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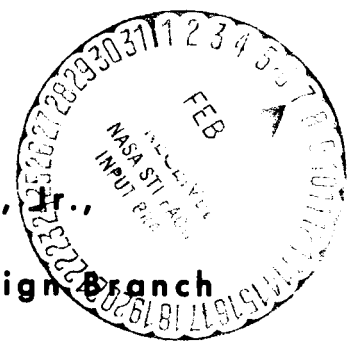
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TRANSLUNAR LM DPS ABORT
TECHNIQUES FOR
ADVANCED LUNAR MISSIONS

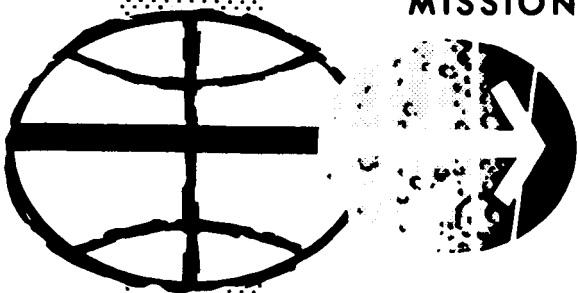
By Gus R. Babb, Jr.,

Advanced Mission Design Branch



MISSION PLANNING AND ANALYSIS DIVISION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS



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ABORT TECHNIQUES FOR ADVANCED LUNAR
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FIGURES

Figure		Page
1	Profile of a single-impulse abort trajectory after an SPS engine failure at lunar orbit insertion	9
2	Single-impulse abort ΔV requirements for a maximum payload trajectory to Copernicus	10
3	Single-impulse abort ΔV requirements of nonfree-return trajectories to Aristarchus and Copernicus . .	11
4	Multi-impulse abort ΔV requirements of nonfree-return trajectories to Copernicus	12
5	Profile of multi-impulse abort after engine failure at lunar orbit insertion	13
6	Abort ΔV available from LM descent propulsion system	14
7	SM separation prior to LM DPS abort propulsion maneuver	15
8	CM DPS abort maneuver on to a high-speed transearth trajectory following SM separation	16
9	Flight profile for abort with SM jettisoned to increase ΔV capabilities	17
10	Abort ΔV requirements for fast TE trajectories, Copernicus mission	18
11	Abort ΔV requirements for fast TE trajectories, Aristarchus mission	19
12	LM electrical energy consumed to provide emergency crew lift support after SM separation	20
13	LM O_2 and H_2O consumed to provide emergency crew life support after SM separation	21

TRANSLUNAR LM DPS ABORT TECHNIQUES
FOR ADVANCED LUNAR MISSIONS

By Gus R. Babb, Jr.

SUMMARY

A method of providing abort capabilities from nonfree-return translunar (TL) trajectories using the LM descent propulsion system (DPS) is presented. Abort ΔV requirements are given which indicate that for low-energy TL trajectories the DPS does not have sufficient ΔV capability to abort the entire spacecraft. An abort technique is examined in which the SM is jettisoned and emergency life support is provided by the LM. The resultant decrease in spacecraft weight enables the DPS to abort the spacecraft from these low-energy trajectories and return it to earth before the life support capabilities of the LM are exceeded. The abort maneuver is delayed as long as possible (8 to 20 hours) to reduce the time required for LM life support and to obtain earth landing longitude control.

INTRODUCTION

The enlarged payload and performance requirements of longer and more complicated lunar landing missions will require the use of nonfree-return TL trajectories. For a mission to the crater Copernicus, for instance, a 110-hr, nonfree-return trajectory can deliver 5000 to 6000 lb more into lunar orbit than a free-return trajectory. This results in a 5000-lb increased lunar science payload or 5000 lb of service propulsion system (SPS) fuel which could be used for lunar orbit maneuvers such as a CSM assist of the LM.

The CSM has the capability of delivering an Apollo (36-hour) LM to a 60-n. mi. orbit over a limited region of the moon using free-return trajectories, but nothing more. With nonfree-return TL trajectories, a large variety of mission possibilities become available.

The advantage of the free-return trajectory is the free abort capability in case of lunar orbit insertion (LOI) failure. However, this advantage is lost as soon as LOI begins, and an SPS failure after a

partial LOI burn presents nearly as difficult an abort problem with a free-return as it does with a nonfree-return.

If nonfree-return trajectories are to be generally used, some technique must be found to abort the mission in case of a complete SPS failure at LOI, which is the first major SPS burn. The only other ΔV source available at this point is the LM DPS which is unfortunately quite limited in ΔV capabilities when maneuvering the CSM with a full load of fuel.

Studies of both single and multi-impulse abort ΔV requirements show that for the 110-hour Copernicus TL trajectory the minimum ΔV required for abort is approximately 2700 fps. This is beyond the 1950 fps that the DPS can generate with the service module loaded, as it is at this point. The problem is 40 000 lb of unused and unusable SPS fuel. There is no provision for emptying the fuel tanks, and the tanks themselves cannot be simply dropped off because the fuel tanks and the CSM life support system are all part of the same structure in the SM. One solution is to jettison the entire SM with its fuel and life support; abort onto a high-speed single impulse transearth (TE) trajectory with the DPS, which can generate 4600 fps with this reduced weight; power down the CM; and live in the LM during the TE phase. The CM can be powered up and the LM jettisoned just prior to entry.

This paper presents abort ΔV requirements and examines the feasibility of abort from nonfree-return trajectories using the LM for life support over a portion of the return trajectory.

ANALYSIS

At the beginning of the LOI burn the spacecraft is composed of the CSM, with nearly full fuel tanks, docked to the LM (which is also fully fueled). The weights of these various vehicles are given in table 1.

It is assumed that there is a total SPS failure at initiation of the LOI burn, i.e., the TL trajectory is in the slow nonfree-return type and with no previous warning the engine fails to start. The problem is to find some way to return the crew to earth from this point.

Single Impulse Abort

In the single impulse abort case the spacecraft makes a single burn directly from the TL coasting orbit onto a transearth trajectory. This profile is illustrated in figure 1. With this type of abort trajectory, the ΔV required rises sharply with an increase in the time between pericyynthion passage (at which the SPS failure occurred) and the application of the abort impulse. This is shown in figure 2 for the Copernicus

landing mission. The required ΔV reaches a peak and levels off to a nearly constant value at a time past pericyynthion of slightly less than 2 hours. Because of the 2 to 4 hours required to ready the LM descent stage for abort, the lower-cost aborts immediately after pericyynthion are not, in practice, available.

The ΔV requirements also increase with an increase in translunar flight time (fig. 3). The slower nonfree-return trajectories which have the greatest payload capabilities also present the most difficult abort problem. In few cases of interest is the required ΔV within the 1950-fps capability of the DPS when moving the fully loaded spacecraft. More generally, the ΔV requirements are between 2500 and 3500 fps.

Multi-Impulse Abort

Because of the high ΔV required for single impulse aborts, a study of multi-impulse abort techniques was undertaken using the optimum n-impulse program developed in reference 1. The results of this study have been published separately (ref. 2). Figure 4 summarizes these results. The minimum three-impulse ΔV to abort from the 110-hour TL trajectory is over 3000 fps if the 2-hour delay after pericyynthion to ready the LM is observed.

The best multi-impulse aborts all had similar profiles (fig. 5). The first DPS burn occurs as soon after LOI as possible and places the CSM LM into a high ellipse around the moon. This requires approximately 600 fps for a 10 000-n. mi. ellipse. A second impulse is made at apocynthion to take care of any plane change that may be required. The third impulse and the second major burn is made somewhat before pericynthion and injects the spacecraft onto the TE trajectory defined by the final state vector. Thus, the spacecraft is inserted into a capture orbit so that lunar gravity can swing the velocity vector around, eliminating large flight-path angle changes.

The higher intermediate orbits require less ΔV for getting in and out of orbit. However, the higher orbits also have a longer period and the time between LOI failure and spacecraft exit of the lunar sphere of influence (SOI) increases. The position of the final V_{∞} vector must be rotated around in a posigrade direction corresponding to the 13° /day apparent revolution of the earth about the moon. This shifts the best firing position for the TE injection impulse farther and farther from pericynthion, making injection increasingly costly in ΔV .

The optimum ellipses had apocynthions of close to 13 000 n. mi. and periods of about 3 days.

The n-impulse program used here gives the best transfer between two specified state vectors for a given time interval Δt . The program was only capable of operations around a single body so that the state vectors had to be obtained from other sources. Initial state vectors (at the time of abort) were obtained from TL trajectories to Copernicus with various flight times. The final state vector was the state vector at the lunar SOI, obtained from the best single-impulse TE trajectory starting from the lunar parking orbit for a Copernicus landing and rotated to account for the abort time. This TE trajectory had an approximate 100-hr flight time. A few other TE state vectors were examined and the differences in ΔV were not found to be very significant. All the final V_{∞} vectors examined were from the same type of low-energy, 100-hour TE trajectories, and subsequent single-impulse studies have indicated that faster trajectories may give better results. Using faster TE trajectories increases the change in absolute velocity (energy state), but it also decreases the required change in the flight-path angle.

Abort Techniques

Both of the abort modes examined required abort ΔV 's on the order of 3000 fps for the slow low-energy TL orbits which are highly desirable for their payload carrying capability. Figure 6 gives the ΔV available for abort versus the CSM SPS fuel remaining onboard. For a fully loaded spacecraft, the DPS can only generate 1950 fps. Thus, it appears that with the kind of slow nonfree-return orbits necessary for the payloads and orbit orientations of advanced lunar missions, the DPS is often - if not always - unable to perform the kind of direct abort with the fully loaded spacecraft that had been envisioned up until this time.

The problem is quite simply too much weight. In particular, the 40 000 lb of SPS fuel. This accounts for nearly half of the mass remaining after the abort burn, and reduces the available ΔV to the point where an abort cannot be made. This SPS fuel is completely useless since the SPS engines do not work, and if in some manner it could be jettisoned the increase in available ΔV would allow the spacecraft to abort from almost any TL trajectory.

There is no current provision for jettisoning the SPS fuel and it appears that modifications to do so would be very difficult. The events that would most likely cause SPS failure with hypergolic fuels would be of the same type that would prohibit emptying the fuel overboard, since pumping the fuel into the firing chamber and pumping it outside is essentially the same problem. In any case, hardware modifications of a rather extensive nature would have to be made to jettison the fuel alone.

One solution is to jettison the whole SM. This not only eliminates the SPS fuel but also the fuel tanks, SPS engine, and structures. Without the SM the ΔV available increases to 4600 fps. Life support must be shifted to the LM at that point and the CM powered down. This is required because the CSM life support and power is all in the SM with the exception of enough in the CM for reentry. The spacecraft, during and after impulse, would simply be the LM with the CM being carried along as a reentry vehicle to be entered and used when earth is reached (fig. 7 and 8).

Once the SM is jettisoned, time becomes a critical factor because of the much more limited life support capabilities of the LM. The time on LM systems can be minimized by coasting out to the neighborhood of the SOI before the abort impulse is made (fig. 9). In this region the relative positions of the earth, moon, and spacecraft change very slowly and a rather large wait has very little effect upon the minimum obtainable time between the abort impulse and reentry. Figure 10 shows the time of TE flight after the abort impulse versus the ΔV required at the impulse based on a 110-hour Copernicus mission. This is the total time that life support must be on the LM systems. Two curves are shown: one is for the impulse occurring 8 hours past pericyynthion ($\Delta t = 8$) and the other is for an impulse occurring 14 hours past pericyynthion ($\Delta t = 14$). The minimum time of flight after the impulse is almost exactly the same in both cases, around 42 hours. This means that the time of the SM jettison and abort impulse can be varied so that a suitable landing area will be in the correct position for the reentry without increasing the total time from impulse to reentry.

Figure 11 shows that the same comments apply for an Aristarchus abort. In addition, the minimum time between impulse and reentry, during which life support is in the LM, is very close to the same for both missions, i.e., 42 hours. The ΔV curves are very steep in this region which indicates that this time (42 hours) is a solid limit and that a large increase in available ΔV is necessary to significantly reduce the minimum time required on LM life support.

On the basis of this information a consumables analysis was made for the LM in deep space with three men aboard. This analysis shows that there is apparently no major problem involved with three men living in the standard LM on an emergency basis for the 45 hours required of this abort. There is some problem, however, with CM power.

During the CM powered-down phase, all users of power in the CM can be turned off except for the computer. The resultant power level is quite low but the 42 hours is long enough that the total energy drain would completely deplete the CM power sources just about the time the reentry parachute opens. This would not leave any power to right the

CM if it landed apex down, nor would there be any power for beacon operations. This latter would be absolutely necessary should the landing take place out of sight of the recovery forces.

The CM power level during TE flight is low enough to be easily delivered by the CSM LM umbilical cord, and the LM batteries have more than enough power and energy to supply it. However, at present there is a relay system that disconnects the power lead of the umbilical when the descent stage is attached. Some method of circumventing this relay should be incorporated so that the descent batteries can supply the CM computer during the TE coast. With this change, the length of the TL flight time is limited only by the LM power sources and can be as long as 57 hours (fig. 12).

Other LM consumables are closely matched to the power output. The O_2 usage could last for a 62 hour flight, while the water would be depleted in 61 hours (fig. 13). Thus, there is only a 5-hr variation in the flight time capabilities of the major LM consumables.

RESULTS

Single-impulse and multi-impulse aborts from slow nonfree-return TL trajectories were examined. Abort ΔV requirements in both abort modes proved to be around 3000 fps which is considerably more than the 1950 fps capability of the LM DPS when moving the massive CSM LM spacecraft configuration remaining after an LOI burn failure.

The abort technique that has been developed in this study to circumvent this problem is to jettison the SM before the abort burn. This reduces the total mass of the spacecraft and the DPS can then provide 4600 fps of ΔV .

The CSM consumables are in the SM, which is left behind, so life support must be transferred to the LM. The life support capabilities of the LM are sufficient if the time spent between jettisoning of the SM and reentry at earth is as short as possible. Therefore, the spacecraft with the SM still attached is allowed to coast out to the region of the lunar sphere of influence (approximately 31 000 n. mi.). The SM is then jettisoned and the spacecraft aborts onto a high speed TE-trajectory. With the 4600 fps of ΔV that is now available, the TE transfer time after the abort burn can be as low as 40 to 45 hours. Emergency life support can easily be maintained onboard the LM for this period of time and, in fact, could be provided for up to 57 hours. This gives a certain amount of performance pad and contingency capability.

The minimum TE transfer time of 42 hours appears to be relatively independent of the original translunar trajectory. Consequently this abort technique should be suitable for all of the landing sites now being considered.

CHANGES TO THE LM

At present there seems to be only two modifications to the LM that are really necessary for this type of abort. First, the relay system involved with the umbilical cord between the CM and LM must be altered so that power can be sent back to the CM from the LM while the descent stage is attached. This requirement was discussed in detail in an earlier section. Second, a third suit umbilical outlet should be installed in the LM for the third crew member in the event of LM pressure loss.

CONCLUSIONS

An abort technique has been described in which the abort ΔV capability of the LM DPS is more than doubled by jettisoning the SM and shifting life support to the LM. The 4600 fps ΔV available with this technique is enough to provide abort capability, in the event of LOI failure, for all normal TL trajectories, including the very slow nonfree-return orbits which have maximum payload capabilities.

The length of time that life support must be provided by the LM is determined by the minimum time interval between the abort ΔV and earth reentry. This minimum time proved to be almost the same (approximately 42 hours) for both of the sites examined (Copernicus and Aristarchus) even though these sites represent a fairly wide spread in orientation of the original TL trajectories. This is well within the 57 hours of life support that the current LM can provide.

There is every indication that this relative invariance in life support requirements will hold true for trajectories to a very wide region of the lunar surface. In addition, the post-Apollo LM vehicles will probably have increased life support capabilities, and the combination of high- ΔV capability plus long TE flight times should allow aborts from any TL trajectories likely to be of interest for some time.

A safe return to earth in the event of an SPS engine failure at LOI can thus be effected by the LM DPS even for very low-energy TL orbits. This allows these maximum payload trajectories to be utilized in the mission planning process without violating current safety constraints.

TABLE I.- WEIGHT SUMMARY

Block II CM (manned), lb	13 000
Block II SM (includes RCS and unusable RCS fuel), lb	11 560
Usable SM fuel, lb	39 740
Total LM weight (unmanned but including full fuel), lb	<u>33 500</u>
Total spacecraft weight, lb	97 800
Usable LM DPS propellant	17 510 lb
DPS engine I_{sp}	299.4 sec

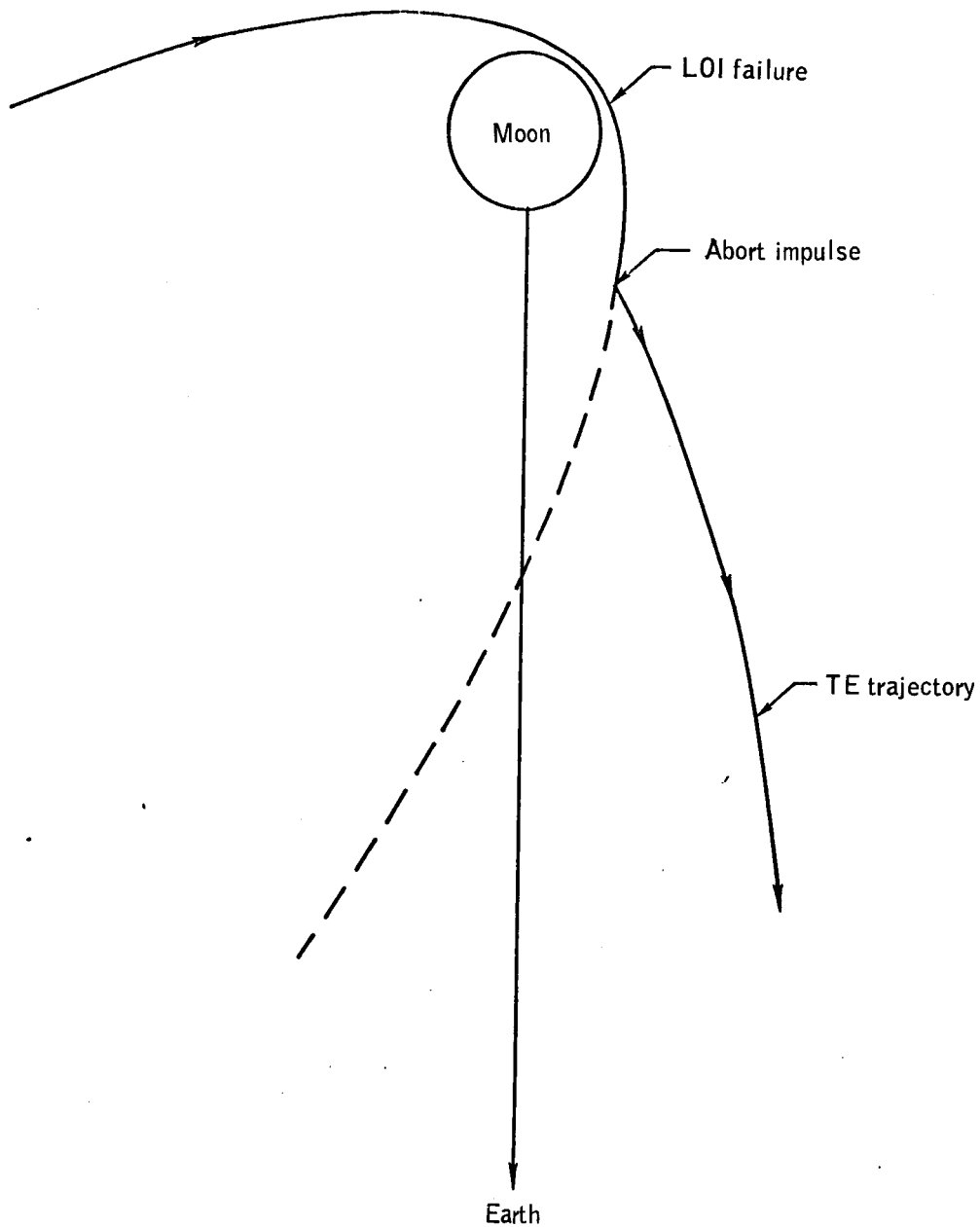


Figure 1.- Profile of a single-impulse abort trajectory after an SPS engine failure at lunar orbit insertion.

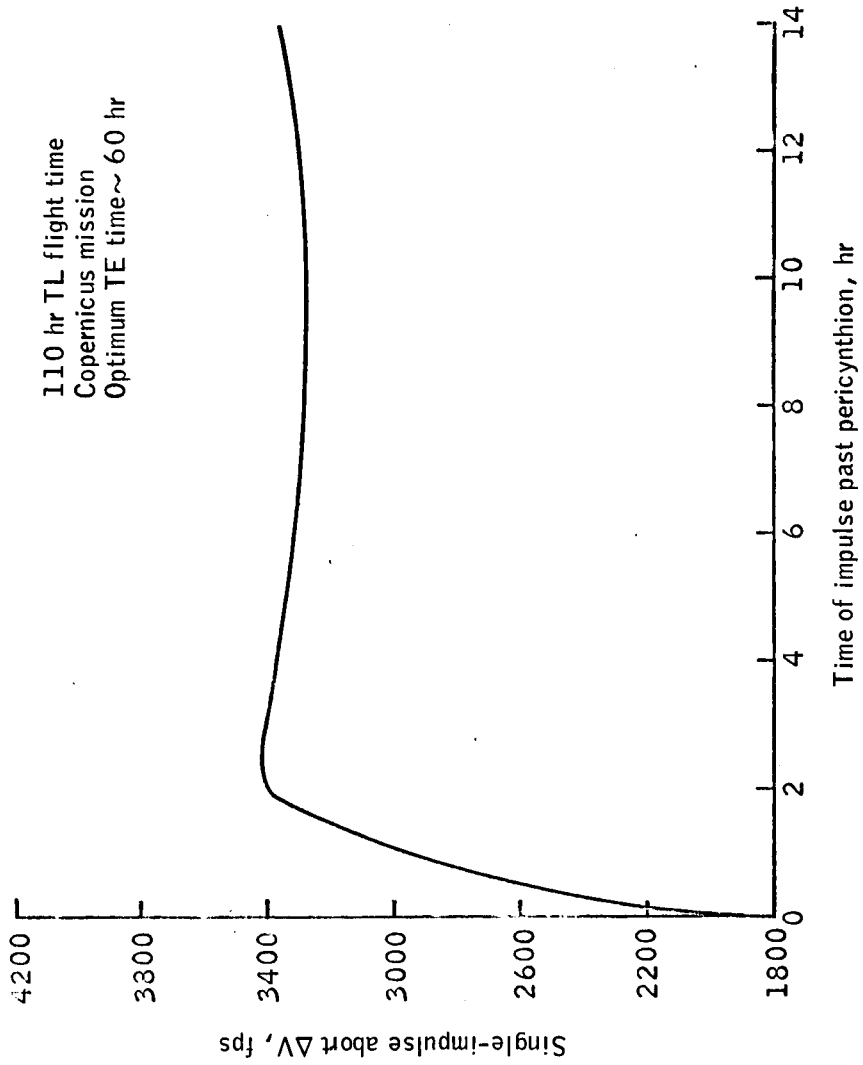


Figure 2.- Single-impulse abort ΔV requirements for a maximum payload trajectory to Copernicus.

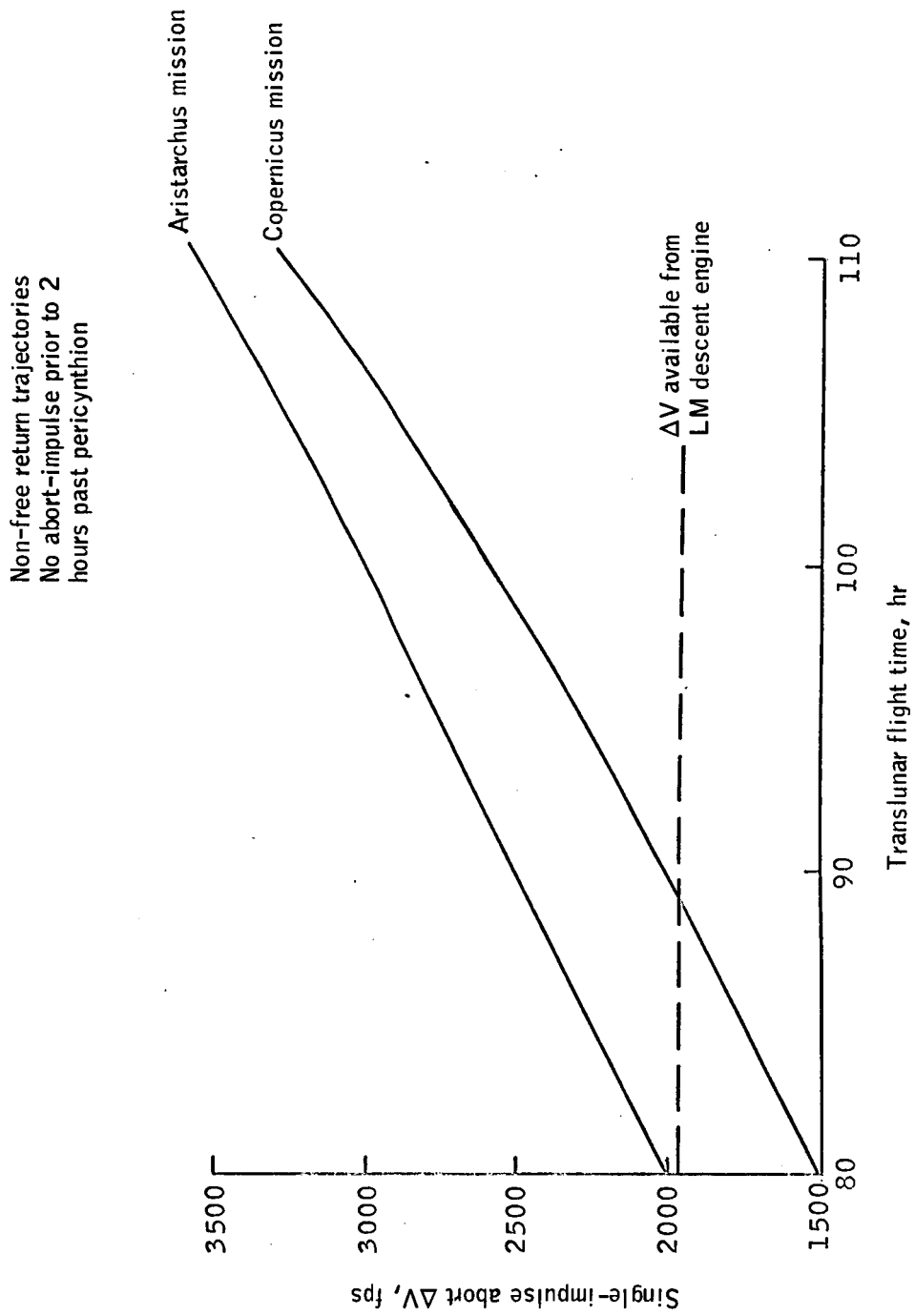


Figure 3.- Single-impulse abort ΔV requirements of nonfree-return trajectories to Aristarchus and Copernicus.

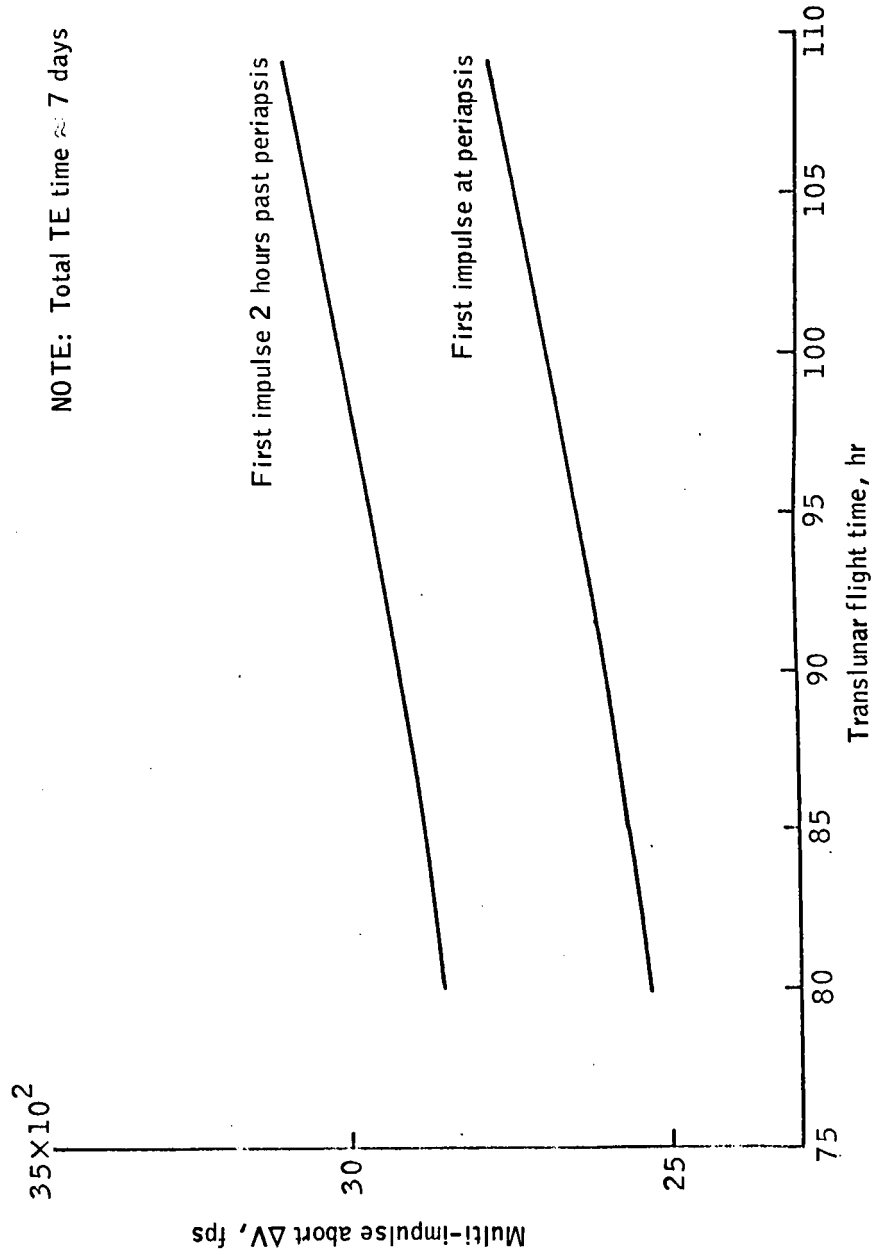


Figure 4. - Multi-impulse abort ΔV requirements of nonfree-return trajectories to Copernicus.

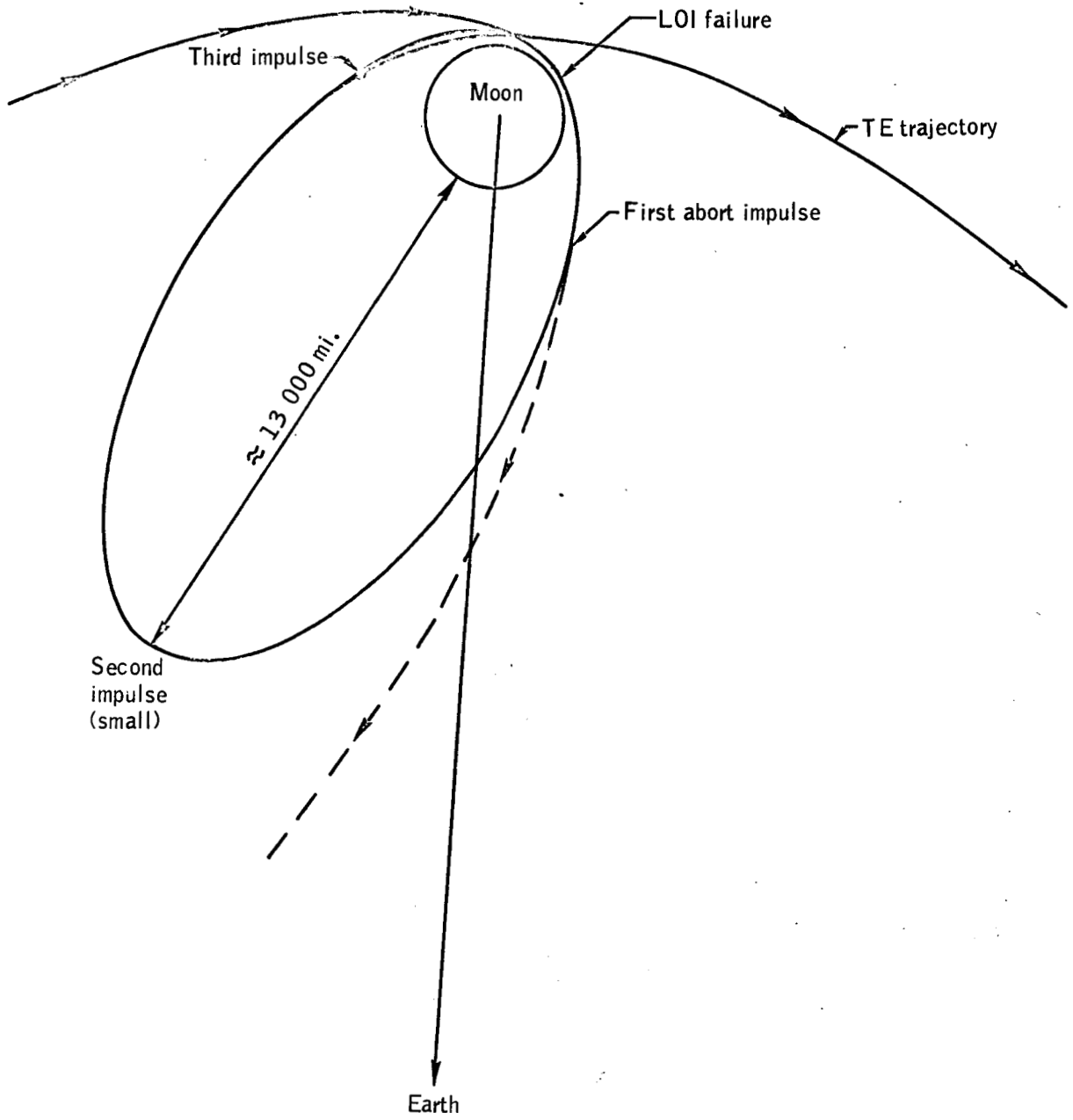


Figure 5.- Profile of multi-impulse abort after engine failure at lunar orbit insertion.

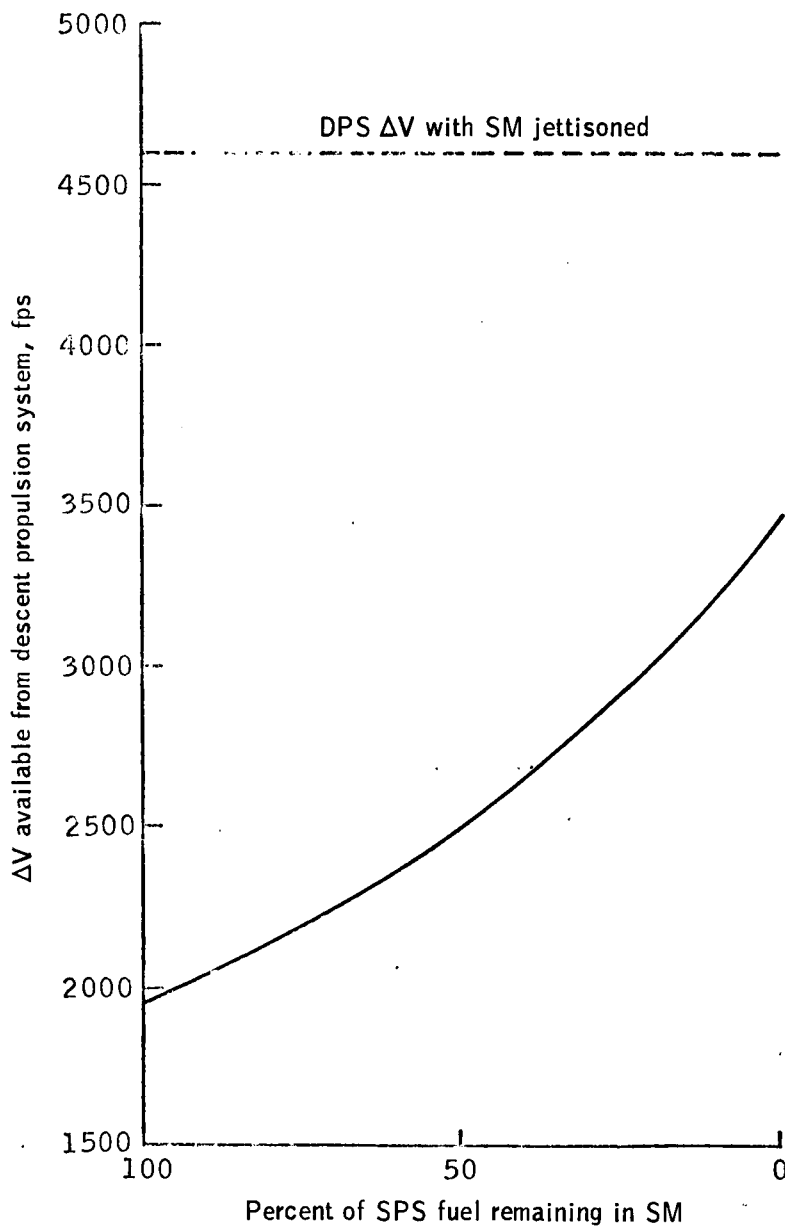


Figure 6.- Abort ΔV available from LM descent propulsion system.

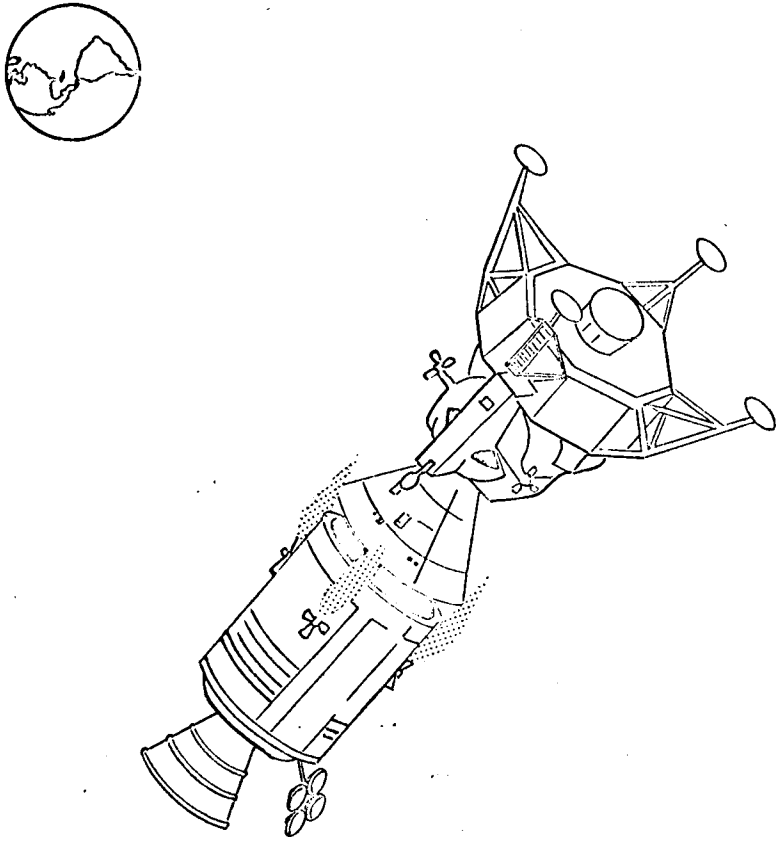


Figure 7.- SM separation prior to LM DPS abort propulsion maneuver.

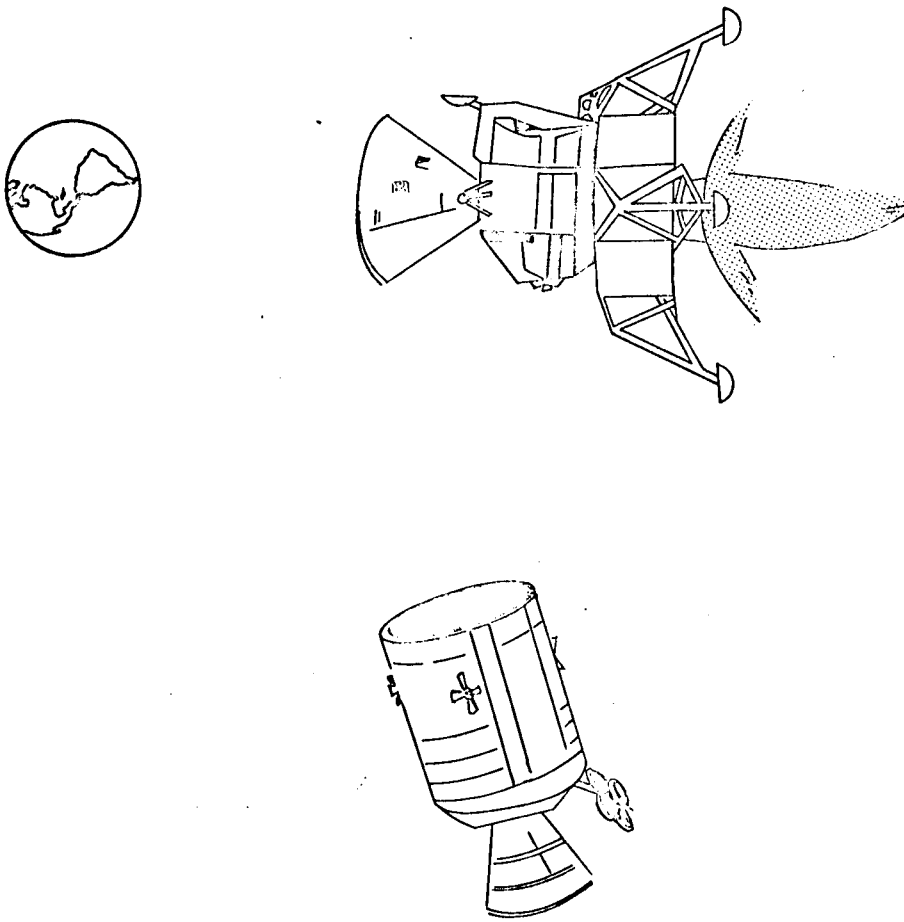


Figure 8. - CM DPS abort maneuver on to a high-speed transearth trajectory following SM separation.

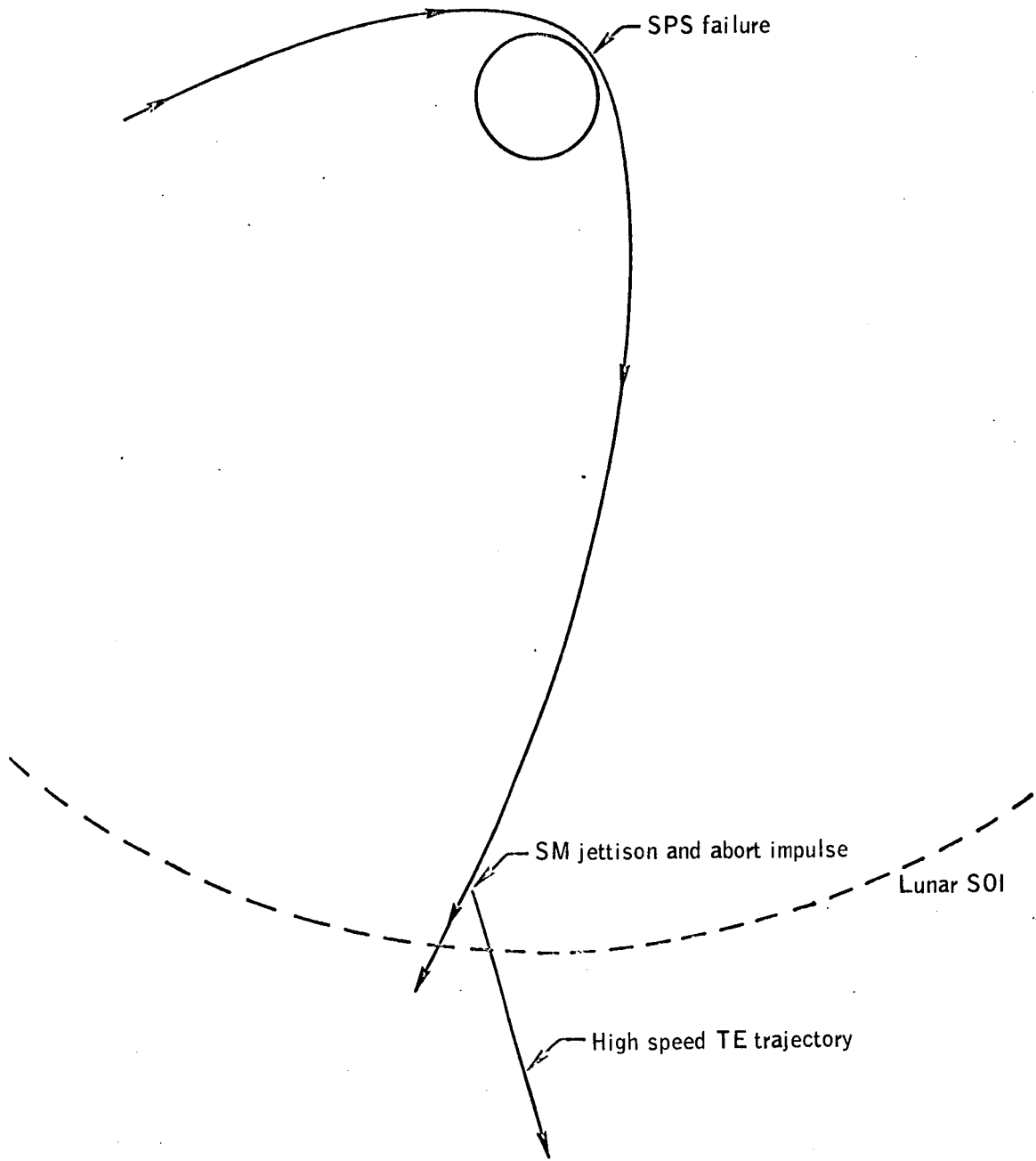


Figure 9.- Flight profile for abort with SM jettisoned to increase ΔV capabilities.

Note: Zero time = Δt after pericyynthion
110-hr TL non-free return trajectory

- Δ = Atlantic landing
(Ascension island)
- \odot = Pacific landing
(Hawaii)

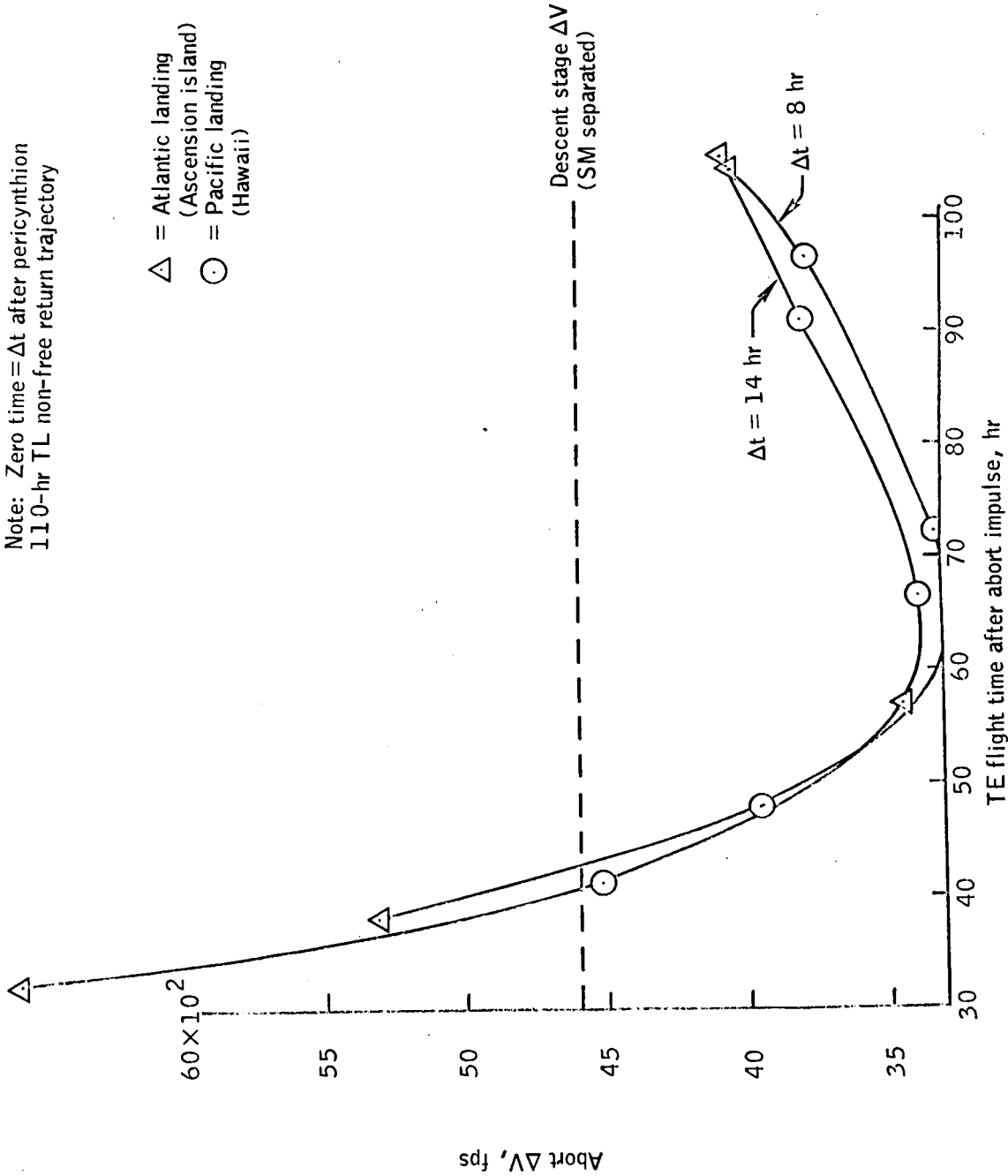


Figure 10.0.- Abort ΔV requirements for fast TE trajectories, Copernicus mission.

Zero time = Δt hr after pericyynthion
 110-hr flight time

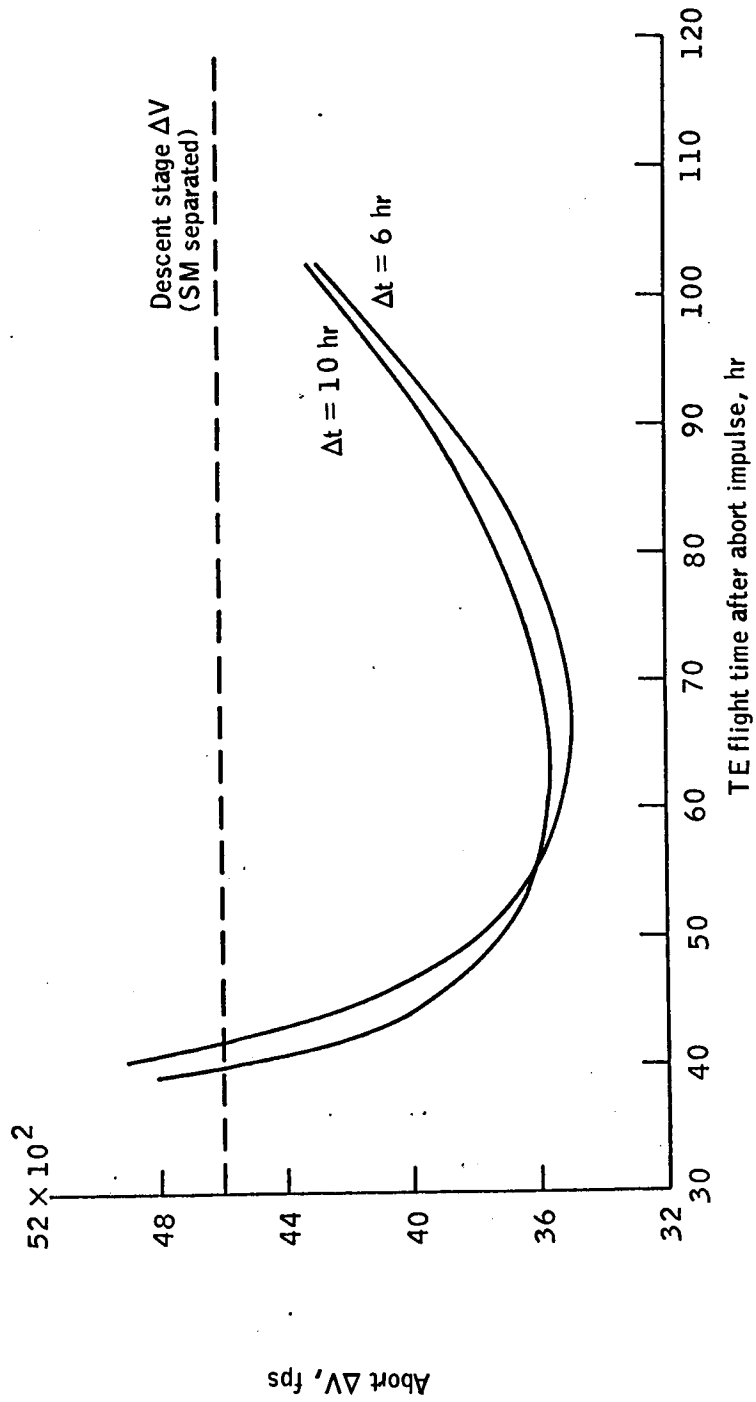


Figure 11.- Abort ΔV requirements for fast TE trajectories, Aristarchus mission.

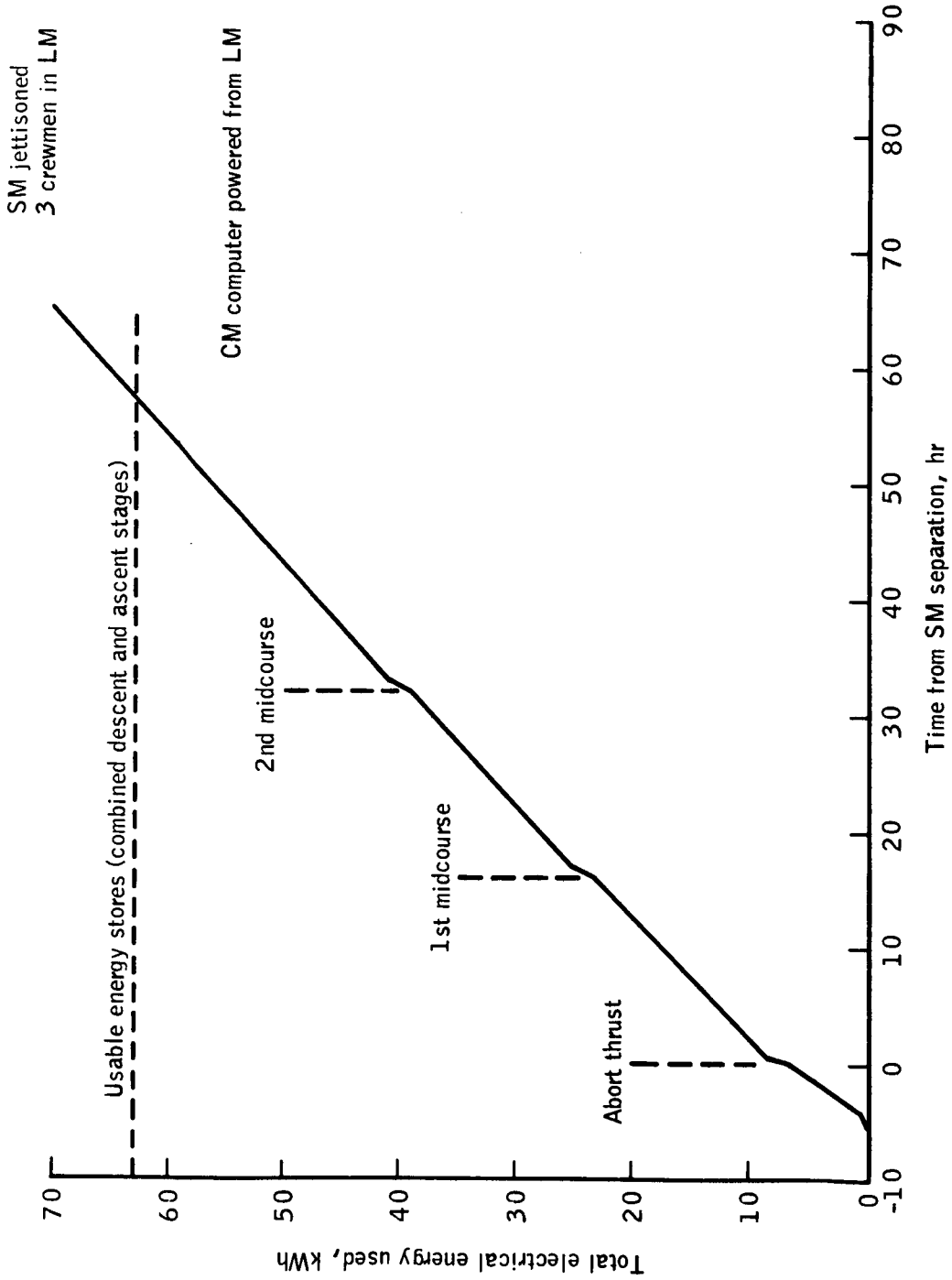


Figure 12.- LM electrical energy consumed to provide emergency crew lift support after SM separation.

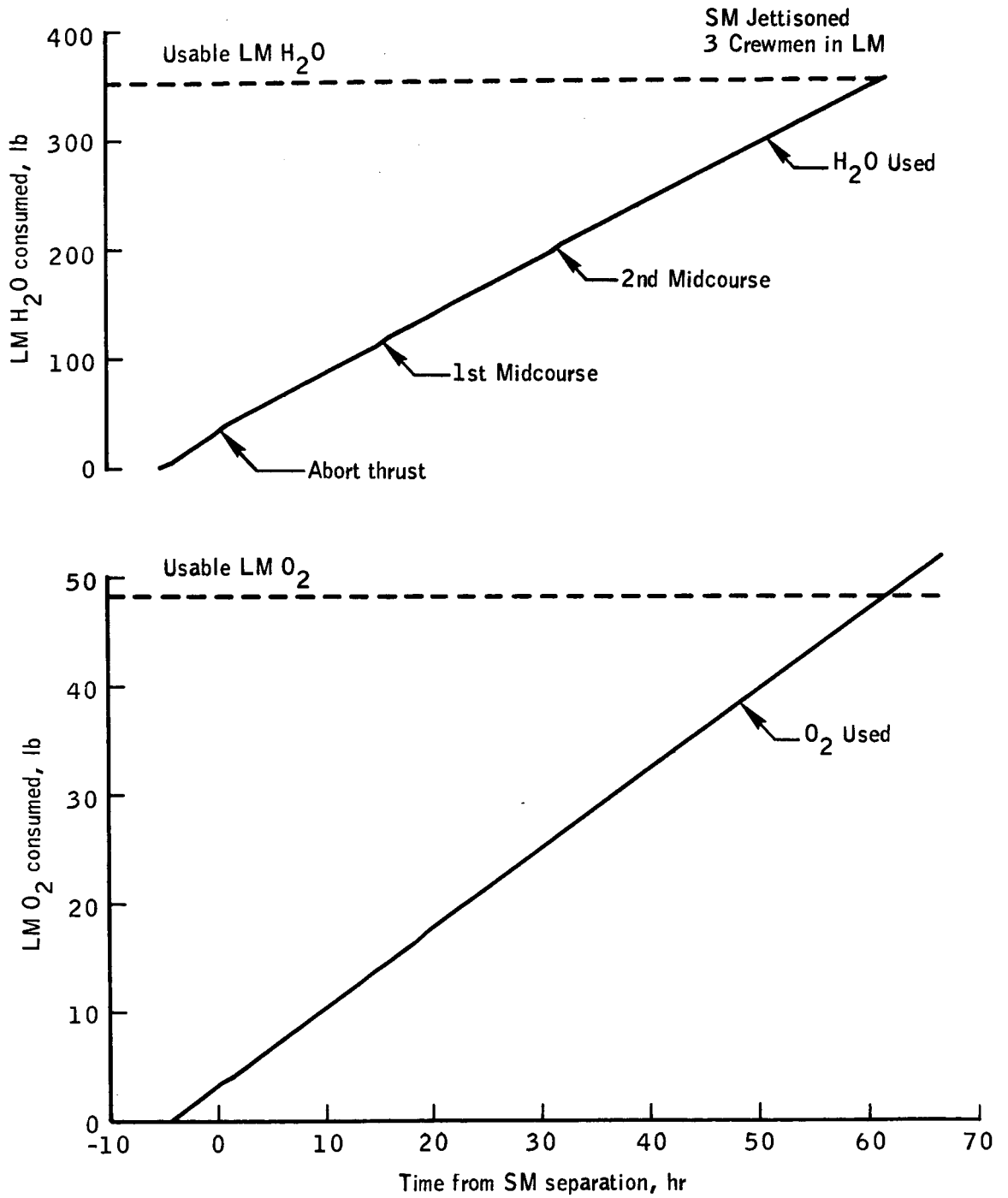


Figure 13. - LM O₂ and H₂O consumed to provide emergency crew life support after SM separation.

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1. Jezewski, Donald J.; and Rozendaal, Harrey L.: An Efficient Method for Calculating Optimal Free-Space N-Impulse Trajectories. MSC IN 67-FM-170, November 1967.
2. Kirkpatrick, James C.: Preliminary Abort ΔV Requirements for A Copernicus Mission. MSC IN 68-FM-116, May 16, 1968.