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APOLLO 8 SPACECRAFT OPERATIONAL ALTERNATE MISSION PLANS

VOLUME I

EARTH ORBIT ALTERNATES

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Orbital Mission Analysis Branch

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PROJECT APOLLO

APOLLO 8 SPACECRAFT OPERATIONAL ALTERNATE MISSION PLANS VOLUME I - EARTH ORBIT ALTERNATES

By David D. DeAtkine, Alexander Woronow, and Richard J. Carr
Orbital Mission Analysis Branch

November 21, 1968

MISSION PLANNING AND ANALYSIS DIVISION
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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APOLLO 8 SPACECRAFT OPERATIONAL ALTERNATE MISSION PLANS

VOLUME I - EARTH ORBIT ALTERNATES

By David D. DeAtkine, Alexander Woronow, and Richard J. Carr

SUMMARY

The plans proposed in this report deal with alternate mission sequences resulting from the following contingencies occurring during the Apollo 8 mission:

1. COI, where the S-IVB fails late in its first burn, and is followed by CSM separation and an SPS burn to orbit.

2. S-IVB failure prior to TLI or insufficient propellant for restart.

3. Premature or non-nominal TLI termination resulting in an ellipse whose energy is such that a ΔV of greater than approximately 3000 fps is required at TLI-plus-3-hours for an SPS midcourse maneuver (this corresponds to an apogee of approximately 60 000 n. mi.).

The proposed earth orbit alternate missions involve several different procedures. The sequence used is determined by the contingency situation itself and such constraints as SPS ΔV capability, recovery requirements, landmark lighting conditions, and RCS deorbit capability. A summary of the alternate missions as a function of the type of failure is shown in table I, and a function of time of failure in table II.

INTRODUCTION

This document presents the operational earth orbit alternate mission plans for the Apollo 8 mission (i.e., C', Alternate 1), and provides Flight Control with a comprehensive set of alternate missions to integrate into mission rules (earth orbit alternate missions are provided for in the current version of the C' mission rules, reference 1, but details are lacking).

This document is restricted to earth orbit alternate missions which stem from either an off-nominal first or second S-IVB burn and assumes a perfectly operational CSM. The types of earth orbital alternates that have been identified include missions that are a C-type, an E-type, or a semisynchronous (12-hour period)-type; circumlunar and lunar orbit alternate mission plans are to be published separately. This study is restricted to the December launch window; however, the same techniques described are applicable to the January window as well.

ABBREVIATIONS

CM	command module
COI	contingency orbit insertion
CSM	command and service modules
EPO	earth parking orbit
ETR	Eastern Test Range
g.e.t.	ground elapsed time
G.m.t.	Greenwich mean time
h_a	apogee altitude above earth
h_p	perigee altitude above earth
LOI	lunar orbit insertion
LTA	LM test article
MCC	midcourse correction
MSFN	Manned Space Flight Network
RCS	reaction control system
RTACF	Real-Time Auxiliary Computing Facility
RTCC	Real-Time Computer Complex

SM	service module
SPS	service propulsion system
t_b	burn time
TEI	transearth injection
TLI	translunar injection

NOMINAL MISSION DESCRIPTION

The C' mission is planned to be a CSM-only, lunar orbit mission (approximately 10 revolutions about the moon). The prime objectives are to verify the MSFN state vector determination capability, test cislunar passive thermal control and onboard navigation techniques, and generally verify the translunar, transearth and lunar orbit mission timelines and operational procedures. See reference 2 for a complete list of C' test objectives.

The mission is planned to be launched within a window which opens on December 20, 1968, and closes December 27, 1968. A January launch window (January 18 through January 24) has been determined in the event of launch delays beyond the December window.

The C' nominal mission description given in table III relates to a particular launch and injection opportunity; namely, December 21, 72° launch azimuth; 12^h51^m G.m.t. lift-off, and injection on the first opportunity. For a complete description of the nominal mission, see reference 3.

SUMMARY OF INPUT DATA

Input data used in the preparation of this document were obtained from the following sources:

Mission C' detailed test objectives . . . . . ref. 2

Mission and spacecraft constraints. . . . . refs. 4 and 5

C' mission rules. . . . . ref. 1

Crew timeline . . . . . ref. 6

Spacecraft parameters . . . . . ref. 4

ΔV allotment. . . . . table IV

DEFINITIONS, GROUND RULES, AND ASSUMPTIONS

Since the mission and spacecraft constraints documents (refs. 4 and 5) do not contain all necessary ground rules and assumptions for earth orbit alternate mission planning, it is necessary to document here the basic guidelines used in planning the alternate missions described.

Definitions

Several terms which are used in this report are defined as follows:

1. Alternate mission - Any deviation from the nominal timeline where further mission objectives are considered before the end of the mission.

2. Abort - Any situation where crew safety requires immediate mission termination; no further mission objectives are considered.

3. Semisynchronous orbit - An elliptic orbit with a 12-hour period, which therefore has two perigee passes per day; the perigee positions are fixed relative to the earth, 180° apart in longitude.

Ground Rules

1. Alternate mission planning will be consistent with current spacecraft, crew, and operational constraints.

2. No additional RTCC processors will be necessary. Additional real-time requirements will be incorporated into the RTACF.

3. Entry velocities between 26 000 and 36 000 fps are assumed to be acceptable.

4. All large deboost SPS maneuvers are positioned such that MSFN coverage is available.

5. Deorbit from the alternate mission is planned such that recovery lighting constraints are met whenever possible.

6. Only water landings are planned (assumed recovery areas are as specified in the C' nominal lunar mission).

7. High apogee passes are planned with consideration for radiation dosage.

8. In general, all maneuvers in the high ellipse are retrograde; that is, abort return time is always lessened by each maneuver.

9. It is assumed to be desirable to stay in orbit following a non-nominal TLI or no TLI for a full-duration (approximately 10 days) mission rather than to abort the mission.

10. In all alternate missions, the lunar mission timeline is adhered to whenever possible.

11. There will be no additional crew training for alternate missions.

12. In the event of a failure of the S-IVB to reignite for TLI, it is assumed desirable to perform an SPS maneuver to achieve an E-type high ellipse for navigation and passive thermal control objectives. In addition, the SPS burns are to be timed such that the SPS injection and SPS deboost simulate the LOI and TEI maneuvers of the lunar mission.

13. SPS injection coverage, although desirable, is not assumed to be mandatory (injection position determined by lighting conditions for high apogee navigation exercises).

14. RCS deorbit capability is maintained for all alternate missions.

Assumptions

1. The high-ellipse phasing maneuver (used to position a later perigee pass) is applied at the first perigee passage, and is not necessarily covered by MSFN, although coverage is, of course, desirable.

2. No shifting of the line of apsides or the line of nodes is attempted in the phasing maneuver of the premature cutoff high-ellipse alternate; only a reduction in orbital period is made.

3. A maximum ΔV of 8000 fps is allowed in the phasing and deboost maneuvers of the premature cutoff ellipse; this leaves approximately 1700 fps of ΔV (based on usable C' propellant loading) for burn dispersion allowances, deorbit, and other maneuvers.

4. The deboost maneuver from the premature cutoff high ellipse achieves a 400-n. mi. apogee (as in the E mission deboost) except where the 8000-fps ΔV limit prevents it.

5. A maximum of 8700 fps of ΔV is allowed for orbit shaping, SPS injection, and deboost; this leaves approximately 1000 fps of ΔV for deorbit and other maneuvers.

6. The December launch window was used in this study; however, the same techniques should be applicable to any launch opportunity.

7. A maximum radiation dose of 1 rad and 2 rads depth and skin dose, respectively, per high-ellipse revolution is assumed.

8. The maximum apogee for the premature TLI cutoff earth-orbit type of alternate mission is assumed to be 60 000 n. mi.; this corresponds to an SPS midcourse at TLI-plus-3-hours of approximately 3000 fps of ΔV. Further studies may alter this change-over point (i.e., the point at which an SPS burn can complete the TLI and a circumlunar mission can be attempted).

ALTERNATE MISSION PLANS

Alternate Mission 1

The sequence of events associated with alternate mission 1 is as follows:

1. COI following S-IVB first burn malfunction.

2. SPS orbit "tuning" maneuvers, following the lunar mission MCC timeline.

3. SPS injection into a 4000-n. mi. ellipse at approximately 69 hours g.e.t., provided sufficient SPS ΔV is available.[a]

4. SPS deboost into a low earth orbit approximately 24 hours after injection.

5. Further MCC's (following the lunar mission timeline) are used to adjust the ellipse for deorbit.

6. Approximately a 10-day mission.

If the S-IVB malfunctions late in its first burn, the CSM has the capability to achieve a low orbit using the SPS. If the ΔV_{COI} is less

[a]See discussion for explanation of ΔV sufficiency criteria.

than approximately 900 fps, the CSM has the capability to later inject into an E-type ellipse, then deboost and continue the mission in a low earth orbit still following the lunar orbit timeline.

If, on the other hand, the CSM has the ΔV capability to achieve and deboost from the high ellipse (which means a $\Delta V_{COI} < 900 \pm 200$ fps, depending upon the insertion ellipse), the plan is to remain in low earth orbit for approximately 68-72 hours, making MCC's consistent with the lunar timeline. These maneuvers could be used for perigee preservation or other orbit tuning burns. In the time period of 68-72 hours g.e.t. (consistent with the nominal mission LOI), the SPS is ignited over U. S. or ETR MSFN stations near maximum northern declination for a 4200-fps burn to inject the CSM onto a 4000-n. mi. h_a.

The injection point is a function of the lighting conditions desired under the apogee for star/landmark navigation sightings. This is illustrated in figure 1, where the injection position on the arc across the maximum northern declination is chosen based upon maximizing daylight under the resulting apogee. It can be seen from this figure that, if injection is constrained to this arc (this is done to maximize injection and deboost burn MSFN coverage), in all launch opportunities (day of launch, launch azimuth, or time) at least half the apogee pass could be placed in daylight. It should be remembered that the day/night terminator remains approximately fixed relative to the orbital line of apsides.

The CSM remains in the high ellipse for approximately 24 hours (until the US-ETR rotates back under the inertial perigee). While in the high ellipse, the crew work and rest cycles would be quite similar to that of the lunar mission. The high ellipse deorbit situation is covered in reference 7, which is based upon the E mission high ellipse. The SPS deboost burn (ΔV = 3700 fps) occurs over the US-ETR; the resulting ellipse is approximately 100- by 400-n. mi. (the same as in the E-type mission). The CSM remains in the low earth orbit for the remainder of the 10-day mission. Remaining SPS propellant can be used for orbit trimming and shaping and deorbit.

In the event the ΔV_{COI} is too large, the CSM would not inject onto the high ellipse, but remain in low earth orbit, for a C-type mission. End of mission recovery area lighting conditions are a problem for earth orbit alternates from missions which lift-off early in the first two days of the December launch window. Nodal regression during the 10 days of earth orbit together with the early lift-off results in early morning (presunrise) landings. This condition can be improved in some cases by nodal plane-change maneuvers; this method, however, is somewhat restricted by ΔV capability.

A typical alternate mission 1 timeline is shown in table V(a) for the December 21, 72° launch azimuth opportunity.

Alternate Mission 2

The sequence of events associated with alternate mission 2 is the same as that shown for alternate 1, except that there is no COI because the S-IVB achieves orbit, but does not reignite with CSM attached.

If, after achieving EPO, the S-IVB for any reason cannot be reignited with the CSM attached, the spacecraft can be separated, and the same sequence outlined for the alternate 1 mission can be followed, except that there is no SPS ΔV limitation as a result of COI. Therefore, the SPS injection to a 4000-n. mi. ellipse would always be performed (assuming the CSM is in a "GO" condition). The same lighting problems for navigation and lighting exists here as in alternate 1.

Shown in table V(b) is a typical timeline for alternate mission 2.

Alternate Mission 3a

The sequence of events associated with alternate mission 3a is as follows:

1. S-IVB shuts down prematurely during TLI; resulting h_a is less than 4000 n. mi.

2. SPS injection onto a 4000-n. mi. ellipse, following the lunar mission MCC timeline.

3. SPS deboost to low earth orbit one day later.

4. Continue 10-day low earth orbit mission.

In the event the S-IVB reignites but malfunctions, necessitating engine shutdown, and the resulting apogee is less than 4000 n. mi., the SPS is used (following the MCC timeline) to boost the apogee to 4000 n. mi. The CSM remains in this orbit for approximately one day, deboosts to low earth orbit, and continues a low earth orbit mission. The maneuver to achieve the 4000-n. mi. apogee would be performed at the second perigee. The deboost is performed over the injection ships, roughly 24 hours after TLI. The CSM then spends approximately six orbits in the 4000-n. mi. ellipse.

A typical alternate mission 3a timeline is shown in table V(c) for the December 21, 72° launch azimuth opportunity.

Alternate Mission 3b

The sequence of events associated with alternate mission 3b is as follows:

1. Premature S-IVB shutdown during TLI; $4000 \leq h_a \leq 25\ 000$ n. mi.

2. SPS phasing maneuver at first perigee.

3. SPS deboost at perigee approximately 24 hours after TLI.

4. SPS MCC's to adjust orbit, following lunar timeline.

5. Mission approximately 10 days in duration.

The plan is to perform an SPS phasing maneuver at the first perigee passage to change the orbital period such that at TLI-plus-24-hours the CSM passes through perigee, and such that perigee is located over one or both injection ships, or Hawaii, Canberra, or Carnarvon MSFN stations (depending upon the day of launch, since perigee moves southerly during the launch window). This is done to cover the large SPS burn used to deboost the CSM into a 100- by 400-n. mi. low earth orbit. The change of orbital period to be performed by the phasing maneuver is shown as a function of the S-IVB premature cutoff apogee altitude in figure 2. The ΔV requirements for phasing and deboost are shown as a function of TLI premature cutoff apogee altitude in figure 3. This maneuver could take place anywhere from 6 hours to 16 hours g.e.t., depending upon apogee altitude.

The deboost perigee line is shown in figure 4. It can be seen that the desired perigee point moves southwestward through the monthly launch window; its longitude position is placed to maximize MSFN coverage. Both figures 3 and 4 show period changes and ΔV requirements for selected days and launch azimuths of the December launch window. The orbital period as a function of TLI cutoff is shown in figure 5. The ΔV and t_b requirement to lower apogee to 400 n. mi. is shown in figure 6.

This alternate mission sequence is to perform a phasing maneuver at first perigee pass (3 to 14 hours after TLI), coast in the phasing ellipse until approximately TLI-plus-24-hours, deboost into the 100- by 400-n. mi. orbit, and proceed with a low earth orbit mission (10 days). There is ample time (20 to 25 hours) spent in the high ellipse to perform navigation exercises; however, in some regions of the launch window the landmark lighting conditions underneath the apogee resulting from the TLI premature cutoff are rather poor.

A typical timeline for alternate mission 3b is shown in table V(d).

Alternate Mission 3c

The sequence of events associated with alternate mission 3c is as follows:

1. Premature S-IVB shutdown during TLI; 25 000 $\leq h_a \leq$ 60 000 n. mi.

2. SPS phasing maneuver at first perigee.

3. SPS maneuver at approximately 72 hours g.e.t. to place CSM in a semisynchronous orbit.

4. Remain in semisynchronous ellipse for approximately the remainder of 10-day mission, then initiate a direct entry from semisynchronous orbit.

The procedure is to perform a retrograde phasing maneuver at the first perigee to alter the orbit period such that a later perigee occurs over a selected Pacific recovery area. At this perigee another SPS maneuver lowers apogee to approximately 22 000 n. mi. The resulting ellipse is a semisynchronous (12-hour period) orbit whose perigees occur over the same two points in the Pacific and Atlantic, once each per day. The spacecraft would then remain in this ellipse for 10 days and deorbit from the semisynchronous orbit into the Pacific prime recovery area. Contingency deorbit can be performed from this ellipse at all true anomalies except a relatively small band about perigee (about 25°). The perigee lines, or recovery lines, are established in the two oceans as shown in figure 7. The lines move southeasterly through the launch window in order to provide water landing deorbit opportunities in the Atlantic. This procedure provides two deorbit opportunities per day to stationary locations, one each in the Atlantic and Pacific oceans. It can be seen in figure 7 that early in the launch window, the Pacific perigee line is west of the 165° west prime recovery longitude.

It may be advantageous to make the final orbit slightly less than 12 hours in period to allow the perigee to advance slowly eastward toward the prime area throughout the mission. The recovery ship could move westward, and therefore reduce the amount of required perigee progression. The disadvantage in allowing the Pacific perigee point to progress toward the prine recovery area is that the Atlantic perigee moves onto land. The remainder of the 10 days is spent in this semisynchronous orbit, allowing ample opportunity for navigation exercises. However, for some portions of the launch window, the landmark lighting conditions under the TLI-established apogee position are somewhat marginal.

A typical timeline for alternate mission 3c is shown in table V(e).

Alternate Mission 3d

If the resulting apogee altitude is greater than 60 000 n. mi., the SPS ΔV required at TLI-plus-3-hours to complete the injection is approximately 3000 fps. At this point, the decision would be to perform this maneuver and continue the lunar mission. Alternate missions from this point on will be described in volume II of this document. It should be noted that if a premature or non-nominal TLI cutoff occurs, the TLI-plus-3-hours midcourse will always be computed by the RTCC, and if ΔV_{MCC} is less than approximately 3000 fps, the midcourse would be performed; in other words, midcourse ΔV, not h_a, determines whether alternate 3c or 3d is used.

CONCLUDING REMARKS

The data given and procedures described in this document represent the operational earth orbit alternate mission plans for the C' lunar orbit mission. The purpose here has been primarily to qualitatively, rather than quantitatively, define the alternate mission procedures. Because of the large amount of data that would be necessary to cover the launch window and corresponding alternate mission situations completely, the data presented is merely meant to be representative of the range of maneuvers which could be used. RTACF procedures and processors will be used to compute these maneuvers in real time, and there will be a reliance upon real-time maneuver targeting for the alternate missions rather than preflight generated data.

TABLE I.- EARTH ORBITAL ALTERNATE MISSIONS FOR THE APOLLO C' ALTERNATE 1 MISSION

Contingency orbit insertion	Earth parking orbit (no S-IVB re-ignition)	Ellipse resulting from the S-IVB TLI burn			
		$100 \leq h_a < 4000$ n. mi.	$4000 \leq h_a < 25\ 000$ n. mi.	25 000 n. mi. $\leq h_a <$ 60 000 n. mi.	$h_a \geq 60\ 000$ n. mi.
After COI, if sufficient SPS propellant remains, (COI ΔV < 900 fps), inject into E-type ellipse ($h_a \sim 4000$ n. mi.). Follow lunar time-line. Deboost to low earth orbit; continue C-type mission. If there is insufficient SPS propellant for above sequence, remain in low earth orbit; conduct C-type mission.	SPS burn to E-type high ellipse ($h_a \sim 4000$ n. mi.). Follow lunar time-line. Deboost to low earth orbit; continue C-type mission.	If the resulting apogee can be rotated to an acceptable position for landmark sighting with < 900 fps ΔV, perform apsidal rotation and proceed with alternate 2 sequence; if not, deboost and continue low earth orbit mission.	Perform phasing maneuver at first perigee to place later perigee over MSFN site. Perform deboost at the later perigee ($h_a \sim 400$ n. mi.). Continue low earth orbit mission.	Perform phasing maneuver at first perigee to place later perigee over recovery site. Perform maneuver at the later perigee to establish semi-synchronous orbit. Remain in this ellipse for remainder of mission. Direct entry from high ellipse.	Go circumlunar or nominal lunar orbit (depending upon ΔV requirements).

TABLE II. - C' EARTH ORBIT ALTERNATE MISSION FLOW CHART

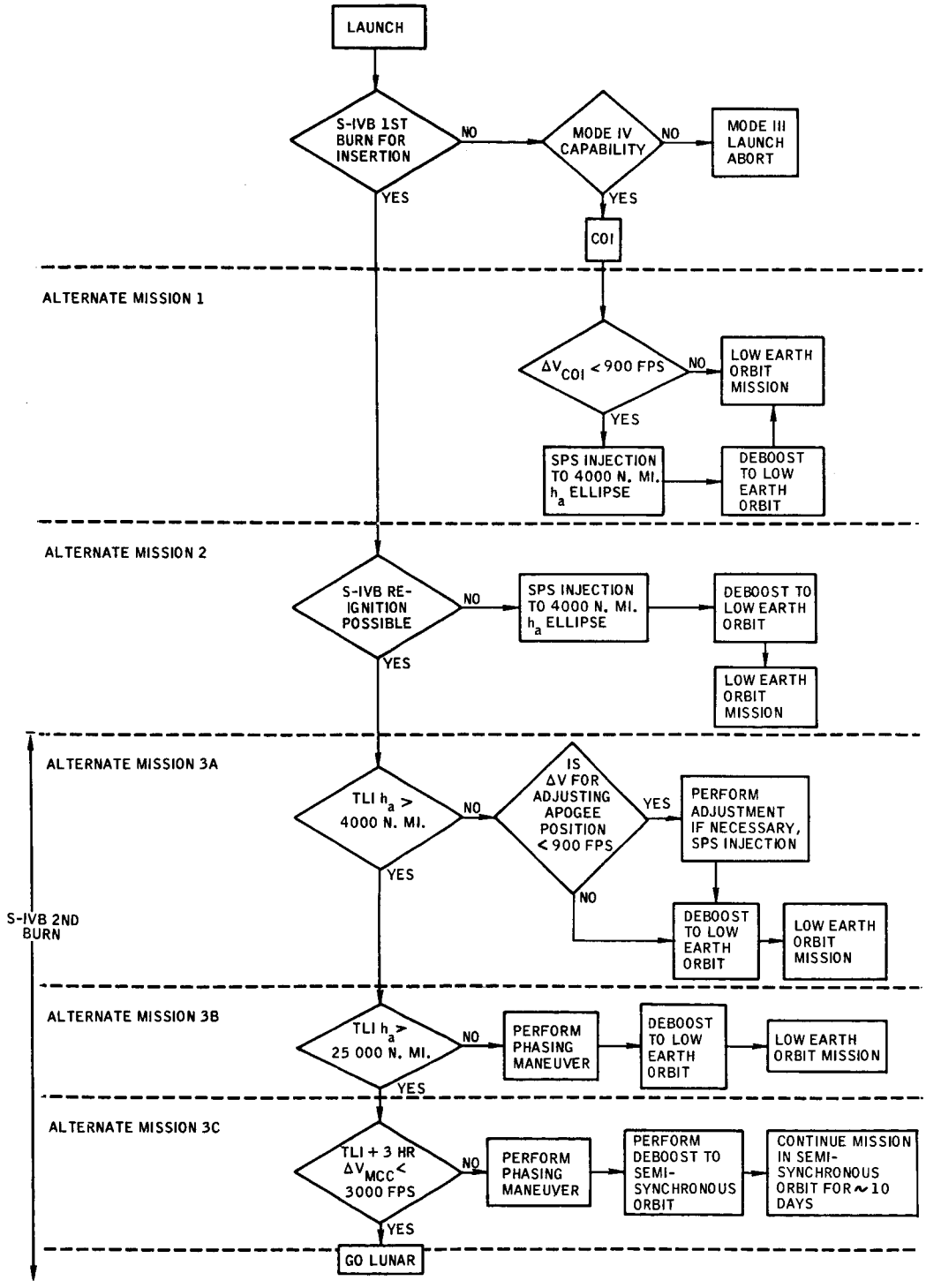


TABLE III.- NOMINAL MISSION EVENTS FOR THE APOLLO C'

ALTERNATE 1 MISSION

[December 21 launch, 72° launch azimuth, injection on first opportunity]

Event	Ground elapsed time of initiation, hr:min:sec	ΔV, fps	Burn time, sec
Lift-off	00:00:00	--	--
Insertion	00:11:20	--	--
TLI	02:50:31	10 635	311.5
MCC1	09:00:00	--[a]	--[a]
MCC2	28:00:00	--[a]	--[a]
MCC3	47:00:00	--[a]	--[a]
MCC4	61:00:00	--[a]	--[a]
LOI(1)	69:07:29	2 991	245.8
LOI(2)	73:30:53	138	9.7
Pass over target	82:08:26	--	--
TEI	89:04:02	2 837	171.3
MCC5	99:00:00	--[a]	--[a]
MCC6	118:00:00	--[a]	--[a]
MCC7	142:00:00	--[a]	--[a]
MCC8	169:00:00	--[a]	--[a]
Entry	171:05:32	--	--
Splashdown	171:19:18	--	--

[a]Nominally zero.

TABLE IV.- ASSUMED SPS ΔV ALLOTMENT FOR THE APOLLO C'

ALTERNATE 1 MISSION

(a) TLI resulting in an apogee > 4000 n. mi.

Maximum ΔV allowed for deboost from TLI apogee, fps	8000
ΔV reserved for deorbit, fps	500
ΔV reserved for low earth-orbit maneuvers (excluding deorbit), fps	1200
Total ΔV available, fps	9700

(b) No TLI or TLI resulting in an apogee < 4000 n. mi.

Maximum ΔV allowed for SPS injection and deboost, fps	8700
ΔV reserved for deorbit, fps	500
ΔV reserved for low earth-orbit maneuvers (excluding deorbit), fps	500
Total ΔV available, fps	9700

TABLE V. - TYPICAL SPS MANEUVER TIMELINES FOR THE EARTH ORBITAL ALTERNATE MISSIONS OF THE APOLLO C' ALTERNATE 1 MISSION[a]

Mission time, hr:min	Event	Duration, min:sec	ΔV, fps	Resulting h_a/h_p, n. mi.	MSFN coverage
(a) Alternate 1					
0:11	COI	00:56	600	182/100	BDA, insertion ship
9:07	MCC[b]	00:10	110	110/110	HAW
61:00	MCC2[b]	--	--	107/107	--
70:10	SPS injection	05:07	4180	4000/105	TEX, MLA, GBI
93:51	Deboost	03:02	3668	400/105	MLA, GBI
100:03	MCC3	00:15	25	400/90	HAW
165:01	MCC4	00:02	363	200/90	CRO
236:51	Deorbit	00:11	282	190/-10	HAW
(b) Alternate 2					
09:11	MCC[b]	00:04	45	125/102	HAW
61:00	MCC2[b]	--	--	115/102	--
70:08	SPS injection	05:27	4160	4000/103	TEX, MLA, GBI
93:48	Deboost	03:15	3680	400/103	MLA, GBI
100:01	MCC3	00:15	27	400/90	HAW
165:40	MCC4	00:02	359	200/90	CRO
236:46	Deorbit	00:12	286	190/-10	HAW

[a]All timelines assume December 21 launch, 72° launch azimuth, injection on first opportunity.

[b]Used, if necessary, to "tune" orbit prior to injection; is nominally zero.

TABLE V.- TYPICAL SPS MANEUVER TIMELINES FOR THE EARTH ORBITAL ALTERNATE MISSIONS OF THE APOLLO C' ALTERNATE 1 MISSION[a] - Continued

Mission time, hr:min	Event	Duration min:sec	ΔV, fps	Resulting h_a/h_p, n. mi.	MSFN coverage
(c) Alternate 3a					
02:51	TLI c/o	--	--	2000/104	Injection ship, HAW
05:02	Boost to high apogee	02:24	1620	4000/106	Injection ships, GWM
19:56	Deboost	04:13	3679	400/106	TEX, MLA, GBI
63:02	MCC[b]	00:19	29	400/90	TEX, MLA
100:15	MCC2	00:02	337	200/90	CRO
167:00	MCC3[b]	--	--	200/90	--
236:35	Deorbit	00:15	283	188/-10	HAW
(d) Alternate 3b					
02:52	TLI c/o	--	--	23 000/110	Injection ship, HAW
15:37	Phasing	00:07	69	22 100/110	None
27:48	Deboost	08:41	7810	400/110	Injection ship
63:21	MCC1	00:15	35	400/90	MLA, GBI
97:43	MCC2	00:02	336	200/90	CRO
167:00	MCC3[b]	--	--	200/90	--
236:09	Deorbit	00:12	292	196/-11	HAW

[a]All timelines assume December 21 launch, 72° launch azimuth, injection on first opportunity.

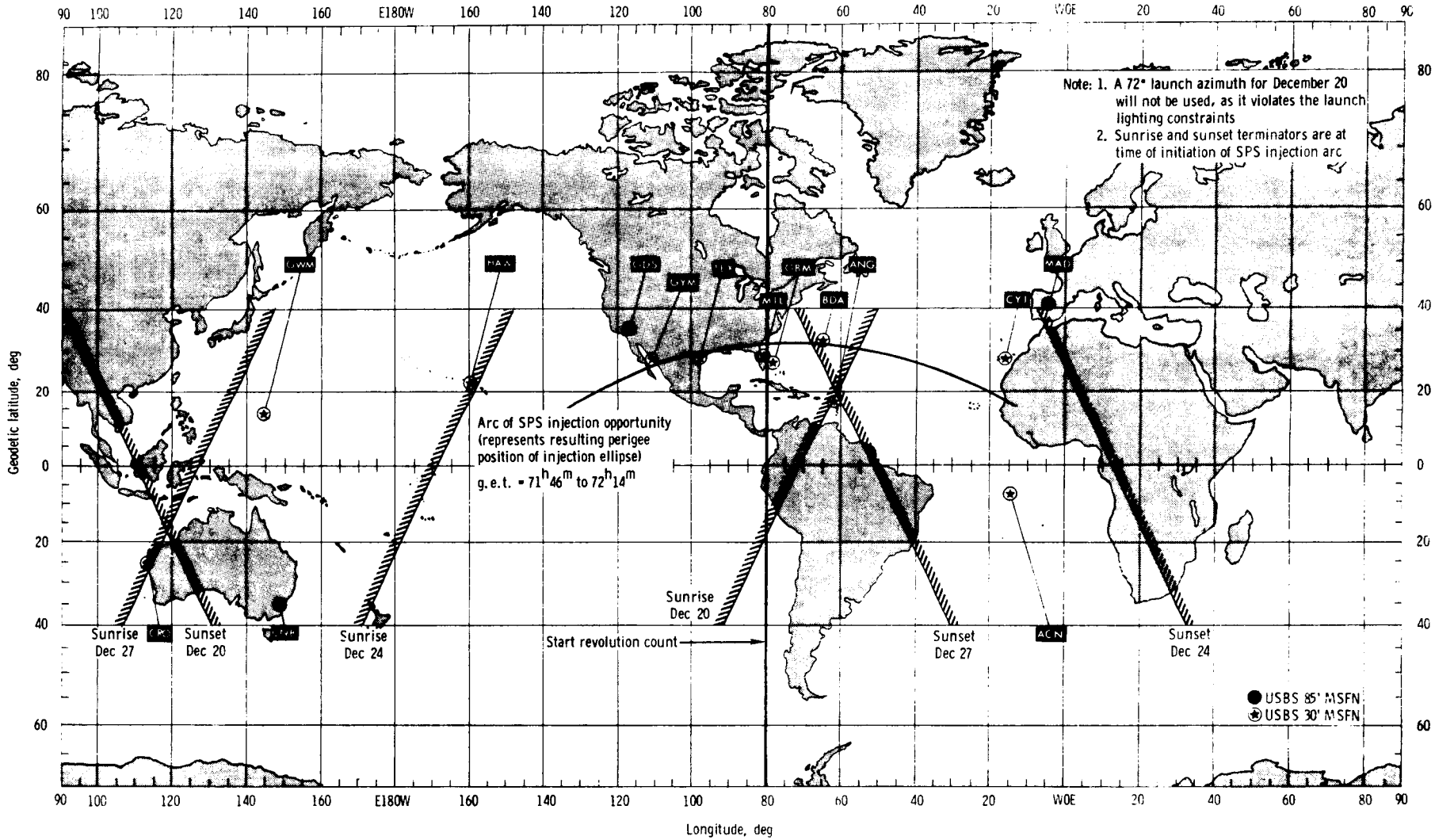
[b]Used for orbit "tuning" prior to retrofire, and is nominally zero.

TABLE V. - TYPICAL SPS MANEUVER TIMELINES FOR THE EARTH ORBITAL ALTERNATE MISSIONS OF THE APOLLO C' ALTERNATE 1 MISSION[a] - Concluded

Mission time, hr:min	Event	Duration, min:sec	ΔV, fps	Resulting h_a/h_p, n. mi.	MSFN coverage
(e) Alternate 3c					
02:52	TLI c/o	--	--	50 000/112	Injection ship, HAW
35:28	Phasing	01:00	638	29 900/111	None
70:29	Deboost to semisynchronous orbit	00:46	518	21 800/110	Injection ship
186:15	MCC1[b]	--	--	--	--
236:11	Deorbit	00:04	50	21 800/25	CRO, TAN, CNB

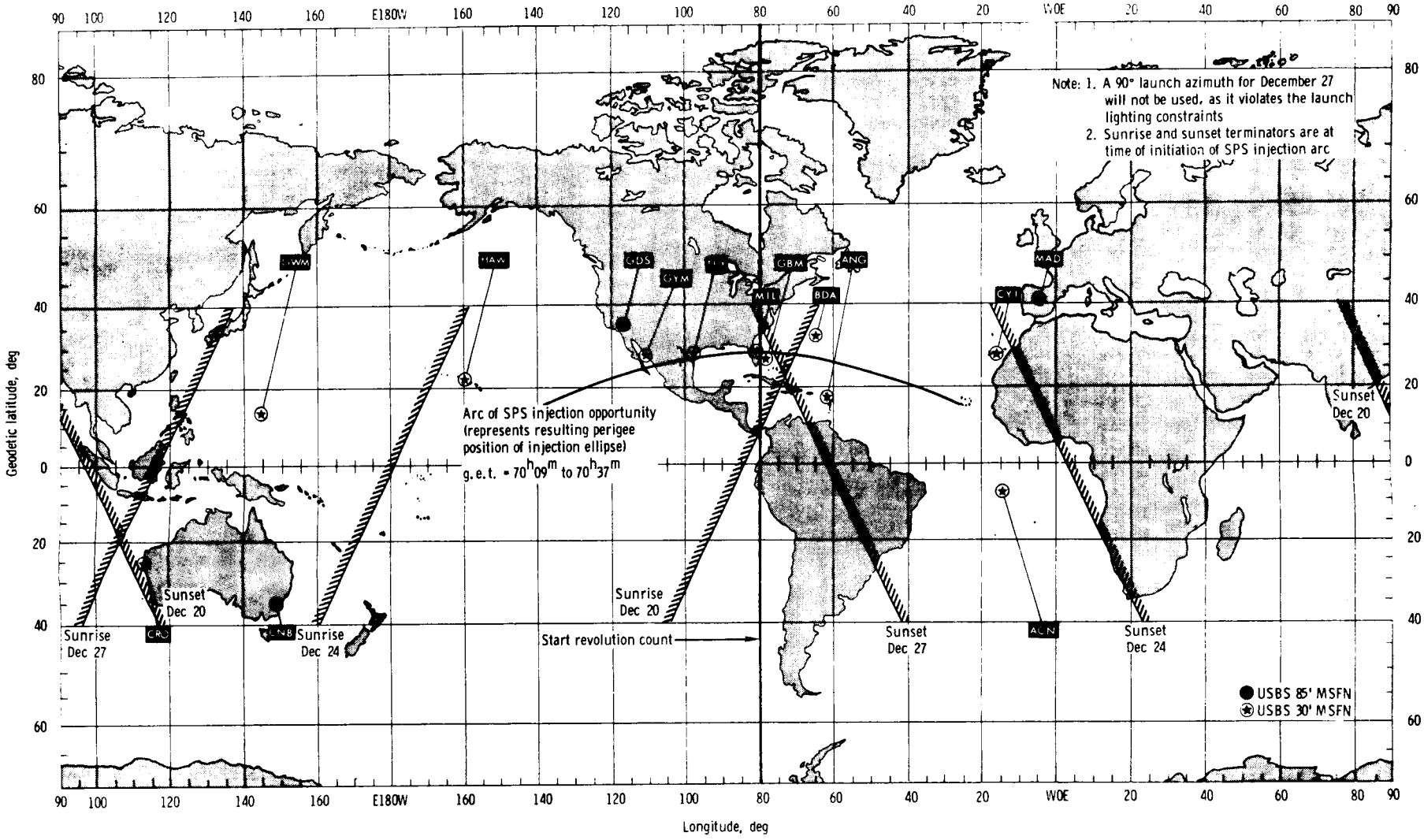
[a]All timelines assume December 21 launch, 72° launch azimuth, injection on first opportunity.

[b]Used for phasing to adjust final perigee point for deorbit; nominally zero.



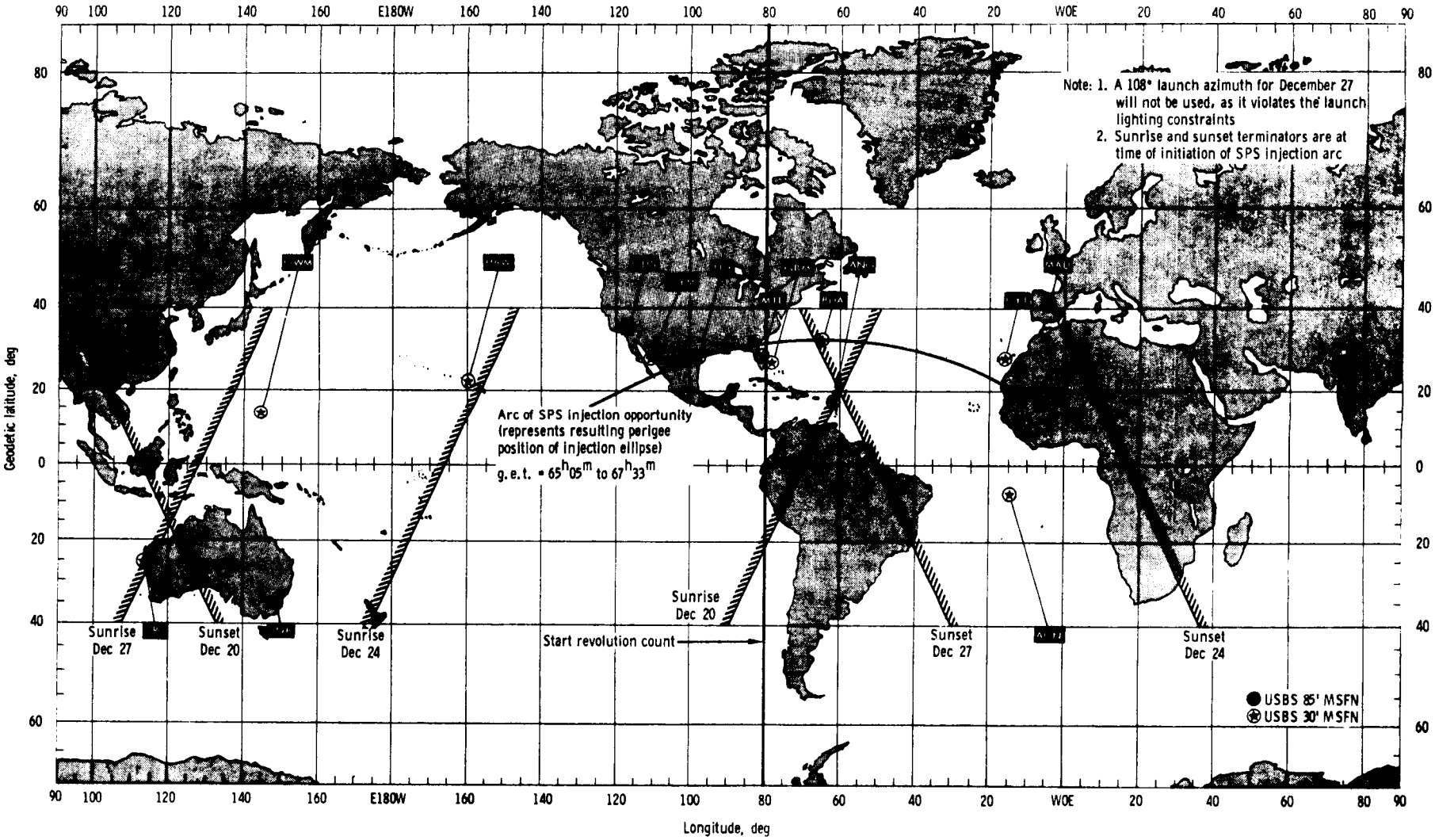
(a) 72° launch azimuth.

Figure 1. - Relative positions of C' alternate mission SPS injection high ellipse and terminators for selected days in December launch window.



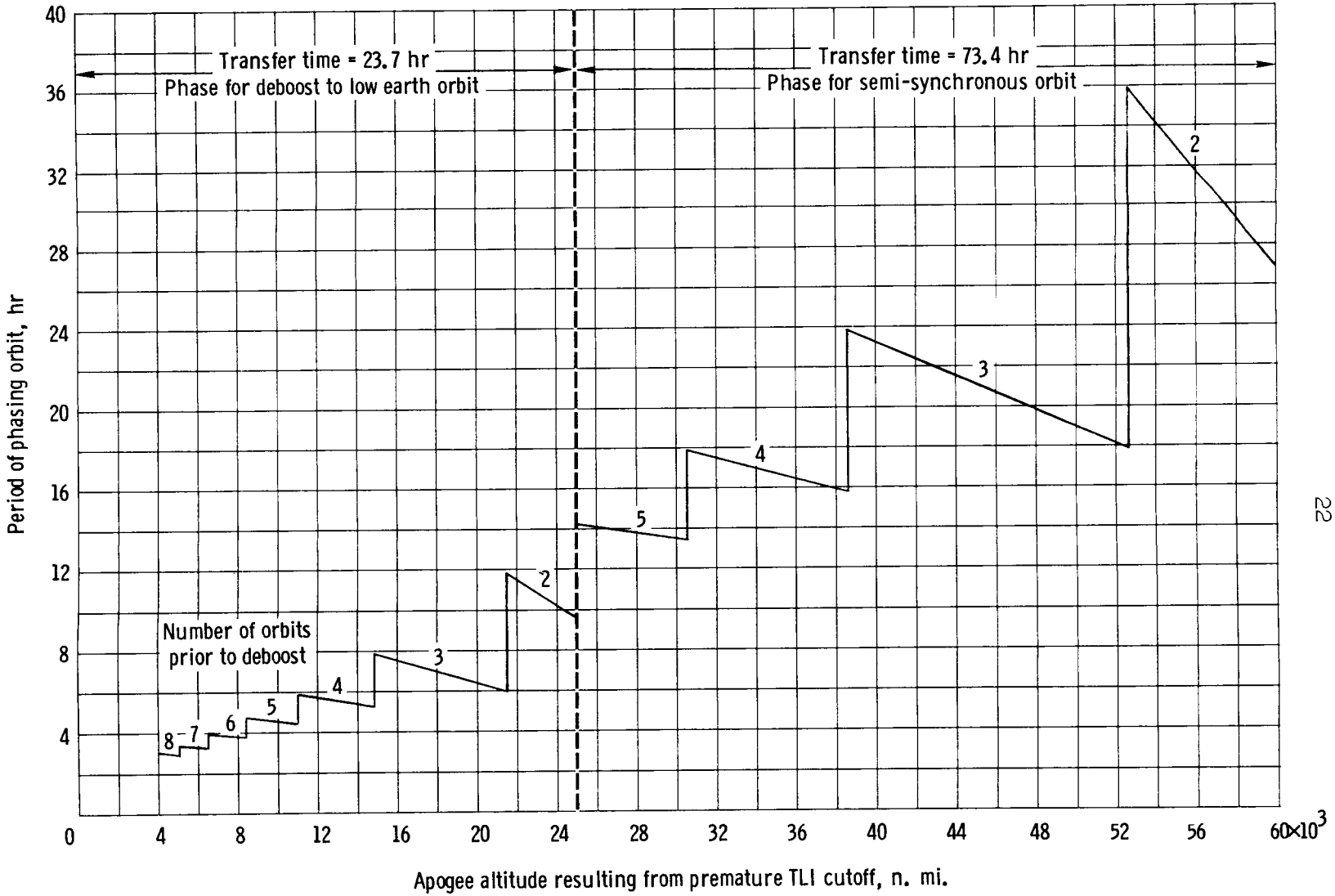
(b) 90° launch azimuth.

Figure 1. - Continued.



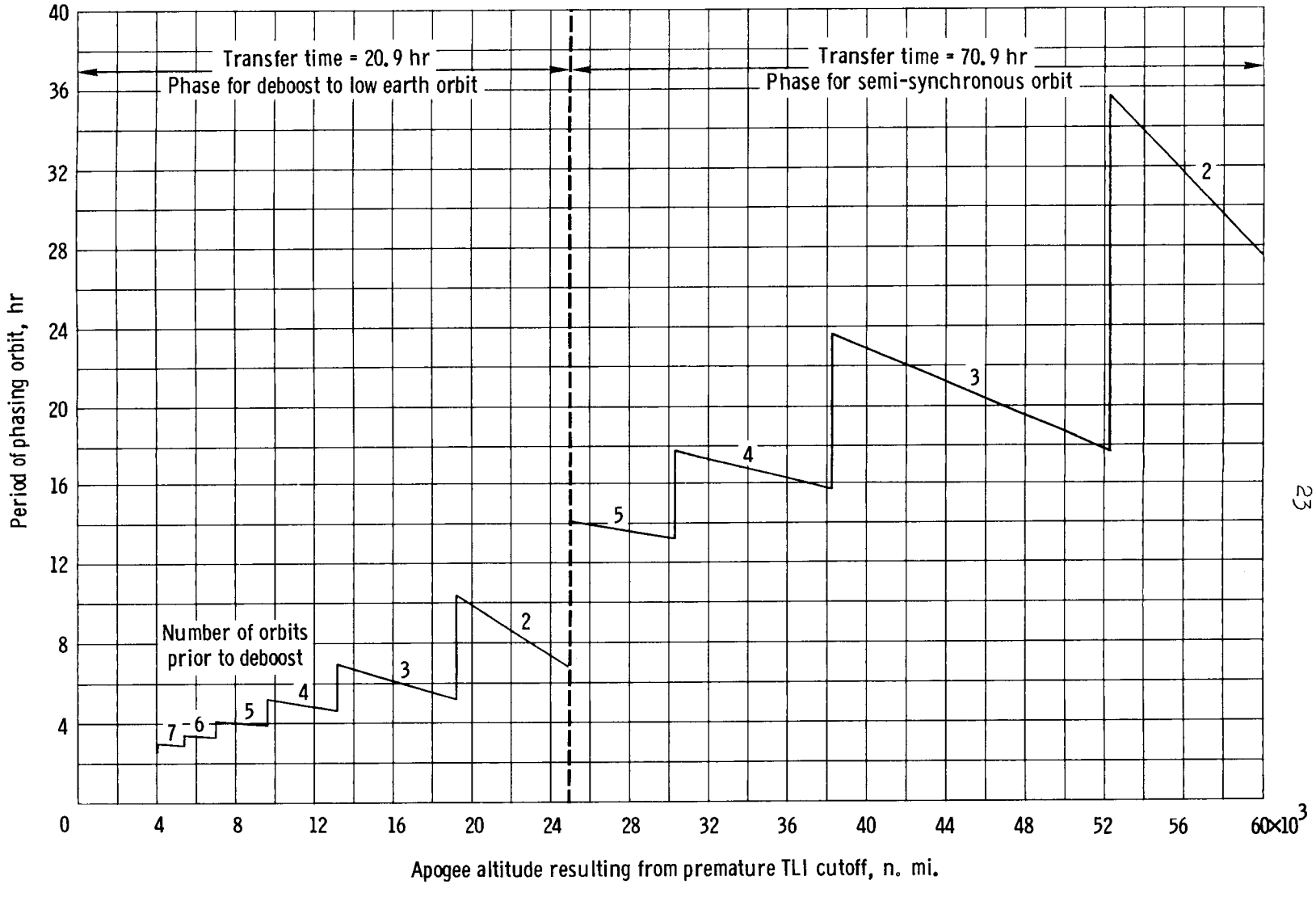
(c) 108° launch azimuth.

Figure 1. - Concluded.



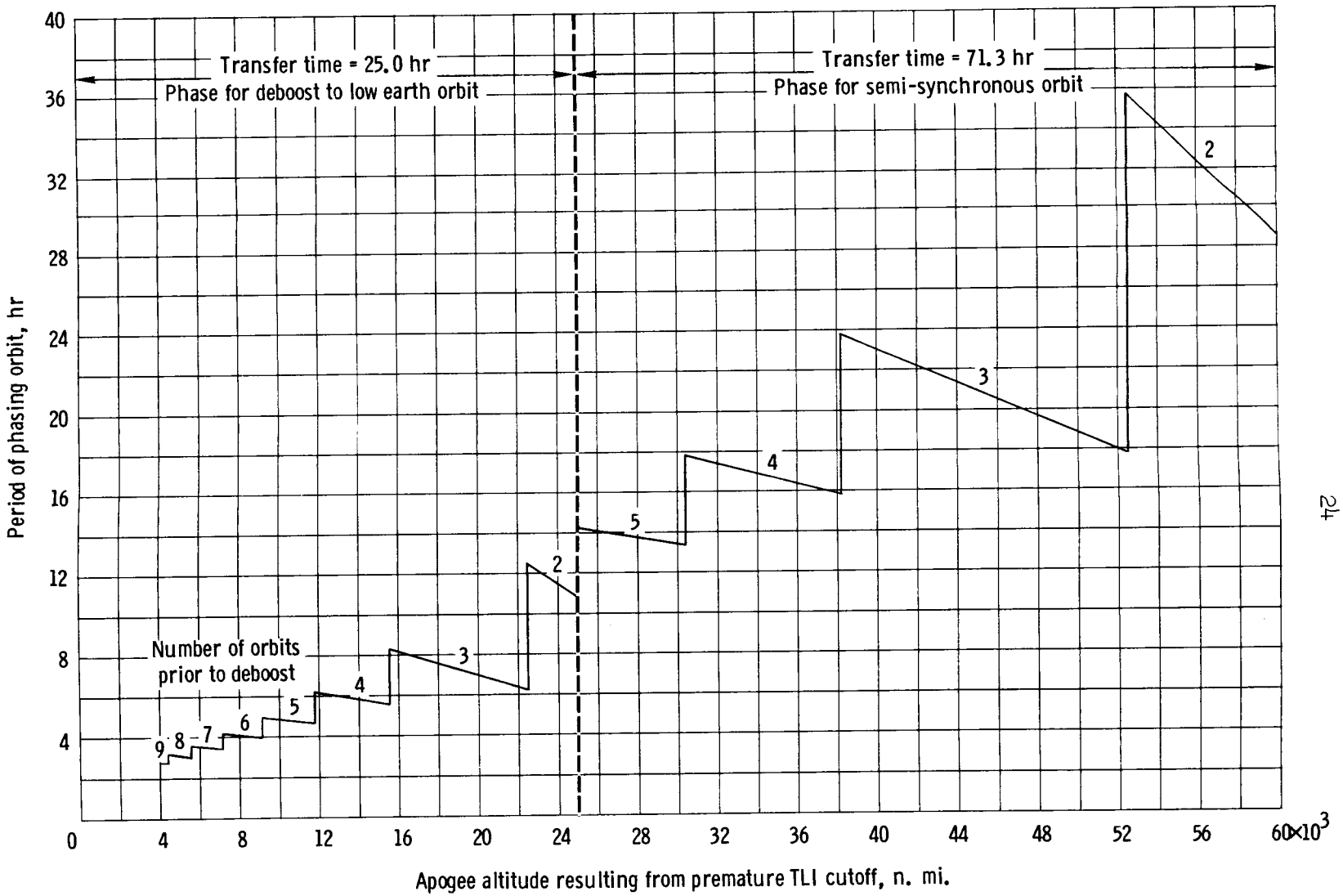
(a) December 20 launch date, 90° launch azimuth.

Figure 2.- Period of phasing orbit versus apogee altitude resulting from premature TLI cutoff.



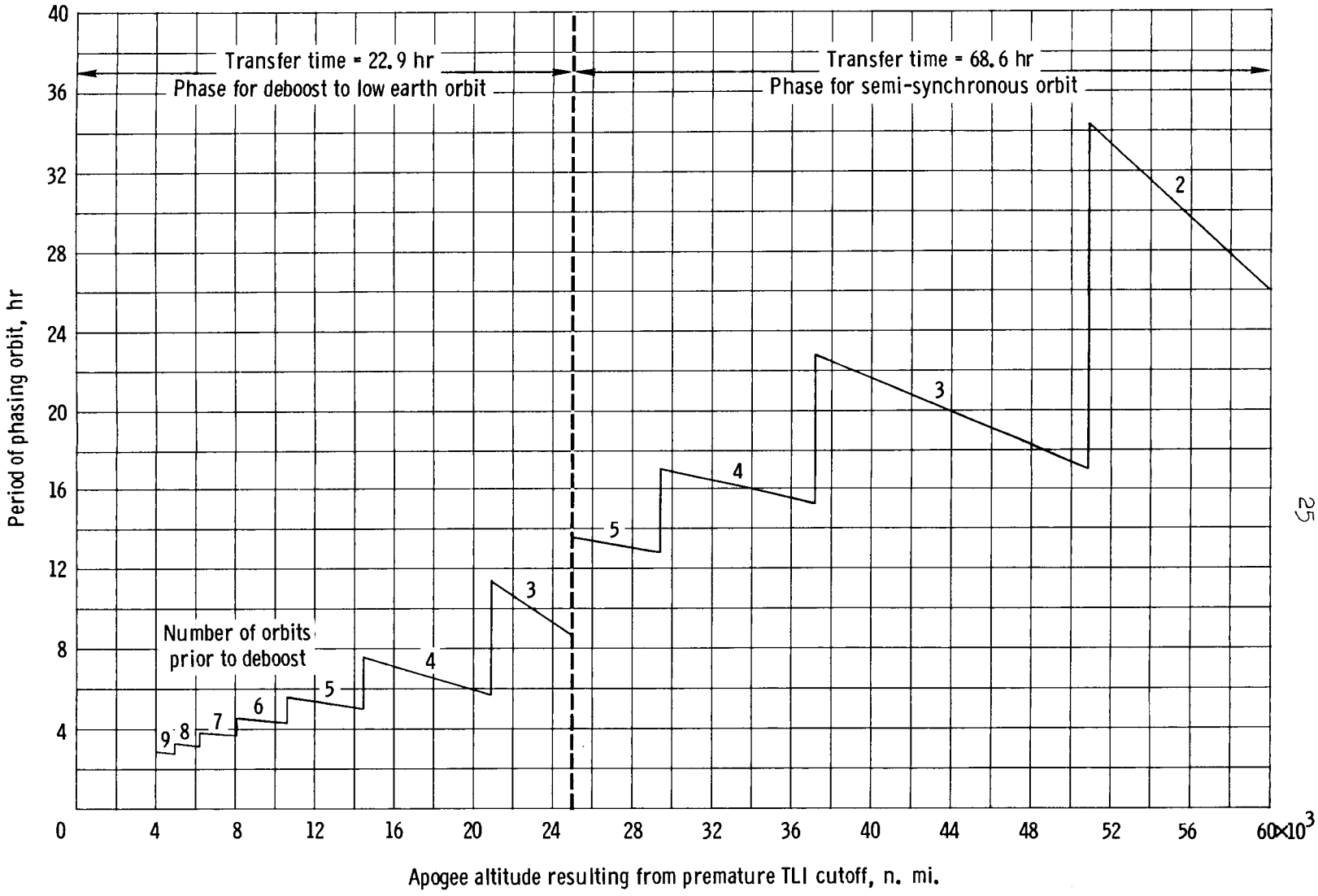
(b) December 20 launch date, 108° launch azimuth.

Figure 2.- Continued.



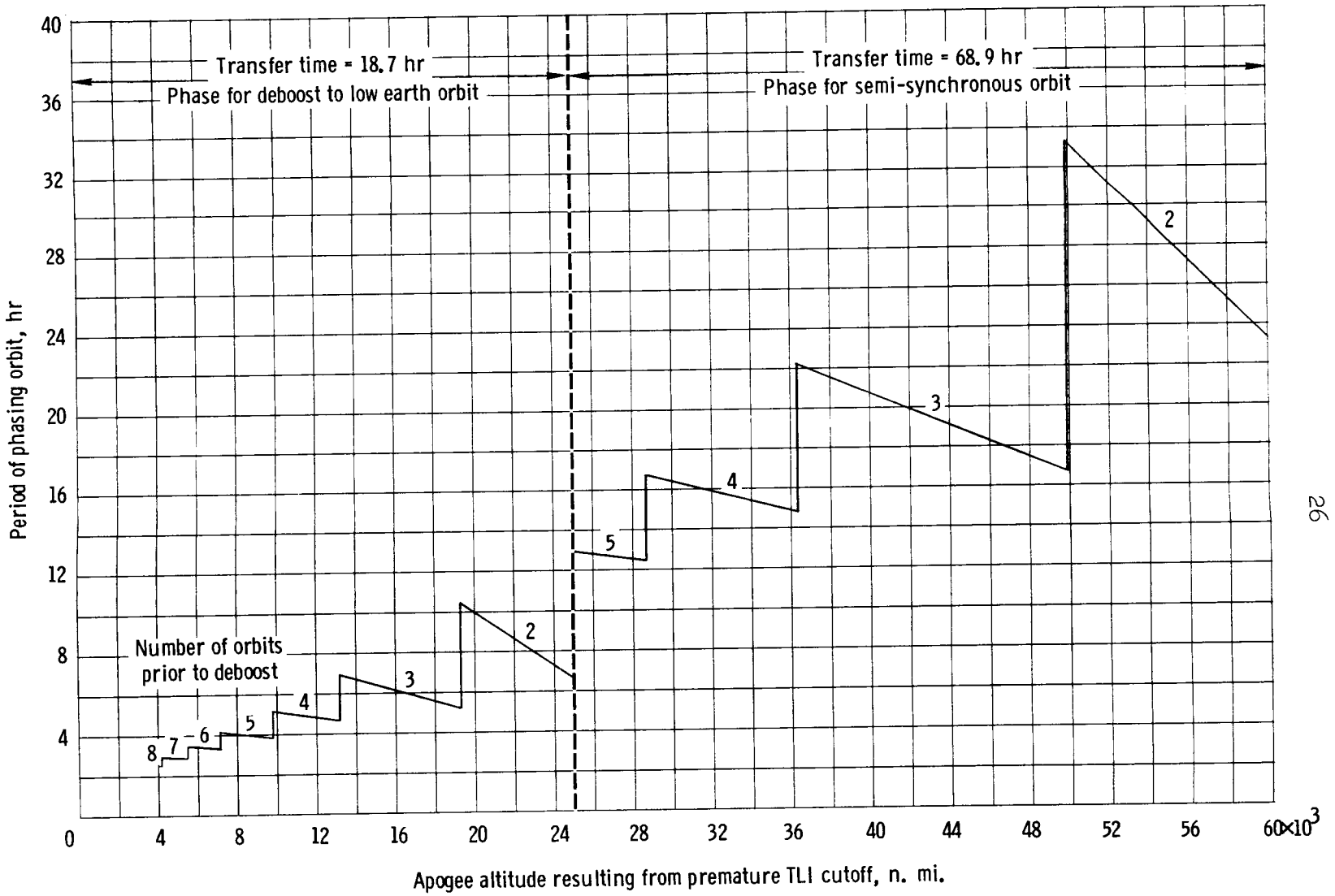
(c) December 24 launch date, 72° launch azimuth.

Figure 2.- Continued.



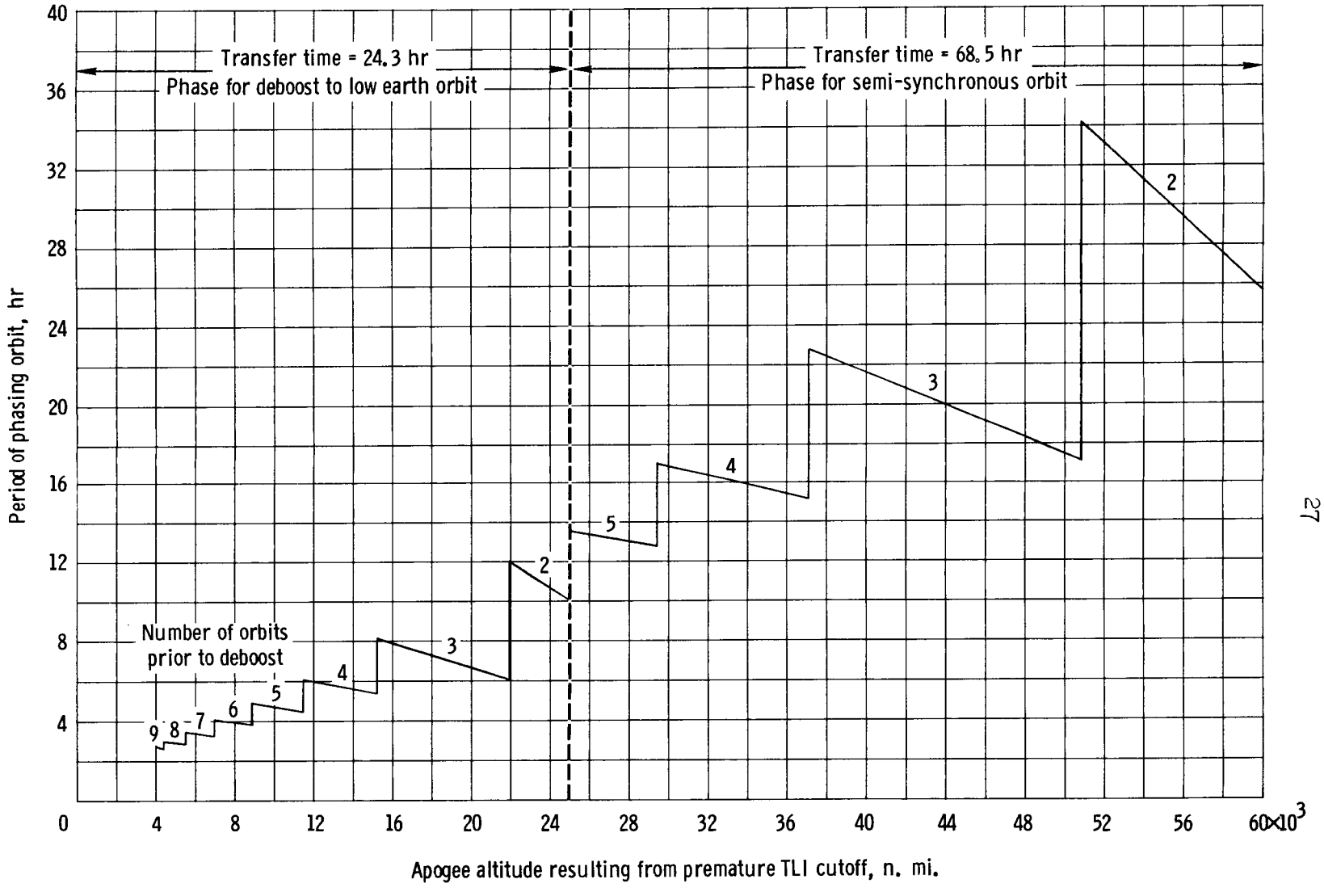
(d) December 24 launch date, 90° launch azimuth.

Figure 2.- Continued.



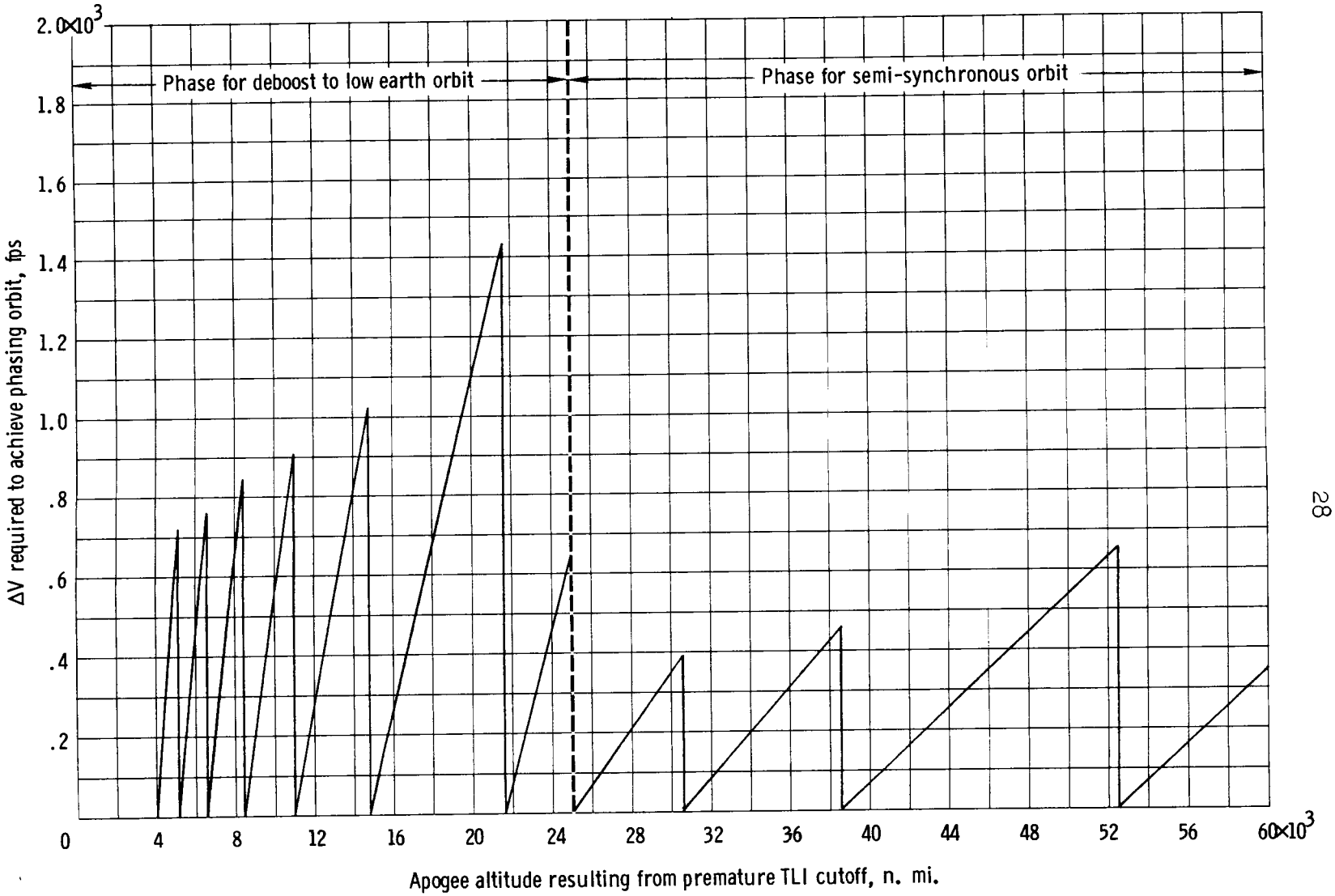
(e) December 24 launch date, 108° launch azimuth.

Figure 2.- Continued.



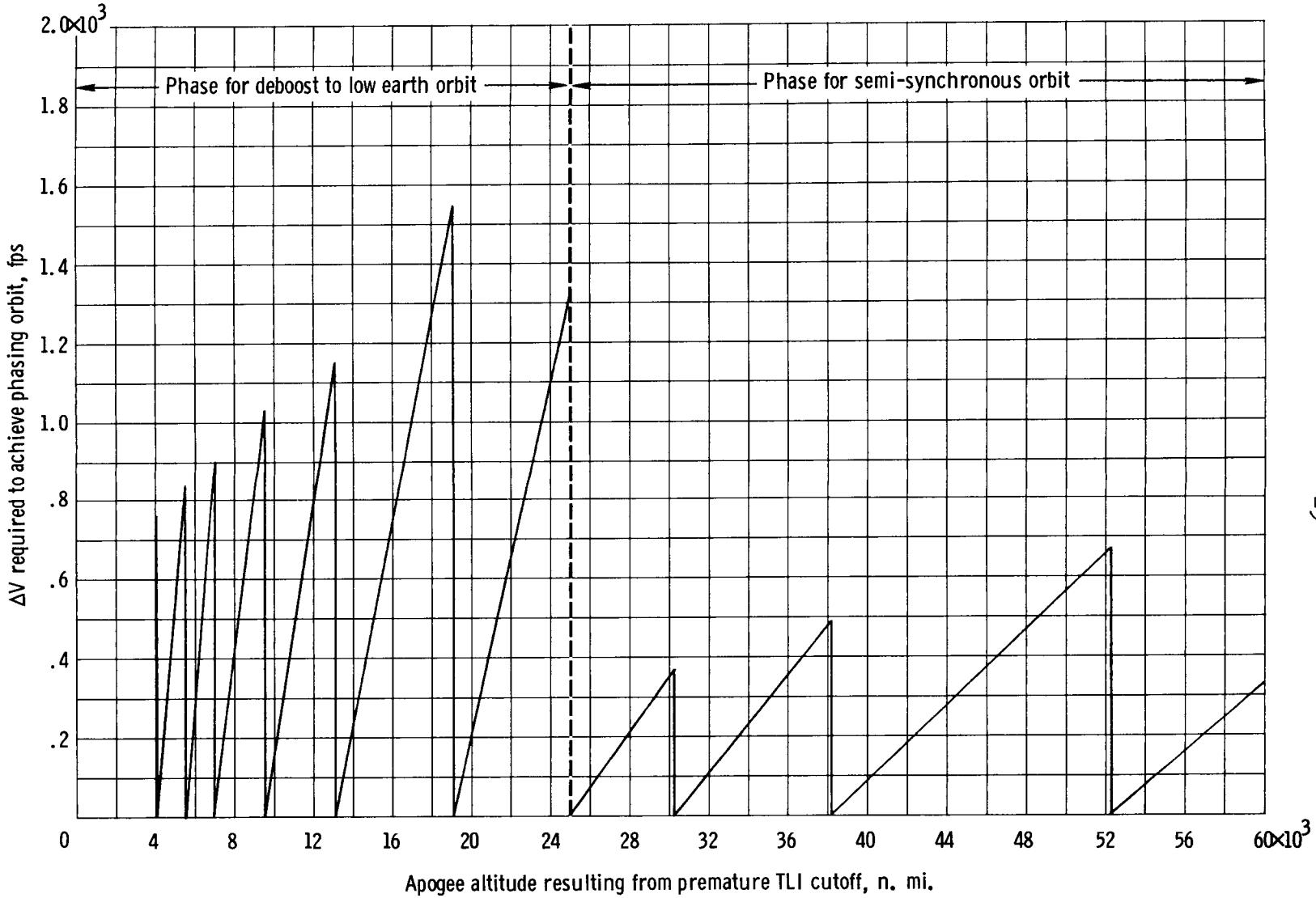
(f) December 27 launch date, 72° launch azimuth.

Figure 2.- Concluded.



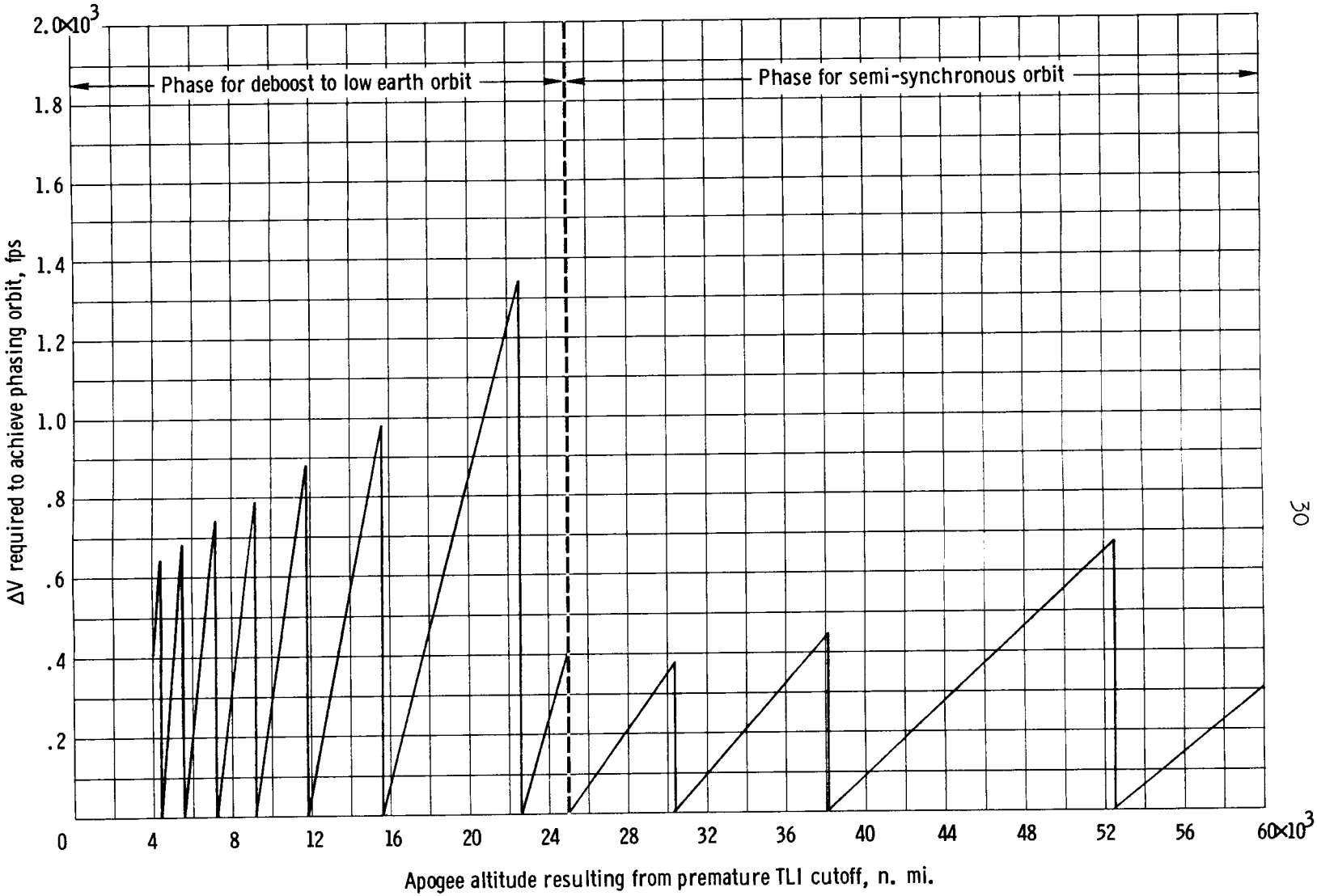
(a) December 20 launch date, 90° launch azimuth.

Figure 3. - ΔV required to achieve phasing orbit versus apogee altitude resulting from premature TLI cutoff.



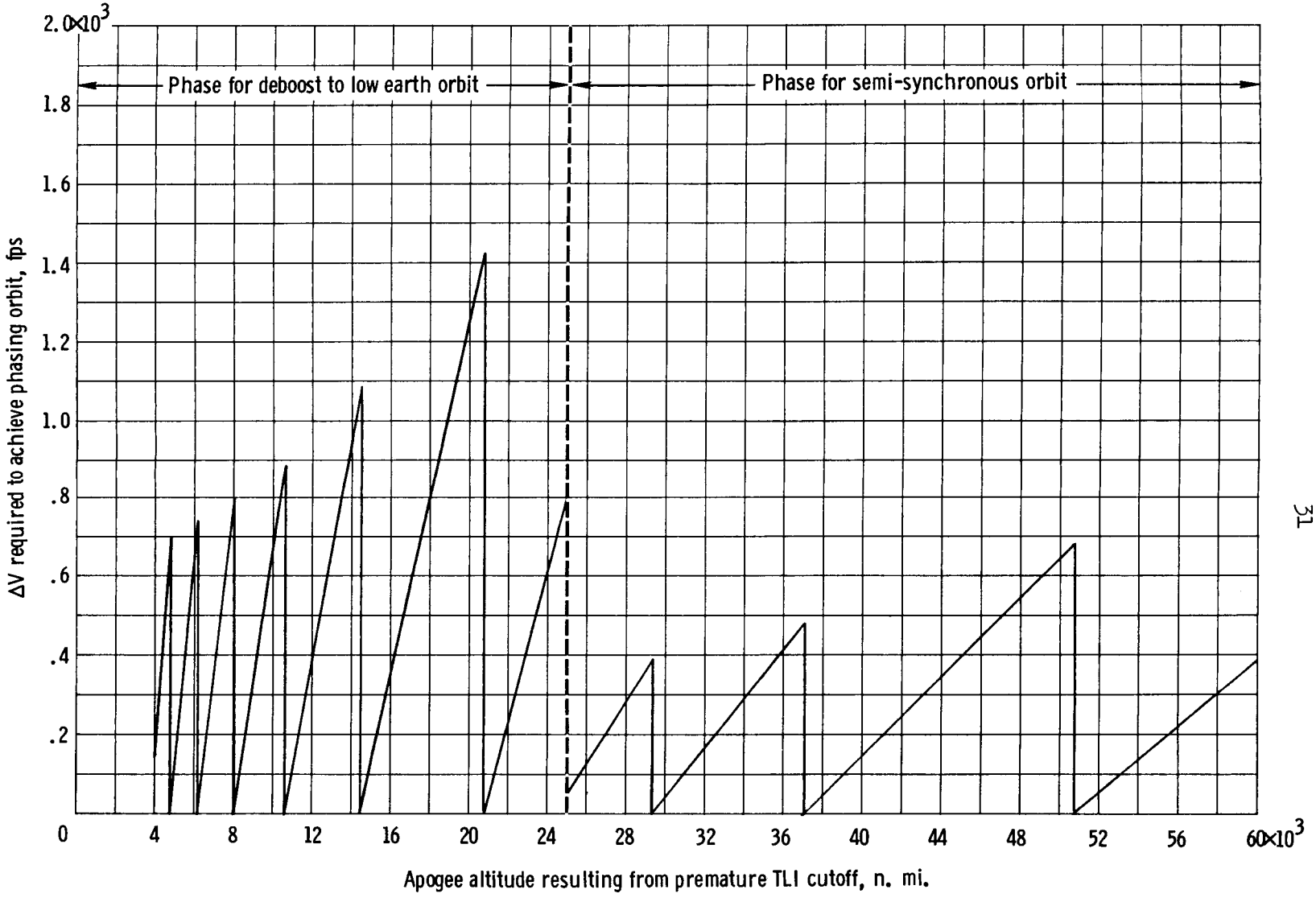
(b) December 20 launch date, 108° launch azimuth.

Figure 3. - Continued.



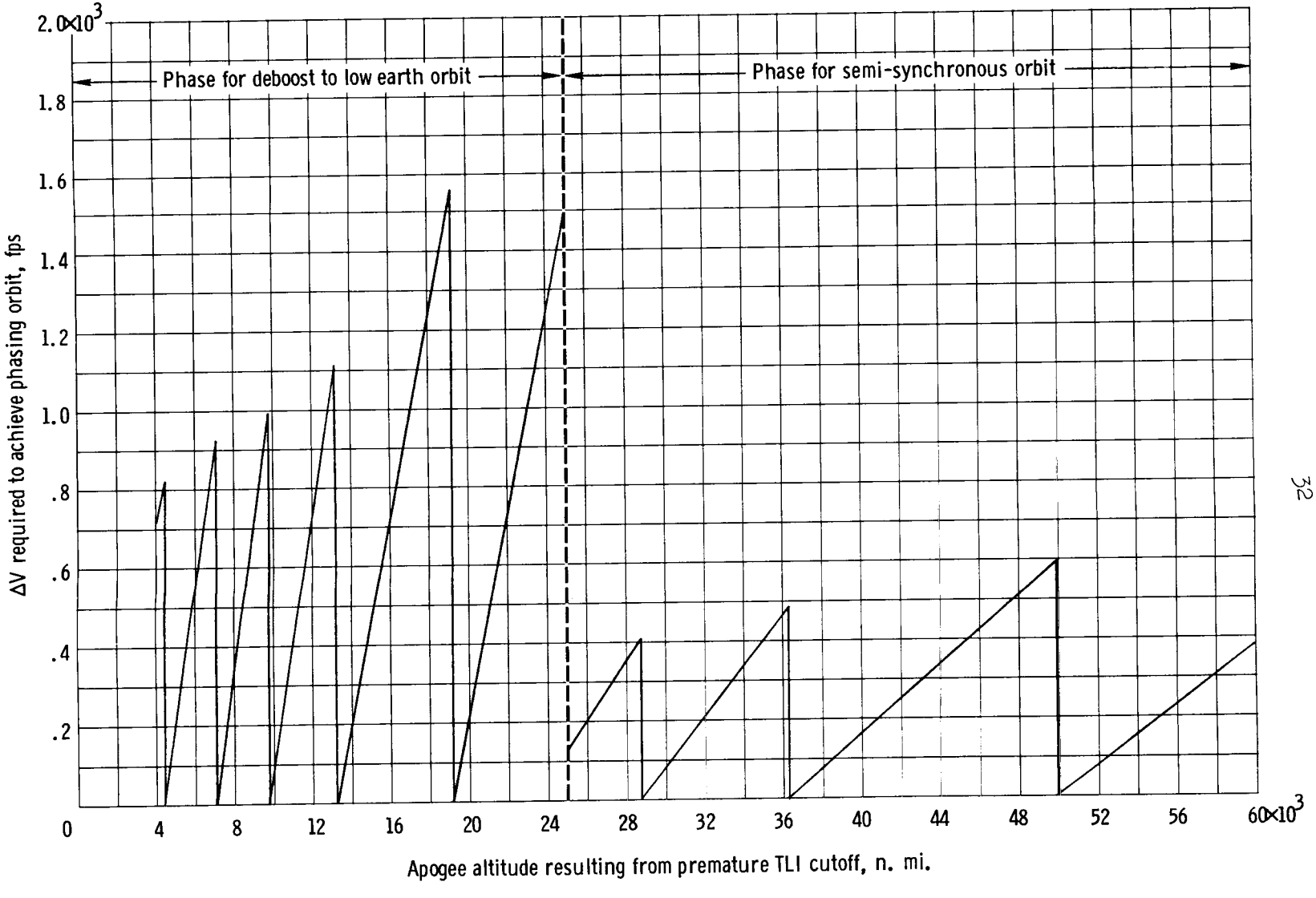
(c) December 24 launch date, 72° launch azimuth.

Figure 3. - Continued.



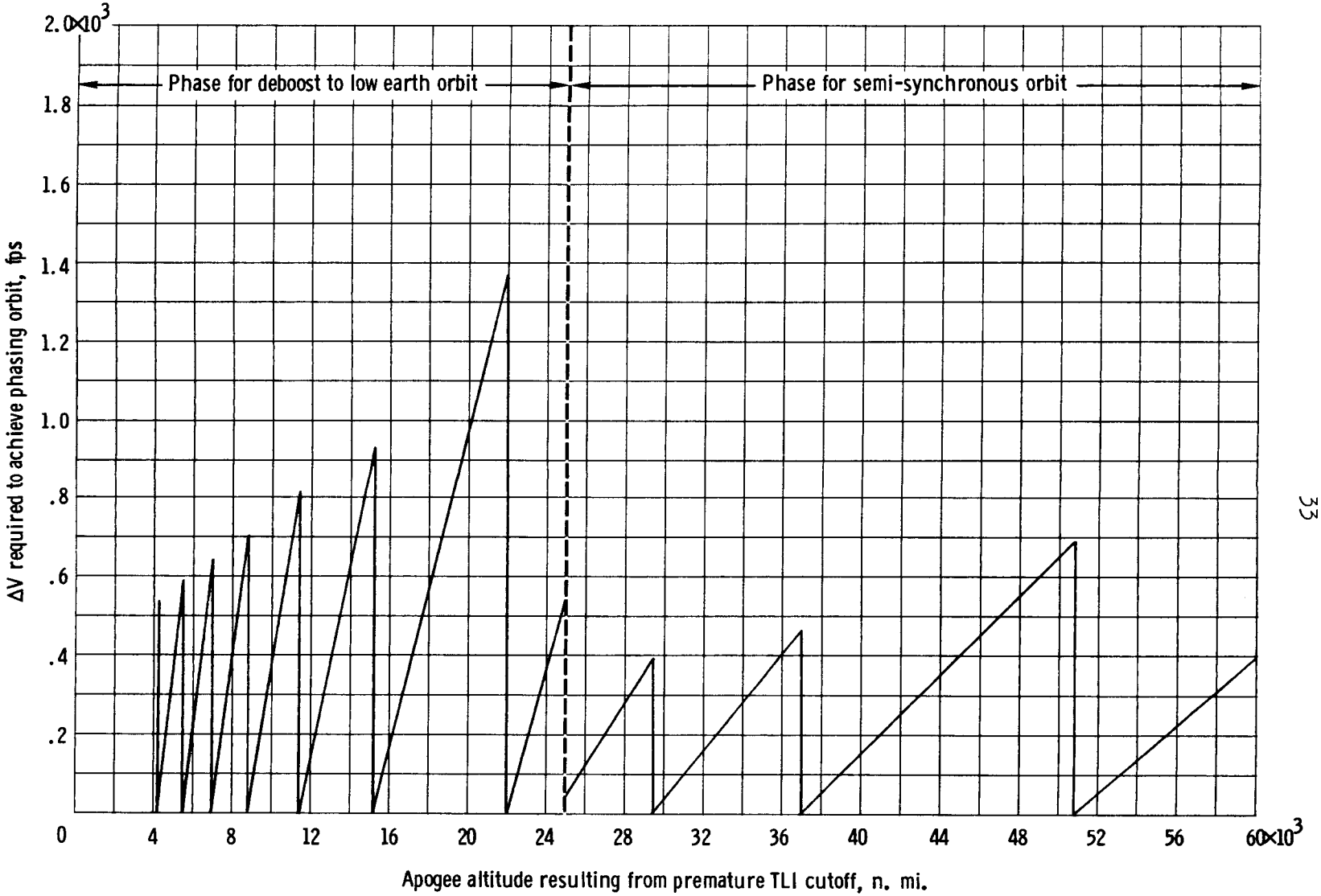
(d) December 24 launch date, 90° launch azimuth.

Figure 3. - Continued.



(e) December 24 launch date, 108° launch azimuth.

Figure 3. - Continued.



(f) December 27 launch date, 72° launch azimuth.

Figure 3. - Concluded.

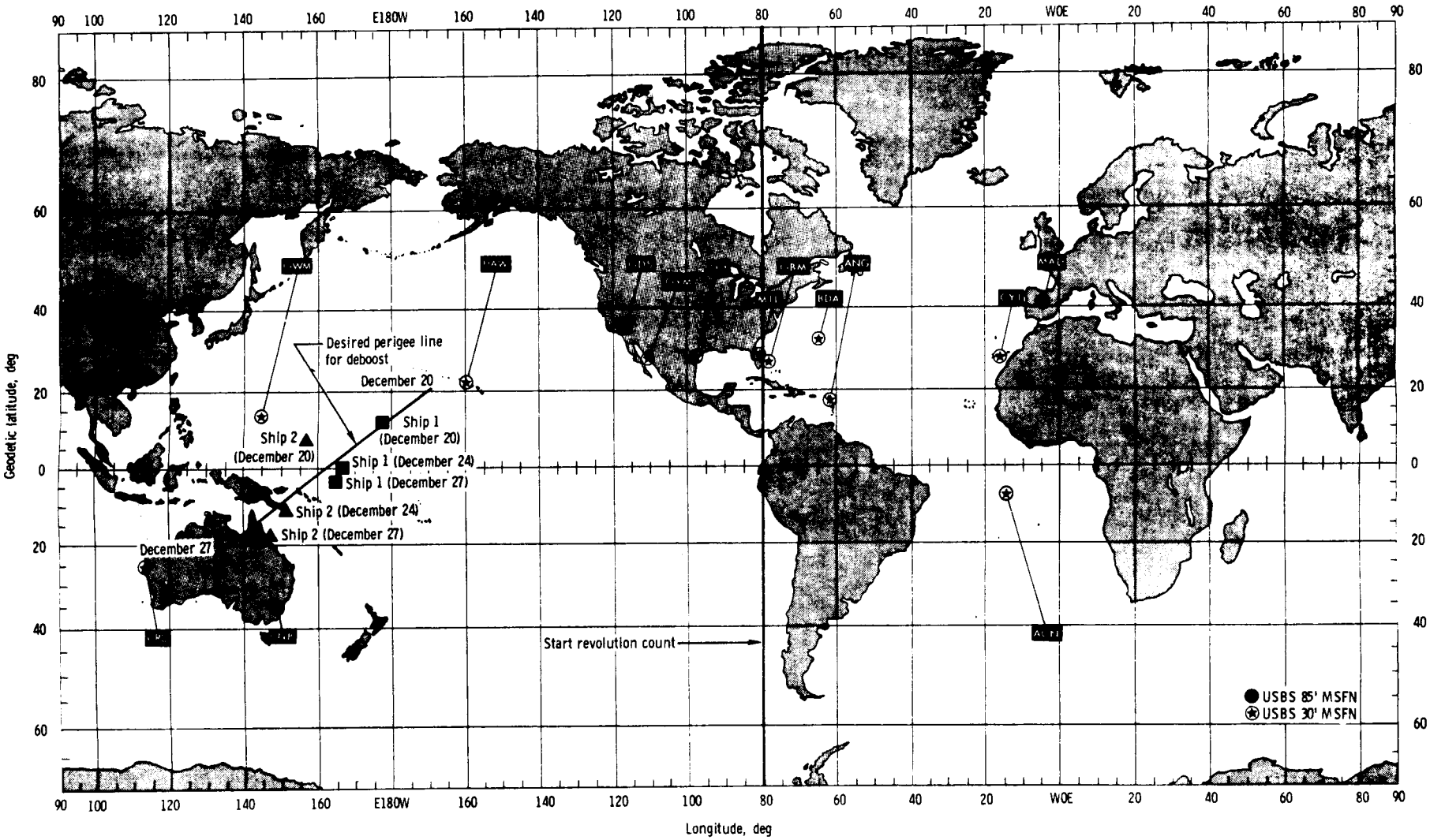


Figure 4. - Desired perigee position for deboost from high ellipse.

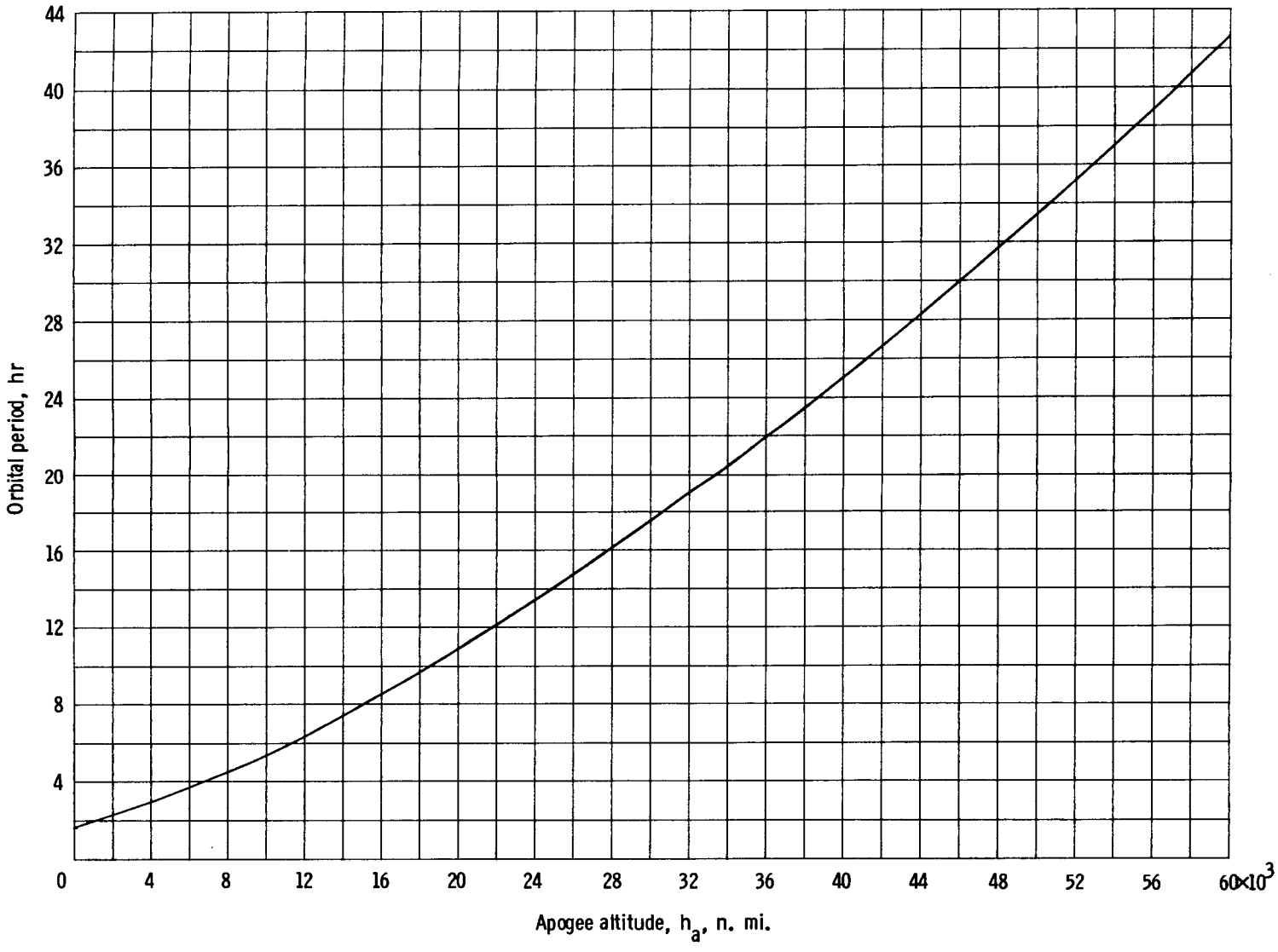


Figure 5. - Orbital period versus apogee altitude for a perigee of 100 nautical miles.

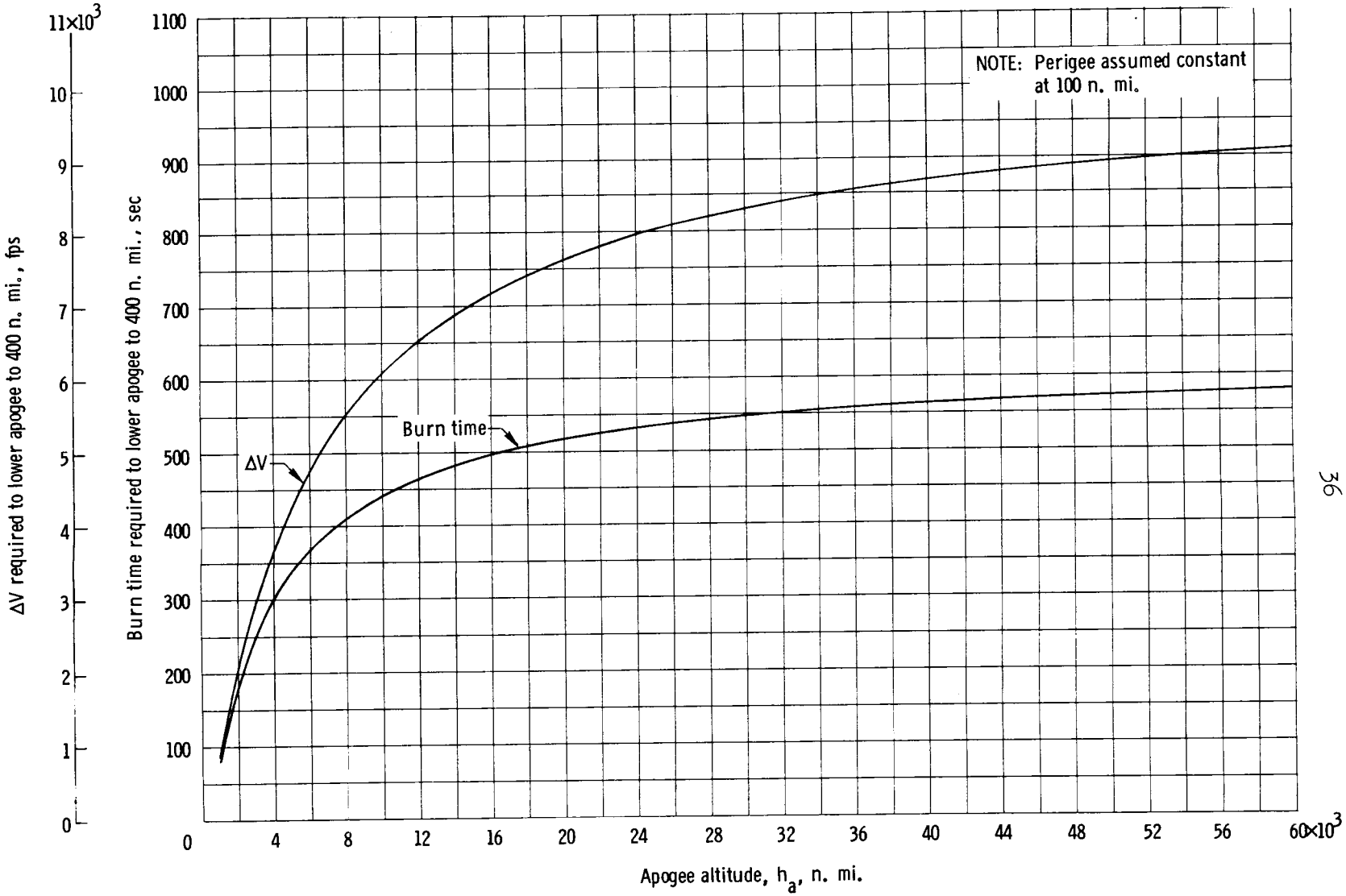


Figure 6. - ΔV and burn time required to lower apogee to 400 nautical miles.

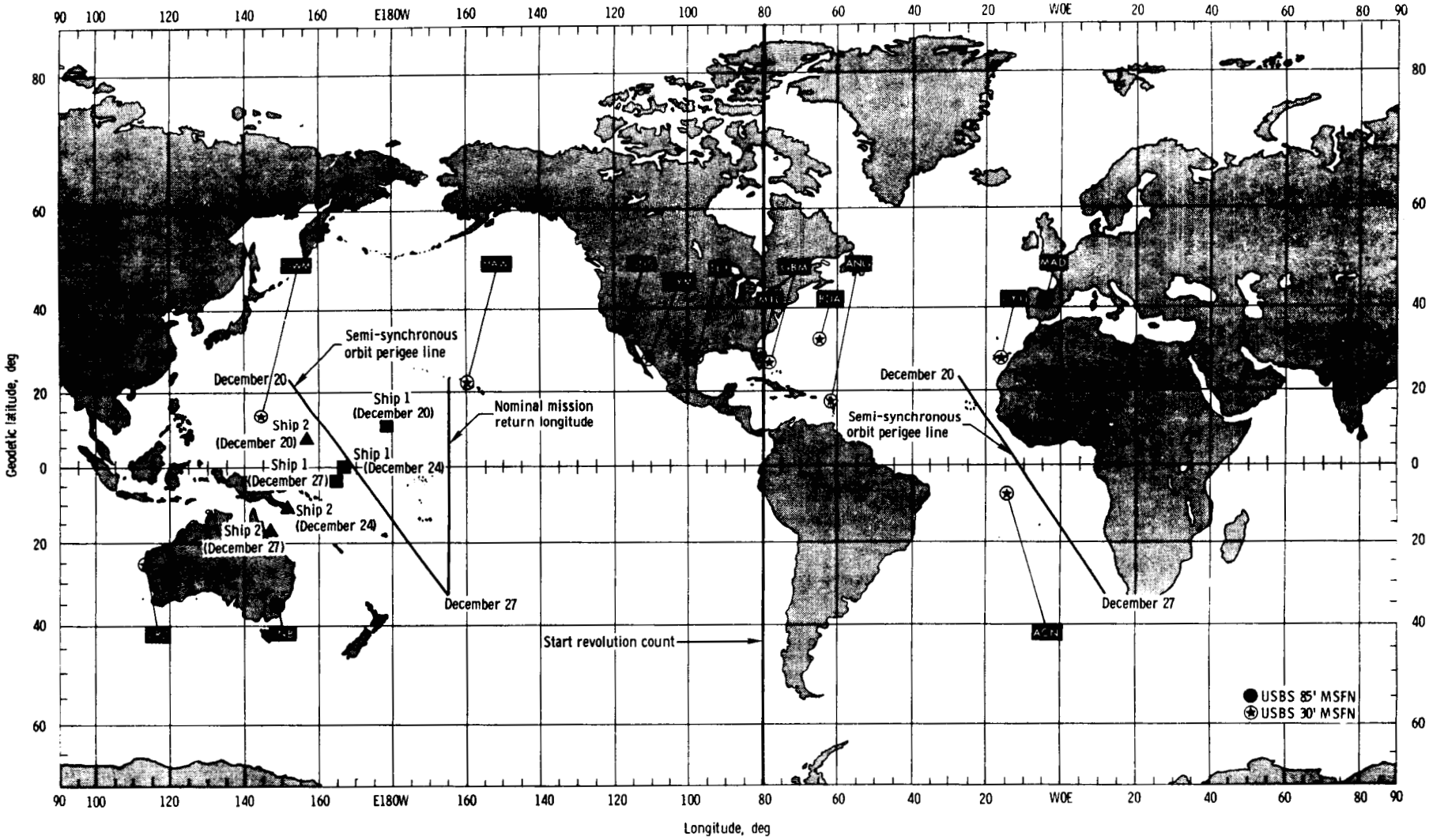


Figure 7. - Desired perigee positions for semisynchronous orbit.

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