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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC INTERNAL NOTE NO. 66-FM-104

September 23, 1966

MISSION PLANNING STUDIES FOR MANNED, CIRCULAR, SYNCHRONOUS-ORBIT MISSIONS

By Charles T. Hyle

Flight Analysis Branch

[redacted]

MISSION PLANNING AND ANALYSIS DIVISION

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

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PROJECT APOLLO

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MISSION PLANNING STUDIES FOR MANNED, CIRCULAR, SYNCHRONOUS-ORBIT MISSIONS

By Charles T. Hyle

1.0 SUMMARY

General mission planning information, such as payload capability, deorbit requirements, and hardware characteristics, was investigated in this study of manned, circular, synchronous-orbit missions for the Apollo Applications Program (AAP). Four circular, synchronous-orbit missions were investigated in detail. Two of these missions were equatorial and two were inclined. The results showed that the Saturn V (S-V) designed for the lunar mission could place 39 543 lb of payload into the equatorial orbits and 71 360 lb of payload into the inclined orbits. Because it is so small, the payload for an equatorial mission could not include a lunar module laboratory (LM lab).

2.0 INTRODUCTION

The AAP has been established to utilize the capabilities and hardware developed during Project Apollo for investigations in space medicine, the physical sciences, and advanced space technology. Current AAP planning includes two types of synchronous-orbit missions. Both are long duration missions--14 to 45 days--requiring a high altitude circular orbit. One type has a 28.5° inclined orbit and the other has an equatorial one.

This report outlines four mission plans, two for the inclined and two for the equatorial orbits, and demonstrates the S-V payload capability for earth synchronous orbits. The report also includes general mission planning information such as deorbit requirements and hardware characteristics. Since no distinct guidance techniques were used in the trajectory thrusting simulations, the report is considered preliminary.

Many useful contributions, including some of the figures, were provided by Charles E. Allday of the Flight Analysis Branch.

2.1 Mission Objectives

There are three primary objectives of an AAP synchronous-orbit mission.

a. The accomplishment of experiments contained in the command module (CM), service module (SM), and LM lab.

b. The demonstration of the capability of the S-V to place a useful payload into a synchronous orbit.

c. The demonstration of ground operations support of a manned synchronous-orbit mission.

This report is primarily concerned with the second objective.

2.2 Definition of Synchronous Orbits

A synchronous orbit is one which has a period, T, equal to that of the earth (T=23.9345 hr).

Several types of orbits satisfy the above definition: A synchronous orbit may be elliptic or circular, inclined or equatorial. Ground tracks of typical inclined elliptic and circular synchronous orbits are shown in figure 1. The ground track of an inclined circular synchronous orbit appears as a figure-eight which degenerates to a point as the inclination is reduced to zero. (See figure 2.) Figure 2 also shows the interesting variation of longitude with time as affected by inclination. Circular synchronous orbits are defined by the following orbital elements:

Inertial velocity, V_I, fps.	10 088
Inertial flight-path angle, γ_I, deg, ...	0
Radius from center of earth to satellite, R, ft .	20 925 738
Altitude, h, n. mi.	19 323

3.0 TRAJECTORY SIMULATIONS AND RESULTS

Computations for the four missions considered in this report were made with the CO3E, three-degree-of-freedom, integrating, particle trajectory program, except where noted. As stated previously, the thrust phase simulations used no explicit guidance techniques. In-plane thrust maneuvers were applied along the current inertial velocity vector, and out-of-plane maneuvers were oriented with respect to the inertial velocity vector. The simulated atmosphere used in the reentry phase was the 1959 ARDC atmospheric model. The earth model was the Fischer ellipsoid. As stated previously, the launch vehicles and spacecrafts for the AAP are those of the Apollo Program.

3.1 Launch Vehicle

3.1.1 General description.-The propulsion capability to be used to launch manned, synchronous-orbit missions is that of the S-V plus part of the service propulsion system (SPS) capability. The S-V is a three stage vehicle whose stages are designated S-IC, S-II, and S-IVB. The launch configuration is shown in figure 3.

Stage one of the S-V is powered by five F-1 engines with a total thrust rating of 7.5 million pounds. The S-II second stage is powered by five J-2 engines with a thrust rating of 200 000 lb per engine. The third stage S-IVB is powered by a single gimballed J-2 engine rated at 200 000 lb.

Pertinent launch vehicle and propulsion characteristics may be found in table 1 and figure 4.

3.1.2 Performance.-Launch vehicle characteristics used for the two equatorial missions described in this report were obtained from reference 1, and those for the two inclined missions were obtained from reference 2.

As indicated by the second mission objective, the demonstration of the capability to launch and insert a useful manned satellite into synchronous orbit is a feat in itself. As will be shown in the following pages, a comparatively small experiment payload (in a circular equatorial orbit) is all that is obtainable with the present S-V configuration.

North American Aviation (NAA) has examined in detail an equatorial synchronous-orbit mission (ref.1). They assumed, however, that costly modifications to provide additional S-IVB restart capability would be made to the launch vehicle. Because of the current fiscal situation, costly modifications are highly undesirable. Therefore, a basic problem to be solved by this study was how much payload can the current S-V configuration insert into a circular, synchronous, equatorial orbit and how much into an inclined, circular, synchronous orbit.

Since reference 1 established an inclined 100-n.mi. circular orbit by a partial burn of the S-IVB prior to the use of the assumed modified capability, this parking orbit was used as initial conditions for the two equatorial missions to be described.

Similarly, reference 2, in its outline of an inclined, circular, synchronous-orbit mission, established a 100-n.mi. parking orbit, which was taken as starting conditions for the two inclined missions described in this report.

The two launch phases used for initial conditions are shown in tables 2 and 3.

All four mission profiles were launched due east and resulted in a 28.5° inclination at the 100-n. mi. orbit.

3.2 Spacecraft

3.2.1 General description.- The orbiting spacecraft, as initially conceived for the AAP, is comprised of the CM, SM, and the LM lab. This configuration is shown in figure 5.

The CM, as the primary crew station, contains life support equipment and spacecraft controls. Since it is also the reentry module, it contains the necessary landing and recovery equipment.

The orbit maneuver capability, including that for deorbit, is provided by the SPS of the SM. The SPS is also required for orbital insertion for all missions examined in this report. An experiments pallet is installed in one of the SM equipment bays and is expected to hold much of the required experiment hardware.

Minor modifications, mainly removing unnecessary hardware, are necessary to the LM lab. One of the main functions of the LM lab in the synchronous mission will be to house a crew member who is making astronomical observations.

3.2.2 Performance. SPS thrust characteristics and CM aerodynamics are listed in tables 1 and 4, respectively. Deorbit propellant requirements are included in the inserted payload.

3.3 Mission 1 - Equatorial

Since an equatorial synchronous orbit remains stationary with respect to the earth, one of the prime considerations for this type of orbit is selecting the desired hover-point (HP).

The rationale for selecting 101° E longitude as the HP for mission 1 was that this longitude was located at the first opportunity for synchronous-orbit insertion.

Following the first of the two available S-IVB burns, the LV coasted in the 100-n. mi orbit, almost to the node, then the remainder of the S-IVB fuel was used to obtain an apogee altitude of the required synchronous orbit and the inclination was reduced to approximately 7°.

Approximately five hours later, just prior to apogee, the SPS was ignited to remove the remaining inclination and to circularize the orbit to 19 323 n. mi. The total payload was 39 543 lb. (See table 5.)

Since no firm duration requirement exists, the mission was concluded after about 14 days. Retrofire occurred at about 2 hours before the 14th day, and the direct descent landing occurred 5 hours later. The landing point was in the Pacific, southeast of Hawaii, at 154.9° W, on the equator.

A detailed description of the boost phase parameters for this mission is provided by table 5 and figure 6. A ground track of mission 1 from launch through reentry is shown in figure 7. Unified S-band radar stations considered for this report are identified in table 6, and their coverage of mission 1 is contained in figure 8.

Following SPS insertion into the sychronous orbit, only 1350 lb of payload may be considered useful for experiments. This weight includes life support and deorbit requirements. Table 7 summarizes distribution of the inserted payload.

Because of the comparatively small payload, this mission plan would not include the LM lab, and the experiments hardware would be stored in the CM and SM experiment pallet.

3.3.1 Mission 1 reentry.- A footprint about the nominal landing point, as defined by various reentry lift-vector orientations, is shown in figure 9. Time histories of pertinent orbital elements and reentry parameters from retrofire to 50 000 ft are shown in figures 10 and 11. Although no guidance was used in the reentry simulation, one half of the value of the aerodynamic lift coefficients were used, which produces the same landing point as a guided reentry. Because these half-lift coefficients were used, the time histories in the figures from 400 000 ft to 50 000 ft do not describe the actual conditions during a guided reentry. They are included mainly to show trends. Tracking station coverage from retrofire to 50 000 ft is presented in figure 12. The CM heat shield is more than adequate to withstand the atmospheric entry from a synchronous orbit.

A heating rate of 216 BTU/ft^2/sec and a total heat load of 25 628 BTU/ft^2 are typical values (ref. 1) resulting from the direct descent from a synchronous orbit. These values are about 30% lower than those expected from a lunar flight.

3.3.2 Deorbit Requirements.- The descent from a synchronous orbit is governed, of course, by the amount and direction of the retrograde maneuver. A descent which produces a flight-path angle (γ_i) of about -6° and an inertial velocity (V_i) of 33 835 fps is referred to the middle of the entry corridor. Such an entry is considered most desirable in that it provides proper ranging to allow for lift-vector control and heat dissipation without exceeding untolerable g loading on the crew. Deorbit requirements for the mission 1 spacecraft were established with figures 13(a) through 13(d). The deorbit propellant required to hit the desired V_i and γ_i at the 400 000 ft "reentry corridor" is 15 183 lbs. Reentry time is about 5 hours and 17 minutes, and longitude increment is 108°. These figures and figure 14 show that the most efficient way to make the deorbit maneuver is to maintain a 180° attitude with respect to the inertial velocity vector.

The deorbit propellant required to hit the desired reentry corridor are provided for various spacecraft weights, by figures 15 and 16.

The thrusting capability of the four RCS SM attitude control (quads) to correct a nonnominal SPS deorbit maneuver is shown in figures 17(a) and 17(f). These solid propellant rockets, rated at 100 lb, are shown to be useful for only small SPS thrust dispersions and then only if the corrective maneuver is made prior to a true anomaly of 245°. That is, assuming 400 of the original 800 lb of RCS propellant is used to produce a 150 fps ΔV, only a 1% nonnominal SPS deorbit maneuver could be corrected, and this could be done only if the correction is made before a true anomaly of 245°.

Use of the RCS as a means of deorbiting from a secondary orbit, say 100-n. mi. circular, could be accomplished; however, the direct descent is recommended.

3.4 Mission 2 - Equatorial

Equatorial mission 2 is basically the same as mission 1, including payload, except for the location of the HP. The launch phase is identical through the second S-IVB burn, which provides the synchronous altitude requirement at apogee and reduces the inclination. However, instead of circularizing the orbit at apogee, a maneuver is performed to increase the perigee to 9783 n. mi. and, therefore, increase the period. One orbit later, at the next apogee, the desired HP has rotated beneath the apogee. The SPS is then used to complete insertion into the synchronous orbit above the HP. This HP, 151° W, was selected for the favorable tracking coverage and the proximity of the Brazilian coast following a direct decent entry. The same HP could have been obtained by using the same phasing maneuvers as were used for mission 1, but the fuel saved by the method just described makes it the obvious choice.

Very useful planning information is provided in figure 18. Its utility can be illustrated as follows: Point A represents the spacecraft at apogee of the 19 323/100-n. mi orbit (the above intermediate orbit). The incremental longitude between this apogee and the desired HP is then read on the abscissa. The intersection of the slanted line from point A and a vertical line through the delta longitude produces point B.

Point B then indicates the required orbit period and inertial velocity--left hand scale--that will produce the desired longitude shift by the time the spacecraft gets back to apogee. Since the chart was made using impulsive quantities, the answer differs slightly from the integrated quantities shown in table 8. The right hand ordinate shows point B to require a 9750-n. mi perigee, compared to 9783-n. mi. integrated value. A good estimate of the ΔV requirement is also obtained by subtracting the inertial velocity of the 100-n. mi. orbit from the inertial velocity of the 9750-n. mi. orbit. This indicates that 3400 fps are needed. The actual value was 3600 fps. Figure 19 shows apogees and perigees produced by various V_i and γ_i at 19 323 n. mi.

Insertion time histories, a ground track, and tracking coverage are shown in figures 20, 21, and 22. Additional details of the launch are itemized in table 8. Since the reentry phase is the same as for mission 1, the entry sequence, shown in table 8, produced a landing at 45° W on the equator. This site is in the Atlantic just off the coast of Brazil.

3.5 Mission 3 - Inclined

The payload potential of the Apollo S-V is substantially more for the inclined synchronous orbit then for the equatorial. In fact, the payload inserted in the inclined orbit was 71 360 lb compared to 39 543 lb for the equatorial mission. This payload increase is, of course, provided by the removal of the plane-change requirement. The insertion phase for the inclined mission is, naturally, different from the equatorial.

Initial conditions of the 100-n. mi orbit, from reference 2, are given in table 3. The S-IVB was not shutdown when the 100-n. mi. orbit was obtained, but continued thrusting along the inertial velocity vector for about five minutes. This produced the desired apogee of 19 323 n. mi. Shortly before arrival at apogee, the S-IVB--minus boil-off losses--was relit to raise the 100-n. mi. perigee to 4975 n. mi.

Transposition and docking with the LM lab was then performed, and the final circularization maneuver was made with the SPS at apogee. There was 25.6 minutes between S-IVB shutdown and the SPS ignition time. As indicated, the increased payload capability means the LM lab would be used on this mission. Details of the launch and payload breakdown are provided in tables 9 and 7 and figure 23. A trace of the launch and a typical revolution are shown by figure 24. Radar coverage is shown in figure 25.

As in mission 1 the shortest insertion time was the criteria used to position the spacecraft with respect to the earth. As shown before, inclined orbits produce a relative motion in the form of a figure eight along the earth's surface. The name HP, therefore, does not apply; however, the point of nodal crossing does remain fixed, and it is referred to as the HP in the inclined missions. This mission, therefore, has its HP at 52° E, and the motion takes place in a longitude spread of about 7° about this point.

3.5.1 <u>Mission 3 Reentry</u>.- Mission 3 required another deorbit study to investigate the effects of spacecraft position in the orbit on the location of the landing point. Again, a direct descent approach was used. In general, it was found that a deorbit maneuver initiated in the Northern Hemisphere landed in the Southern Hemisphere about 108° E. of the point of maneuver initiation.

Various landing sites can be selected and the associated point of retrofire determined for the mission 3 orbit with figures 26 and 27. Figures 26 and 27 and figure 28 provide the information to determine the landing location from any deorbit point in a circular synchronous orbit inclined 28.5°. All quantities described above are plotted against "central angle", which is just the angular displacement in the orbit measured from the descending node. Tracking station coverage from retrofire to 50 000 ft is presented in figure 29.

The landing point selected for mission 3 is located just off Brisbane, Australia, at 26.13° S and 163.3° W. Because the entry parameters are essentially the same as for mission 1, figures providing this information are not repeated. However, a detailed history of the mission 3 reentry is provided by table 10.

3.6 Mission 4 - Inclined

The inclined mission 4 is basically the same as mission 3 except for the location of the HP. Again, this was accomplished by phasing prior to synchronous orbit insertion because of the fuel savings. The boost phase was identical through S-IVB jettison, but the spacecraft coasted for one orbit before circularizing. During this phasing period, the earth's rotation caused the second apogee to occur over the desired point, and the SPS was used to raise perigee from 4975 n. mi. to 19 323. The stationary or nodal point was 152° W, which is very close to the mission 2 HP. The 71 360 lb payload, which is detailed in table 7, describes a figure-eight with respect to the earth. (See fig. 30.) Additional boost phase details are available in table 11 and figure 31. Radar coverage is provided in figure 32. As for all the other missions, this mission was terminated after about 14 days. A suggested landing point is 28.3° N and 47.1° W. The required deorbit maneuver would then take place at 28.5° S and 152° W.

4.0 CONCLUSIONS

Using Apollo S-V hardware, a payload of 39 543 lb can be placed in an equatorial, circular, synchronous orbit and a total of 71 360 lb can be placed in a 28.5° inclined, circular, synchronous orbit. Both payloads include a useful allotment for experiments to be conducted during a manned synchronous orbit mission. The size of the first payload, however, precludes a LM lab for equatorial missions.

Useful mission planning information for such synchronous missions is provided in tables and plots. Included are launch, payload, phasing, tracking, deorbit, and entry requirements.

TABLE I. - APOLLO-SATURN V PROPULSION CHARACTERISTICS

(a) S-IVB

[Thrust = 205 000 lb, flow rate = 480 lb/sec]

Mission	Maneuver	Dry weight, lb	Weight of propellant used, lb
1 and 2	Plane change and synchronous apogee	---	179397.
3 and 4	Continue first burn	---	128905.
3 and 4	Raise perigee	---	23093.

(b) Service module

[Thrust = 20 000 lb, flow rate = 6[illegible].5526 lb/sec]

Mission	Maneuver	Dry weight, lb	Weight of propellant used, lb
1	Circularization	11550.	24857.
1	Deorbit		15183.
2	Phasing		18796.
2	Circularization	11550.	6061.
2	Deorbit		15183.
3 and 4	Circularization	11550.	21583.
3 and 4	Deorbit		17197.

TABLE II.- SATURN V ASCENT TRAJECTORY TO 100-N.MI. CIRCULAR ORBIT FOR EQUATORIAL MISSIONS 1 AND 2

Event	Time, sec	Latitude geocentric, deg	Longitude, deg	Altitude, ft	Velocity fps		Flight-path angle, deg		Azimuth relative geocentric, deg	Load factor, g
					Relative	Inertial	Relative	Inertial		
Liftoff	0.0	28.49	279.36	0	0	1 340	-	0.00	-	1.191
Inboard engine cutoff	155.1	28.48	280.14	187 959	7 259	8 517	23.40	19.78	90.79	4.537
S-IC burnout	159.1	28.48	280.22	199 586	7 627	8 890	22.60	19.25	90.84	3.317
S-II ignition	162.9	28.48	280.30	210 519	7 581	8 851	21.87	18.61	90.89	0.000
Change mixture ratio	165.4	28.48	280.00	217 530	7 608	8 881	21.47	18.27	90.93	0.709
Ullage cases jettison	189.1	28.47	280.89	279 601	7 962	9 263	17.97	15.38	91.28	0.832
LES jettison	194.1	28.47	281.01	291 714	8 046	9 352	17.27	14.81	91.35	0.847
Change mixture ratio	442.7	27.96	289.56	568 746	16 215	17 591	1.12	1.03	96.19	1.708
S-II burnout	535.9	27.36	294.71	592 377	21 127	22 503	0.87	0.82	98.82	2.041
S-IV B ignition	541.4	27.31	295.06	594 034	21 125	22 501	0.77	0.72	99.00	0.000
100-n.mi. circular orbit	693.4	25.45	305.20	607 612	24 205	25 582	0.000	0.00	104.02	0.705

TABLE III. - SATURN V ASCENT TO 100-N MI CIRCULAR ORBIT FOR INCLINED MISSIONS 3 AND 4

Event	Time, sec	Geodetic latitude, deg	Longitude, deg	Velocity[a], fps	Altitude, ft	Flight-path angle[a], deg	Azimuth[b], deg
Lift-off	0.0	28.65	-80.64	1340	0	0.0	---
End vertical rise, begin zero-lift flight	12.00	28.65	-80.64	1343	512	3.84	90.0
Shutdown of S-IC inboard engine	154.57	28.64	-79.85	8648	191838	20.10	90.7
Shutdown of S-IC outboard engines, begin coast	158.57	28.64	-79.76	9093	203885	22.94	90.7
Jettison of S-IC, S-II ignition, begin pitch-up maneuver[c]	162.37	28.64	-79.68	9053	215281	18.99	90.8
End pitch-up maneuver, change mixture ratio	172.37	28.64	-79.45	9188	243846	17.58	91.0
Jettison S-IC/S-II interstage adapter section	188.57	28.63	-79.07	9459	287004	15.67	91.2
Jettison launch escape system	193.57	28.63	-78.95	9550	299613	15.11	91.3
Change mixture ratio	386.28	28.33	-72.77	15302	572189	2.15	94.9
Shutdown of S-II, begin coast	534.19	27.51	-65.19	22446	606342	0.52	98.9
Jettison S-II, S-IVB ignition	538.99	27.47	-64.89	22445	607204	0.44	99.0
End integrated trajectory (100 n. mi. circular parking orbit)	694.13	25.59	-54.69	25582	607293	0.00	104.1

[a]Inertial quantities

[b]Relative quantities

[c]Pitch-up of one-deg/sec for 10 sec approximates optimum trajectory, after which a pitch-down rate of 0.0993 deg/sec is maintained to first S-IVB burnout

TABLE IV.- COMMAND MODULE ENTRY CHARACTERISTICS

CM weight, lb		11800
Reference area, ft^2		129.4
Entry aerodynamics		
M	C_L	C_D
0.	.2368	.9608
.7	.2368	.9608
.9	.2464	1.0733
1.1	.4526	1.2349
1.2	.4318	1.2129
1.35	.4684	1.3367
1.65	.5300	1.3419
2.00	.5520	1.2815
2.40	.4834	1.3086
3.00	.4630	1.2635
4.00	.4289	1.2350
6.0 - 25.	.4169	1.2447

TABLE V. - MISSION 1 - CIRCULAR SYNCHRONOUS EQUATORIAL ORBIT WITH A HOVER POINT AT 101.1° E

Event	Time, day:hr:min:sec	Geodetic latitude, deg	Longitude, deg	Altitude, n. mi.	Inertial velocity, fps	Inertial flight-path angle, deg	Inertial azimuth, deg	Inclin-ation, deg	Weight, lb
Lift-off	0:00:00:00	---	---	---	---	---	---	---	---
100-n. mi. circular orbit	0:00:11:33	25.3	-54.8	100.0	25582	0.0	104.02	28.57	284458
Restart S-IVB near first descending node	0:00:23:13	7.16	-11.4	100.0	25582	0.0	117.8	28.57	283815
S-IVB burnout	0:00:29:27	-1.9	13.9	164.0	33291	6.5	96.9	7.1	104418
Circularize with SPS	0:05:37:26	0.0	101.1	19323.	5264	1.07	82.88	7.1	64400
SPS cutoff	0:05:43:57	0.0	101.1	19323.	10088	0.0	90.0	0.0	39543
Start retrofire	13:21:42:33	0.0	101.5	19323.	10088	0.0	90.0	0.0	39543
End retrofire	13:21:46:33	0.0	101.24	19323.	5183	-0.55	90.0	0.0	24390
400 000 ft	14:02:52:08	0.0	-171.5	65.83	33825	-6.40	90.0	0.0	---
50 000 ft	14:02:59:16	0.0	-154.9	8.23	23688	-17.00	90.0	0.0	---

TABLE VI. - GROUND STATIONS USED FOR SYNCHRONOUS ORBIT STUDY[a]

Station Name	Call Letters	Latitude$_g$, deg	Longitude, deg
Cape Kennedy	CNV	28.48	-80.58
Grand Bahama	GBI	26.62	-78.35
Antigua	ANT	17.14	-61.79
Ascension Island	ASC	-7.97	-14.40
Bermuda	BDA	32.35	-64.65
Canary Island	CYI	27.73	-15.60
Pretoria[c]	PRE	-25.94	28.36
Tananarive[b]	TAN	-19.02	47.31
Carnarvon	CRO	-24.90	113.72
Hawaii	HAW	22.12	-159.67
Canton Island[b]	CTN	-2.79	-171.65
Guaymas	GYM	27.96	-110.72
Corpus Christi	TEX	27.66	-97.38
Guam	GUA	13.58	144.93
Canberra	CAN	-35.31	149.00
Johannesburg	JHB	-25.88	27.71
Madrid	MAD	40.42	-3.67
Goldstone	GST	35.39	-116.85
Cherry Point[c]	CPT	34.88	-76.88

[a]All stations are unified S-band stations except Pre, Tan., Can., CPT, and JHB.

[b]A & Tel - Acquisition & Telemetry.

[c]T - Tracking.

TABLE VII.- SYNCHRONOUS ORBIT WEIGHT DISTRIBUTIONS

Missions 1 and 2:	
Command module, lb	11800.
Command module experiments, lb	200.
Service module, lb	11550.
Service module experiments, lb	693.
Deorbit propellant, lb	15183.
Reserve fuel, lb	117.
Total weight in orbit, lb	39543
Missions 3 and 4:	
Command module, lb	11800.
Command module experiments, lb	500.
Service module, lb	11550.
Service module experiments, lb	3500.
Lunar module laboratory, lb	11249.
Lunar module laboratory experiments and/or maneuver propellant, lb	15336.
Deorbit propellant, lb	17197.
Reserve fuel	228.
Total weight in orbit, lb	71360.

TABLE VIII. - MISSION 2 - CIRCULAR SYNCHRONOUS EQUATORIAL ORBIT WITH A HOVER POINT AT 151.3° W

Event	Time, day:hr:min:sec	Geodetic latitude, deg	Longitude, deg	Altitude, n. mi.	Inertial velocity, fps	Inertial flight-path angle, deg	Inertial azimuth, deg	Inclin-ation, deg	Weight, lb
100-n. mi. circular orbit	0:00:11:33	25.3	-54.8	100.0	25582	0.0	104.02	28.57	284458
Restart S-IVB near first des-cending node	0:00:23:13	7.16	-11.4	100.0	25582	0.0	117.8	28.57	283815
S-IVB burnout	0:00:29:27	-1.9	13.96	164.0	33291	6.5	96.9	7.1	104418
Raise perigee for phasing orbit	0:05:37:26	0.0	101.1	19323.	5264	1.07	82.88	7.1	64400
End first SPS	0:05:42:22	0.0	100.7	19323.	8676	0.055	90.	0.	45604
Circularize with SPS	0:22:27:45	0.0	-151.3	19323.	8676	0.06	90.	0.	45604
End of second SPS	0:22:29:20	0.0	-151.3	19323.	10088	0.0	90.	0.	39543
Start retrofire	13:18:30:00	0.0	-151.3	19323.	10088	0.0	90.	0.	39543
End retrofire	13:18:34:00	0.0	-151.3	19323.	5183	-.55	90.	0.	24360
400 000 ft	13:23:39:35	0.0	-64.8	65.83	33825	-6.40	90.	0.	---
50 000 ft	13:23:46:43	0.0	-45.2	5.23	23688	-17.00	90.	0.	---

TABLE IX. - MISSION 3 - INCLINED CIRCULAR SYNCHRONOUS ORBIT WITH THE NODAL POINT AT 52° E

Event	Time, day:hr:min:sec	Geodetic latitude, deg	Longitude, deg	Altitude, n. mi.	Inertial velocity, fps	Inertial flight-path angle, deg	Inertial azimuth, deg	Weight, lb
Continue S-IVB burn	0:00:11:34	25.25	-54.7	100.	25582.	0.0	104.02	289930.
S-IVB cutoff	0:00:16:03	18.70	-34.5	136.2	33446.	4.73	112.05	161025.
Restart S-IVB	0:05:15:16	-23.08	57.50	19240.	5276.	4.64	72.55	156913.
Second S-IVB cutoff	0:05:16:04	-23.04	57.43	19244.	7461.	4.44	72.5	133820.
Circularize with SPS	0:05:41:40	-21.54	55.85	19323.	7412.3	.445	70.67	92943.
End SPS	0:05:47:20	-21.13	55.67	19323.	10088.	0.0	70.22	71360.
Next node crossing (hover point)	0:09:01:17	0.0	51.96	19323.	10088.	0.0	61.46	---
Retrofire	13:15:07:52	27.67	53.82	19323.	10088.	0.0	97.9	44775.
End retrofire	13:15:12:22	27.54	53.67	19323.	10088.	0.0	98.35	27578.
400 000 ft	13:20:21:50	-28.68	139.2	65.84	33833.	-6.26	89.58	---
50 000 ft	13:20:29:22	-26.13	163.3	---	---	---	---	---

TABLE X.- DESCENT TRAJECTORY FROM CIRCULAR SYNCHRONOUS INCLINED MISSION 3

Time, sec	Geodetic latitude, deg	Longitude, deg	Altitude, n.mi.	Inertial velocity, fps	Inertial flight-path angle, deg	Inertial azimuth, deg
0.	27.67	53.82	19323.	10088.	0.	97.9
270.6	27.54	53.67	19322.	5181.8	-.63	98.4
4365.6	25.88	46.45	18532.	5847.9	-23.36	102.79
8461.6	23.14	40.42	16135.	7746.4	-38.93	107.37
12557.6	17.97	38.30	11793.	11289.0	-46.69	112.64
16653.6	2.23	53.43	4595.	19994.0	-42.78	118.46
18837.6	-28.68	139.21	65.8	33833.	-6.26	89.57
19265.7	-26.83	159.61	8.2	1837.3	-21.67	87.59

TABLE XI. - MISSION 4 - INCLINED CIRCULAR SYNCHRONOUS ORBIT WITH THE NODAL POINT AT 152° W

Event	Time, day:hr:min:sec	Geodetic latitude, deg	Longitude, deg	Altitude, n. mi.	Inertial velocity, fps	Inertial flight-path angle, deg	Inertial azimuth, deg	Weight, lb
100-n. mi. circular orbit	0:00:11:34	25.25	-54.7	100.	25582.	0.0	104.02	289930
S-IVB cutoff	0:00:16:03	18.70	-34.5	136.2	33446.	4.73	112.05	161025
Restart S-IVB	0:05:15:16	-23.08	57.5	19240.	5276.	4.64	72.55	156913.
Second S-IVB cutoff	0:05:16:04	-23.04	57.43	19244.	7461.	4.44	72.5	133820.
Circularize with the SPS	0:19:15:38 0:19:21:18	-21.52 -21.11	-148.17 -148.35	19323. 19323.	7412. 10088.	.445 0.0	70.65 70.20	92943. 71360.
Next node crossing	0:22:35:00	0.0	-152.0	19323.	10088.	0.0	61.46	71360.
Suggested deorbit point	13:22:40:00	-5.0	-159.1	---	---	---	---	---
50 000 ft	14:04:01:30	2.0	-43.6	---	---	---	---	---

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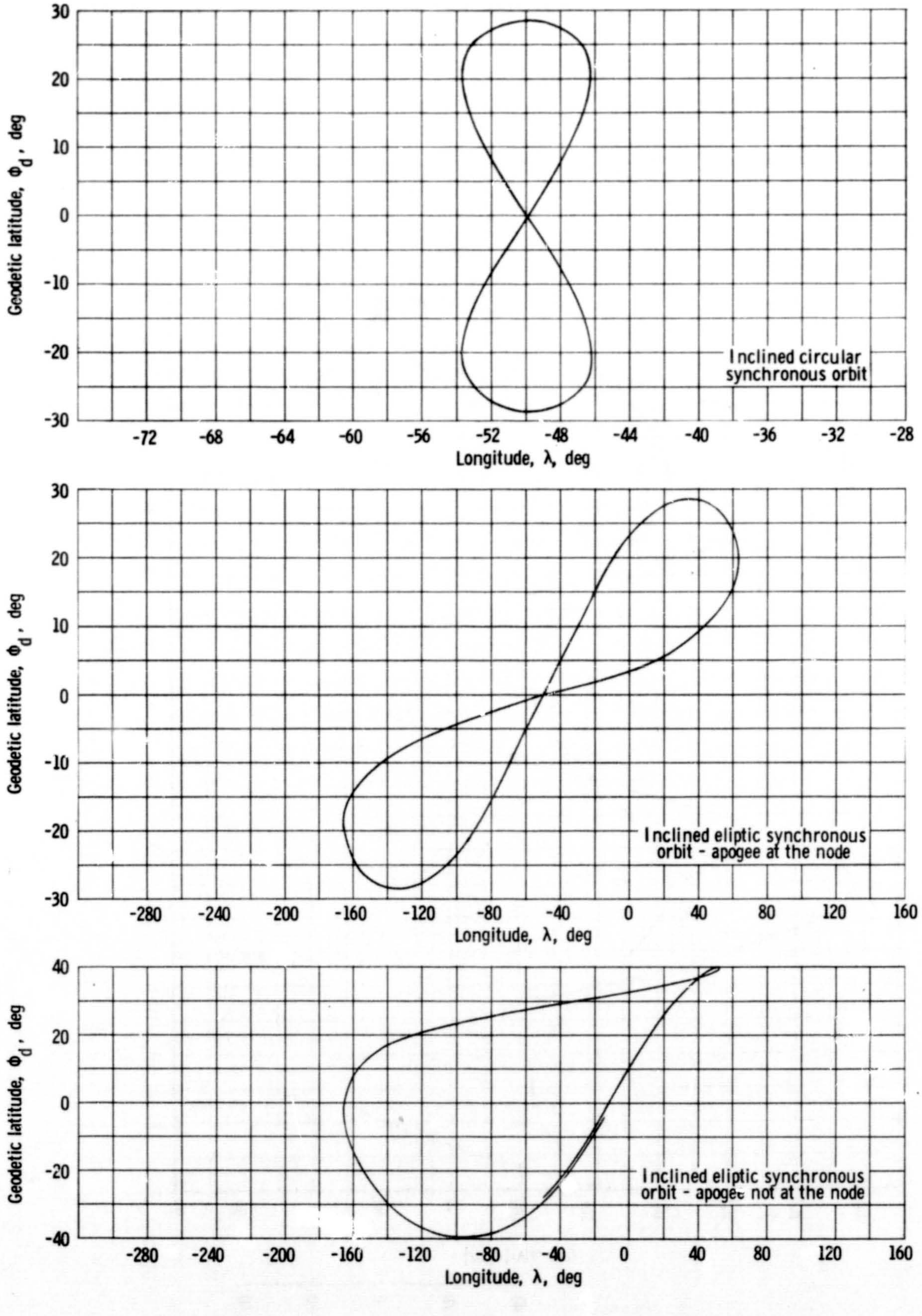


Figure 1. - Typical synchronous orbit groundtracks.

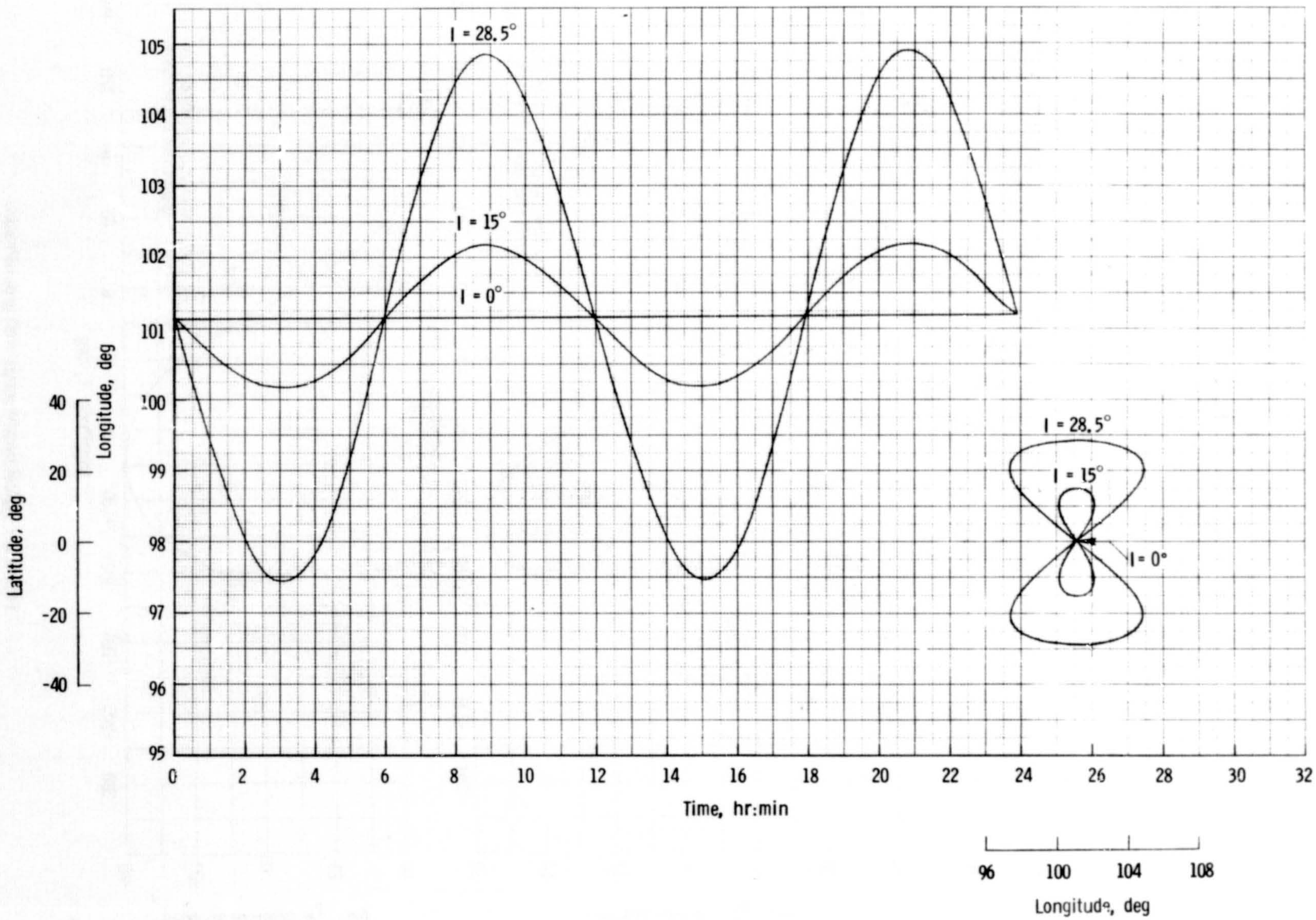


Figure 2. - Effect of inclination on the longitude time history of a circular synchronous orbit.

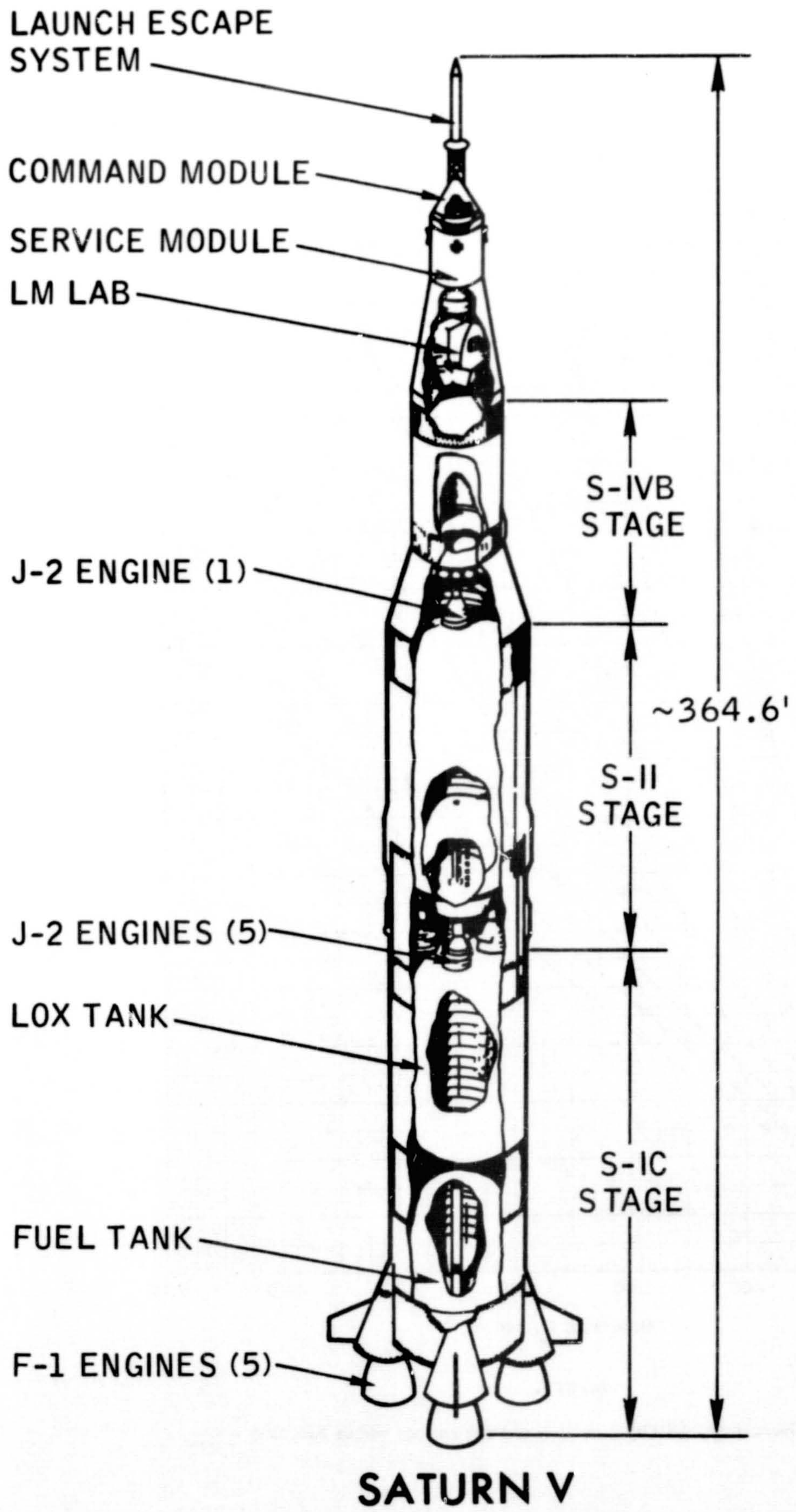
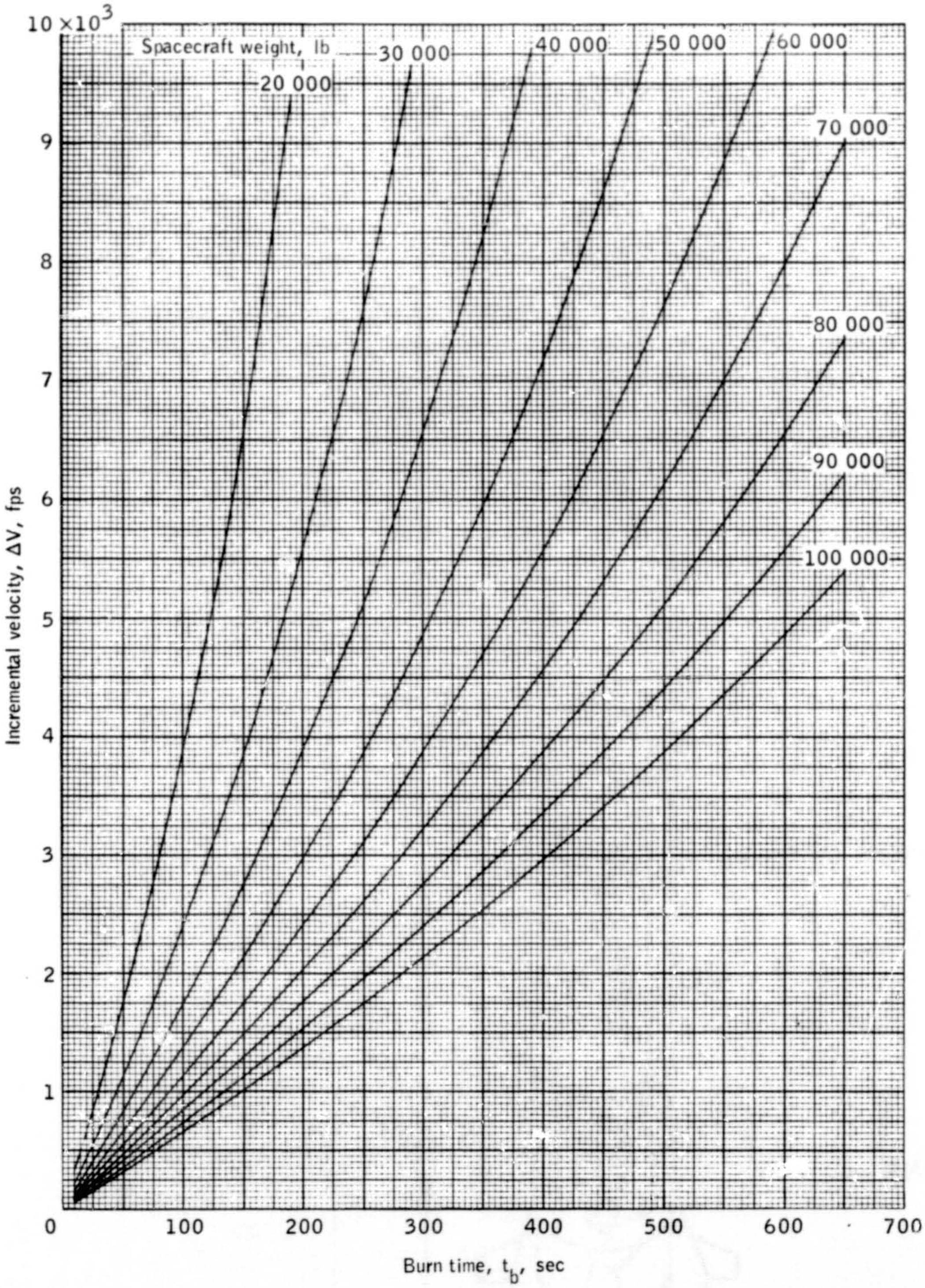
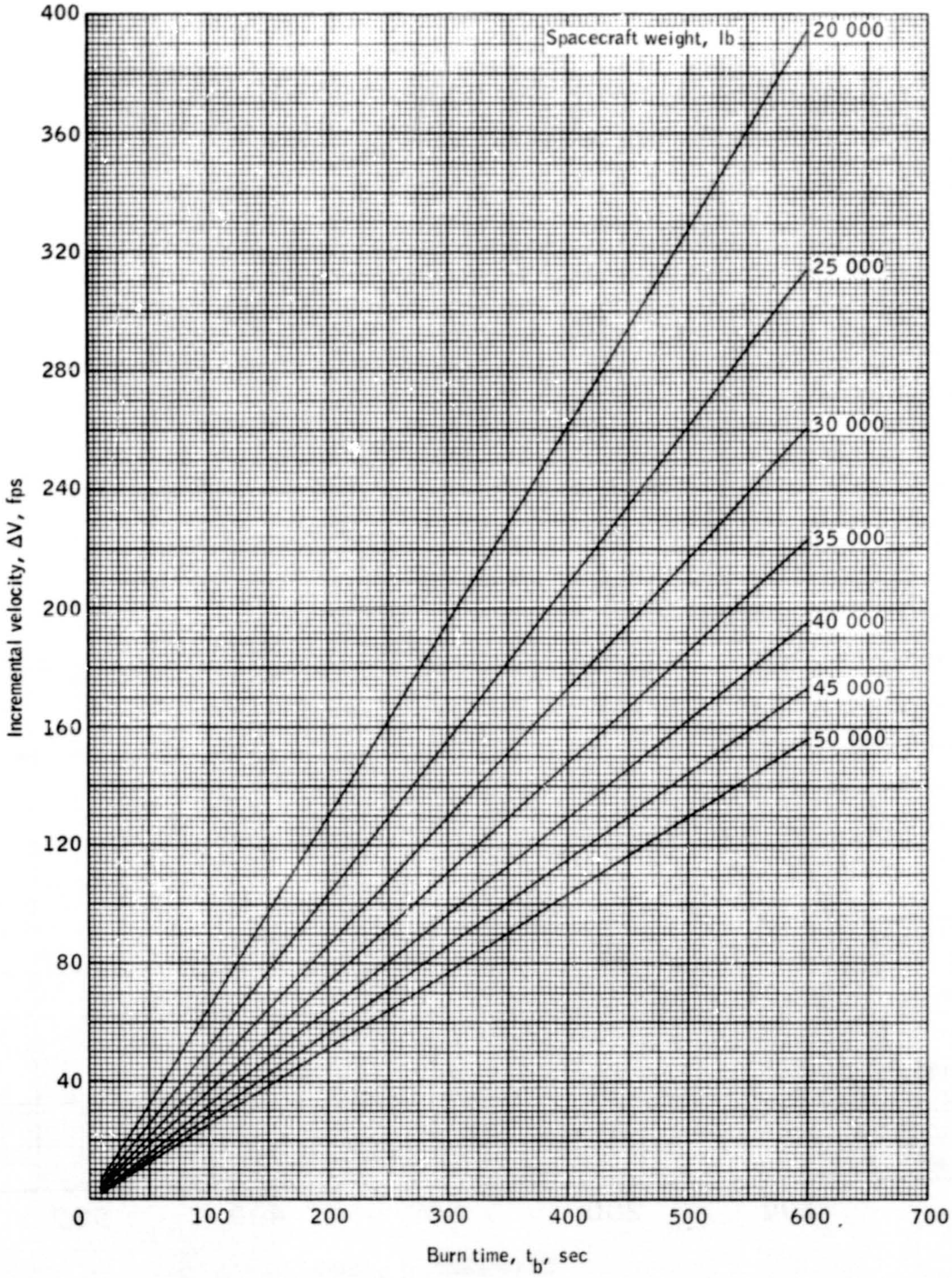


Figure 3. - Launch vehicle to be used for the AAP synchronous orbit missions.



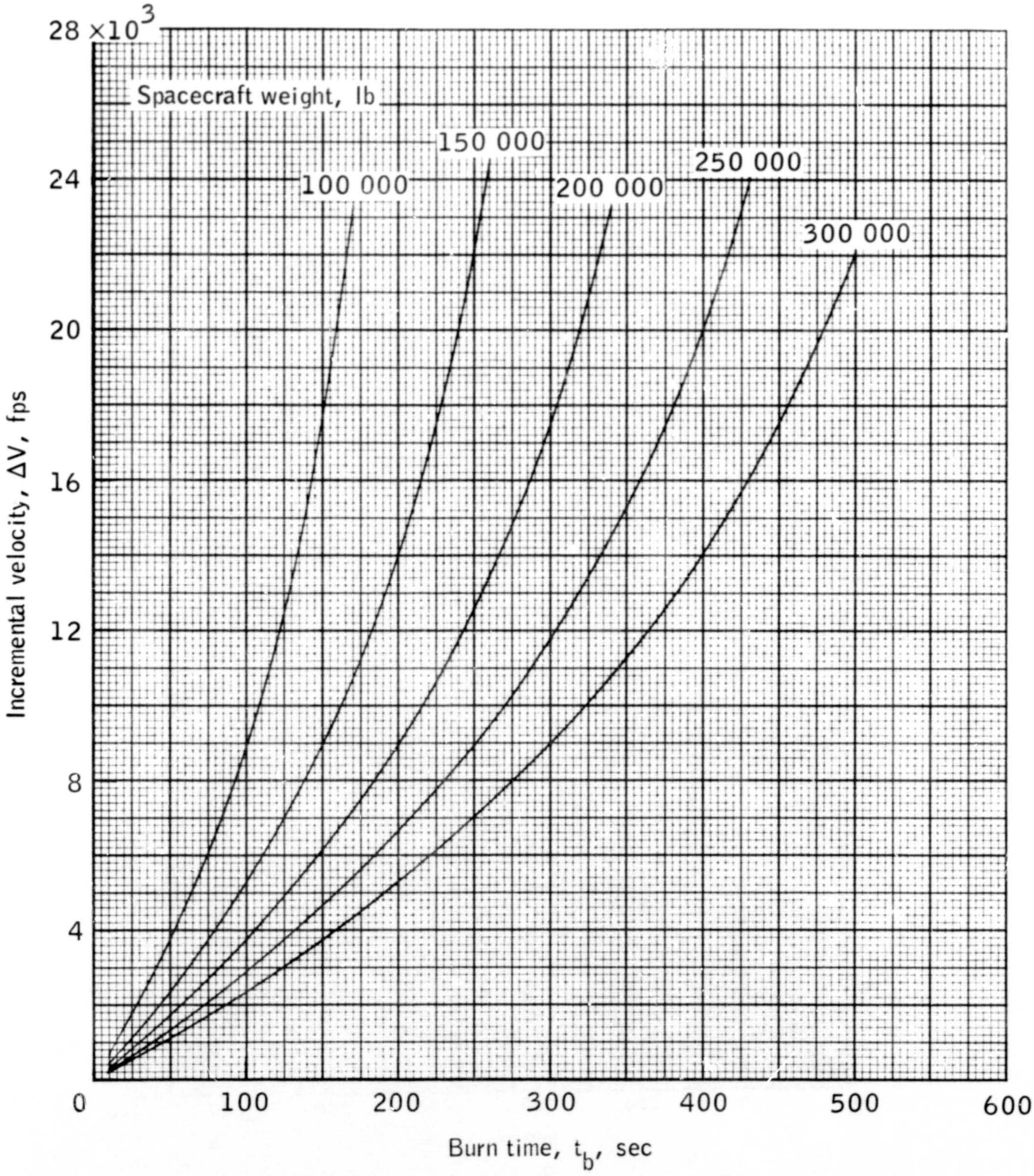
(a) SPS.

Figure 4.- Incremental velocity capability of three engines versus burn time.



(b) RCS.

Figure 4.- Continued.



(c) S-IVB.

Figure 4.- Concluded.

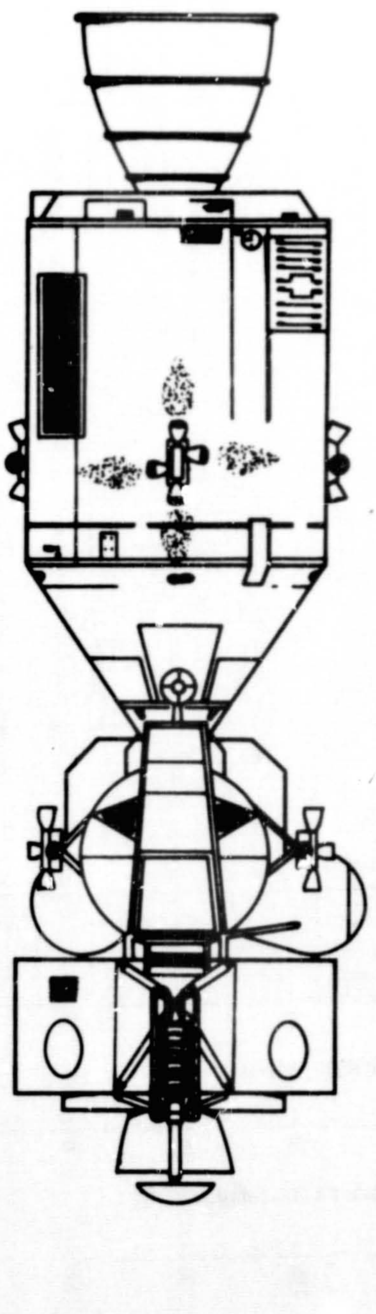
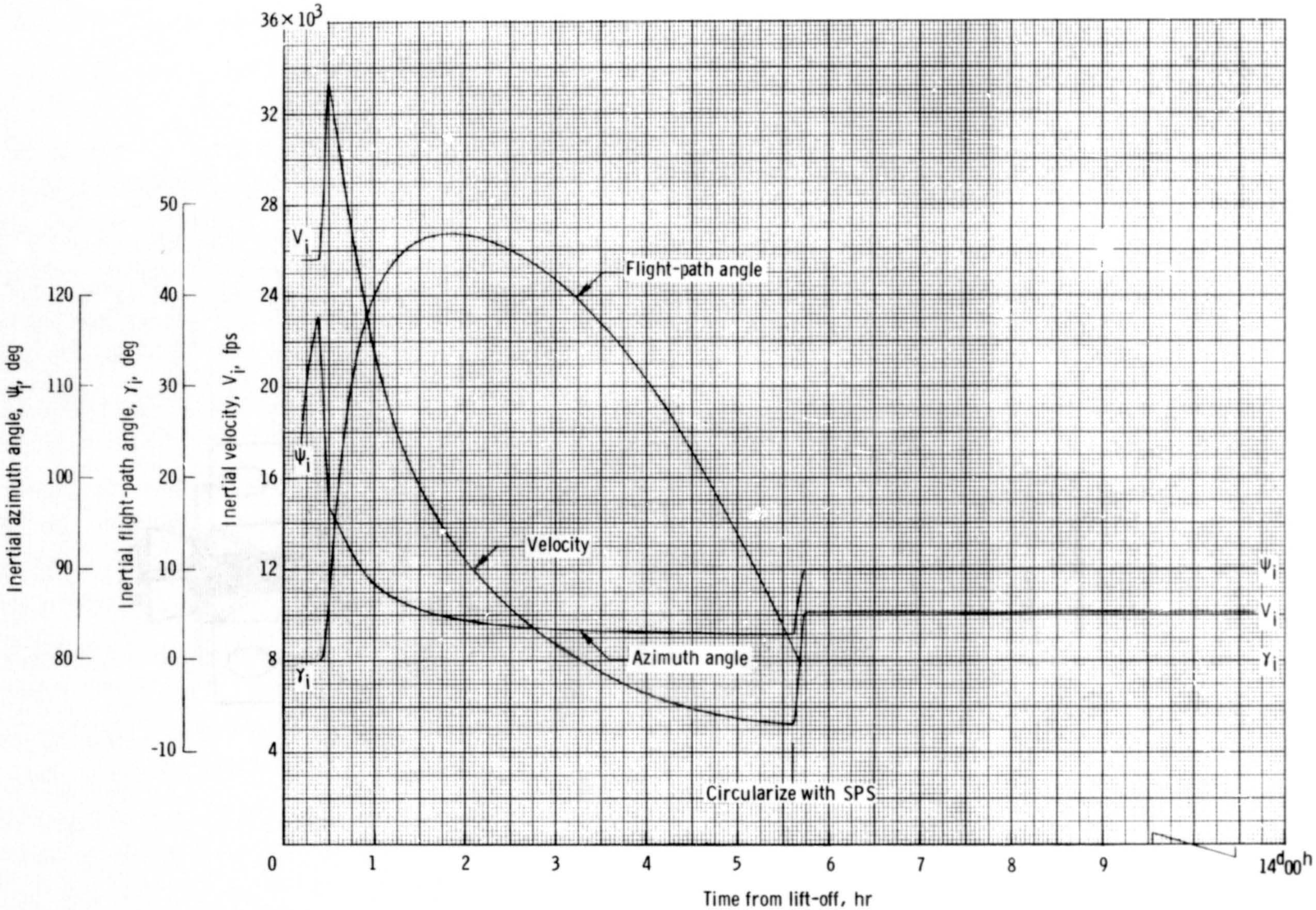
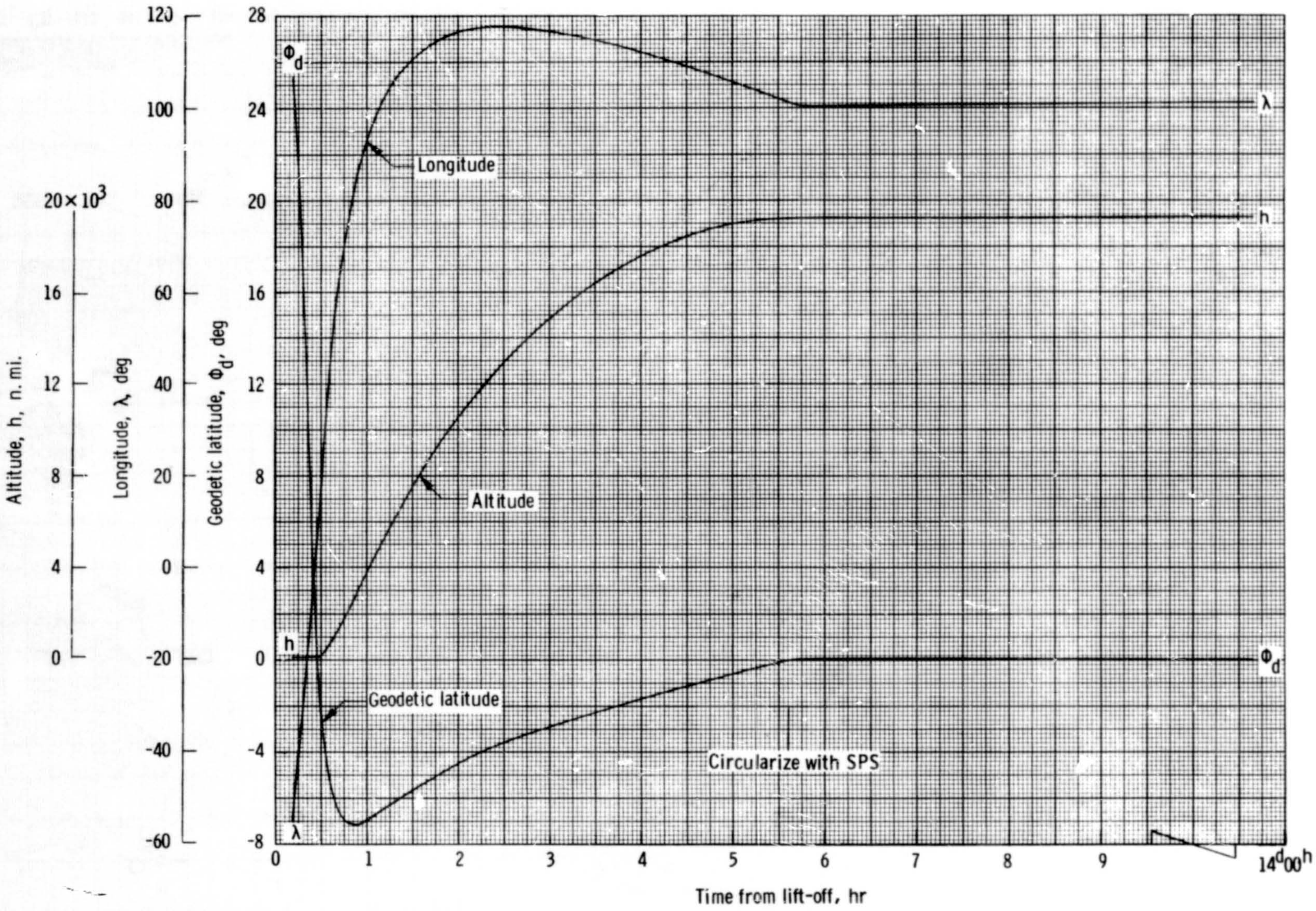


Figure 5.- LM laboratory.



(a) Time history of inertial velocity, inertial flight-path angle and inertial azimuth angle.

Figure 6. - Circular synchronous equatorial mission 1 ascent.



(b) Time history of geodetic latitude, longitude and altitude.

Figure 6. - Concluded.

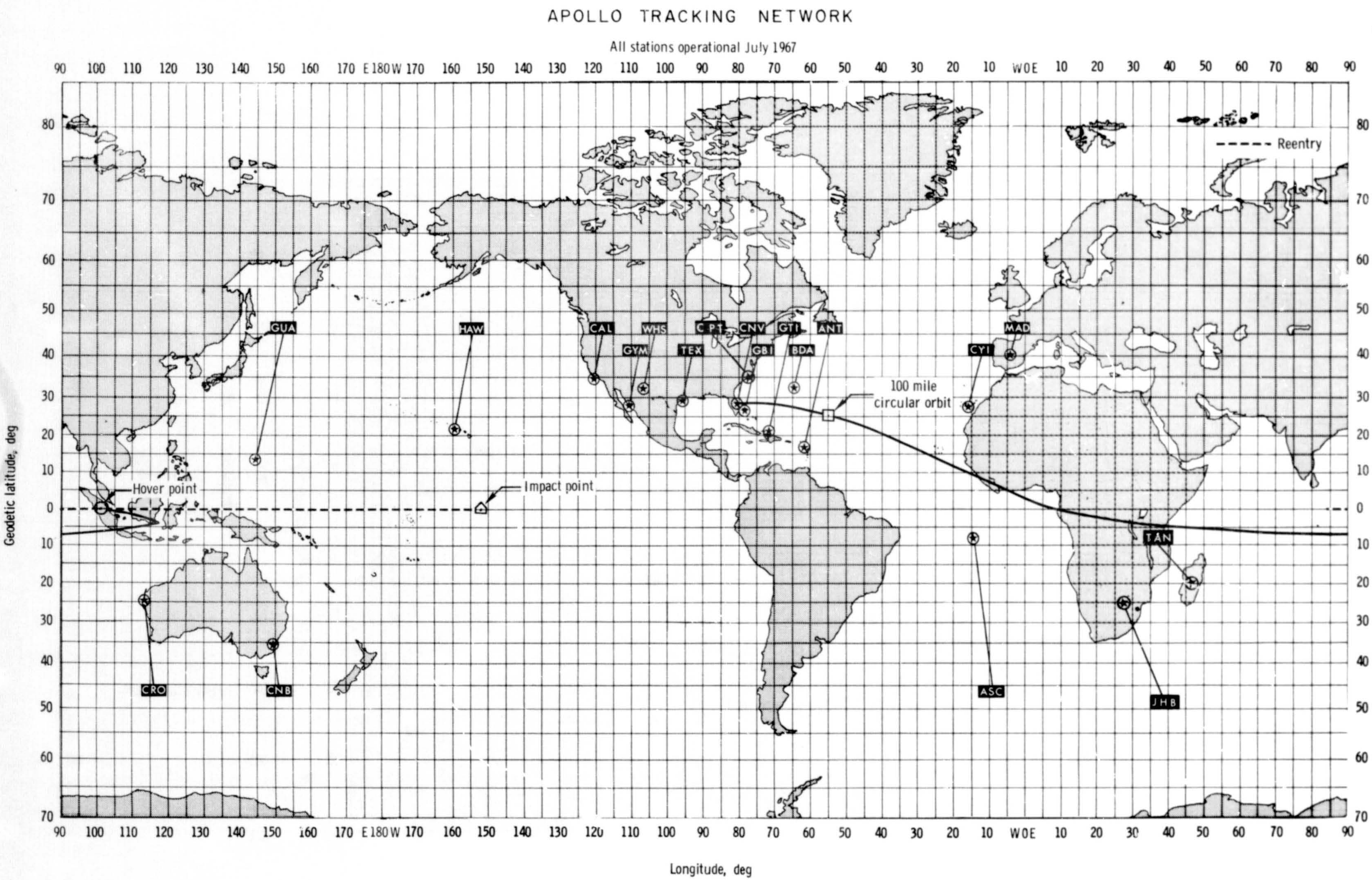


Figure 7. - Groundtrack for circular synchronous equatorial mission 1.

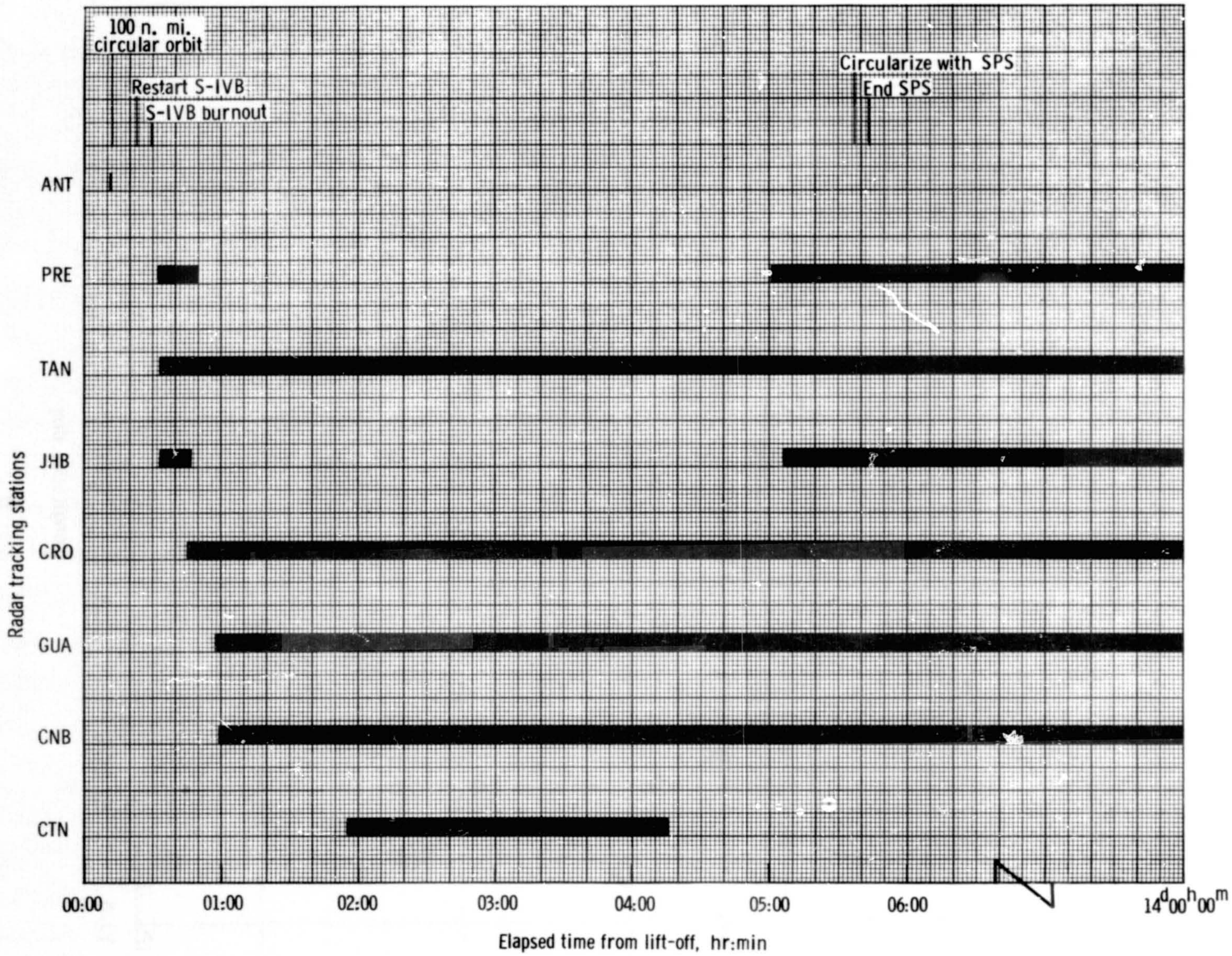


Figure 8. - Tracking station coverage through synchronous orbit insertion - Mission 1.

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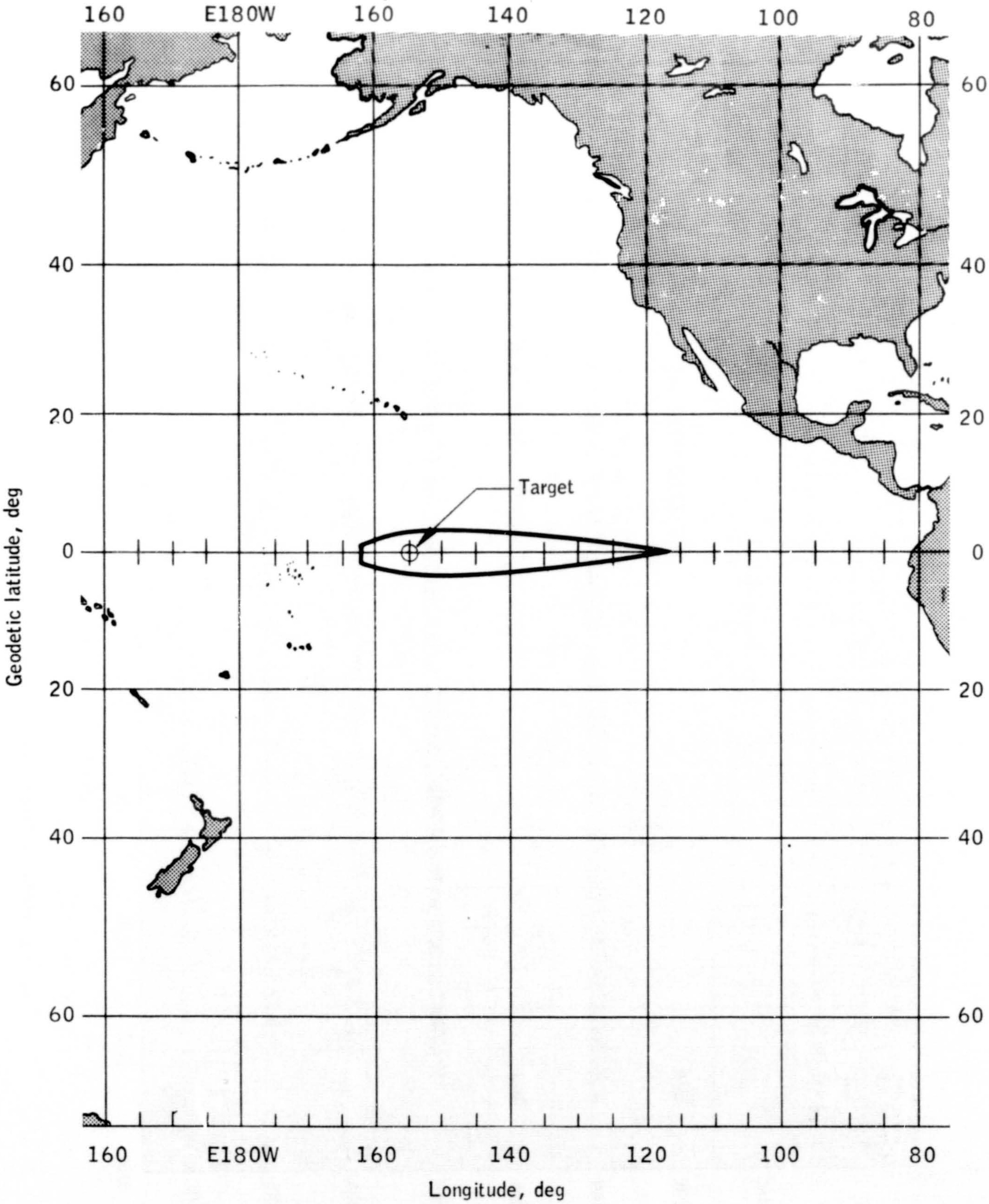
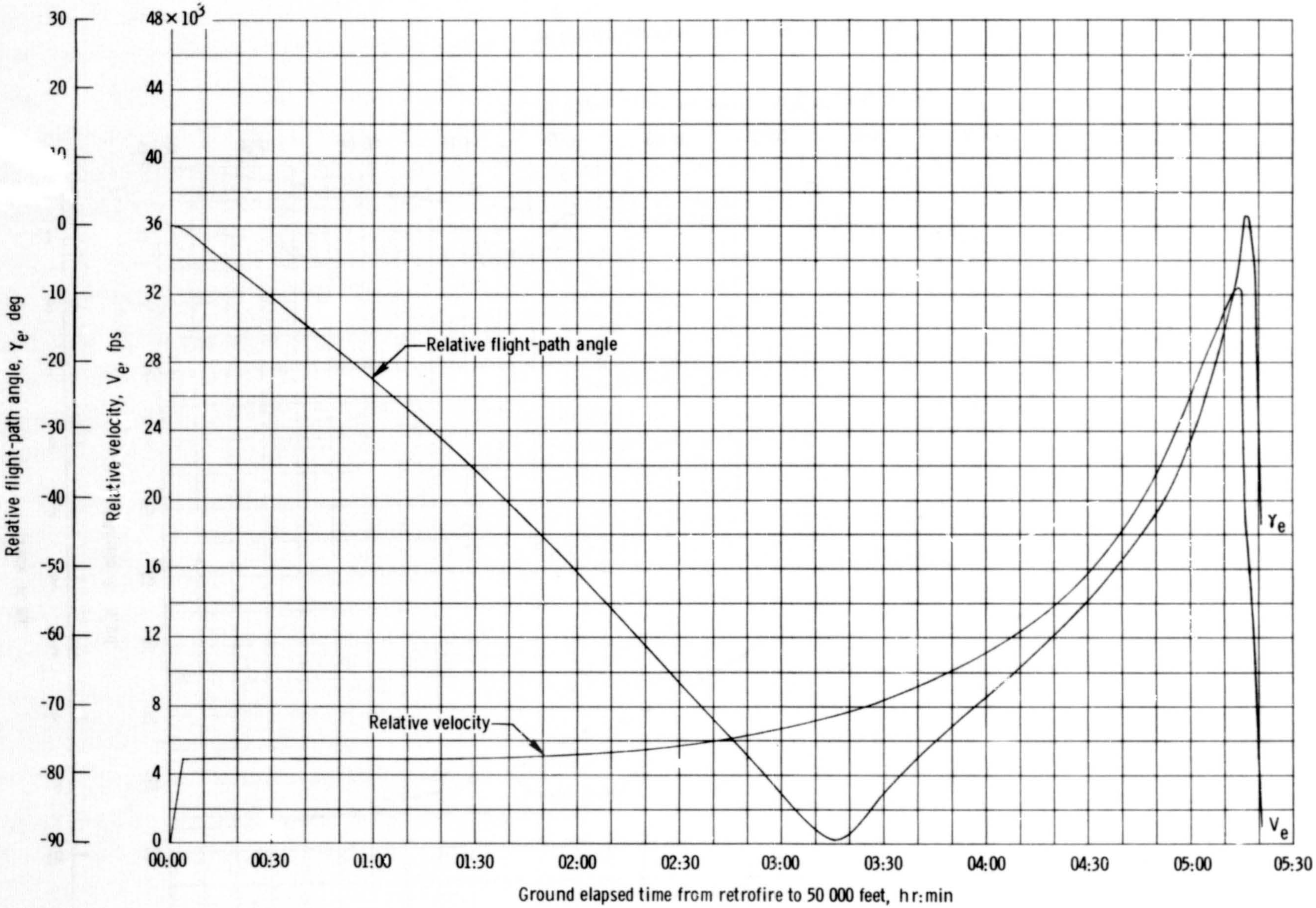
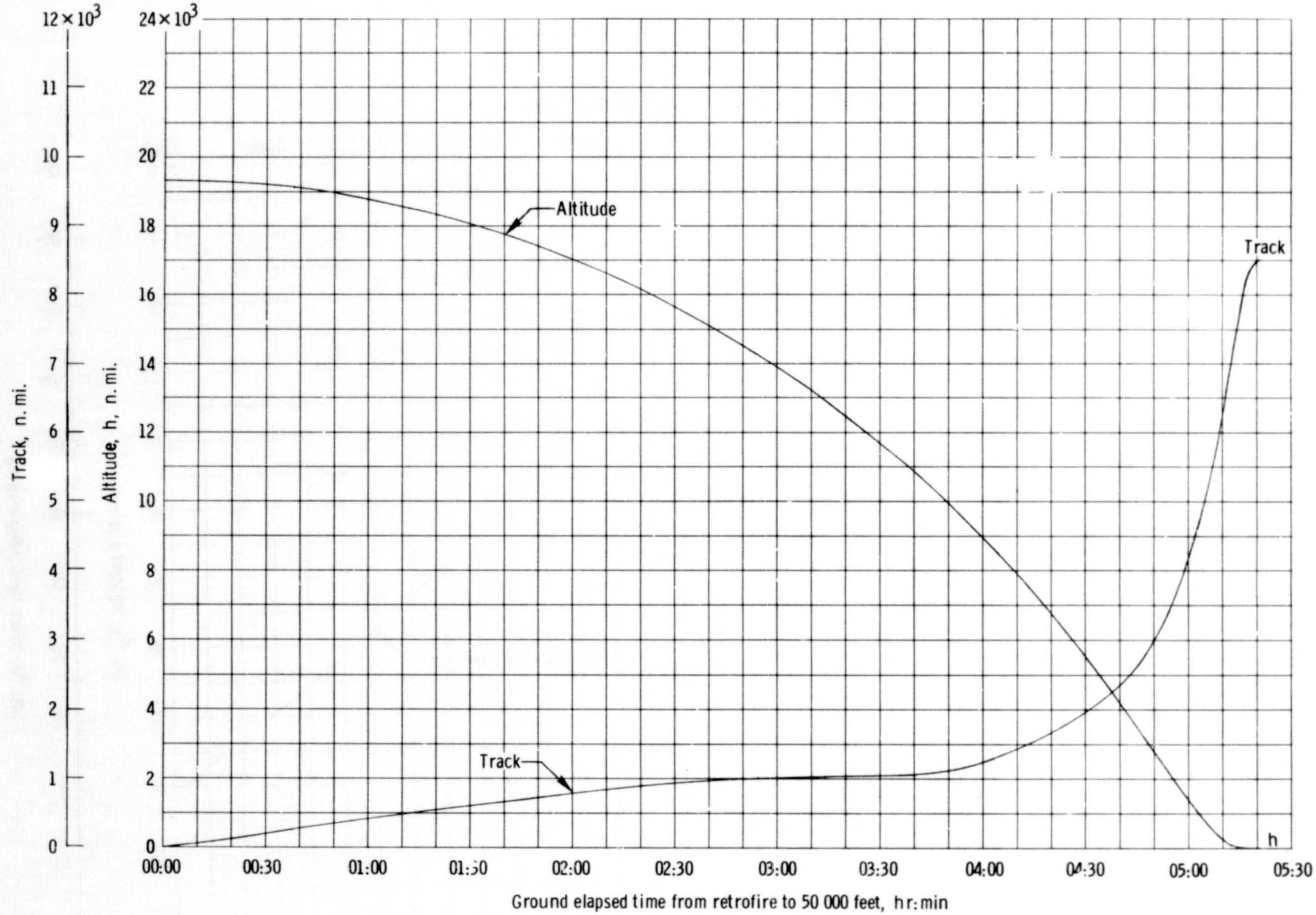


Figure 9.- Entry footprint about the nominal landing point from circular synchronous equatorial mission 1.



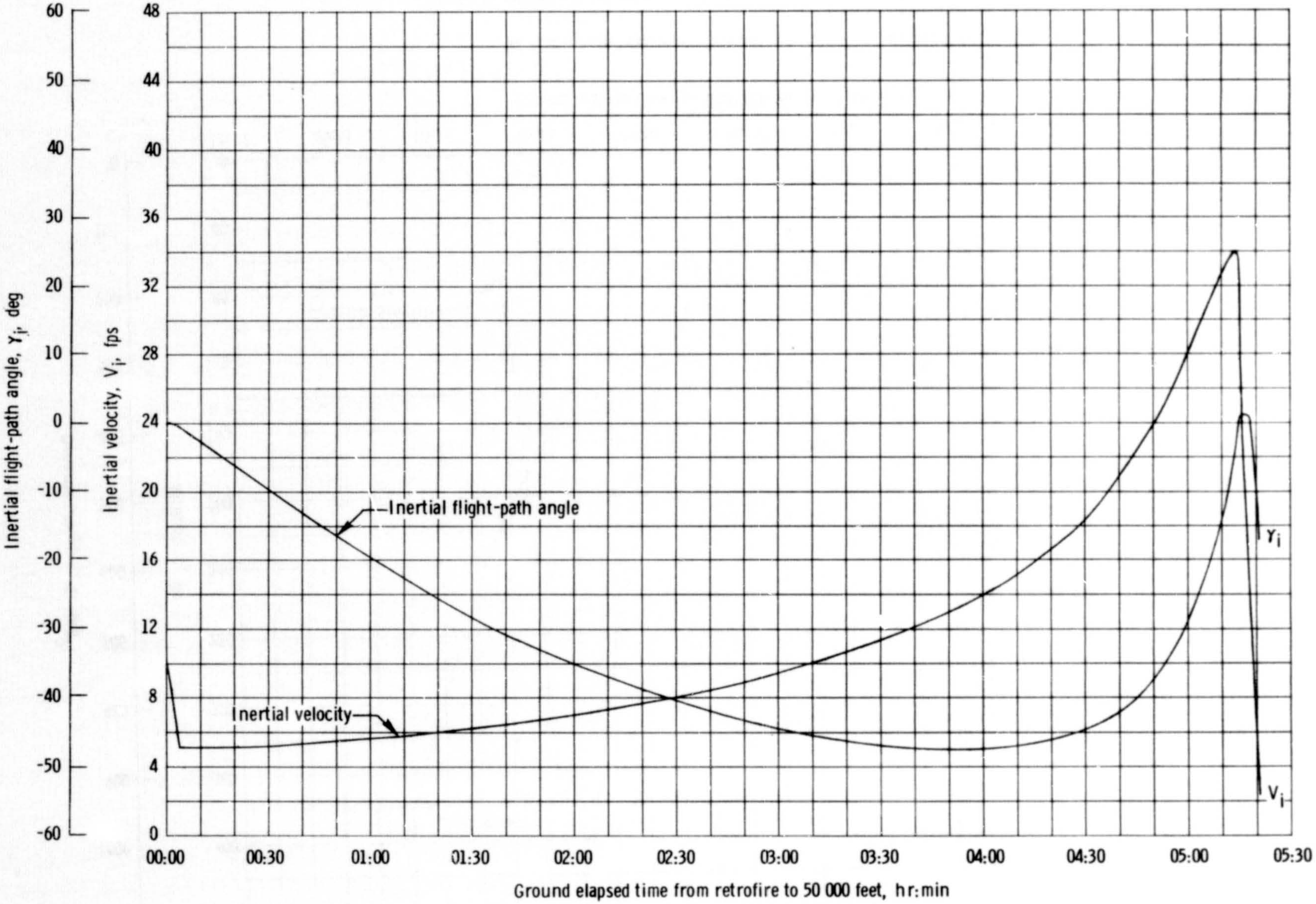
(a) Relative velocity and relative flight-path angle versus ground elapsed time.

Figure 10. - History of typical orbital parameters from deorbit to 50 000 feet - Mission 1.



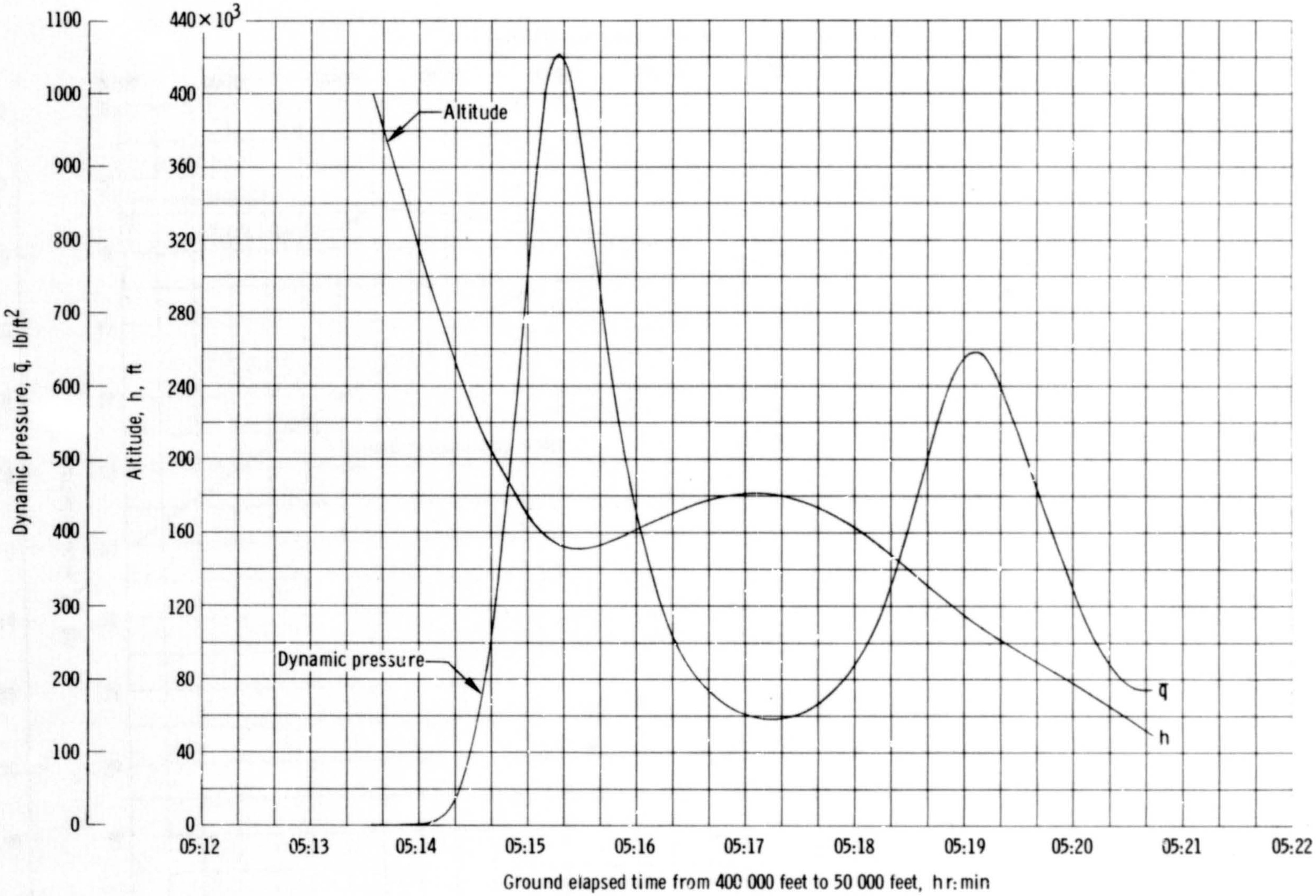
(b) Altitude and track versus ground elapsed time.

Figure 10. - Continued.



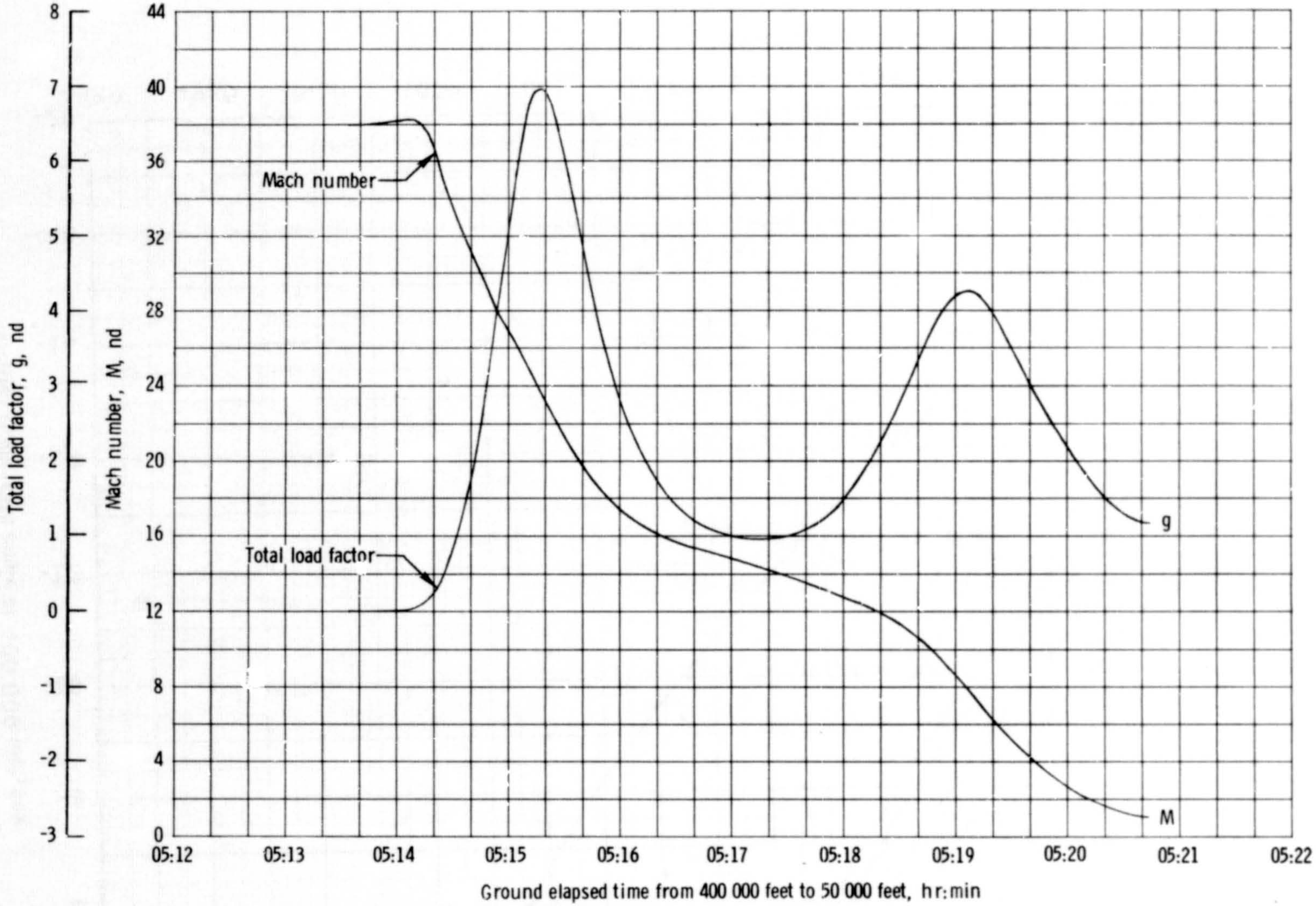
(c) Inertial velocity and inertial flight-path angle versus ground elapsed time.

Figure 10. - Concluded.



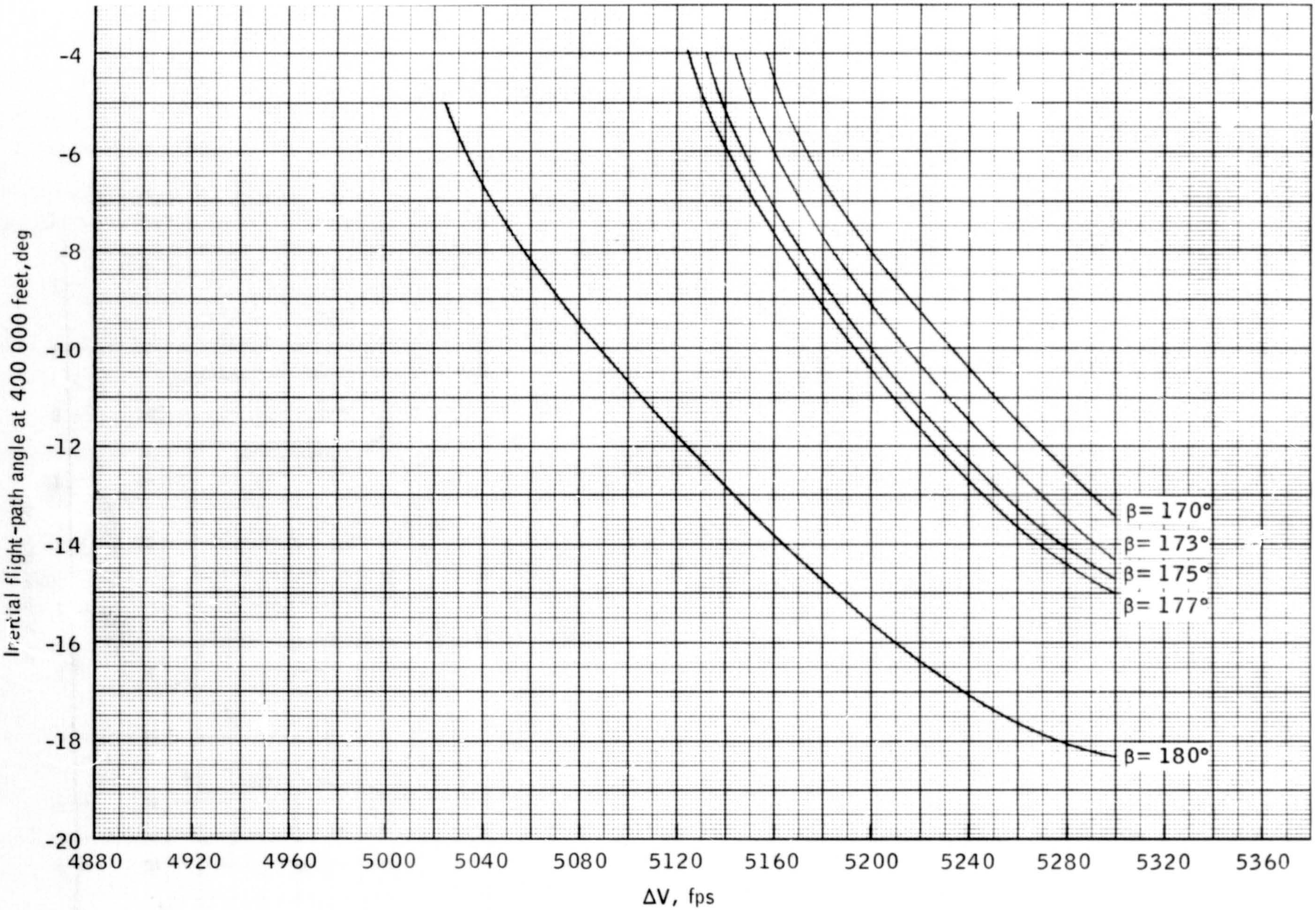
(a) Altitude and dynamic pressure versus ground elapsed time.

Figure 11. -History of typical entry parameters from 400 000 to 50 000 feet - Mission 1.



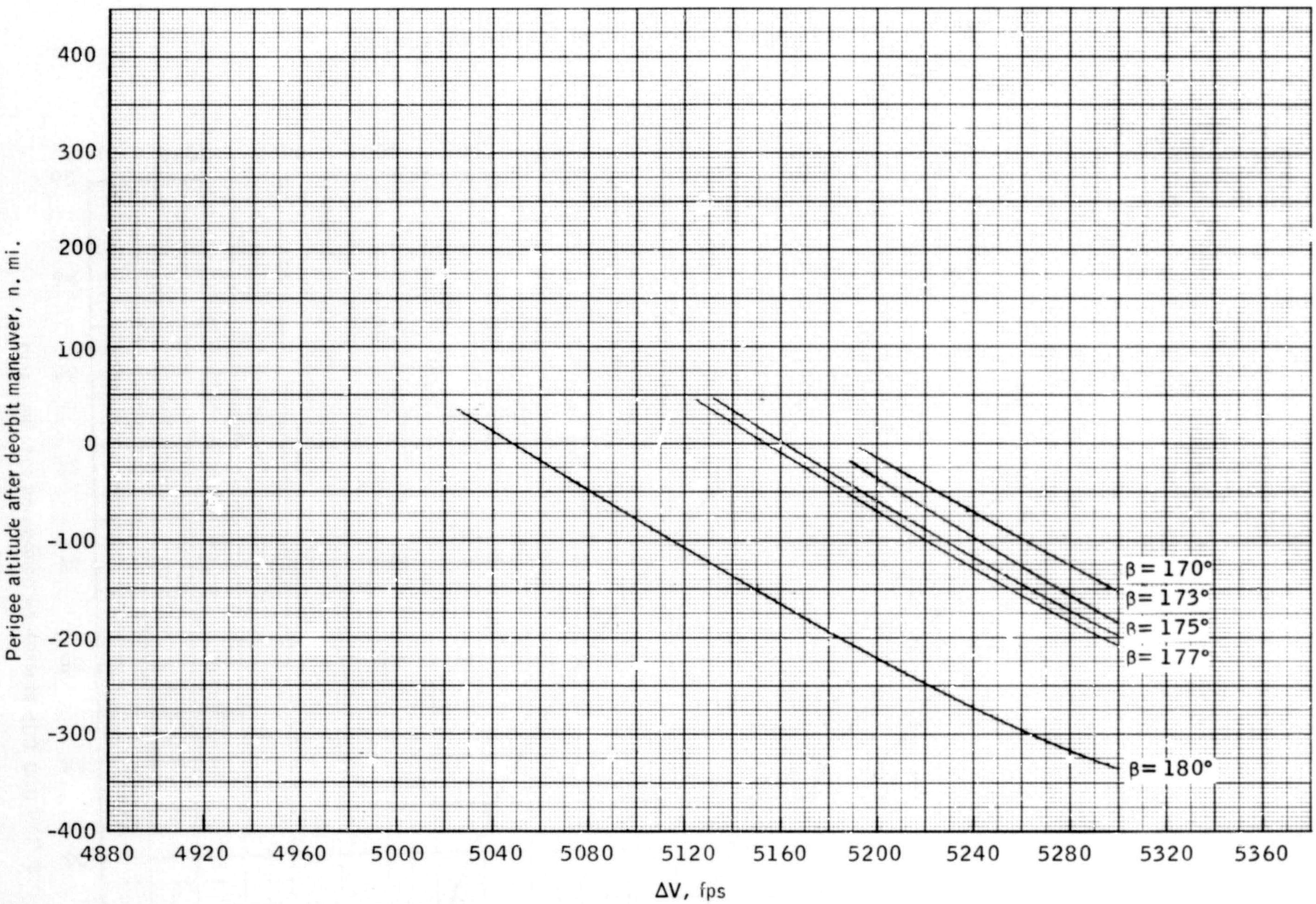
(b) Mach number and load factor versus ground elapsed time.

Figure 11. - Concluded.



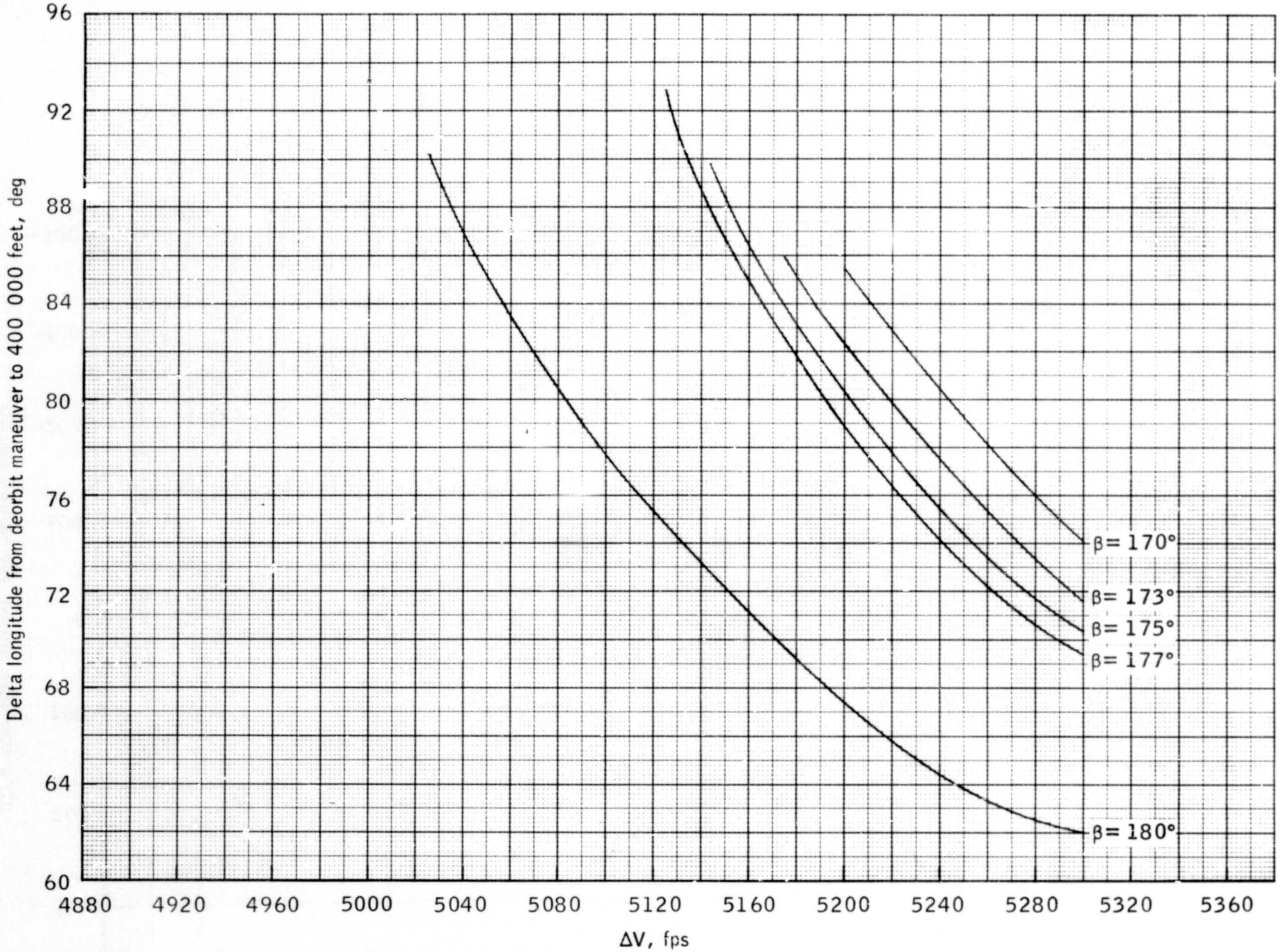
(a) Inertial flight-path angle at 400 000 feet versus ΔV.

Figure 12.- Deorbit requirements for circular sychronous equatorial mission 1.



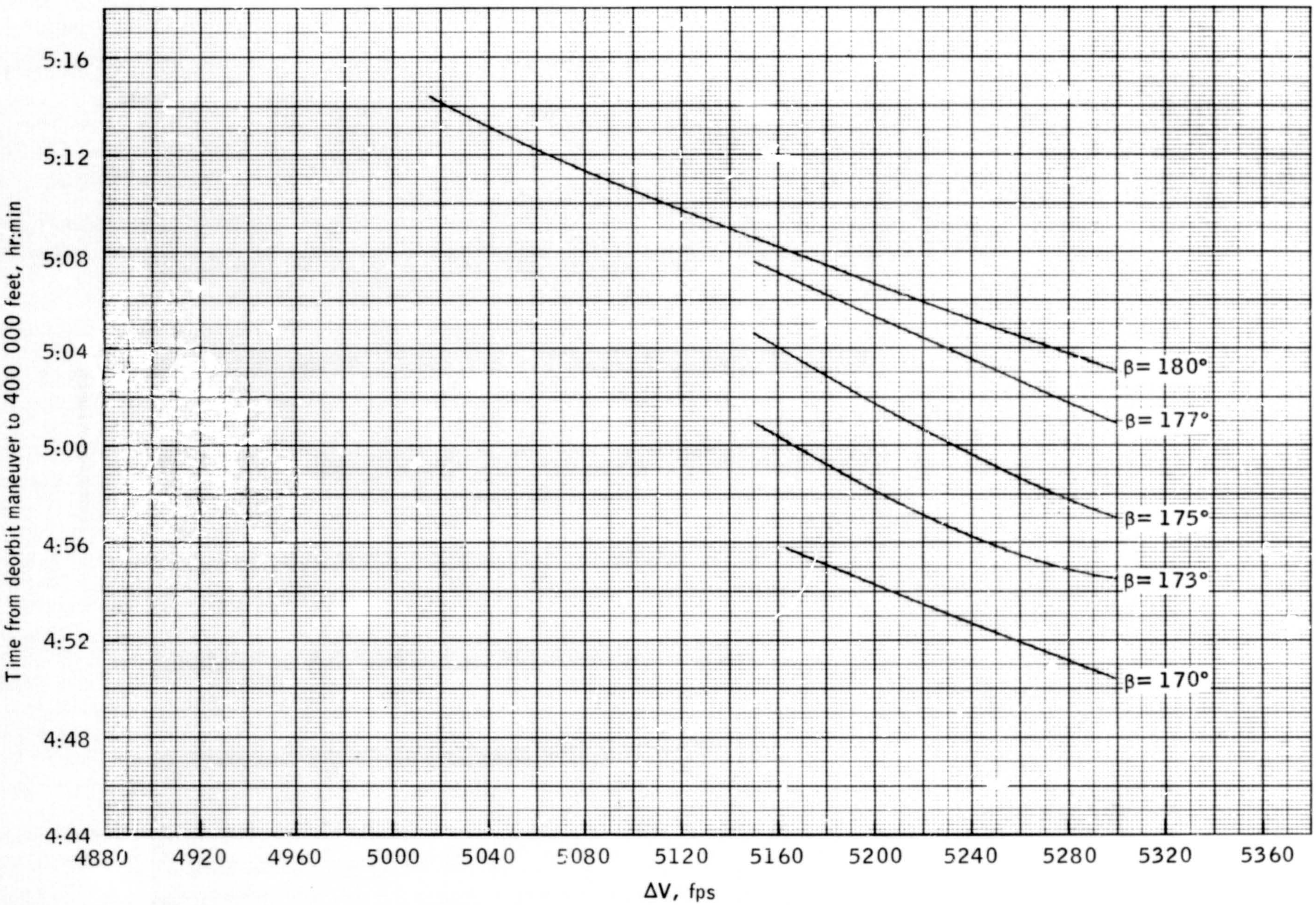
(b) Perigee altitude after deorbit maneuver versus ΔV.

Figure 12.- Continued.



(c) Delta longitude from deorbit maneuver to 400 000 feet versus ΔV.

Figure 12.- Continued.



(d) Time from deorbit maneuver to 400 000 feet versus ΔV.

Figure 12.- Concluded.

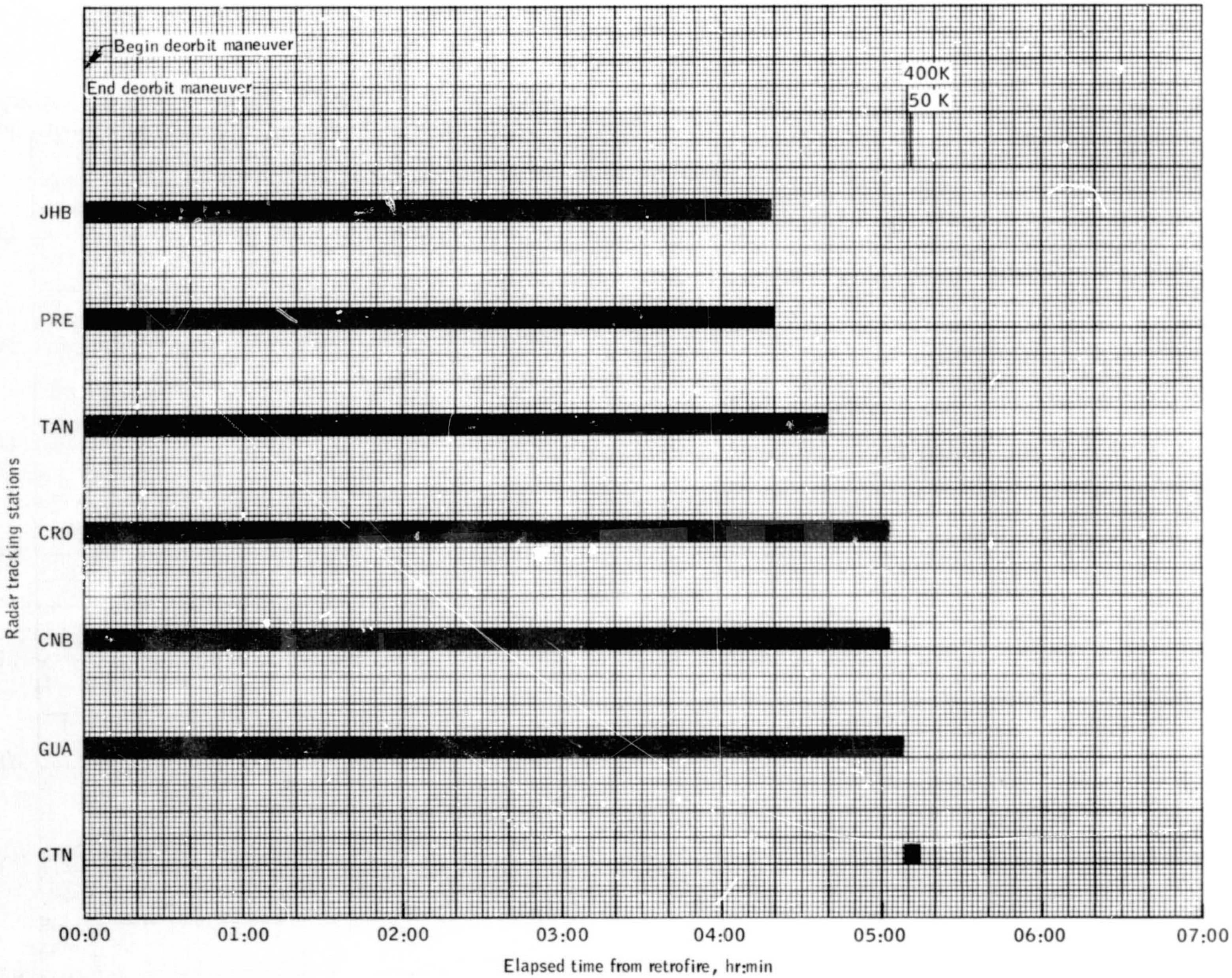


Figure 13.- Tracking station coverage from retrofire to 50 000 feet - Mission 1.

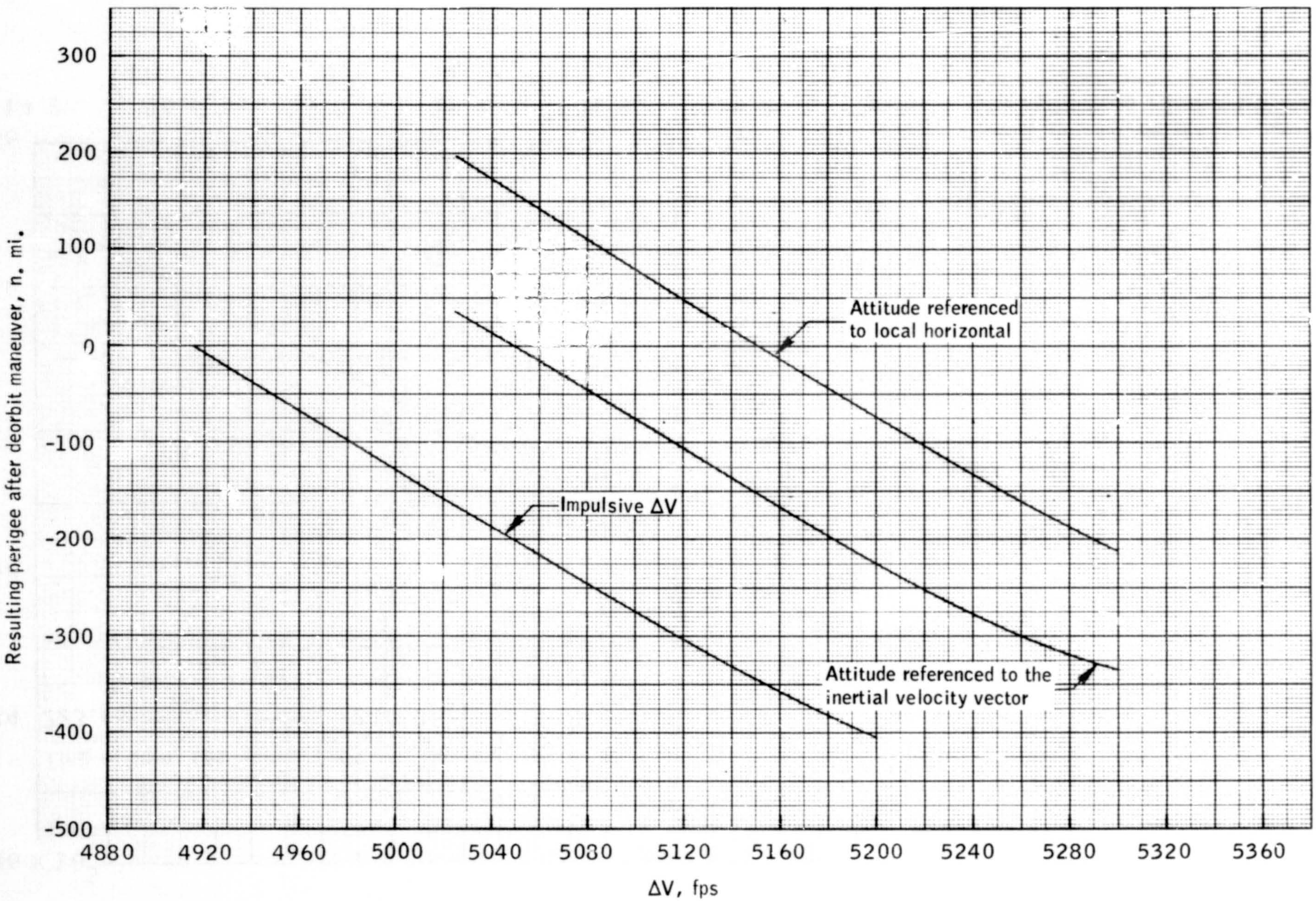


Figure 14.- Deorbit requirements for various attitude references.

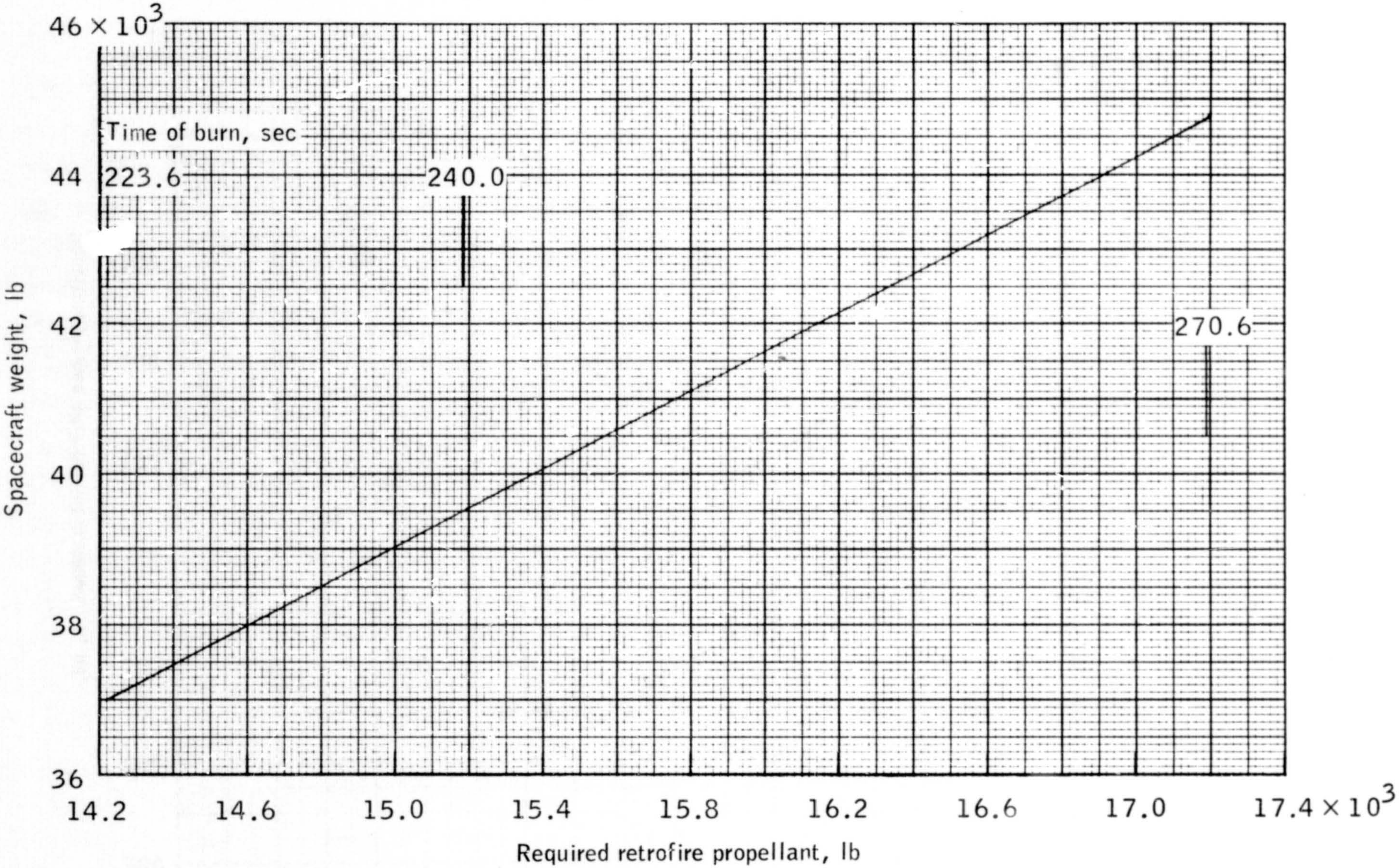


Figure 15.- Deorbit propellent required to hit the reentry corridor for various spacecraft weights.

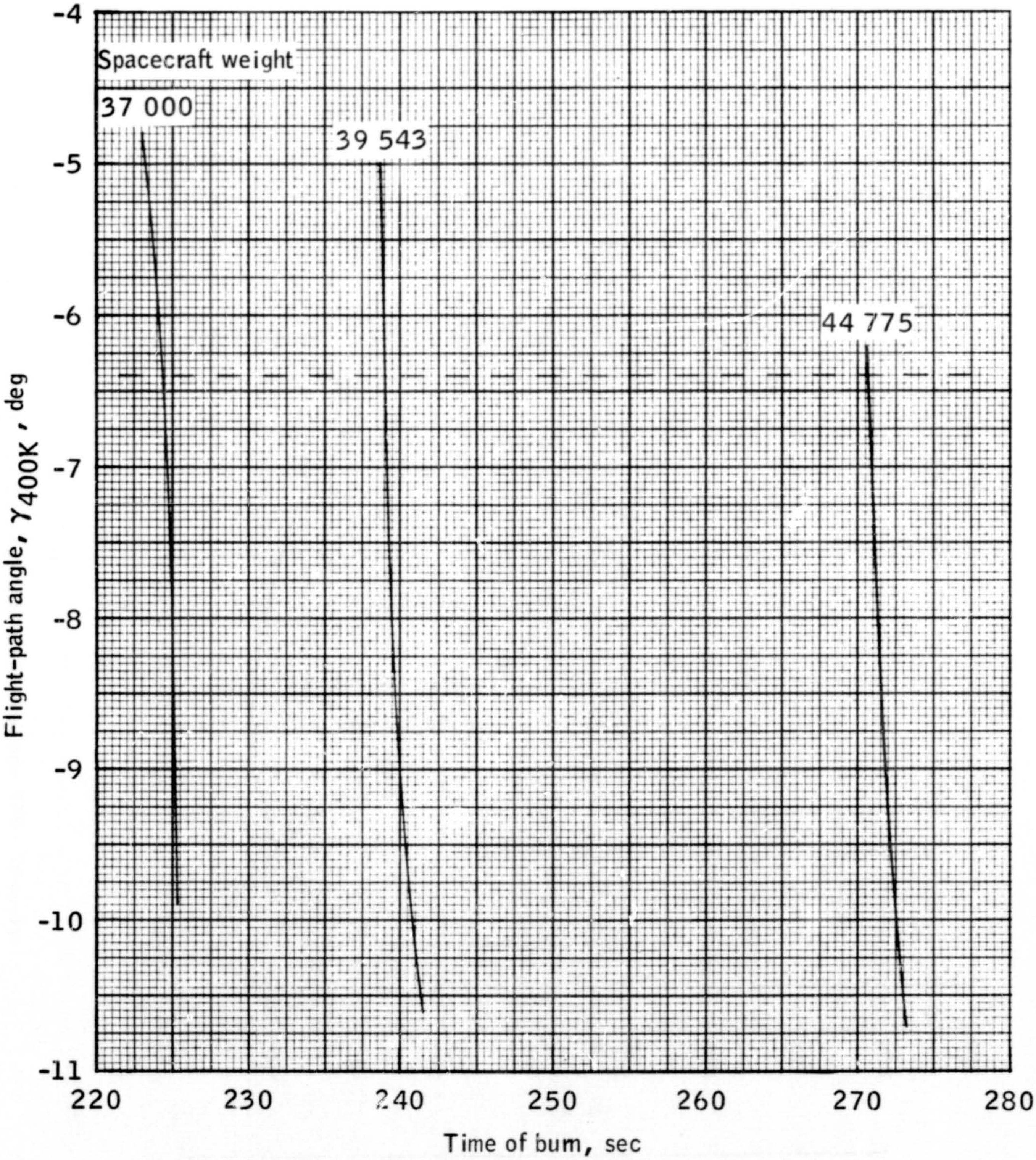
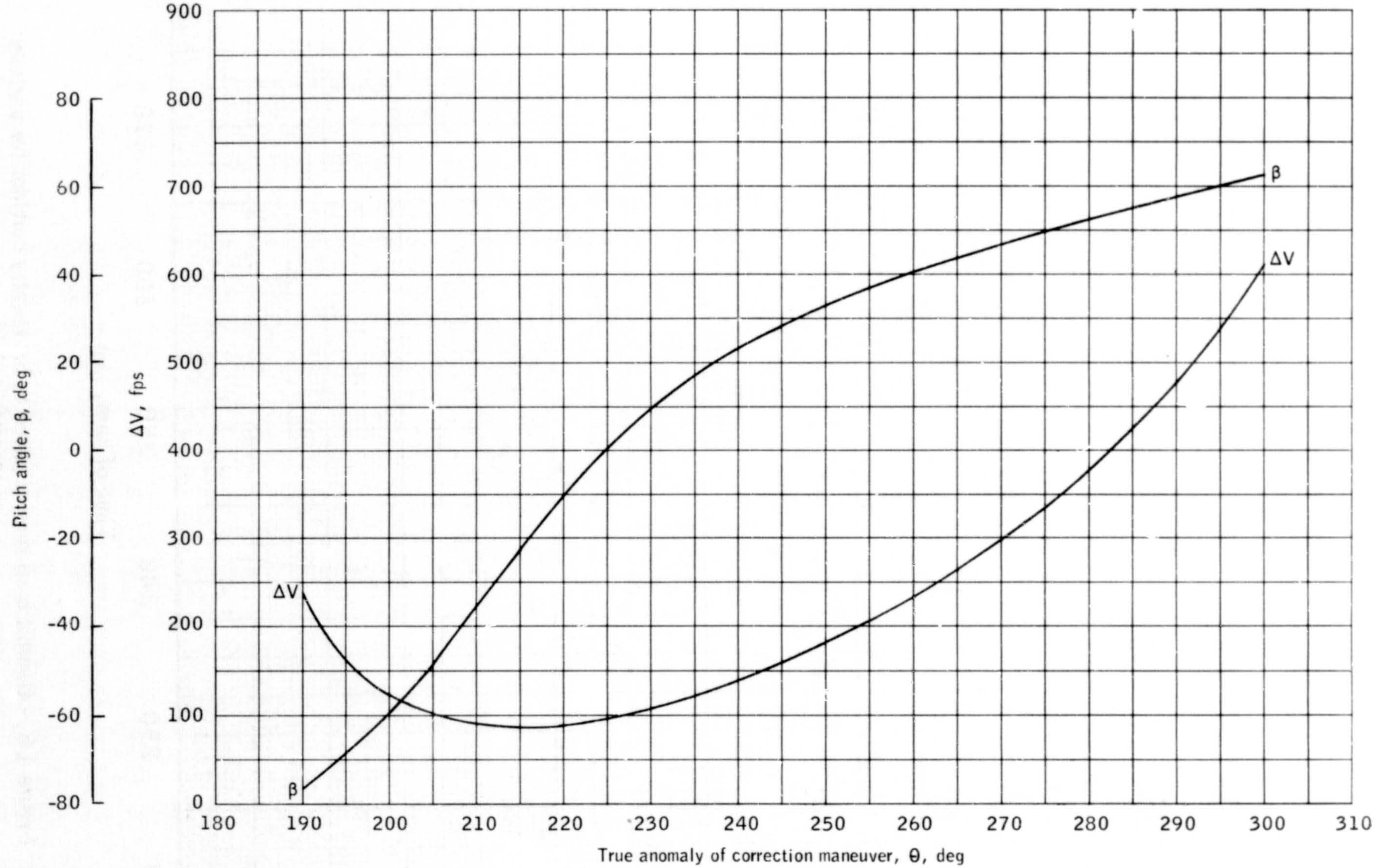
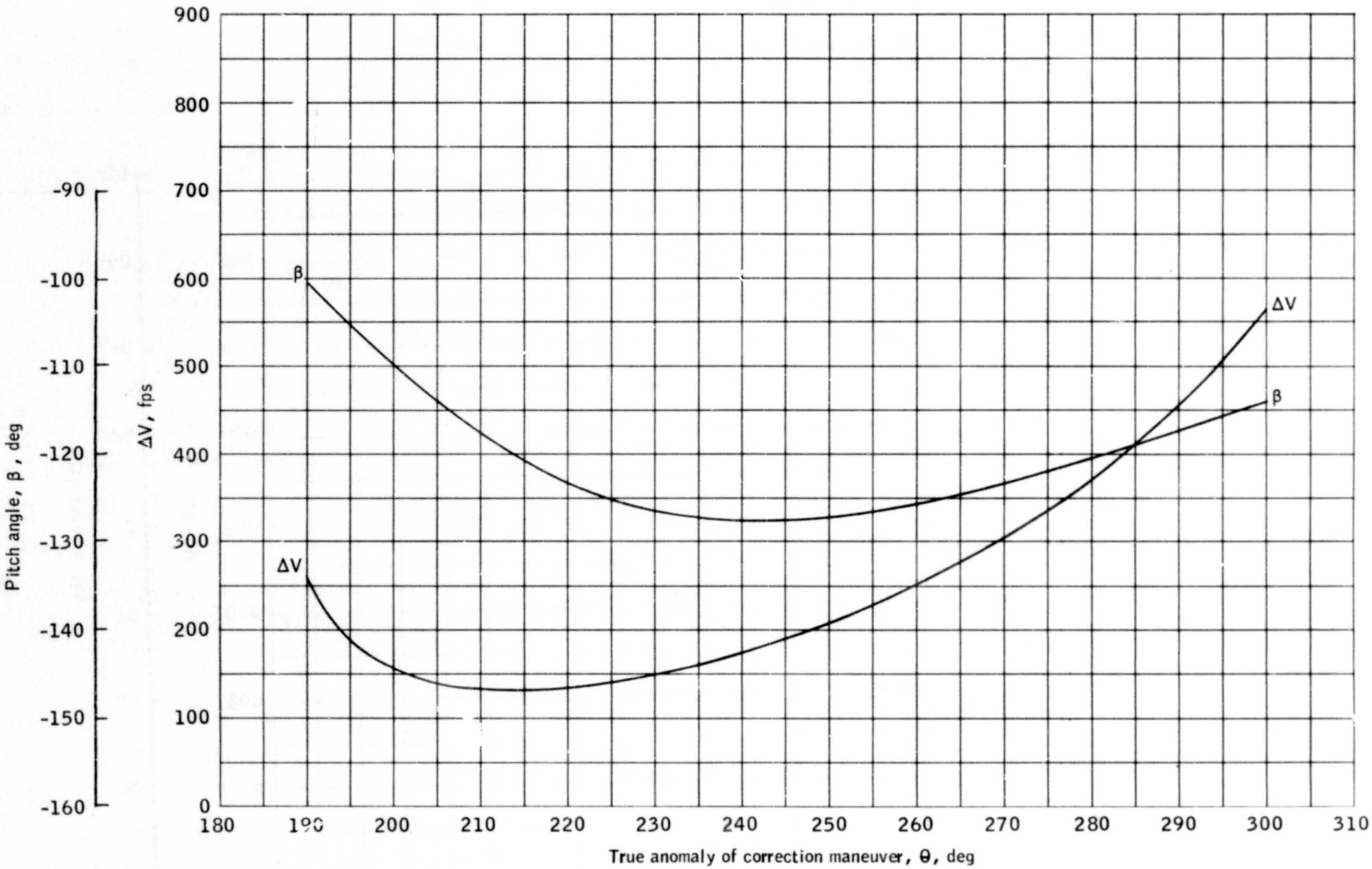


Figure 16.- Deorbit burn time required to hit reentry corridor for various spacecraft weights.



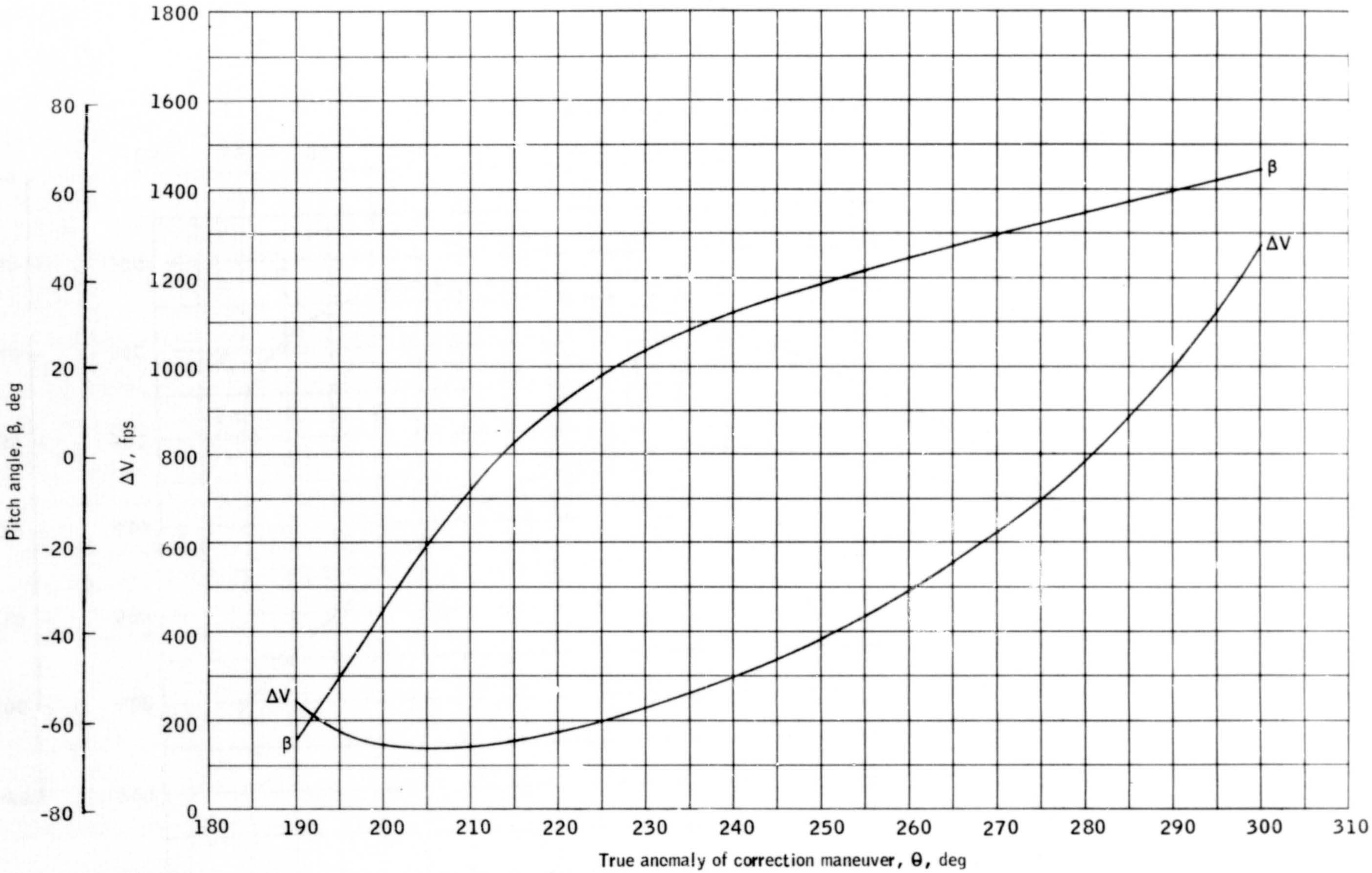
(a) 1% SPS overthrust.

Figure 17.- RCS requirements to correct a non-nominal SPS deorbit maneuver to the middle of the entry corridor.



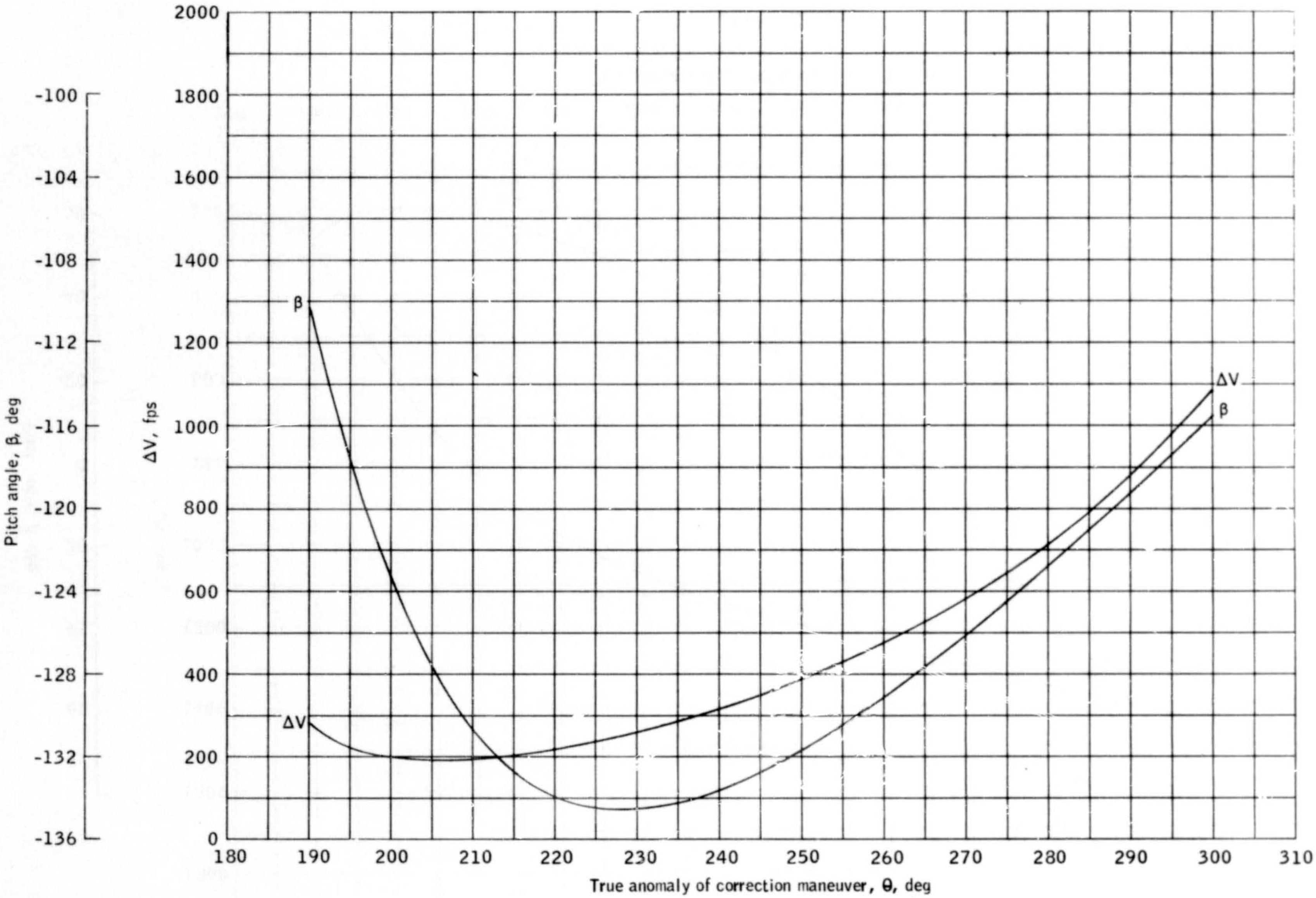
(b) 1% SPS underthrust.

Figure 17.- Continued.



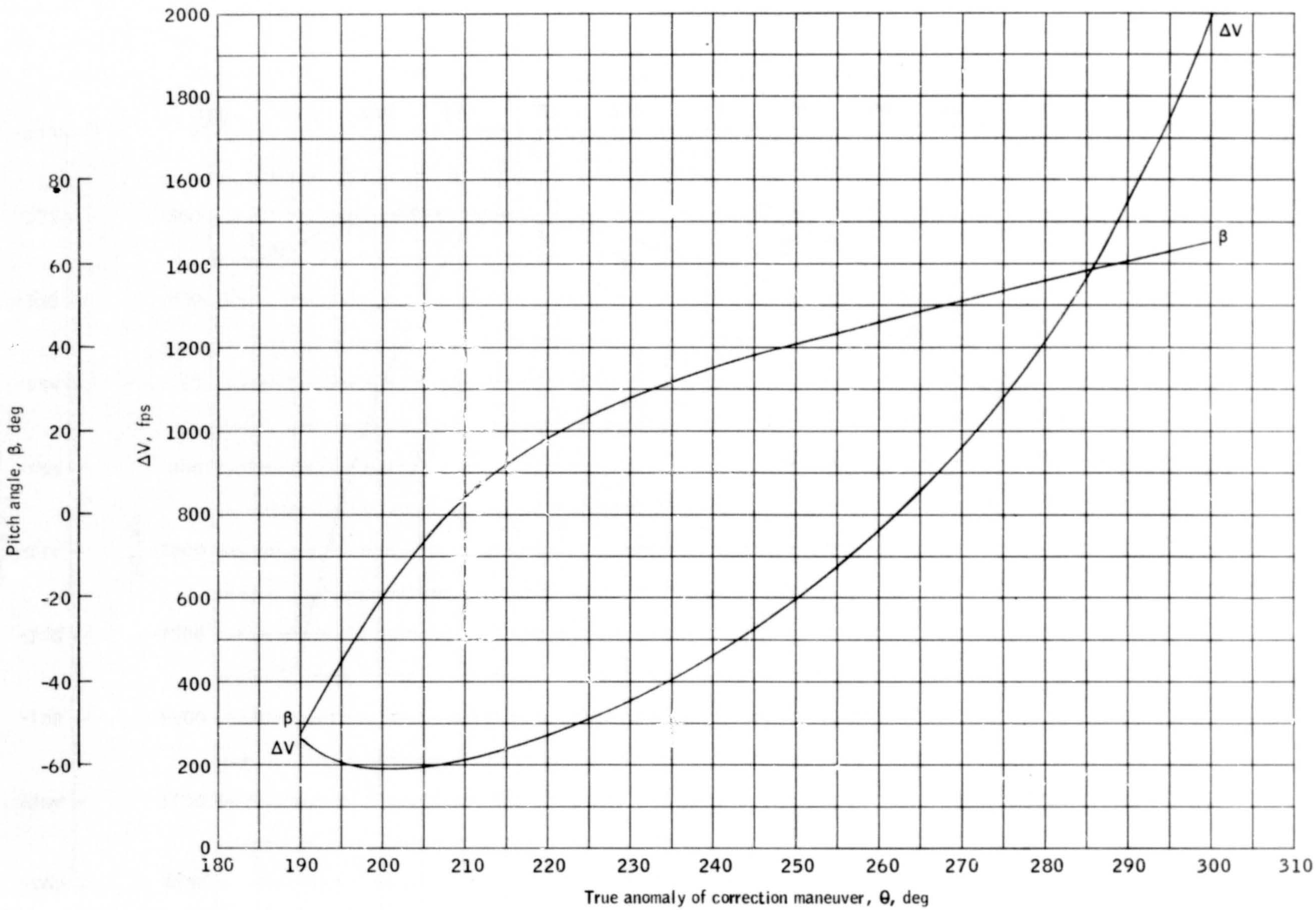
(c) 2% SPS overthrust.

Figure 17.- Continued.



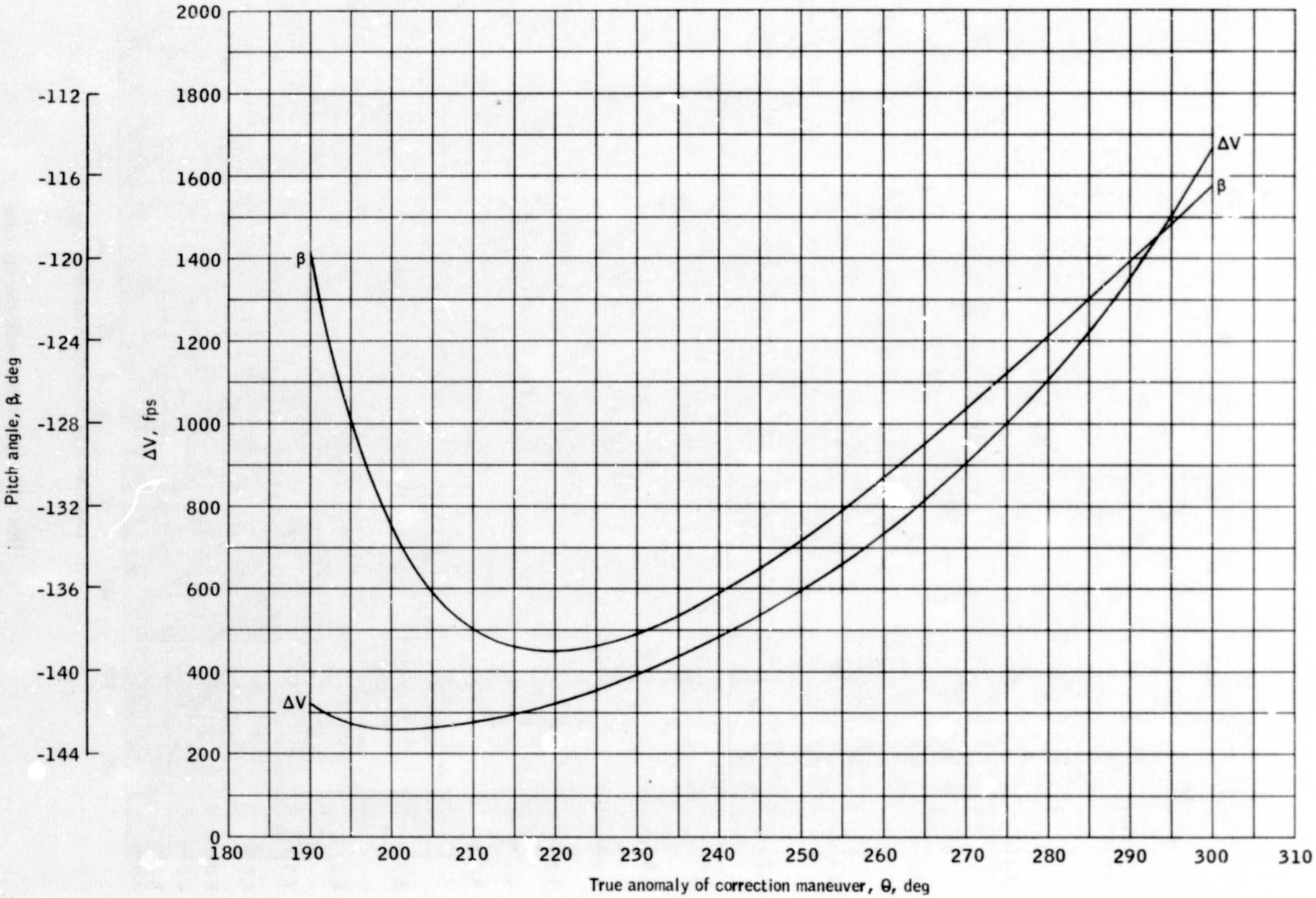
(d) 2% SPS underthrust.

Figure 17.- Continued.



(e) 3% SPS overthrust.

Figure 17.- Continued.



(f) 3% SPS underthrust.

Figure 17.- Concluded.

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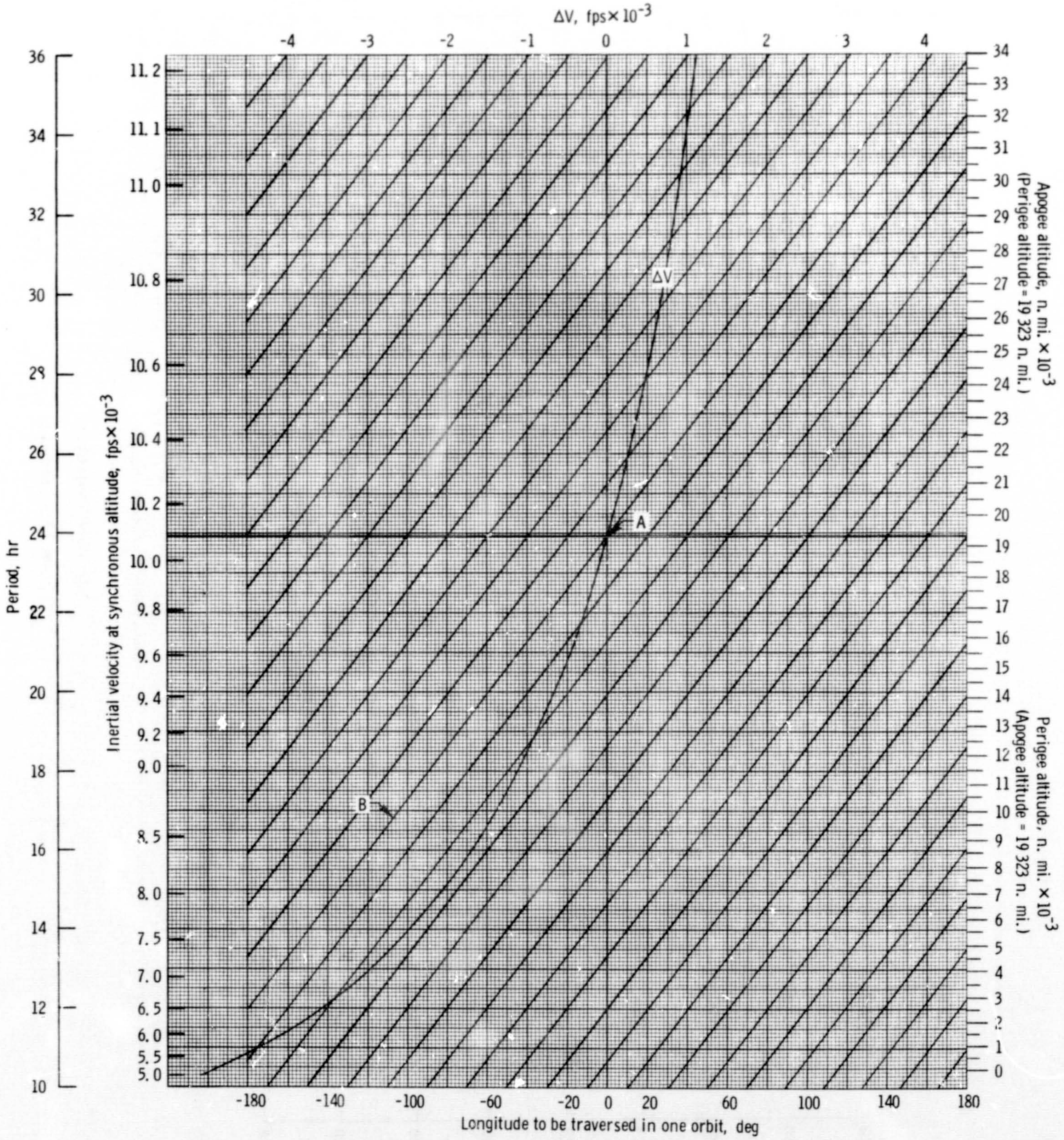


Figure 18. - Synchronous orbit phasing guide.

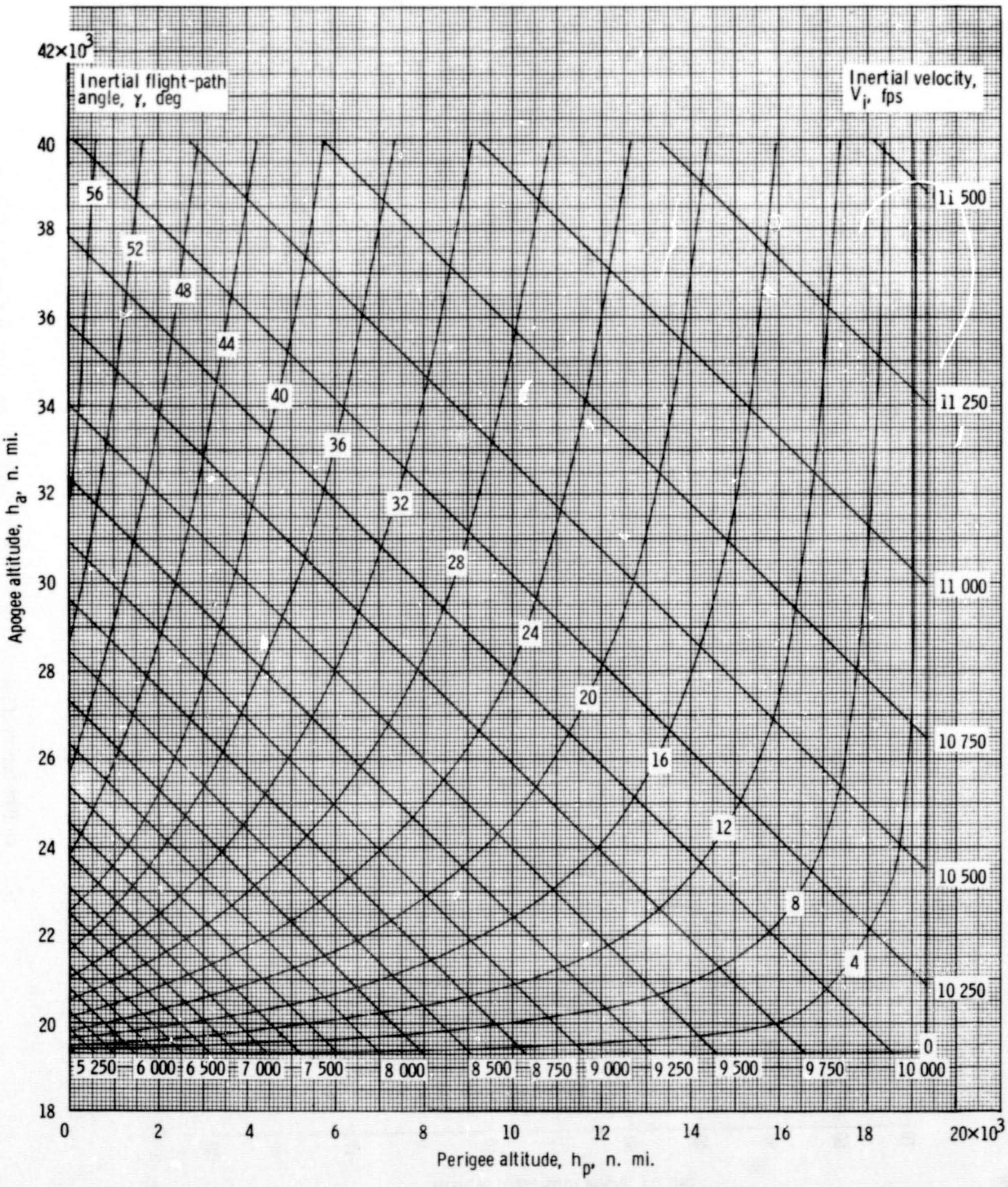
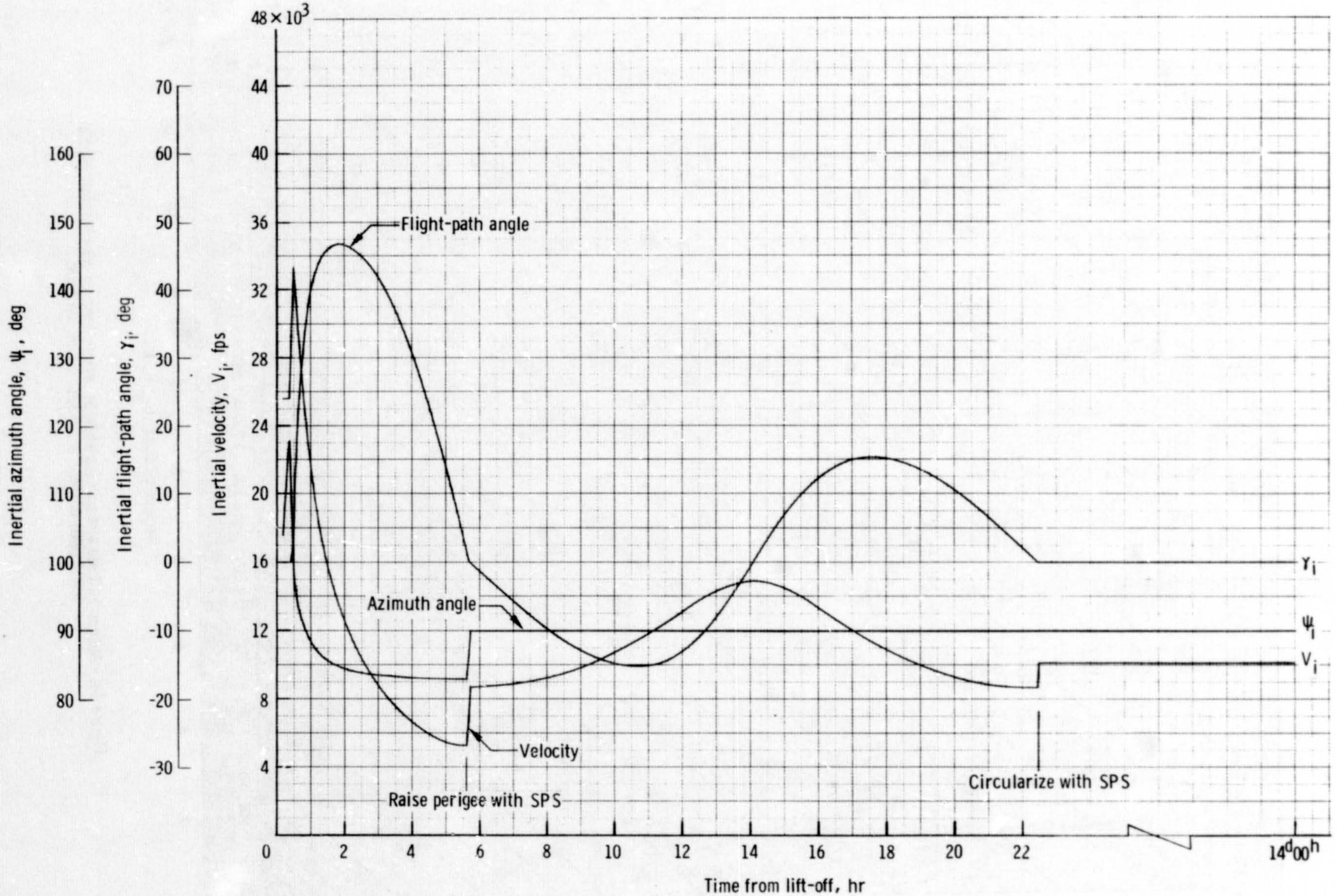
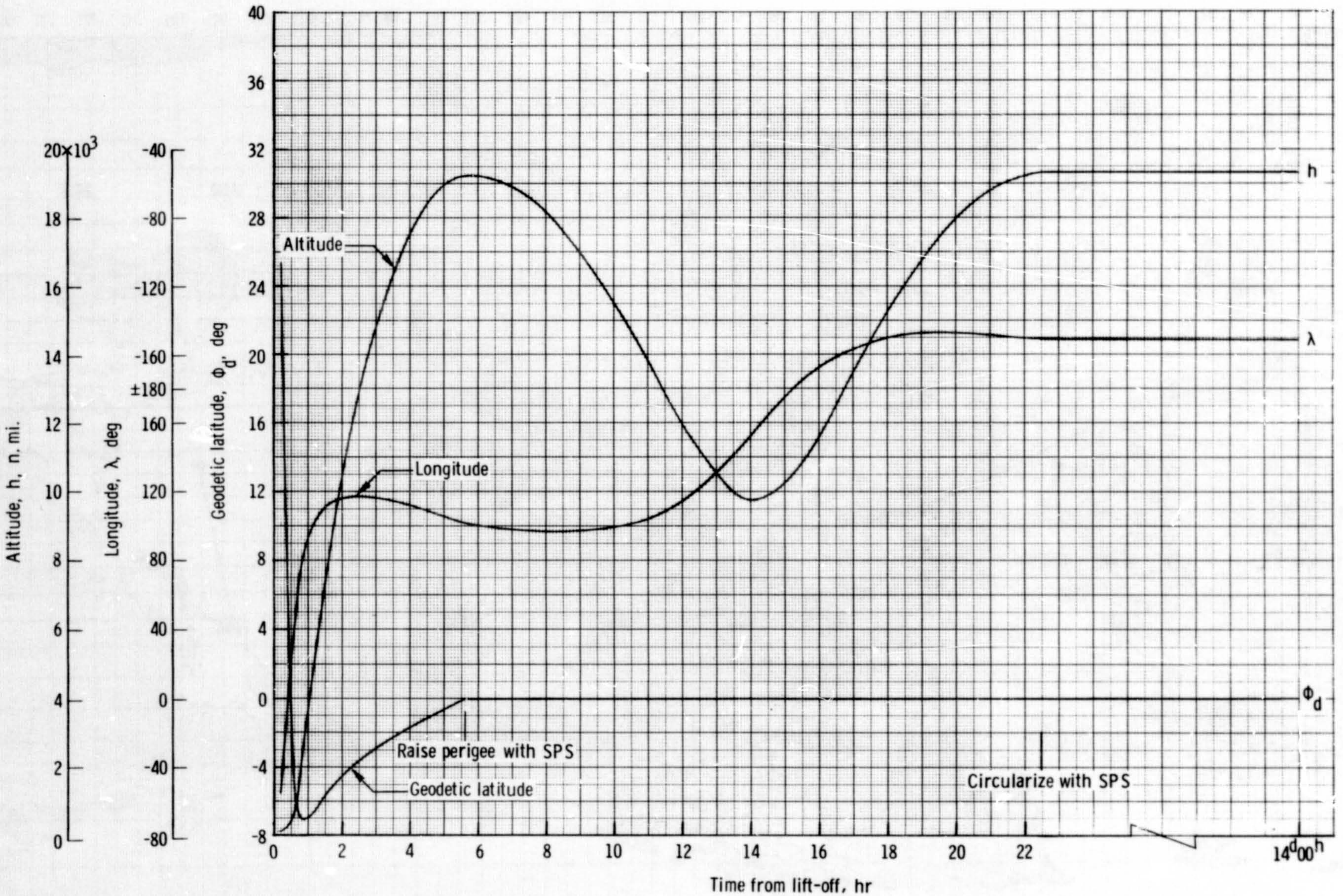


Figure 19. - Apogee altitude and perigee altitude versus inertial velocity and inertial flight-path angle at a 19 323 n. mi. altitude.



(a) Time history of inertial velocity, inertial flight-path angle and inertial azimuth angle.

Figure 20. - Circular synchronous equatorial mission 2 ascent.



(b) Time history of geodetic latitude, longitude and altitude.

Figure 20. - Concluded.

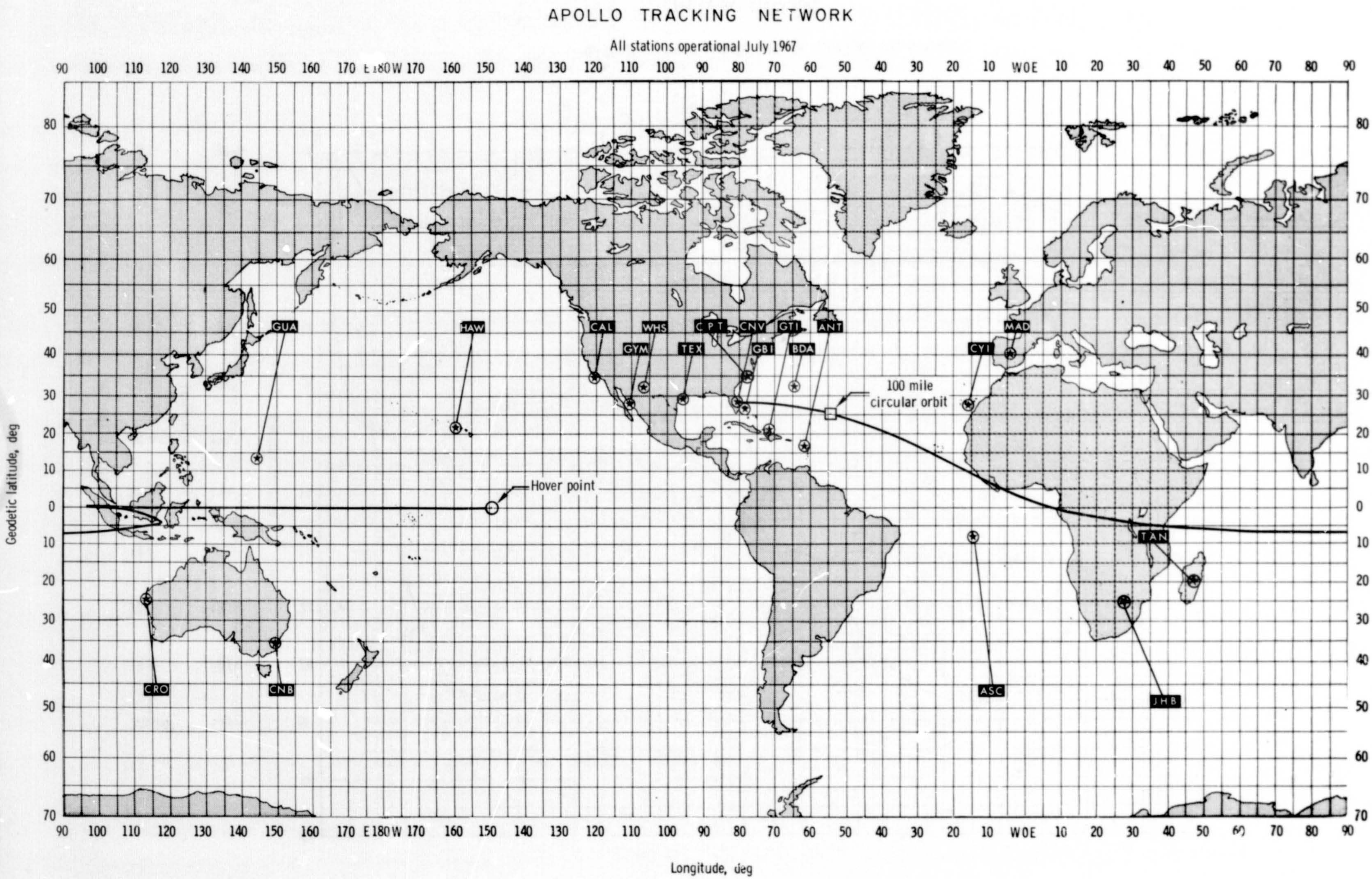


Figure 21. - Groundtrack for circular synchronous equatorial mission 2.

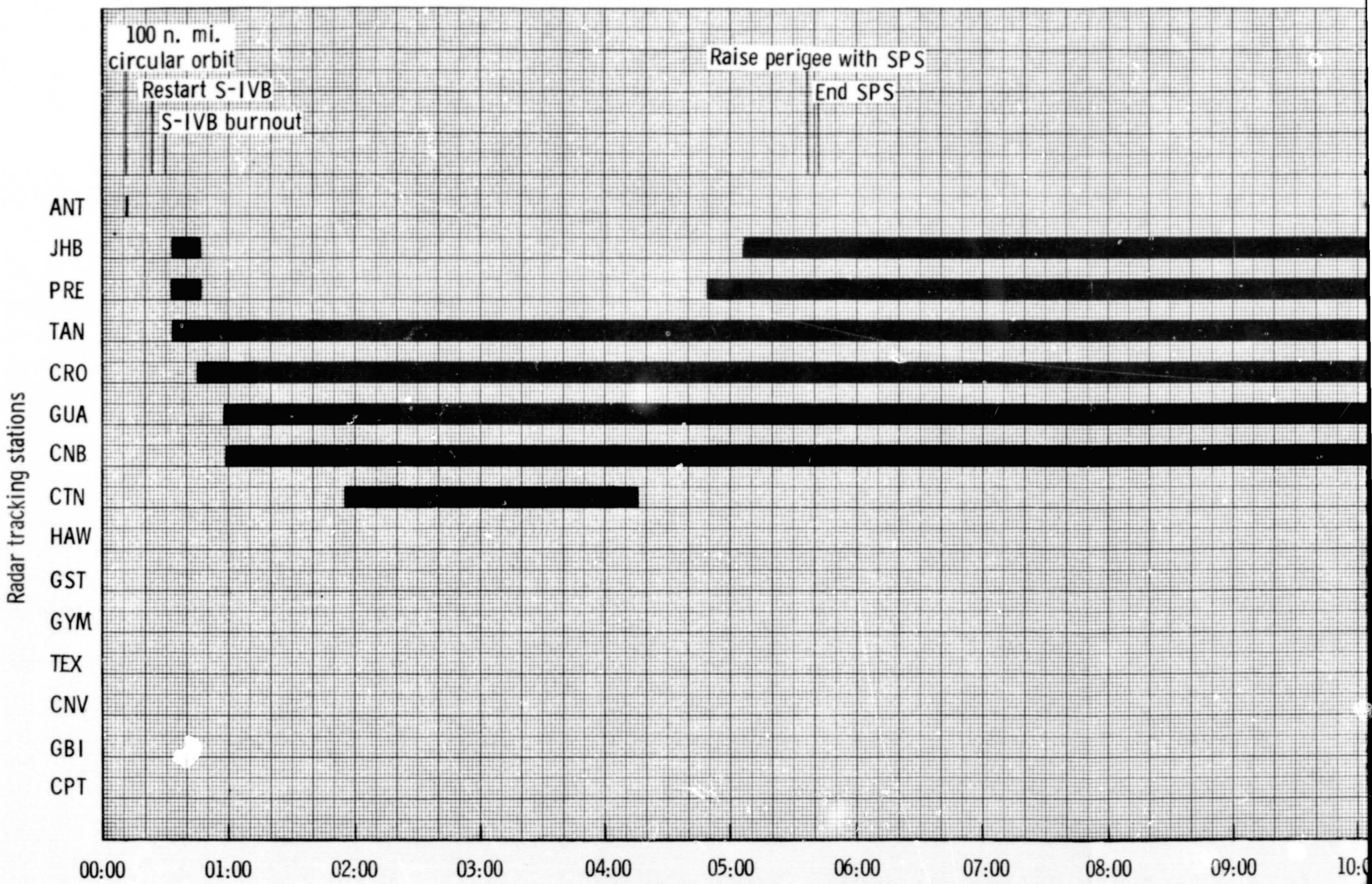
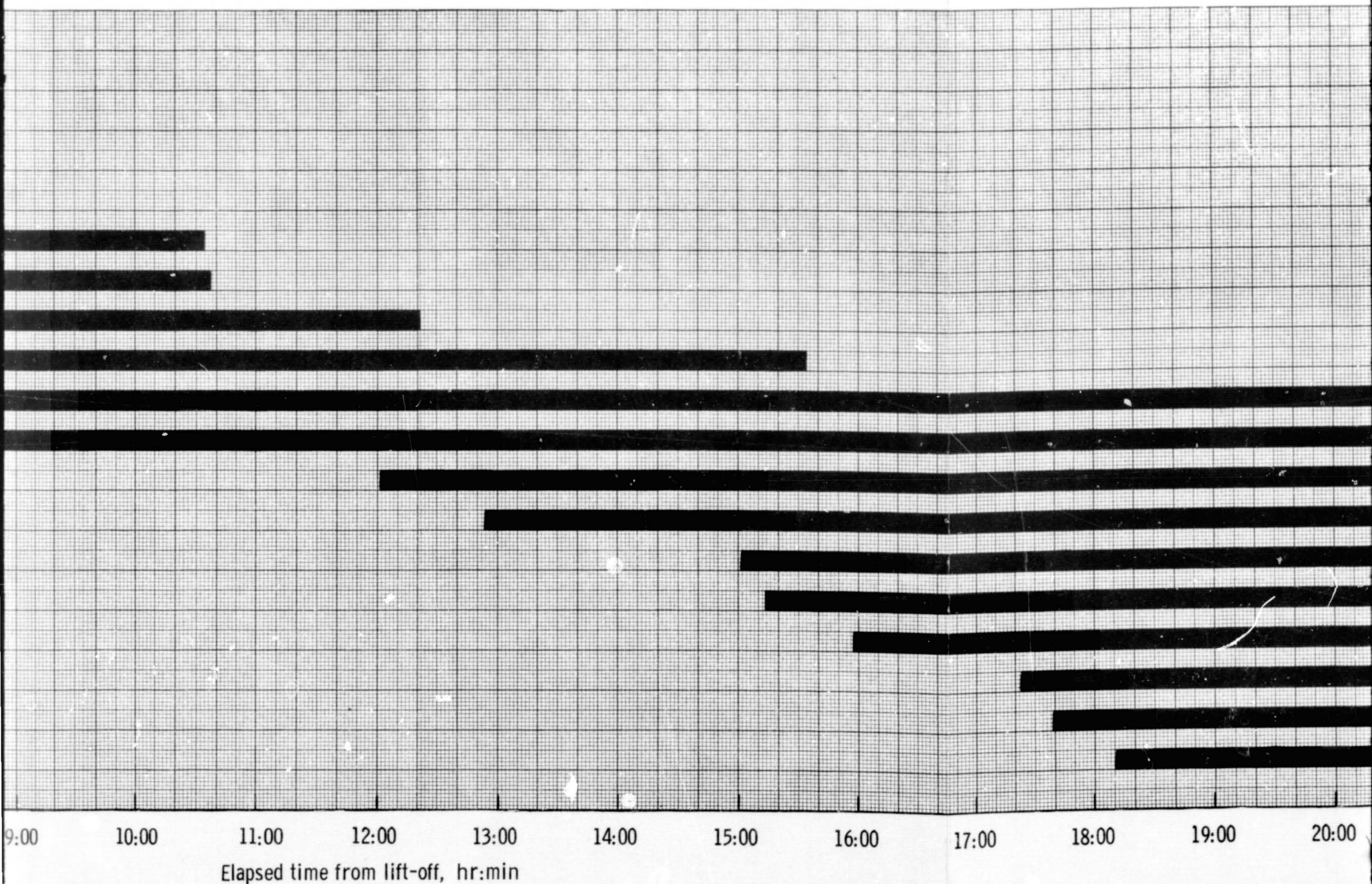


Figure 22. - Trackin

FOLDOUT FRAME 1

FOLDOUT FRAME

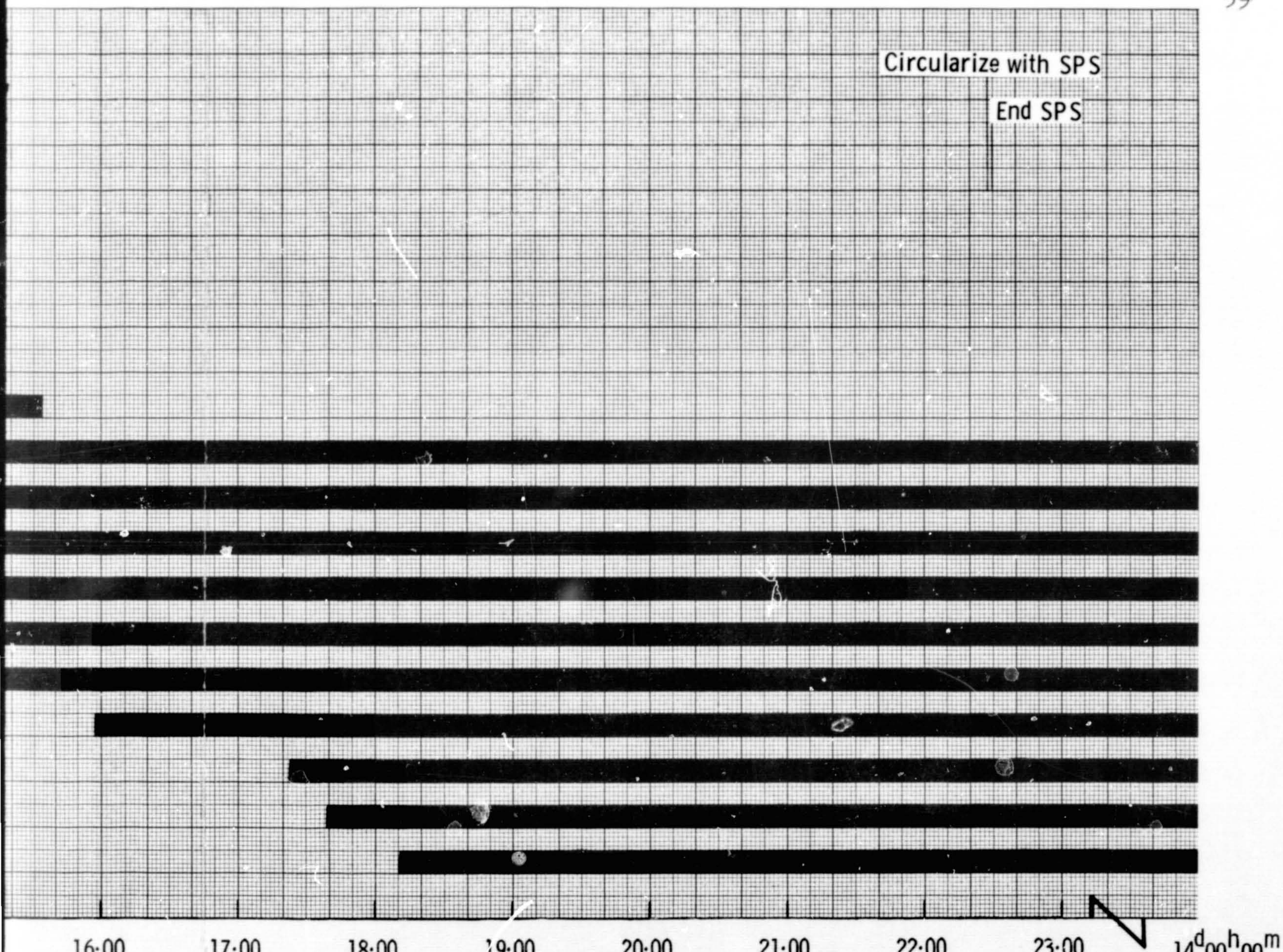


ure 22. - Tracking station coverage through synchronous orbit insertion - Mission 2.

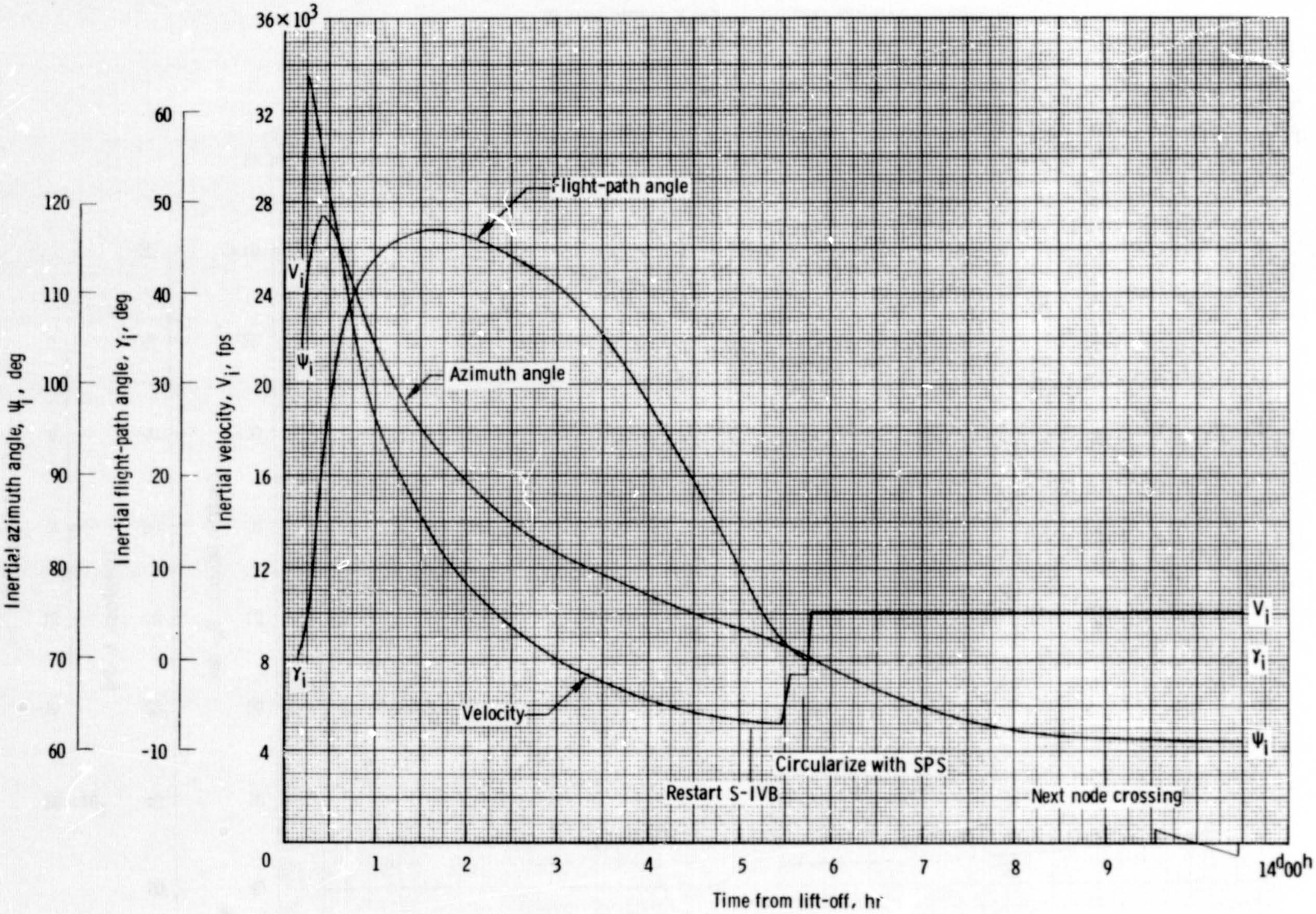
UT FRAME

FOLDOUT FRAME 2

FOLDOUT FRAME 3



FOLDOUT FRAME 3



(a) Time history of inertial velocity, inertial flight-path angle and inertial azimuth angle.

Figure 23. - Circular synchronous inclined mission 3 ascent.

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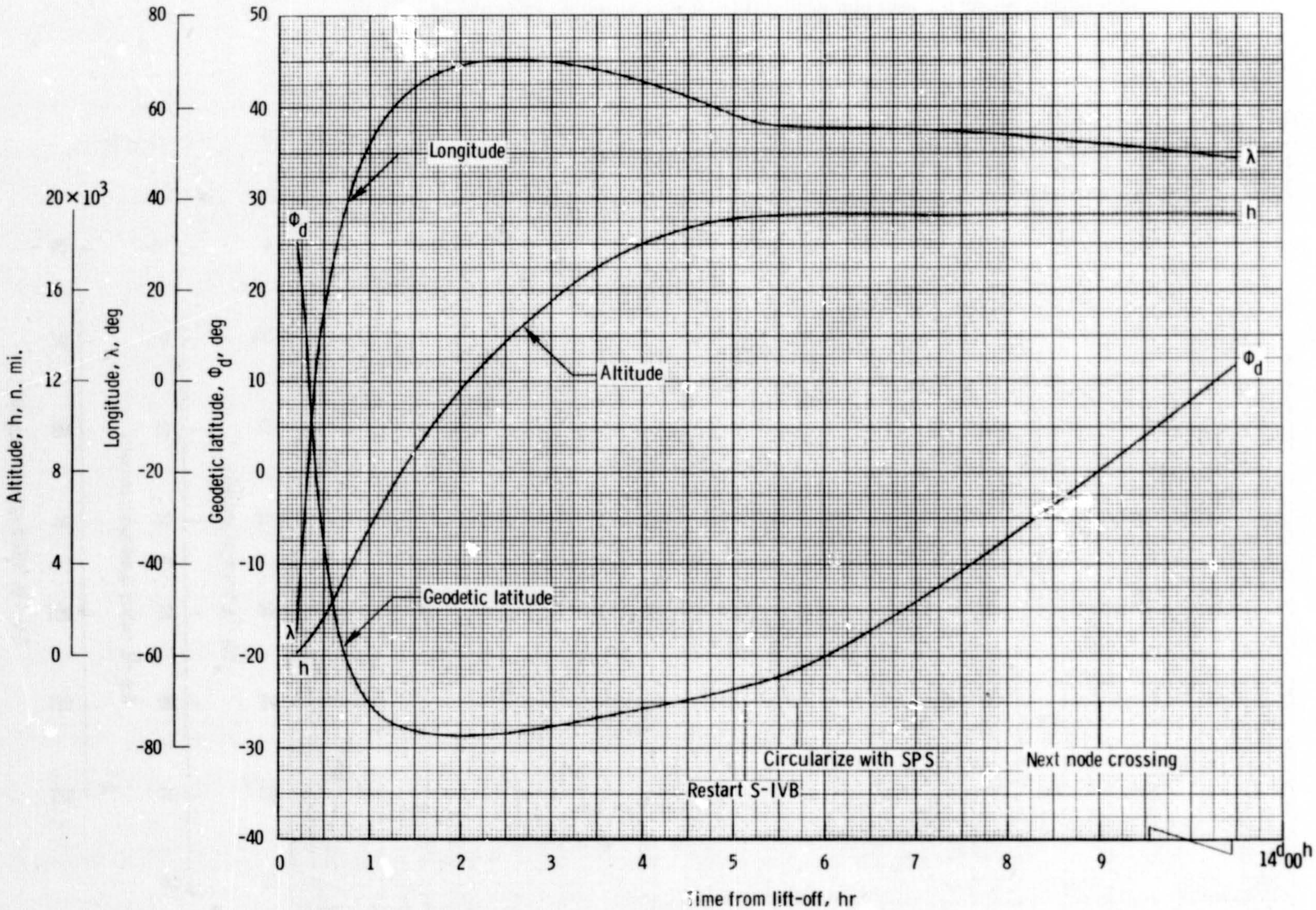


(b) Time history of geodetic latitude, longitude and altitude.

Figure 23. - Concluded.

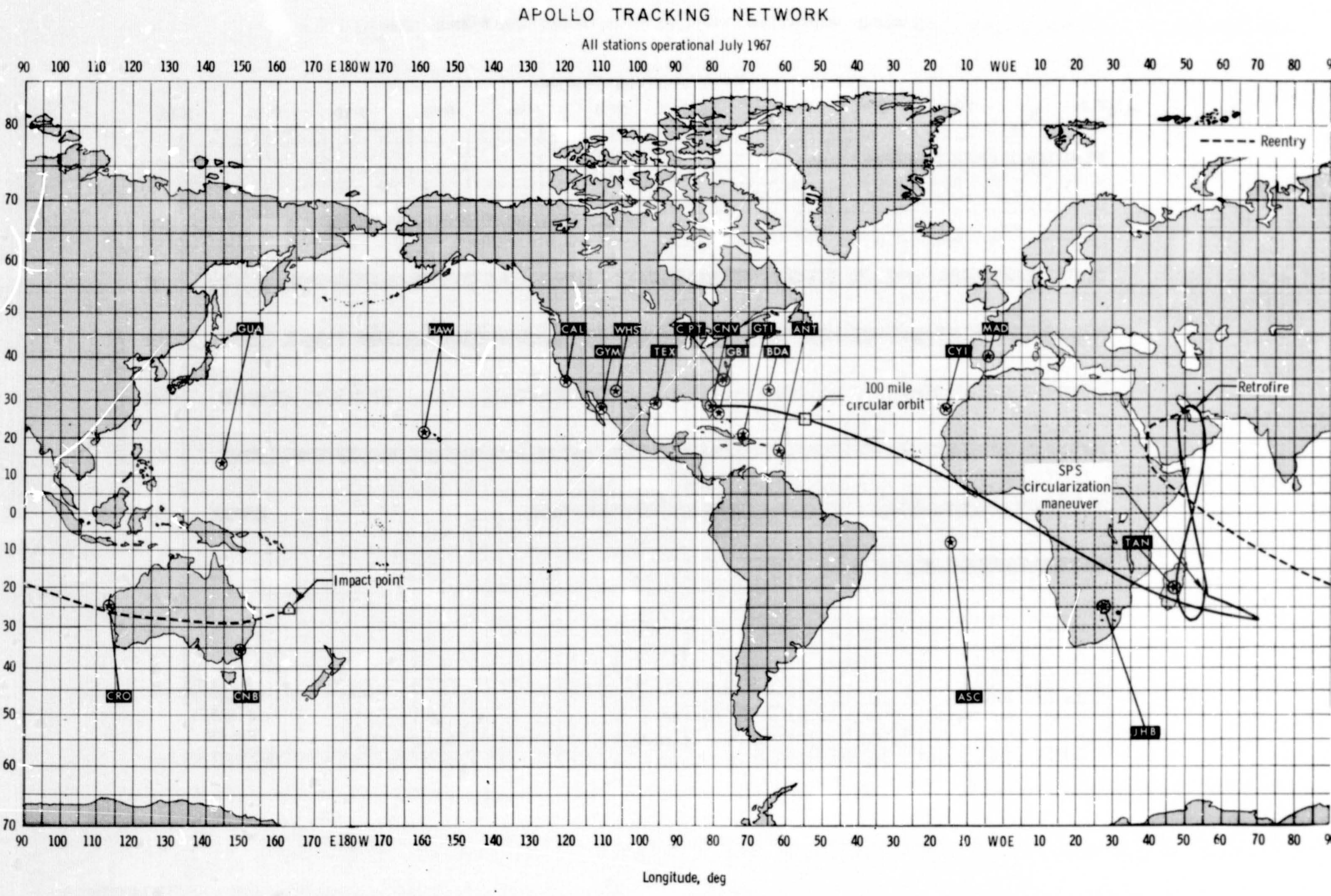


Figure 24. - Groundtrack for circular synchronous inclined mission 3 ascent.

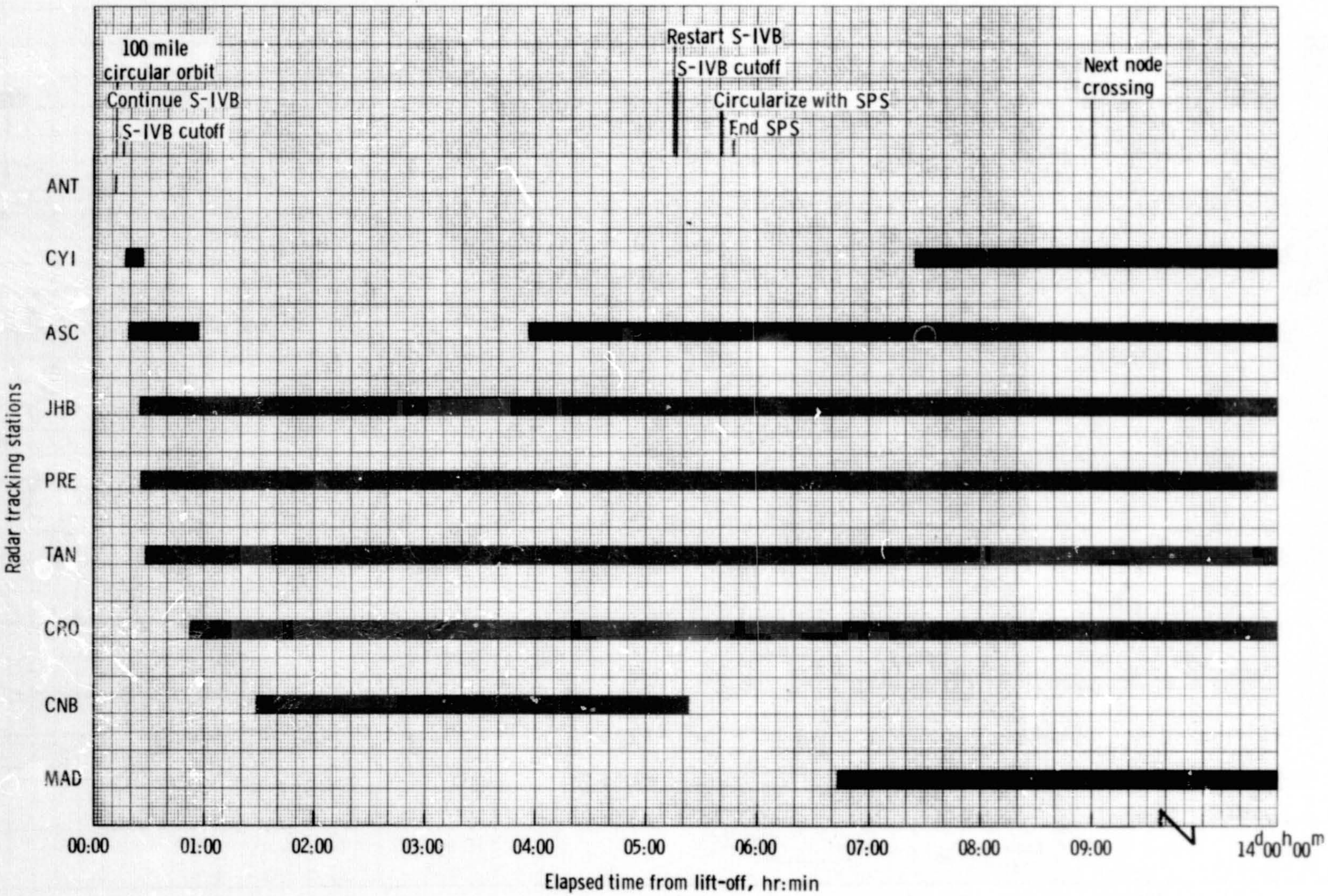


Figure 25. - Tracking station coverage through synchronous orbit insertion - Mission 3.

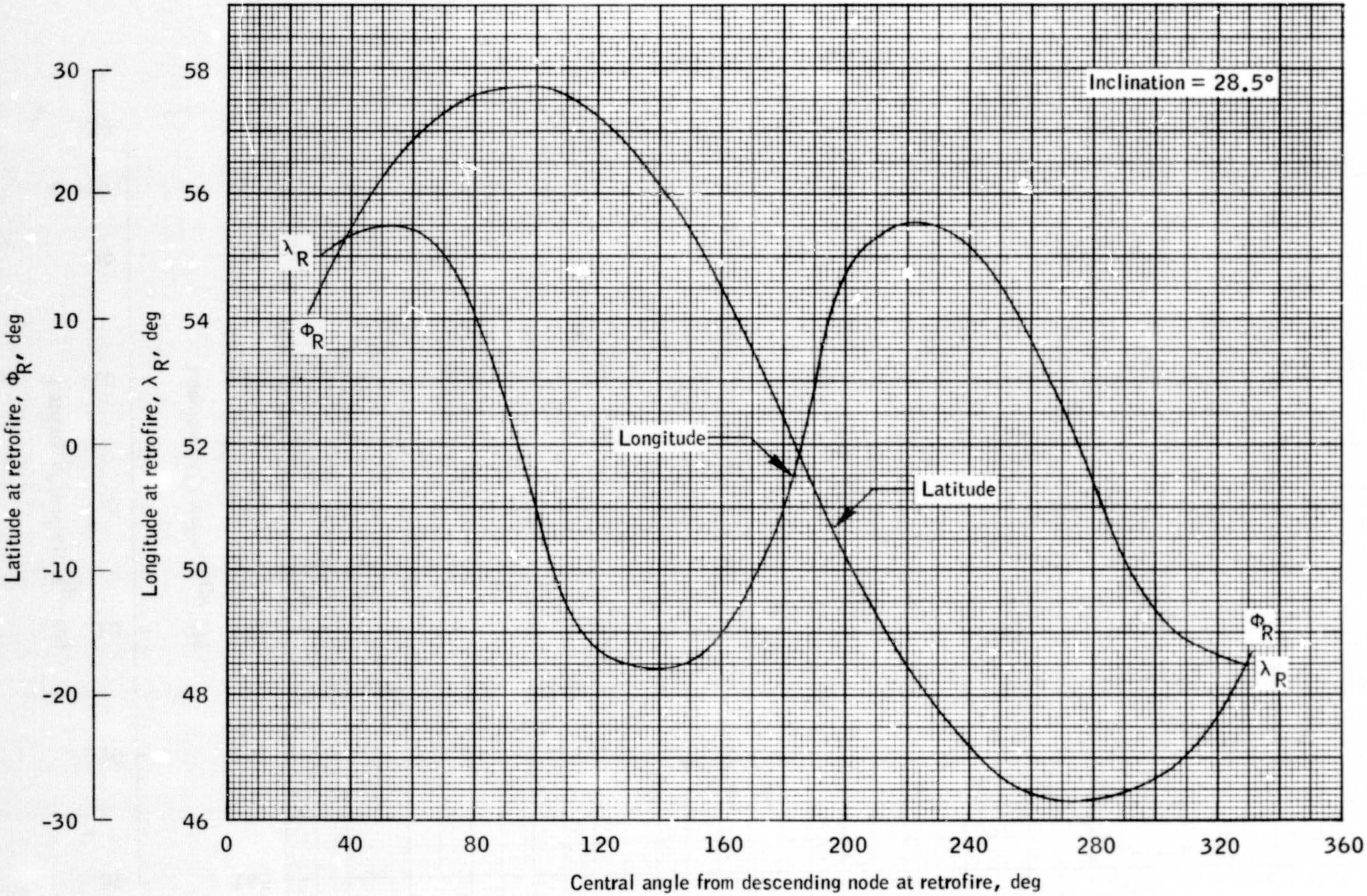


Figure 26.- Latitude and longitude at retrofire versus central angle from descending node.

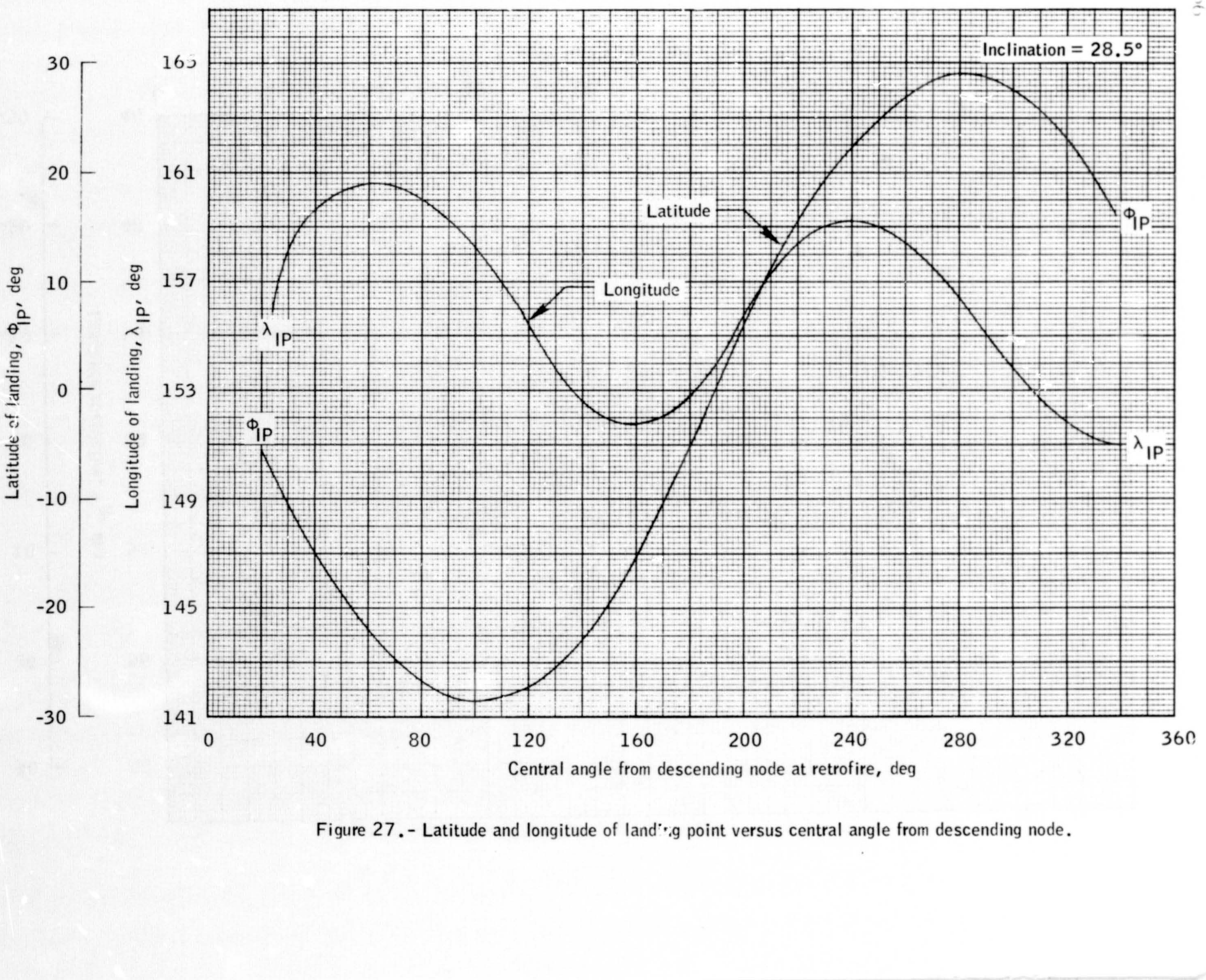


Figure 27.- Latitude and longitude of landing point versus central angle from descending node.

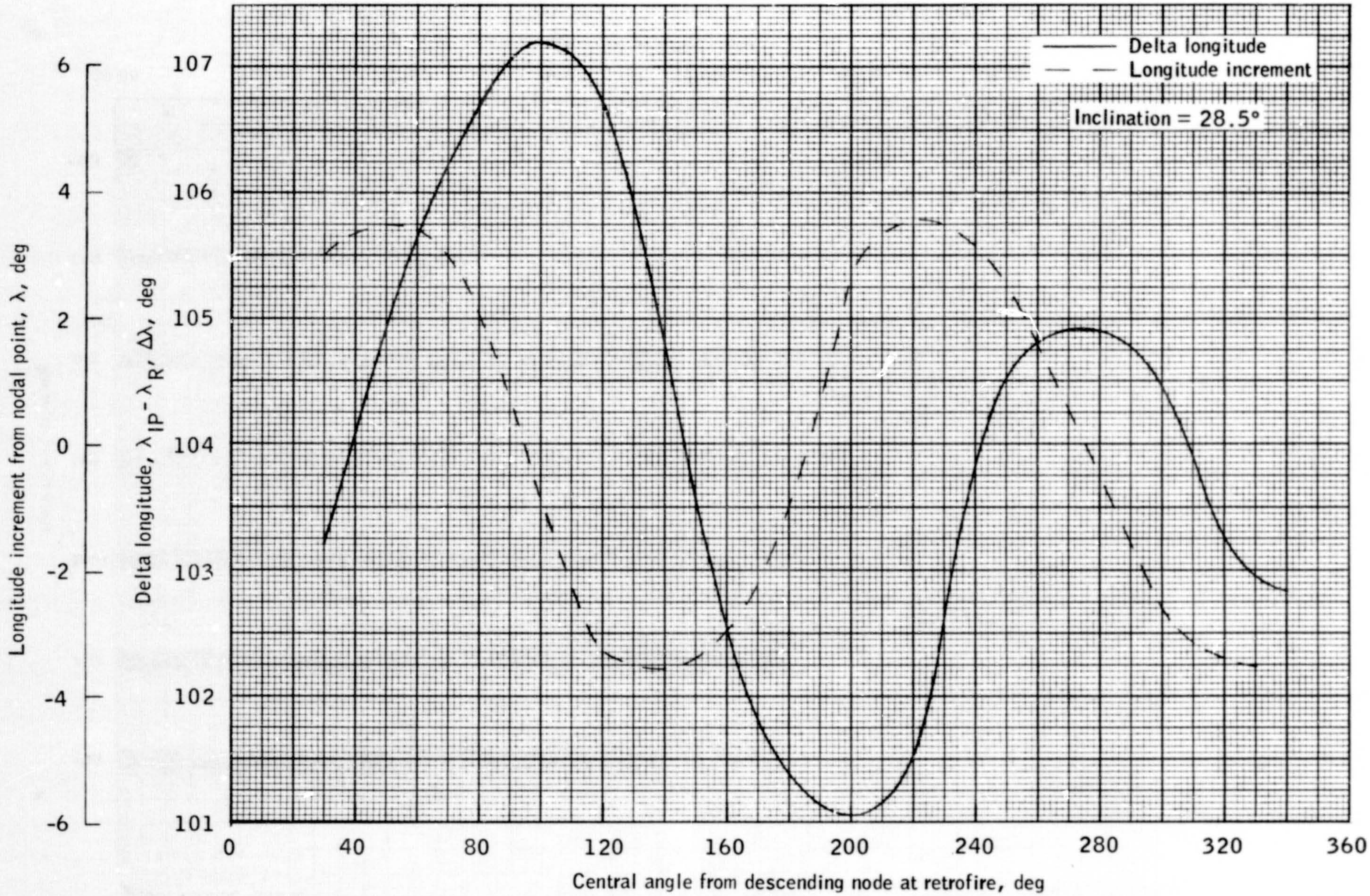


Figure 28.- Longitude increment between retrofire and landing versus central angle from descending node.

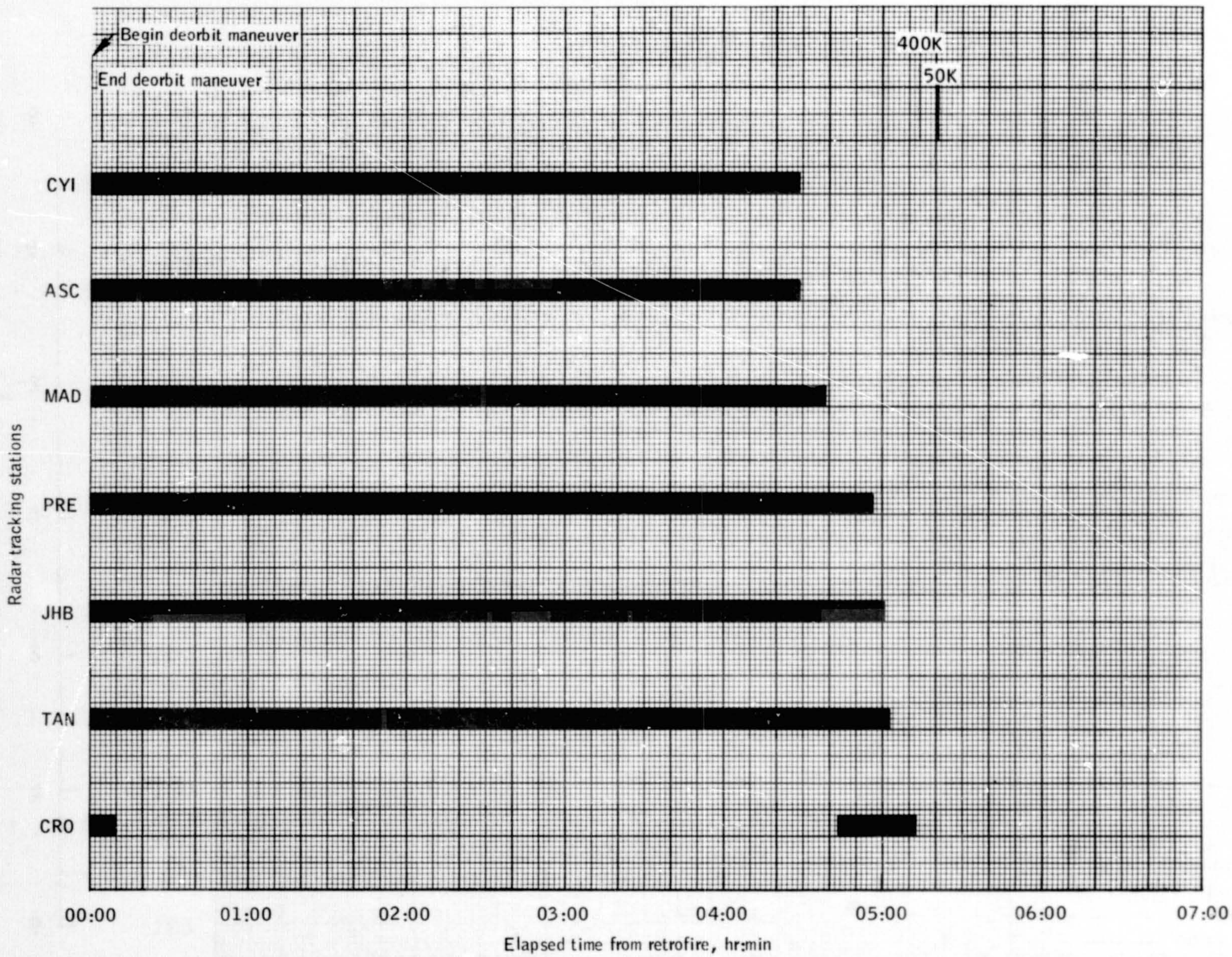


Figure 29.- Tracking station coverage from retrofire to 50 000 feet - Mission 3.

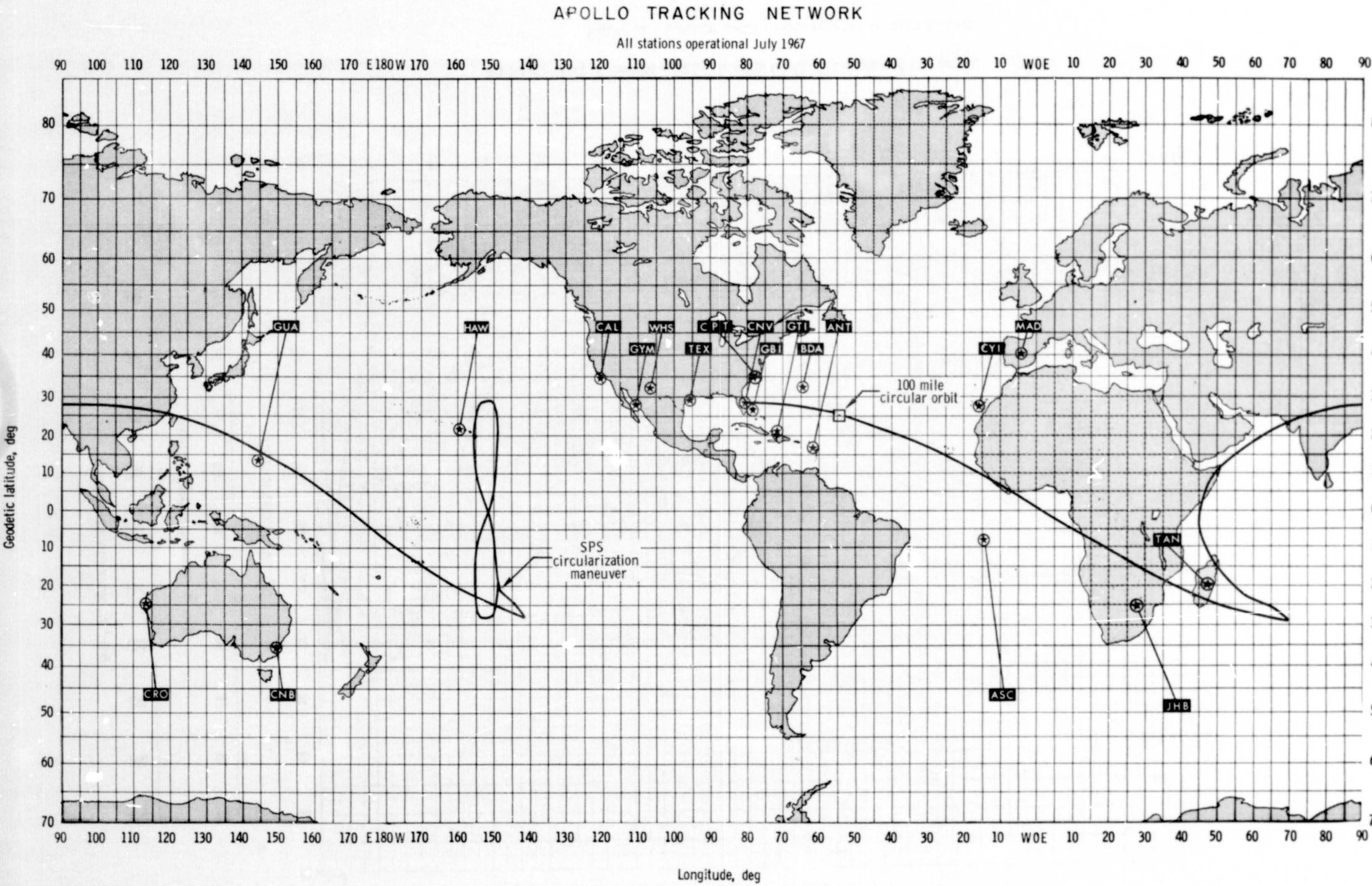
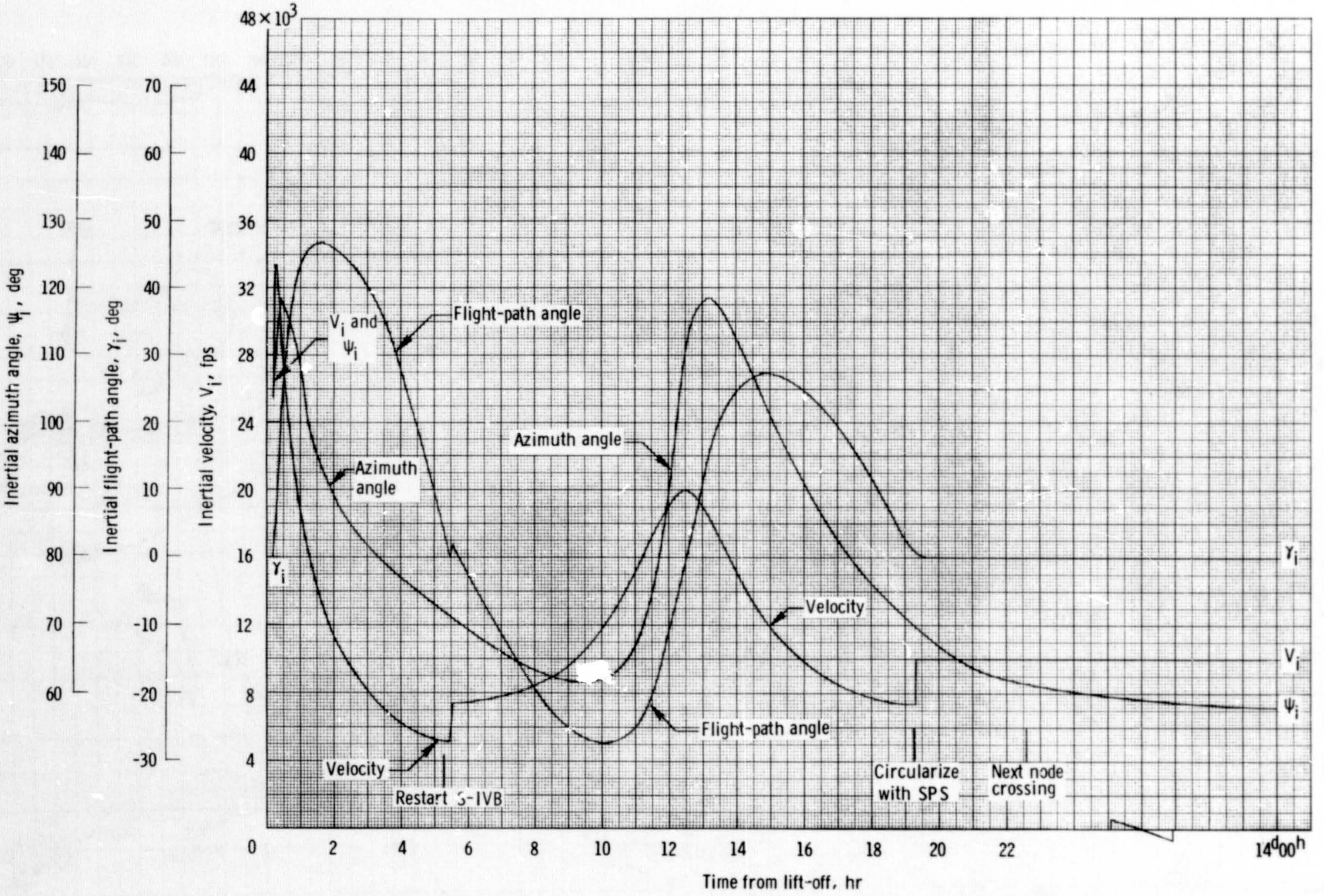
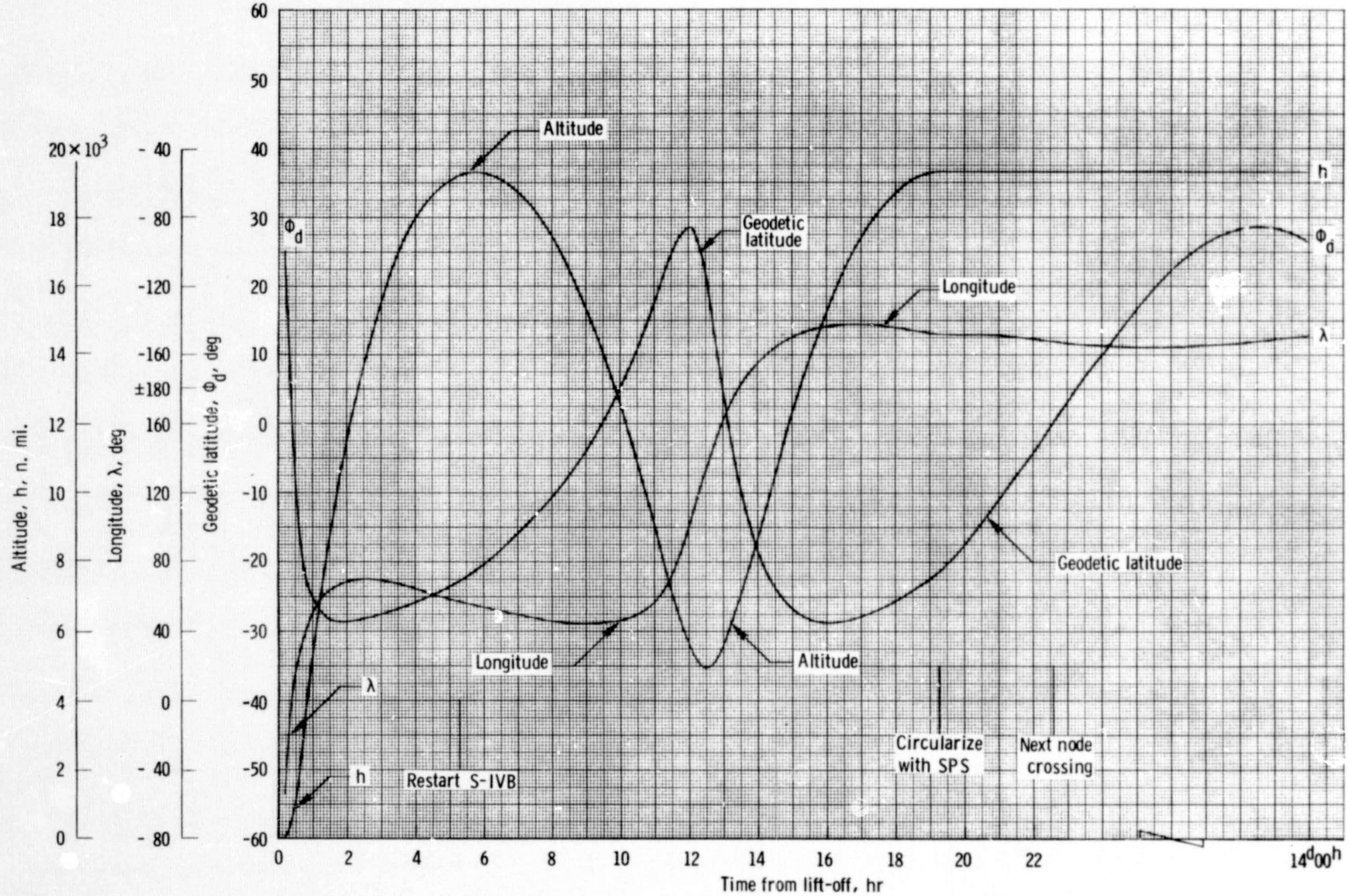


Figure 30. - Groundtrack for circular synchronous inclined mission 4 ascent.



(a) Time history of inertial velocity, inertial flight-path angle and inertial azimuth angle.

Figure 31. - Circular synchronous inclined mission 4 ascent.



(b) Time history of geodetic latitude, longitude and altitude.

Figure 31. - Concluded.

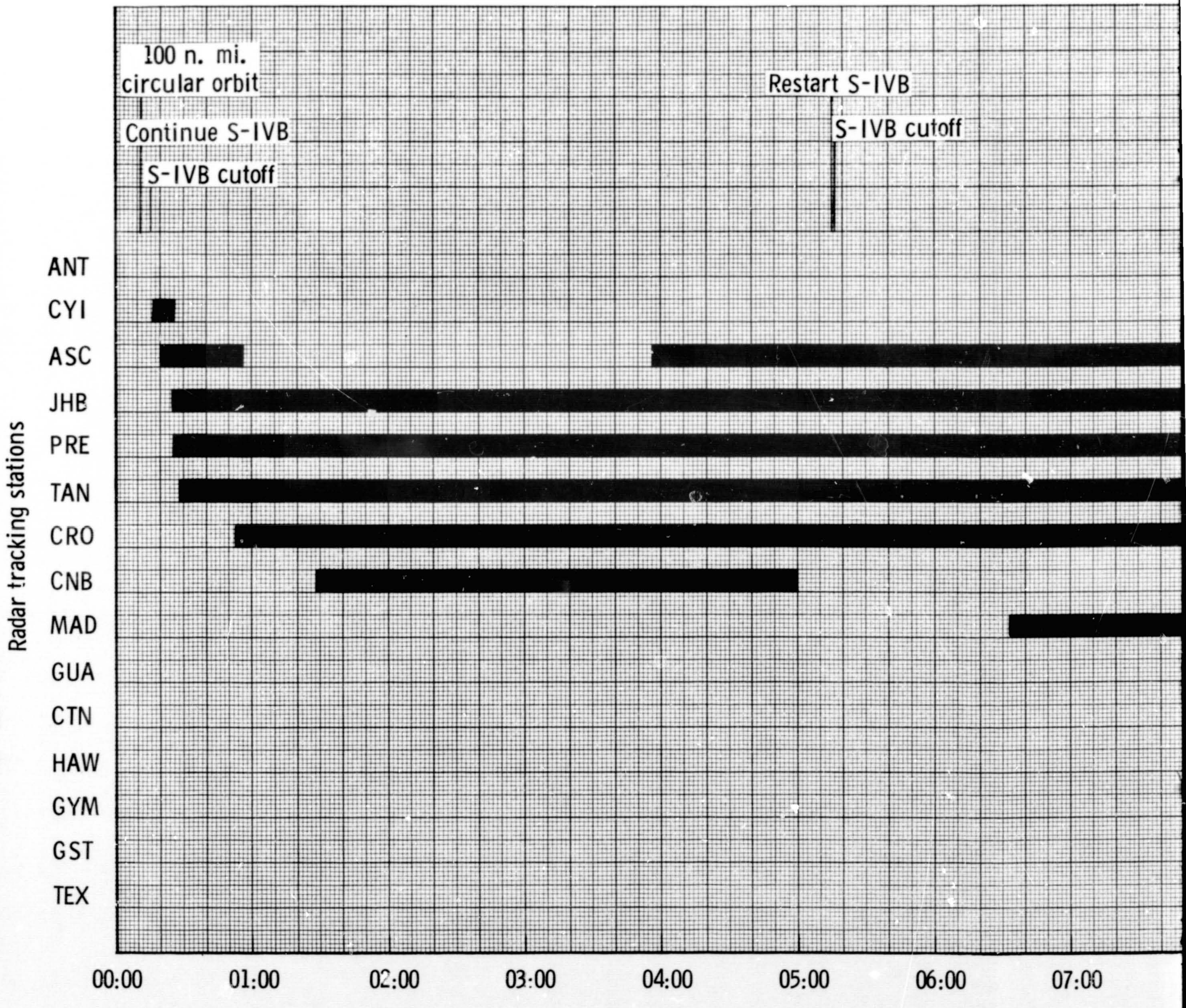


Figure 32. - Tr

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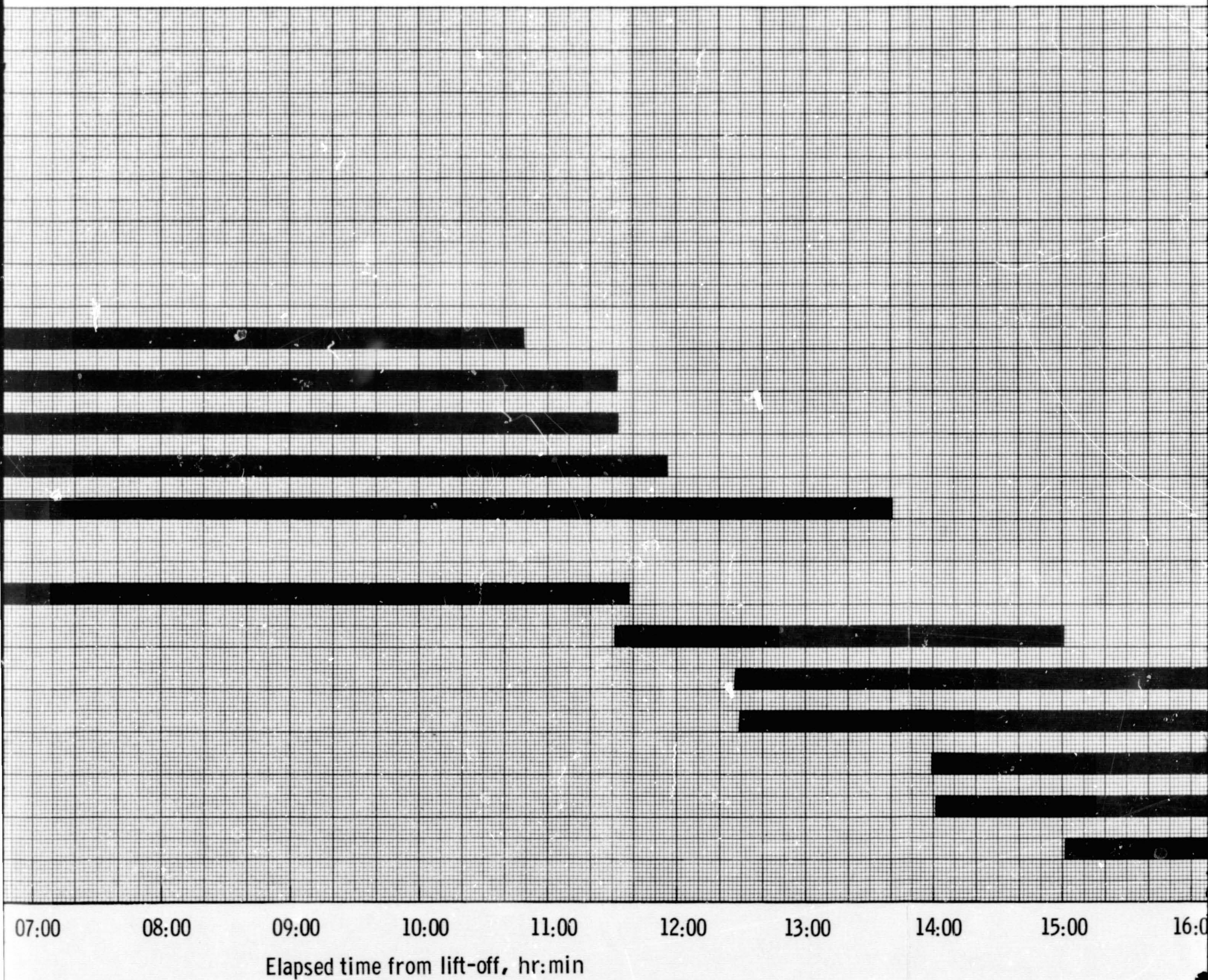

Figure 32. - Tracking station coverage through synchronous orbit insertion - Mission 4.

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FOLDOUT FRAME 2

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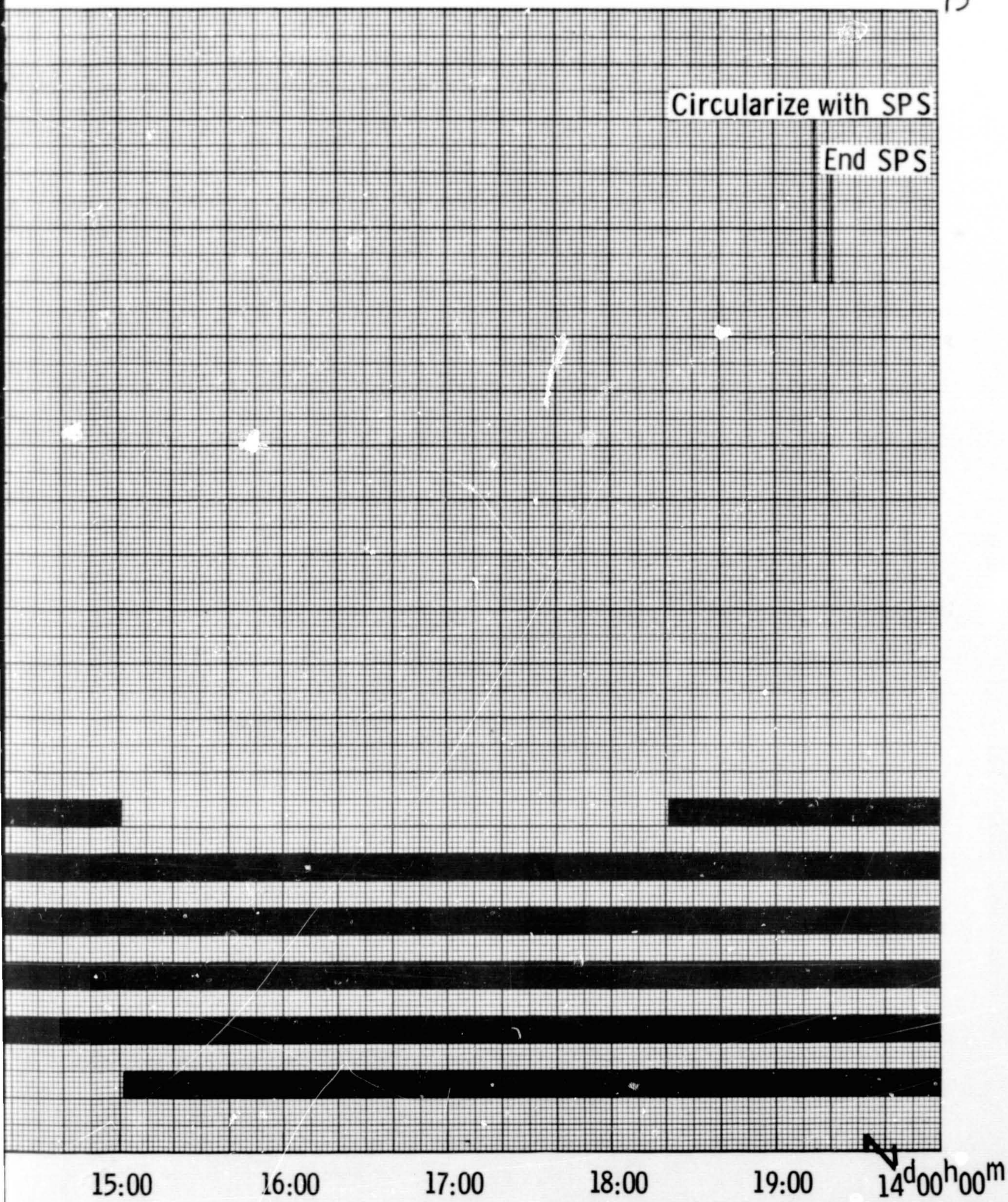

FOLDOUT FRAME 3

REFERENCES

1. Apollo Extension System, Mission Description-509, North American Aviation. INC-SID Report 65-1725, January 31, 1966.

2. Apollo Extension Systems Trajectory Study, TRW Report 4226-6010-RC000, September 3, 1965.