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ABORT OPTIONS FOR HUMAN LUNAR MISSIONS BETWEEN EARTH ORBIT AND LUNAR VICINITY

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Apollo mission design emphasized operational flexibility that supported premature return to Earth. However, that design was tailored to use expendable hardware for short expeditions to low-latitude sites and cannot be applied directly to an evolutionary program requiring long stay times at arbitrary sites. This work establishes abort performance requirements for representative on-orbit phases of missions involving rendezvous in lunar-orbit, lunar-surface and at the Earth-Moon libration point. This study submits reference abort delta-V requirements and other Earth return data (e.g., entry speed, flight path angle) and also examines the effect of abort performance requirements on propulsive capability for selected vehicle configurations.

NOMENCLATURE

CEV	Crew Exploration Vehicle
CM	Command Module
CSM	Command and Service Module
EOD	Earth Orbit Departure
L1	Earth-Moon cis-lunar libration point
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LOD	Lunar orbit departure
LOI	Lunar orbit insertion
LOR	Lunar orbit rendezvous – mission mode
LPR	Libration point rendezvous – mission mode
LSAM	Lunar surface access module – it comprises the descent and ascent lunar stages
SM	Service module
TLI	Trans Lunar Injection

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INTRODUCTION

The Exploration Systems Architecture Study (ESAS) [Ref. 5] developed at NASA is serving as the source for subsequent performance and mission design requirements for the lunar Crew Exploration Vehicle (CEV) program. Two key architecture decisions, pertinent to this paper, arose from that study. First, an all-up (or tandem) configuration was selected as opposed to a convoy configuration. The tandem configuration is similar to that of the Apollo program, where the Command and Service Module (CSM) attached to the Lunar Surface Service Module (LSAM) departs the Earth for the outbound transfer, whereas in the convoy configuration the CSM is transferred alone to rendezvous with a pre-emplace LSAM in lunar orbit. Second, the only Lunar Orbit Rendezvous (LOR) is to occur after the lunar surface stay and prior to Earth return.

Even though the convoy mission required a smaller overall initial mass in Low Earth Orbit (LEO), the all-up architecture presents several advantages over the proposed convoy option. First, its vehicle abort capability exceeds that of the convoy configuration. In particular, a tandem vehicle with three available independent propulsion systems –Service Module (SM), and LSAM ascent and descent stages– presents a more complete Earth return coverage for declared aborts following Trans-Lunar Injection (TLI). Second, the availability of multiple propulsion stages in a tandem vehicle represent an inherent functional redundancy to execute an abort to Earth return that is absent in a convoy configuration. If one or more propulsion stages become inoperable, it may still be possible to use the remaining propulsion systems to execute the abort (with a possible relaxation of the return trip time capability). On the other hand, a failed SM in the convoy configuration on the outbound lunar transfer trajectory would be catastrophic, except possibly in the unique case of a nominal free-return flyby trajectory. Third, the tandem configuration does not require an outbound rendezvous between the CSM and LSAM at lunar arrival, as is the case with the convoy configuration, thus eliminating a critical space maneuver sequence. Fourth, in case of an emergency, the tandem configuration provides a backup habitat for the crew. It is important to note that the Apollo 13 mission benefited from all the advantages described above. Following an explosion in the SM, the crew survived in the lunar lander whose propulsion system was used to perform the abort to Earth return. Five, due to launch vehicle size limitations in ESAS, two launches are required to support the all-up Earth departure. This implies additional crew safety since the crew would not be launched into space onboard a massive and consequently potentially more dangerous rocket. The operational drawback during flight derived from having two launches is the additional Earth rendezvous maneuver between the CSM and the LSAM which would be equivalent to the one that took place in the Apollo missions when the CSM had to separate, turn around and dock with the lunar module.

The LOR architecture that emerged from the ESAS (Figure 1) employs an Earth Departure Stage (EDS) to perform the TLI maneuver, placing the stack CSM/LSAM on a trans-lunar trajectory with the EDS jettisoned after TLI, as in the Apollo missions. At lunar arrival, a Lunar Orbit Insertion (LOI) maneuver delivers the CSM/LSAM into a Low Lunar Orbit (LLO). Unlike the Apollo program, which used its service module to execute LOI, this architecture uses the descent stage of the LSAM to perform LOI. Subsequently, the LSAM descents to the surface and, after a given surface stay time, only the ascent stage ascends back to lunar orbit to rendezvous with the awaiting CSM.

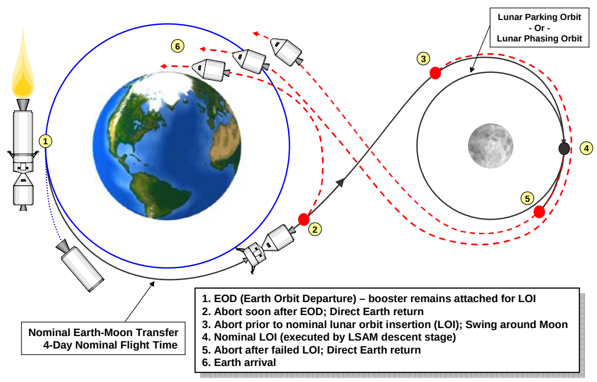

Figure 1 Abort to Earth return from nominal LOR lunar transfer trajectory.

In addition to LOR, and based on previous interest in the L1 libration point as a potential staging point for missions to the moon and possibly elsewhere [Ref. 2], Libration Point Rendezvous (LPR) mission abort cases were also examined in this work. As described in the Lunar Architecture Focused Trade Study Final Report [Ref. ?], which examined a number of lunar based exploration architectures, the LPR mission employed two dedicated vehicles (Figure 2). The first vehicle carries the crew from LEO to the L1 vicinity, and once at L1, the crew transfers into a pre-emplaced lander. The lander departs L1 bound for the lunar surface via an intermediate low lunar phasing orbit.

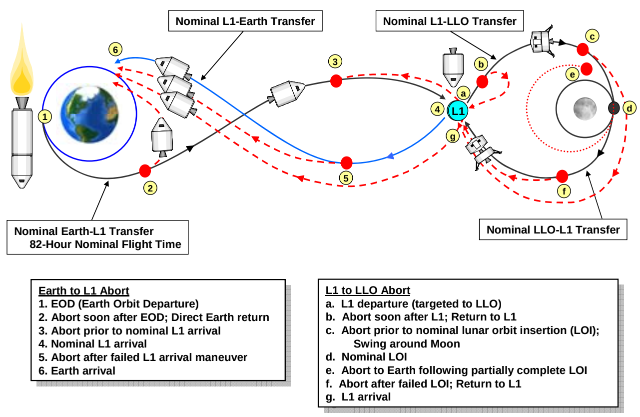

**Figure 2 Abort to Earth return from nominal Earth-L1 transfer trajectory.
Also shown is the abort from a nominal L1-Moon transfer, back to L1.**

Whereas this work presents abort ΔV requirements for LOR-style and libration point missions, vehicle performance results are only shown for the former since it constitutes the preferred architecture for the future manned lunar missions. Performance cost results and other associated parameters, such as Earth return entry speed and flight path angle, for an abort from the trans-lunar trajectory are presented. This includes an abort to Earth from trans-lunar trajectories at times prior to and after lunar encounter, and Earth return abort data for cases of fully or partially failed LOI.

PERFORMANCE STUDY. ΔV REQUIREMENTS.

Abort ΔV requirements for the cases considered in the study (LEO-LLO and LEO-L1-LLO) were obtained using Copernicus[4], a multi-body trajectory design and optimization tool developed by C. Ocampo. All aborts consisted of a single maneuver that has been optimized to minimize the DV while satisfying the associated constraints.

The chosen epoch for the abort ΔV requirements is based on a maximum 57.1º lunar arrival maximum wedge angle. This wedge angle arises from a worst case geometry and TLI epoch that combines the 28.5º Earth departure parking orbit with the maximum lunar inclination (about the Earth equator) of 28.6º for a coplanar TLI.

Comment [j1]: Jerry, do you remember why the epoch is different for the LEOLLO and LEOL1 cases?

Two main scenarios have been considered:

1. <u>LEO to LLO.</u>
   The parameters and conditions of this case can be found in Table 1. The nominal transfer time is four days, the initial orbit around the Earth is a 407x407 km altitude orbit with a 28.5deg inclination and the parking orbit around the Moon is a 100x100 km altitude. The inclination of the parking orbit will be a parameter is this study and it spans from 0 to 180deg. The Earth return transfer time after the abort maneuver is executed will spans from hours to several days. Finally, the only condition for the Earth return is a vacuum perigee altitude of 38.24 km.

   Within this scenario two subcases were studied:
   - Total failure (0% burn).
     In this case the LOI is never carried out. The time when the abort maneuver is executed spans from 7 to 140h after the TLI maneuver; in this way we are taking into account cases where the abort maneuver is executed before and after the nominal LOI.
   - Partial burns of 25%, 50% and 75%.
     The magnitude of the LOI maneuver was 25%, 50% or 75% of the nominal one. It is assumed that there is no error in the maneuver direction. The time when abort maneuver is executed spans from 2 to 50h after the nominal LOI.

Table 1: Parameters for the LEO to LLO case

Parameter	Value	Units	
Epoch	2006/10/1/15/6/0.00	UTC	
TOF	4	day	96 h
Departure from Earth. Orbital elements			
Semi Major Axis Eccentricity Inclination Longitude of the asc. node Argument of periapse True anomaly	6785.1 0 28.5 free free free	km deg deg deg deg	407 x 407 km altitude
Arrival to Moon. Orbital elements			
Semi Major Axis Eccentricity Inclination Longitude of the asc. node Argument of periapse True anomaly	1837.4 0 0, 45, 90, 135, 180 free free free	km deg deg deg deg	100 x 100 km altitude retrograde orbit Parameter for trade study
Gravitational field			

Earth Moon	398600.432893 4902.800582	km^3/s^2 km/s^2	
Time when the abort maneuver is executed	7, 14, 21, …, 140 2, 4, 6, …, 50	h h	Only for the 0% case (complete failure) For the 25%, 50% and 75% cases (partial failure) this time is after the nominal LOI (96h + epoch)
Return time to Earth after abort	7, 14, 21, …, 140 8, 14, 20, 140	h h	Only for the 0% case (complete failure) For the 25%, 50% and 75% cases (partial failure)
Earth return parameters			
Vacuum perigee altitude	38.24	km	
Entry interface altitude	121.92	km	400,000 ft where entry speed and flight path angle are calculated

Comment [j2]: Confirm these numbers for the partial burn abort time

2. <u>LEO to LLO through L1.</u>
   The parameters and conditions for this case can be found in Tables 2 and 3. This case can be divided into two cases:
   - LEO to L1
     For the nominal transfer, the initial orbit around the Earth is the same as in the LEO to LLO case and the transfer time is now 82h (compared to the 96h of the previous case).
     The Earth return conditions are the same as above and only the total failure (0% burn) has been carried out.
   - L1 to LLO
     The nominal transfer time is 60h and the orbit around the Moon and the Earth return conditions are the same as in the LEO to LLO case. Only the total failure (0% burn) has been carried out.

Table 2: Parameters for the nominal LEO to L1trajectory

Parameter	Value	Units	
Epoch	2006/10/2/18/0/0.00+82h	UTC	
TOF	3.4166666666	day	82 h
Departure from Earth. Orbital elements			
Semi Major Axis Eccentricity Inclination Longitude of the asc. node Argument of periapse True anomaly	6785.1 0 28.5 free free free	km deg deg deg deg	407 x 407 km altitude
Arrival to Moon vicinity	Earth/Moon L1 point at 82h + epoch		
Gravitational field			
Earth Moon	398600.432893 4902.800582	km^3/s^2 km/s^2	
Time when the abort maneuver is executed	6, 12, 18, …, 144	h	For the 0% case (complete failure)
Return time to Earth after abort	6, 12, 18, …, 144	h	For the 0% case (complete failure)
Earth return parameters			
Vacuum perigee altitude	38.24	km	
Entry interface altitude	121.92	km	400,000 ft where entry speed and flight path angle are calculated

Table 3: Parameters for the nominal L1 to LLO trajectory

Parameter	Value	Units	
Epoch	2006/10/1/15/6/0.00	UTC	
TOF	2.5	day	60 h
Departure from Moon vicinity	Earth/Moon L1 point at 82h + epoch		
Arrival to Moon. Orbital elements			
Semi Major Axis Eccentricity Inclination Longitude of the asc. node Argument of periapse True anomaly	1837.4 0 0, 10, 20, …, 180 free free free	km deg deg deg deg	100 x 100 km altitude retrograde orbit Parameter for trade study
Gravitational field			
Earth Moon	398600.432893 4902.800582	km^3/s^2 km/s^2	
Time when the abort maneuver is executed	6, 12, 18, …, 98	h	For the 0% case (complete failure)
Return time to L1 after abort	6, 12, 18, …, 98	h	For the 0% case (complete failure)

Comment [j3]: confirm

Overall, 16,050 individual optimized abort trajectories were used in the study. Additional cases were generated, but only used to tune the optimizer for best results.

Comment [j4]: We should put all the results in a website and so people can download them. We have to put the link right here.

RESULTS AND DISCUSSION

Libration Point Rendezvous Mission. The LPR mission (figure 2) is split into two phases: the LEO to L1 phase and the L1 to LLO phase.

The abort ΔV requirement for this flight phase, in which the CSM carries the crew from LEO to L1 for rendezvous with the awaiting LSAM, are shown in figure 3. The nominal LEO to L1 transfer takes 82 hours (3.42 days) and aborts after the nominal L1 arrival time assume a completely failed L1 insertion maneuver. For a given abort time, the Earth return abort ΔV cost becomes large for very short Earth return times, on the order of a day or less. This abort cost is reduced significantly for all abort times if sufficient time (on the order of 2 days or more) is allowed for Earth return. Further, there is generally an Earth return time for a given abort time that produces the minimal abort ΔV. This condition correlates approximately to a fixed value of 7.5 days as the result of the addition of the abort and Earth return times. This fixed value of 7.5 days occurs because the spacecraft is practically following the same trajectory in all these cases. If we take the nominal case of 3.4167 days and no L1 insertion maneuver is performed then an optimal correction maneuver can make the spacecraft return to the Earth with the specified vacuum perigee in about 4.1days (so the total time is approximately 7.5 days). Since this optimal correction maneuver is so small (in the order of 20m/s) this will be close to the optimal return trajectory in case of abort (ORTCA). If the abort maneuver has to be executed before or after the nominal L1 insertion time then an optimal correction maneuver can be applied so the trajectory follows practically the ORTCA. So for each abort time we can find an optimal maneuver that practically follows the ORTCA that is why the resulting total time is approximately 7.5 days. If the nominal trajectory transfer time (3.5 days) changes then the ORTCA and total time (7.5 days) will change accordingly.

Comment [j5]: The DV is never 0 is small but not zero. 17, or 20 m/s

Comment [j6]: Confirm

Chart with DeltaV Req. Contour lines.- Mark nominal arrival time and DeltaV = 0 line (as contour)

After the crew arrives at L1 and docks with the awaiting LSAM, the crew transfers into the LSAM for the second phase of the mission, specifically, from L1 to LLO at a specified inclination. This part of the transfer carries a nominal flight time of 60 hours (2.5 days). Statistical data for this flight phase as a function of lunar arrival inclination are shown in figure 4. These data represent the maximum, mean, and minimum abort ΔV requirement for all selected abort and Earth return times, for each lunar inclination target. In general, the abort ΔV cost increases with increasing lunar orbit inclination. The maximum abort ΔV is about 5 times that of the mean, which tends to cover Earth return aborts for all abort times assuming that there is sufficient time for the return flight (e.g., assuming there are no consumables or power related issues that would drive the crew to a fast return time).

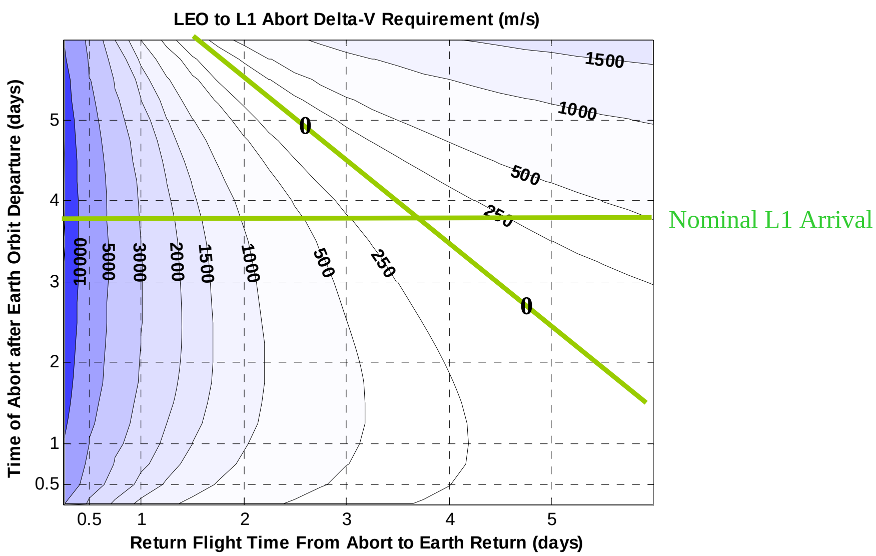

Figure 3 **Abort ΔV performance requirement for a nominal LEO to L1 transfer.**

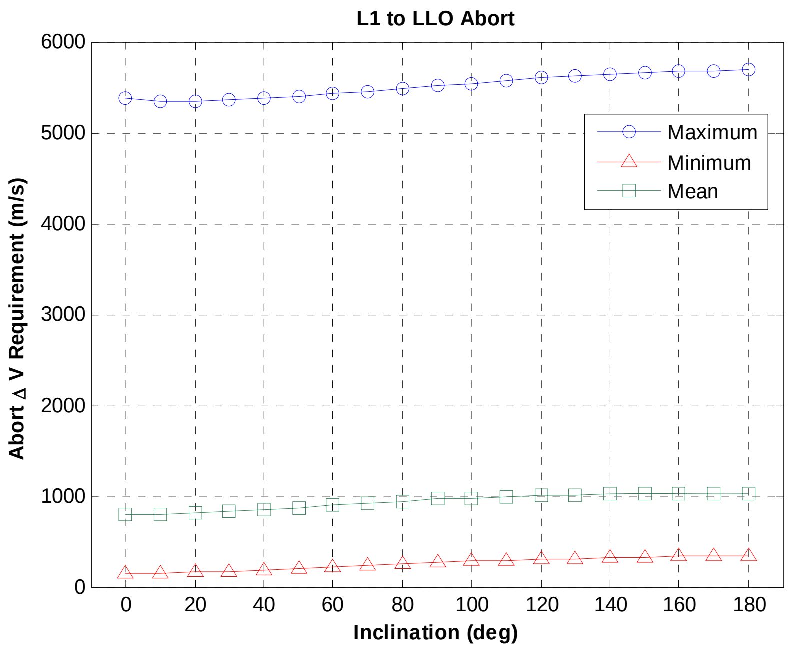

Figure 4 **Maximum, minimum and mean overall abort ΔV performance requirement, as a function of lunar orbit inclination, for a nominal L1 to LLO transfer.**

<u>LOR Mission Abort ΔV requirement</u>

For a LEO to LLO nominal mission, the abort ΔV requirement is generally larger for smaller lunar arrival inclinations (see Figure 5). The maximum abort ΔV corresponds to the minimum Earth return time. The data in Figure 5 show the maximum, mean, and minimum ΔV requirement to return a spacecraft to Earth following an abort. Note that the mean abort ΔV requirement for a given inclination is significantly lower than the maximum for that inclination. This reflects a general prevalence of the relatively lower abort ΔV requirements. The abort ΔV cost becomes significantly higher for return times less than about a day.

Comment [j7]: In the figure you cannot see that. I think that we should make this figure so we can see the mean and min values. The max values are distorting the picture. I'd rather have the max values in a different scale or in another figure.

Comment [j8]: I don't know what it means

Comment [j9]: Is this in general or for a particular inclination. Please cite the figure or figures we should look at to confirm this.

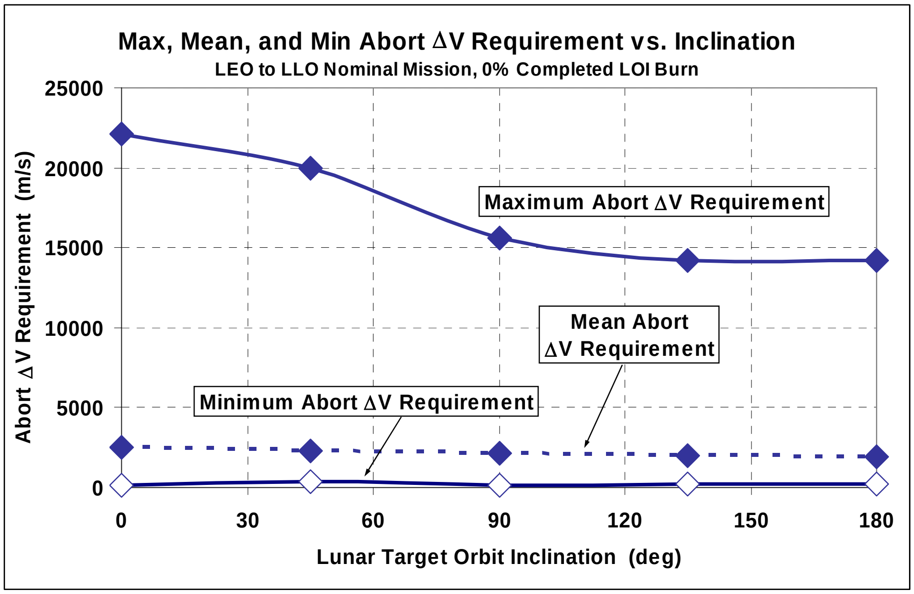

Figure 3 **Abort ΔV performance requirement versus lunar target orbit inclination. Data represent maximum, mean, and minimum values in the entire sample space of abort trajectories.**

The abort ΔV requirements and associated relative Earth return entry speeds for a nominal LEO to LLO mission targeted to lunar arrival inclinations of 0°, 90°, and 180° are shown in Figure 6 (a, b, and c). Note that the relative entry speeds are the result of an attempt to minimize the abort ΔV cost without particular regard to the entry location and heading. These insertion point parameters (i.e., latitude, longitude, azimuth) are an outgrowth of this ΔV minimization. If a particular entry latitude, longitude, and azimuth were desired for a range of abort times, the abort ΔV cost would generally increase (as more constraints are added to the Earth return trajectory). A desired entry interface latitude would be significantly affected by the lunar departure epoch and its associated lunar declination with respect to the Earth equator. The return flight time would also have an effect, though less significant. Further, there may be instances where a single abort ΔV maneuver would not provide enough flexibility to achieve a particular entry condition. In this case, an additional degree of freedom in the form of another maneuver or set of maneuvers may be needed. One possibility would be performing an abort into a phasing orbit, followed by subsequent deorbit at the proper location. This approach assumes the ability to restart the engine at least once following declared abort. In the event that only one abort maneuver was available, there would be restrictions in the achievable entry conditions (i.e., latitude, longitude, altitude, azimuth, and flight path angle combinations) that would be available to the crew. These sensitivities would have to be determined in future studies.

Comment [j10]: If we want to return in 2h then it would become very significant. Why don't remove this sentence.

Comment [j11]: *I really like these remarks. Very good!*

The results for 0°, 90°, and 180° lunar inclination targets show the lowest abort ΔV requirement for higher orbit inclinations (around 180º). However, the 90° lunar arrival inclination represents a practical abort ΔV performance limit for lunar inclination targets for "manned" missions. The parking orbit inclination selected would depend on the landing site latitude and surface stay time, though it would most likely be retrograde due to lower abort ΔV requirements following failed or partially failed LOI. Polar lunar orbit inclinations (i.e., 90°) might be used to support extremely high latitude or polar landing site latitudes.

Retrograde lunar inclination targets generally result in lower Earth return abort ΔV requirements (see Figure 5). The powered flight performance cost (ΔV) between LLO and the lunar surface varies between $\pm$5 m/s with the lower cost associated with posigrade (0°-90° inclination) cases[3]. However, the retrograde orbit insertion (and its associated powered maneuvers) is preferred due to lower abort ΔV requirements that overshadow the small powered flight cost differential. A free-return fly-by mission would employ a retrograde lunar approach. The Apollo program used retrograde parking orbits to provide free return (i.e., no abort ΔV required) fly-bys, following failed LOI, for the first two lunar landing missions (Apollo 11 and 12). Subsequent missions employed retrograde parking orbit targeted trajectories modified to provide low ΔV abort-to-Earth cost with a lower nominal mission ΔV. Note that an abort from a posigrade lunar inclination-targeted trajectory would have a larger associated abort ΔV requirement. Further, if no abort were performed following a failed LOI (for a posigrade insertion), the spacecraft would tend to enter into a large and undesirable Earth orbit or may swing around the moon into heliocentric space. The latter case was demonstrated during the Apollo program when a spent SIVB trans-lunar injection stage was disposed of via its diversion from the nominal Earth-Moon transfer trajectory to a passage around the front (posigrade) side of the moon.

In general, for a given abort time, the abort ΔV requirement is increased with reduced Earth return flight time. A confluence point of abort ΔV performance appears around 96 hours (the nominal lunar arrival time). Note that LOI nominally executes at lunar arrival, but for this study, the LOI is assumed to be completely failed (i.e., it does not execute). The abort ΔV values are specified for mission times both before and after the nominal planned LOI maneuver. For longer Earth return times, there is a high variability in the abort ΔV cost for an abort time occurring around this nominal planned LOI time. As the Earth return flight time is reduced to a small value (e.g., around a day or less), the abort ΔV requirement sees a large increase, for all times of abort.

For the case of a 90° lunar inclination target, an 11,000 m/s Earth return entry speed limit covers all abort time and all Earth return times greater than about 55 hours. Depending upon the time of abort, the abort ΔV cost could range from about 700 to 2000 m/s. For a 180° lunar inclination target case, the 11,000 m/s Earth return entry speed limit covers about the same minimum 55 hour Earth return time while the abort ΔV cost is a bit less than 1800 m/s, for all post-EOD times of abort.

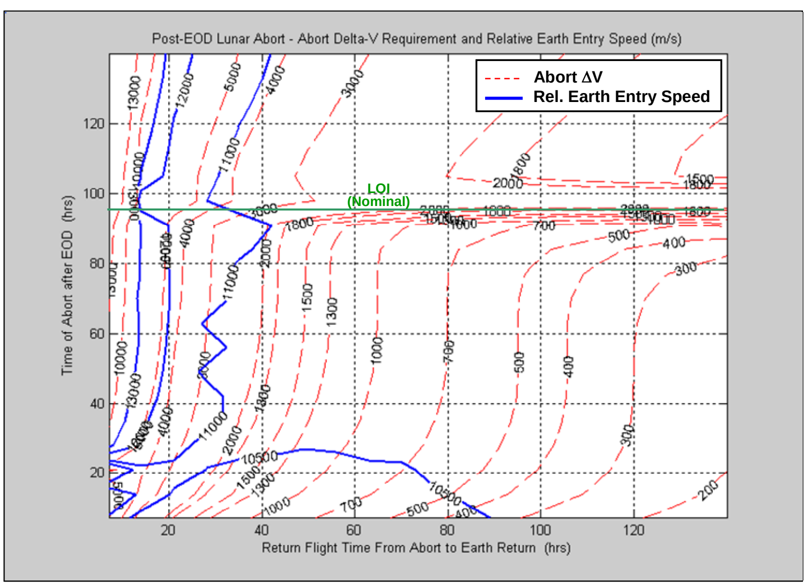

Figure 6(a) **Abort ΔV requirement and relative Earth entry speed versus post-EOD abort time and Earth return flight time, for a 0° lunar orbit inclination target.**

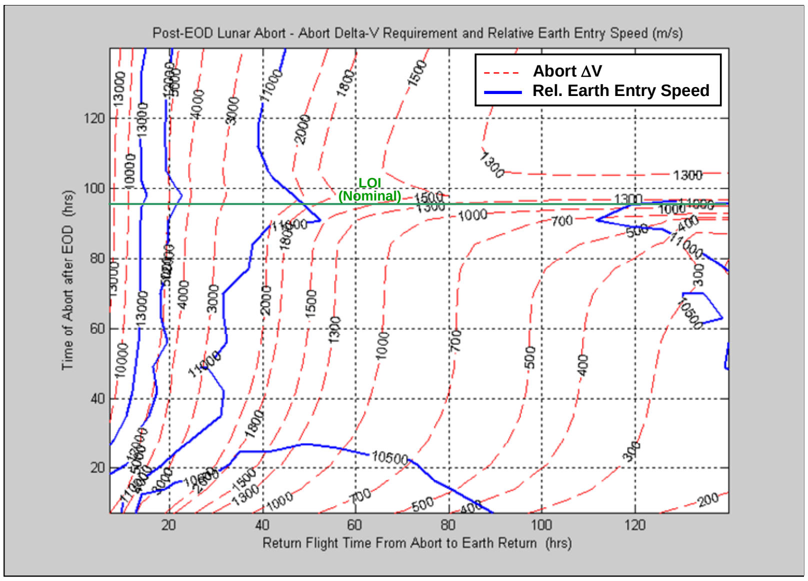

Figure 6(b) **Abort ΔV requirement and relative Earth entry speed versus post-EOD abort time and Earth return flight time, for a 90° lunar orbit inclination target.**

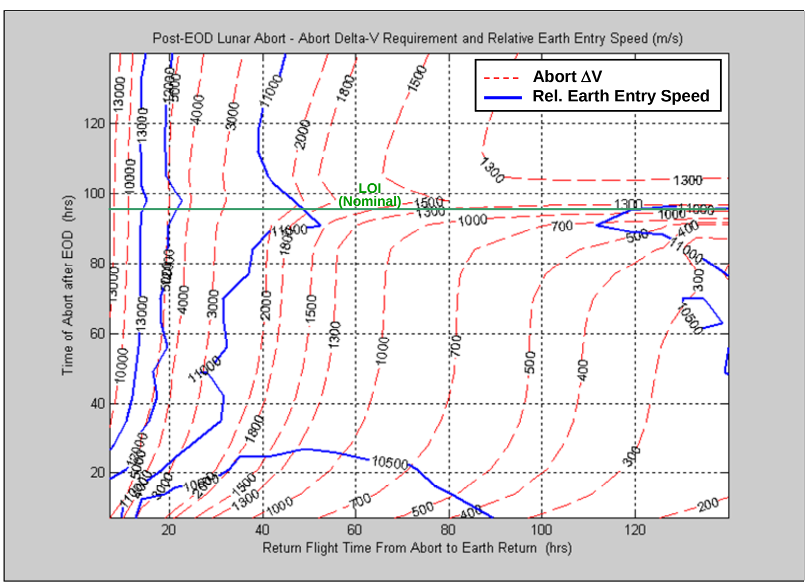

Figure 6(c) **Abort ΔV requirement and relative Earth entry speed versus post-EOD abort time and Earth return flight time, for a 180° lunar orbit inclination target.**

Comment [j12]: Figures 6(b) and 6(c) look the same. Is that true????

<u>LOR Mission Abort Residual ΔV</u> The abort ΔV requirement provides a general roadmap for achieving and Earth return from an aborted nominal trans-lunar trajectory given selected times of abort and Earth return times after the initiation of the abort. This roadmap includes associated parameters, such as relative entry speed and inertial flight path angle. The ability of a spacecraft to achieve the ΔV requirements depends upon its available propellant mass for a given overall spacecraft mass as well as the number of distinct and available propulsion systems and their specific impulse (Isp). The specific impulse is a measure of the engine performance.

A vehicle based on the ESAS results is measured against the abort ΔV requirements to assess its ability to achieve the required abort performance. The configuration shown in Figure 7 represents an "all-up" spacecraft as it appear after TLI, after the EDS has been jettisoned. This Apollo style configuration includes propulsion systems on the CEV (i.e., the service module or SM), the LSAM descent stage, and the LSAM ascent stage. For this study, the CEV command module is assumed to have no significant translational maneuvering capability.

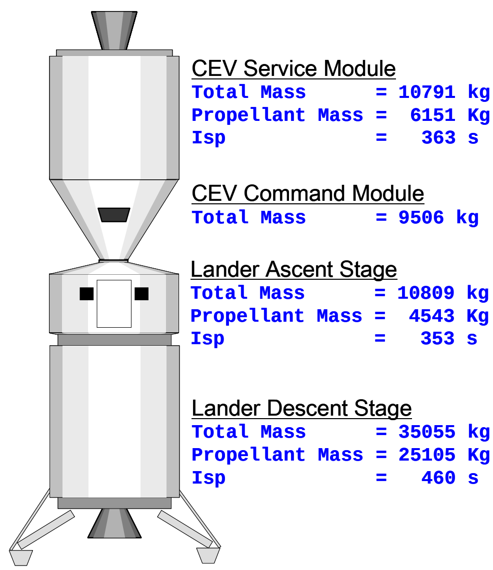

Figure 7 **Vehicle configuration and component propulsion system masses designed to support a human LOR mission.**

The liquid oxygen/hydrogen LSAM descent stage (Isp = 460 s) performs the LOI and the lunar deorbit and powered lunar descent burns. The liquid oxygen and methane LSAM ascent stage (Isp = 353 s) performs the powered lunar ascent as well as the translational maneuvers in the rendezvous sequence that brings it back to the CEV in lunar orbit. The liquid oxygen and methane CEV SM (Isp = 363 s) nominally performs the TEI that places the crew on a path back to Earth after the completed lunar mission.

The nominal mission sequence for this spacecraft includes the following: TLI performed by the EDS which is discarded after TLI completion and before LOI, LOI performed by LSAM descent stage, deorbit and powered descent to the lunar surface performed by the LSAM descent stage, powered ascent and translational rendezvous maneuvers (following lunar surface stay) performed by the LSAM ascent stage, and the TEI maneuver performed by the CEV SM. These stages are designed to support a typical lunar mission. However, in the event of a declared abort, these stages may be used together to accomplish desired Earth return transfers with desired flight times.

The ability of a spacecraft to achieve a given abort ΔV requirement is measured in terms of residual abort ΔV. The residual ΔV represents the ΔV capability remaining in a set of selected available propulsion systems in a vehicle configuration after that spacecraft has achieved its abort ΔV requirement. If the residual abort ΔV is a positive, then there is additional propulsive ΔV remaining in the given spacecraft after completion of the abort. If the residual abort ΔV is zero or a negative number, then the given spacecraft configuration is not capable of achieving the given abort ΔV requirement (for a given abort initiation and Earth return time).

<u>Earth Return Abort – Complete LOI Failure</u> The contour plots found in Figure 8 show the residual abort ΔV as a function of abort initiation time and Earth return time, for selected propulsion system availabilities. Overlaid on these plots are the corresponding Earth return entry speeds (i.e., 11 km/s and 12 km/s contours). The entry speed information provides insight into entry speed and associated aero-heating limits that may occur for particular aborts. Violations of entry speed or associated aero-heating limits may obviate a particular abort, even if there exists enough propulsive capability on the spacecraft to execute that particular abort.

If the stage in the lower right corner of each plot is grayed out, then that propulsion stage is considered to be unavailable to contribute to the overall abort ΔV. In general, the greatest residual ΔV results occur in cases of early abort times and longer Earth return flight times. The plot in the upper left shows the case with the most available performance where all propulsion stages (CEV SM, LSAM descent and ascent stages) are available. In this case a 2-day Earth return time could comfortably accommodate and aborted mission for any time of abort (after TLI). In addition, a flight time of two days or greater for the Earth return provides for a relative entry speed less than 11 km/s. The configuration in the upper right plot shows a CEV SM propulsion system that is unavailable. However, the stage can be jettisoned to reduce weight to enhance performance of the remaining stages, due to reduced mass. Depending on the emergency, the SM may be retained in the spacecraft configuration despite an unavailable propulsion system due to other critical systems (such as power and consumables). Overall abort capability degrades as the configuration moves to one with a failed LSAM ascent stage (lower left). Finally, the worst abort performance occurs when the large LSAM descent stage is unavailable. In this case, a late abort time near or after the nominal lunar arrival time (i.e., following failed LOI), may necessitate several days (3.5 to 4 or more) return flight time. This long Earth return time will place an additional burden on the available consumables. Overall, the all-up configuration provides reasonable propulsive ΔV capability for aborted return to Earth for a number of propulsion system failure cases.

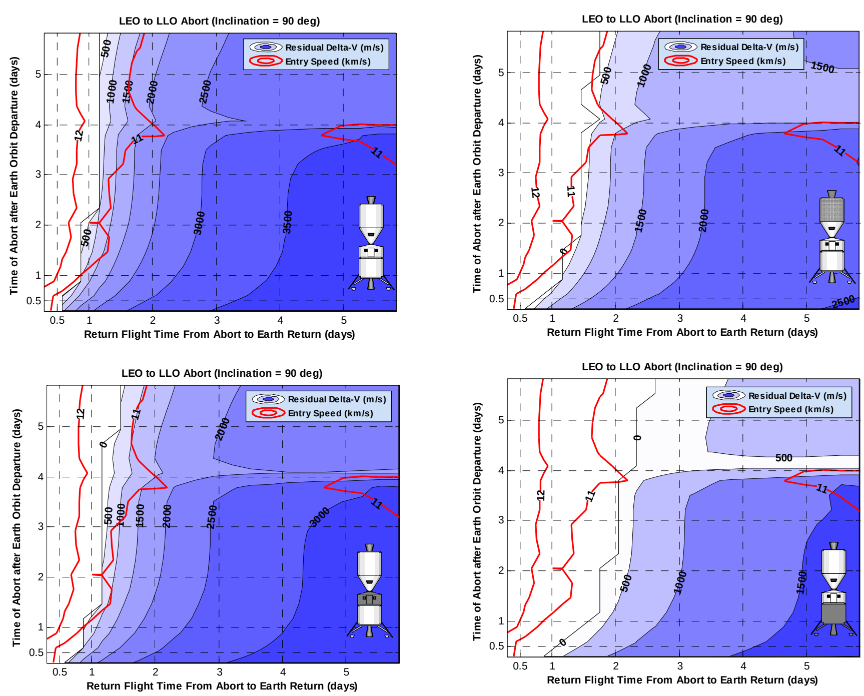

Figure 8 **Abort residual ΔV available after performing required abort ΔV, for selected vehicle propulsion stage availabilities.**

SUMMARY

In general, the larger abort ΔV cost is associated with a faster Earth return flight time and, consequently, a higher entry speed. The abort ΔV requirement is also generally lower for insertion into a retrograde equatorial (i.e., 180° inclination) low lunar orbit (LLO). The average abort ΔV over the (abort time vs. Earth return time) sample space for any LLO inclination target is significantly lower than the maximum abort ΔV for the corresponding LLO inclination. This result reflects the relatively low abort ΔV cost in the sample space. The abort ΔV cost rises significantly for Earth return trip times less than about a day. For Earth return flight times greater than a day, the slope of the abort ΔV cost with Earth return flight time is shallower.

CONCLUSIONS

Based on the assumptions used in this study, an abort ΔV capability of about 2000 m/s and an entry speed of capability of 11,000 m/s would allow an abort from any time on the outbound trajectory back to Earth in a minimum of about 55 hours. Constraining the Earth entry location and heading will, most likely, require a higher ΔV availability. It is recommended that the effects of these constraints be assessed in future studies. The retrograde lunar orbit inclination targets provide lower abort ΔV costs than posigrade orbits. The lunar polar (i.e, 90° inclination) orbit target provides a reasonable abort ΔV performance limit for manned lunar missions.

REFERENCES

1. Robertson, E., Geffre, J., “Lunar Architecture Focused Trade Study Report”, ESMD-RQ-0005, Rev. A, February 4, 2005.
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5. ESAS study reference --- the one that includes the LOR as the preferred architecture.
6. Apollo abort study.
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To Do:
1) Restructure references in chronological order (from front to back).
2)

Comment [j13]: I haven't checked the format but I think that we should put the references in the order that we cite them.