, an on-orbit plane change to align the CEV orbit for an in-plane LSAM ascent, and the Moon-to-Earth TEI maneuver sequence as well as post-TEI TCMs. Additionally, the CEV will have to execute TEI maneuver sequences while observing Earth atmospheric entry interface objectives for lunar high-latitude to equatorial sortie missions as well as near-polar sortie and long duration missions. The combination of these objectives places a premium on appropriately designed trajectories both to and from the Moon to accurately size the translational ΔV and associated propellant mass in the CEV reference configuration and to demonstrate the feasibility of anytime Earth return for all lunar missions. This report examines the design of the primary CEV translational maneuvers (or maneuver sequences) including associated mission design philosophy, associated assumptions, and methodology for lunar sortie missions with up to a 7-day surface stay and with global lunar landing site access as well as for long duration (outpost) missions with up to a 210-day surface stay at or near the polar regions. The analyses presented in this report supports the Constellation Program and CEV project requirement for nominal and anytime abort (early return) by providing for minimum wedge angles, lunar orbit maintenance maneuvers, phasing orbit inclination changes, and lunar departure maneuvers for a CEV supporting an LSAM launch and subsequent CEV TEI to Earth return, anytime during the lunar surface stay.

INTRODUCTION

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In addition to a successful nominal mission, human lunar mission design must provide the crew with options for a safe return to Earth in the event of an off-nominal situation. Mission planners are charged with the task of providing safe Earth-return coverage that is as complete as possible, given mission constraints such as vehicle performance, vehicle structural or thermal limitations, and crew physiological limits. The lunar mission design carries a primary goal to provide a crew with a successful trip to and from the Moon as well as inherent abort-friendly characteristics. Trans-lunar injection (TLI) and lunar orbit insertion (LOI) maneuvers are executed by the Earth departure stage (EDS) and the lunar surface access module (LSAM), respectively. However, while the crew exploration vehicle (CEV) does not participate in these maneuvers, the LOI targets (i.e., lunar orbit inclination and longitude of the ascending node [LAN]) support anytime early return maneuvers which would be executed by the CEV. For these reasons, we examine the CEV nominal and early return missions from an overall flight design perspective.

The mission design in this report was originally developed in support of the Exploration Systems Architecture Study (ESAS)[1] **(Reference 1).** It represents a snapshot of the NASA mission design as it existed soon after the release of the ESAS, but before the selection of the CEV prime contractor. The CEV and LSAM mission design are part of an overall architecture intended to provide for short or long duration missions to the lunar polar region as well as short duration mission (up to 7 days on the lunar surface) to all other lunar landing sites. This architecture also contains features that provide the crew with an anytime early Earth return capability.

MISSION DESIGN OVERVIEW

A crewed lunar mission consists of a two-launch sequence as shown in Figure 1. The first launch has the Cargo Launch Vehicle (CaLV) carrying the lunar surface access module (LSAM) and the EDS to low Earth orbit (LEO). The LSAM/EDS parking orbit determines the time of the opening of the trans-lunar injection (TLI) window. Subsequent launch of the CEV on the Crew Launch Vehicle (CLV) initiates the CEV rendezvous with the awaiting passive LSAM/EDS stack in LEO.

Following the CEV rendezvous with the LSAM/EDS stack in LEO, the crew prepares the vehicle for TLI. After the opening of the injection window, the EDS executes TLI placing the CEV/LSAM/EDS stack on a path to the Moon. After TLI and before lunar orbit insertion (LOI), the EDS stage is jettisoned and up to four trajectory correction maneuvers (TCM) can be executed for the CEV/LSAM stack.

Following a nominal 4-day flight time, the LSAM descent stage executes a one or three burn LOI, placing the CEV/LSAM stack in the desired 100 km circular orbit altitude lunar destination orbit (LDO). A 24 hour period follows LOI, allowing for operations activities such as vehicle checkout, crew sleep cycles, and preparation of the LSAM for descent and landing. After this period, the LSAM performs a de-orbit and subsequent powered flight landing to the surface.

The lunar surface stay can last up to seven days. During this time, the CEV performs an on-orbit plane change maneuver to allow the LSAM ascent stage to execute an in-plane ascent at the end of the surface stay, leaving the spent LSAM descent stage on the lunar surface. The ascent initiates a sequence of rendezvous maneuvers, performed by the active LSAM, with a passive CEV. Following CEV/LSAM docking and a post-rendezvous checkout and departure preparation period, the CEV jettisons the LSAM ascent stage and performs a trans-Earth injection (TEI) maneuver to return the crew to Earth. This maneuver may consist of a single or three burn sequence, depending upon the required TEI departure plane change. The TEI can be targeted to either water or land landing. Following TEI, the CEV can perform up to 3 planned TCMs. Anywhere from 3.5 to 4.5 days after TEI, the CEV arrives at the 400,000 ft (121.92 km) Earth entry interface (EI) altitude. About an hour prior to EI, the crew jettisons the CEV service module, leaving the crew module to complete the Earth entry to touchdown (or splashdown).

MISSION TYPES – SORTIE VS OUTPOST

There are two fundamental mission design types addressed in this report: a global access, short duration sortie mission and a polar, short or long duration (outpost) mission. The global access sortie landing site latitudes range between -85° and 85° while the polar mission landing sites lie within 5° latitude of either the north or south pole. Early interest in the polar missions focused on the south pole as a preferred polar landing site.

For both mission types, the CEV possesses the capability to allow the crew to perform an anytime early return from the Moon. For both nominal and early return cases, the CEV conducts two major maneuvers to ultimately place the crew on a path back to Earth. First, the CEV executes an on-orbit alignment plane change (also known as the ascent plane change) that allows the LSAM to subsequently perform an in-plane ascent and rendezvous with a now passive CEV. Second, after a post-rendezvous and pre-TEI checkout phase, the CEV jettisons the LSAM and executes a TEI of up to three burns. In both mission types, the TEI maneuver can accommodate up to a 90° relative declination (between the Earth return target V-infinity vector and the initial parking orbit plane).

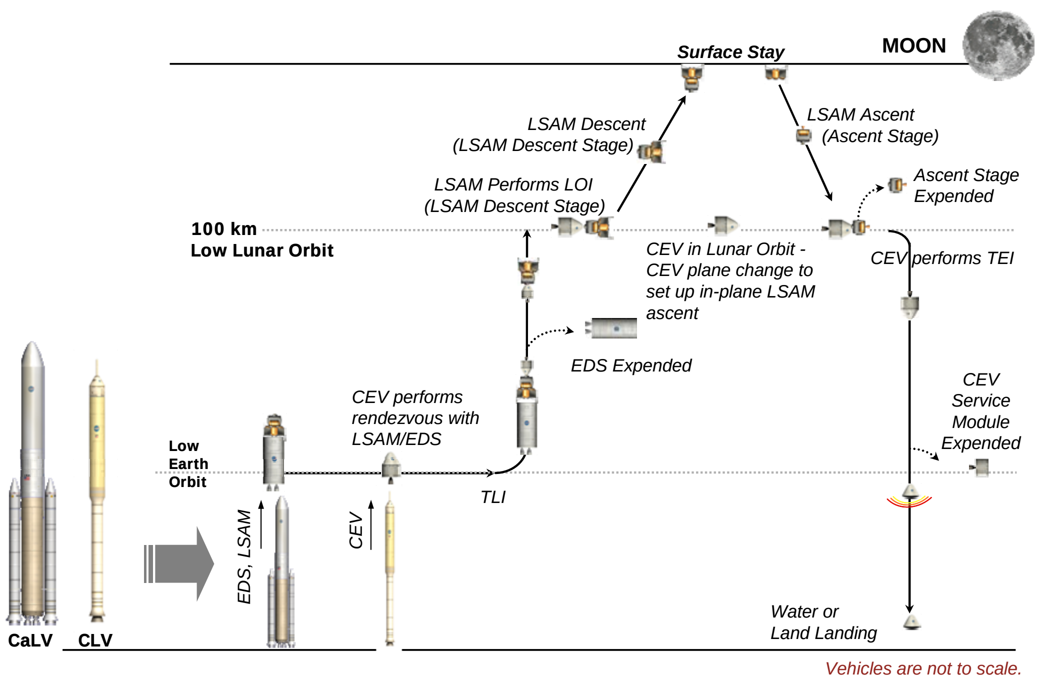

Figure 1 Mission Design Overview

Global Access Sortie Missions

Pure plane changes are costly in a low lunar orbit (i.e., the 100 km altitude circular CEV parking orbit). The global access sortie mission design minimizes the ascent plane change via TLI/LOI arrival orbit targeting, allowing the LSAM ascent (and rendezvous) to occur at any time during a lunar surface stay **(Reference 3)** of up to seven days. Specifically, the TLI and LOI maneuvers combine to target the CEV/LSAM stack to a particular LDO inclination and LAN*, supporting a particular lunar landing site latitude and longitude. For some lunar landing site locations, the required LDO inclination and LAN targets can result in up to a 90° relative declination between the incoming LOI V-infinity vector and the LDO achieved at the conclusion of the LOI burn(s). The possibility of this large LOI relative declination drove the mission design to include a three-burn LOI sequence to reduce its delta-V (ΔV) cost.

A TLI propulsive ΔV capability of 3150 m/s reflects the capability of the EDS to push the CEV/LSAM stack toward its LOI orbit target, which in turn reflects the lift capability of the CaLV. A

* The LDO is a circular 100 km altitude parking orbit.

1250 m/s LOI ΔV represents the propulsive capability of the LSAM (descent stage) to insert the CEV/LSAM stack onto the desired LDO. Note that global lunar landing site access is achieved only when augmenting this propulsive ΔV with up to three days of extended lunar orbit loiter (in addition to the nominal 24-hour post-LOI loiter period) prior to LSAM descent.

Polar Outpost or Sortie Missions

Outpost mission design differs from a global access sortie mission in that the LOI maneuver always targets to a 90° inclined (polar) orbit with no constraints on LAN. This allows for a minimum ΔV single burn LOI with a ΔV of about 850 m/s. This covers a worst-case geocentric transfer angle of about 57.1°. This maximum geocentric wedge angle stems from the worst possible geometry associated with an Earth departure from a 28.5° LEO targeting to the Moon at the time of the Moon's maximum orbit inclination of 28.6°. In this worst case geometry, the geocentric wedge angle is the sum of the pre-TLI inclination and the Moon's orbit inclination. A polar landing site can lie anywhere within a 5° latitude band of the pole. Like the global access sortie missions, a polar sortie mission can possess a lunar surface stay time of up to seven days. Outpost missions can last up to 180 days with up to an additional 30 days of contingency surface time.

TEI MISSION DESIGN CONSIDERATIONS

Mission requirements called for the CEV to safely return the crew to Earth at any time and from any lunar orbit (inclination and LAN). To assure sufficient propellant for the CEV to accomplish this, the authors originally selected a worst-case lunar departure geometry as a bounding case. This geometry affects the lunar departure V-infinity magnitude and thus, the TEI ΔV. The Moon-to-Earth flight time also plays a significant role in the TEI propellant cost and must also be considered.

The performance cost of the TEI maneuver sequence depends upon the magnitude of the TEI V-infinity vector and the relative declination between that V-infinity vector and the initial lunar parking orbit plane. The magnitude of the TEI V-infinity vector depends upon the Earth-Moon geometry at TEI as well as the Earth return orbit inclination target. The Moon's inclination, relative to the Earth equator, varies between 18.3º and 28.6º over an 18.6 year period (metonic cycle). The lunar orbit inclination and the lunar declination (the location of the Moon relative to the Earth equatorial plane) at the time of TEI will affect the V-infinity and thus the TEI ΔV (for a given relative declination). The worst case departure geometry occurs when the Moon is at its maximum inclination in the metonic cycle (i.e., 28.6º) coincidentally when the Moon is at nodal crossing (zero declination) and the Earth return trajectory is targeted to a polar (90º) Earth inclination. This geometry, shown in Figure 2, shows that this confluence of worst case geometrical conditions results in a 118.6º geocentric wedge angle[†]. The TEI V-infinity vector associated with a maximum 118.6º geocentric wedge angle was used to support determination of an upper limit for the CEV TEI ΔV requirement.

Figure 3 shows the variation of the TEI V-infinity magnitude with Earth return inclination (and associated geocentric wedge angle for a TEI with the Moon at nodal crossing) along with the variation in V-infinity magnitude due to Earth return flight time. This figure reflects how shorter flight times can result in significant increases in the V-infinity magnitude (and associated TEI ΔV). For this mission design, flight times between 3.5 and 4.5 days were selected as they provided relatively small V-infinity vectors with reasonable Earth return flight times. The one-day variation in flight time allows for 360º of Earth return longitude control, thus providing for full EI state vector targeting.

The TEI maneuver sequence, shown in Figure 4, can consist of either a single burn or three burns, depending upon the magnitude of the relative declination change required to carry the CEV from its initial parking orbit plane to the final target Earth return V-infinity vector. For zero or small lunar departure relative declinations, the CEV will possess sufficient propellant to execute TEI using only a single burn.

[†] Note that Figure 2 shows a case where the CEV returns to Earth by over-flying the north pole. Subsequent skip entry approach requirements dictate a preference for a south pole over-flight upon Earth return. However, the TEI ΔV limit remains unchanged.

However, global access and anytime return requirements for crew safety dictate that there may be times when the relative declination becomes large. In this case, the CEV TEI ΔV, and associated propellant requirement, can be reduced through the use of a three-burn sequence. A 90º plane change capability during the TEI sequence will allow the CEV to begin in any lunar orbit orientation and intercept any V-infinity vector direction. With this capability, no required TEI window exists and the crew can begin the maneuver sequence to leave LLO at any time*.

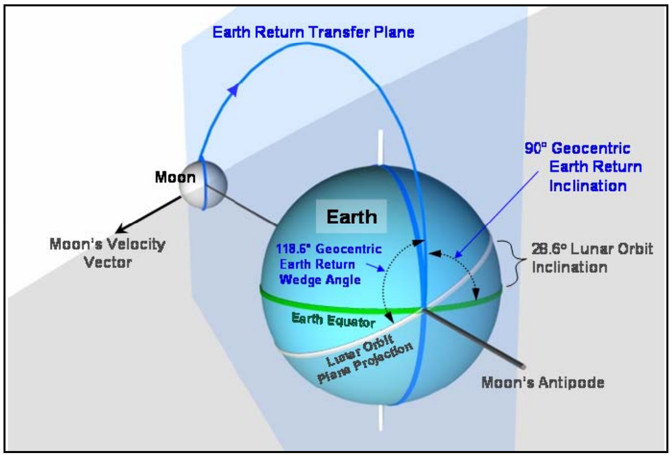

Figure 2 Moon to Earth Worst Case Transfer Geometry

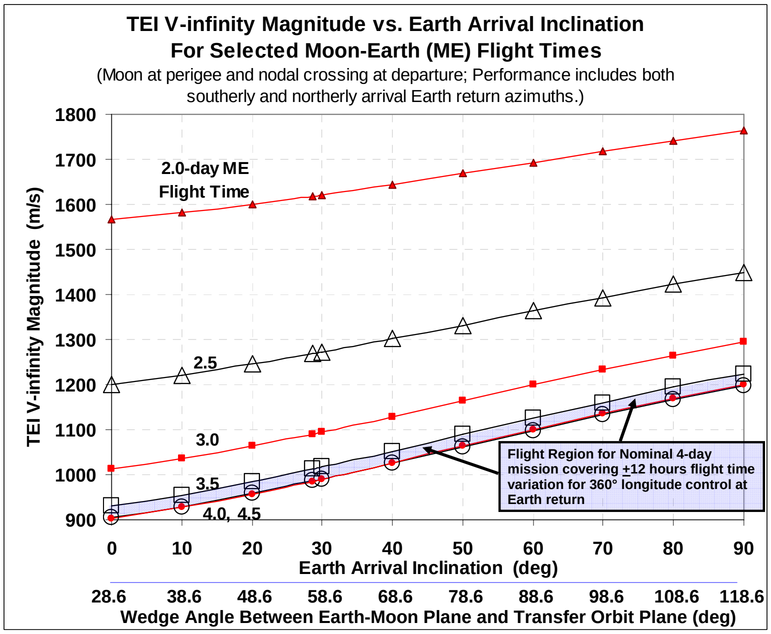

Figure 3 TEI V-Infinity Magnitude vs Earth Arrival Inclination For Selected Moon Earth Flight Times

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* The anytime departure is based on a planar requirement. Proper phase location of the CEV is required for a minimum ΔV TEI. Therefore, for an approximately 2-hour LLO orbit period, the crew could execute an anytime TEI with up to a 2-hour delay.

The first TEI burn (TEI 1) raises apoapse to create an intermediate transfer orbit that will be used to perform a plane change. Note that initial analyses employed a planar TEI 1 maneuver to obtain reasonable performance results. A more rigorous optimization would include a small plane change in this first burn. After TEI 1, the CEV coasts to the vicinity of the apoapse of the transfer orbit where it executes TEI 2*, which is essentially a plane change maneuver. This maneuver is performed at high altitude to reduce the plane change cost. The TEI 2 maneuver employed a "fail-safe" technique which provides added propellant to maintain a positive periapse altitude (e.g., 100 km) during a finite burn plane change. A minimum finite burn TEI 2 maneuver ΔV shows a variation in the periapse altitude during the course of the burn. For large plane changes (e.g., 90º plane change), the periapse altitude can become negative very soon (around 5%-10% into the burn) and remain negative until the burn is nearly complete. Clearly, this represents a safety hazard to the crew in the event of a premature engine shutdown during the 2nd TEI burn. The additional cost of a "fail-safe" burn is around 11%-12% of the optimal ΔV cost of the 2nd burn. Following TEI 2, the CEV coasts to TEI 3, which occurs in the vicinity of the periapse. TEI 3 places the CEV onto the Earth return V-infinity vector, targeted to a specific set of Earth EI conditions.

The entire three-burn maneuver sequence occurs over a period of about 24hours. The 24-hour total TEI time provides a compromise between minimizing the TEI ΔV and the total Earth return flight time (TEI 1 to EI), while maintaining full EI targeting to support land and water landings. An additional consideration was the intent to complete the TEI maneuver sequence itself as quickly as possible. After TEI 3 completion, a propulsion system failure may be survivable with the crew already on its way to an Earth return EI target state. A propulsion system failure prior to successfully completed TEI 3 could strand the crew in the vicinity of the Moon. Given confidence in a reliable propulsion system the cost of the TEI maneuver sequence could be further reduced by increasing the intermediate transfer orbit apoapse and allowing appropriate adjustments to the TEI 3-to-EI flight time. Given a perfectly executed TEI 3, the CEV could coast to Earth return for either a water or land landing. There are three TCMs provided after TEI 3 with small propulsive ΔV allotments (e.g., 1-3 m/s).

The total 3-maneuver TEI sequence ΔV cost is summarized in Figure 5 as a function of lunar departure V-infinity magnitude and the relative declination requirement. Also overlaid on this figure is the TEI ΔV cost increase due to reduced Moon-Earth flight times. The highlighted region near the bottom of the figure reflects the minimum 3.5-day flight time. A longer flight time is not shown as it does not drive the TEI ΔV cost. The data is this figure reflect the previously described confluence of worst-case geometric conditions. For the one-day TEI maneuver sequence followed by a minimum 3.5 day Moon-Earth flight time, this confluence of worst case geometry provides a bounding ΔV requirement for the CEV. For the 3.5 day minimum Moon-Earth flight time with a 90º Earth return inclination target (requiring a maximum 118.6º of geocentric wedge angle change) and for a 90º (maximum) relative declination, the TEI ΔV requirement is 1476 m/s. This ΔV requirement includes a "fail-safe" TEI 2 plane change.

Following ESAS, the CEV team at NASA-Johnson Space Center selected a 1450 m/s CEV TEI translational ΔV budget. This included 1435 m/s for a worst case geometry three-burn TEI maneuver sequence with a non-"fail-safe" TEI 2 burn combined with a 15 m/s propulsive reserve. The slight reduction in the TEI ΔV budget was based on the assumption that the CEV would not often encounter a situation where it faced this confluence of worst-case geometries. Subsequent analysis focused on better understanding the extent to which these conditions would actually occur over a metonic cycle as a function of the date of lunar departure and the lunar landing site latitude and longitude.

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* Note that a minimum ΔV TEI maneuver sequence will require that the burns generally do not occur on the line of apsides.

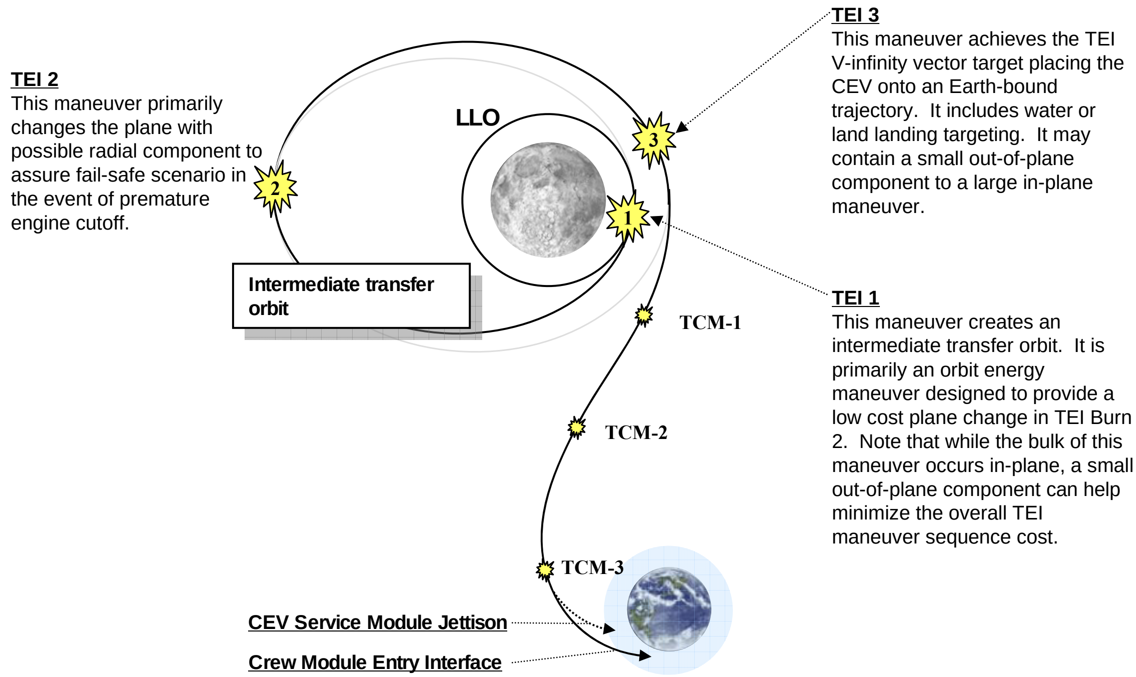

Figure 4 TEI Maneuver Sequence

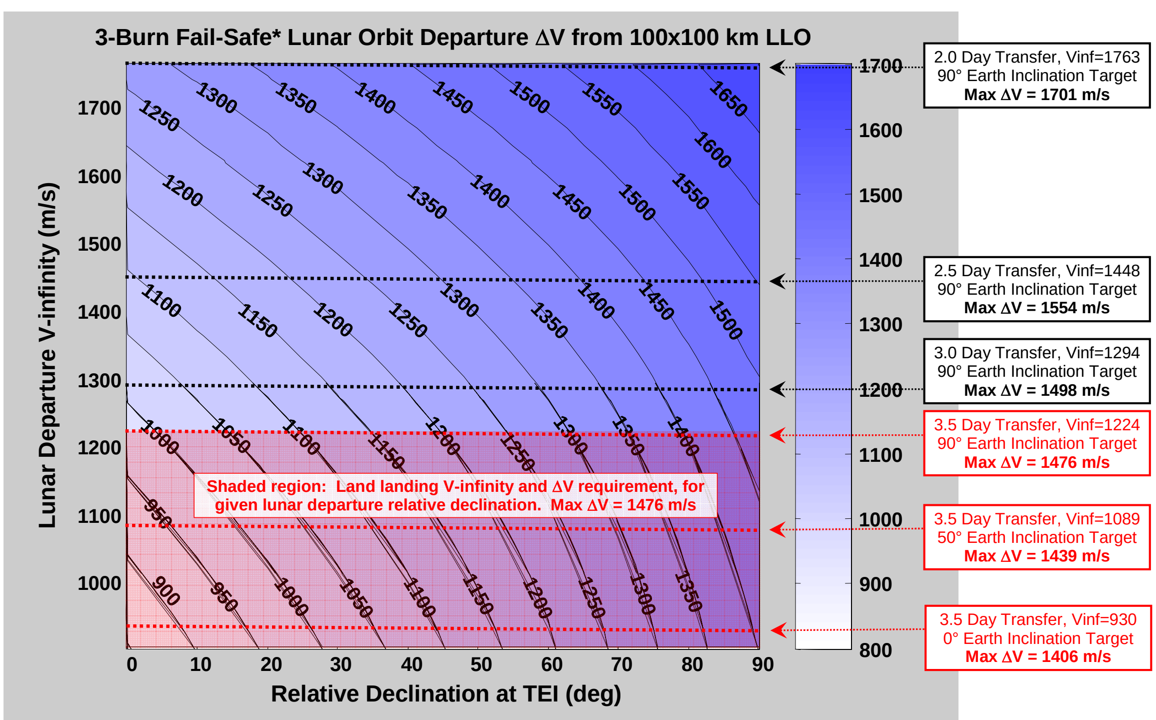

Figure 5 TEI V-Infinity Magnitude vs Earth Arrival Inclination For Selected Moon Earth Flight Times

MISSION PROFILE

Launch/Earth Rendezvous (FD1 vs FD3)

The Constellation Architecture Requirements Document (CARD) **(Reference CARD)** requires that the CLV provide a daily launch opportunity for not less than 4 consecutive days. If the initial launch attempt is scrubbed for weather, hardware, or other issues, then three additional launch opportunities will be available. The previously launched CaLV determines the date and time of the opening of the TLI window, so the lunar mission depends upon the successful launch of the CLV on one of these opportunities. A flight day one (FD1) rendezvous of the CEV with the awaiting LSAM/EDS stack requires about 7 hours from launch to docking, while a FD3 rendezvous requires around 50 hours. Since the time between the first launch opportunity and the opening of the TLI window is fixed, a successful launch of the CLV on the first opportunity will require the crew to loiter several days in LEO prior to TLI. If the longer FD3 rendezvous is planned for the 4th, and last, launch opportunity, this loiter time will be approximately 1.5 days longer than if a FD1 rendezvous were used. Therefore, a focus on the FD1 rendezvous profile will help minimize this LEO loiter time. The first three CLV launch opportunities could employ an FD3 rendezvous. Then the 4th launch opportunity could employ a FD1 rendezvous with no increase in the maximum LEO loiter time, as compared to an all-FD1 approach. Operational considerations will help determine the selection of a mixed FD3/FD1 approach. Following rendezvous and docking after launching on the last possible opportunity, a pre-TLI checkout time of 12 hours precedes TLI.

Rendezvous Phase Description

See Table 1 for Earth rendezvous mission profile and timeline.

a. The CLV liftoff at the KSC launch site is on July 13th at 20h:41m:30s GMT.

b. Following the CLV second stage engine cutoff, the CEV separates from the CLV and coasts to apogee, at which time the CEV will perform the phasing burn. The phasing burn (a posigrade, Hohmann transfer maneuver) is designed to place the CEV 43 km behind the LSAM at the time of the coelliptic maneuver.

c. One hour after the coelliptic maneuver, the Terminal Phase Initiation (TPI) burn occurs. TPI is a Lambert targeted maneuver, designed, after approximately 150 degrees of transfer angle, to place the CEV in the vicinity of the LSAM / EDS.

d. Terminal phase rendezvous and proximity operations begin after TPI. During the 150 degree transfer, the CEV will perform mid-course correction burns, if necessary. Near the end of the transfer time, the CEV will perform small braking maneuvers to slow its approach rate to the LSAM, finally arriving at a stable orbit condition.

e. Following docking, the CEV / LSAM will coast for approximately 4 days, until preparation for TLI begins.

Table 1

EARTH RENDEZVOUS MISSION PROFILE/TIMELINE

Trajectory Event	Mission Elapsed Time	Delta Velocity Magnitude of Translational Maneuver		Resultant Apogee / Perigee	
	dd:hh:mm:ss	m/s	ft/s	km	nmi
CLV Liftoff	00:00:00:00				
Main Engine Cutoff	00:00:09:18			180 / -60	97 / -32
Phasing Maneuver	00:00:26:11	99	325	267 / 180	144 / 97
Height Maneuver	00:01:12:21	32	105	292 / 267	158 / 144
Coelliptic Maneuver	01:23:03:28	8	26	293 / 289	158 / 156
Terminal Phase Initiation	02:00:03:28	2	7	298 / 291	161 / 157
Terminal Phase Finalization	02:00:41:09	2	7	299 / 294	161 / 159

Note: At the time of rendezvous initiation,
the LSAM / EDS is in a 299 x 294 km orbit.

Total Rendezvous Delta V = 143 m/s (470 ft/s)

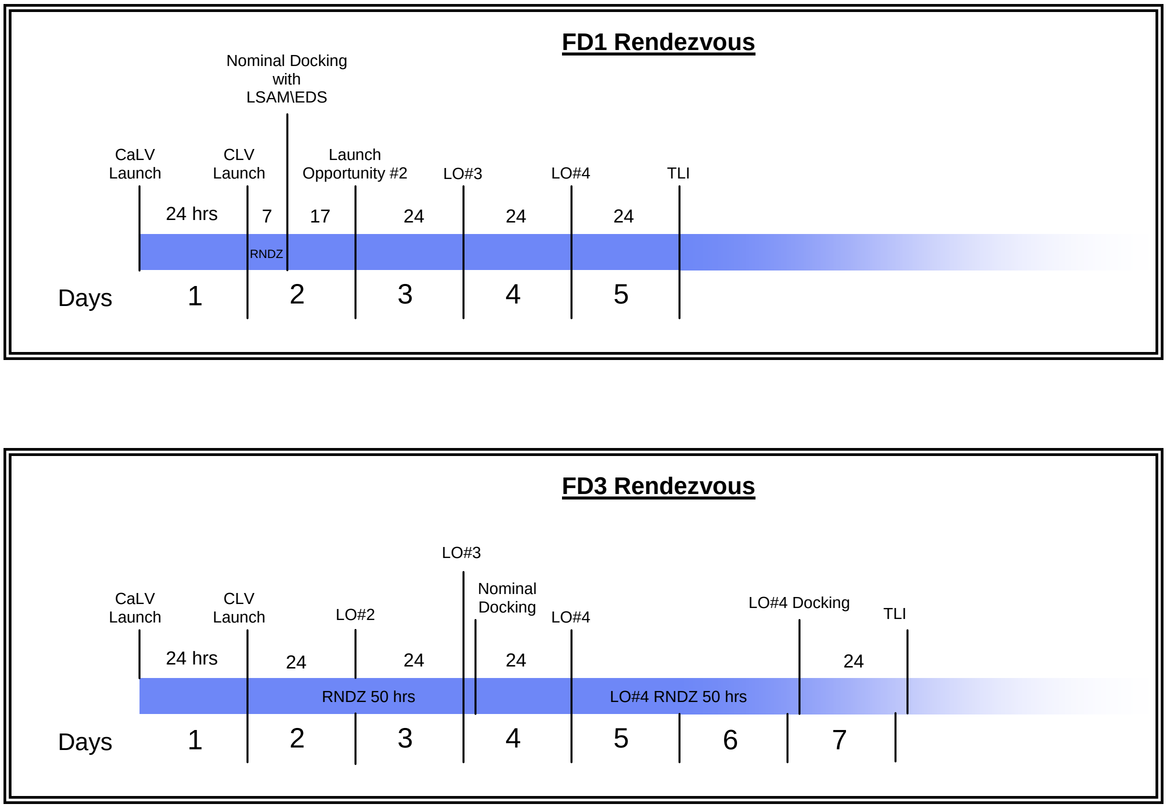

Figure ?????

TLI to LOI

As stated in Section 3.0 of the CARD requirement CA0352-HQ states that "The Constellation Architecture shall provide the capability to perform an expedited return of the crew from the surface of the Moon to the surface of the Earth in 120 hours (TBR-001-005) or less after the decision to return has been made." This requirement, combined with the global access requirement, demands a crew capability to depart any lunar landing site location before the nominal end of mission. The outbound (Earth to Moon) flight phase for a global access sortie mission employs the previously mentioned targeting scheme designed to minimize the required CEV on-orbit plane change, performed at any time during the lunar surface stay to align its orbit for an in-plane LSAM ascent and rendezvous. The ESAS work produced a globally-applicable lunar outbound targeting technique to support early analysis of global access sortie missions and preliminary vehicle sizing.

LOI Maneuver Sequence. During the Apollo program, the LOI maneuver consisted of a single burn. In the Constellation Program, the potential for large LOI wedge angle changes during lunar arrival (LOI) necessitates availability of a three-burn sequence (see Figure 6). The inclination and LAN targets associated with a particular landing site latitude and longitude are achieved at the conclusion of the LOI maneuver sequence on arrival to the LDO. The spacecraft achieves a LDO inclination and LAN target through a combination of the TLI and LOI burn, which is reinforced with "as-needed" with post-TLI Trajectory Correction Maneuvers (TCMs). This technique would allow the crew to perform an in-plane LSAM landing and be assured that the on-orbit plane change, executed in this case by the CEV to set up the LSAM for a near in-plane ascent, does not exceed a specified (minimized worst-case) value for either a nominal or early return scenario. Here, the phrase "near in-plane" refers to the fact that, while an in-plane launch to the CEV parking orbit could occur when it contains the landing site, the LSAM ascent may occur only when the CEV is in the proper phase location to effect a timely rendezvous sequence. Thus, the LSAM must execute a small amount of ascent/rendezvous plane change. This can be accomplished using ascent yaw steering or possibly a small out-of-plane maneuver during the rendezvous sequence.

The overall nominal design example for a maximum surface stay time sortie reflects a total (Earth launch to Earth return) mission duration of about 21.5 days with 7 days on the lunar surface. Assuming a pre-emplaced EDS and LSAM in LEO, the CLV launches the CEV into orbit where it docks with the EDS/LSAM. After about **3-days** time from CLV launch, the EDS executes a TLI maneuver targeted to a particular lunar orbit inclination and LAN realized at the conclusion of the LOI burn sequence. After a day of loiter for checkout prior to LSAM descent, the CEV/LSAM orbit contains the landing site, allowing for an in-plane LSAM descent to the surface. After approximately 6 days on the lunar surface, the CEV performs and orbit plane change in preparation for the LSAM ascent on surface day 7. The modified CEV orbit inclination and LAN provides for an in-plane nominal launch of the LSAM on day 7*. About a day after the LSAM ascent and rendezvous with the CEV, the CEV performs a TEI maneuver placing it on a 3.5 to 4.5 day flight to Earth. The Earth return flight time depends upon the Earth-Moon geometry at TEI-3 as well as the desired longitude of the entry interface state‡. A more detailed look at the mission design for the surface stay portion of the sortie mission from lunar descent through lunar ascent appears in Figure 7.

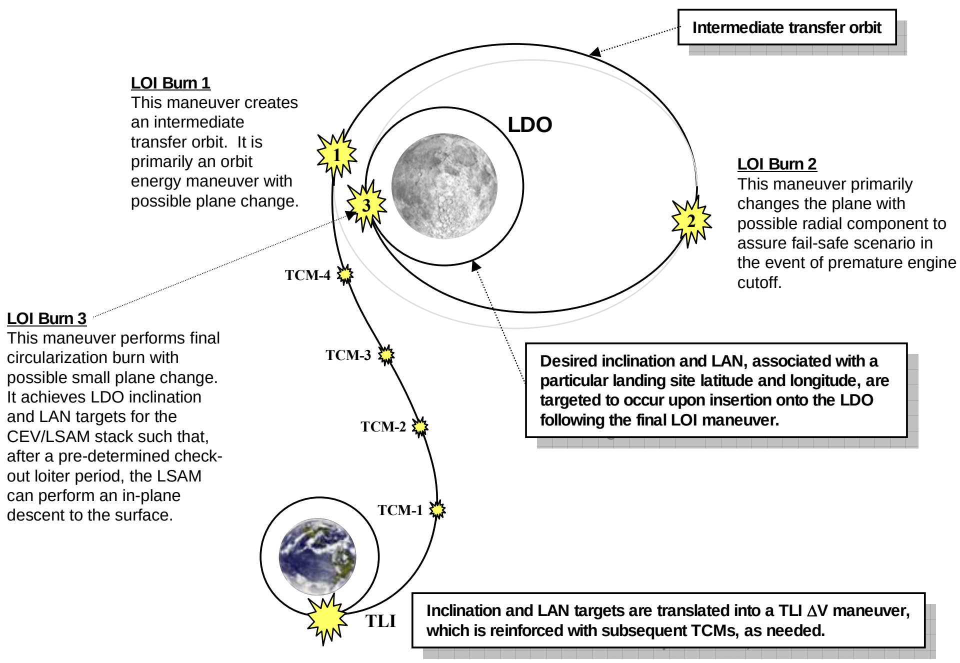

Figure 6 **LOI Maneuver Sequence.** Three-burn LOI maneuvers provide lunar capture to desired LDO inclination and LAN associated with a particular landing site latitude and longitude.

* Note that the LSAM is not required to perform a large ascent plane change due to the cooperative CEV on-orbit plane change maneuver. Generally however, the CEV will not be concurrently in the minimum rendezvous time phase location, at the time of LSAM launch, and in a plane containing the landing site. Thus, the LSAM will perform some yaw steering during the ascent flight phase.

‡ Note that the Earth return flight time also affects the latitude of Earth entry interface. However, the effect of flight time on EI longitude is much greater than it is on the latitude.

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Note that, in addition to the magnitude of the on-orbit plane change required to align the CEV for in-plane LSAM ascent, the LSAM performance and lunar launch windows are also affected by this sortie mission design (References 3, 4, and 5). The performance benefit of using an on-orbit plane change, as compared to a large ascent yaw steer, to effect this alignment between the CEV and LSAM was confirmed in a separate study (Reference 3).

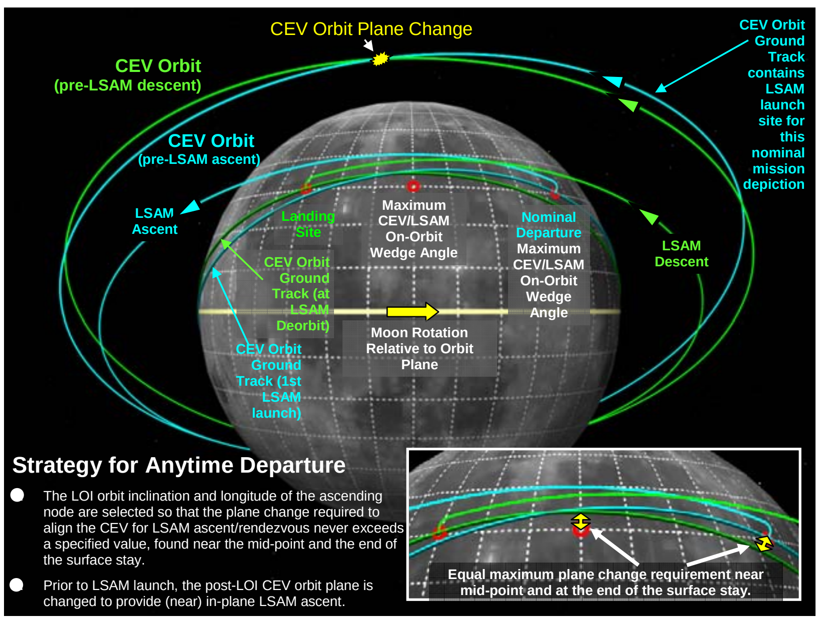

Figure 7 CEV Parking Orbit Prior To LSAM Descent And LSAM Ascent, Including On-Orbit CEV Plane Change.

Inclination and LAN Target Design for LOI. The LOI arrival inclination and LAN are an integral part of the overall mission design and relate the required on-orbit plane change (required to align the CEV for in-plane LSAM launch) to the surface stay time and landing site latitude. Once the location of the landing site

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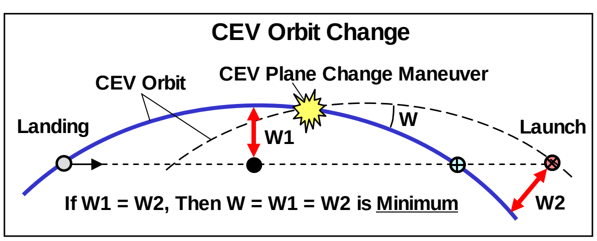
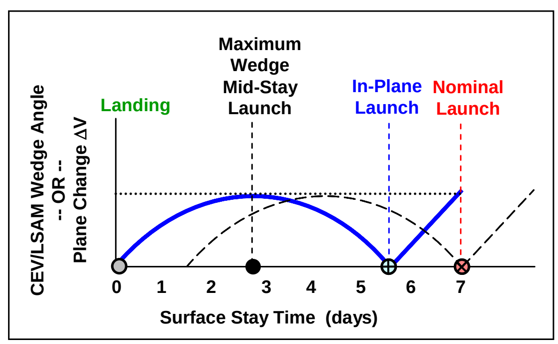
[‡] Note that the Earth return flight time also affects the latitude of Earth entry interface. However, the effect of flight time on EI longitude is much greater than it is on the latitude.

and duration of the lunar surface stay are selected for a given sortie mission, mid-fidelity trajectory processors such as NASA JSC's EXLX* ,calculated the cost to perform either a single-maneuver or three-maneuver burn sequence to place the CEV in the required LDO (i.e., achieve the inclination and LAN targets). EXLX computes both the TLI and LOI maneuver combination. Selectable minimizations of the TLI and LOI combination were employed in an attempt to minimize the overall initial mass in LEO. Note that the selection of the inclination and LAN at LOI completion also determines the inclination and LAN of the post-LSAM ascent orbit. Therefore, the arrival orbit (inclination and LAN) targets will affect the TEI ΔV requirement, since the post-CEV plane change orbit (supporting in-plane LSAM launch) is constrained.

Minimum CEV Plane Change Supporting In-Plane LSAM Ascent. The LOI inclination and LAN targets are dependent on a number of factors including: the lunar landing site latitude and longitude, surface stay time, loiter time after LOI and prior to LSAM deorbit, and the size of the low lunar orbit. The diagrams in Figure 8 reflect a technique employed to provide the minimum possible on-orbit CEV plane change (ascent plane change), while at the same time providing for anytime LSAM liftoff. This technique achieves a minimum plane change angle by launching the LSAM such that it intercepts the CEV orbit plane 90° downrange from the launch site (Reference 2). The left portion of Figure 8 shows the CEV orbit containing the landing site (solid blue curve) at the beginning of the 7-day surface stay. The solid blue curve of the right portion of the figure reflects an LSAM ascent, immediately after landing, to have zero wedge angle (or correspondingly zero descent/landing plane change ΔV requirement). As time progresses, the landing site moves to the right, relative to the CEV parking orbit. About three days after landing, the minimum attainable plane difference between the CEV parking orbit and a post-LSAM ascent/rendezvous has reached a maximum. This difference, shown of the left portion of the figure, as W1, is reflected on the right portion with a maximum wedge angle (and associated plane change ΔV). Approximately mid-way between the fifth and sixth day of the surface stay, the CEV orbit again contains the landing site and an in-plane LSAM ascent can occur this time. Thks is shown in the left portion of the figure and reflected in the right portion as a zero wedge angle. At the end of the 7-day surface stay, the landing site now becomes the launch site. The left portion of the figure shows the minimum attainable wedge angle between the landing/launch site and the CEV parking orbit to have a magnitude of W2. The CEV would have to perform a wedge angle of W2 degrees to contain the LSAM landing/launch site and provide an in-plane LSAM launch (per the left and right side of the figure, respectively). If the magnitude of the CEV plane change near the midpoint (W1) and end (W2) of the 7-day mission is equal, then the maximum plane change is minimized. With this technique, the on-orbit plane change required to provide an in-plane LSAM ascent never exceeds W = W1 = W2 throughout the entire surface stay. If the CEV carries enough propellant to accomplish this wedge angle (W), then it will possess the ability to set up an in-plane LSAM ascent anytime during the lunar surface stay*.



Orbit Plane Change



CEV/LSAM Wedge Angle/Plane Change ΔV

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* EXLX represents a set of Earth-Moon trajectory tools. The "X" in the EXLX acronym can represent "O" for "Orbit" or "S" for "Site", depending upon the tool used. In this case, the tool used was the Earth Orbit to Lunar Orbit (EOLO) tool.

* Note that this scenario, which evolved from the ESAS (Reference 1), uses the CEV to perform the on-orbit plane change. However, this on-orbit maneuver could also be performed by the LSAM if it possessed enough propellant.

Figure 8 **CEV Orbit Plane Change To Set Up LSAM In-Plane Ascent.** Comparison of CEV orbit plane change and corresponding magnitude of plane change wedge angle and associated plane change ΔV for an example 7-day lunar surface stay.

This technique applies to any surface stay time. The example used in this report reflects the maximum surface stay time of seven days. This technique could also be applied to surface stays greater than seven days, though it tends to be less effective as a mission design strategy for longer surface stay times (about 10–11 days or longer) as the cost of the CEV orbit plane change can grow, significantly. If a long duration sortie mission were desired along with an early return capability, an alternate mission design strategy or a constrained landing site location (e.g., near pole or near equator) may be more appropriate.

Once the CEV arrival inclination and the associated maximum on-orbit wedge angle are computed, the corresponding LSAM ascent orbit inclination can then be determined. Subsequently the LSAM performs and in-plane ascent and rendezvous with the CEV. The magnitude of the on-orbit plane change is determined by the time, during the lunar surface stay, that it is executed. The plane change occurs at the line of common node of the current CEV orbit and the anticipated LSAM post-ascent orbit. This placement of the maneuver minimizes the orbit plane change ΔV requirement and, in the general case, results in an orbit node shift and inclination shift to place the CEV in the LSAM ascent target orbit.

The difference between the CEV inclination at LSAM descent and the corresponding LSAM ascent inclination, shown in Figure 9, is minimized for equatorial or polar orbits and maximized in the mid-latitude regions. The CEV parking orbit has an inclination slightly higher (posigrade orbits) or lower (retrograde orbits) than the landing site latitude. This provides two encounters during a surface mission where the CEV orbit contains the landing site.

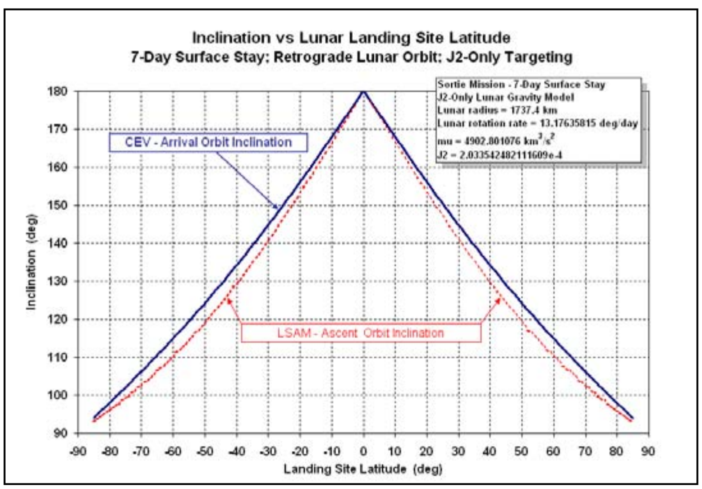

Figure 9 **CEV And LSAM Inclination Targets Vs Landing Site Latitude For An Example 7-Day Lunar Surface Stay.**

For a J_2-only gravitational environment, the inclination is independent of landing site longitude. Figure 10 shows that the analytical (J_2-only lunar gravity) approach reveals a maximum CEV on-orbit plane change* requirement of about 5.9°. For a 100-km circular lunar orbit altitude, the associated on-orbit plane change would cost about 168.2 m/s.

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* Note that the minimized ΔV plane change represents an on-orbit wedge angle rotation which will, in general, change both the inclination and the LAN of the initial CEV orbit so that it is aligned with the LSAM inclination an LAN targets for in-plane ascent.

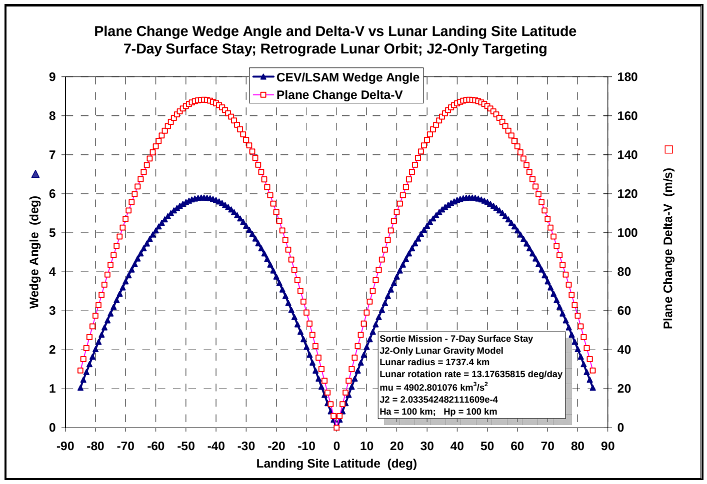

Figure 10 **Plane Change Wedge Angle And Associated ΔV As A Function Of Lunar Landing Site Latitude, For An Example 7-Day Surface Mission.**

Lunar Arrival and Departure LAN. Once the CEV parking orbit inclination has been established, the LAN associated with a given landing site longitude can be determined to support an in-plane LSAM landing. Computation of the CEV parking orbit LAN is dependent on several factors including: lunar surface stay time, landing site latitude and longitude, and the associated CEV parking orbit inclination. The arrival orbit inclination may be retrograde (90° to 180° inclination) or posigrade (0° to 90°). For crewed missions, the retrograde orbit is preferred as it provides a lower propellant requirement for outbound aborts back to Earth. While powered descent and ascent are more expensive for retrograde orbits than posigrade, this cost is small (Reference 7). Powered-flight descent from a full retrograde (180°) orbit has about a 10 m/s additional ΔV cost as compared to a full posigrade (0°) descent, for an overall powered ΔV cost of around 2000 m/s. LAN calculations for a CEV arrival and LSAM departure orbit are shown in Figure 10. Practically speaking, for a lunar arrival LAN calculation, the mission planner must offset the arrival LAN (using the lunar rotation rate and orbit precession) to allow for post-LOI loiter time for the crew to reconfigure and checkout the spacecraft for powered lunar descent. LOI inclination and LAN targeting must accommodate these offsets. LSAM ascent inclination and LAN targets, at the end of the surface stay, must also incorporate non-spherical lunar gravitational affects.

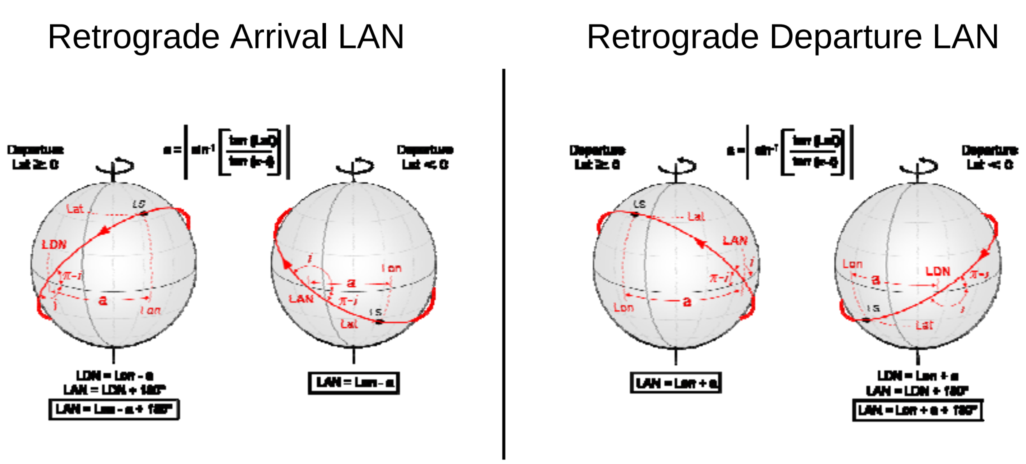

Figure 10 **Retrograde Lunar Arrival and Departure LAN Calculations.**

Improved Targeting Solutions. The analytic solutions apply only to CEV flight in a J_2-only gravity field. Computing a set of CEV arrival inclination and LAN targets via the analytic technique, and then propagating the trajectory with a higher order lunar gravity model, can result in a higher than analytically predicted CEV plane change requirement. A more detailed numerical optimization employing a 50x50 resolution of the LP150Q lunar gravitational model results in a slightly higher wedge angle of about 6.2° with an associated plane change cost of 177 m/s **(Reference 8)**. In both the analytic and numerically optimized cases, the maximum CEV plane change requirements occurs in the mid-latitude regions near 45° latitude.

The targeting goal of the orbital inclination and LAN is to minimize the any-time abort CEV plane change. The analytical solution however was limited to a J_2-only gravity field. Bringing the targets into a LP150Q 50×50 gravity field requires a small retargeting of the solutions to re-establish the desired conditions. Figure 14 shows the original solution (red) propagated in the LP150Q field, and the retargeted solution (blue).

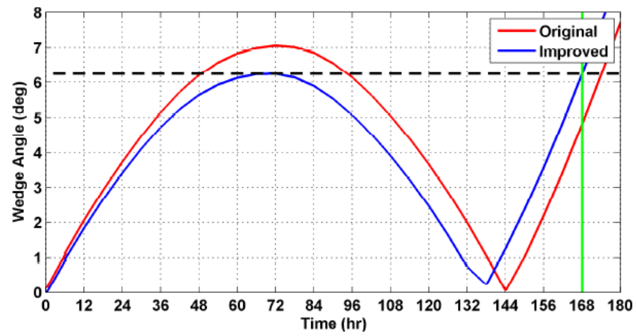

Figure 14 Improved solution (Landing site: 46°N 10°E) for a 7-day lunar surface stay.

By scanning the globe and re-targeting the solutions we see that the worst case CEV plane change is 6.2° at a latitude of 46°N 10°E, as seen in Figure 15.

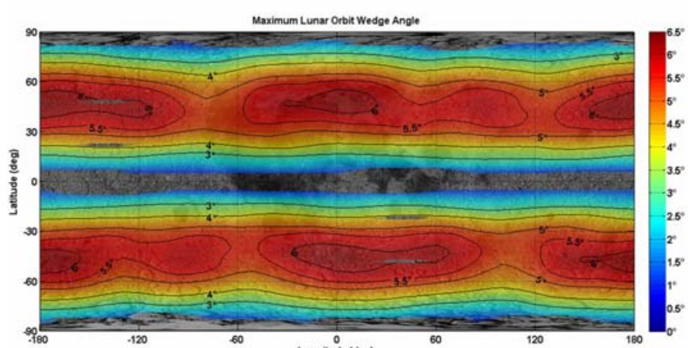

Figure 15 Global scan of worst case wedge angle (improved solution).

As expected, the northern and southern hemisphere solutions are symmetric. Table 3 summarizes some of these solutions.

Table 3

MAXIMUM CEV PLANE CHANGE AND ASSOCIATED LANDING SITE LATITUDE AND LONGITUDE

		Pre-Improv		Post-Improv		
Lat	Lon	Worst Case	7 day	Worst Case	7 day	Note:
46°	10°	7.1°	4.8°	6.2°	6.2°	Maximum abort wedge angle
33°	125°	4.7°	6.9°	5.3°	5.3°	Maximum nominal departure wedge angle
45°	−125°	7.0°	4.5°	6.1°	6.1°	Maximum difference (worst abort to nominal departure)
32°	125°	4.6°	6.9°	5.3°	5.3°	Minimum difference (worst abort to nominal departure)

Powered Descent and Abort to Orbit Considerations

A 24-hour coast period follows LOI, allowing for operations activities such as vehicle checkout, crew sleep cycles, and preparation of the LSAM for descent and landing. After this period, the CEV separates from the LSAM. One revolution later, the LSAM performs a de-orbit burn, placing itself in an elliptical transfer orbit. The periapse of this orbit should be chosen to minimize the overall delta-V while still maintaining an adequate safety margin for a failed PDI and terrain clearance during the descent. Recent work has shown that a periapse of 50,000 ft (15.24 km), also used during Apollo, appears to be the best choice. Near periapsis, following an approximately one hour coast, the LSAM ignites the descent engines and steers to a soft landing. The Powered Descent burn lasts only a few minutes. Figure 11 shows the flight sequence.

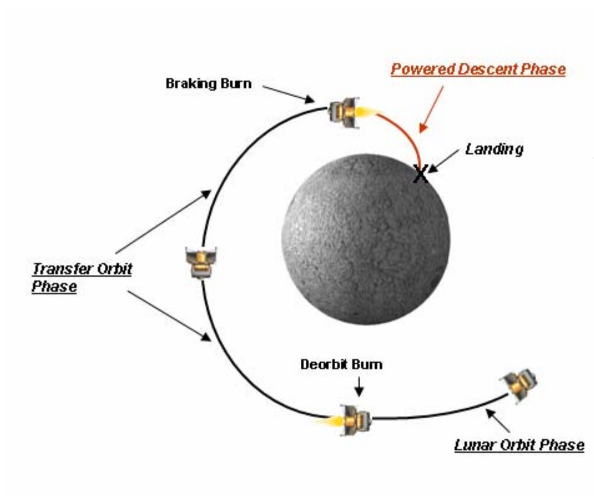

Figure 11 Lunar Descent Scenario

For a nominal LSAM de-orbit and descent, there is no impact to the CEV, since the LSAM separates prior to the de-orbit burn. However, in case of a failed PDI, or an abort back to orbit during the descent, the CEV may be required to adjust its orbit to rendezvous with the LSAM. In a relative motion sense, at the time of an early failed PDI, the LSAM will be ahead and below the CEV, butessentially in the same orbital plane as the CEV. At the time of mid- to late-PDI aborts, the LSAM will be below and behind the CEV, though again essentially in the same orbital plane as the CEV. For a rendezvous following a PDI abort, the LSAM ascent vehicle is the active vehicle and it carries the capability for completing a nominal rendezvous sequence. Typically, the descent abort to orbit scenarios do not require the CEV to perform the rendezvous, but the capability should be preserved in case of further problems with the LSAM.

Orbit Maintenance / CEV Plane Change Considerations

Need introductory words … The surface stay involves a continuous program of CEV orbit maintenance …. blah blah To investigate the orbit maintenance costs, the CEV was propagated using the LP150Q gravity model with a 50 × 50 field for a duration of 8 days; 1 day to account for the post LOI check-out time prior to LSAM descent, and 7 days for a nominal mission. The periapse raise maneuver was performed when the CEV altitude exceeded a minimum 50 km altitude. At the conclusion of the mission, a Hohmann transfer was used to return the CEV to a 100 km circular altitude.

A scan through a range of inclinations and right ascensions of the ascending node (RAAN) show a maximum budged as high as 32 m/s as seen in Figure 12.

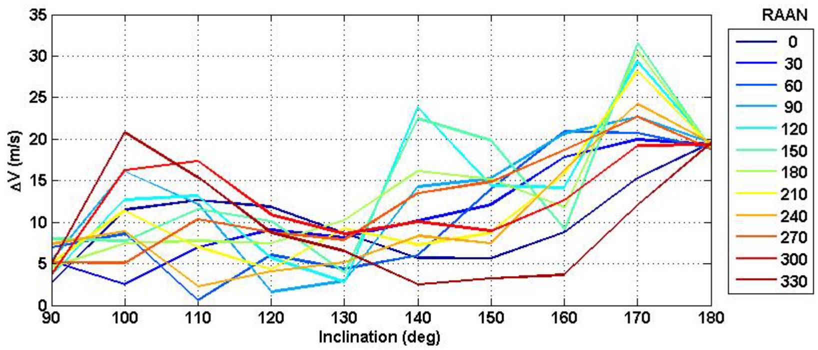

Figure 12 ΔV cost as a function of inclination and RAAN.

Although the maximum ΔV cost is as high as 32 m/s, the distribution is not normal as can be seen in Figure 13.

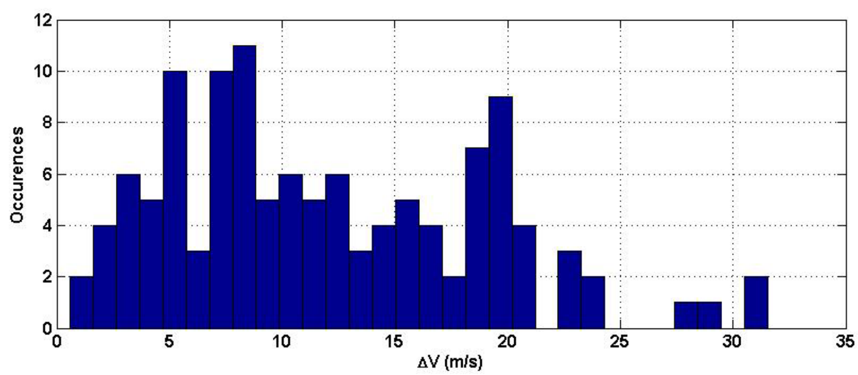

Figure 13 ΔV distribution.

This is a very cursory look at the altitude maintenance costs for a 7-day lunar mission. Additional analysis can be performed to improve burn strategy and investigate potential benefits of other altitude constraints.

LSAM Ascent/Rendezvous

Prior to the nominal LSAM ascent from the lunar surface, the CEV will perform a plane change maneuver that will allow the launch site to be in the CEV orbital plane at the nominal liftoff time. The

LSAM will then lift-off at the in-plane time and complete the ascent and rendezvous sequence. However, in case of an early (or late) lift-off, the CEV will not be in an orbit that passes directly over the launch site.

This orbit misalignment can be removed in one of two ways: (1) yaw steering during the LSAM ascent, or (2) additional plane change cost, primarily to be provided by the CEV. Figure 16 shows a cost comparison of the two methods over a 7-day stay at Aristarchus Crater. The yaw steering is more expensive in nearly all cases. A similar result is seen at a majority of other latitudes. The primary reason for this result is that it is relatively inexpensive at the Moon to change inclination during ascent, but relatively expensive to change the longitude of the ascent node (LAN). Since the Moon rotates so slowly, the CEV's LAN can be very different than the LSAM in-plane LAN, which drives the delta-V penalty very high.

As a result, it is recommended that in a majority of abort cases, the CEV perform a plane change maneuver. The yaw steering capability should be used for small out-of-plane steering corrections, but not for anytime abort considerations. For reference, Figure 17 shows the yaw steering costs around the in-plane point, for a particular inclination. The frequency of a favorable phase angle liftoff time is also included on the figure.

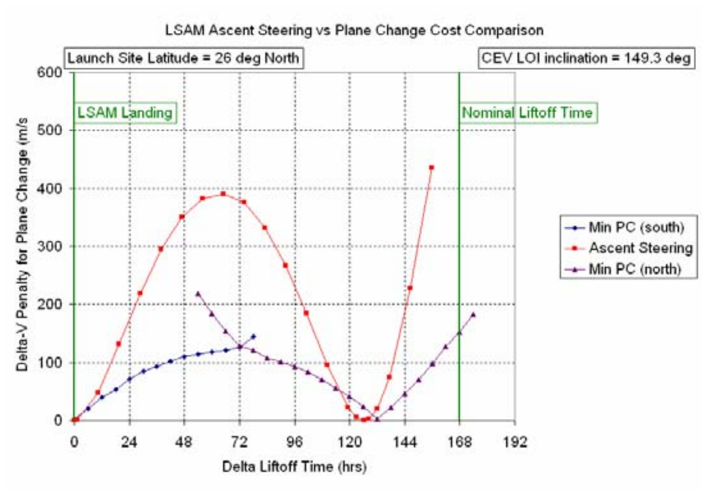

Figure 16 LSAM Yaw Steering vs Plane Change Cost Comparison

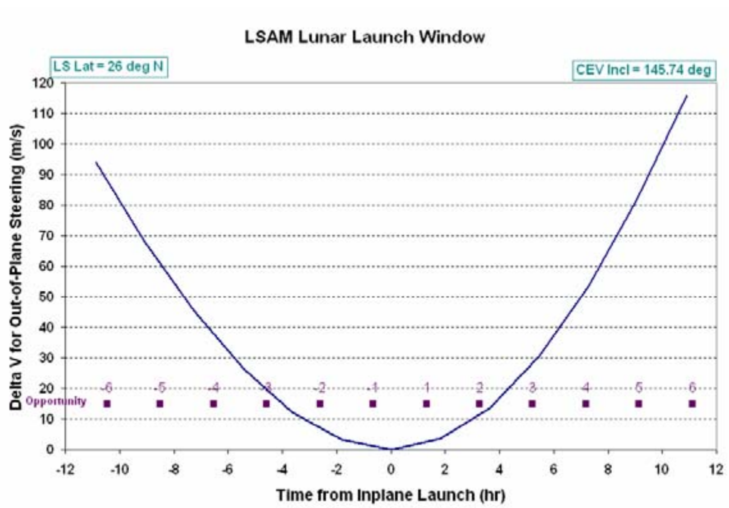

Figure 17 A Typical LSAM Yaw Steering Cost

TEI to EI

The Moon to Earth trajectory phase begins with either a 1 or 3-impulse TEI and ends at an Earth EI point. The trajectory is designed to minimize the TEI ΔV while achieving the desired entry conditions at Earth. For vehicle design and sizing purposes, the worst case Earth-Moon geometry should be used. This geometry consists of: (1) Moon at perigee, (2) Moon at maximum inclination, and (3) Moon at nodal crossing (i.e., the ascending node). To add to this worst case geometry, the transfer trajectory should be designed with: (1) a lunar parking orbit normal to the departure V_∞ vector, (2) a transfer orbit inclination of 90° (i.e., arrival azimuth of 0°), and (3) the shortest transfer time.

The 3-impulse TEI initiates a departure from a 100 km circular orbit at the Moon. The first impulse transitions the CEV from a 100 km circular orbit to a 100 x 15925 km elliptical orbit (orbital period of about 24 hours). The second impulse executes the required plane change. The third impulse sends the CEV towards Earth. The trajectory's flight time from the TEI's third impulse to the Earth entry interface can be varied from 3 to 4 days. The 24 hour variation in flight time enables targeting a specific longitude.

Earth entry interface targets correspond to either water or land landings. For land landings, more entry interface target parameters must be satisfied. More parameters are required because primary and alternate landing sites are at fixed locations and have fixed landing areas. Water landing sites are variable sized areas located in the vast oceans. Earth entry interface parameters that are used as targets include the latitude, longitude, flight path angle, azimuth angle, and entry speed.

Analysis of the return trajectory was performed in two different ways. The first way was to evaluate individual trajectories over a metonic cycle (19 years) of the Moon orbiting the Earth. The range of time was chosen to be 1 January 2018 to 1 January 2037. The second way was to concentrate on a particular date. Figure 18 shows a subset of dates that indicates the worst case TEI ΔV date falls on 18 September 2024. This worst case corresponds to the worst case Earth-Moon geometry and worst case transfer trajectory previously discussed.

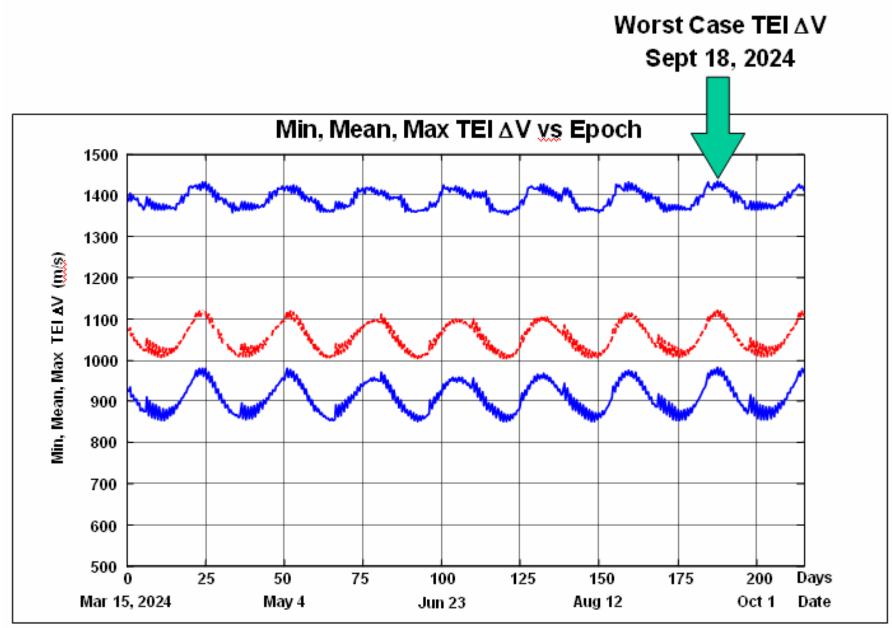

Figure 18 **TEI ΔV vs. Epoch: South to North Polar Return Targeted For the Longitude of Edwards Air Force Base**

Figure 19 shows a contour map of the TEI ΔV corresponding to lunar landing site latitudes and longitudes. The highest TEI ΔV values are confined to specific regions on the lunar surface. These regions may change size and shift location with time. If the CEV is not designed to accommodate the worst case TEI ΔV, then full lunar access may not be feasible. Past analyses show that the TEI ΔV can be reduced by introducing a variable period (up to 7 days) of loitering in orbit; this action produces more favorable departure geometries.

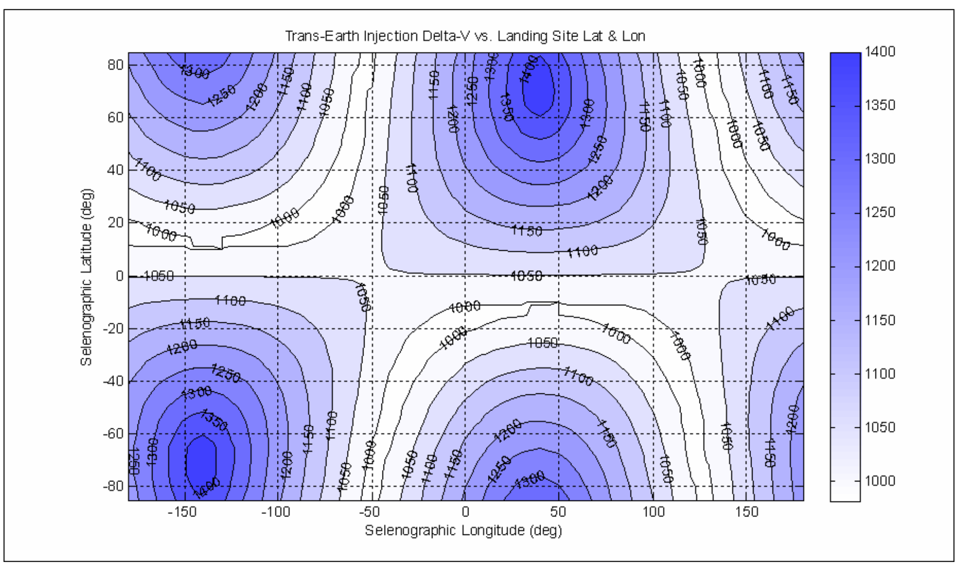

Figure 19 **TEI ΔV vs. Lunar Landing Site: 18 September 2024 Worst Case Geometry**

CONCLUSION

Need about a half a page for this.

End conclusion.

ACKNOWLEDGMENT

The authors wish to acknowledge Sam Wilson for his help with the techniques described in this report. Sam Wilson's astrodynamics expertise, insight, and common sense have proved invaluable. His hand can be found in many aspects of the lunar mission design as it exists today.

NOTATION

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