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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC INTERNAL NOTE NO. 68-FM-149

June 20, 1968

CIRCUMLUNAR COMMUNICATIONS USING TWO SATELLITES IN LUNAR POLAR ORBIT

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PROJECT APOLLO

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CIRCUMLUNAR COMMUNICATIONS USING TWO

SATELLITES IN LUNAR POLAR ORBIT

By John T. McNeely and Ellis W. Henry

SUMMARY

An analysis has been made to determine the feasibility of deploying two satellites in lunar polar orbit and using these satellites to relay communications to the far side of the moon. The command and service modules (CSM) on an Apollo free-return lunar landing mission can be used to transport these communications satellites (COMSAT's) to lunar proximity if the Saturn V launch vehicle stack weight is increased or the service propulsion system (SPS) fuel is off-loaded or both. The Saturn V launch vehicle is capable of injecting more payload on a translunar trajectory than the present stack weight. Also, after expending the entire ΔV budget, approximately 1000 lb of fuel can be brought back to earth. This fuel could be replaced with additional payload such as COMSAT's. The analysis showed that the two COMSAT's can weigh as much as 1600 lb each.

COMSAT propulsion and guidance systems requirements as well as the effects of the earth and the sun on the COMSAT's orbits were investigated. It is shown that third-body perturbations cause a satellite initially placed in a 5000-n. mi. radius circular polar orbit to impact the lunar surface within 14 months if correction maneuvers are not made. Also, communication coverage suffers significantly if phasing between the COMSAT's is not kept within certain limits by periodic maneuvers. A ΔV of 10 fps every 3 months for each COMSAT will correct the orbits for third-body perturbations, and a ΔV of 6 fps each month for one COMSAT will hold phasing between the COMSAT's to within $\pm 10^{\circ}$. Earth-based tracking is obviously essential to these correction maneuvers. Considering the above requirements, approximately 42 percent of the total COMSAT weight must be allocated to the propulsion and guidance systems if the expected lifetime in lunar orbit is 2.5 to 3 years.

INTRODUCTION

This study was made at the request of the Advanced Planning Support Office of the Flight Operations Directorate to determine the technique and ΔV budget for deployment of communication satellites into lunar orbit. The proposed use of these satellites is to provide a communications link between the earth and the CSM when the CSM is on the backside of the moon. Several satellite orbit configurations that can be used to relay communications to the backside of the moon were investigated in reference 1. As shown in reference 1, two satellites in high altitude polar orbits can provide better than 99 percent communications coverage over a period of one sidereal month, and this configuration is assumed for the study. The study is divided into six parts: (1) lunar orbit configuration, (2) COMSAT deployment and phasing, (3) COMSAT guidance requirements, (4) COMSAT orbit lifetime study, (5) COMSAT velocity requirements, and (6) COMSAT payload weight. The equations essential to the trajectory analysis associated with the study were obtained from references 2 and 3.

LUNAR ORBIT CONFIGURATION

The lunar orbit configuration assumed as discussed in the Introduction, is shown schematically in figure 1. The large dashed circle represents the orbit of the COMSAT's, which are in the same plane, phased 180° apart, and have an orbital radius of 5000 n. mi. each and an orbital inclination of 90°. This orbital radius was chosen after considering that a lower orbital radius would result in poorer communications coverage but a lower ΔV budget, whereas a higher orbital radius would result in increased communication coverage but increased communication distance from the CSM to the COMSAT's and an increased ΔV budget.

The communication coverage of this COMSAT configuration has a blackout region on the lunar surface consisting of a 354-n. mi. wide band. This band rotates and moves across the lunar surface as the COMSAT's circle the moon and the orbit nodal position regresses. The band width decreases to 0 for a CSM orbital altitude of 17 n. mi. or greater. If the COMSAT orbital radius is increased to 10 000 n. mi., the width of the lunar surface band decreases to 176 n. mi., and the CSM orbit altitude of no blackout decreases to 4.2 n. mi. However, the communication distance between the COMSAT's and the CSM is undesirably doubled. Also, as will be shown later, the higher radius orbits are affected much more by third-body perturbations and thus require more ΔV for corrections.

Communications blackouts will occur with any lunar orbit configuration in which both the CSM and the COMSAT is occulted by the moon. This is illustrated in figure 2, which shows the earth-moon-satellite geometry for one synodic month. Since the orbital plane of the COMSAT's retains its inertial orientation as the moon revolves around the earth, the plane cuts the "no-communication region" (NCR) at positions near B and D. When this occurs, it is possible to have both a COMSAT and the CSM occulted by the moon, a situation resulting in a communication blackout. For the COMSAT configuration previously described, there exists approximately 1.5 days when communication is interrupted at B and at D. Between B and D and also D back to B, there is approximately 12 days when communication is continuous.

Table I, which was taken from reference 1, shows the communications coverage that can be achieved over one sidereal month for the configuration describ d above for the two COMSAT's. The CSM was assumed to be in a lunar equatorial orbit at an altitude of 80 n. mi. The communication blackouts lasted from 8 to 46 minutes and were 5 or 6 CSM revolutions apart. After the fourth and eighth blackout, there was approximately 12 days of continuous communications. Over the sidereal month the CSM could communicate with the earth directly 62.04 percent of the time. The CSM could communicate through the first COMSAT 24.97 percent of the time, and through the second COMSAT 12.36 percent of the time. This left 0.63 percent of the time with no communication. It is not believed that a CSM orbit altitude reduction to 60 n. mi. would significantly decrease the total communication time.

COMSAT DEPLOYMENT AND PHASING

A technique and ΔV budget were developed for deploying a COMSAT into a 5000-n. mi. radius circular polar lunar orbit. A three-impulse technique was used and is shown schematically in figure 3, which gives a view from above the lunar north pole. Both COMSAT's are deployed at the lunar sphere of influence (SOI) by the CSM, which is on a nominal Apollo free-return trajectory. At the SOI, the necessary azimuth and flight-path angle changes are made by the COMSAT propulsion system in order to achieve the 90° inclination and desired pericynthion altitude. The second maneuver is made at pericynthion on the approach hyperbola to achieve an elliptical lunar orbit. A third impulse is made at apocynthion of the ellipse as orbit requirements dictate.

Phasing of the COMSAT's 180° apart in the 5000-n. mi. radius circular orbit is obtained by using two deployment techniques. The techniques are shown schematicly in figure 4, along with the typical ΔV cost. At the SOI, one COMSAT is directed to a trajectory with a 5000-n. mi. pericynthion radius for a ΔV of 800 fps. The second COMSAT is directed to a trajectory with a 2790-n. mi. pericynthion radius, requiring a ΔV of 560 fps. Upon arriving at pericynthion, the first COMSAT circularizes, requiring a ΔV of 2415 fps, while the second COMSAT deboosts into a 2790 by 5000-n. mi.

(perigee by apogee radius) ellipse for 2044 fps. After one-and-one-half revolutions, the second COMSAT circularizes at a 5000-n. mi. radius for 367 fps. Guidance is assumed perfect, the COMSAT's are phased 180° apart and the elapsed time since their deployment at the SOI is approximately 36 hours.

COMSAT GUIDANCE REQUIREMENTS

The COMSAT's are assumed to be ground-controlled after deployment from the CSM. Ground control will impose additional requirements on the Mission Control Center during the period of lunar orbit insertion (LOI) for the lunar mission. The COMSAT's will pass their respective pericynthion positions approximately 45 minutes to 1 hour after the CSM makes its LOI maneuver.

A study was made of the pericynthion errors due to the pointing, proportional, and cutoff errors of the guidance system onboard the COMSAT's. The purpose of the study was to assess the effect of these errors on the COMSAT velocity budget. Figures 5, 6, and 7 show the 3σ navigational accuracy in terms of projected range, altitude, and track errors at pericynthion on the approach hyperbola. The radius of periapsis is 2790 n. mi. for figures 5(a), 6(a), and 7(a), and 5000 n. mi. for figures 5(b), 6(b), and 7(b). The assumed CSM state vector error at the time of satellite separation at SOI is 4 n. mi. in position and 4 fps in velocity.

Figures 5(a) and 5(b) show the variation in position error (3σ) as a function of proportional error. The pointing error was constrained to be not greater than 1°, and the cutoff error was constrained to be not greater than 1/2 fps. The variation in position error with pointing error can be seen in figures 6(a) and 6(b). The proportional and cutoff errors were constrained to be not greater than 1 percent and 1/2 fps, respectively. The variation in position error with cutoff error is shown in figures 7(a) and 7(b). In this case, the proportional and pointing errors were constrained to 1 percent and 1°, respectively. This figure shows that position error is very insensitive to cutoff error. However, a nominal value of 1/2 fps was chosen as this seems necessary when making phasing maneuvers in lunar orbit. For example, consider a 5000-n. mi. radius circular orbit about the moon. The semi-major axis can be increased or decreased by 25 n. mi. for 6 fps which corresponds to about 3 deg/day in phasing. The cutoff error is the most sensitive parameter in this case.

If we choose nominal values for proportional, pointing, and cutoff errors, namely 1 percent, 1°, and 1/2 fps, respectively, and a value for the pericynthion radius of 2790 n. mi., the projected range error at pericynthion is 236 n. mi., the projected altitude error is 161 n. mi., and the projected track error is 108 n. mi. For a pericynthion radius of

5000 n. mi., the projected range error at pericynthion is 270 n. mi., the projected altitude error is 245 n. mi., and the projected track error is 190 n. mi. These guidance errors are fairly pessimistic; however, the resulting pericynthion errors are not considered to be excessive. The result of these errors is reflected in the COMSAT ΔV budget which is discussed in a later section.

COMSAT ORBIT LIFETIME STUDY

In conjunction with the main context of this study, it was felt necessary to investigate the effects of perturbation forces upon the initial orbit over a significant period of time. There are a number of analytic expressions for third-body perturbations, but it is difficult to determine whether they correctly apply to the special case of eccentricity equal to zero. Also, while the polar orbit is inclined 90° to the moon equator, the analytic expressions frequently use a specialized coordinate system, thus requiring a coordinate transformation before the equations are applicable. For accuracy and simplicity, an N-body numerical integration program, which was readily available, was used for certain discrete cases at various altitudes. The results were somewhat unexpected, and the brief number of cases investigated are inconclusive as far as generality of effects. It is now felt that more investigation is desirable and should be treated in detail in a separate paper.

The significant findings as pertaining to this feasibility study can be summarized briefly: For lunar polar orbits initially circular at radii of 4000, 5000, and 6000 n. mi., the perturbation effects limit the uncorrected lifetime (i.e., time to surface impact) to 20, 14, and 10 months, respectively. The plane of motion of the satellite remains essentially polar throughout this lifetime. It was determined that these effects were almost totally the result of the earth perturbing the lunar satellite, thus the higher altitude orbits receive the more pronounced effects.

The data just summarized are shown on figures 8 through 12. Each of figures 8, 9, and 10 show periapsis, apoapsis, and their sum (major axis) for the radii considered. The inclination and its variation is shown in figure 11 for the 5000-n. mi. radius.

The satellite plane exhibits a slow (about 10 deg/year) inertial precession of the selenocentric ascending node, as shown in figure 12. The moon rotates on its polar axis approximately 13 deg/day, so that the selenographic node changes also about 13 deg/day. Thus, the precession of the satellite plane has only a very small effect on the selenographic node.

The figures and data for this section includes all significant perturbations due to remote gravitational bodies, but does not include the triaxial effects of the nonspherical moon. It was verified numerically that the aspherical effects were very small by comparison with those of the earth as a third body for the altitudes considered. As the periapsis altitude becomes appreciably near the surface, the neglected effects become more pronounced. Thus the impact time is approximate rather than exact, but should not change more than a few days. If the satellite orbit is corrected before it becomes appreciably noncircular, by far the most significant perturbation is due to the earth.

COMSAT VELOCITY REQUIREMENTS

The COMSAT ΔV requirements shown in table II were calculated considering the orbit configuration, guidance requirements, and perturbation effects discussed in preceding sections. The ΔV requirement for inserting a COMSAT into a 2790 by 5000-n. mi. radius elliptical orbit is 2604 fps. An additional 367 fps is required for circularizing the orbit at 5000 n. mi. for a total of 2971 fps. The ΔV cost for inserting a COMSAT directly into a 5000-n. mi. radius circular orbit is 3215 fps. Considering the pericynthion errors previously described, each COMSAT has 100 fps. allotted for correction maneuvers in lunar orbit for initial phasing. This amount is sufficient only if ground tracking is available to predict the actual errors at pericynthion and the pericynthion burn retargeted to account for these errors. The COMSAT which was initially directed to the lower pericynthion altitude has 329 fps allotted for correction maneuvers in lunar orbit for future phasing. The other COMSAT has 85 fps allotted for these maneuvers. This unequal ΔV capability brings the total ΔV capability of each COMSAT to 3400 fps.

Considering that the COMSAT orbit inclination is 90°, the orbital elements can likely not be determined accurately except when the moon occults the orbit track, which it does twice each month. It was assumed that ground-based tracking could determine the semi-major axis of the COMSAT's orbits to within 3 n. mi. or the period of the orbits to within 1.2 minutes, so that the drift rate between the two COMSAT's could be cut to 0.7 deg/day. The correction accuracy is approximately 1/4 deg/day based on 1/2 fps cutoff error, 1 percent proportional error, and 1° pointing error. The velocity required to hold COMSAT phasing within $\pm$ 10 degrees should therefore be less than 6 fps per month. This ΔV need only be input by one COMSAT.

In addition to the phasing (COMSAT drift) problem, ΔV must be expended to keep the COMSAT's orbits circular. As was pointed out previously, a COMSAT initially in a 5000-n. mi. radius circular polar lunar orbit will impact the lunar surface in 14 months if no corrections are made. The problem is only lessened by choosing a lower orbital radius, but this decreases the amount of communication coverage achievable. If the orbital radius is 5000 n. mi., the orbit can be corrected every 3 months for 10 fps per correction. Adding this velocity requirement to the 6 fps per month needed to keep the COMSAT's phased properly, the total velocity requirement for one COMSAT is slightly less than 10 fps per month whereas the velocity requirement for the other COMSAT is 10 fps every 3 months. The velocity available for correction is 329 fps in one COMSAT and 85 fps in the other COMSAT. This is sufficient for approximately 2.5 to 3 years of operation. If the guidance and tracking capabilities discussed earlier are realistic and cannot be improved, then additional lifetime can only be obtained at the expense of decreased communications coverage and/or decreased payload weight.

COMSAT PAYLOAD WEIGHT

In order to determine the amount of excess payload that can be carried on a lunar landing mission, a nominal CSM trajectory and ΔV budget must be chosen. Also, spacecraft weights must be specified. The SPS ΔV requirements which define the nominal trajectory for this study are shown in table III. The LOI ΔV is 3100 fps. The ΔV allotted for the lunar orbit plane change and for variations in lunar stay time and transearth injection is 2800 fps. The amount of 790 fps has been allocated for LM rescue. The total SPS ΔV to be expended with the CSM and LM attached is 3230 fps. The total SPS ΔV to be expended with the LM detached is 3830 fps. This ΔV budget is intended to be sufficient to include most of the presently planned Apollo launch windows, which are discussed in reference 4. A summary of the CSM and Saturn V S-IVB stage weights is given in table IV.

At the present time, the excess payload that can be carried on an Apollo free-return lunar landing mission is very small. Two methods, but three modes, of increasing CSM payload are shown in table V. In mode 1, the stack weight (total weight of the CSM, LM, and SLA) is increased, and SPS fuel is not off-loaded. The S-IVB is capable of injecting more weight on a translunar trajectory than the present stack weight limit. In mode 2, the stack weight remains the same and SPS fuel is off-loaded. Considering the SPS ΔV requirements previously discussed (complete ΔV budget), over 1000 lb of fuel is brought back to earth. This fuel could be replaced with additional payload such as the COMSAT's. It should be emphasized that the amount of excess fuel is very dependent upon spacecraft weights, which are continually growing. The 1000 lb of excess fuel discussed here is obtained with the spacecraft weights given in table IV and for approximately 75% of the Apollo launch windows. In mode 3, the stack weight is increased and SPS fuel is off-loaded. Since this mode uses both methods of increasing the payload, the largest excess payload is obtainable using this mode.

The excess payload that can be carried on an Apollo free-return lunar landing mission is shown in table VI for the three modes discussed above. In the case of mode 1, the stack weight is increased to 102 374 lb. In this configuration, should the COMSAT's not be deployed, the lunar landing mission, including LM pickup, could still be completed. This is assuming that the SPS tanks were initially full. The total weight of one COMSAT (assuming two are carried) is 850 lb. For a specific impulse of 260 seconds for the satellite propulsion system and a stage weight to fuel weight ratio of 0.25, the COMSAT payload is 496 lb. For mode 2, with all tanks full, the stack weight is 100 674 lb. Approximately 1700 lb of fuel can be off-loaded and replaced with two COMSAT's weighing 850 lb each. The payload weight of each COMSAT is the same as in mode 1, i.e., 496 lb. In this case, if the COMSAT's do not deploy, the nominal landing mission must be aborted because there would be no fuel for LM pickup should that become necessary. Mode 3 is a combination of modes 1 and 2. The stack weight is increased to 102 374 lb and SPS propellant is off-loaded. Again, the nominal landing mission must be aborted if the COMSAT's cannot be deployed. The two COMSAT's could weigh as much as 1600 lb each with a useful payload of 934 lb each assuming a specific impulse of 260 seconds and a stage weight to fuel weight ratio of 0.25.

CONCLUSIONS

The CSM, on an Apollo free-return lunar landing mission, can be used to transport two COMSAT's weighing as much as 850 lb each to lunar proximity. This CSM excess payload is obtained by off-loading SPS propellant or by increasing the Saturn V launch vehicle stack weight. If both payload increasing methods are utilized, the two COMSAT's could weigh as much as 1600 lb each. The following conclusions can be made about the COMSAT's and their orbit:

1. The relative phase angle between the COMSAT's must be held close to 180° for maximum communications coverage. The drift rate between the COMSAT's can be cut to 0.7 deg/day if earth-based tracking is sufficient to determine the semimajor axis of the COMSAT's orbits to within 3 n. mi. or the period of the orbits to within 1.2 minutes. This correction maneuver cost approximately 6 fps each month and need only be made by one COMSAT.

2. The earth and the sun perturb the COMSAT's orbits so greatly that they will impact the lunar surface within 14 months if correction maneuvers are not made. Correction maneuvers made by both COMSAT's every 3 months for 10 fps each will compensate for third-body effects.

3. Earth-based tracking is essential since the maneuvers discussed above cannot be made without accurate information concerning COMSAT posi-

tion and velocity. It would be undesirable to have the CSM do this tracking.

4. If approximately 42 percent of the total COMSAT weight is allocated to the propulsion and guidance systems, sufficient ΔV is available for an expected lifetime in lunar orbit of 2.5 to 3 years.

TABLE I.- COMMUNICATION COVERAGE ACHIEVABLE

WITH TWO POLAR ORBIT SATELLITES[a]

(a) Tracking intervals

Length of, blackout, min	Time to, next blackout	CSM revolutions, to next blackout
21	596 min	5
46	693 min	6
8	609 min	5
38	12 days	--
38	608 min	5
16	693 min	6
43	589 min	5
31	12 days	--

[a]Taken from reference 1. The CSM is in an 80-n. mi. altitude circular equatorial orbit. The two satellites are in 5000-n. mi. radius circular polar orbits, in the same plane, and phased 180° apart.

(b) Percentages of coverage

Communication	Coverage, percent/sidereal month
CSM line of sight	62.04
Relay through first satellite . . .	24.97
Relay through second satellite	12.36
Total communication	99.37
No communication	0.63

TABLE II.- SATELLITE ΔV REQUIREMENTS

Event	ΔV Allowance, fps	
	Satellite in, low orbit	Satellite in, high orbit
Azimuth and flight-path angle changes at SOI	560	800
Lunar orbit insertion	2044	2415
Circularize lunar orbit	367	0
Correction maneuvers in lunar orbit for initial phasing	100	100
Correction maneuvers in lunar orbit for future phasing	329	85
Total	3400	3400

TABLE III.- SPS ΔV REQUIREMENTS DEFINING THE NOMINAL APOLLO TRAJECTORY FOR THIS STUDY

Event		ΔV Allowance, fps
Mission Dependent	Lunar orbit insertion	3100
	Lunar orbit plane change, variations in lunar stay time, and transearth injection	2800
Mission Independent	Outbound midcourse	130
	Contingencies in lunar orbit	178
	LM rescue	790
	Inbound midcourse	62

TABLE IV.- STAGE WEIGHT SUMMARY[a]

(a) CSM - LM	
CM dry (including crew), lb	13 000
SM (including 202 lb residuals), lb	10 900
SPS (unusable), lb	917
Total CSM dry	24 817
Total usable SPS, lb (I_{SP} = 310 sec)	39 741
Total LM (full tanks, less crew), lb	32 216
SLA, lb	3 900
Total stack weight (full tanks), lb	100 674
(b) S-IVB	
S-IVB stage (dry), lb	26 400
Instrument unit, lb	4 100
Flight performance reserve, lb	2 800
Flight geometry reserve, lb	2 500
SLA, lb	3 900
Total weight staged after TLI, lb	39 700

[a]Maximum total weight of the standard Saturn V in an earth orbit is 297 534 lb.

TABLE V.- METHODS OF INCREASING CSM PAYLOAD

Mode	Method
Mode 1	Stack weight is increased SPS is not off-loaded
Mode 2	Stack weight is not increased SPS is off-loaded
Mode 3	Stack weight is increased SPS is off-loaded

TABLE VI.- EXCESS PAYLOAD CAPABILITY OF THE CSM ON AN APOLLO FREE RETURN LUNAR LANDING MISSION[a]

Mode	Stack weight, lb	Satellite weight, lb[b]	Satellite payload weight, lb[c]	Effect of failure to deploy satellites
1	102 374	850	496	Complete nominal landing mission
2	100 674	850	496	Abort nominal landing mission
3	102 374	1600	934	Abort nominal landing mission

[a]Satellites are deployed prior to LOI

[b]Weight of one satellite assuming two are carried

[c]$I_{SP} = 260; \frac{W_{stage}}{W_{fuel}} = 0.25$

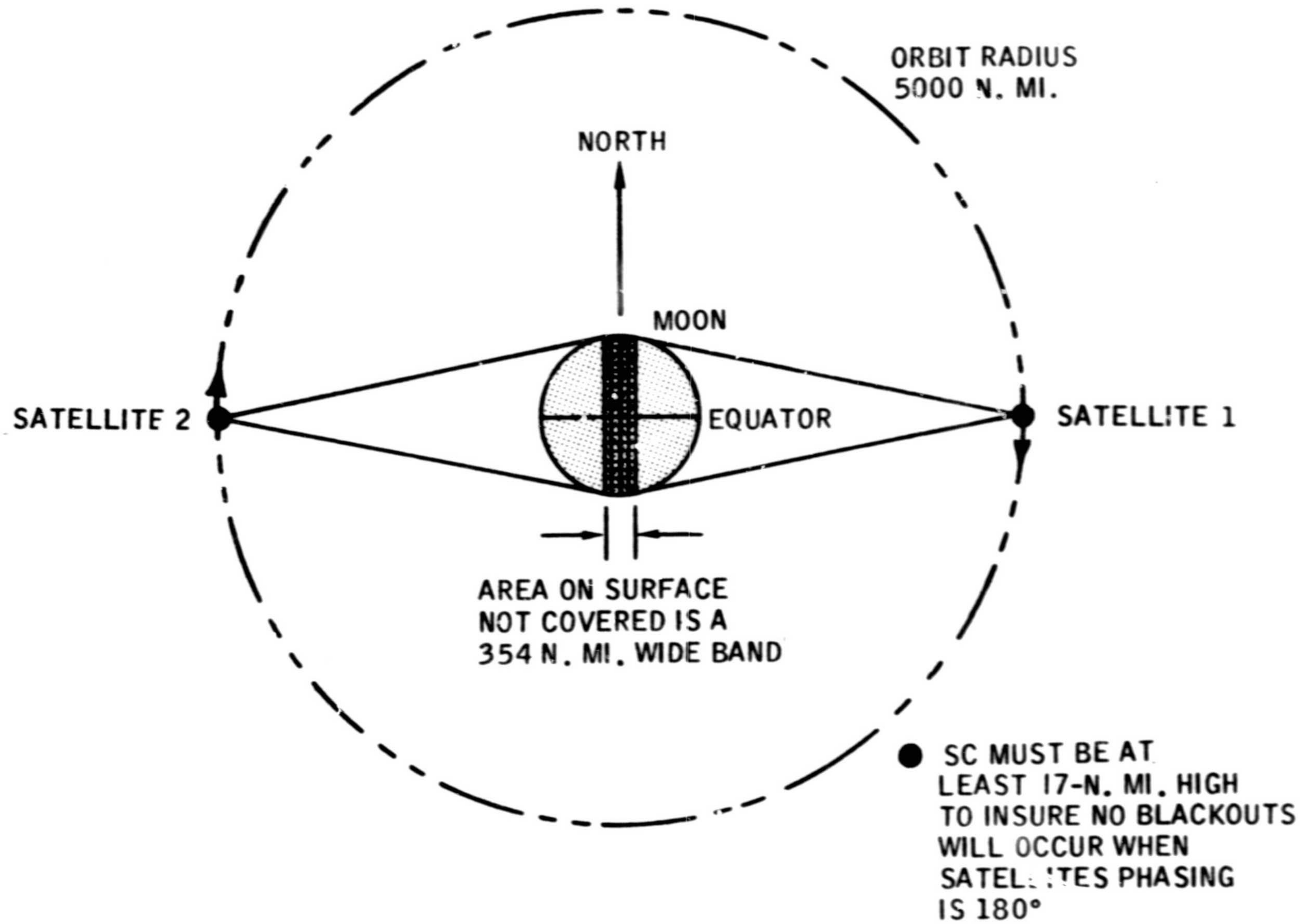


Figure 1.- Orbit configuration for two communications relay satellites.

Reference : MPAD 3293

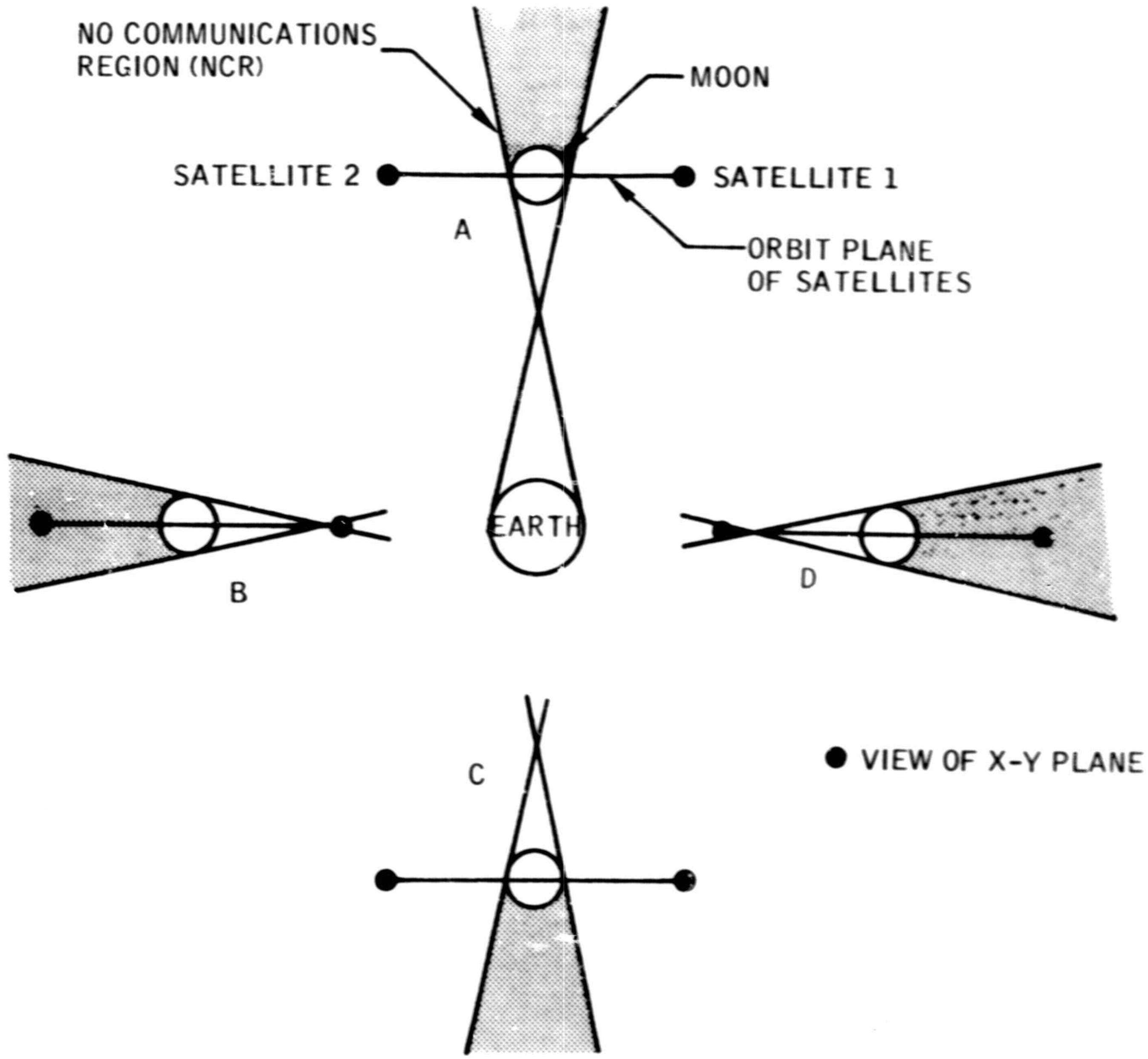


Figure 2.- Orientations of the earth, moon, and satellite for one synodic month.

Reference : MPAD 3284

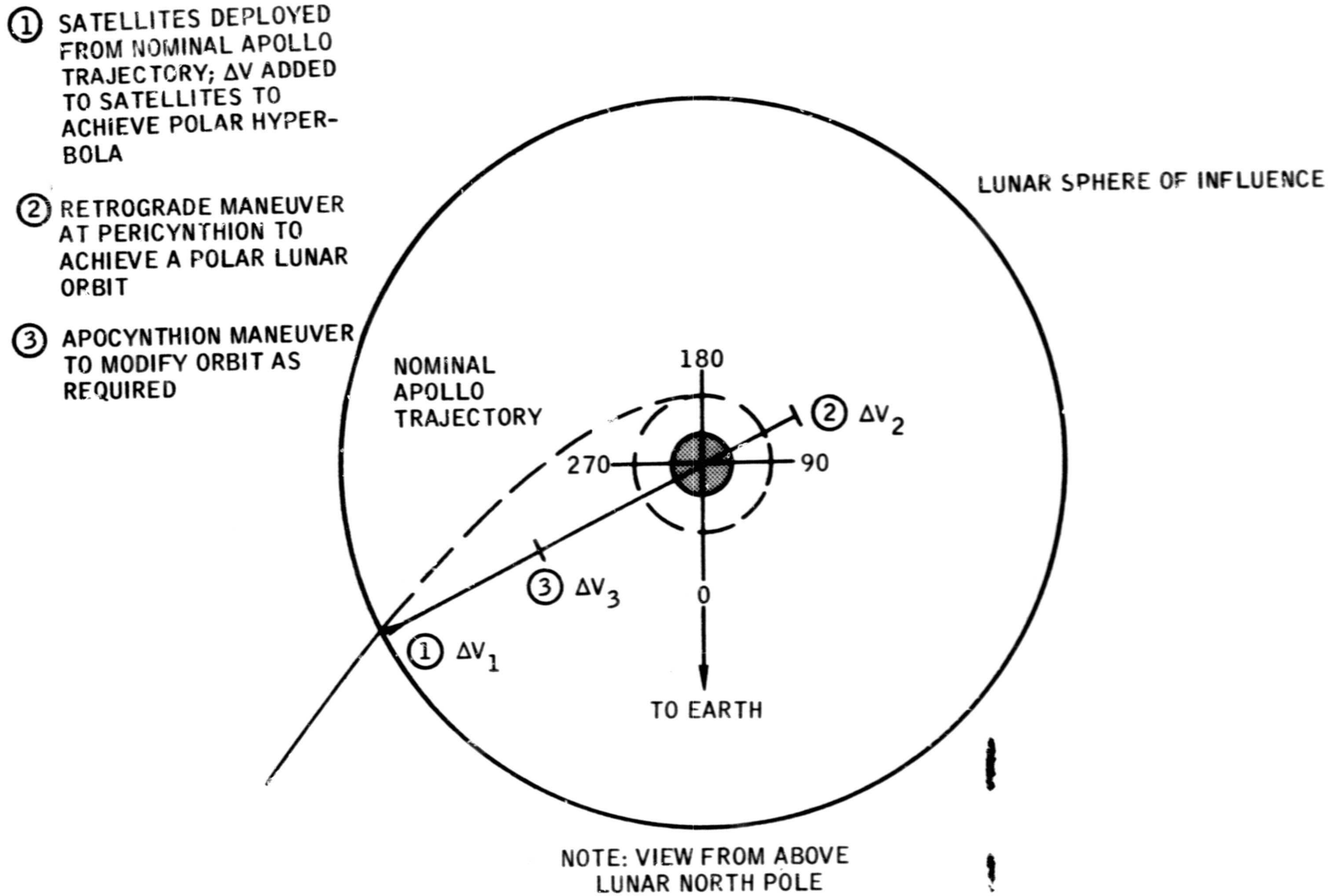
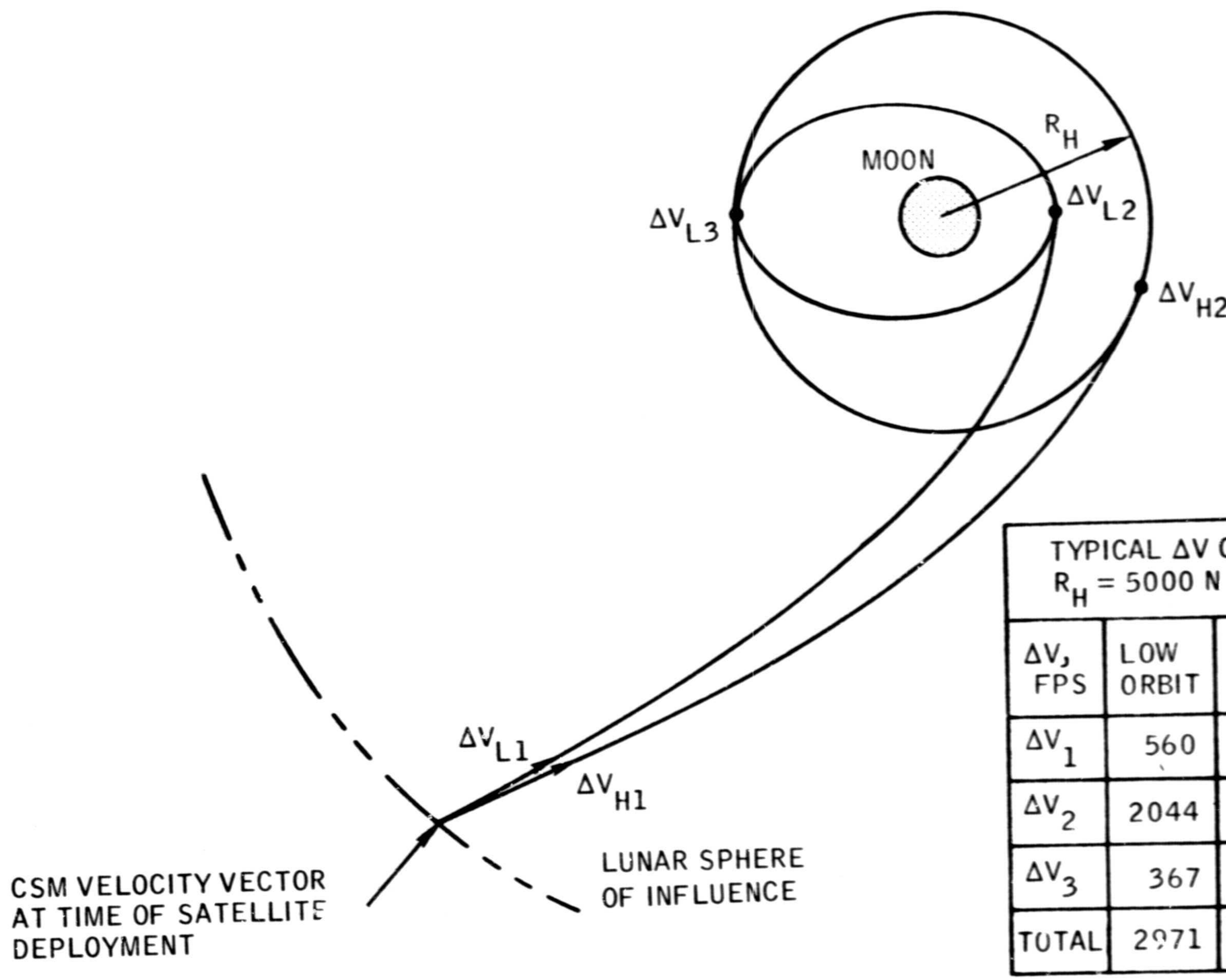


Figure 3.- Three-impulse technique for polar lunar orbit insertion.

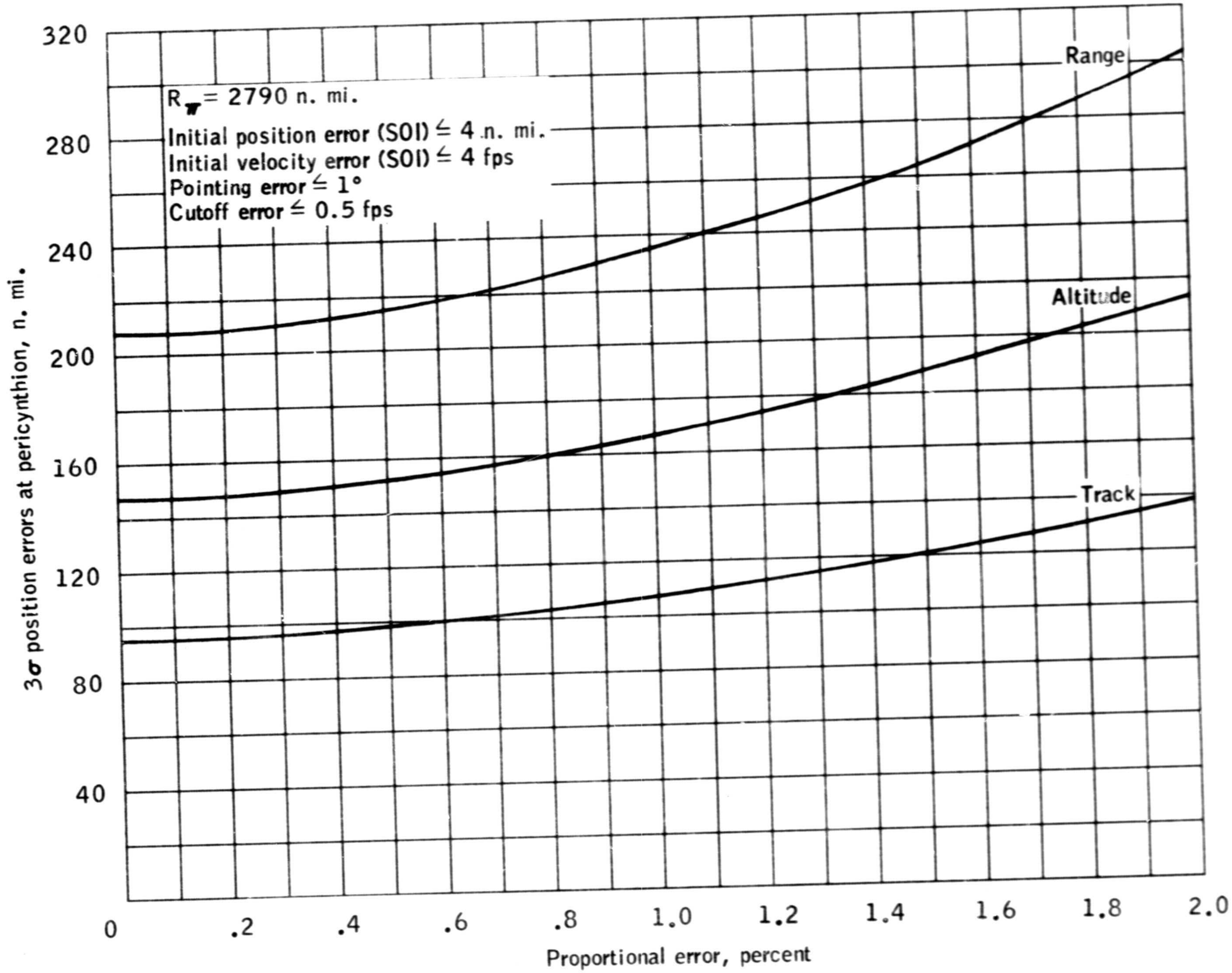
Reference : MPAD 3295



TYPICAL ΔV COST R_H = 5000 N. MI.		
ΔV, FPS	LOW ORBIT	HIGH ORBIT
ΔV_1	560	800
ΔV_2	2044	2415
ΔV_3	367	0
TOTAL	2971	3215

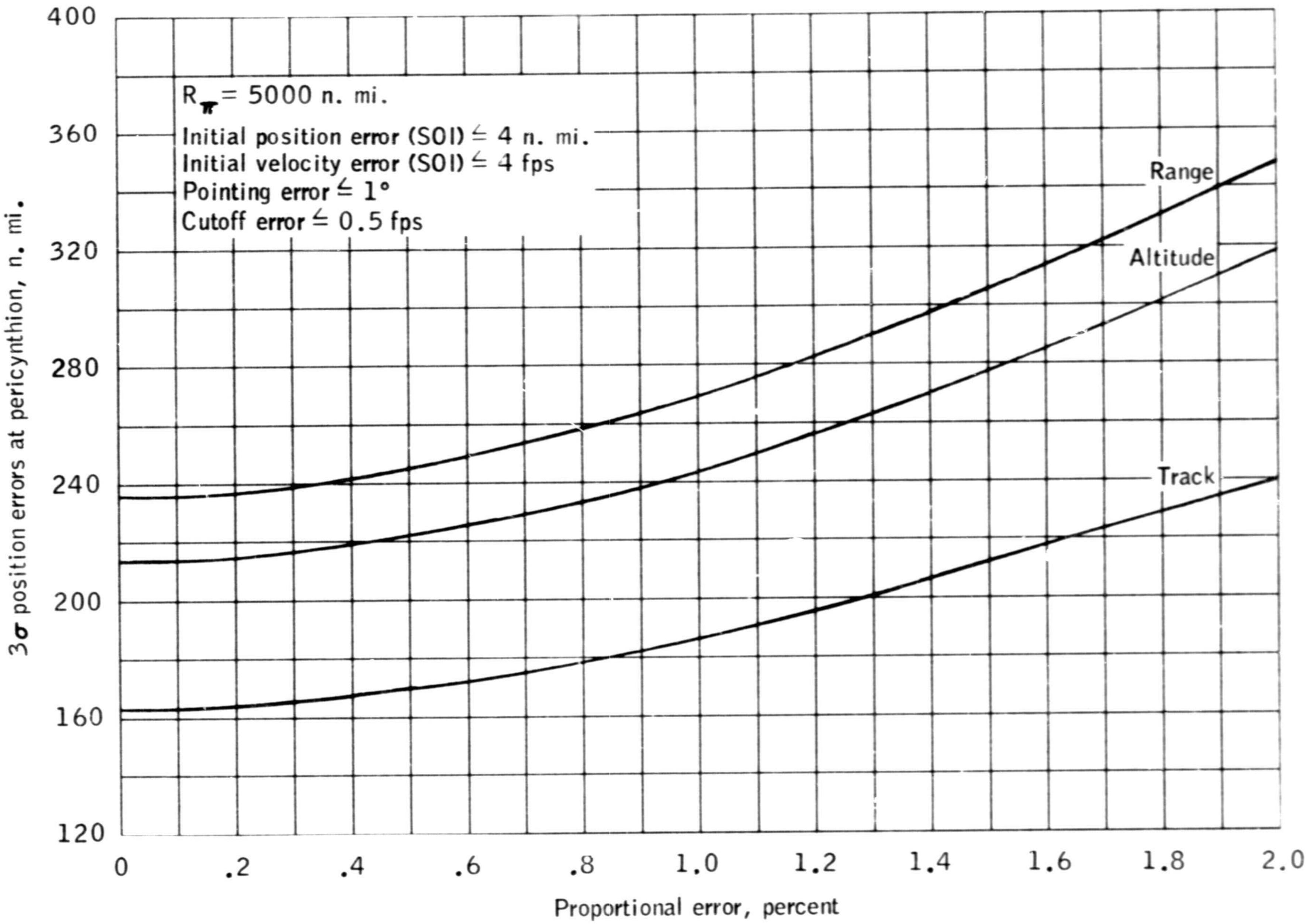
Figure 4.- Satellite phasing technique.

Reference : MPAD 3294



(a) Radius of pericynthion = 2790 nautical miles.

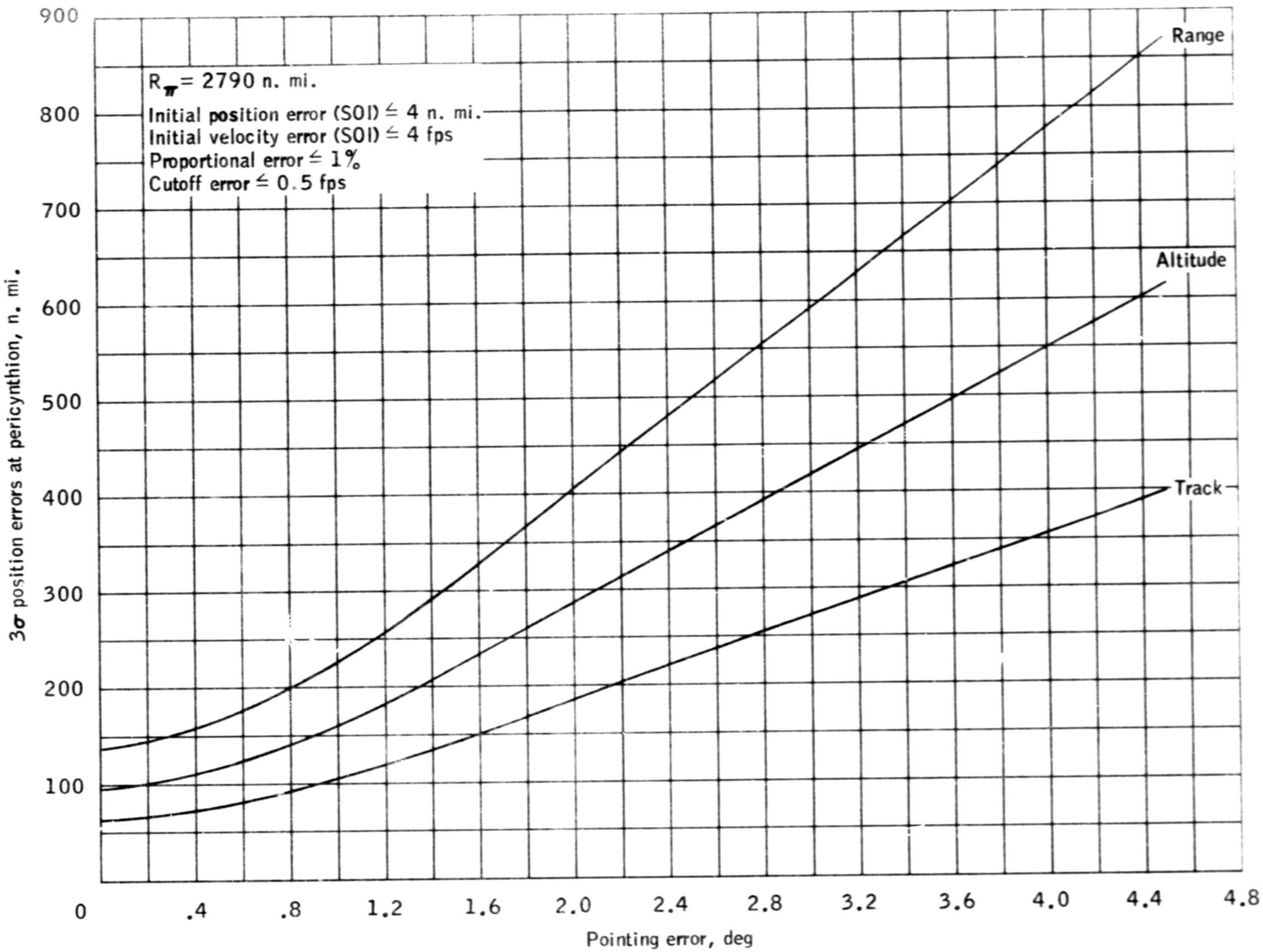
Figure 5.- 3σ satellite position error at pericynthion versus proportional error associated with the maneuver at SOI.



(b) Radius of pericynthion = 5000 nautical miles.

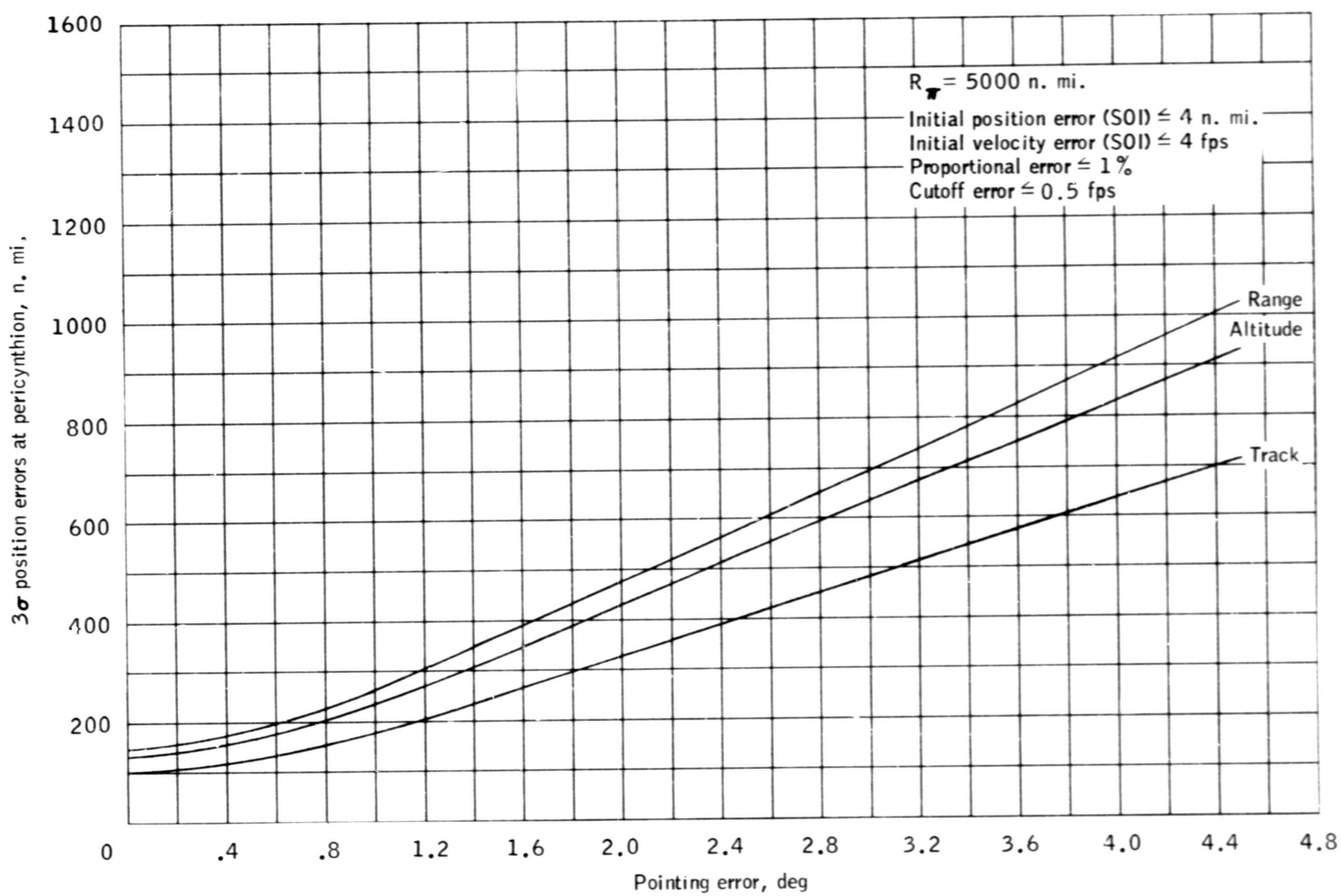
Reference: MPAD 3289

Figure 5.- Concluded.



(a) Radius of pericynthion = 2790 nautical miles.

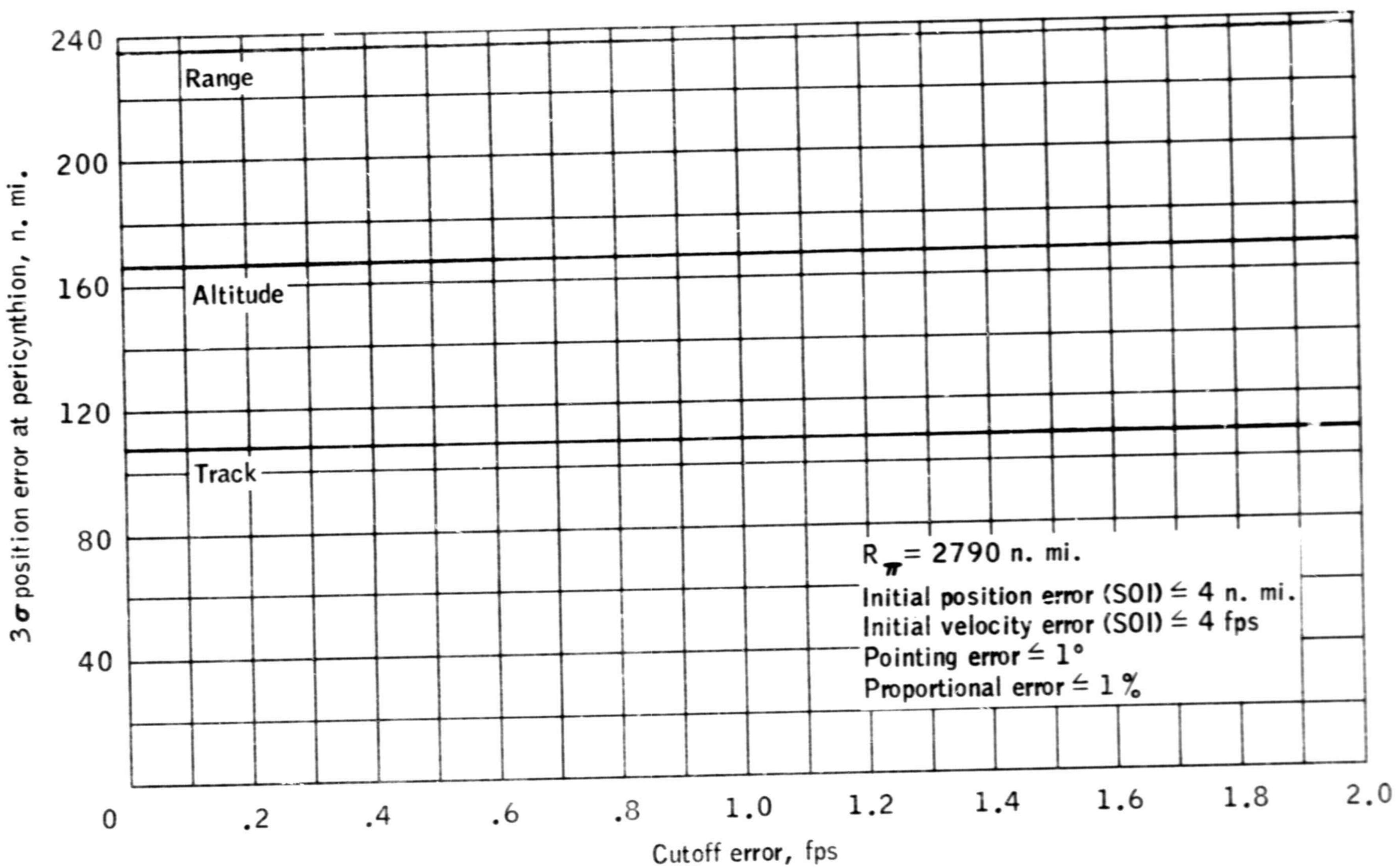
Figure 6.- 3σ satellite position error at pericynthion versus pointing error associated with the maneuver at SOI.



(b) Radius of pericynthion = 5000 nautical miles.

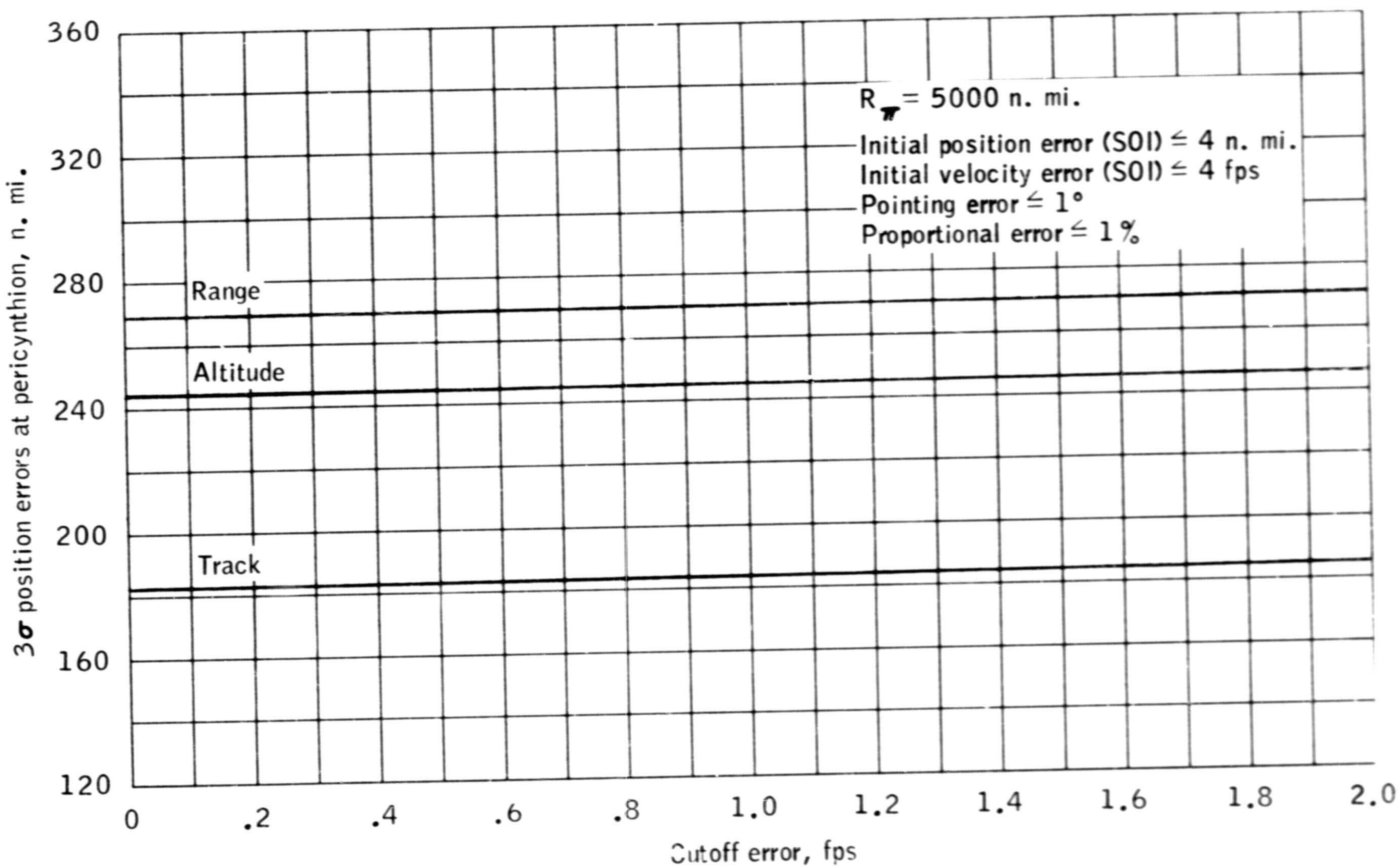
Reference : MPAD 3287

Figure 6.- Concluded.



(a) Radius of pericynthion = 2790 nautical miles.

Figure 7.- 3σ satellite position error at pericynthion versus cutoff error associated with the manuever at SOI.



(b) Radius of pericynthion = 5000 nautical miles.

Reference: MPAD 3288

Figure 7.- Concluded.

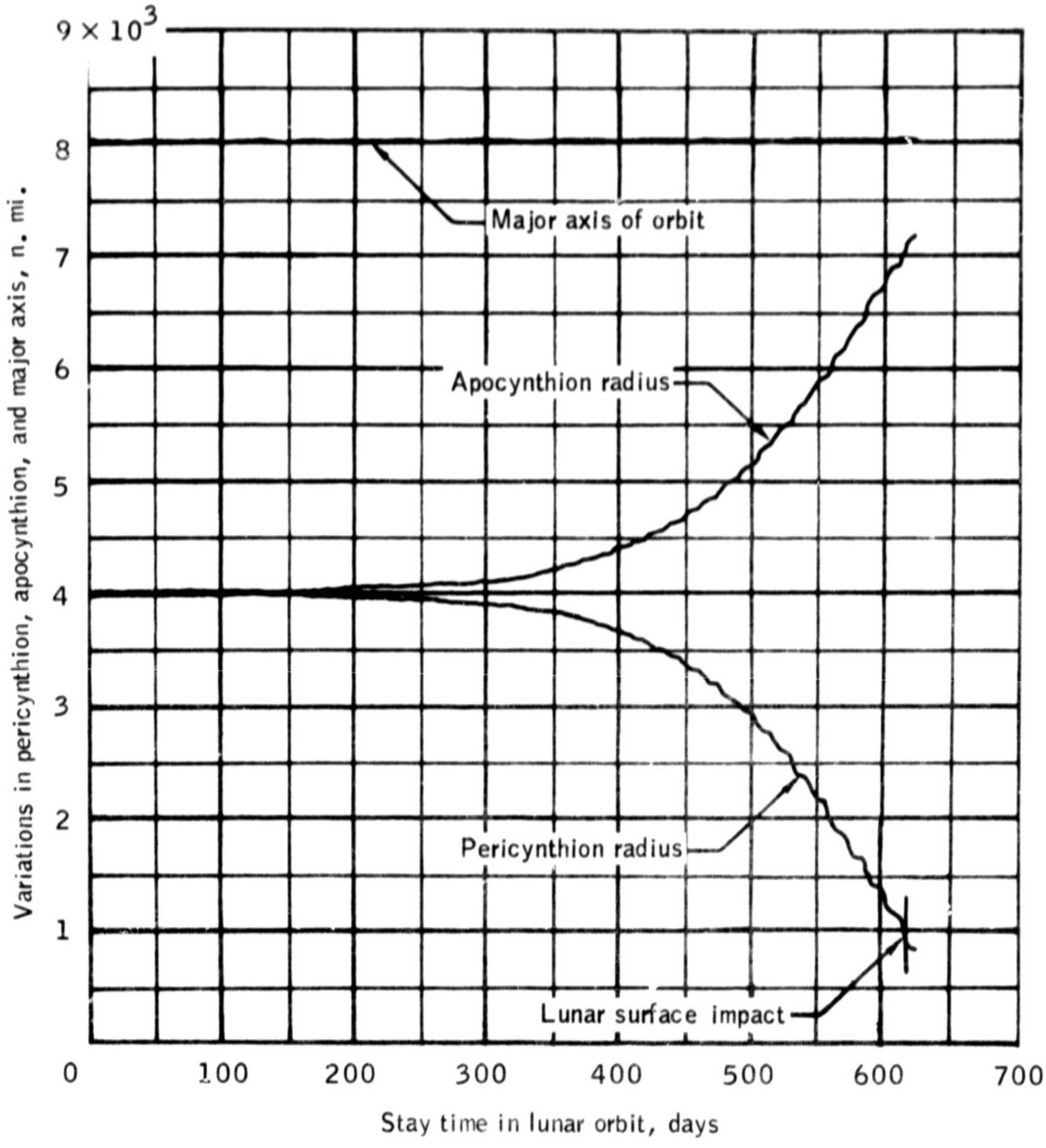


Figure 8.- Variations in pericynthion, apocynthion, and major axis due to third-body perturbations on an initially circular polar lunar orbit of radius 4000 nautical miles.

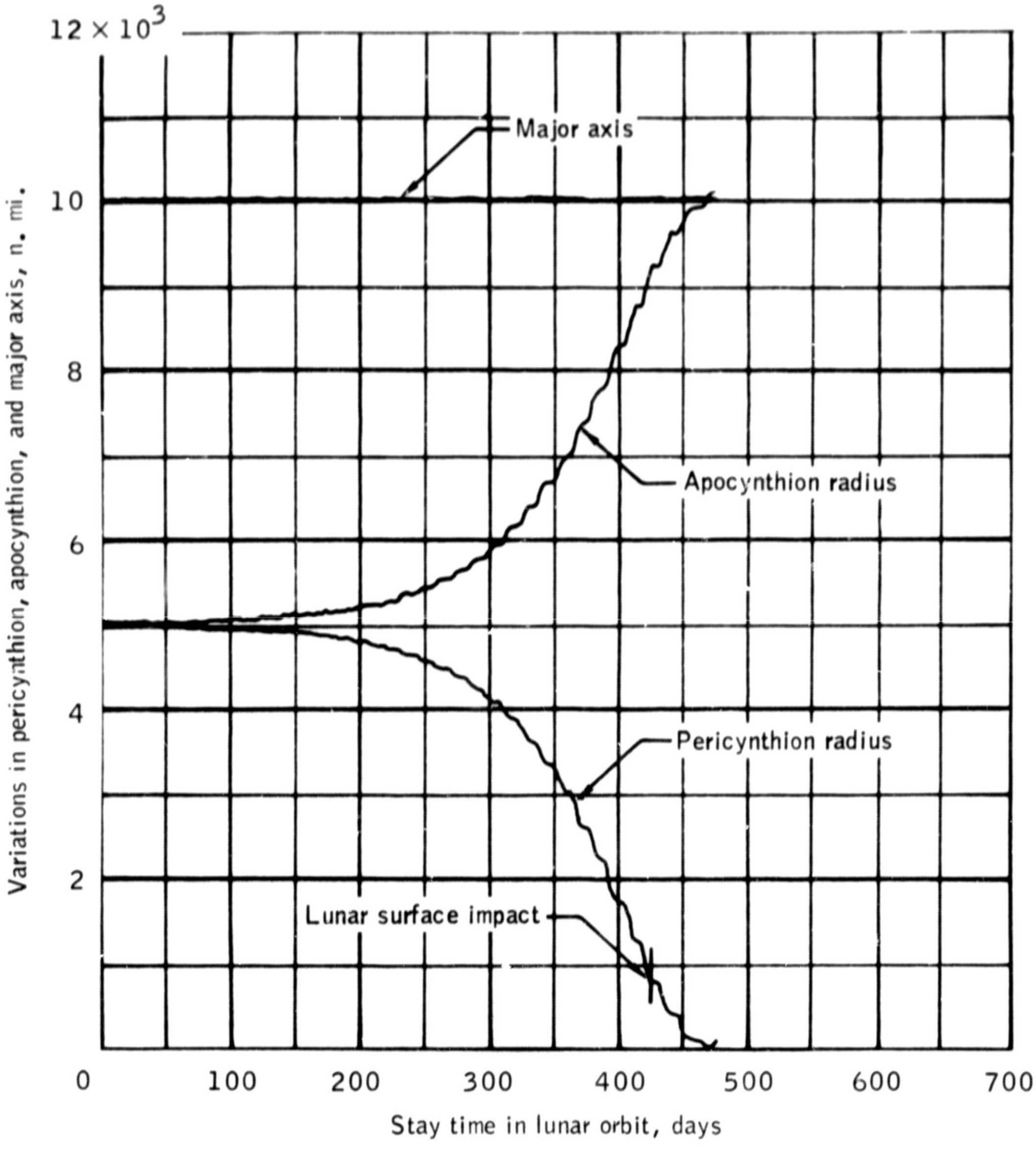


Figure 9.- Variations in pericynthion, apocynthion, and major axis due to third body perturbations on an initially circular polar lunar orbit of radius 5000 nautical miles.

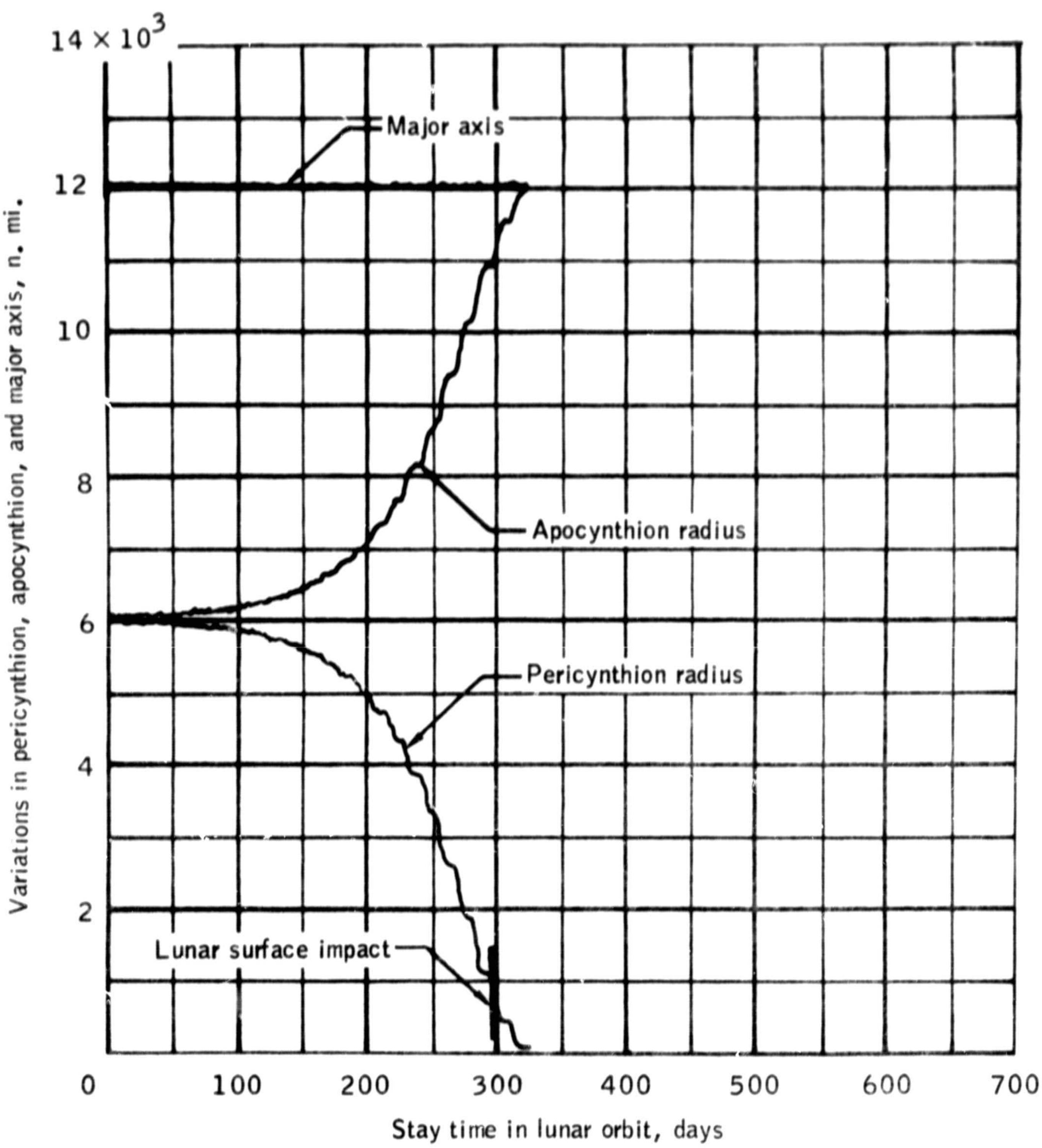


Figure 10.- Variations in pericynthion, apocynthion, and major axis due to third body perturbations on an initially circular polar lunar orbit of radius 6000 nautical miles.

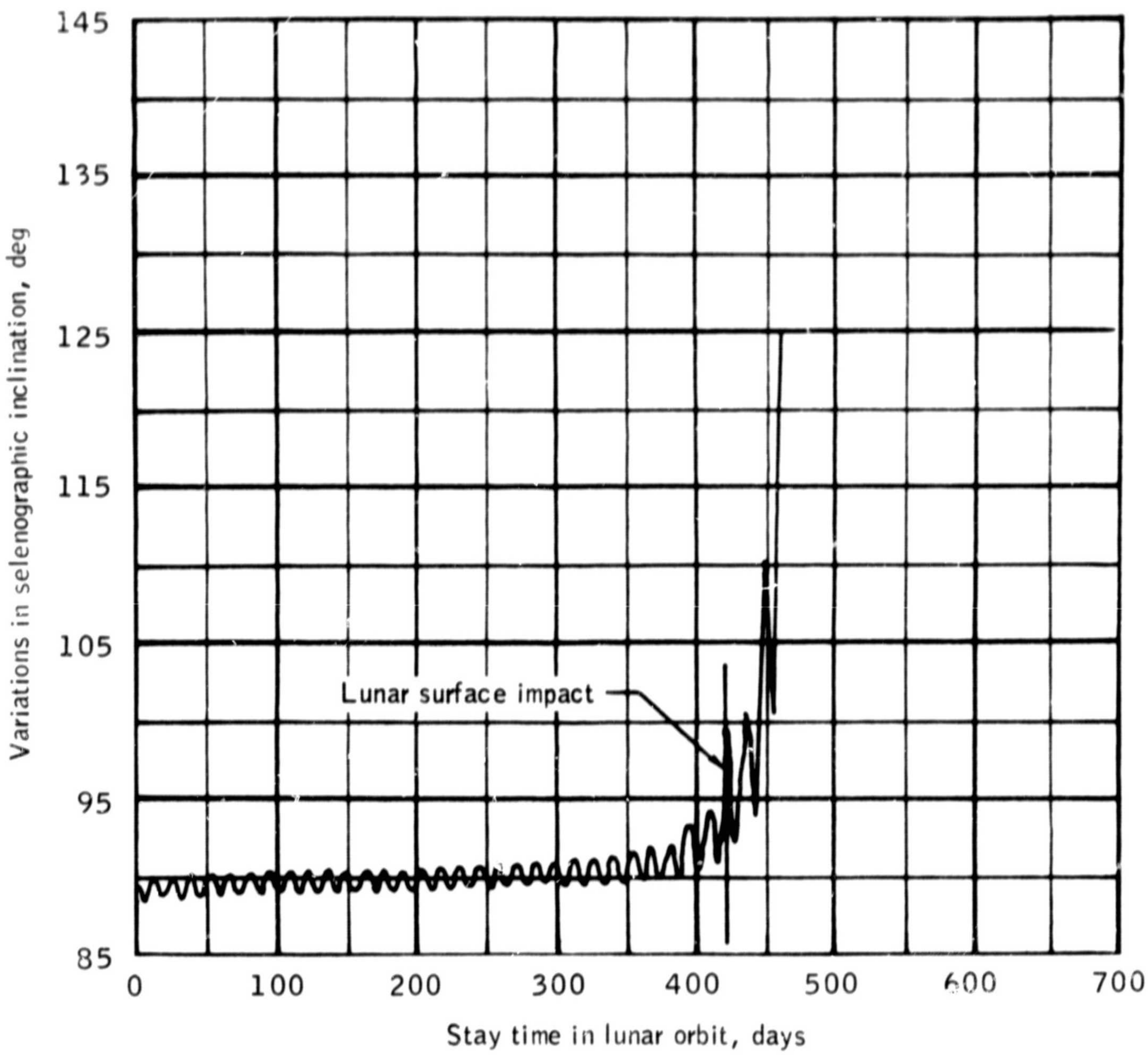


Figure 11.- Variations in selenographic inclination due to third-body perturbations on an initially circular polar lunar orbit of radius 5000 nautical miles.

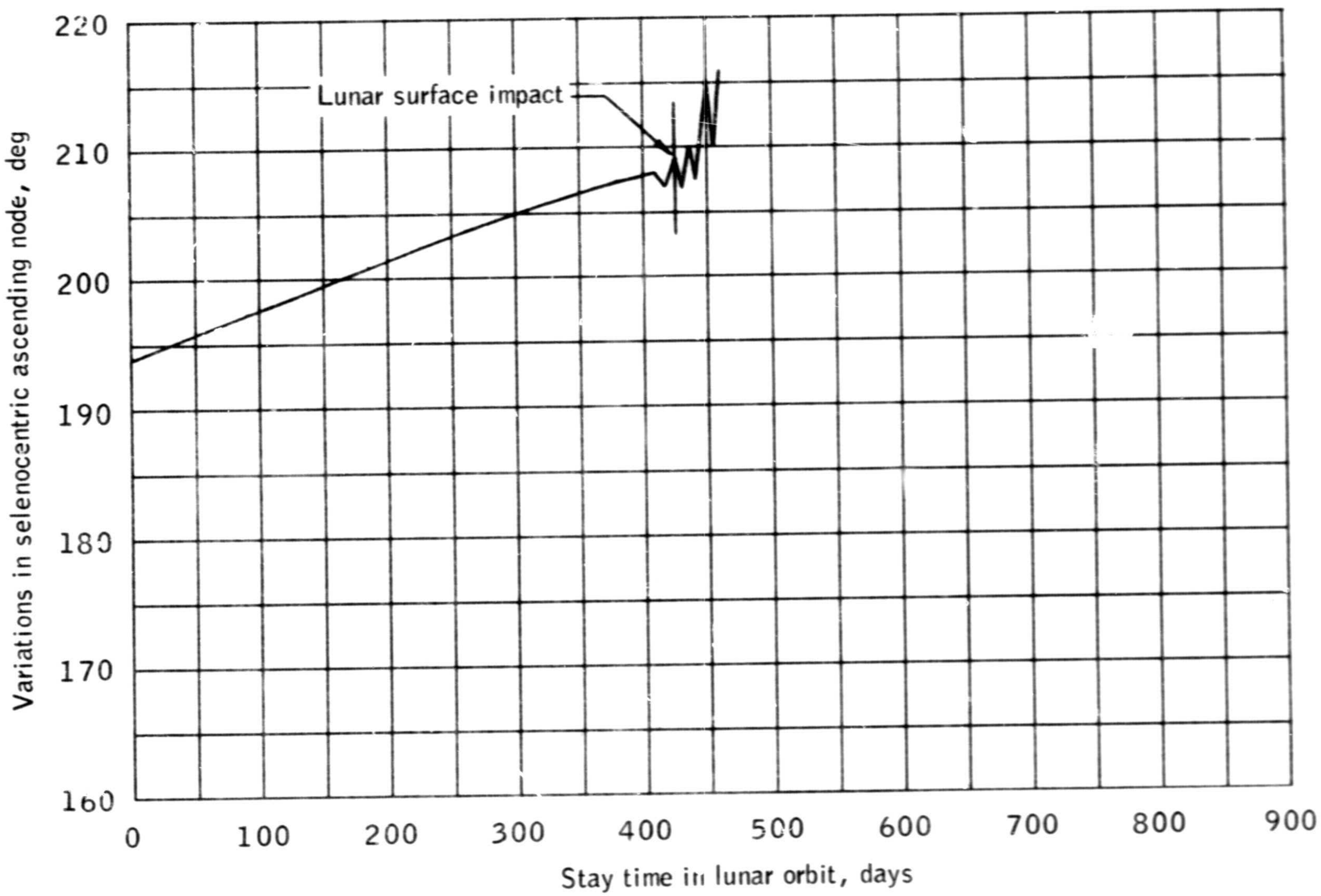


Figure 12.- Variations in selenocentric ascending node due to third-body perturbations on an initially circular polar lunar orbit of radius 5000 nautical miles.

REFERENCES

1. Wolfer, Barry M.; and MacLeod, John B.: Preliminary Investigation of Lunar Backside Communications Using Modified Lunar Orbiter Spacecraft. MSC publication, June 17, 1966.

2. Ehricke, Krafft A.: Principles of Guided Missile Design. D. Van Nostrand Company, Inc., 1960.

3. Bond, Victor R.; and Henry, Ellis W.: One-Way and Flyby Interplanetary Trajectory Approximations Using Matched Conic Techniques. MSC IN 67-FM-16, February 3, 1967.

4. Low, George M.: Apollo Spacecraft Weight and Mission Performance Definition. MSC publication, December 12, 1967.