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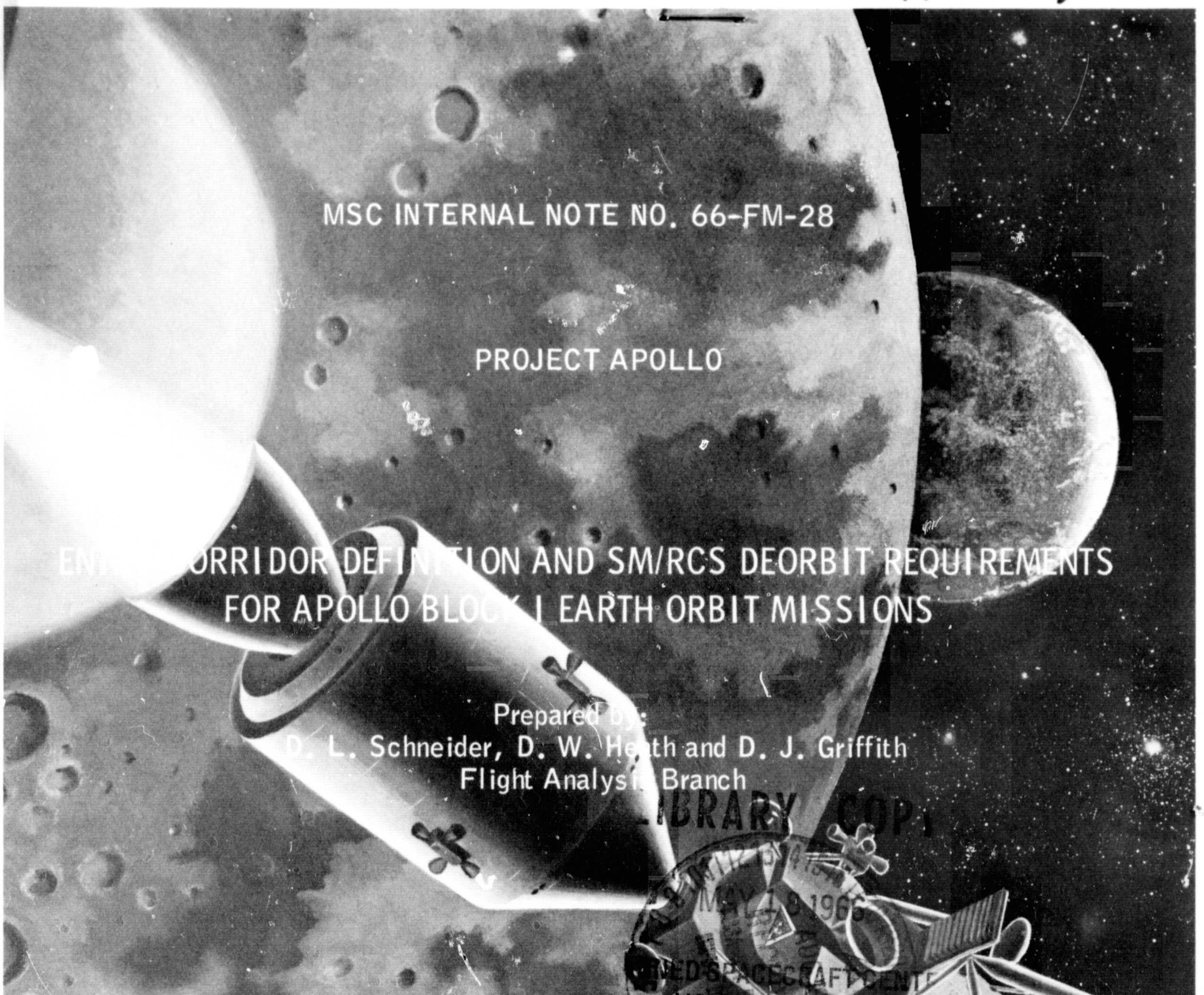

MSC INTERNAL NOTE NO. 66-FM-28

PROJECT APOLLO

ENTRY CORRIDOR DEFINITION AND SM/RCS DEORBIT REQUIREMENTS FOR APOLLO BLOCK I EARTH ORBIT MISSIONS

Prepared by:
D. L. Schneider, D. W. Heath and D. J. Griffith
Flight Analysis Branch

MISSION PLANNING AND ANALYSIS DIVISION
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REFERENCES

1. MSC Memorandum, "Results of the AS-204A FOP Meeting," dated December 8, 1965, (Unclassified).

2. MSC Memorandum, "Apollo Block I CM Abort Reentry ECS Evaluation," prepared by Systems Analysis Group, (Unclassified).

3. TRW Document, 3640-6012-TU-000, "Mission Requirements for Apollo Spacecraft Development Mission Apollo Saturn 204A (CSM 012/205 CSM 014)," (U), dated August 30, 1965, (Confidential).

4. NAA Document SID 65-806, "Mission Compatability Evaluation Mission AS-204A Spacecraft 012 Project Apollo," (U), dated September 15, 1965, (Confidential).

5. MSC Internal Note No. 65-FM-57, "Reentry Phase of the AS-204A Preliminary Reference Trajectory," (U), dated October 20, 1965, (Confidential).

6. MSC Memorandum "CSM ECS Water-glycol Evaporator Status and Recommendations," dated January 21, 1966, from Apollo Spacecraft Program Office, (Unclassified).

7. MSC Internal Note No. 65-FM-58, "Preliminary Reference Trajectory for a Long Duration Manned Earth Orbital Apollo Mission AS-204A," (U), dated April 26, 1965, (Confidential).

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ENTRY CORRIDOR DEFINITION AND SMRCS DEORBIT REQUIREMENTS

FOR APOLLO BLOCK I EARTH ORBIT MISSIONS

By David J. Griffith, Donald P. Schneider, and David W. Heath

1.0 SUMMARY

This document defines the entry corridor and the minimum deorbit velocity requirements of the service module reaction control system (SMRCS) for Block I Apollo missions. The velocity requirements are based on impulsive maneuvers but may be correlated to finite thrusting.

Sufficient information for planning any Block I mission can be obtained from the results presented in the figures of this document. This information can also be used to determine propellant reserve requirements for the SMRCS when the SMRCS is used for ejection from either a circular or an elliptical orbit.

The results indicate that an ejection near the apogee of an elliptical orbit requires less ΔV - and, therefore, less SMRCS propellant reserves - than an ejection from a circular orbit. The ΔV required for ejection was found to be a function of several factors:

a. Orbital shape and altitude.

b. Mode and time of entry.

c. Efficiency of the total impulse translations.

d. Deorbit attitude.

2.0 INTRODUCTION

The purpose of this document is to define the entry corridor and present minimum SMRCS deorbit velocity requirements for the Apollo Block I missions. Sufficient parametric data may be obtained from the enclosed figures to design any Block I mission. These data may also be used for determining SMRCS propellant reserves requirements when considering use of the SMRCS for deorbit from both circular and elliptical orbits. The ΔV requirements quoted are based on impulsive maneuvers but may be correlated to finite thrusting by use of the following relations:

$$\Delta\theta = \frac{360^\circ}{T} \left(\frac{t_b}{2}\right)$$

where

$\Delta\theta$ = true anomaly

T = orbital period

$\frac{t_b}{2}$ = one-half burn time.

For a given ΔV the propellant required is given by

$$W_p = W_o \left(e^{\frac{\Delta V}{I_{sp}\, g_o}} - 1\right).$$

For the range of ΔV's available from the RCS the above relation can be reduced to

$$W_p = W_o \times (\text{decimal part of the exponent of } e)$$

where

W_o = initial spacecraft weight.

Therefore,

$$t_b = \frac{I_{sp}}{T} \times W_p.$$

The entry corridors presented were supplied by the Reentry Studies Section of Mission Analysis Branch.

4.0 METHOD OF ANALYSIS

When conditions occur that require use of the SMRCS as a backup deorbit mode, the crew is confronted with (1) marginal available propellant, (2) low thrust-to-weight ratio, and (3) the possibility of manual thrust-vector control. As a consequence, for entry survival deorbit near the entry time limit of 45 minutes (shallow-entry) is mandatory. The family of deorbit conics which best meet these conditions are minimum propellant deorbits to a fixed entry angle (i.e., steepest entry flight-path angle for a given amount of propellant).

The method of analysis was to take the polynominal of line c on figure 1 (SMRCS target boundary) and use an iterative technique for the solutions of minimum ΔV's required for deorbit from several feasible circular and elliptical orbits. By using this technique the absolute minimum ΔV required for entry within the established criteria could be determined.

4.0 RESULTS AND DISCUSSIONS

4.1 Atmospheric Entry Corridor

The reentry corridor for Block I missions is shown in figure 1. There are many ways of determining what criteria should be used for a overshoot boundary for a given spacecraft. The best way to determine the overshoot boundary is by examining the spacecraft structural and subsystem limitations.

The Block I heat shield was designed to withstand heat rates below 203 $BTU/ft^2/sec$ and a total heat load below 27000 BTU/ft^2. Between the overshoot boundary and undershoot boundary, line a and line g in figure 1, these heating limits are not exceeded.

4.1.1 Forty-five minutes time overshoot corridors. - Upon reviewing the basic Block I subsystems, it was found that all spacecraft electrical power during reentry was obtained from three parallel connected silver oxide zinc storage batteries after command service module (CSM) separation. It was determined that, with normal equipment and subsystem electrical power drain, 45 minutes of reentry time from CSM separation until touchdown would allow 48 hours of postlanding battery power. Additional powered reentry time, approximately 1 hour and 45 minutes from CSM separation to touchdown, may be obtained by using the 48 hours postlanding battery supply. (See reference 1.)

The 45 minutes reentry flight time includes a minimum of 5 minutes between CSM separation and the 400000 feet altitude reentry interface. The separation distance at the reentry interface for the same CSM separation maneuver can be increased by lengthening the time between CSM separation and the reentry interface. The flight time is 33 minutes between the reentry interface and drogue chute deployment, which occurs at 25000 feet altitude. Normal drogue and main chute time is about 7 minutes.

Line a in figure 1 is the overshoot boundary based on a rolling reentry at 14 deg/sec, which results in a 45 minutes flight time from CSM separation to touchdown when the nominal AS-204A trim aerodynamics supplied in reference 3 are used. The reentry range from the reentry interface to touchdown along line a is approximately 6500 n. mi.

Line b in figure 1 is based on nominal trim aerodynamics from reference 3. The CM lift vector is assumed up throughout the entry phase. The flight time from CSM separation to touchdown in 45 minutes.

For the region above line a in figure 1, it is always recommended that a initial lift vector down attitude be held until some threshold load factor is reached then a rolling reentry mode should be used. The exact load factor threshold has not been determined at this time but this procedure of lift vector down initially reduced the down range dispersions for recovery and slightly improves the CM capture capability within this region.

4.1.2 <u>One thousand and four hundred seconds entry time overshoot corridors</u>. - In a crew-suited mode of reentry, a Block I environmental and control system (ECS) failure of the type listed in references 1 and 2 may cause the reentry time to be restricted to the 1400 seconds (23.3 minutes from 400000 feet altitude to touchdown. There is presently no firm commitment by the Apollo Spacecraft Program Office on this 1400 seconds reentry restriction. During low power periods, transients heat loads can cause venting of sufficient water from the glycol water boiler to obstruct the common boiler venting duct used by both the crew ECS and the equipment glycol ECS. Reference 2 has shown that if a shirtsleeve mode of reentry is acceptable, the 1400 seconds is no longer a restriction. This, of course, assumes that the crew has sufficient time to de-suit prior the deorbit maneuver.

Line c in figure 1 is an overshoot boundary for a rolling reentry, a 1400 seconds flight time, and the non-standard atmosphere shown in figure 5. This line represents a conservative upper boundary for a crew-suited reentry with an ECS failure of the type listed in reference 2, provided a 0 lift reentry is used.

Three test case reentries along line c have been sent to the Crew System Space Medicine Branch for analysis of the load factor time histories and the corresponding high suit temperatures and humidity conditions that could exist if the ECS failure given in references 1 and 2 occurred. At the present time, the Crew System Space Medicine Branch is also determining what factors, based on previous experimental data, will require the use of line c as an overshoot boundary.

Line d in figure 1 shows what effect lift-vector-down from 400000 feet altitude to .2 g and, then, $\frac{1}{2}$ lifting coefficient (C_L) until 25000 feet altitude will do to the 1400 seconds overshoot boundary. The $\frac{1}{2}$ C_L trajectory is presently used for targeting purposes inside the available maneuver envelope. Line e is figure 1 is the same as d except that line e is based on the non-standard atmosphere. Line f in figure 1, may be compared with line e, since after .2 g they differ only in their lift vector attitudes.

Line g shows an overshoot boundary based on a 1400 seconds flight time for a constant lift-to-drag ratio of .4 and the non-standard atmosphere shown in figure 5.

4.1.3 Ten g entry undershoot corridors. - Line h in figure 1 represents the 10 g undershoot boundary based on the nominal trim aerodynamics presented in reference 3 and a reentry CM roll rate of 14 deg/sec throughout the flight.

Line i shows the 10 g undershoot boundary based upon a lift-vector-up attitude throughout a reentry with the nominal CM trim aerodynamics given in reference 3.

Although figure 1 is based on the CM weight presented in reference 3, ±500 pounds variation in the reentry weight does no noticeable effect the overshoot boundaries. The two points marked service module service propulsion system (SMSPS) and SMRCS in figure 1 are the selected velocities and flight-path angles used in the AS-204A reference trajectory presented in reference 5. These two points are determined by selection of the ΔV used for deorbit and the spacecraft orbital characteristics and position in the orbit at the time of the retrofire.

4.1.4 Ballistic or rolling entry for various entry times. - Figure 3 shows a series of velocities and flight-path angles that result in constant reentry flight time when a rolling reentry mode is used.

The lines are based upon a lift-vector-down attitude until .2 g and, then, a roll rate of 14 deg/sec until 25000 feet altitude. A nominal parachute time of 7 minutes is assumed. Therefore, the constant time lines shown in figure 3 includes the time from 400000 feet to touchdown.

4.1.5 Half-lift entry for various entry times. - Velocities and flight-path angles that result in constant reentry time lines in figure 4 are based upon a lift-vector-down attitude held until .2 g and, then, a $\frac{1}{2}$ C_L until 25000 feet altitude. Again a nominal parachute time of 7 minutes was included in the total time of flight from 400000 feet altitude.

4.2 Deorbit Target Conditions for Entry

4.2.1 Target conditions for a SMSPS deorbit maneuver. - Figure 2 represents the present available information for determining what velocities and flight-path angles are desirable for reentry from a SMSPS deorbit maneuver. Line b is the overshoot line and line h is the undershoot line, as already shown in figure 1. Between these two boundaries, a dashed line has been drawn which represents the most desirable velocities and flight-path angles for reentries from a SMSPS deorbit maneuver. This dashed line in figure 2 allows for the largest deorbit dispersions from the SMSPS deorbit maneuver and assures the CM capture and adequate ranging maneuverability.

The amount of fuel left at the end of the AS-204A mission is enough that during the SMSPS deorbit maneuver the pitch angle and the length of burn can be varied to obtain velocities and flight-path angles corresponding to the dashed line in figure 4.

4.2.2 Deorbit target conditions for a SMRCS deorbit maneuver. - Since the SMRCS is sensitive to the position of deorbit, the apogee of an elliptical orbit must be properly placed for effective recovery of the spacececraft in a planned recovery area. This placement of the apogee of the ellipse in the proper position for a SMRCS deorbit is done on the last SMSPS burn prior to deorbit to allow for a possible failure of the SMSPS on the last burn and still leave the SMRCS deorbit maneuver within the same recovery area.

The line marked NAA in figure 2 was the line previously used for determining a SMRCS fuel budget. By reserving enough SMRCS fuel to obtain a velocity and flight-path angle on the line marked NAA in figure 2, the SMRCS reentry maneuver could use the guidance and navigation reentry steering logic if only the SMSPS engine or subsystems were inoperative. If the reentry duration time was critical or the deorbit maneuver dispersion was large, a rolling reentry could be used after the SMRCS deorbit maneuver.

Line c in figure 2 is a rolling reentry with a non-standard atmosphere and the nominal trim aerodynamics presented in reference 3. This limit line is the present selected target boundary for SMRCS deorbit fuel budgeting from elliptical orbits.

Line c in figure 2 is for a rolling reentry with a standard atmosphere and the nominal trim aerodynamics.

4.3 Deorbit Velocity Requirements

4.3.1 Elliptical orbits. - Figures 6(a) through 6(d) present inertial velocity and flight-path angle at 400000 feet for various retrograde ΔV's applied at apogee for various ellipses. The SMRCS target boundary and 45 minutes overshoot boundary defined in figure 1 are also shown. The data clearly indicates the effect of perigee altitude on the minimum ΔV required to stay below the selected boundaries. For a given ΔV, changes in apogee altitude have a small effect on the flight-path angle displacement from the target boundary (line c, figure 1). The data presented in figure 13 show the minimum retrograde ΔV is not significantly different for apogees greater than 180 n. mi. This fact indicates that the apogee altitudes above 180 n. mi. would be based on operational considerations other than ΔV.

4.3.2 Circular orbits. - The two selected overshoot entry corridors and ΔV requirements for circular orbits are presented in figure 7. A comparison of the circular orbit data with the elliptical orbit data presented in figure 6 shows that a higher ΔV is required for circular orbits to stay below the selected target boundary.

4.4 Deorbit Attitude Requirements

Inertial velocity and flight-path angle at 400000 feet for various ΔV's and retro attitudes are presented in figure 11 for apogee deorbit from an 85/130 n. mi. elliptical orbit. This figure indicates that horizontal retro (β = 0 degrees) is optimum for the range of ΔV's within the capability of the SMRCS. Lowering the retro attitude below the velocity vector for a given retro ΔV increases the entry velocity and makes the entry flight-path angle shallower. This has an undesirable effect of raising the minimum ΔV requirement for the selected entry overshoot boundary.

4.5 Deorbit True Anomaly Requirements

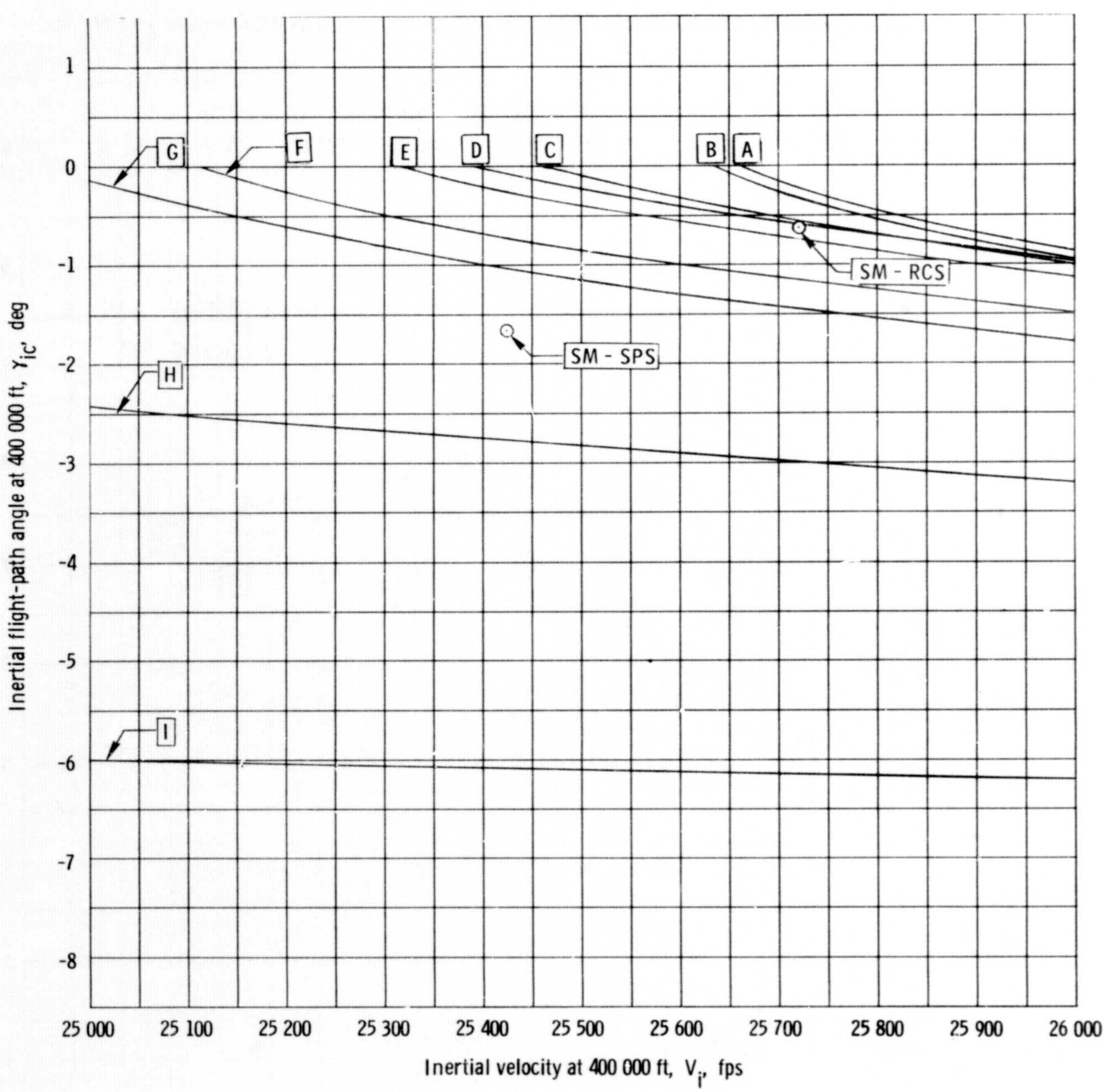
For a fixed ΔV of 100 ft/sec, figure 12 indicates the horizontal retro is still optimum for a low ΔV, even within a bank of ±60 degrees of true anomaly. For a given retro attitude the true anomaly of applied retro velocity had no effect on the entry velocity but had a pronounced effect on the entry flight-path angle. The optimum true anomaly varies with retro attitude and is near apogee (θ = 180 degrees) for near 0 deorbit attitudes.

4.6 SMRCS Propellant Requirements

SMRCS propellant requirements versus ΔV for various spacecraft weights are presented in figure 9(a) through 9(c) for the nominal and $\pm 3\sigma$ engine performances. A translation thrusting efficiency factor of 95% was included in the propellance weight requirements. The RCS propellant required varied from 2.4 lb/ft/sec for the 20000 pounds spacecraft to 4.8 lb/ft/sec for the 40000 pounds spacecraft.

4.7 Orbital Lifetime Requirements

In figure 10(a) to 10(c) the orbital lifetime is presented as a function of spacecraft weight for various perigee and apogee altitudes. The results, based on the selected atmosphere drag coefficient and tumbling area, indicate that, for a perigee altitude of 85 n. mi. and a desired 14 to 18 day lifetime, apogees near 280 n. mi. have to be selected for spacecraft weights between 20000 and 25000 pounds. For a selected orbital lifetime, apogee altitude is seen to decrease with an increase in spacecraft weight.



A Rolling reentry (14°/sec) overshoot boundary. Total time from CS M separation to touchdown is 45 minutes.

B L/D = .3431 overshoot boundary, lift vector up from 400 K ft altitude to 25K ft altitude. Total time from CSM separation to touchdown is 45 minutes.

C Rolling reentry (14°/sec) overshoot boundary with atmospheric density deviated as shown in Figure 5. Total time from 400K ft altitude to touchdown is 1400 seconds.

D L/D = .3431 overshoot boundary, lift vector down until .2g and then 1/2 lift until 25K ft altitude. Total time from 400K ft altitude to touchdown is 1400 seconds.

E L/D = .3431 overshoot boundary, lift vector down until .2g and then 1/2 lift until 25K ft altitude with atmospheric density deviation. Total time from 400K ft altitude to touch down is 1400 seconds.

F L/D = .3431 overshoot boundary, lift vector down until .2g and then 0° bank angle until 25K ft altitude with atmospheric density deviation. Total time from 400K ft altitude to touchdown is 1400 seconds.

G L/D = .4 (constant) overshoot boundary, lift vector down until .2g and then lift vector up until 25K ft altitude with atmospheric density deviation. Total time from 400K ft altitude to touchdown is 1400 seconds.

H L/D = .3431 rolling reentry (14°/sec) 10g undershoot boundary. Lift vector down to .2g then rolling reentry to 25K ft altitude with a maximum load factor of 10g's during reentry.

I L/D = .3431 10g undershoot boundary, lift vector up until 25 K ft altitude with a maximum load factor of 10g's during reentry.

Figure 1. - An operational CM reentry corridor for Block I vehicles for Apollo earth orbit missions.

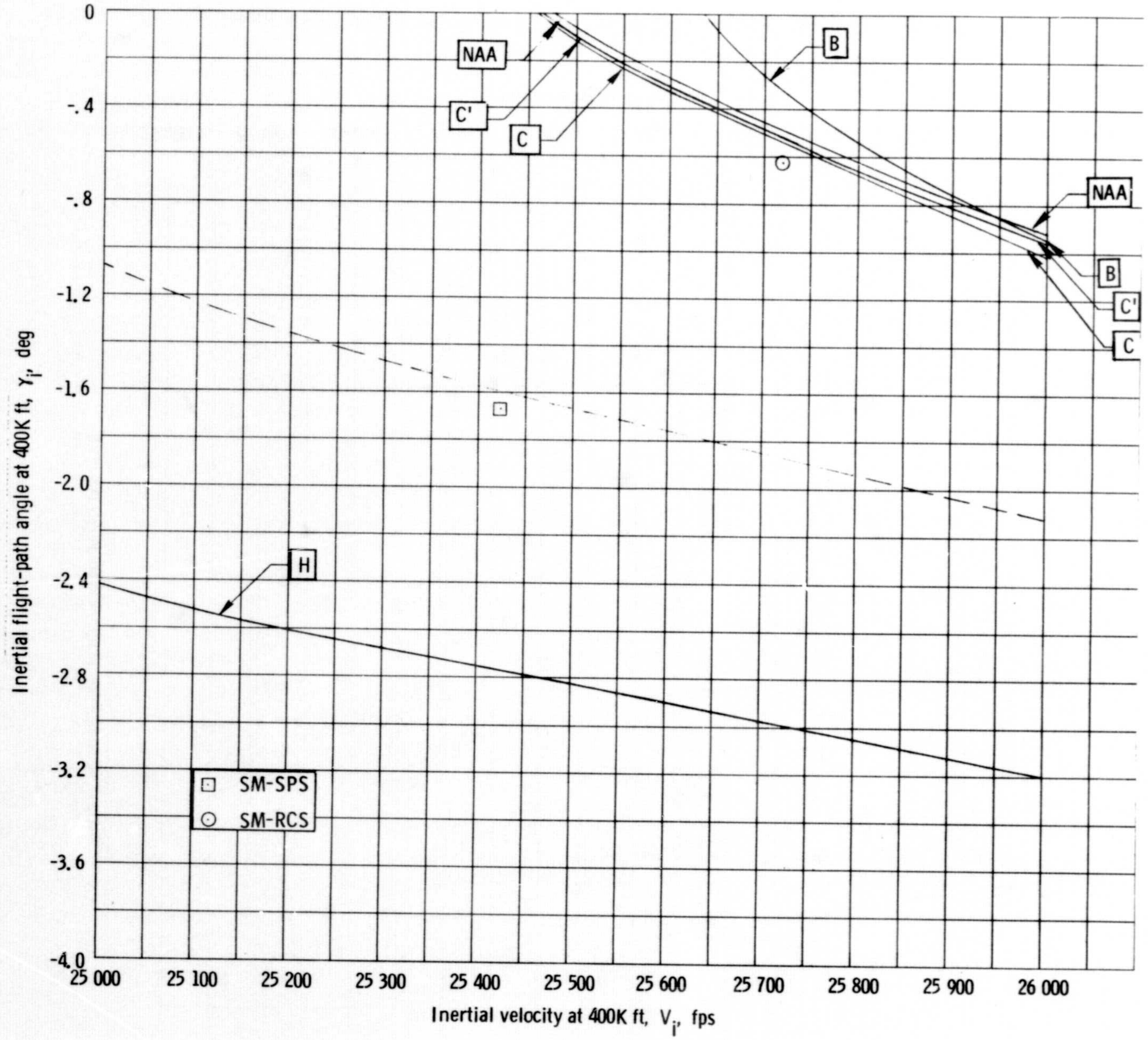


Figure 2. - An operational CM reentry corridor for Block 1 vehicles for Apollo earth orbit missions.

B — L/D = .3431 overshoot boundary, lift vector up from 400K altitude to 25K ft altitude. Total time from CSM separation to touchdown is 45 minutes.

C — Rolling reentry (14°/sec) overshoot boundary with atmospheric density deviated as shown in Figure 5. Total time from 400K ft altitude to touchdown is 1400 seconds.

C' — Rolling reentry (14°/sec) overshoot boundary with standard atmosphere. Total time from 400K ft altitude to touchdown is 1400 seconds.

NAA — Overshoot boundary defined by N. A. A. See reference 4.

H — L/D = .3431 rolling reentry (14°/sec), 10g undershoot boundary. Lift vector down to .2g then rolling reentry to 25K ft altitude with a maximum load factor of 10g during reentry.

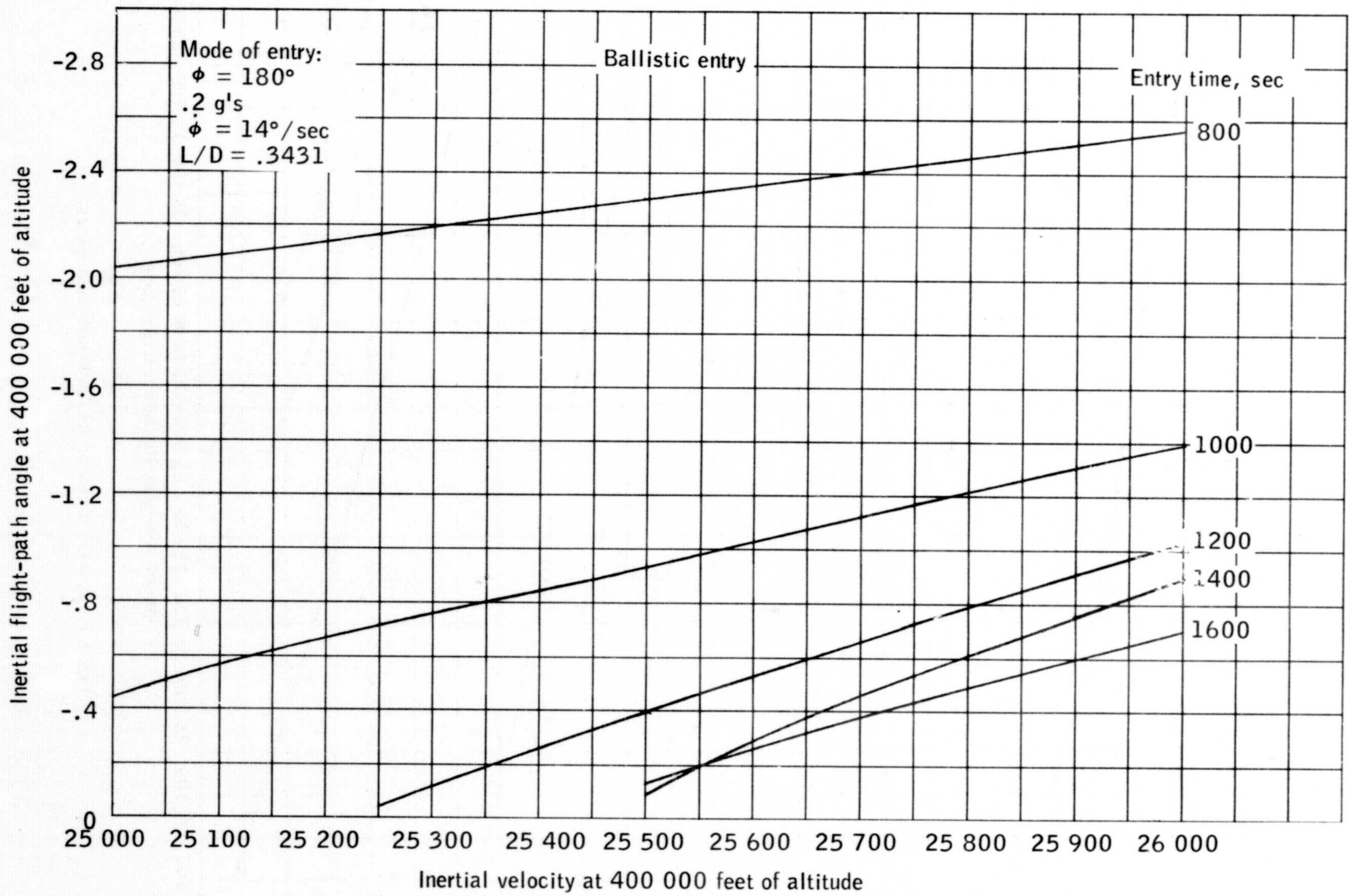


Figure 3.- Operational corridor for Apollo earth orbit entries.

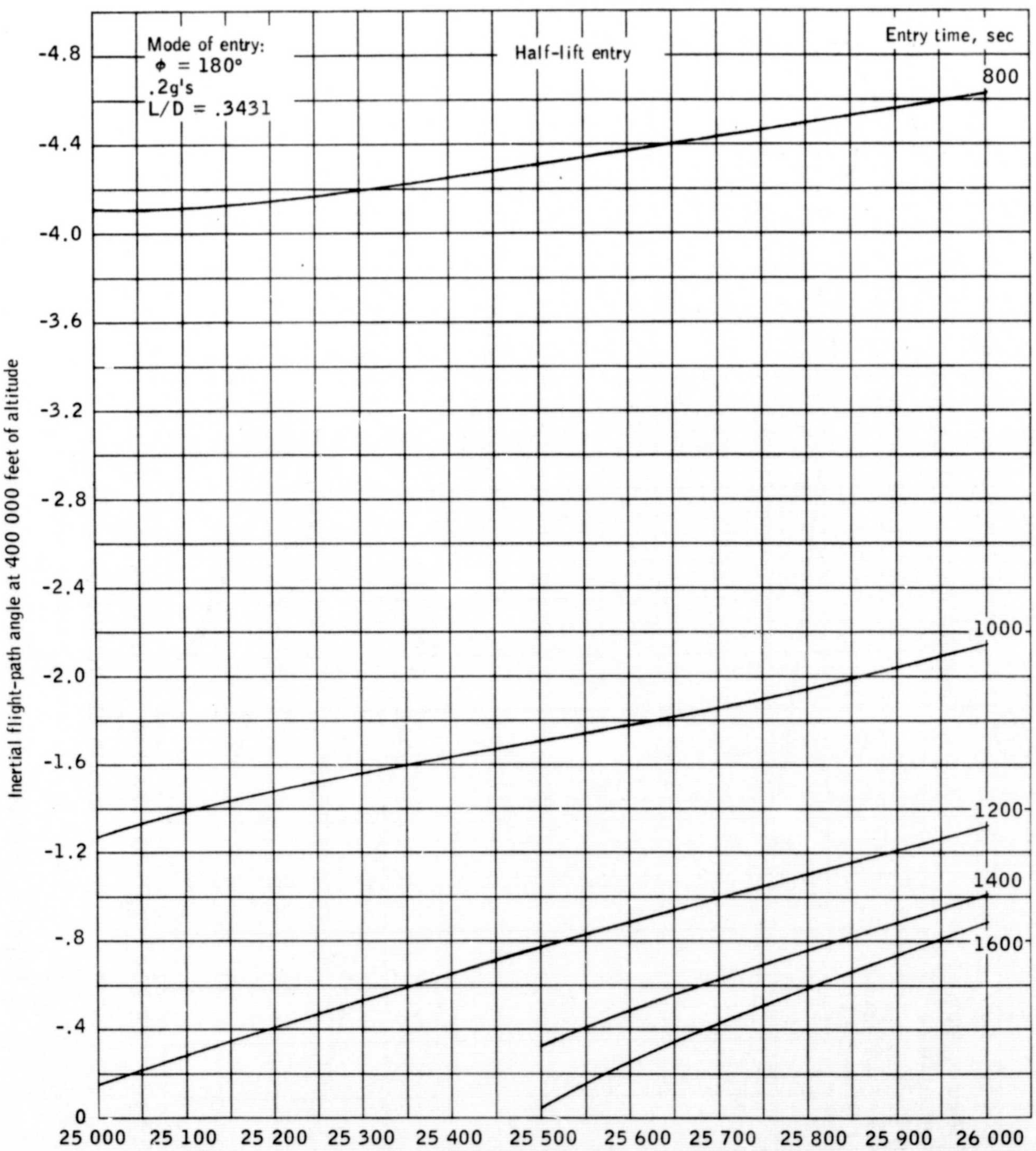


Figure 4.- Operational corridor for Apollo earth orbit entries.

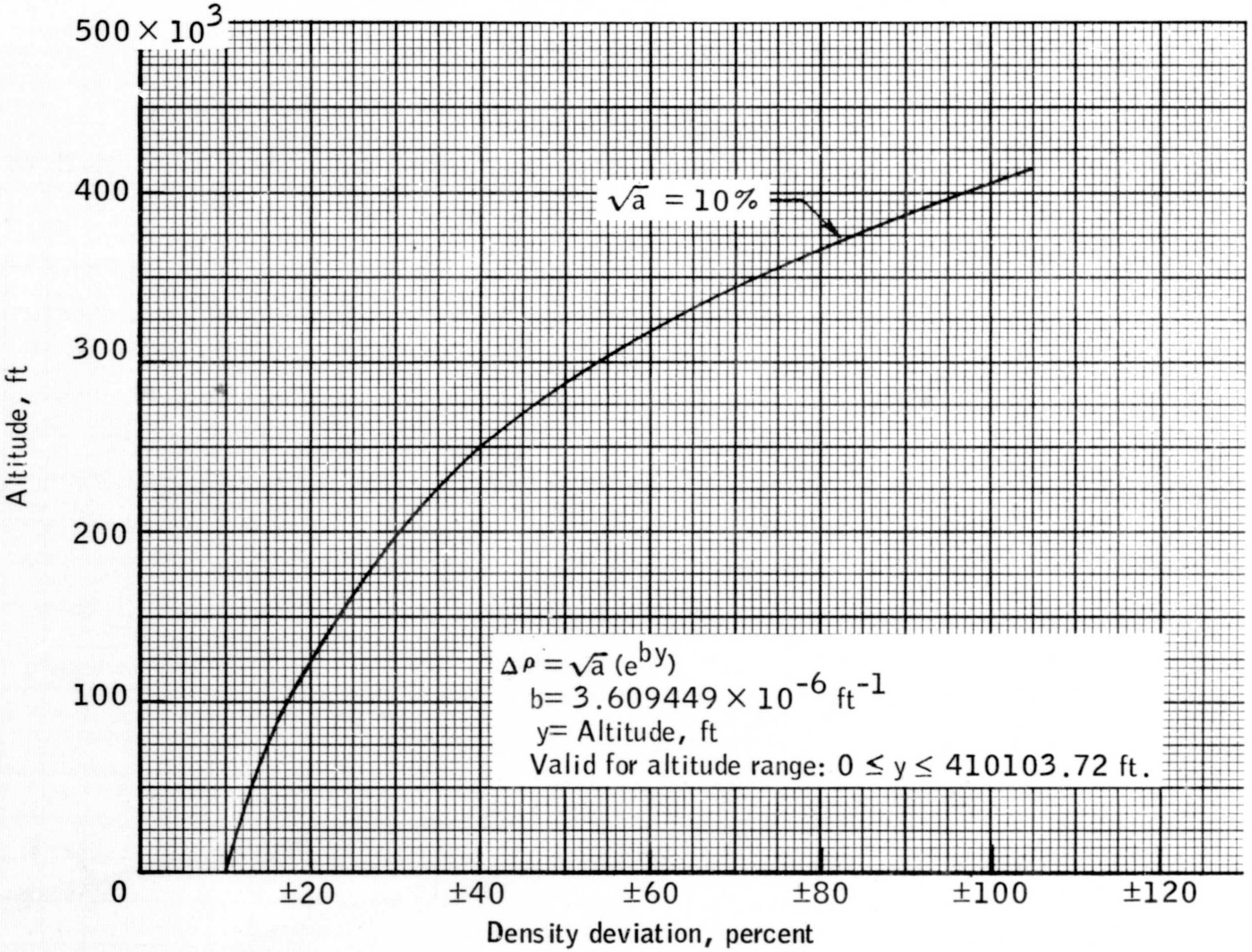


Figure 5.- Altitude versus density deviation.

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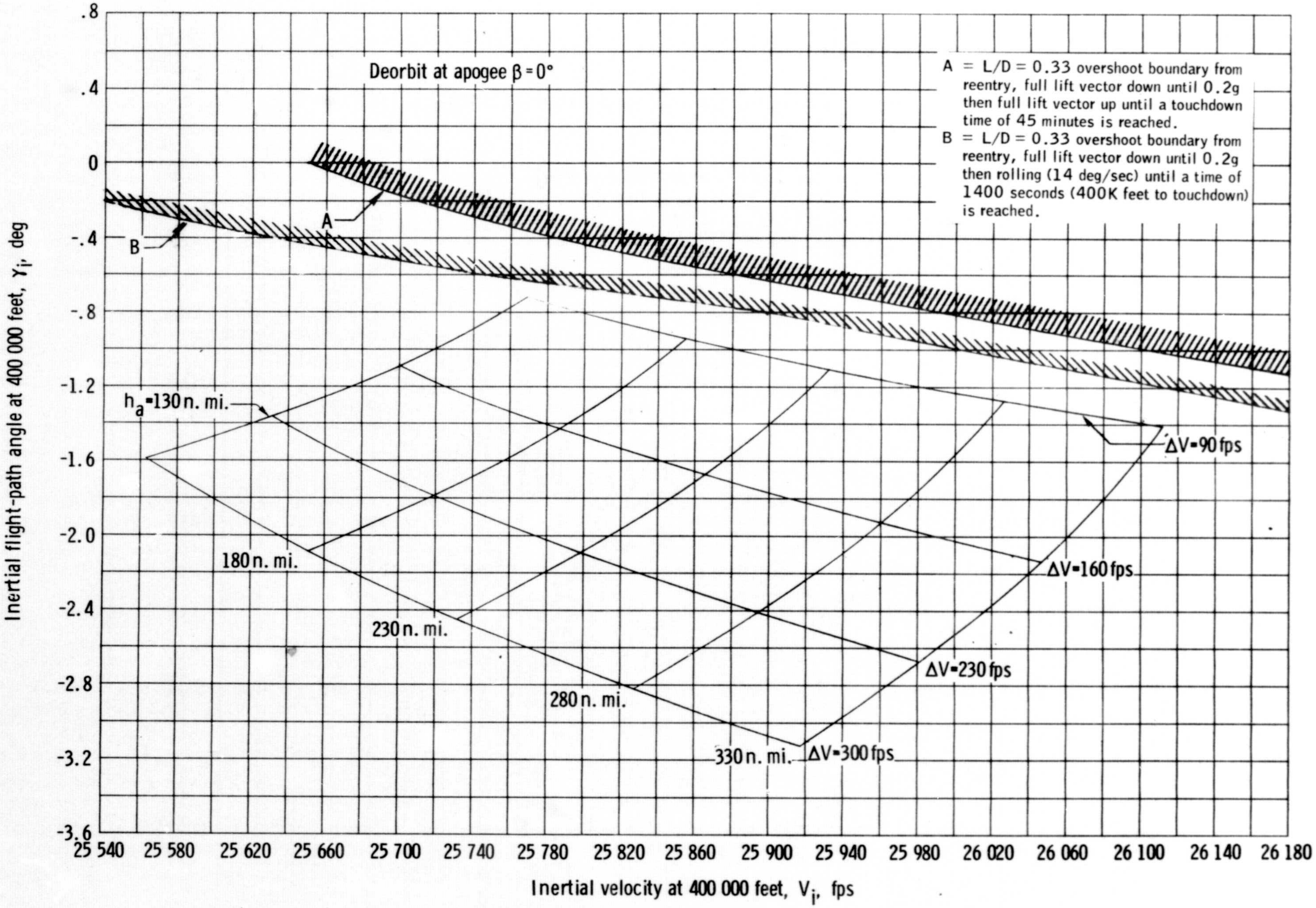
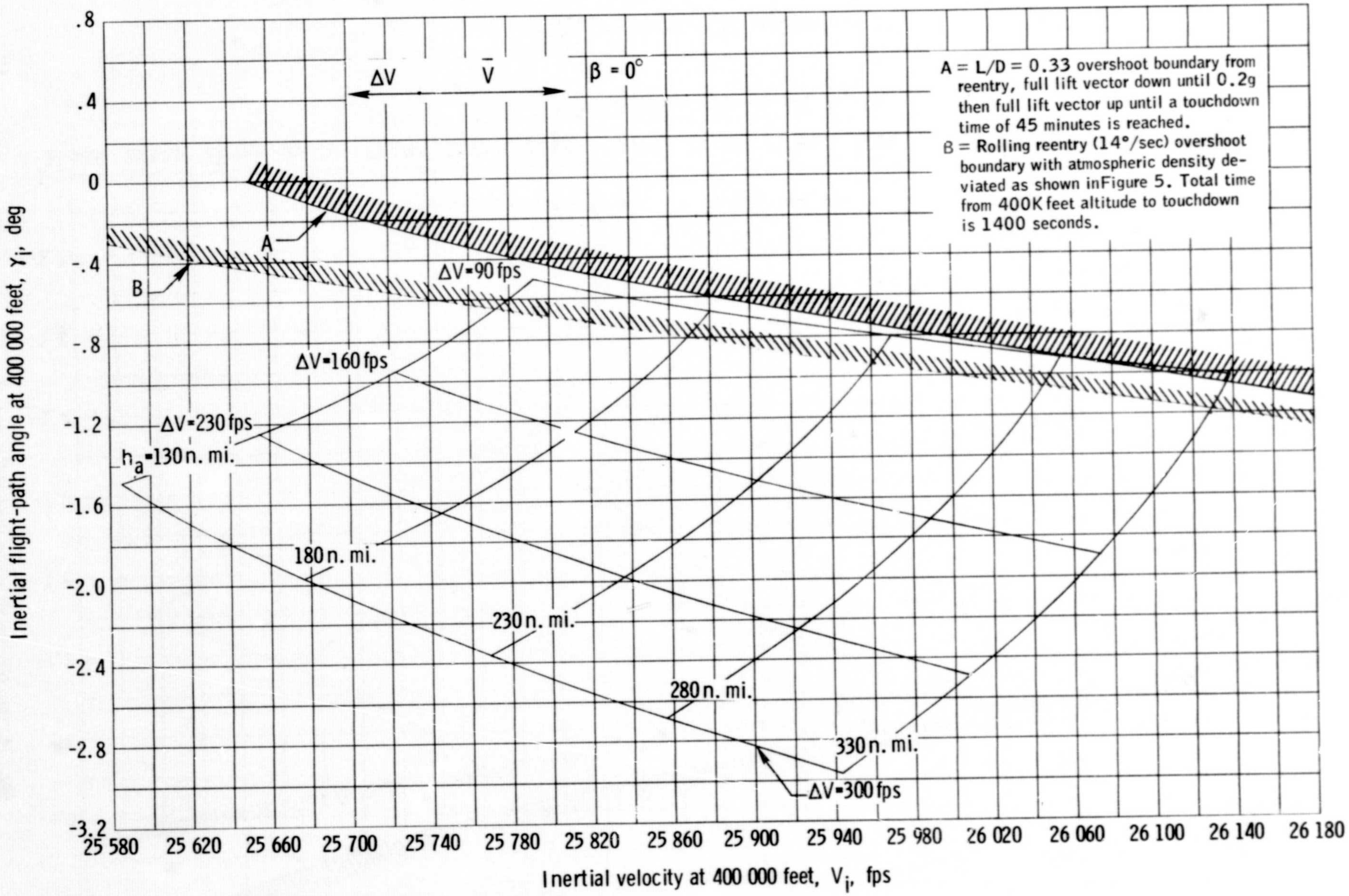


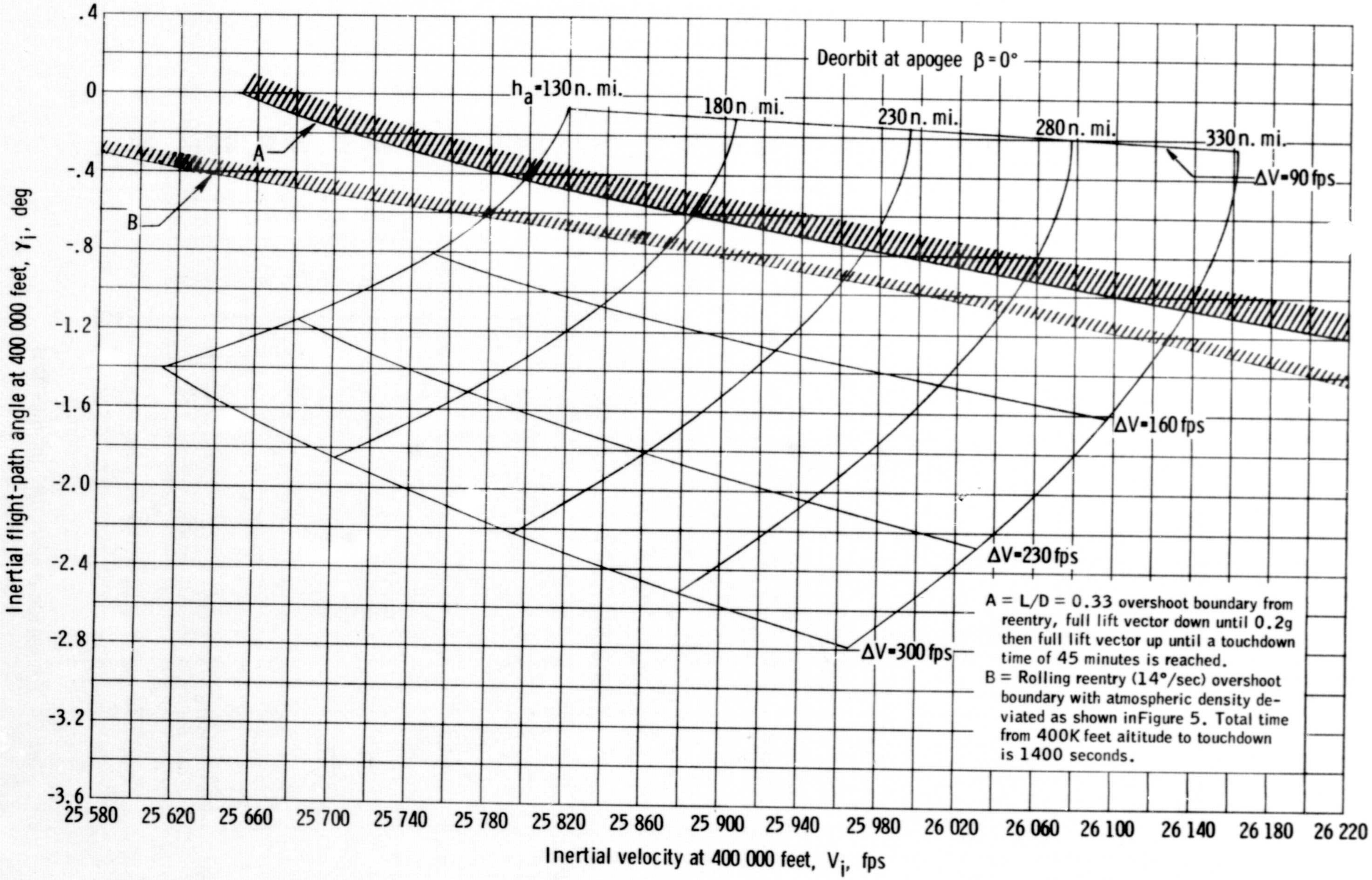
(a) Perigee altitude = 85 nautical miles.

Figure 6. - Entry corridor and ΔV requirements for elliptical orbits.



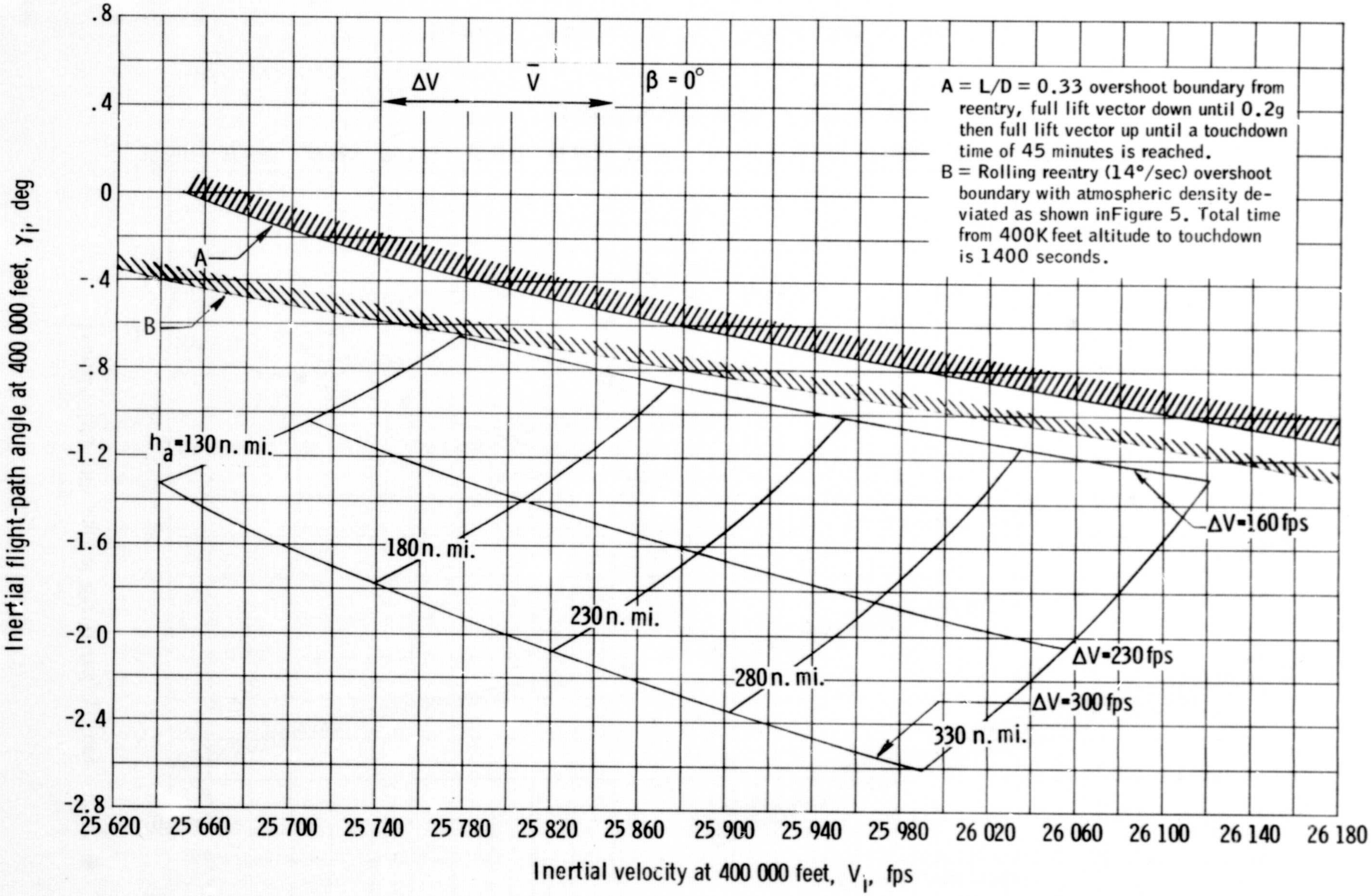
(b) Perigee altitude = 100 nautical miles.

Figure 6. - Continued.



(c) Perigee altitude = 115 nautical miles.

Figure 6. - Continued.



(d) Perigee altitude = 130 nautical miles.

Figure 6. - Concluded.

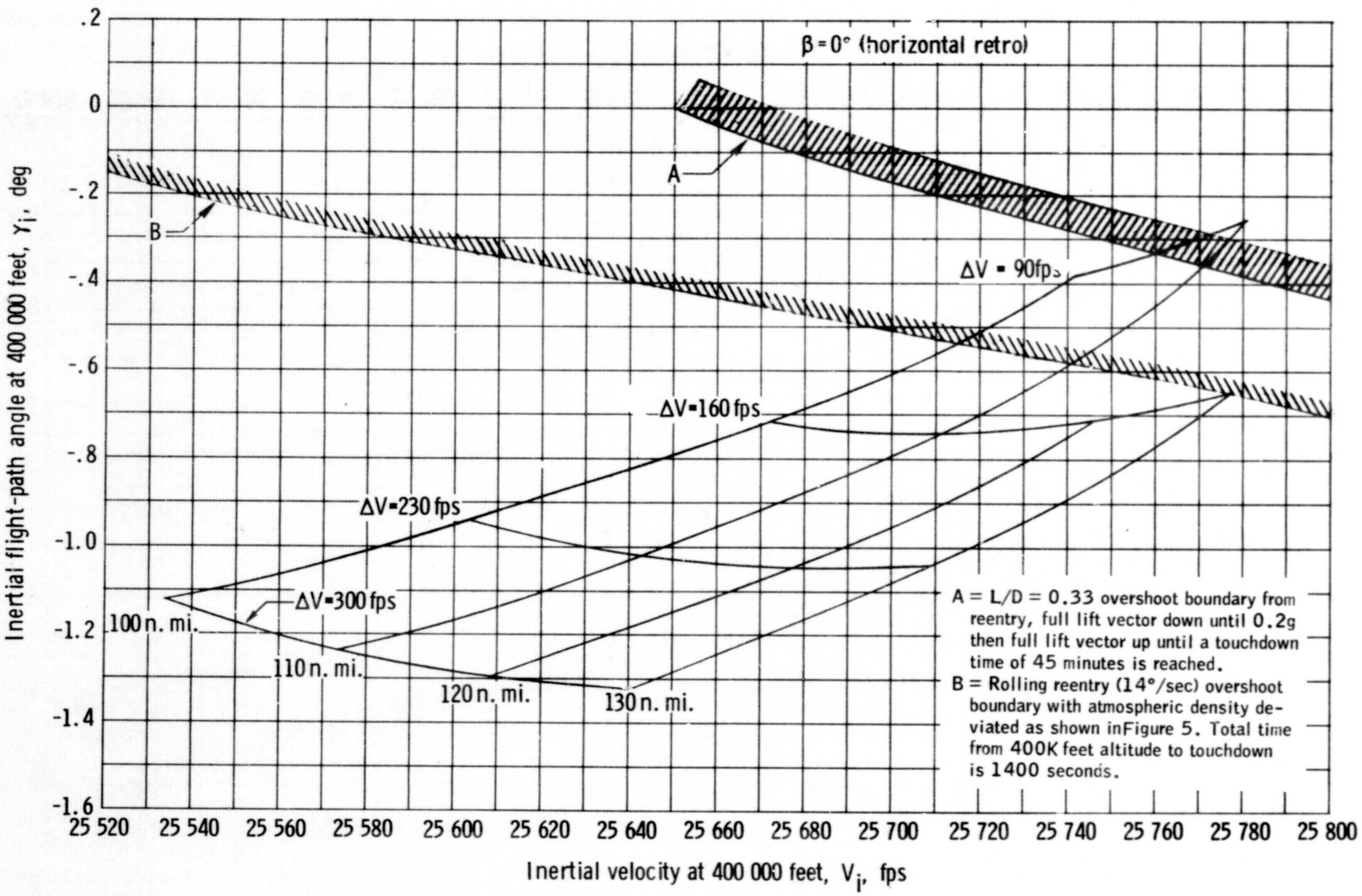


Figure 7. - Entry corridor and ΔV requirements for circular orbits.

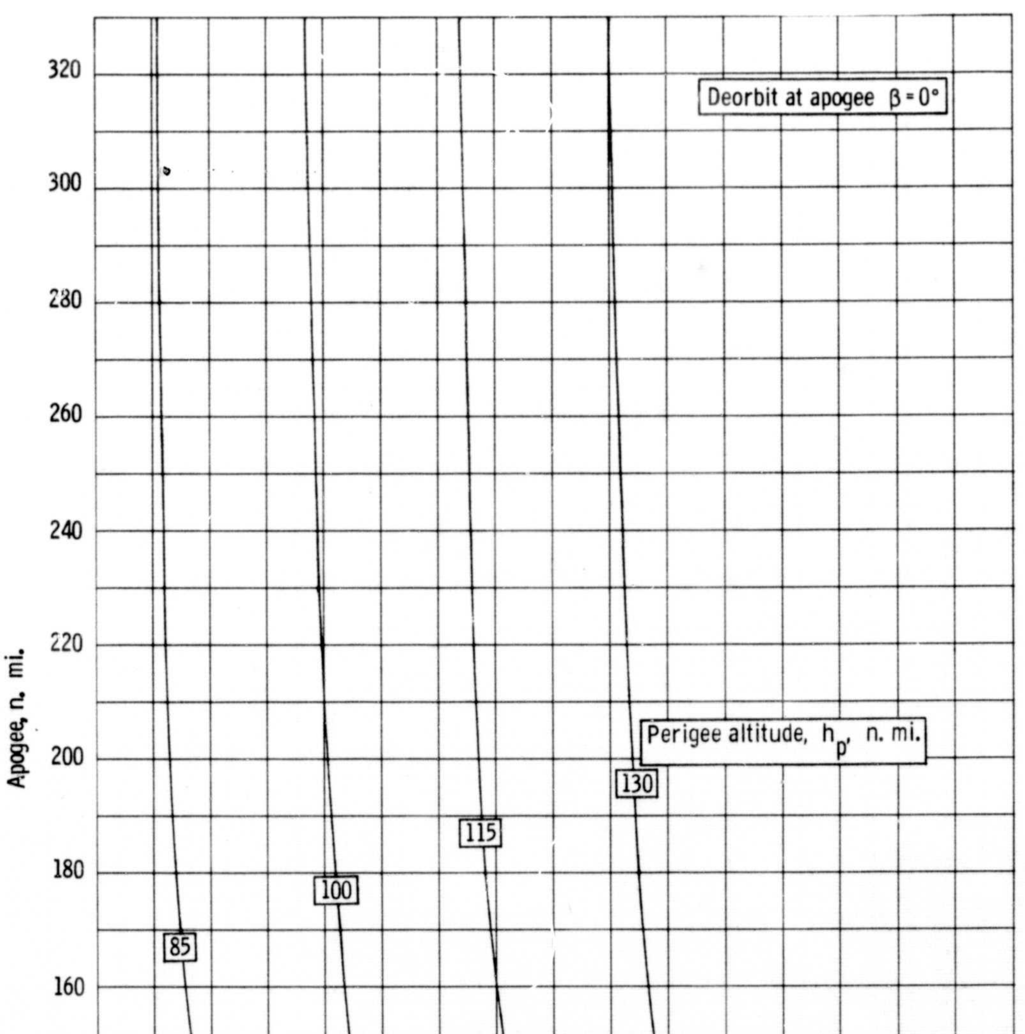
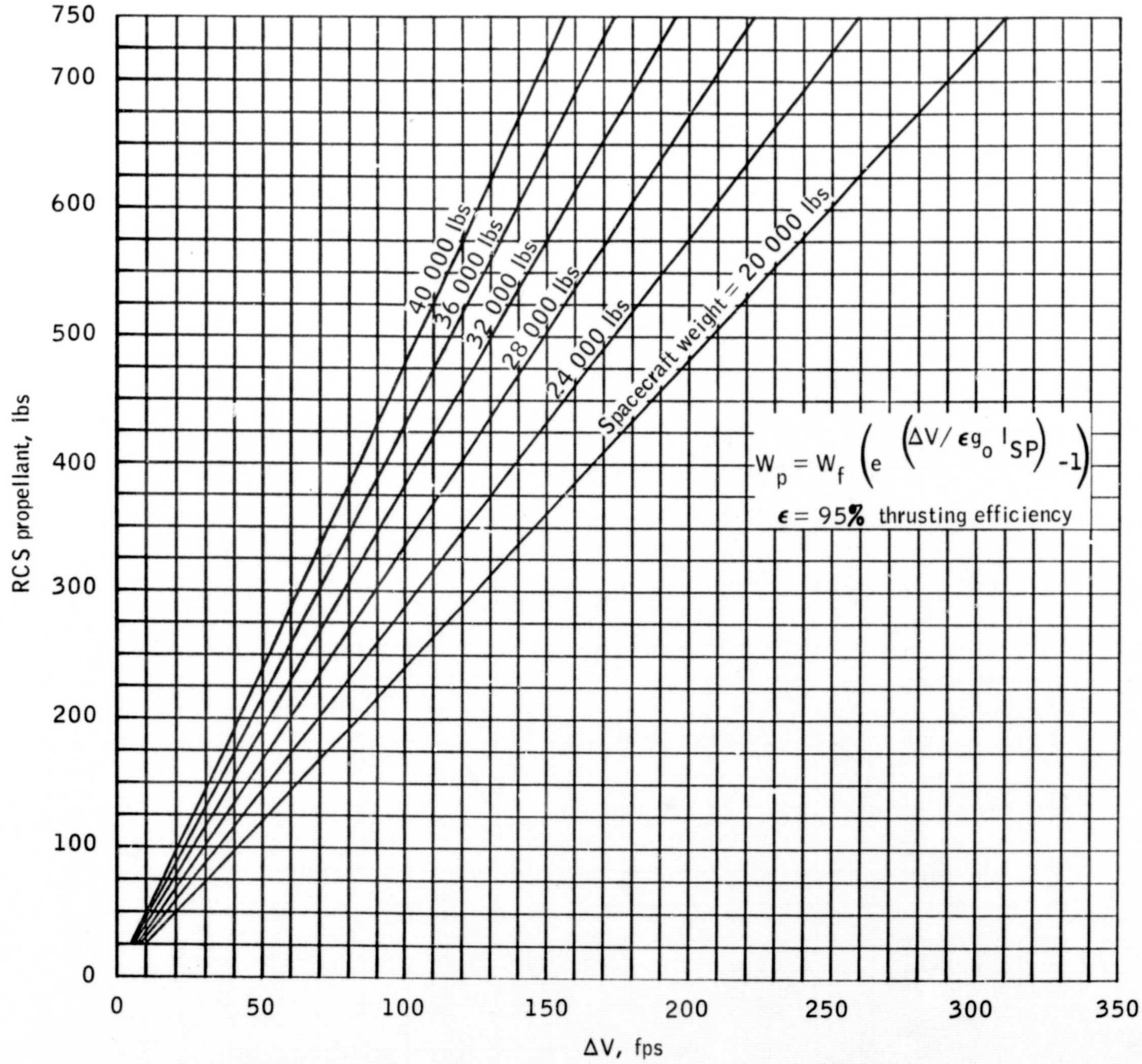
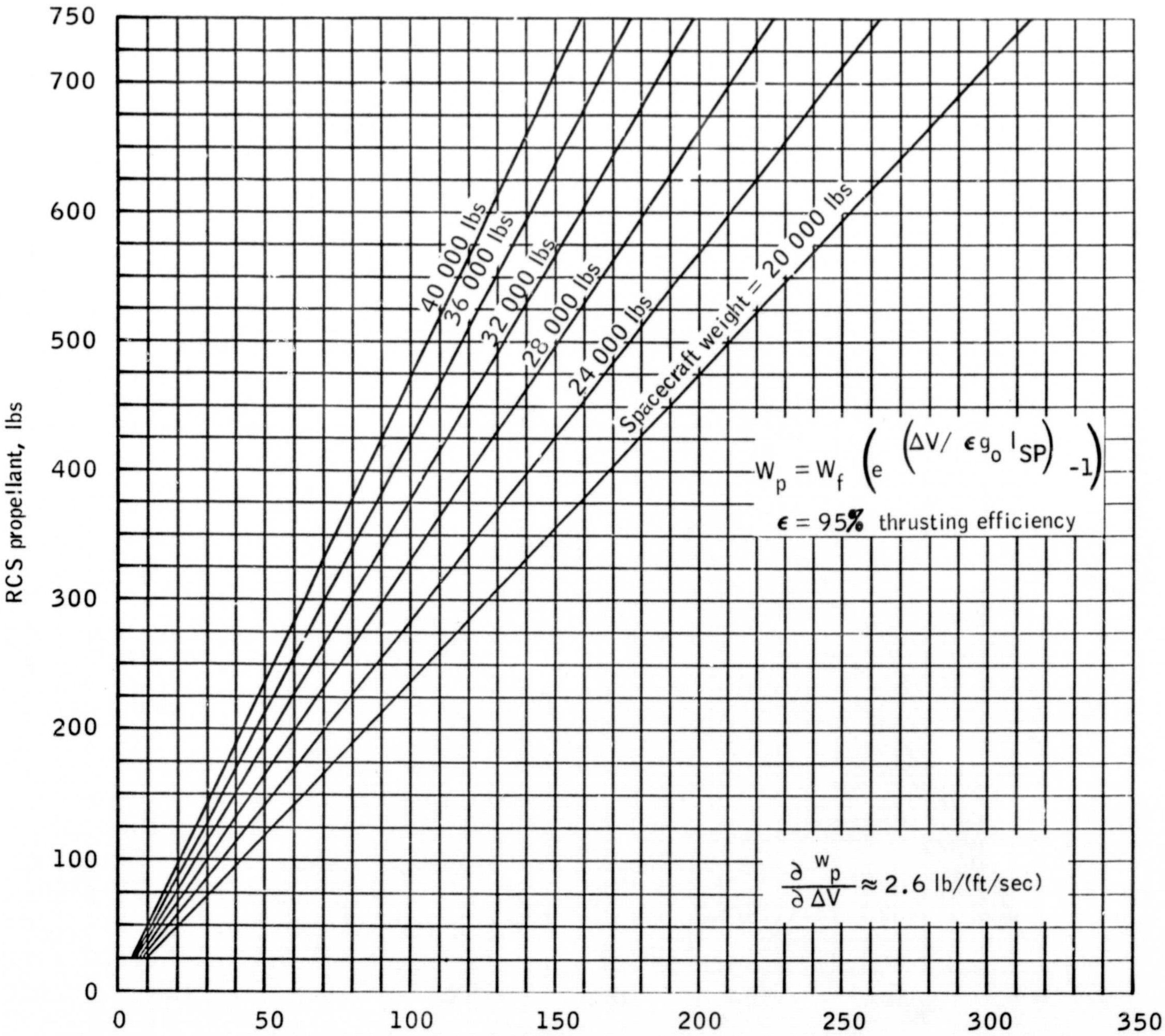


Figure 8. - Minimum retrograde ΔV's for near earth orbits to achieve an entry time of 1400 seconds (based on corridor line Figure 1.).



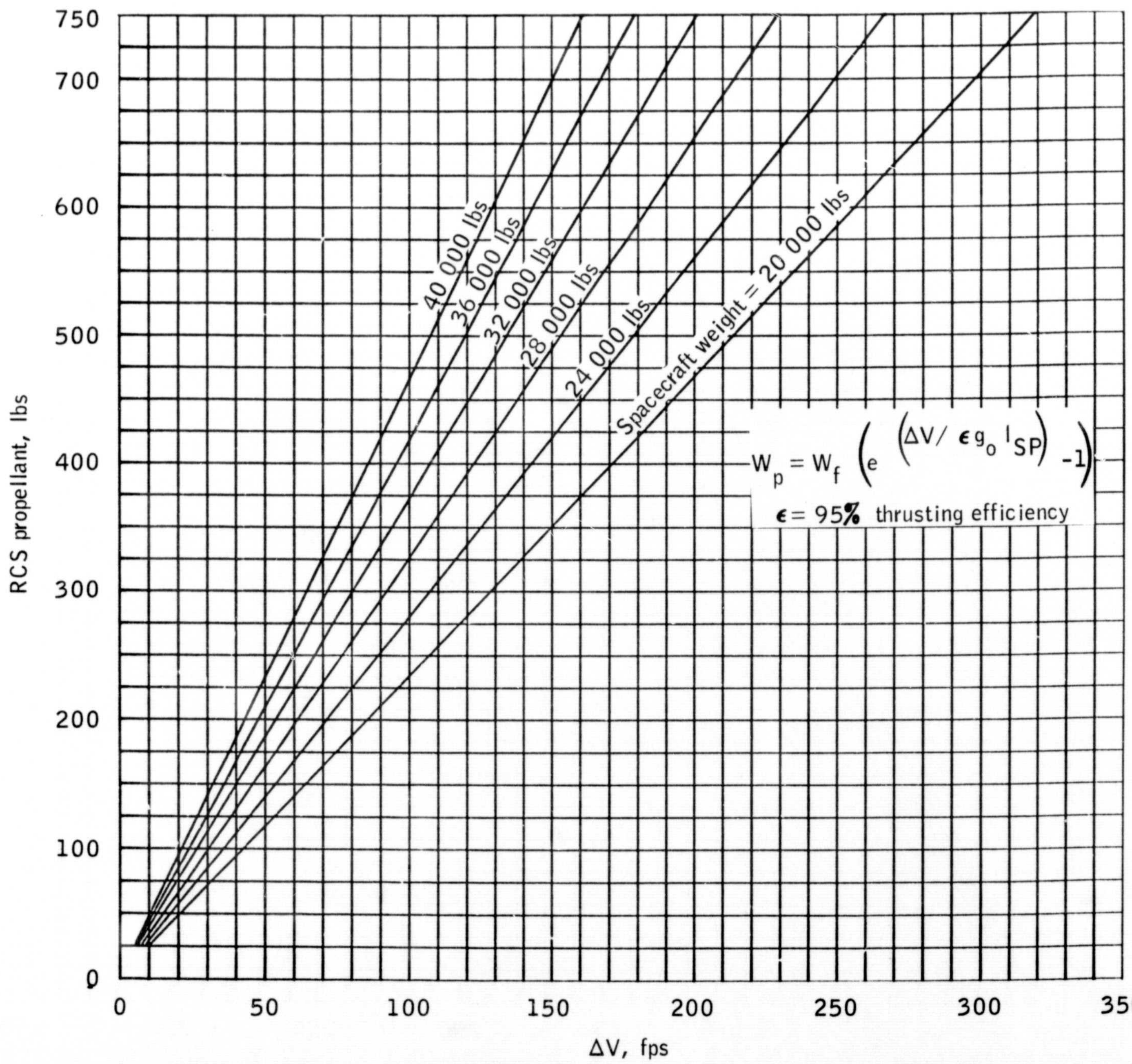
(a) -3σ engine performance.

Figure 9.- Propellant weight versus ΔV for various spacecraft weights.



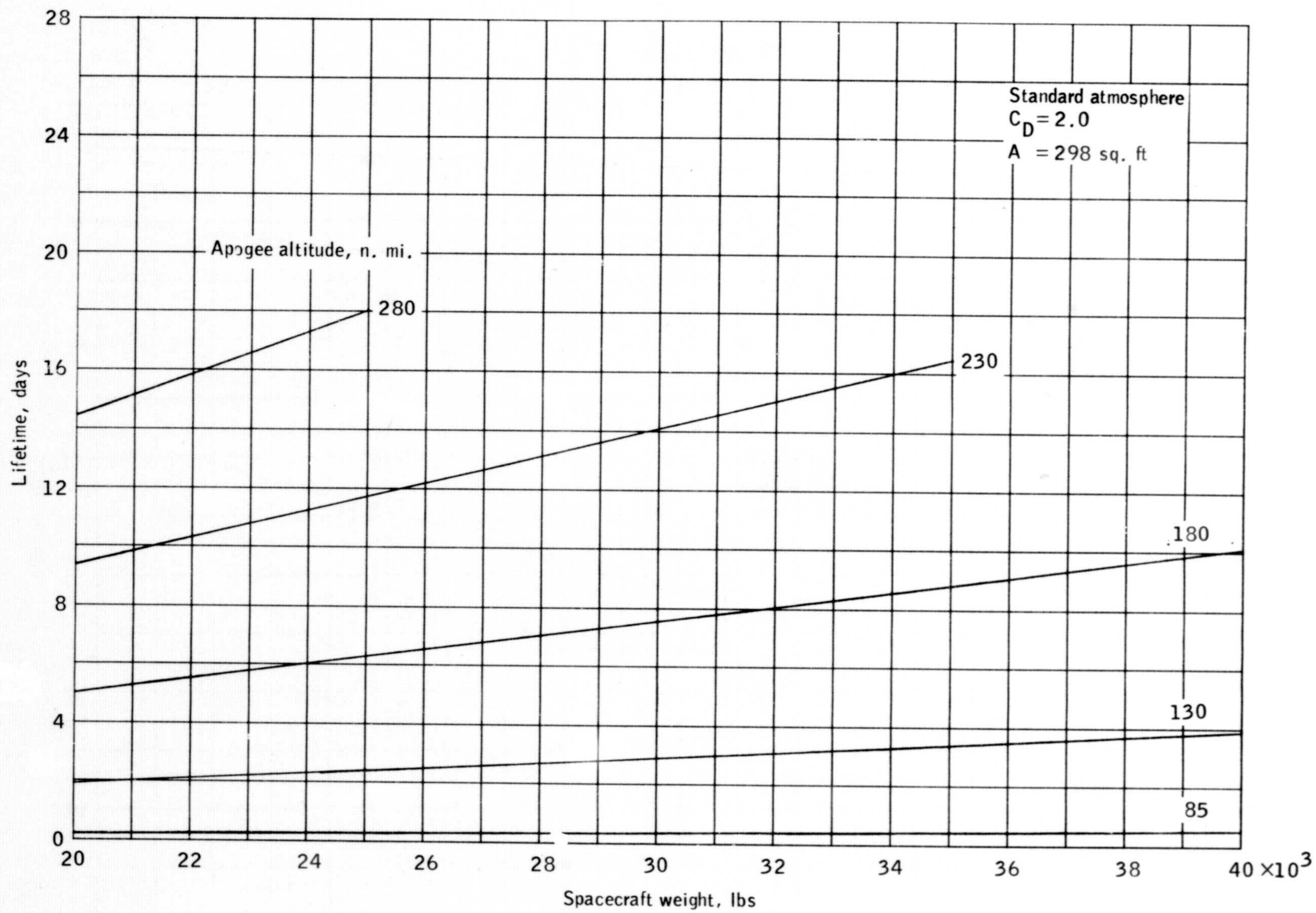
(b) Nominal engine performance.

Figure 9.- Continued.



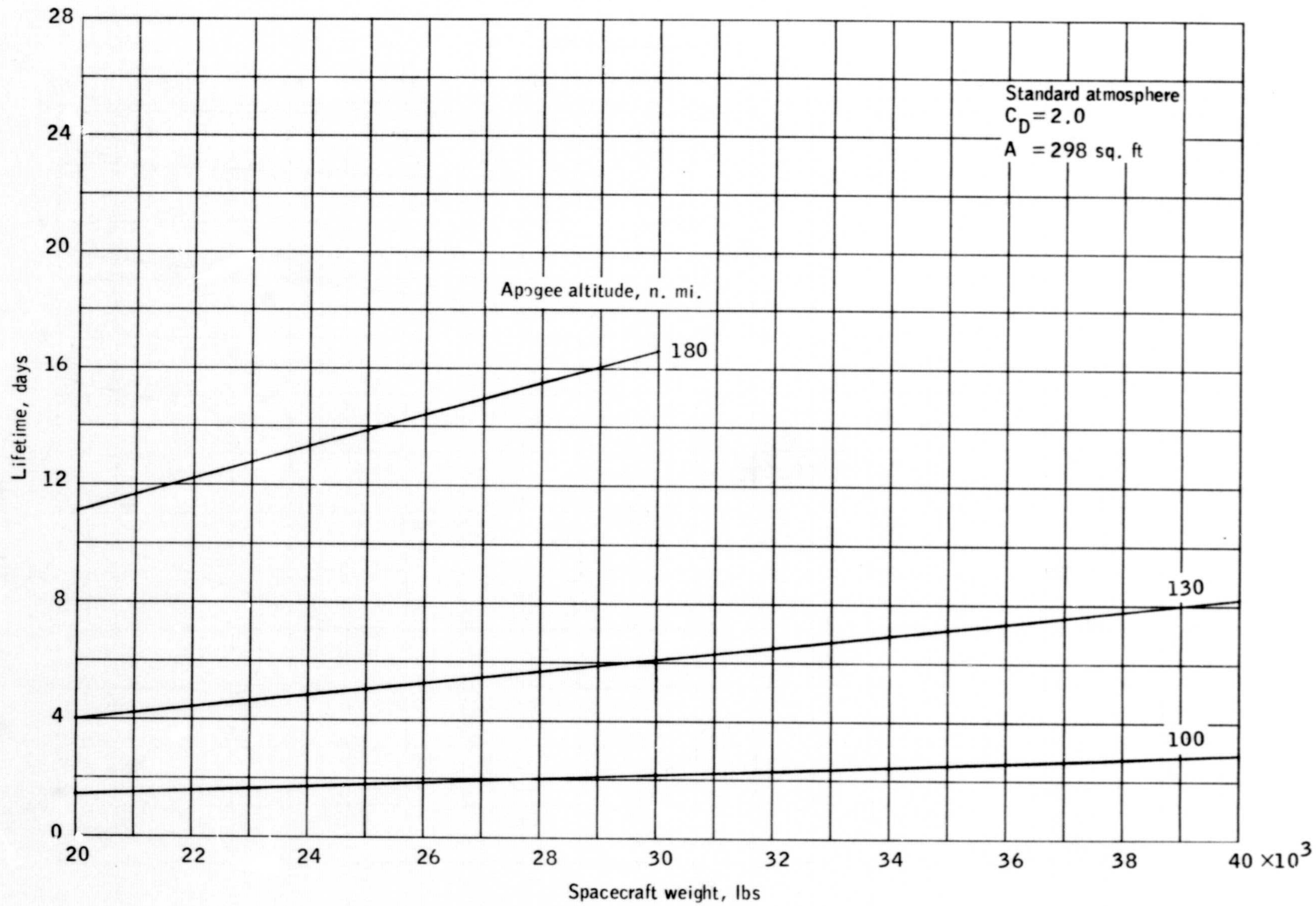
(c) $+3\sigma$ engine performance.

Figure 9.- Concluded.



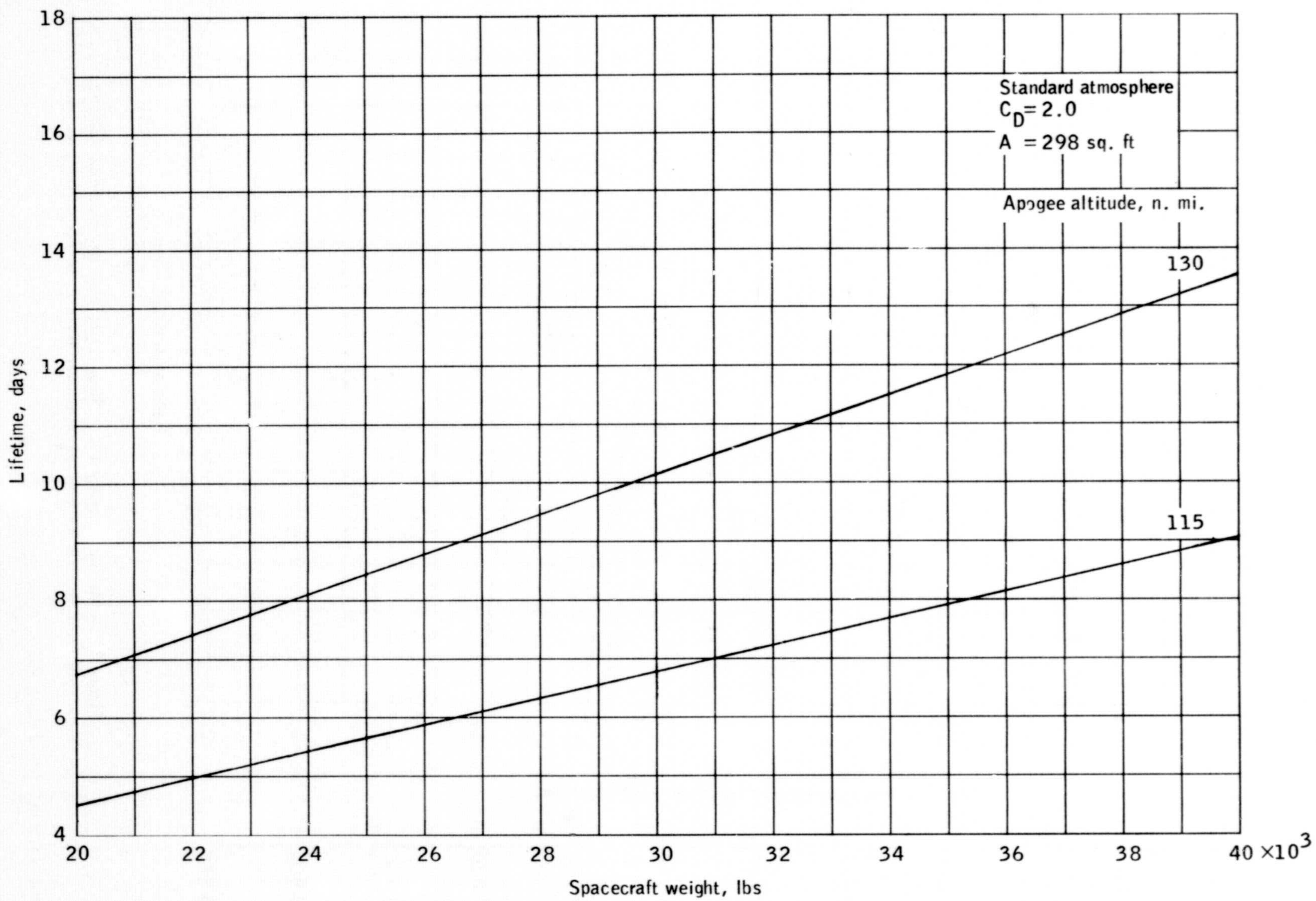
(a) Perigee altitude = 85 nautical miles.

Figure 10.- Lifetime of near earth orbits for various spacecraft weights.



(b) Perigee altitude = 100 nautical miles.

Figure 10.- Continued.



(c) Perigee altitude = 115 nautical miles.

Figure 10.- Concluded.

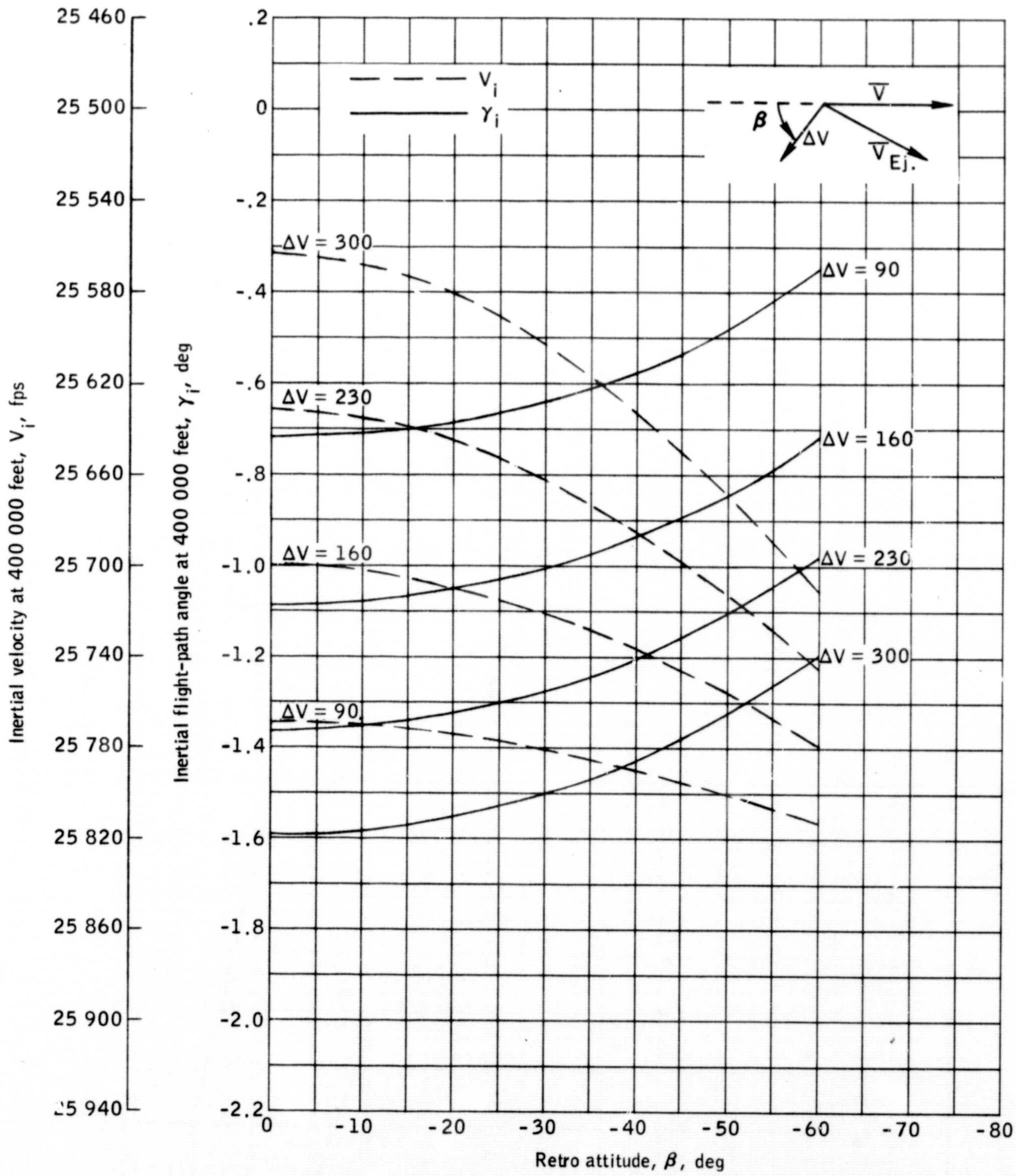


Figure 11.- Inertial velocity and inertial flight-path angle at 400 000 feet as a function of retro attitude for various retrograde ΔV's in an 85/130 n. mi. elliptical orbit.

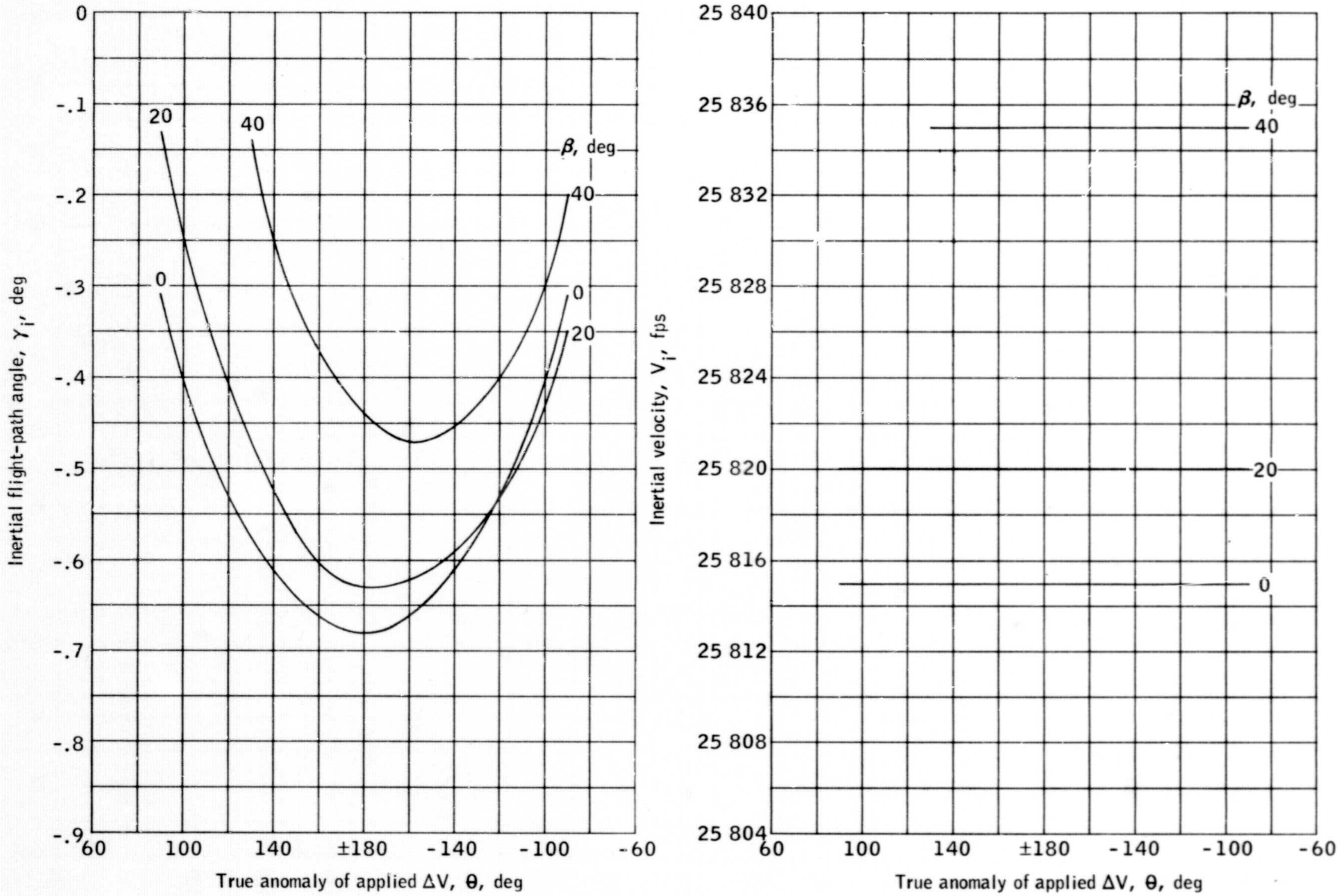
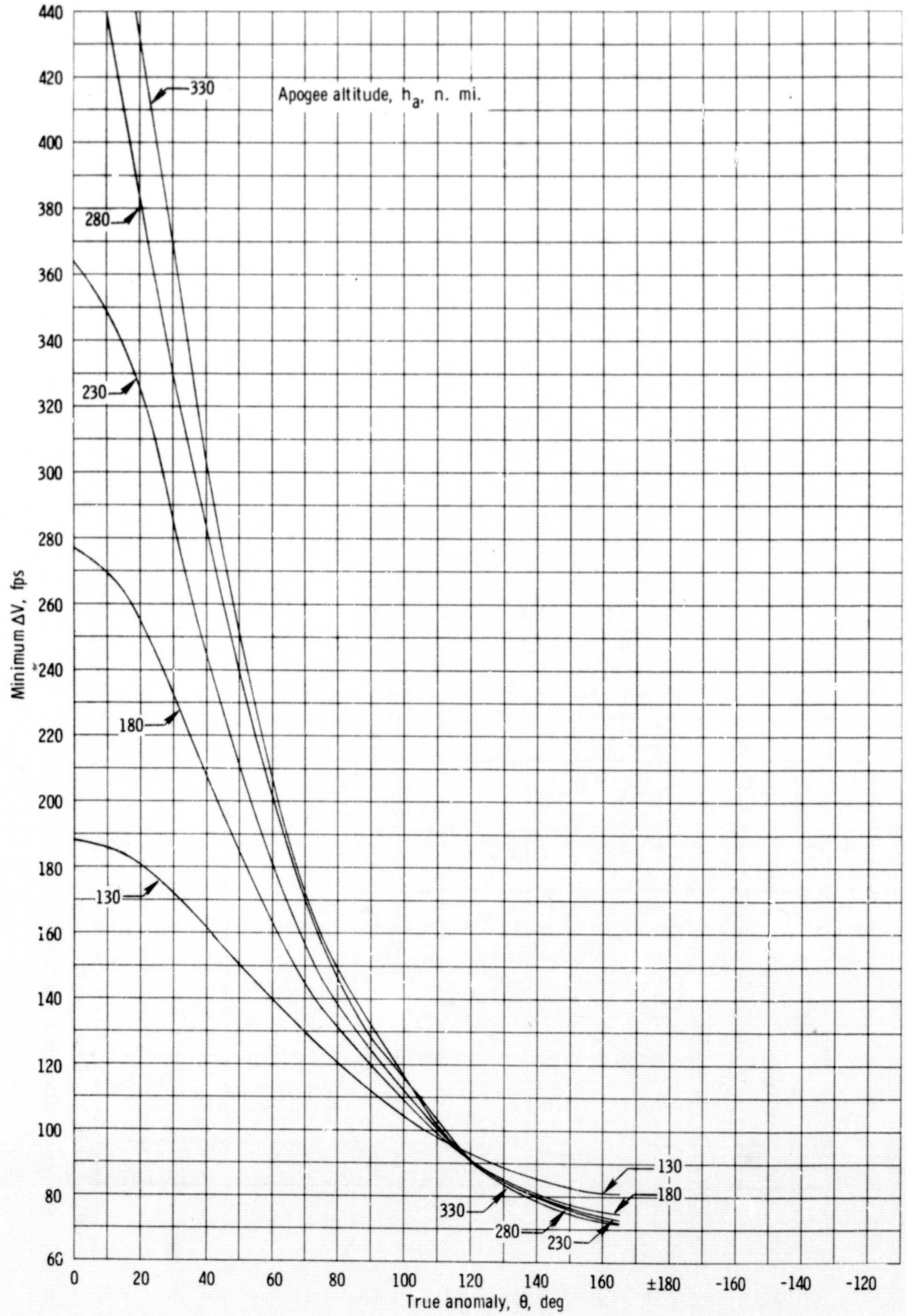
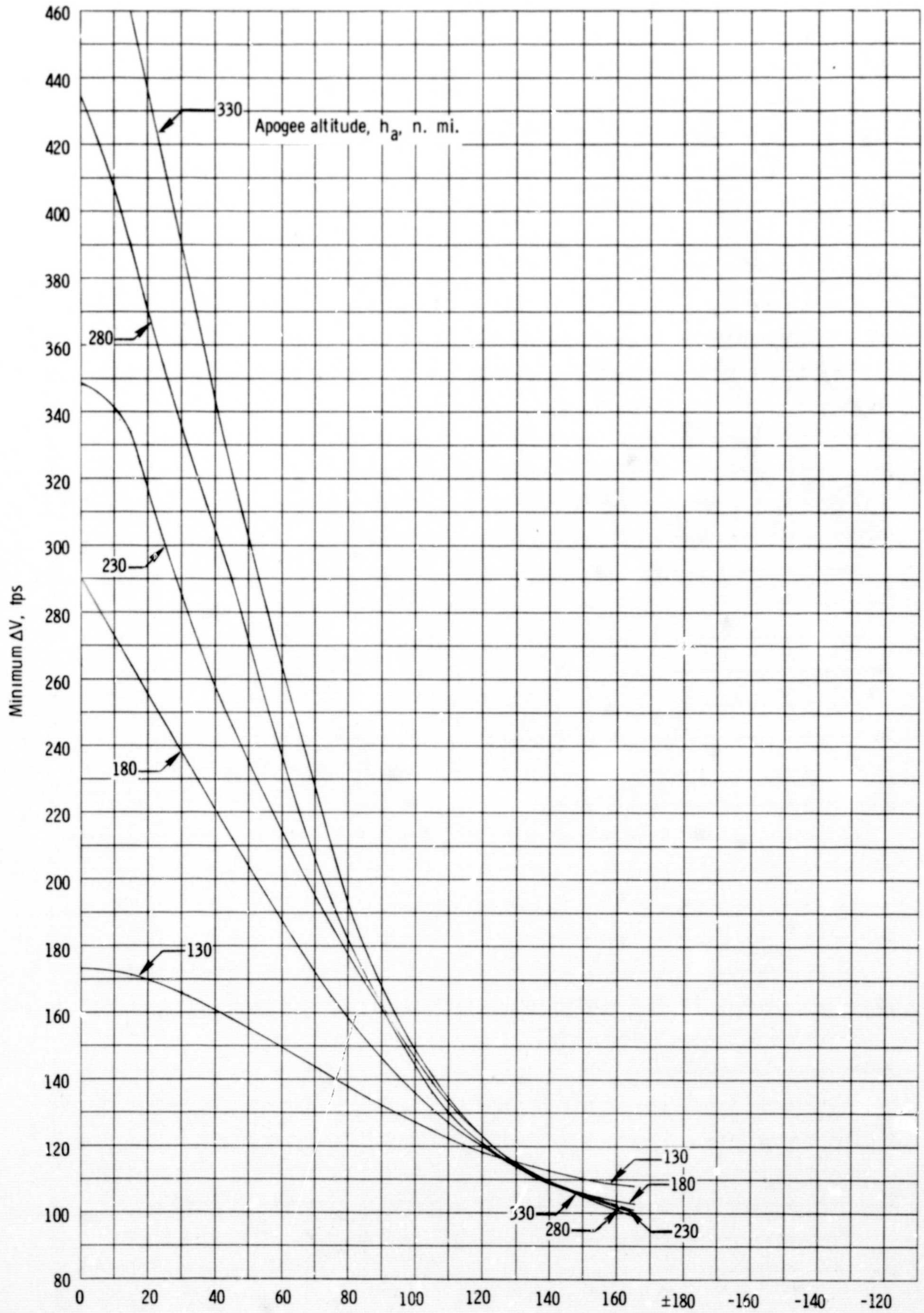


Figure 12.- Inertial flight-path angle and velocity at 400 000 feet as a function of retrograde true anomaly and ΔV of 100 feet per second for an elliptical orbit where $h_p = 100$ nautical miles, and $h_a = 150$ nautical miles.



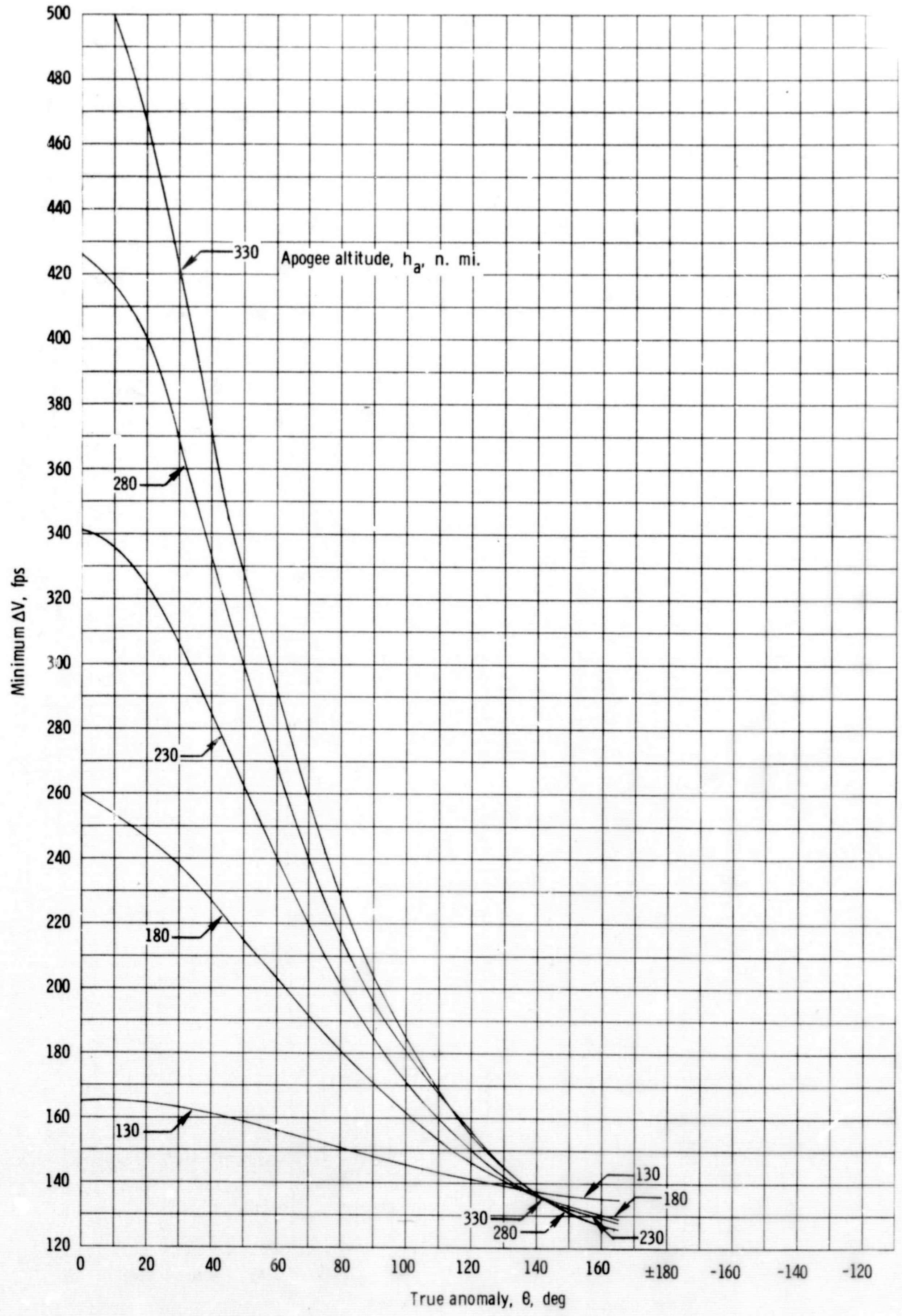
(a) Perigee altitude = 85 nautical miles.

Figure 13. - Minimum retrograde ΔV's for near earth orbits to achieve an entry time of 1400 seconds from various retrograde true anomalies.



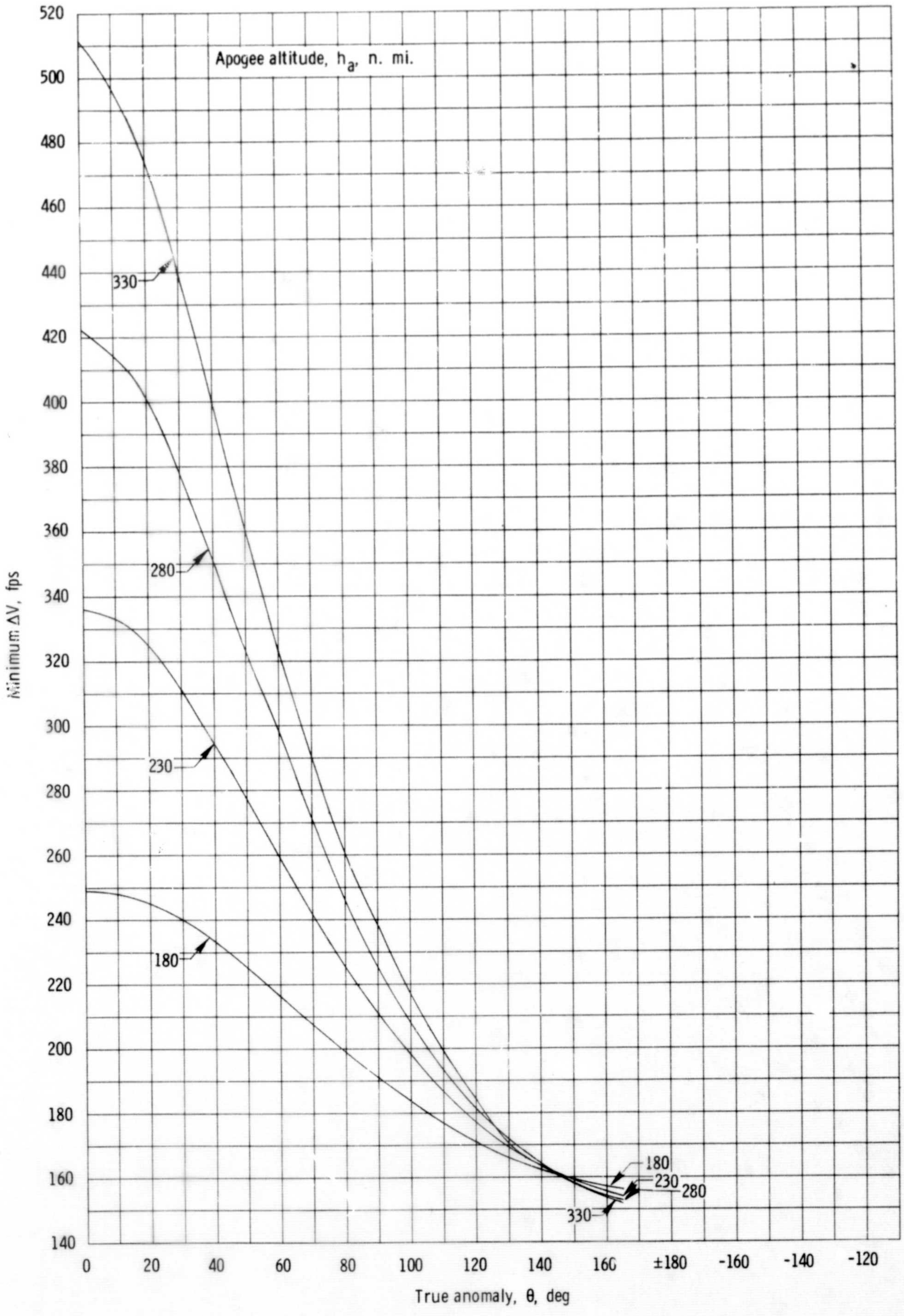
(b) Perigee altitude = 100 nautical miles.

Figure 13. - Continued.



(c) Perigee altitude = 115 nautical miles.

Figure 13. - Continued.



(d) Perigee altitude = 130 nautical miles.

Figure 13. - Concluded.