

# TECHNOLOGY OF LUNAR EXPLORATION

## ABORT CONSIDERATIONS FOR MANNED LUNAR MISSIONS

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### ABSTRACT

This paper considers direct and indirect abort trajectory requirements of the lunar orbit and descent phase of a manned lunar mission. Indirect aborts, treated through an intermediate lunar parking orbit, are shown to be possible from any point on the trajectory down to the final touchdown, but regions from which direct aborts can commence are restricted. Velocity increment and time-to-return for several direct and indirect abort sequences are established with emphasis on how the initial trajectory is shaped to facilitate the abort maneuver.

### INTRODUCTION

Manned lunar flights, unlike those in which no humans are carried, must inherently be capable of a successful mission abort at any time. As a design requirement, abort capability must be considered early in mission planning if the parameters it affects (such as vehicle configuration, mission profiles, and trajectories) are to approach realistic conditions as closely as possible.

A previous paper (1)<sup>4</sup> discussed the problem of aborts from an Earth-to-moon trajectory and showed how a series of abort "way-stations" can be computed in advance for the nominal transfer trajectory. In effect, it established an abort timetable for the flight crew, relating points along the nominal trajectory to attainable Earth landing sites.

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In this paper, the problem is extended to include possible aborts once lunar orbit has been established. Complex flight mechanics and trajectory analyses are bypassed in favor of an overall mission analysis that seeks comparative rather than precise results of spacecraft evaluation and system requirements. The result is the examination of techniques suitable for preliminary mission analysis of the various possible modes of abort from lunar orbits.

The various abort situations considered are shown schematically in Fig. 1. For simplicity, all the trajectories are assumed to be coplanar. This paper is concerned with the spacecraft after it has been inserted at point A into a lunar parking orbit, assumed to be either circular or elliptical. The circular orbit might be used for first-pass reconnaissance, followed by the elliptical orbit for descent to point F, where the terminal landing maneuver begins. Assumed typical nominal values of the orbits are as follows: circular orbit, 100 naut miles altitude; and elliptical orbit, 100 naut miles aposelenium (at point A) 50,000 ft. periselenium.

Fig. 1 defines direct and indirect aborts pictorially. As shown, both types of aborts commence from an arbitrary point, such as B or C, on either a circular or elliptical parking orbit. Indirect returns via a parking orbit are also considered from the abort points anywhere on the terminal descent trajectory (FH). An alternate form of indirect return, via rendezvous in the circular parking orbit (DEAP or FGAP), is also discussed.

#### DIRECT-RETURN ABORTS

##### Nominal Moon-to-Earth Return Trajectory

For accurate analysis of direct moon-to-Earth aborts, it is necessary to compute a series of return trajectories originating at the abort points B or C and satisfying allowed tolerances in transit time and vacuum perigee altitude. Finite thrust levels should be considered during powered flight, and the freefall trajectory should be computed with allowance for at least the gravitational fields of Earth, the moon, and the sun.

However, an alternate procedure, minimizing computation yet representing a real mission, was used to obtain the gross comparison desired in this study. Instead of computing a completely new moon-to-Earth trajectory for every abort point, one nominal return trajectory was established. Then, from any arbitrary abort point B or C, transfer trajectories, such as

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BP or CP<sub>2</sub>, are used to place the vehicle on the standard nominal trajectory. By limiting the insertion point P to within 3 lunar diameters of the moon's center of mass, the nominal return trajectory can be represented as a hyperbola in selenocentric coordinates, and the transfer trajectories as ellipses.

A nominal return trajectory (shown in Table 1) was computed on an IBM 704 using a program that includes the Earth, moon, and sun gravitational fields, Earth's oblateness, and lunar orbit eccentricity. This trajectory has an initial altitude of 123.7 naut miles above the lunar surface, transit time of 77.6 hr, and a vacuum perigee radius of 3264 naut miles. An hyperbola with eccentricity  $e = 1.485$ , approximating the nominal return trajectory in the vicinity of the moon, is shown in Fig. 2. The small differences between the computed trajectory and the hyperbolic approximation are shown in Table 1.

### Transfer From Circular Parking Orbit

The assumed geometry of direct abort is shown in Fig. 3. A two-impulse maneuver applies velocity increments ( $\Delta V$ ) at points 1 and 2, transferring the spacecraft from the parking orbit to the nominal return trajectory. The transfer ellipse is constrained tangent to the return trajectory at point 2. Thus, the first impulse puts the vehicle on the transfer ellipse, and the second adds velocity as required to match the return trajectory.

Defining the angles in the figure, the problem can be stated as follows: Given:  $r_1, r_2, \eta_1, \eta_2, \gamma_2$ , find orbital elements ( $\Delta V_1, \Delta V_2$ ) of transfer ellipse.

For each departure point  $\eta_1$  there are an infinite number of possible insertion points  $\eta_2$ . However, to limit  $r_2$  to within 3 lunar diameters (as mentioned previously),  $\eta_2$  should be less than 111 deg. Moreover,  $\eta_2$  cannot be less than  $\eta_1$ ; hence  $\eta_1 \leq \eta_2 \leq 111 \text{ deg.}$

Equating the parameter  $p$  (semi-latus rectum) of the transfer ellipse at points 1 and 2, and substituting for eccentricity  $e$  in terms of the known flight path angle,  $\gamma_2$  at point 2 yields the following equation:

$$\tan \gamma_2 = \frac{\left[ \frac{(1 - r_2/r_1)}{r_2/r_1 \cos \eta_{T_2} - \cos (\eta_{T_2} - \Delta \eta_{1-2})} \right] \sin \eta_{T_2}}{1 + \left\{ \frac{(1 - r_2/r_1)}{r_2/r_1 \cos \eta_{T_2} - \cos (\eta_{T_2} - \Delta \eta_{1-2})} \right\} \cos \eta_{T_2}} \quad [1]$$

This equation can be solved implicitly for  $\eta_{T_2}$ . The eccentricity  $e$  is given by

$$e = \frac{1 - r_2/r_1}{r_2/r_1 \cos \eta_{T_2} - \cos (\eta_{T_2} - \Delta \eta_{1-2})} \quad [2]$$

These equations were programmed for solution on an IBM 7090, and the results are as follows. Fig. 4 shows the elapsed time ( $\Delta t$ ) to the insertion point vs the location (true anomaly  $\eta_2$ ) of the insertion point, for both direct and indirect returns from the abort point  $\eta_1$ . (In Fig. 1, this is a comparison of transit times on paths  $CP_2$  and  $CAP_2$ , respectively.) Within the  $\eta_2$  region of interest, a break-even point is seen to exist for each abort location  $\eta_1$  at which the direct abort takes just as long to reach point  $P_2$  as the indirect abort.

Fig. 5 gives the corresponding impulsive velocity requirements for the two-impulse direct return and the single-impulse indirect return. A cross-plot of these results (time saved by direct return vs the associated  $\Delta V$  penalty) for various abort locations is shown in Fig. 6. The sharp knees represent regions of minimum  $\Delta V$  penalty because further increases in  $\Delta V$  produce only slight increases in time saved, whereas small decreases in  $\Delta V$  significantly reduce the time saved. Hence, a best operating envelope, such as shown in Fig. 6, can be drawn connecting these regions, to uniquely define the potentialities of this mode of direct abort for each initial abort location. The resulting values are shown in Table 2.

#### Transfer From Elliptical Parking Orbit

Similar results for direct aborts from the elliptical parking orbit are presented in Fig. 7-9. As expected, direct abort from the elliptical orbit requires slightly more  $\Delta V$  from a given location than the circular orbit. The best operating points are shown in Table 3.

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In general, it appears that the two-impulse direct returns are of questionable usefulness. The time savings are relatively small (less than 3 hr compared with 77 hr moon-to-Earth transit time) regardless of the velocity increment applied.  $\Delta V$  is not particularly limited, because if the mission is aborted while still in the parking orbit, over 12,000 fps normally used for landing and take-off are available. It may be that more time could be saved in returning to Earth by performing an indirect return and then using the available  $\Delta V$  to accelerate the moon-Earth return trajectory. This possibility requires further consideration.

### INDIRECT-RETURN ABORTS

#### Return from Parking Orbits

Indirect return from the parking orbit can be handled readily by remaining in the orbit to point A (Fig. 1) and performing a one-impulse transfer to the return trajectory at that point. Required  $\Delta V$  and elapsed  $\Delta t$  for such cases are shown in Figs. 4, 5, 7, and 8. Two-impulse transfers are employed if it becomes necessary to change the periselenium altitude or orientation of the return trajectory, to account for variations with time in the moon's position relative to Earth and sun. Indirect return can be made from any location  $\eta_1$  on the parking orbit.

#### Abort From Terminal Descent

The terminal abort trajectory is indicated as segment FH in Fig. 1, and the geometry considered is shown in Fig. 10. The abort begins at an arbitrary point 1, with an initial impulse that places the vehicle on a transfer ellipse whose aposelenium occurs at point 2, tangent to the original parking orbit. In this case, the terminal descent and subsequent abort return was from the 50,000-ft periselenium of the elliptical parking orbit.

#### Nominal descent trajectory

The selection of a nominal descent trajectory requires that many factors be balanced, such as propulsion, guidance and control system performance, abort capability, and error sensitivity. No attempt is made herein to select a best nominal descent trajectory based on these factors, but by examining available trajectories some insight is gained into the abort considerations influencing descent trajectory shaping.

A number of descent trajectories were available in which either elapsed time or propellant consumption (or both) was minimized by applying the gradient, or steepest descent,

method of variational calculus (2). The required thrust-angle program was determined for optimum trajectories for both constant thrust/weight ratio and constant thrust and for continuous and intermittent burning. Two representative continuous burning, constant-thrust trajectories are shown in Figs. 11-13. As shown, these are not usable trajectories because the tangential velocity component should be reduced to zero at least a few thousand feet above the surface, and the thrust-angle program must be simplified to permit mechanization in a real system. However, these modifications could be made without changing the essential shape of the trajectories.

Basically, these are low flight-path-angle trajectories that retain high tangential velocity down to low altitude. This feature is desirable for abort, because it minimizes the  $\Delta V$  required for abort from a given altitude. Thus, the vehicle could descend quite close to the surface, permitting a closer view without committing all the available  $\Delta V$  in the event of abort. However, the high tangential velocity increases the error sensitivity of the range to touchdown, introduces terrain clearance reaction-time problems, and may eliminate the possibility of obtaining closer reconnaissance at the lower altitudes.

These trajectories are also very economical of propellant, requiring within 1% of the ideal minimum impulsive  $\Delta V$ . As constant velocity vertical descent or hover time is added, however, propellant consumption will increase by about 1% for every 20 sec.

Comparing the trajectories with initial thrust-to-weight ratios ( $F/W_i$ ) of 2 and 6 (Figs. 11 and 12), the former is seen to be an extremely shallow-angle approach, accentuating the conditions discussed previously. However, it also offers lower vertical velocity and longer elapsed time than the higher thrust trajectory, which is desirable to allow more time for crew participation during piloted landings. Both trajectories offer the same low propellant consumption. But because the  $F/W_i = 2$  trajectory favors the above-mentioned abort considerations to a greater extent, it is used as a nominal terminal descent on which abort requirements are subsequently based. These abort requirements are therefore close to the minimum attainable and will undoubtedly increase as the nominal trajectory is reshaped to reduce its touchdown-error sensitivity and to simplify its thrust-angle program.

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### Impulsive abort trajectories

Using the  $(F/W_1) = 2$  terminal descent trajectory as the point of departure, two-impulse transfer ellipses were computed assuming impulsive thrusting. In addition to the tangency constraint already discussed, the transfers were limited to an increment in true anomaly ( $\Delta \eta_{1-2}$ ) of 15 deg. This limit is considered necessary to provide a relatively rapid ascent with adequate terrain clearance. Downrange distance covered during the ascent is 215 naut miles, which is consistent with the distance covered during descent. Such a constraint on  $\Delta \eta_{1-2}$  will increase the  $\Delta V$  requirement about 10% above that required for a Hohmann transfer ( $\Delta \eta_{1-2} = 180$  deg). Hence, for the transfer ellipse,  $r_1$ ,  $r_2$ ,  $\eta_1$ ,  $\eta_2$  and  $\gamma_2$  are all known, and the orbital elements can be determined directly. The resulting  $\Delta V$  requirements are given as a function of abort altitude in Fig. 14. Abort  $\Delta V$  is minimal down to low altitudes because of the shape of the descent trajectory.

### Direct Rendezvous in Parking Orbit - Indirect Return to Earth

This case is represented by path DEAP in Fig. 1. A lunar landing vehicle, initially at point D, aborts and rendezvous with a mother vehicle, initially at D'. Rendezvous is completed at E, and the combined vehicle then proceeds on an indirect return to Earth via path AP.

Whether or not this maneuver is feasible can be determined by investigating the overall trajectory characteristics, in terms of the velocity impulse and guidance accuracy requirements. A discussion of the mathematical model and the resulting abort trajectory requirements follows.

### Velocity requirements

The effect of orbital perturbations in causing out-of-plane motion between the landing craft and the mother vehicle is small because of the short time of flight during the descent phase of the lunar landing mission. Therefore, the gravitational forces can be assumed to follow the inverse-square law, and the orbital transfers can be considered to be coplanar transfers.

The geometry is shown in Fig. 15. The mother vehicle's orbit is assumed to be the previously considered 100-naut mile circular parking orbit. It is also assumed that the abort transfers are achieved by impulsive velocity corrections.

The transfer (abort) trajectory from the descent orbit back to the mother vehicle is not arbitrary because two constraints were imposed:

1 The two vehicles must arrive at the same point in space at the same time (time constraint).

2 The rendezvous must take place at the aposelenium of the abort trajectory. This will generally yield the minimum  $\Delta V$  requirement for direct rendezvous, i.e., less than 180 deg central angle  $\Delta \eta$ . (See Ref. 3 for typical results of this type.)

To determine the abort trajectory that meets the above constraints, it was necessary to equate the time for the aborted vehicle to travel CD to the time it takes the mother vehicle to traverse BD. Thus,  $t_{BD} = t_{CD}$ , where

$$t_{BD} = \frac{r_2^{3/2}}{\mu^{1/2}} (\pi + \eta_1 - \eta_c - \eta_2)$$

$$t_{CD} = \left[ \frac{r_2 / (1 + e_{AB})}{\mu^{1/2}} \right]^{3/2} \left\{ \pi - \left[ 2 \tan^{-1} \left( \sqrt{\frac{1-e_{AB}}{1+e_{AB}}} \tan \frac{\eta_c}{2} \right) \right. \right.$$

$$\left. \left. - e_{AB} \sqrt{1-e_{AB}^2} \frac{\sin \eta_c}{1+e_{AB} \cos \eta_c} \right] \right\}$$

However, the unknown is the true anomaly of the abort point  $\eta_c$ , and, as a result, the abort trajectory had to be determined by trial and error. The necessary iteration to determine the true anomaly was performed by the IBM 7090 digital computer. Once  $\eta_c$  was found, all the properties of the abort trajectory (velocity impulse, error coefficient, etc.) could be determined.

The abort trajectory's orbital elements, the time of transfer and the velocity impulse requirements that meet the aforementioned constraints are shown as a function of the anomaly of the abort point  $\eta$  in Fig. 16. In the case of abort shortly after separation from the mother vehicle, the time to



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rendezvous can be reduced greatly at a small  $\Delta V$  penalty by waiving the requirement for tangential intercept.

### GUIDANCE PERFORMANCE REQUIREMENTS

The guidance performance requirements for a direct abort can be related to error coefficients for the parking orbit and transfer trajectories. In this way, errors at the final point (i.e., the point of rendezvous) can be determined in terms of errors made at the initial point. Because the mother, descent, and abort orbits are analytic and the deviations from the nominal trajectory are small, it is possible to assume small perturbations. Thus, only linear terms of the Taylor series expansion are retained, making it possible to derive exact expressions for these error coefficients. Error coefficients, showing the effect of injection errors for the descent and abort trajectories were derived and are presented here:

#### Descent Trajectory

$$\frac{\partial r_1}{\partial r_2} = \left[ \frac{(1 - e) \cos \eta_1}{1 - e \cos \eta_1} - 1 \right] \frac{r_2 \frac{\partial e}{\partial r_2}}{(1 - e \cos \eta_1)}$$

$$+ \frac{(1 - e)}{1 - e \cos \eta_1} ; \frac{\partial e}{\partial r_2} = \left[ r_2 V_M^2 - \mu \right] \frac{V_M^2}{\mu e}$$

$$\frac{\partial r}{\partial V_M} = \left[ \frac{(1 - e) \cos \eta_1}{1 - e \cos \eta_1} - 1 \right] \frac{r_2 \frac{\partial e}{\partial V_M}}{(1 - e \cos \eta_1)}$$

$$\frac{\partial e}{\partial V_M} = \left[ r_2 V_M^2 - \mu \right] \frac{2r_2 V_M}{\mu^2 e}$$

$$\frac{\partial r}{\partial \gamma_0} = 0$$

Abort Trajectory

$$\frac{\partial r_2}{\partial r_1} = \frac{1}{1 - e_{AB}} \left\{ (1 + e_{AB} \cos \eta_c) + \frac{r_1}{\mu^2 e_{AB}} \left[ v_{AB}^2 r_1 - \mu \right] v_1^2 \cos^2 \gamma_{AB} \cos \eta_c + \frac{v_{AB}^2 \cos^2 \gamma_{AB}}{\mu^2 e_{AB}} \left( \frac{r_2}{r_1} \right) \left[ v_{AB}^2 r_1 - \mu \right] \right\}$$

$$\frac{\partial r_2}{\partial v_{AB}} = \frac{2r_1^2 v_{AB} \cos^2 \gamma_{AB}}{\mu^2 e_{AB} (1 - e_{AB})} \left[ \cos \eta_c + \frac{r_2}{r_1} \right] (v_{AB}^2 r_1 - \mu)$$

$$\frac{\partial r_2}{\partial \gamma_{AB}} = \frac{v_{AB}^2 r_1^2 \cos \gamma_{AB} \sin \gamma_{AB}}{\mu^2 e_{AB} (1 - e_{AB})} \left[ \cos \eta_c + \frac{r_2}{r_1} \right] (2\mu - v_{AB}^2 r_1)$$

These error coefficients are presented as functions of true anomaly of the abort point  $\eta_1$  in Fig. 17. The abort trajectory becomes considerably less sensitive to initial errors if the injection point is for  $\eta_1 > 60$  deg, which is a favorable factor for accomplishing the rendezvous in this region, despite the increased  $\Delta V$  requirement compared with that at lower values of  $\eta_1$ . However, as the velocity is increased the tolerance to errors in time (or  $\eta_1$ ) becomes critical.

Indirect Rendezvous From Elliptical Parking Orbit

In addition to the direct rendezvous previously described, a variety of indirect paths can be followed which require greater time to rendezvous but less  $\Delta V$ . An extreme example for the

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nominal parking orbits considered is a single-impulse maneuver in which the landing vehicle is allowed to remain in the inner orbit until it coincides with the mother vehicle at point A in Fig. 1 after 14 orbits, or approximately 28 hr. This time can be reduced to less than 4 hr by two-impulse maneuver in which the lander, on its first pass through point A, is inserted into an ellipse of higher energy than the mother-ship parking orbit so that its position coincides with the mother ship the second time it reaches A.

A further time reduction to about 2 hr can be obtained by the three-impulse maneuver illustrated in Fig. 18. Here, the lander is placed immediately on an ellipse AB, which is changed by an impulse at point B to a new ellipse BC, which intercepts the mother vehicle at point C. The impulsive velocity increment required to perform this maneuver is shown as a function of abort position in Fig. 19. Time and  $\Delta V$  for the various abort with rendezvous modes are listed in Table 4.

Another approach that has been considered is to descend from the circular orbit to 50,000 ft (start of powered descent) via an elliptical orbit that has a period equal to the period of the circular orbit and has a periselenium of 50,000 ft. The total  $\Delta V$  required to inject into this elliptic orbit from a 100 naut mile circular orbit is approximately 460 fps. The advantage of this "synchronous orbit" approach is that, if an abort is required prior to start of terminal descent, the vehicle can continue along the elliptic path to the injection point and achieve rendezvous by applying an additional 460ft/sec  $\Delta V$ . Maximum time for rendezvous would be equal to the period of orbit, or 126 minutes.

### Direct Rendezvous From Lunar Surface - Indirect Return to Earth

In this case, a landing vehicle is launched from the surface of the moon and rendezvous with the mother vehicle in a known orbit. Launch occurs from a point outside of the plane of the mother orbit. The mother vehicle is assumed to be in a 100-mile circular orbit inclined 10 deg to the lunar equator. The lander vehicle is launched from the equator at points 13 deg and 26 deg from the descending node line, corresponding to 24- and 48-hr stay times, respectively.

Fig. 20 shows the geometry of the situation. Results obtained from the investigations are shown in Fig. 21. Total velocity increment is plotted as a function of downrange angle of intercept for two-impulse tangential transfer trajectories. The second impulse is applied at the tangential intercept point to match the velocity and heading angle of the two

vehicles. The initial heading angle at launch (shown on Fig. 21) determines the downrange angle at intercept and the angle at which the lander vehicle intersects the plane of the vehicle in orbit. From such curves, launch windows can be determined for each value of  $\Delta V$  provided in the lander vehicle.

For example, for a 48-hr dwell time on the moon, the minimum  $\Delta V$  required is approximately 5950 fps and corresponds to a downrange intercept angle of 50 deg. The heading angle required at launch to produce this condition is approximately 102.5 deg. However, if after the same dwell time on the moon the ferry vehicle had available a total velocity impulse of 6250 fps, then the permissible downrange angle of intercept would extend to a range between 23 deg and 140 deg, with the corresponding heading angles at launch varying between 111 deg and 94 deg respectively. This yields a launch-time window of 22 min.

The geometry of the nominal moon-to-Earth trajectory of Fig. 2 differs from that shown in Fig. 20 in that the inclination  $i_m$  is higher, as shown in Fig. 22. Hence, the  $\Delta V$  required to rendezvous with a parking orbit in this plane would be somewhat higher than is shown in Fig. 21 for an equatorial launch site, due to the increased heading-angle change required at intercept.

## CONCLUSIONS

- 1 In the preliminary planning of manned lunar-mission profiles, abort considerations can significantly influence trajectory selection and operations sequencing.
- 2 In certain instances, simple approximations to the flight mechanics can be used in determining required abort velocity increments and time-to-return. Coplanar central force-field equations can be used in most abort situations near the moon. However, the validity of approximating direct aborts by a series of elliptical transfers to a nominal moon-to-Earth return trajectory requires further study.
- 3 Direct abort and return to Earth by transferring to a nominal return trajectory appear of very limited value because return-time saved is low and velocity requirements are high.
- 4 In selecting a nominal terminal descent trajectory for a manned mission, abort considerations and fuel economy favor shallow-angle descents. This must be balanced against error

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sensitivity, crew participation in landing, and terrain clearance requirements, which tend to favor steep descents.

5 Indirect returns, involving abort and direct or indirect rendezvous, can be accomplished in a number of ways. Indirect rendezvous takes up to 1-1/2 hrs longer than direct rendezvous but requires much less velocity increment.

### ACKNOWLEDGMENTS

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Table 1 Nominal return trajectory

	Computed trajectory	Hyperbola, $e = 1.485$
Velocity at periselenium, fps	8159	8159
Radius at $\eta = 90^\circ$ , $10^6$ ft	16.410	16.247
Velocity at $\eta = 90^\circ$ , fps	5831.9	5854.4

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Table 2 Direct aborts from circular parking orbit,  $\Delta V$  penalty vs  $\Delta t$  saved

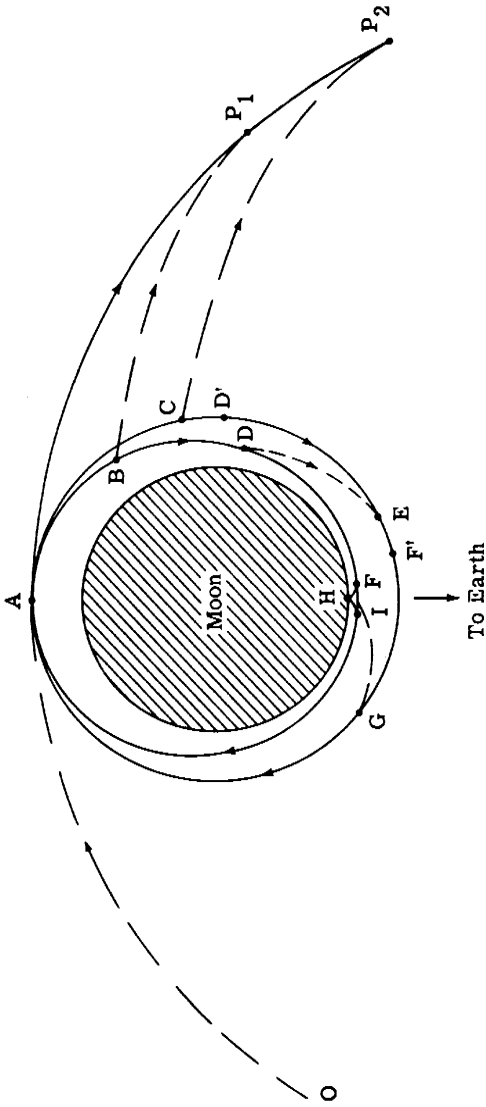
Abort point, $\eta_1$	$\Delta t$ saved, hr	$\Delta V$ Penalty, fps	Insertion point, $\eta_2$
30°	1.88	2700	60.5°
45°	1.95	4350	80.0°
75°	2.40	6080	115.0°

Table 3 Direct aborts from elliptical parking orbit;  $\Delta V$  penalty vs  $\Delta t$  saved

Abort point, $\eta_1$	$\Delta t$ saved, hr	$\Delta V$ Penalty, fps	Insertion point, $\eta_2$
30°	1.95	2750	63°
45°	1.80	4250	85°
75°	2.90	6500	115°

Table 4 Indirect Rendezvous  $\Delta V$  vs  $\Delta t$

No. of impulses	No. of orbits	Time to rendezvous, $\Delta t$	Velocity Increment at rendezvous $\Delta V$ , fps
1	14	28	125
1	1 ("synchronous")	2	460
2	1 to 2	2 to 4	325
3	1	2	130 to 380 (Fig. 19)



DEPICTED TRAJECTORIES

Nominal		Abort	
OA	= Earth - Moon Transfer	Direct Returns	
AP <sub>1</sub> P <sub>2</sub>	= Moon - Earth Return	BP <sub>1</sub>	= From Elliptical Parking Orbit
ACG	= Circular Parking Orbit	CP <sub>2</sub>	= From Circular Parking Orbit
ABF	= Elliptical Parking Orbit	Indirect Returns	
FH	= Terminal Descent	BAP	= From Elliptical Parking Orbit
HI	= Lunar Ascent	CAP	= From Circular Parking Orbit
		FLAP	= From Terminal Descent
		DEAP	= Rendezvous from Elliptical Parking Orbit
		FGAP	= Rendezvous from Terminal Descent

Fig. 1 Abort situations considered

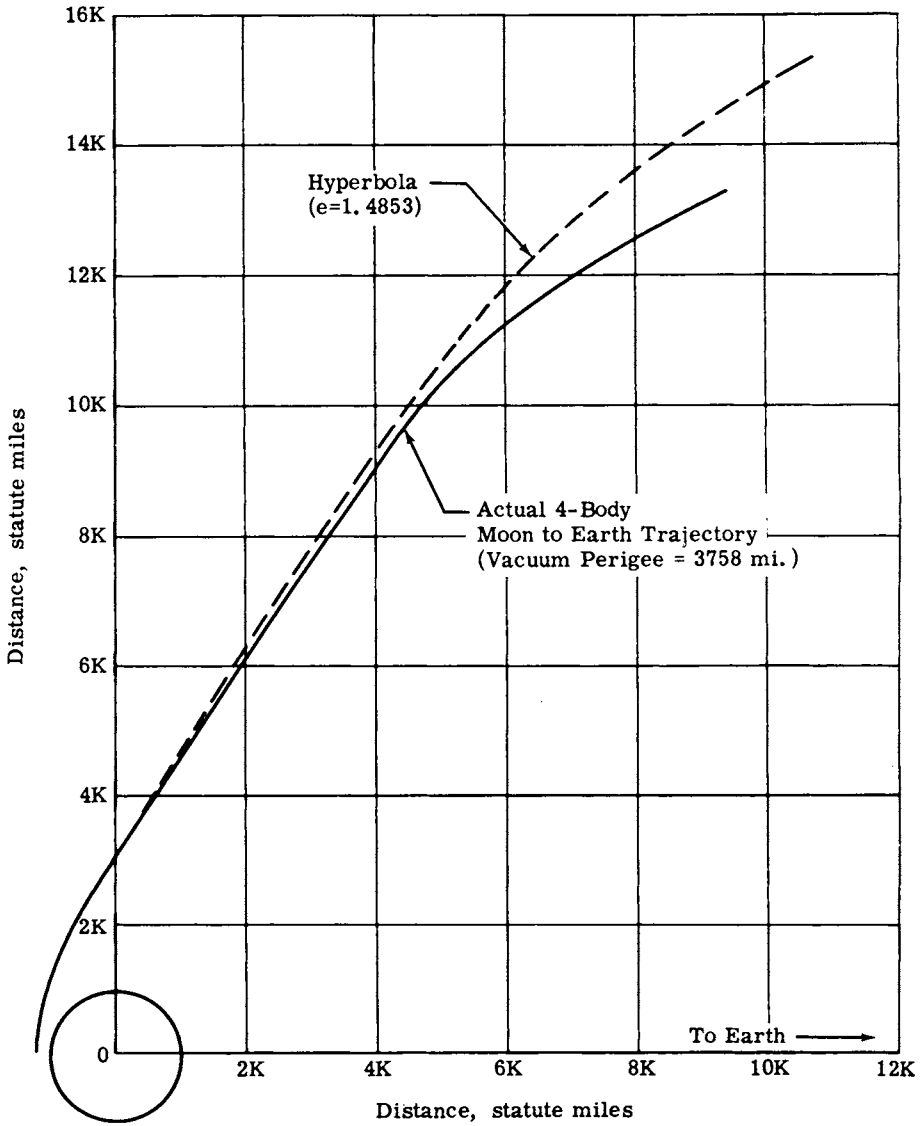


Fig. 2 Moon-to-Earth trajectory



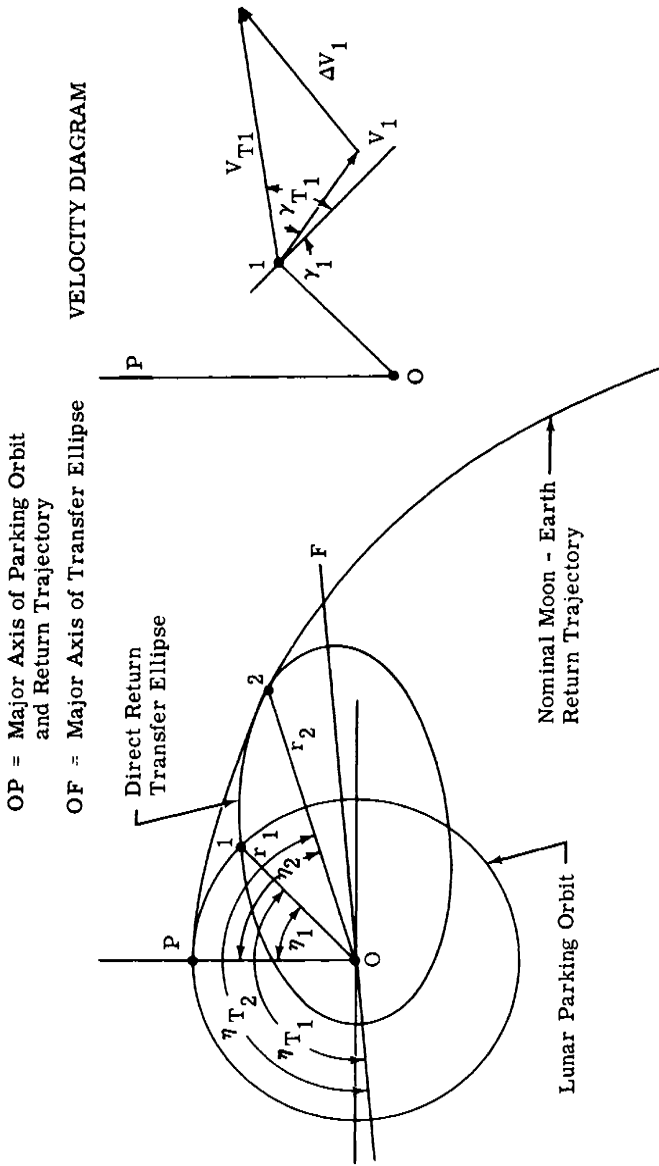


Fig. 3 Direct about geometry

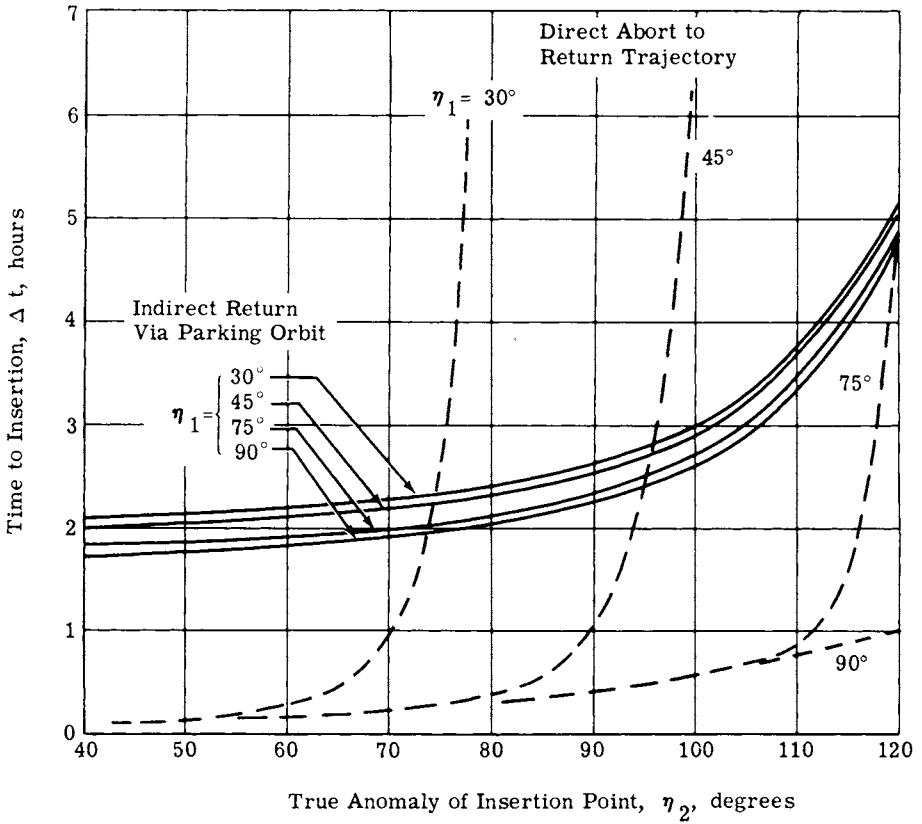


Fig. 4 Elapsed time for direct and indirect returns from 100-naut mile circular parking orbit

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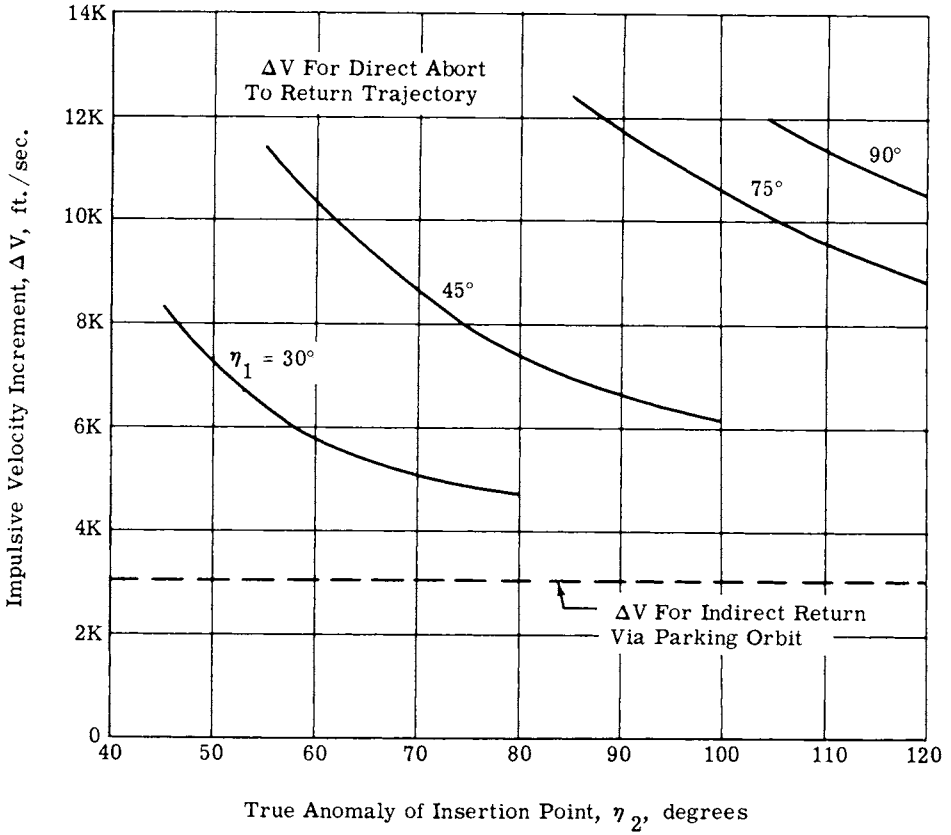


Fig. 5 Velocity increment for direct and indirect returns from 100-naut mile circular parking orbit

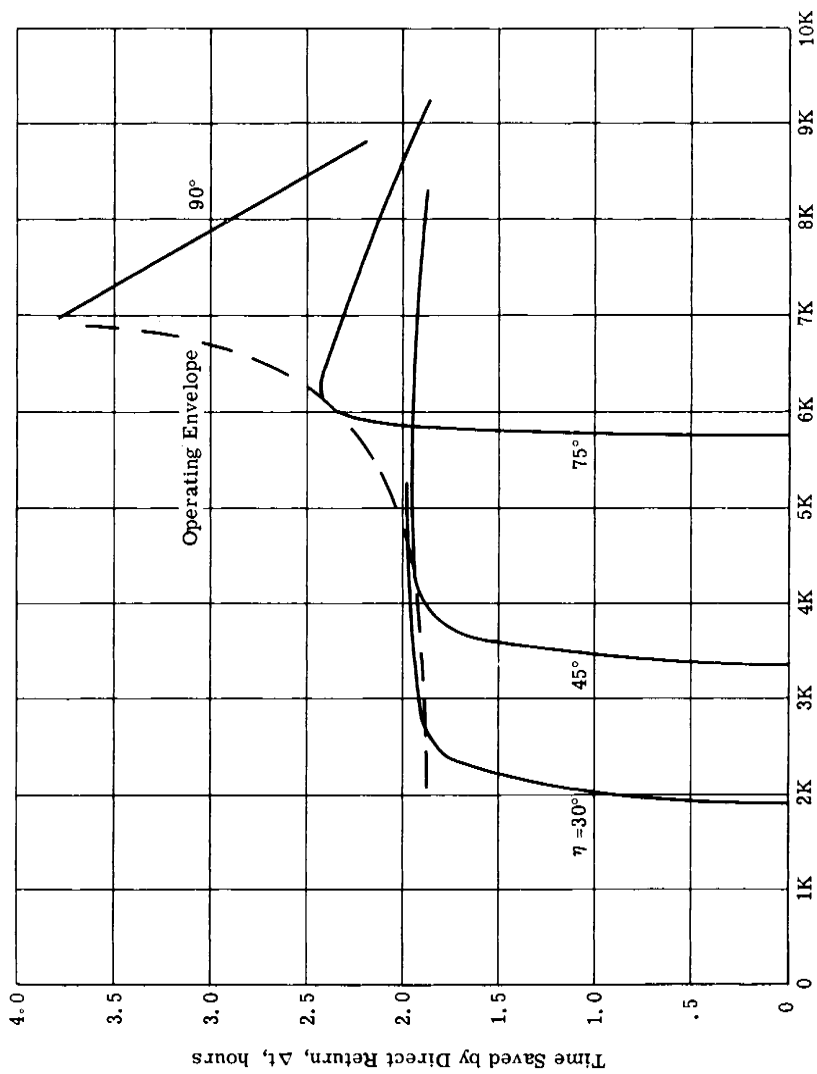


Fig. 6 Time saved vs velocity increment penalty for direct return abort from 100-naut mile circular parking orbit

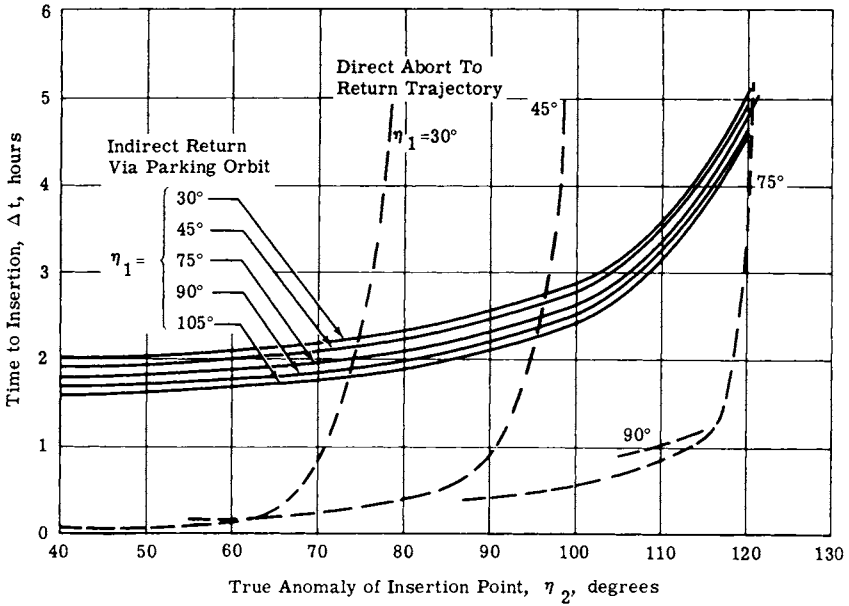


Fig. 7 Elapsed time for direct and indirect returns from elliptical parking orbit

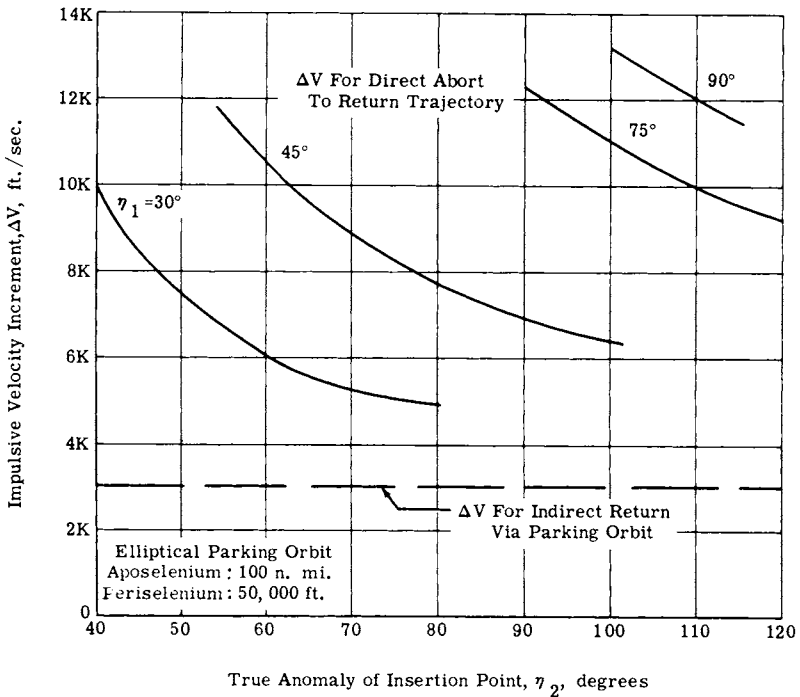
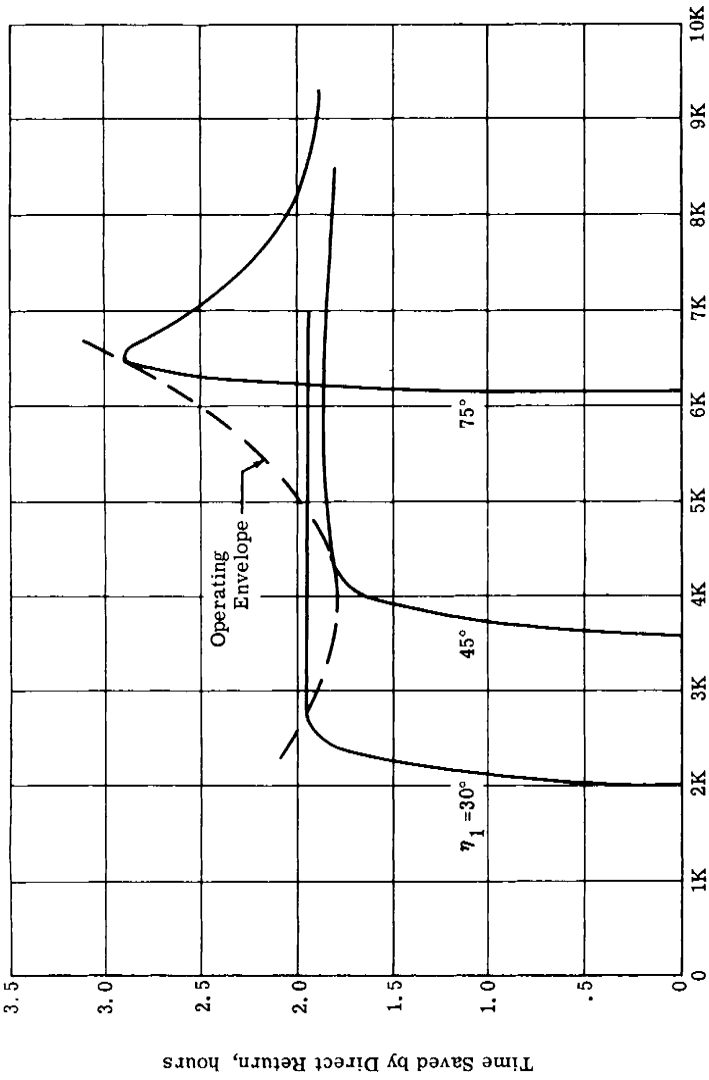


Fig. 8 Velocity increment for direct and indirect returns from elliptical parking orbit



Velocity Increment Penalty,  $\Delta V$ , for Direct Return, ft./sec.

Fig. 9 Velocity increment vs time saved for direct abort from elliptical parking orbit

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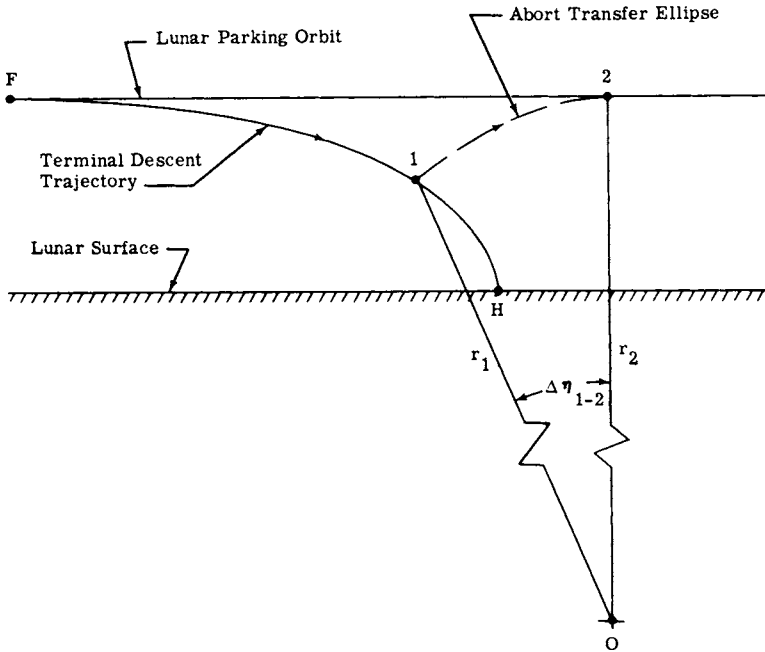


Fig. 10 Geometry of abort from terminal descent

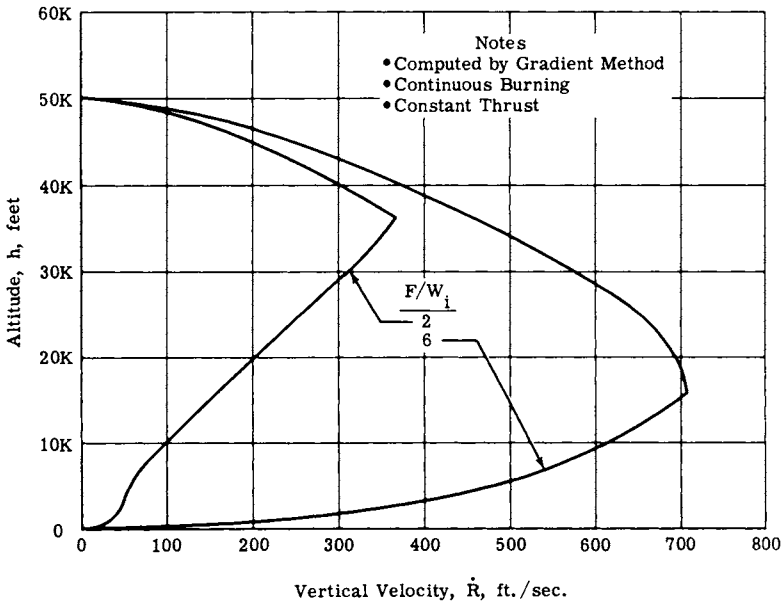


Fig. 11 Minimum-time terminal descent trajectories, vertical velocity vs altitude

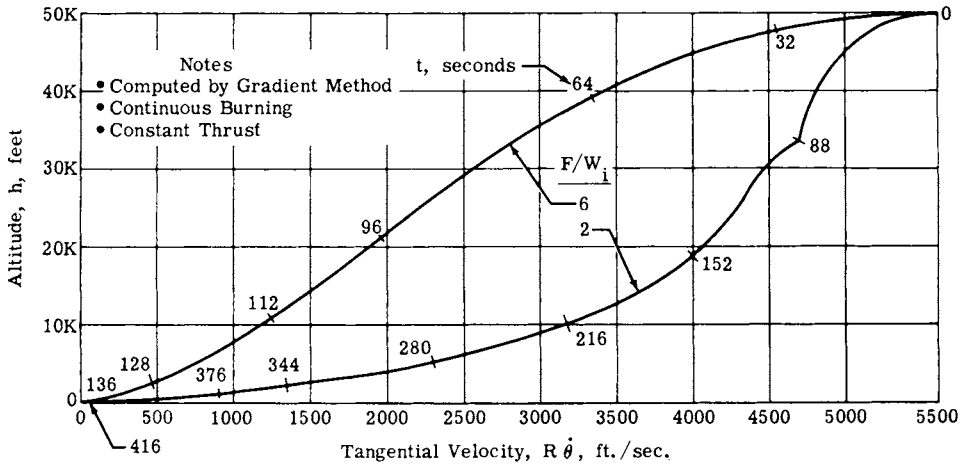


Fig. 12 Minimum time terminal descent trajectories, tangential velocity vs altitude

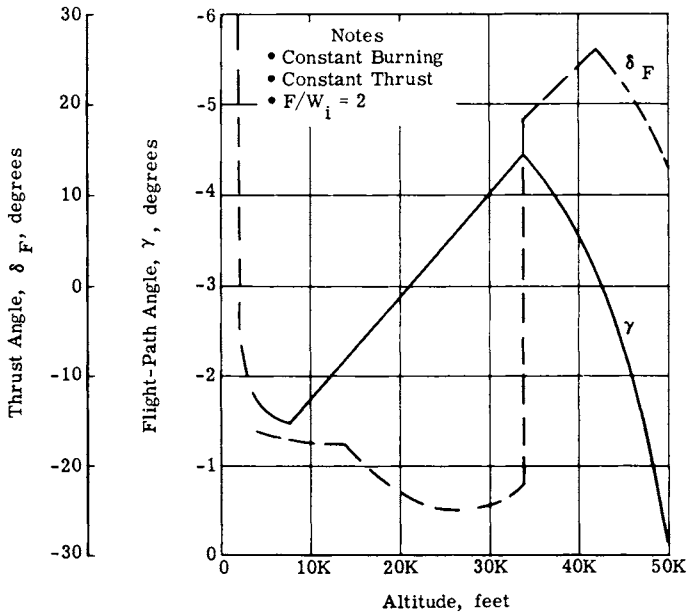


Fig. 13 Terminal descent trajectory, thrust and flight-path angle vs altitude



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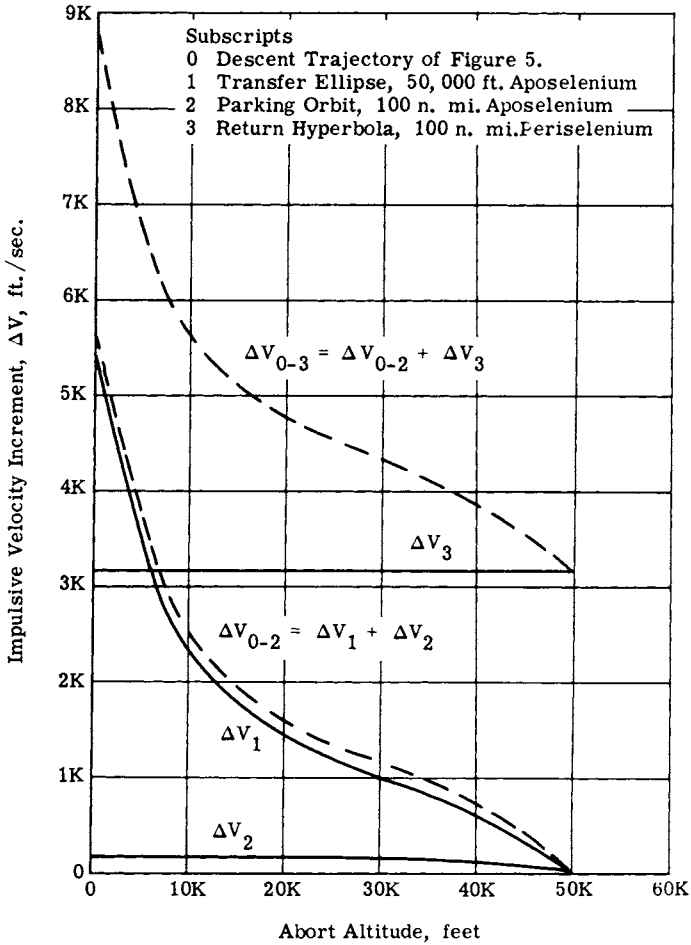
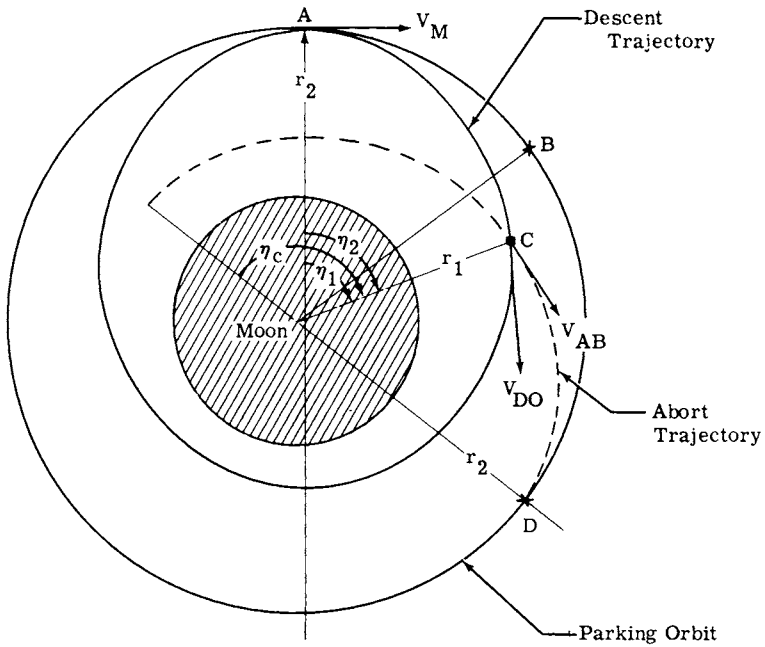


Fig. 14 Abort from terminal descent, velocity increment vs abort altitude



- A = Point of Injection
- B = Position of Mother Vehicle at Time of Abort
- C = Abort Point
- D = Rendezvous Point

Fig. 15 Coordinate system used in direct rendezvous abort analysis

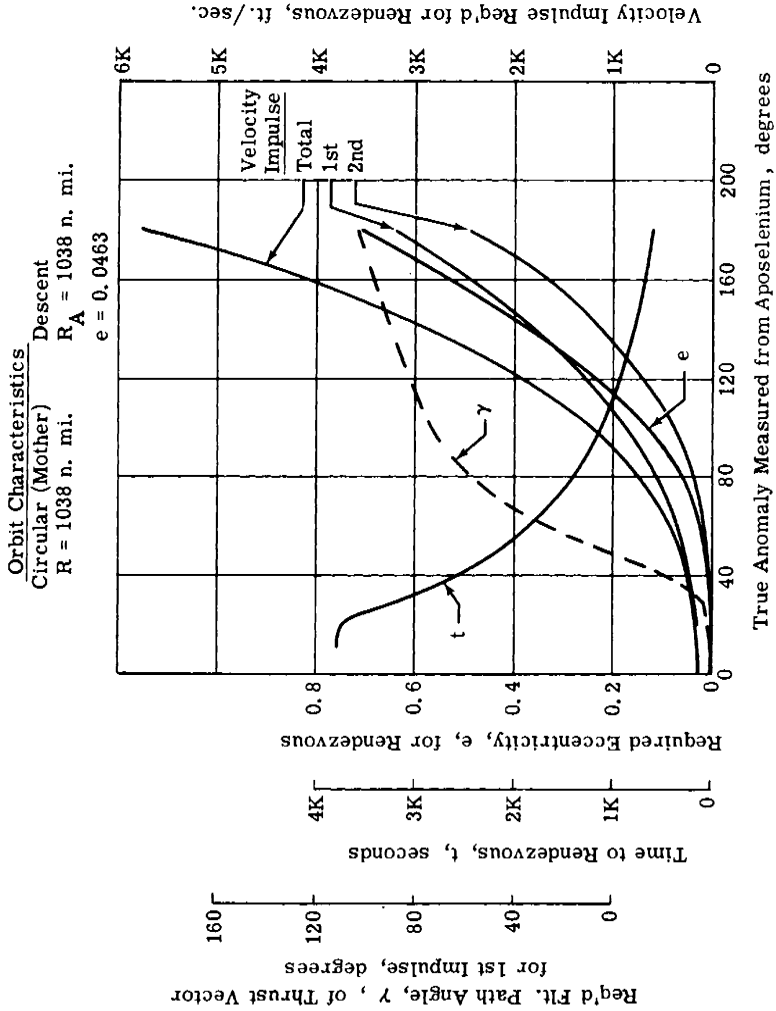
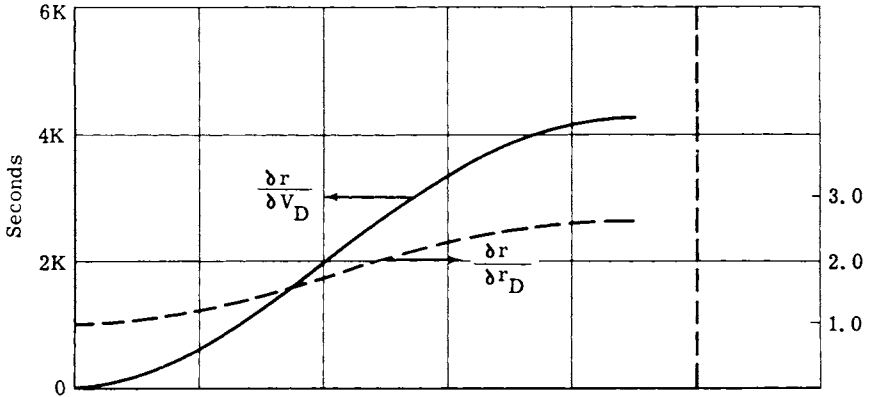


Fig. 16 Abort trajectory characteristics for direct rendezvous

A. Error in de-orbit initial conditions on descent trajectory.



B. Error in abort trajectory initial conditions on rendezvous point.

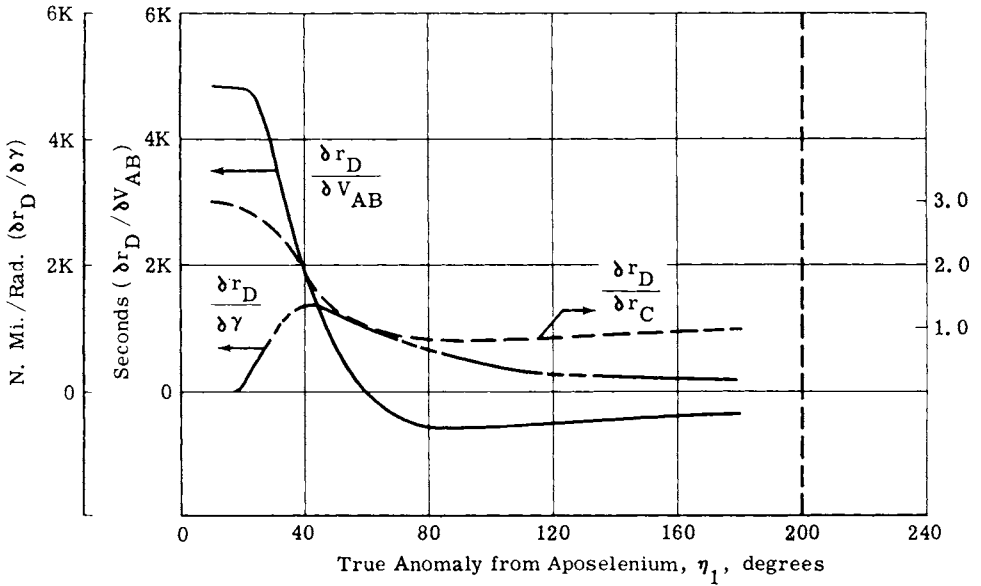
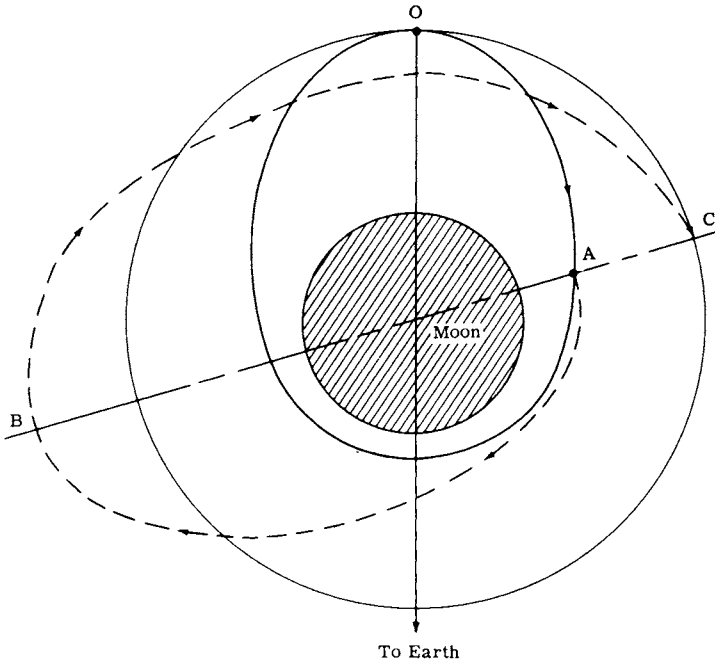


Fig. 17 Error coefficients for direct rendezvous abort

# TECHNOLOGY OF LUNAR EXPLORATION



- OA = Travel Along Descent Orbit
- A = Point of Vehicle Abort, 1st Impulse Applied
- AB = One-Half of 1st Transfer Orbit
- B = Apolune of 1st Transfer Orbit, 2nd Impulse Applied
- BC = One-Half of 2nd Transfer Orbit
- C = Rendezvous Point on Circular Orbit, 3rd Impulse Applied

Fig. 18 Three-impulse indirect rendezvous

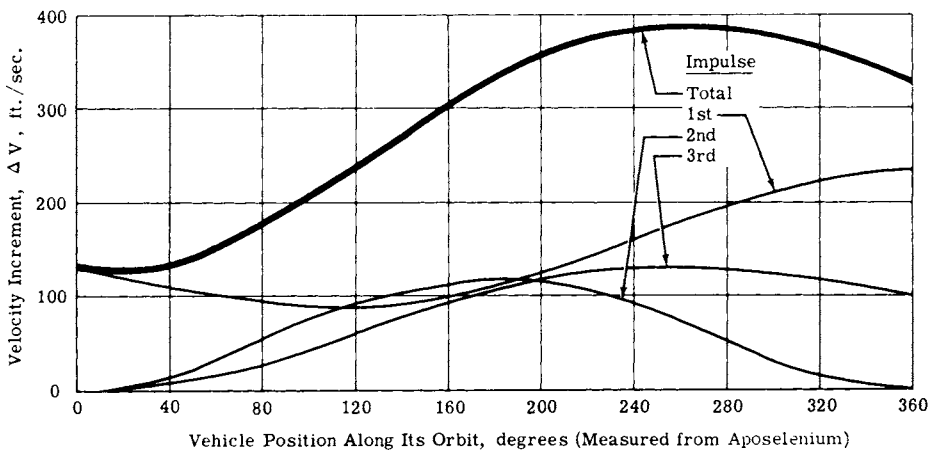


Fig. 19 Impulse required for three-impulse indirect abort rendezvous (rendezvous time, 130 min)

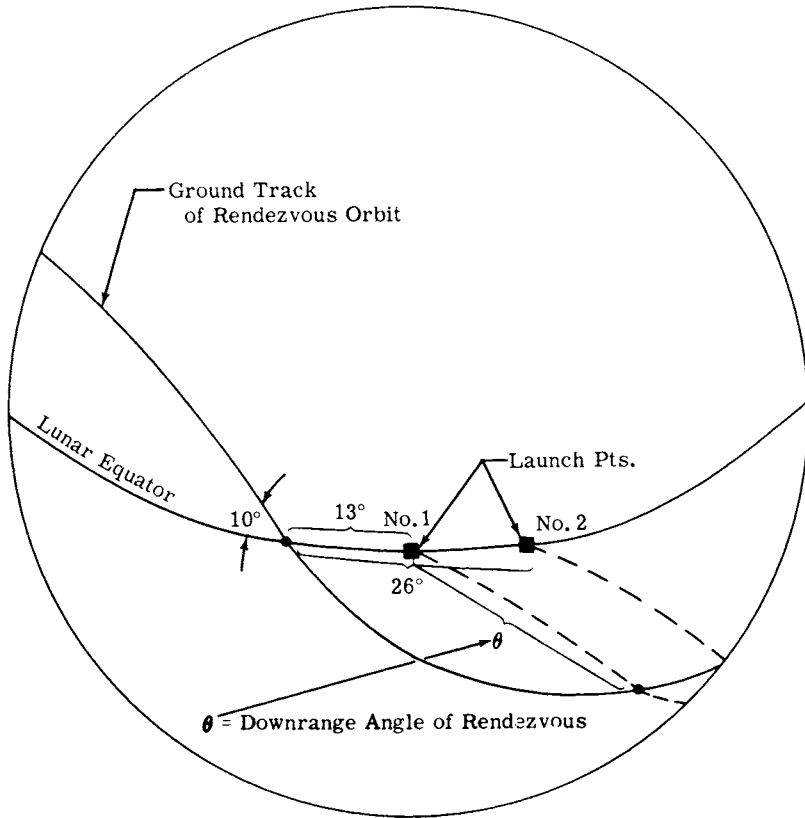


Fig. 20 Geometry of launch from lunar surface

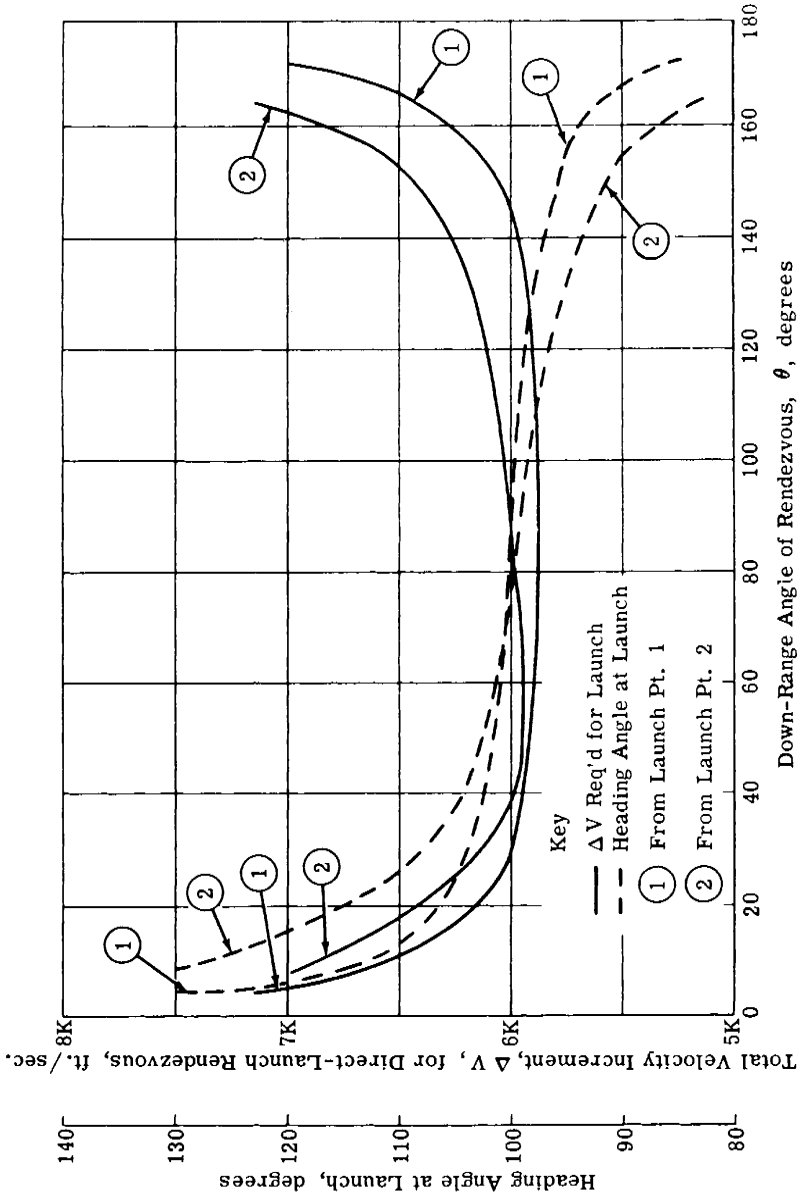


Fig. 21 Requirements for direct launch and rendezvous

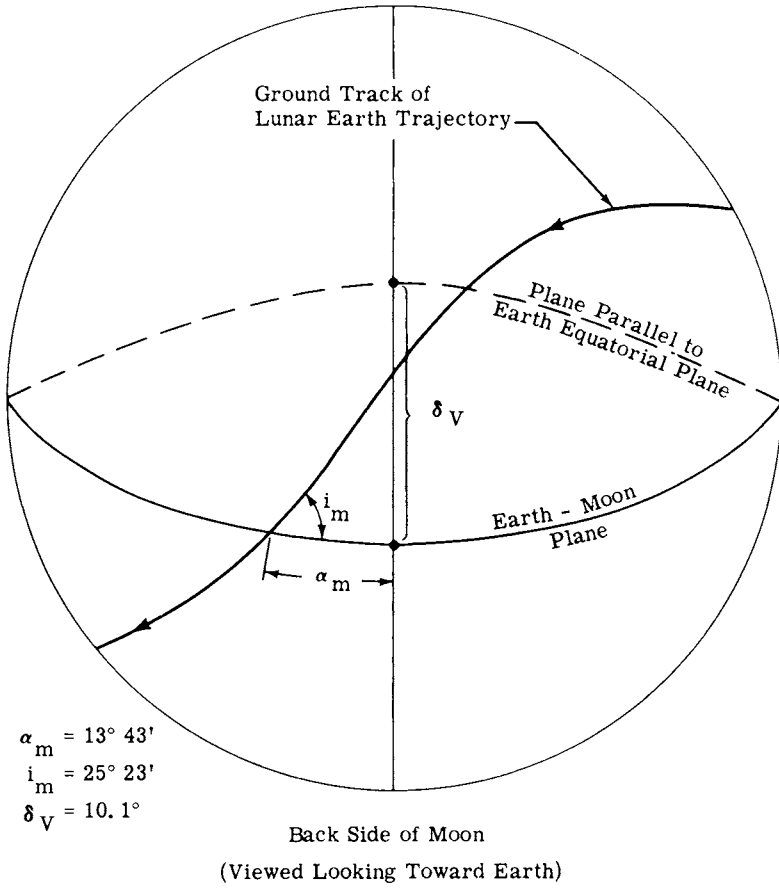


Fig. 22 Geometry of nominal moon-to-Earth trajectory