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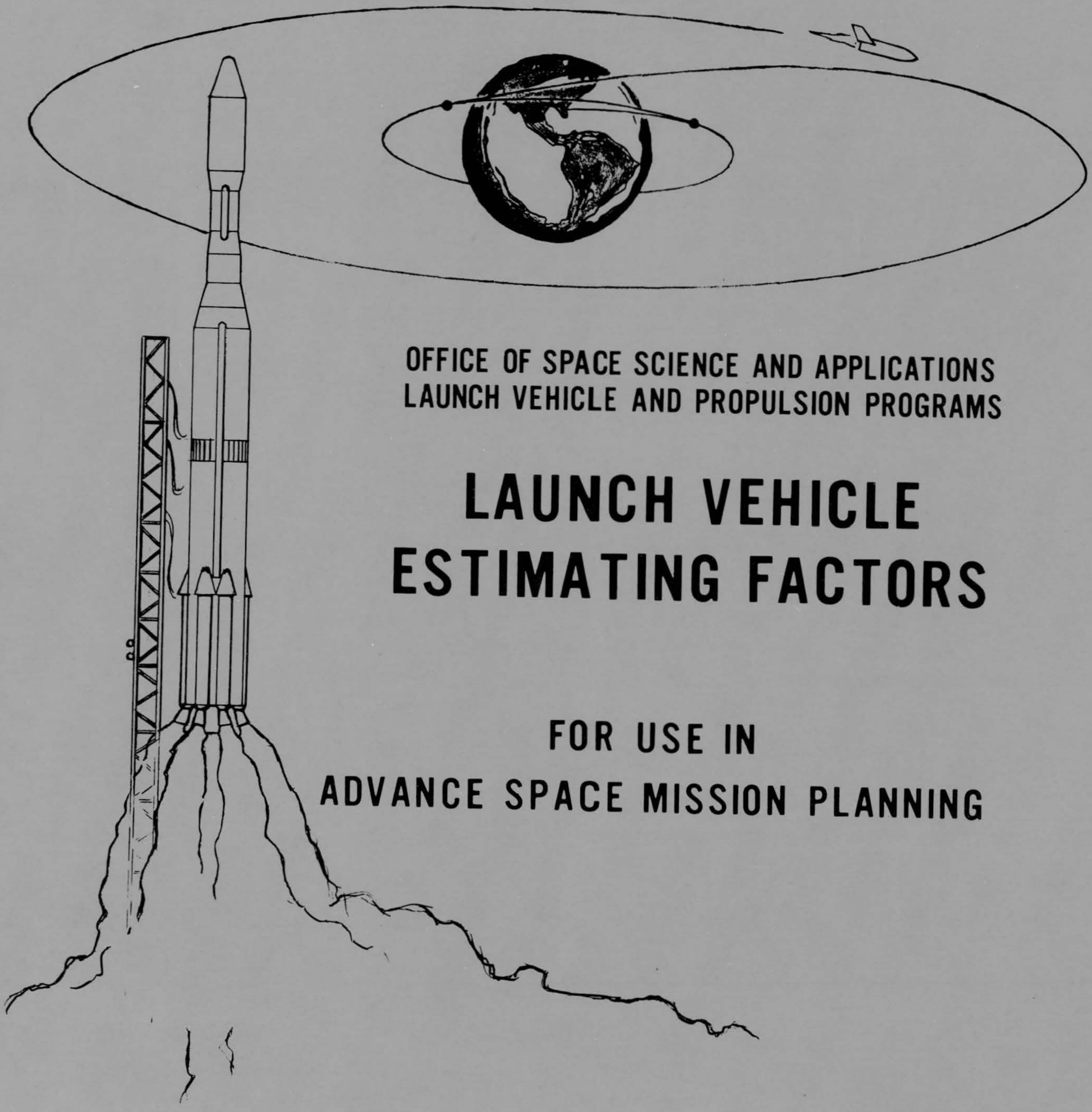

OFFICE OF SPACE SCIENCE AND APPLICATIONS
LAUNCH VEHICLE AND PROPULSION PROGRAMS

LAUNCH VEHICLE ESTIMATING FACTORS

FOR USE IN ADVANCE SPACE MISSION PLANNING

NA

PREFACE

January 1971

This launch vehicle reference document contains mission characteristic and launch vehicle capability data for use by NASA and particularly the Office of Space Science and Applications (OSSA) in the preparation of future mission plans.

The data in this reference book are derived from sources that are considered to be sufficiently accurate for advance planning. In no instance should these data be used for detailed mission planning without concurrence of the Director of Launch Vehicle and Propulsion Programs.

This edition reflects updates in vehicle performance and possible availability and introduces a section on the Space Shuttle. The Shuttle section has been coordinated with the Office of Manned Space Flight.

This document is intended for the official use of NASA and DoD, but may be purchased by nongovernment organizations from the Government Printing Office.

Data contained in this document are reviewed and updated annually. Questions should be directed to Mr. J. E. McGolrick, Mr. B. C. Lam, or Mr. J. W. Haughey on telephone 202/962-4553.

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SECTION I

INTRODUCTION

SECTION I

INTRODUCTION

This document is a tool for establishing a first estimate of launch vehicle requirements by those engaged in the preparation of advance mission plans.

The sections in this document are organized such that the user, having defined a mission in terms of payload weight and orbital specification or space destination, may normally proceed through the following steps:

(1) Determination of basic mission velocity requirements - Section II, Mission Factors and Conversion Charts

(2) Determination of launch site effects and incremental correction on velocity requirements as a function of orbit inclination or launch azimuth at each site - Section III, Launch Site Factors

(3) Determination of the total characteristic velocity (V_C, defined in Section VI) required to accomplish the mission, through combination of the results of (1) and (2)

(4) Determination of the launch vehicle(s) that can deliver the prescribed payload weight at the required characteristic velocity from a specified launch site - Section IV-A, Generalized Performance Data for Expendable Launch Vehicles Using Chemical Propulsion, or Section IV-C, Space Shuttle Performance.

This procedure is appropriate for most launch vehicles; however, because of various constraints, performance capabilities of certain vehicles cannot be generalized. This is particularly true for Earth orbital missions. Therefore, specific Earth orbital performance data for selected expendable launch vehicles and for the space shuttle without upper stages are contained in Section IV-B, Earth-Orbital Capability for Selected Expendable Launch Vehicles and the first part of Section IV-C, Space Shuttle Performance, respectively. For the space shuttle with a solid propellant upper stage, Section IV-C contains sufficient performance data to allow calculation of the vehicle performance once a required sequence of ΔV's are determined from Section II. For other expendable vehicles involving nonrestartable final stages not included in Section IV-B, information can be obtained by contacting one of the people listed in the Foreword.

Performance data for selected solid-propellant velocity packages, which may be used to increase the characteristic velocity of some launch vehicles with small payloads, are presented in Section IV-D, Velocity Packages.

Section IV-E, Performance of Solar-Electric Propulsion Systems, and IV-F, Performance of Nuclear-Thermal and Nuclear-Electric Propulsion Systems, contain representative data for possible future nonchemical stages and electrically propelled spacecraft.

Section IV-G, Retro-Propulsion and Apogee Kick Motor System Data, provides a basis for estimating weights of planetary orbiter retro-propulsion systems and apogee kick stages.

Section V, Shroud Configurations, indicates nominal payload physical dimension constraints for potentially available shrouds.

Section VI, Definition of Terms, is a brief glossary of terms used in this document.

SECTION II

MISSION FACTORS AND CONVERSION CHARTS

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MISSION FACTORS AND CONVERSION CHARTS

The purpose of this section is to provide data for the preliminary estimation of launch energy requirements for a variety of space missions. It should be emphasized that these data are intended only as first approximations. For more detailed information, the reader is referred to information available in the references cited at the end of this section. Section VI should be consulted to clarify definitions of terms.

The performance required to obtain a specific unpowered spacecraft trajectory is most succinctly characterized by the summation of all the discrete and impulsive changes in velocity that must be imparted to the spacecraft. The characteristic velocity so obtained then depends only upon the mission itself and the assumed sequence of impulses employed to inject the spacecraft into its final ballistic trajectory.

The information contained in this section may be used to obtain the characteristic velocity (V_C) requirements for a variety of missions. These characteristic velocity requirements are based upon a reference altitude of 100 nautical miles (n. mi.) above the Earth's surface. By definition, then, the characteristic velocity for a 100 n. mi. circular orbit is assumed to be equal to the actual orbital speed referred to Earth-centered nonrotating coordinates. The characteristic velocity for any other mission is then obtained by adding the required velocity increments to the orbital speed at an assumed 100 n. mi. parking orbit.

In many cases, the characteristic velocity requirements presented here may be compared directly to launch vehicle capabilities shown in Sections IV-A and IV-C, provided that the appropriate penalty contained in Section III for launches other than eastward from Cape Kennedy is added to the basic mission requirements. Exceptions occur for certain missions for launch vehicles using stages that have limited coast or restart capability. These launch vehicles may be unable to perform the particular sequence of impulses assumed in deriving these general characteristic velocity requirements, and more specific performance data, unique for each such vehicle, must be used or computed. Examples of such specific performance data are given in the figures of Section IV-B, which show the Earth orbital performance available from certain expendable launch vehicles using a direct ascent launch rather than an initial 100 n. mi. parking orbit. Data are available in Section IV-C that allow shuttle performance calculations where solid propellant shuttle upper stages are used.

Solar System Missions

Ecliptic Plane Probes

The characteristic velocity requirements for probes to regions lying in the ecliptic plane are shown in Figures II-1 through II-5.

Figure II-1 contains the velocity requirement data as a function of trip time and distance from the Sun for direct flights to regions farther out from the Sun than Earth (outer ecliptic region). Note that the planets (and the asteroid Ceres) are represented by bands of distance from the Sun. These bands are intended to show the limits of radial distance caused by the eccentricity of the planet orbits (see Table II-1 for other details).

As seen in Figure II-1, trajectories to regions far from the Sun, with reasonable flight times, require very high launch velocities. These high velocity requirements can be alleviated somewhat by employing Jupiter gravity-assisted trajectories as shown in Figure II-2. The use of a close encounter with a planet to modify the heliocentric trajectory of a spacecraft has been studied by many investigators, and Jupiter appears to have the most dramatic potential as a swingby target for automated vehicles. Figure II-2 may be compared with Figure II-1 to see the rather substantial savings in either launch velocity or trip time that may be obtained using the Jupiter-assisted trajectories to the outer regions of the ecliptic plane.

For probes to regions closer to the Sun than Earth (inner ecliptic region), minimum energy direct flights require less than 6 months, so only the minimum velocity requirements are shown in Figure II-3. Again, the bands of distance from the Sun representing Venus and Mercury are shown for general information only.

Gravity-assisted trajectories may be used to advantage, under some circumstances, for regions close to the Sun. Although Venus gravity-assisted trajectories to Mercury and the vicinity of 0.2 a. u. have been shown to be superior to direct flight in some cases (see References II-1 and II-2), Jupiter swingbys can be very effective in reducing the large launch velocity required for very close Solar probes. Trajectory data for close Solar probes employing Jupiter swingby are shown in Figure II-4. Recalling that direct flights take less than 6 months, the substantial penalties in flight time associated with the Jupiter swingby are apparent.

In the event that a heliocentric probe orbit is desired which will range through distances from the Sun that lie on both sides of Earth's orbit, Figure II-5 is included to show the characteristic velocity required to achieve a given perihelion and aphelion combination.

Out-of-Ecliptic Probes

Data for direct flights to regions out of the ecliptic plane are presented in Figure II-6; wherein the required launch characteristic velocity is shown for reaching a given point, defined by celestial latitude and radial distance from the Sun. These are optimum values in the sense that the characteristic velocity has been minimized at all points. The corresponding flight times are also shown.

Since any out-of-ecliptic launch requires directing the hyperbolic excess velocity vector away from the ecliptic plane, launch azimuth constraints (see Section III) may become significant. Those optimum trajectories that require launches outside the nominal azimuth limits from the ETR lie in the shaded region of Figure II-6. Probes to points in the shaded region would probably require launch from the WTR to achieve the high declination angles required, and a launch site velocity penalty would have to be added to the basic velocity requirement of Figure II-6. For launches in the acceptable region from the ETR, the velocity penalty caused by launching in a noneasterly direction is fairly small, being always less than 390 ft/sec. Further information is contained in Reference II-3.

It is apparent that direct launches to points at high celestial latitudes may require prohibitively large launch velocities. Once again, Jupiter swingbys have been shown to be useful in reducing the launch velocities required for out-of-ecliptic probes. Figure II-7 shows a plot of the altitude above the ecliptic plane versus distance from the Sun in the ecliptic plane, for fixed values of characteristic velocity. These data are based on optimized trajectories, in which the magnitude of the characteristic velocity is minimized for each point, and the direction of flight leaving Earth and the encounter conditions at Jupiter are free.

Planet Flybys and Orbiters

Launch Requirements. For missions to the planets, the relative positions of the Earth and the target planet in their respective orbits must lie within certain angular limits for reasonable launch velocities. As a consequence, the synodic period of revolution of the target planet with respect to Earth is of paramount importance in establishing launch opportunities. If the orbits of Earth and the target planet were perfectly circular and coplanar, the launch energy requirements would be identical at each launch opportunity as established by the synodic period.

Actually, significant differences can occur in the minimum launch energies required for different launch opportunities because the orbits of the planets are neither circular nor coplanar. Because of such differences, Figures II-8 through II-15 present characteristic velocity data peculiar to particular launch opportunities occurring during the period of interest.

Figure II-8 shows launch velocity requirements for flights to Mercury from 1977 to 1990. For Mercury, three opportunities exist each year, but Figure II-8 shows data only for the two of interest. The solid bar shows the absolute minimum launch velocity for each year and would represent the minimum requirement for a flyby mission. Approach velocities corresponding to these trajectories are extremely large (40,000 to 56,000 ft/sec) and are not considered practical for Mercury orbiter missions. The open bars, on the other hand, present data for a different launch opportunity each year and represent the launch velocity corresponding to the trajectory which minimizes the approach velocity at Mercury.

No consideration is given in Figure II-8 to presenting data for a particular opportunity width for each specific mission because of the rapidly changing trajectory parameters which are characteristic of Mercury missions. A typical velocity increment, above the listed characteristic velocity, would be 400 ft/sec for a 15-day opportunity. Trade-offs between mission requirements and opportunity width must be made on an independent basis for each mission because of the need to consider many more constraining parameters, such as those related to departure and arrival geometry, than can be discussed here.

Swingby trajectories employing a gravity assist from Venus have been shown to be beneficial during some years for both flyby and orbiter missions to Mercury. Figure II-9 shows launch velocity requirements for Venus-Mercury missions between 1973 and 1985. In general, launch opportunities up to 30 days do not require large characteristic velocity increments as would a 30-day opportunity for direct Mercury missions; however, in 1975 and 1983 only 20-day opportunities are possible for unpowered swingbys. A powered swingby can be used to extend these two launch windows; this mode is the only one possible in 1980. The velocity increments (at Venus passage) shown for the powered mode in 1975, 1980, and 1983, are maximum for a 30-day opportunity. For more information on this subject, the reader should consult References II-4, II-5, and II-6. Specific questions pertaining to both direct Mercury and Venus-Mercury missions should be addressed to NASA OSSA Launch Vehicle and Propulsion Programs.

Figure II-10 shows the launch velocity requirements for Mars and Venus from 1972 to 1990. The velocity required to permit a 30-day launch opportunity is shown along with the minimum launch velocity for each opportunity. In some opportunities, Type I trajectories have the minimum launch velocity requirement; in others, Type II trajectories do. In cases where there is a strong tradeoff between launch velocity and flight time, both Type I and II trajectories are shown.

Because of excessive flight times, absolute minimum energy launches to Jupiter and the outer planets are not attractive. Therefore, the data in Figure II-11, II-12, and II-13 show launch velocity requirements for particular flight times to Jupiter, Saturn, and Uranus, respectively, along with the velocity to permit a 30-day launch opportunity width.

For Neptune, the yearly changes are negligible during the period from 1974 to 1990, so the velocity requirement is shown in Figure II-14 as a function of trip time only.

The yearly changes in velocity requirement again become significant for probes to Pluto, because of the large eccentricity and inclination of that planet's orbit. Characteristic velocity data for two flight times are shown in Figure II-15. Only one opportunity per year exists.

Figures II-11 through II-15 are based upon data contained in Reference II-7, and that report should be consulted for further information.

For missions to the planets beyond Jupiter, the use of a Jupiter gravity-assisted trajectory may be advantageous in reducing either trip time or launch characteristic velocity. Data in Figures II-16 show launch velocity requirements for particular flight times for Saturn and Uranus. Similar data for Neptune, and Pluto missions using Jupiter swingby are shown in Figure II-17. The velocity increments for 20-day opportunities are shown instead of 30-day opportunities since they are significantly less. Each of these missions has five opportunities in the 1970-1990 time period, corresponding to five successive Jupiter opportunities.

An unusual opportunity for multiple planet swingbys occurs in the 1975 to 1981 time period. The "Grand Tour" using Jupiter, Saturn, and Uranus swingbys to Neptune has launch opportunities in each year from 1976 to 1980. The opportunity to perform this mission occurs only every 179 years. Figure II-18 shows launch velocity requirements for particular flight times for the "Grand Tour". Two other multiple planet swingby combinations of considerable interest are shown in Figure II-19. These are a Jupiter-Saturn-Pluto mission and a Jupiter-Uranus-Neptune mission. Further information can be obtained from References II-6, II-13, II-14, II-15, and II-16.

In general, the Jupiter swingby mode will be beneficial for flyby missions, but the increases in approach velocity at encounter with the target planet caused by the higher energy trajectories may preclude the use of the Jupiter swingby mode for orbiter missions.

The frequency of opportunities for Jupiter swingby trajectories to the outer planets depends upon the synodic period of the outer planets relative to Jupiter.

	Synodic Period of Outer Planets Relative to Jupiter, Earth Years
Saturn	19.8
Uranus	13.7
Neptune	12.7
Pluto	12.0

Planetary Orbit Energy Requirements. For planetary orbiters, the magnitude of the required retro-velocity increment, which must be provided by the spacecraft propulsion unit, depends upon the approach velocity relative to the target planet, the mass of the planet, and the periapsis and eccentricity of the desired satellite orbit. If the only mission requirement is that of being captured by the target planet, an orbit of high eccentricity but very low periapsis is most economical. The required retro-impulse, which is applied at periapsis, is smaller at lower periapsis altitudes. If the retro-impulse is applied at very high periapsis altitudes, the required impulse, at worst, approaches the magnitude of the initial hyperbolic excess velocity relative to the planet.

For circular orbits or orbits of small eccentricity, the required retro-impulse is, of course, larger than that required for capture. In some cases, an altitude exists that minimizes the required velocity increment for a circular orbit.

Data that permit determination of retro-velocity requirements for all potential target planets except Pluto are given in Figures II-20 through II-27. The first six of these figures give the approach velocities associated with the trajectories for which launch velocity requirements were given in Figures II-8, and II-10 through II-14.

Figure II-26 shows escape velocities for different periapses for all the planets and Earth's Moon. The relation between retro-impulse, approach velocity, and escape velocity is shown in Figures II-27a and II-27b.

The following procedure yields the retro-impulse requirement:

(1) Determine the approach velocity from Figures II-20 through II-25

(2) Select the periapsis and apoapsis radii for the desired final orbit*

(3) Determine the escape velocity at periapsis for the planet being orbited from Figure II-26

(4) Calculate the ratio of apoapsis to periapsis of the final orbit and the ratio of approach velocity to escape velocity at periapsis

(5) Use Figure II-27a or II-27b to determine the ratio of retro-impulse to escape velocity at periapsis, and multiply this ratio by the escape velocity obtained in (3).

* It is convenient to choose the orbit such that the ratio of apoapsis to periapsis is one of those plotted in Figure II-27a or 27b. Since it is sometimes convenient to specify orbital altitudes above the surface, Table II-1 is included to facilitate the conversion. The Table also includes various other data for the planets.

An estimate of the retro-propulsion system weight can be made by using this data in conjunction with information in Section IV-G of this document.

Earth Orbit Requirements

Figure II-28 depicts the velocity required for Earth orbits. The circular orbit characteristic velocities assume a Hohmann transfer from the reference 100 n. mi. initial parking orbit. The curves for circular and eccentric orbits in Figure II-28 are not related to any particular launch site. However, the curve labeled circular equatorial orbits from ETR show the characteristic velocity requirements to establish a circular orbit with zero degree inclination after launching due east from ETR. The calculation is based upon the plane change being optimally divided between the two impulses of a Hohmann transfer. Synchronous altitude is indicated on this curve.

More general Earth orbital data are contained in Figure II-29, where the total characteristic velocity requirement for orbits of arbitrary perigee and apogee are shown. The velocity contours of Figure II-29 are based upon the assumption of a transfer orbit with perigee at 100 n. mi. and apogee as shown along the abscissa, followed by a second impulse to raise perigee or, if sufficiently large, to establish a new apogee with the apogee of the transfer orbit becoming the perigee of the final orbit. The more efficient maneuver involves establishing the transfer orbit with apogee being the final apogee value, while using the second impulse to establish a new perigee, rather than transferring first to the new perigee and then raising the apogee. The difference between the two techniques is significant only for very high energy orbits. The coast time in the initial transfer orbit may be limited by system considerations. The coast time from 100 n. mi. to any transfer apogee may be estimated from Figure II-28.

The total characteristic velocity shown in Figure II-29 includes the velocity impulse required at the apogee of the initial transfer orbit. This impulse is shown as a separate item in Figure II-30 to assist in defining possible energy management problems or to estimate independent kick motor requirements.

As mentioned above, Figures II-28, II-29, and II-30 assume an initial altitude of 100 n. mi. This restriction does not apply to the more general method, discussed below, of estimating Earth orbital impulse requirements. This general technique must be used if the initial orbit is not at 100 n. mi., or if a plane change is included.

This general method has been developed for estimating the velocity increments required to transfer between Earth orbits subject only to the restriction that the impulses are applied at either perigee or apogee. An intermediate circular orbit may be required if the initial and final orbits do not have a common line of apsides.

Figures II-31a and II-31b illustrate the relationship between the horizontal inertial velocity at a specified reference altitude, and the altitude of the other apsis, for any Earth orbit. Figure II-31a presents this relationship on log-log paper to allow the consideration of apsis altitudes up to 500,000 n. mi. Figure II-31b is restricted to the lower altitudes and is plotted on linear coordinates with velocity increments of 200 ft/sec.

In brief, either Figure II-31a or II-31b, as appropriate, can be used to find the horizontal velocity which must exist at the reference altitude before and after an impulse is applied. The magnitude of the required impulse for coplanar orbits is then simply the absolute magnitude of the difference between these velocities. For low altitudes this difference can be obtained more accurately from Figure II-31c where circular velocity has been subtracted from the orbital velocity.

For orbits which are not coplanar, but which have a common line of apsides, Figure II-32a or II-32b is also needed to compute the vector magnitude of the impulse required to alter the altitude of the other apsis of the initial orbit and change the plane of the orbit simultaneously. To use these figures, the initial velocity and the final velocity are obtained from Figure II-31a or II-31b for the reference altitude (i.e., the altitude at which the impulse is assumed to occur). As in the coplanar case, these inertial velocities are functions only of the other apsides, before and after the impulse, and are not dependent on the plane change. By entering Figure II-32a or II-32b at the computed value of the ratio of these velocities and interpolating at the specified plane change angle, the ratio of required velocity increment to final velocity is read from the ordinate. This is the total velocity increment required to change the other apsis and the plane of the orbit simultaneously.

If the initial and final orbits have a common apsis altitude, a single-impulse maneuver is possible. If not, a two-impulse maneuver may be analyzed by hypothesizing an intermediate transfer orbit. In general, the intermediate transfer orbit can be formed with either apsis of the initial orbit connected to either apsis of the final orbit, but, in general, the sum of the velocity increments will be minimized if the lowest and highest altitudes possible are used for the two impulses. In particular, if a plane change is needed, it should be performed at the highest altitude possible. Although a true optimum maneuver may dictate that a small part of the total plane change should be done at the lower altitude, the improvement is usually small. In any event, regardless of the number of impulses used, the user must specify a logical sequence of impulses and treat these one at a time, always using the altitude at which the particular impulse is applied as the "reference altitude" of Figures II-31a or 31b. Reference II-17 contains example solutions based upon this technique.

Certain special types of Earth orbits are noteworthy. For example, because the Earth's oblateness causes a precession of the orbital plane about the polar axis, it is possible to select an orbit that precesses at the same angular velocity as that of Earth about the Sun. As a consequence, the orbit maintains a constant orientation with respect to a line from the Sun to Earth. Figure II-33 presents the characteristic velocity and inclination requirements for circular "Sun-synchronous orbits". The launch azimuth penalty for a WTR launch is included in the characteristic velocity. The characteristic velocity for an ETR launch with no azimuth restrictions would differ from those shown by less than 20 ft/sec.

In addition to causing a precession of the orbital plane, the oblateness of Earth also causes the line joining perigee and apogee to rotate within the orbital plane. This rotation can be eliminated if an orbital inclination of 63.4° is selected.

Missions to the stable libration points in Earth-Moon space can be considered as special Earth orbits. These points lie about 208,000 n. mi. from the center of Earth along lines 60° on either side of the line joining Earth and the Moon. Since the influence of the Moon would be much smaller than that of Earth during establishment of this position, a good first approximation to the mission requirement assumes that the characteristic velocity requirement is the same as that for a circular orbit with 208,000 n. mi. radius.

Additional data on Earth orbits may be found in Reference II-18 and II-19.

Lunar Mission Requirements

Figure II-34 presents characteristic velocity requirements and equivalent hyperbolic excess velocity at the Moon for lunar missions as a function of trip time for the Moon at perigee and at apogee. In general, characteristic velocity requirements and approach velocities will lie between these pairs of curves. Retro-propulsion requirements for orbiter missions can be found using the equivalent hyperbolic excess velocity together with data in Figures II-26, and II-27a or 27b by following the procedure described on page II-7. If this procedure is used, orbits should be restricted to those with apolunes less than 22 lunar radii (equivalent to an altitude of about 19,600 n. mi.). For additional information see References II-18, II-19, and II-20.

Cautionary Note

Mission requirements data presented here are based upon specific trajectories. In using these data in conjunction with launch vehicle capability data in Section IV, it should be borne in mind that operations involving specific parking orbits, plane changes, and orbit circularization may require staging, coasting, or multiple burning of a stage beyond the capability of a particular launch vehicle. Related detailed questions should be referred to NASA OSSA Launch Vehicle and Propulsion Programs.

Conversion Charts

Figures II-35, II-36a, and II-36b are included for rapid conversion from the characteristic velocity used in this document to hyperbolic excess velocity, Earth Mean Orbital Speed (EMOS), and the energy parameter C_3. Figure II-37 contains conversions from traditional engineering units (ft/sec, lbm, and lbf) to metric units (m/sec, kg, and nwt).

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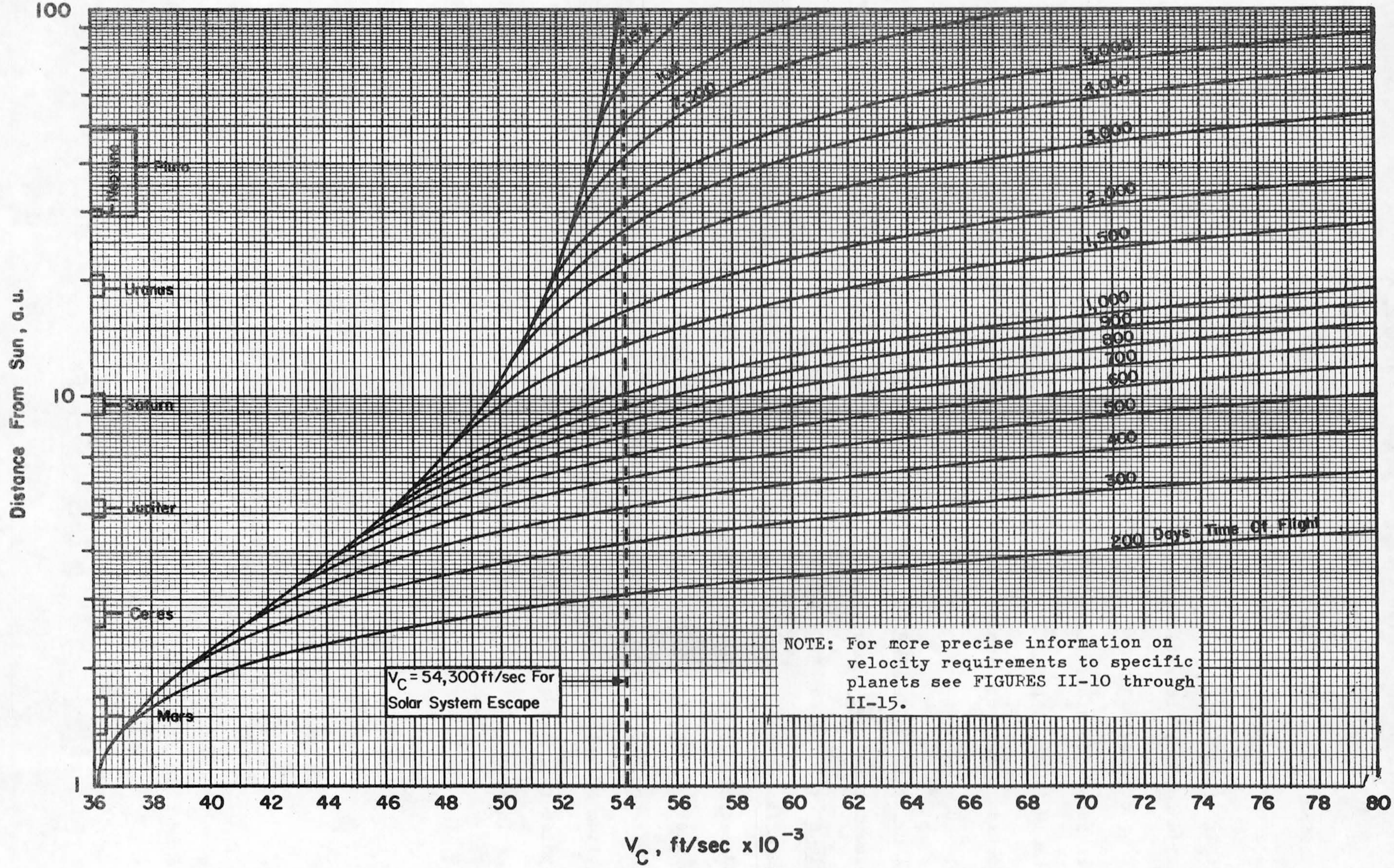


FIGURE II-1. VELOCITY REQUIRED FOR BALLISTIC PROBES TO OUTER ECLIPTIC REGIONS

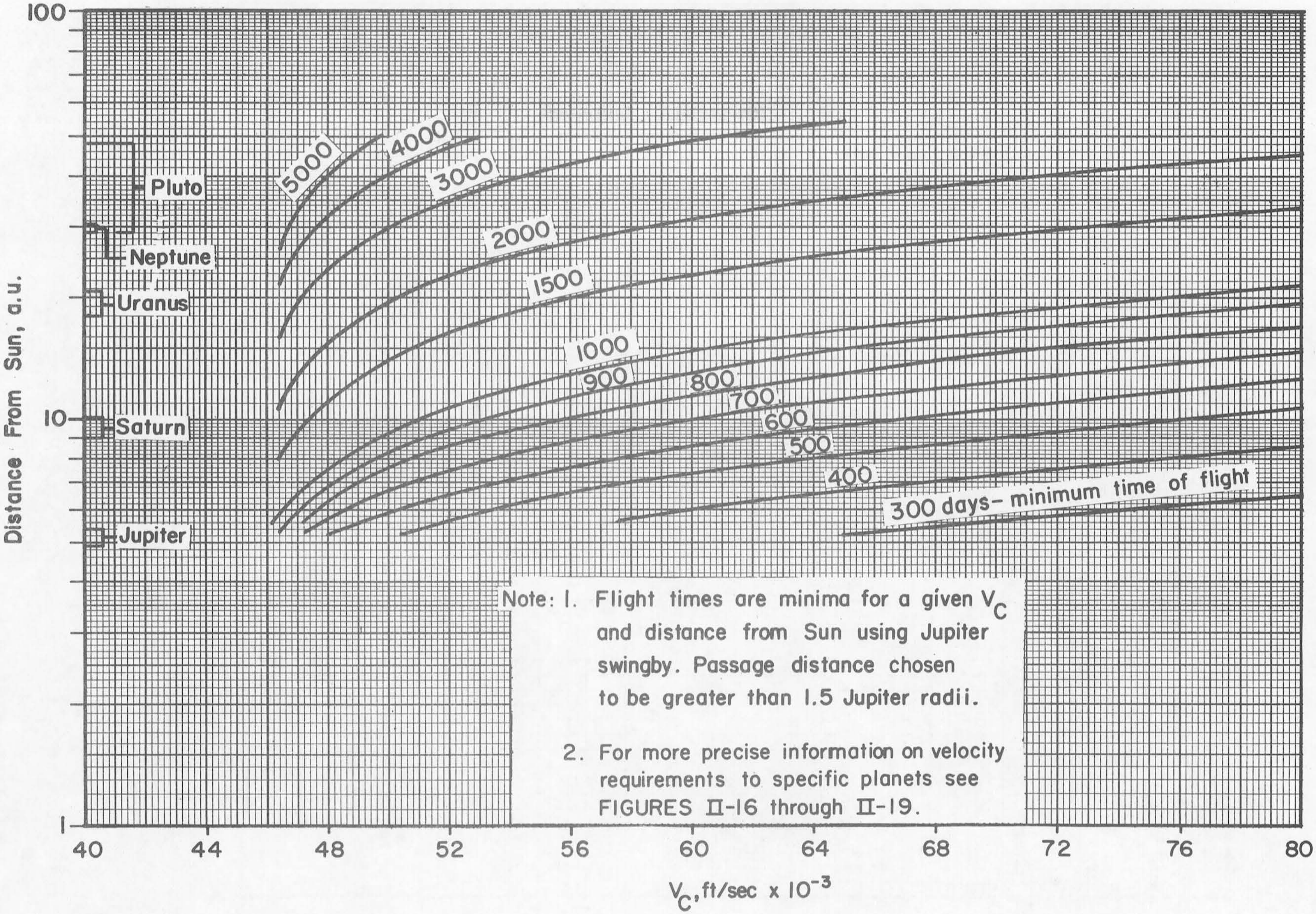


FIGURE II-2. VELOCITY REQUIRED FOR PROBES TO OUTER ECLIPTIC REGIONS WITH JUPITER SWINGBY

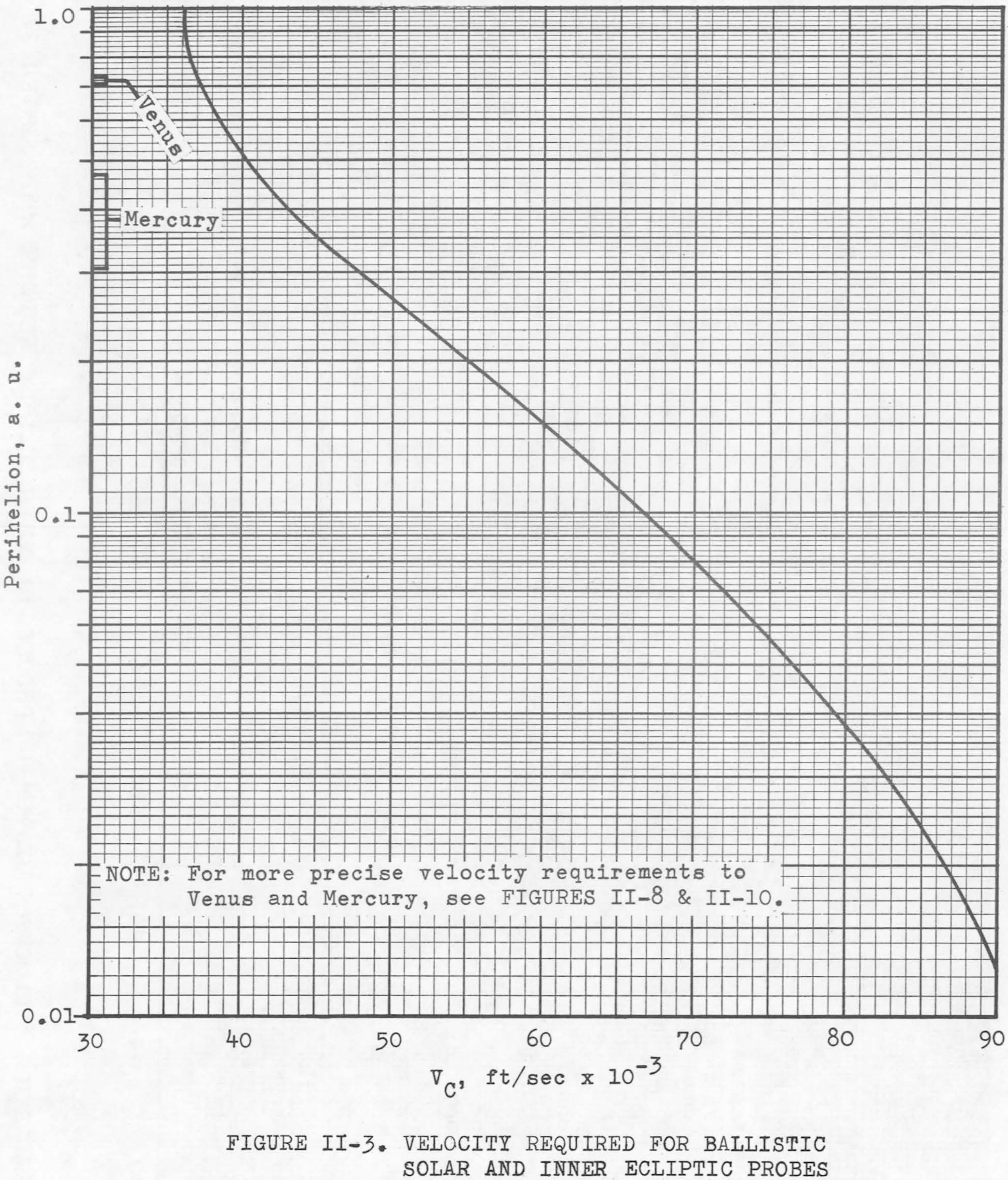


FIGURE II-3. VELOCITY REQUIRED FOR BALLISTIC SOLAR AND INNER ECLIPTIC PROBES

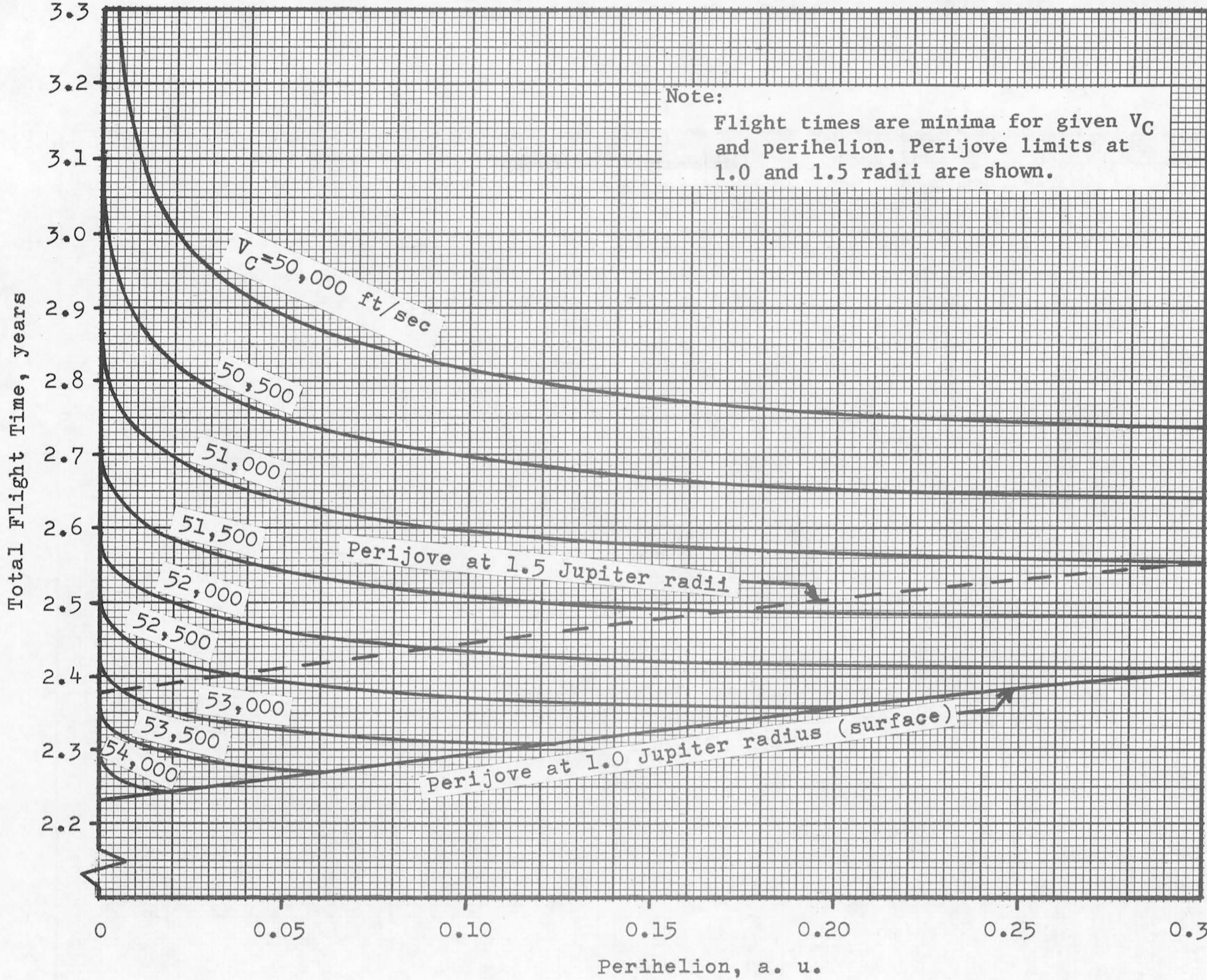


FIGURE II-4[illegible] TRAJECTORY DATA FOR SOLAR PROBES USING JUPITER SWINGBY

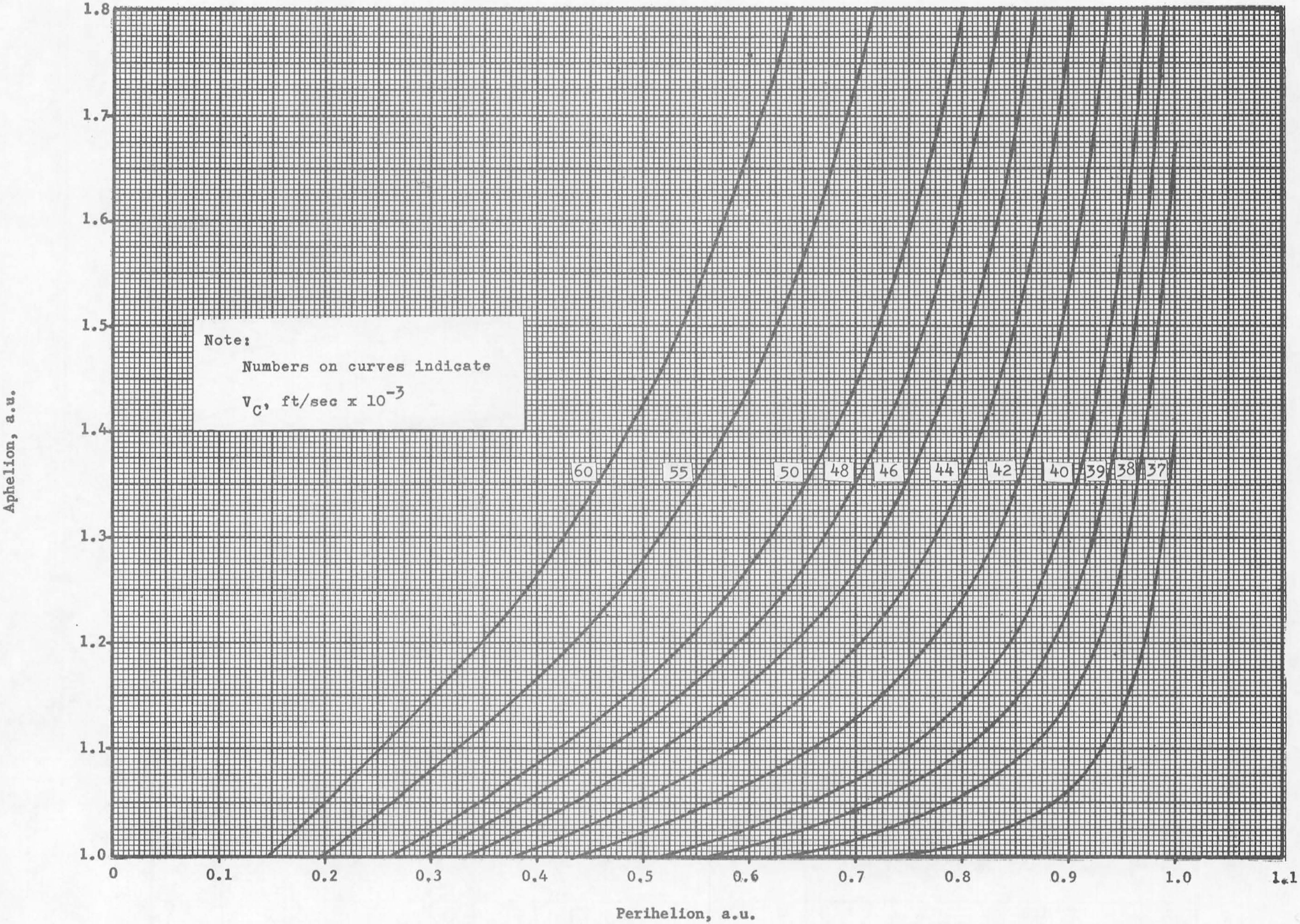


FIGURE II-5. VELOCITY REQUIREMENTS FOR BALLISTIC PROBES IN THE ECLIPTIC PLANE WITH VARIOUS PERIHELION AND APHELION DISTANCES

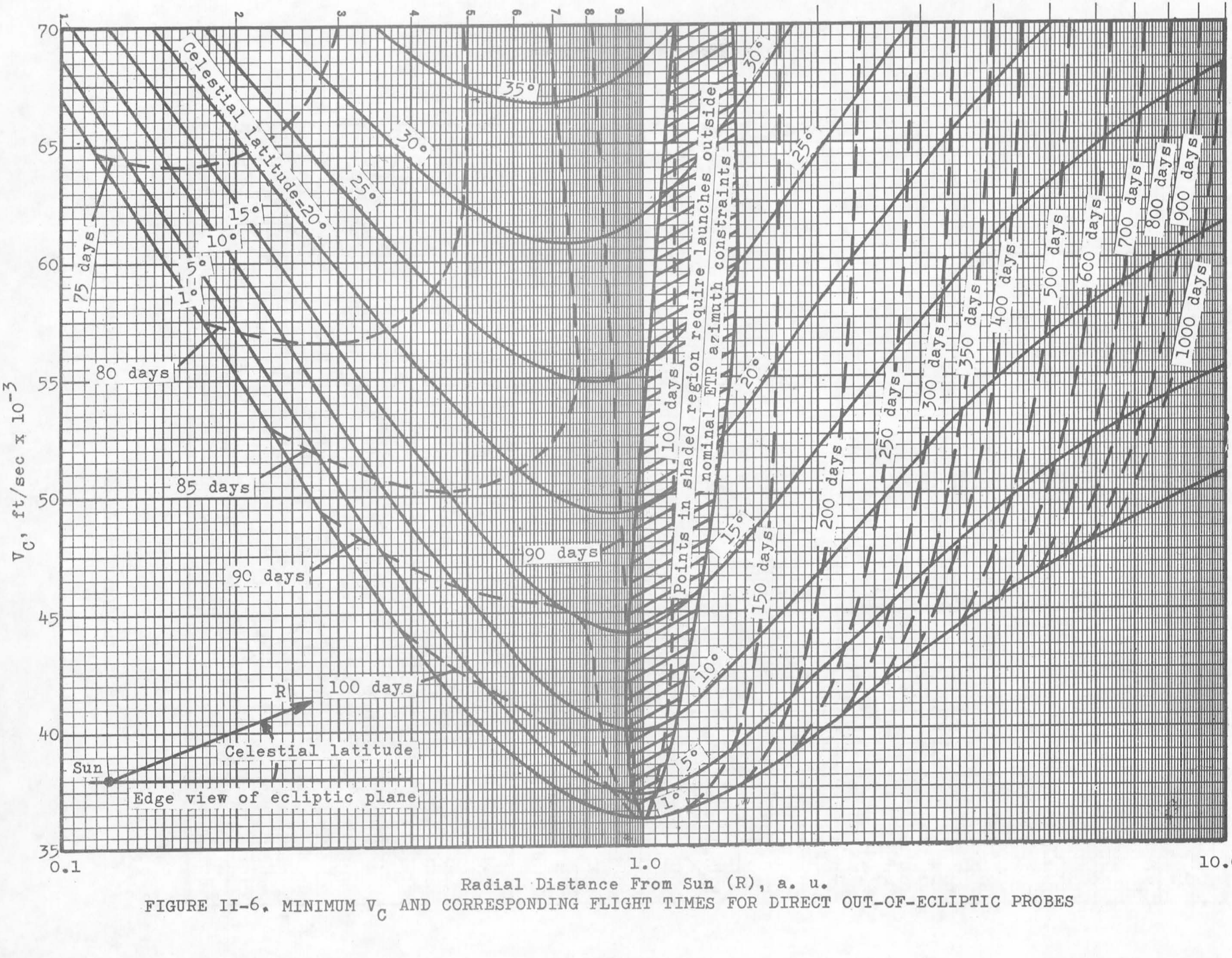


FIGURE II-6. MINIMUM V_C AND CORRESPONDING FLIGHT TIMES FOR DIRECT OUT-OF-ECLIPTIC PROBES

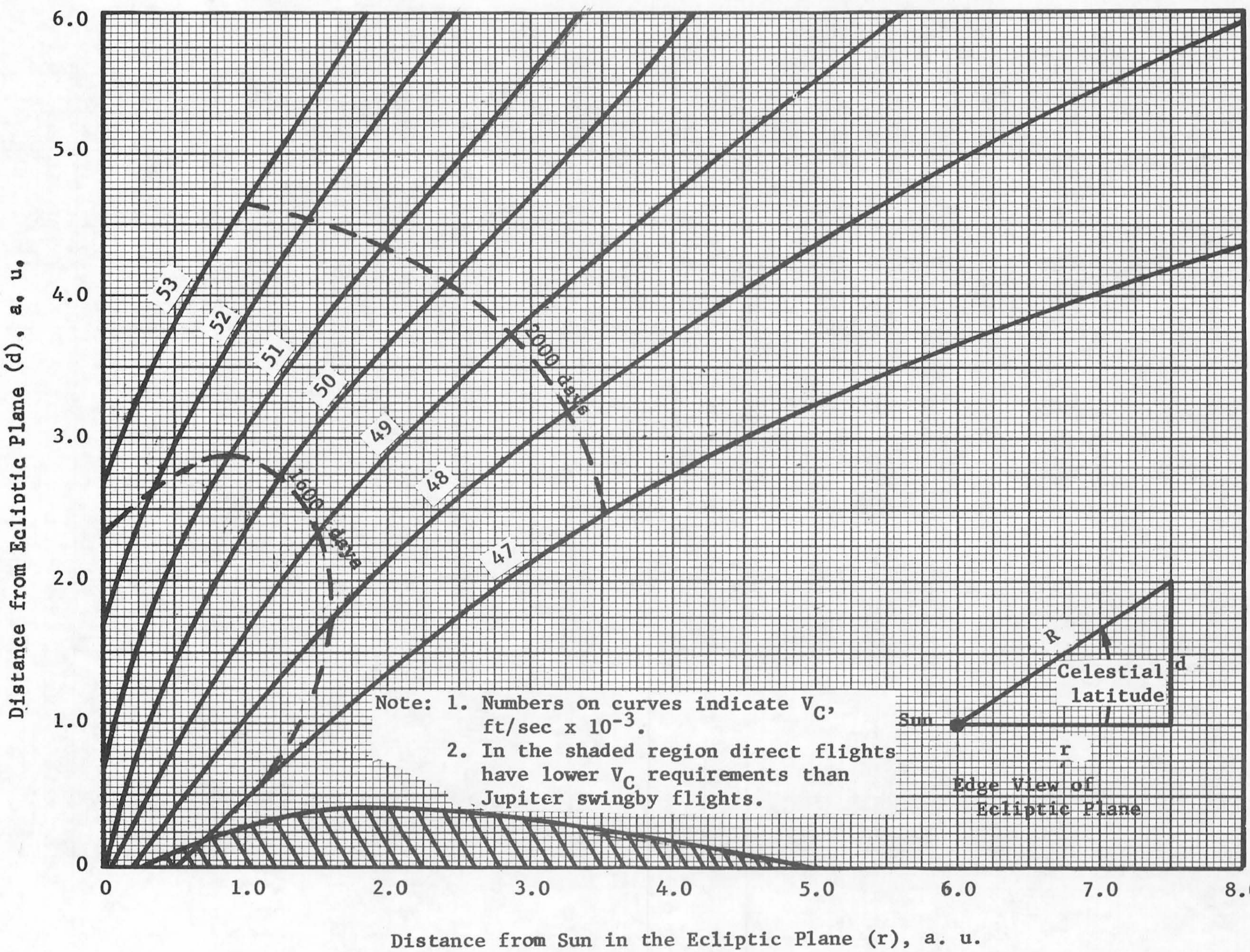


FIGURE II-7. ACCESSIBLE REGION BOUNDARIES FOR OUT-OF-ECLIPTIC

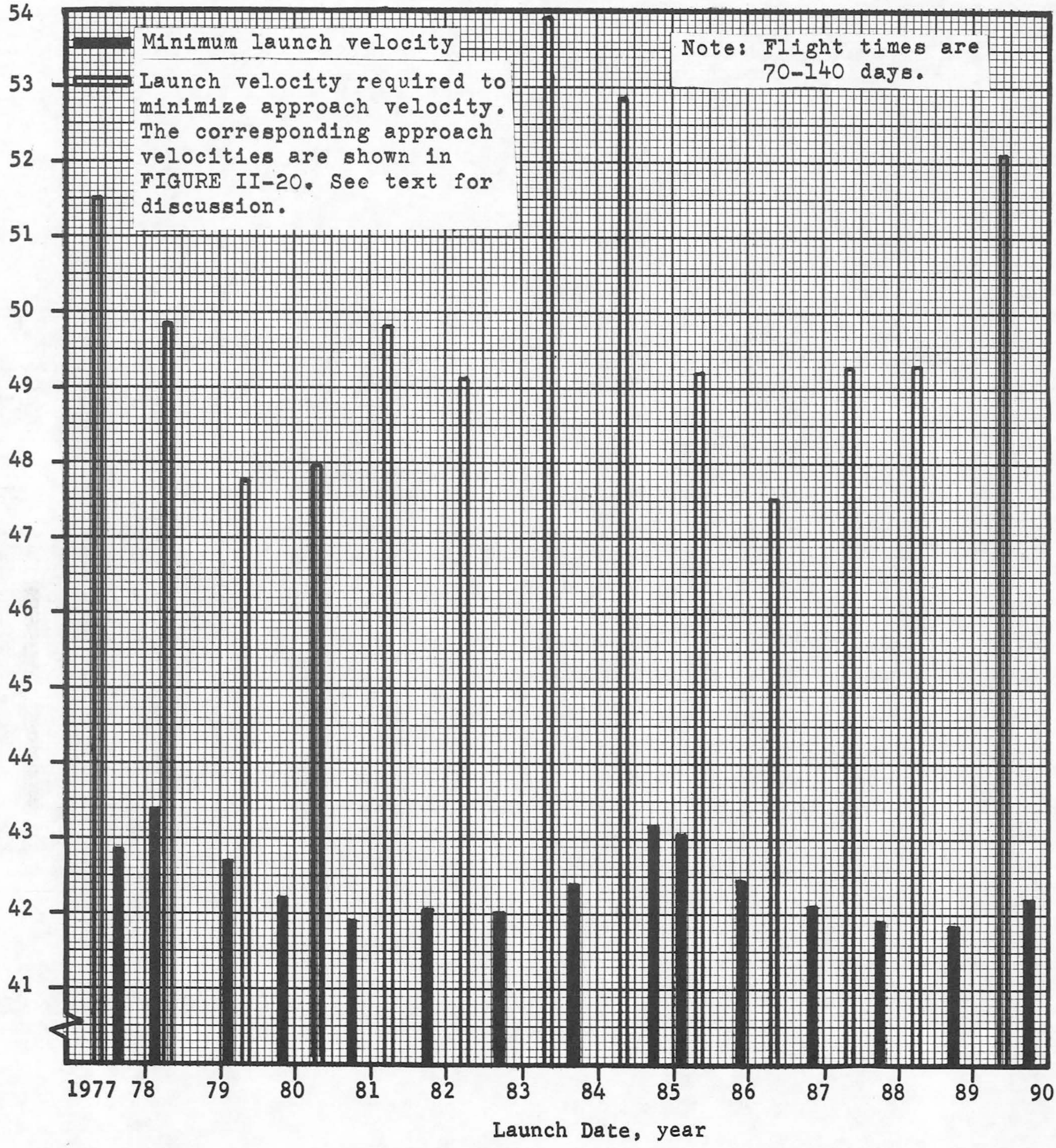


FIGURE II-8. LAUNCH CHARACTERISTIC VELOCITY FOR DIRECT MERCURY MISSIONS

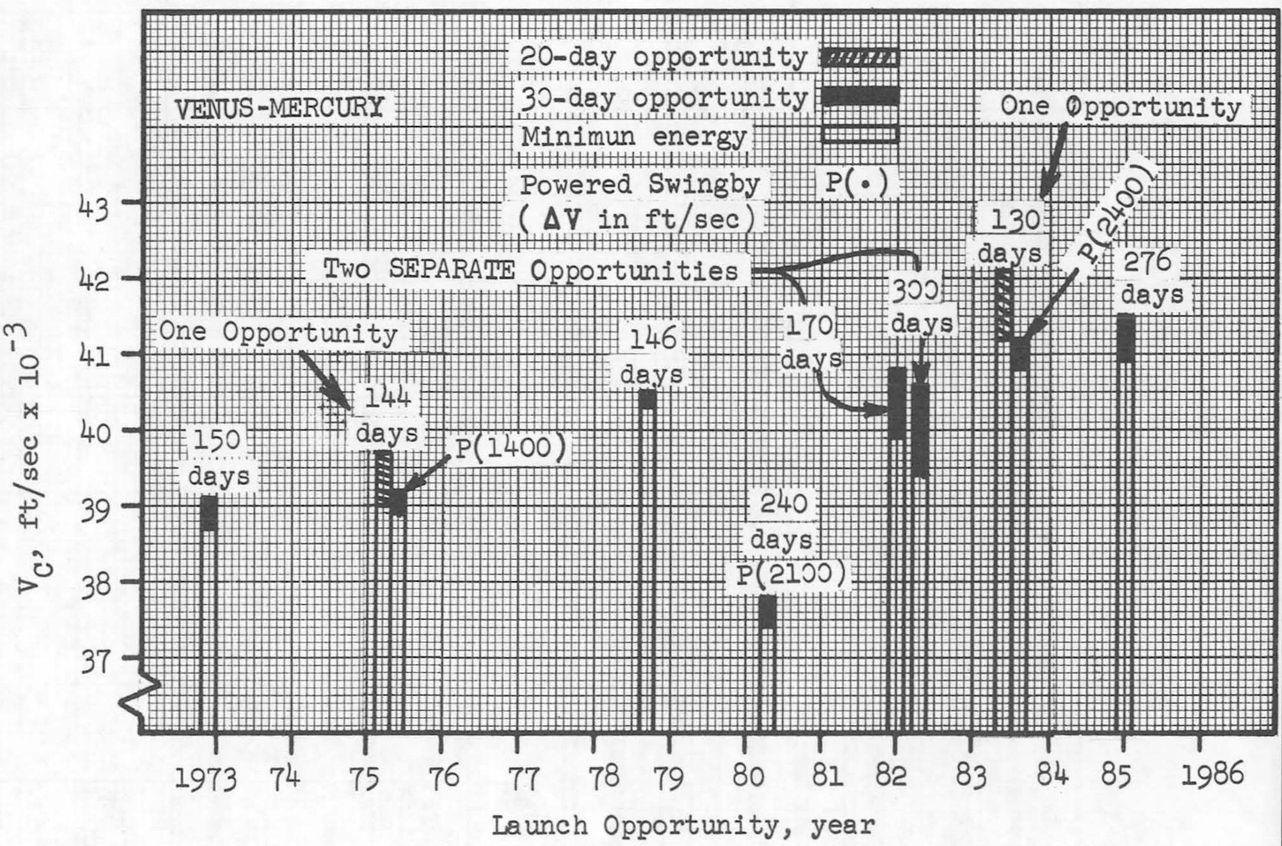
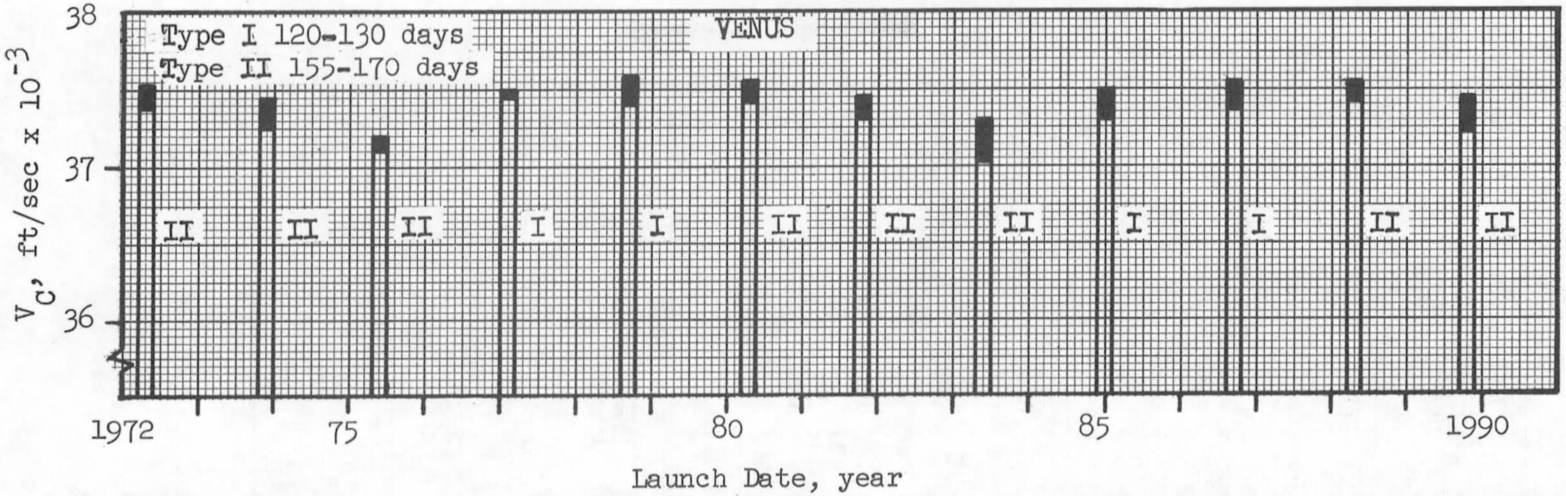


FIGURE II-9. CHARACTERISTIC VELOCITY REQUIREMENTS FOR MERCURY MISSIONS USING VENUS SWINGBY



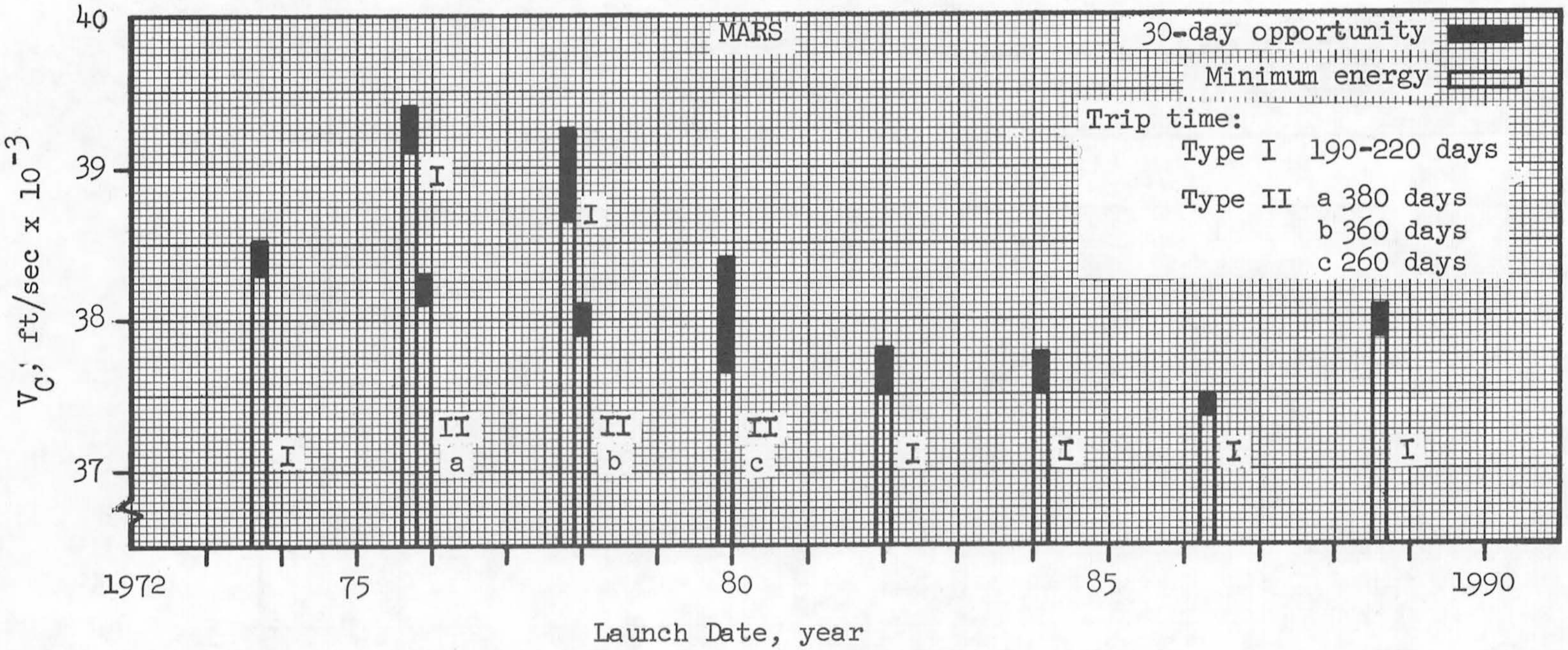


FIGURE II-10. CHARACTERISTIC VELOCITY REQUIREMENTS FOR VENUS AND MARS

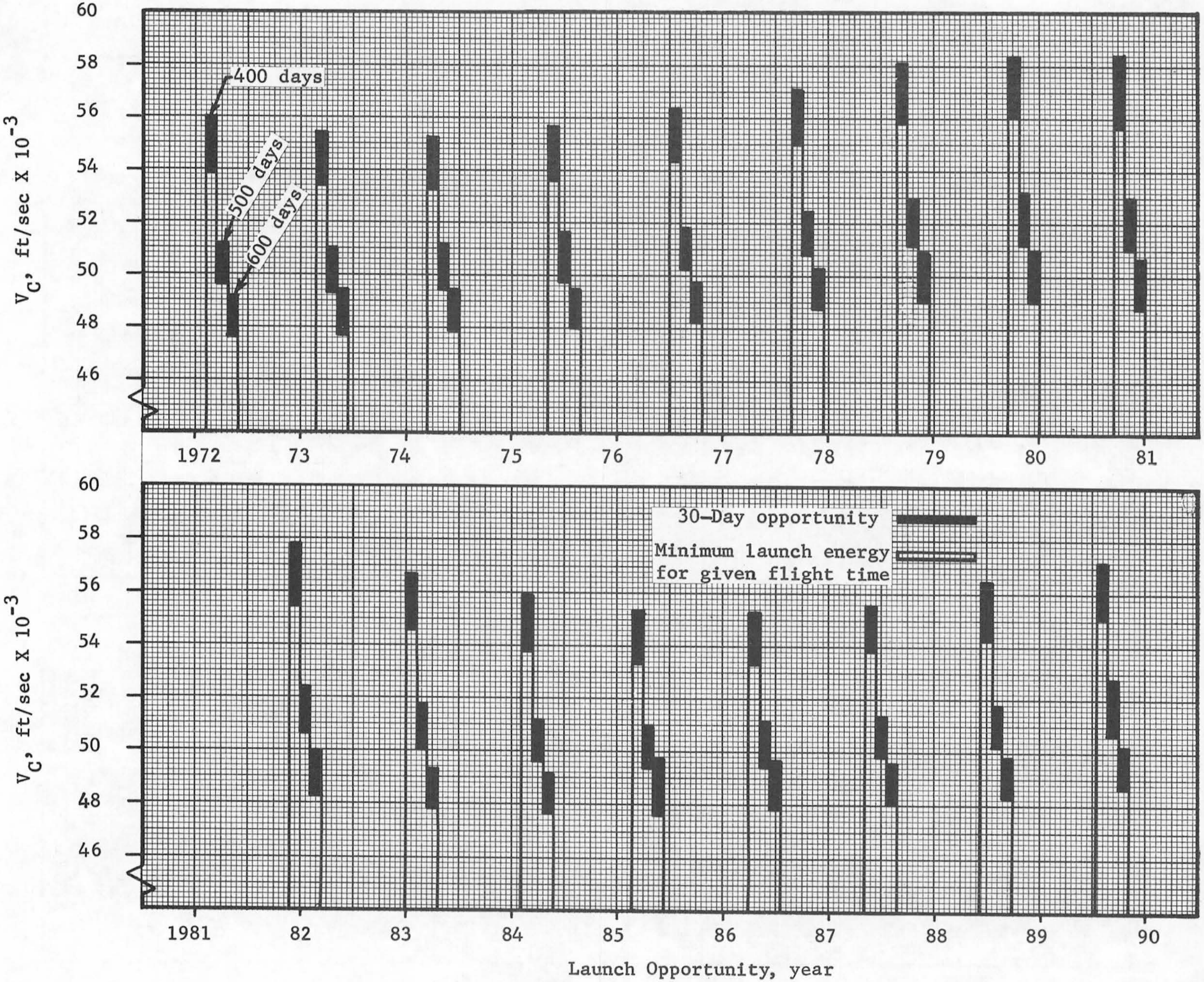


FIGURE II-11. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER
(400, 500, AND 600 DAY FLIGHT TIMES)

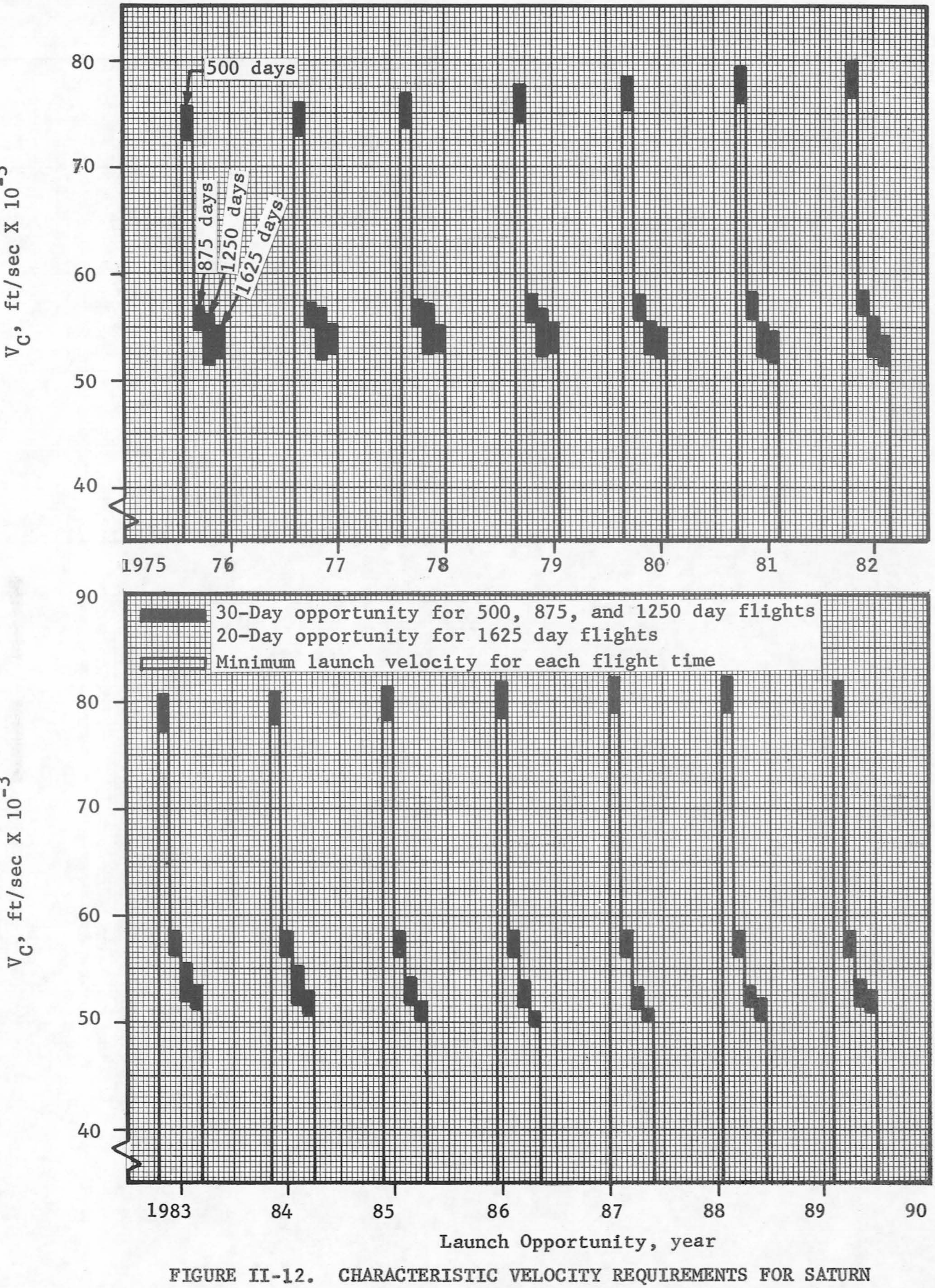


FIGURE II-12. CHARACTERISTIC VELOCITY REQUIREMENTS FOR SATURN (500, 875, 1250, AND 1625 DAY FLIGHT TIMES)

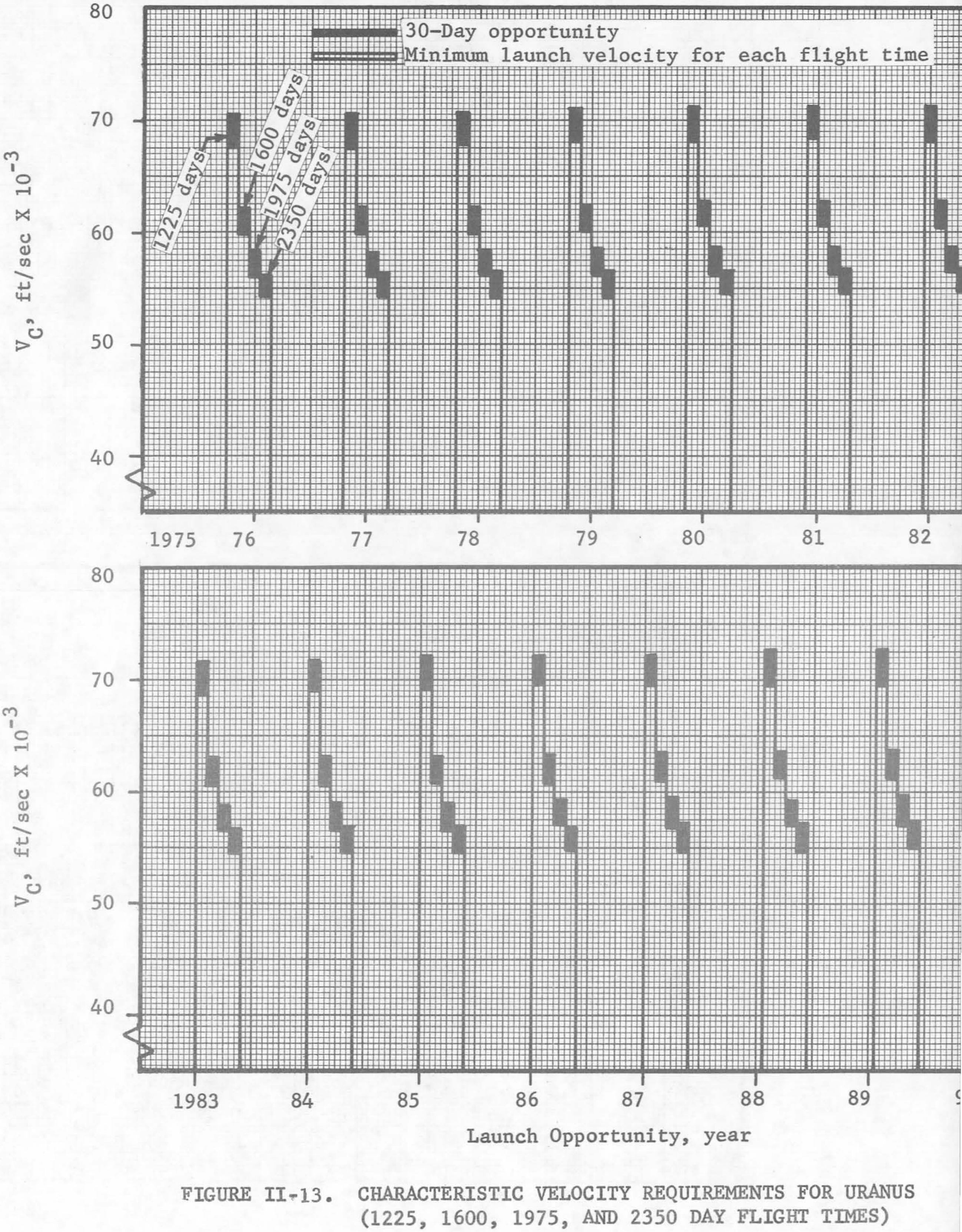


FIGURE II-13. CHARACTERISTIC VELOCITY REQUIREMENTS FOR URANUS (1225, 1600, 1975, AND 2350 DAY FLIGHT TIMES)

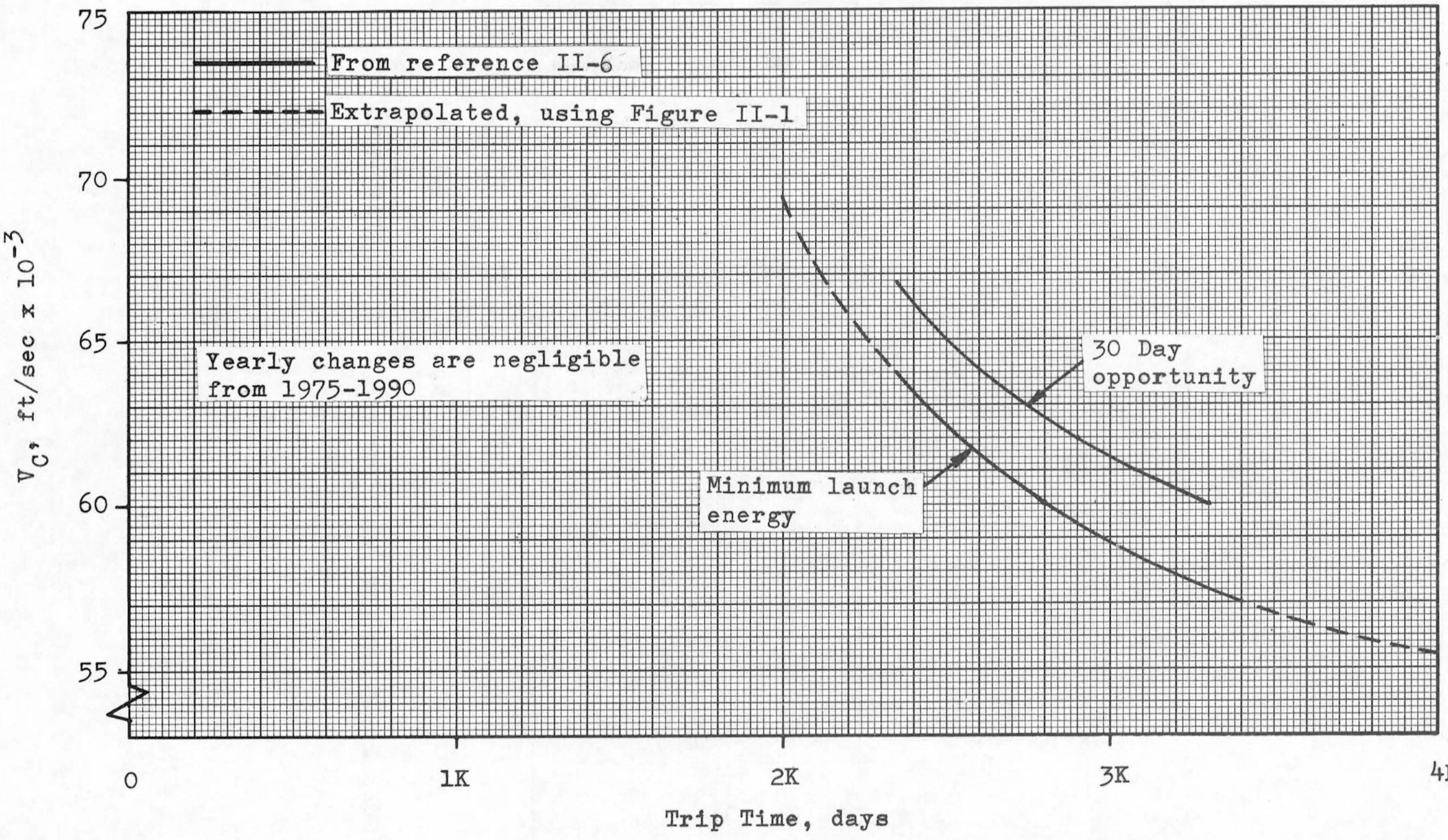


FIGURE II-14. CHARACTERISTIC VELOCITY REQUIREMENTS FOR NEPTUNE

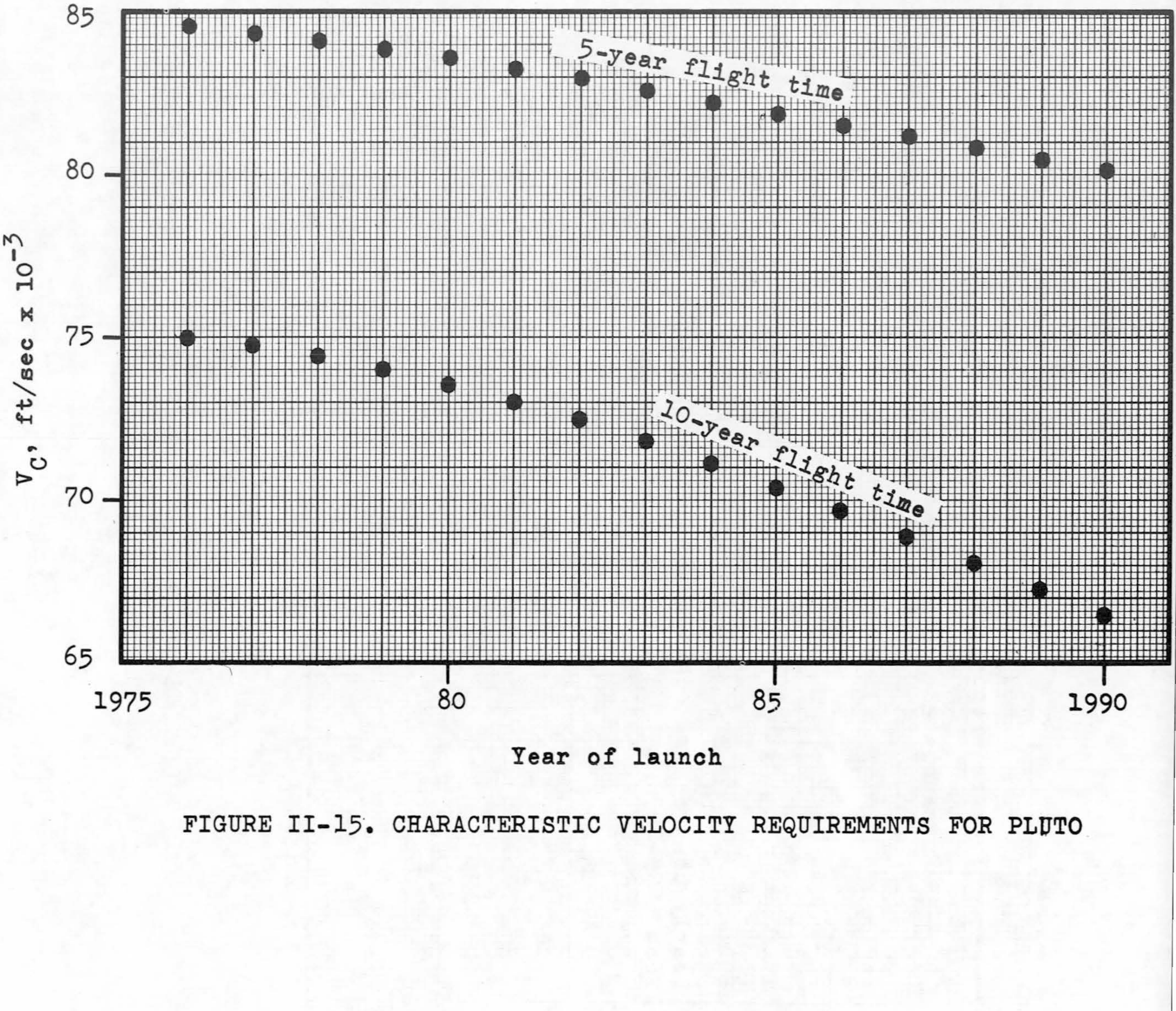
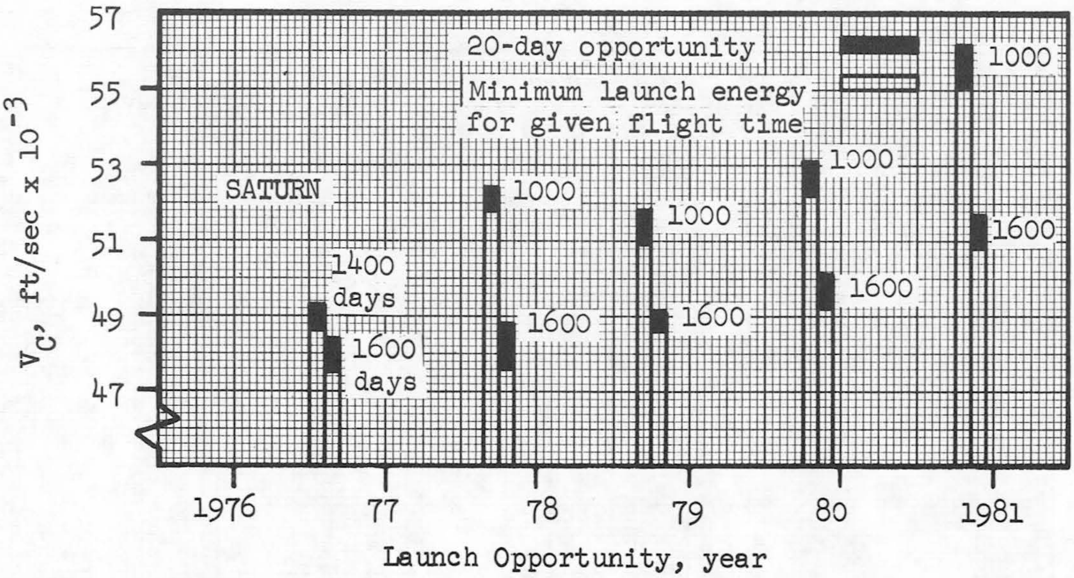


FIGURE II-15. CHARACTERISTIC VELOCITY REQUIREMENTS FOR PLUTO



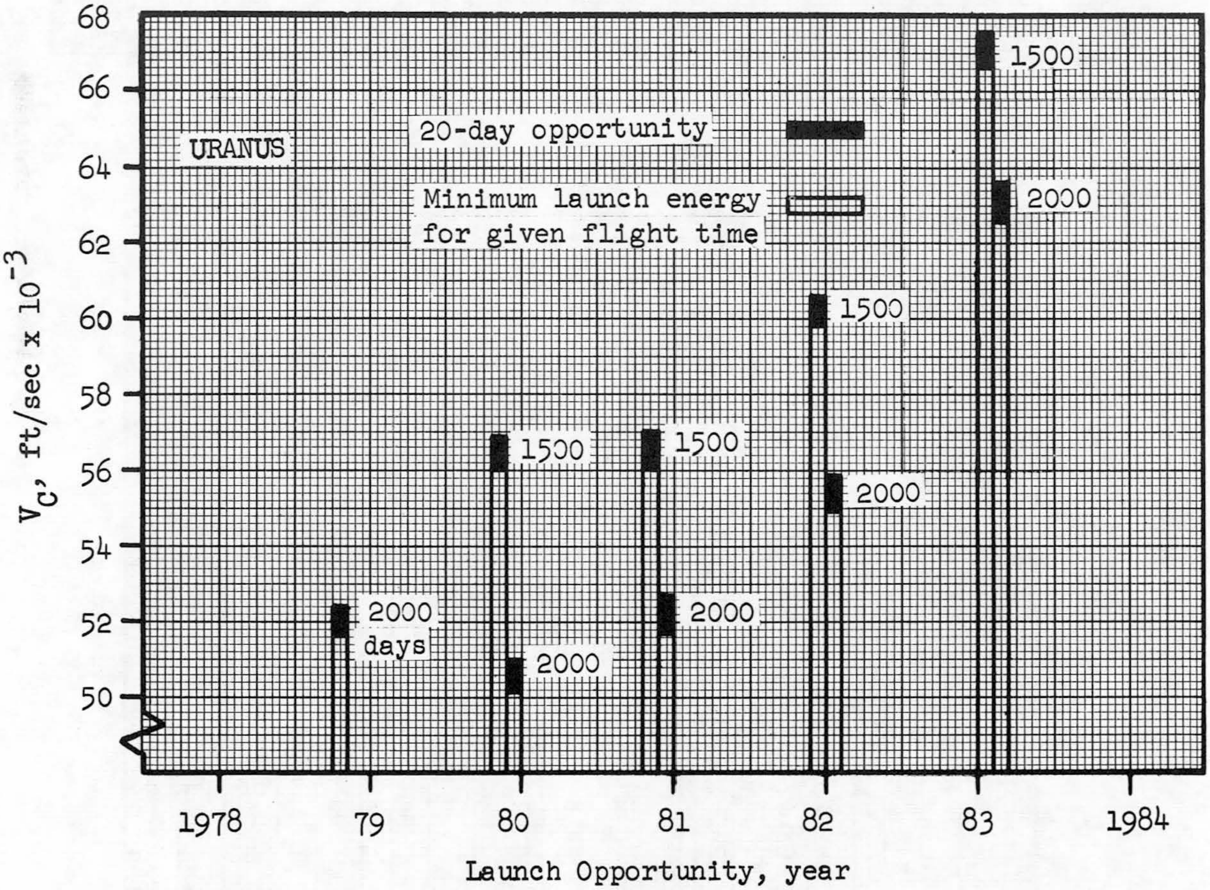
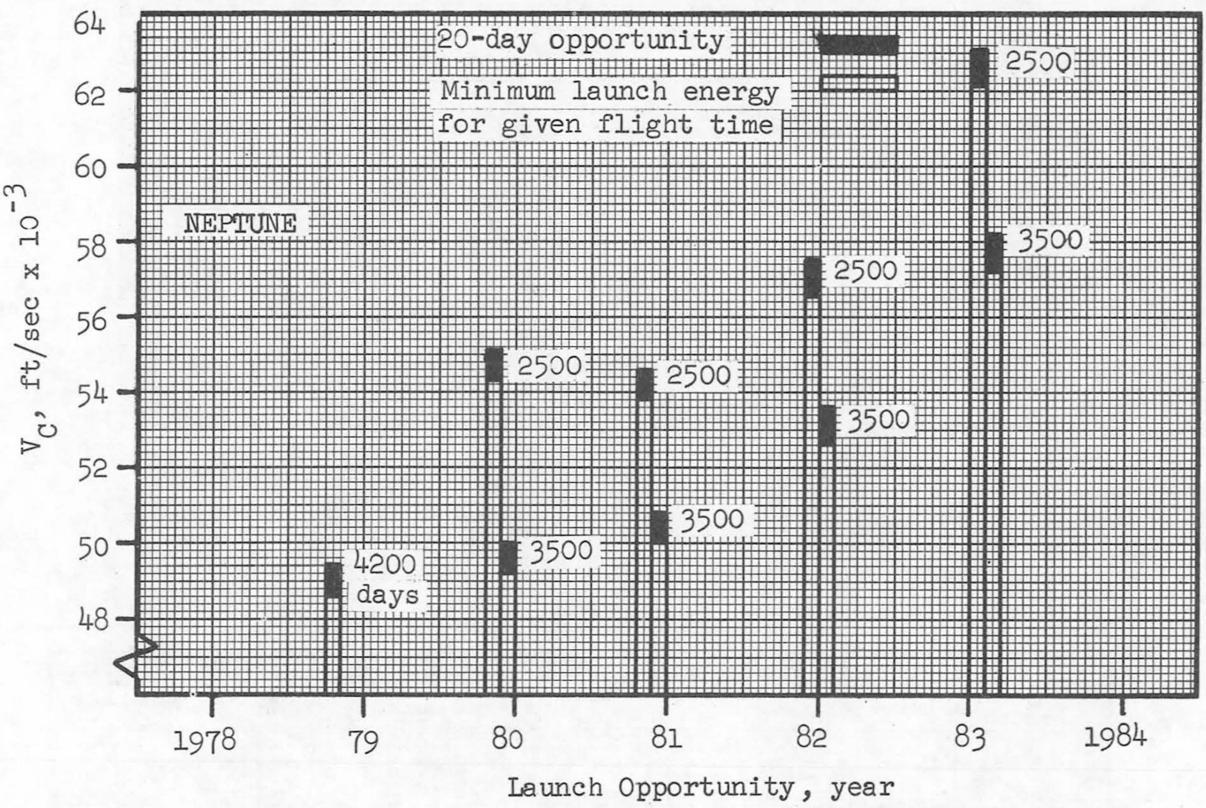


FIGURE II-16. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER SWINGBYS TO SATURN AND URANUS



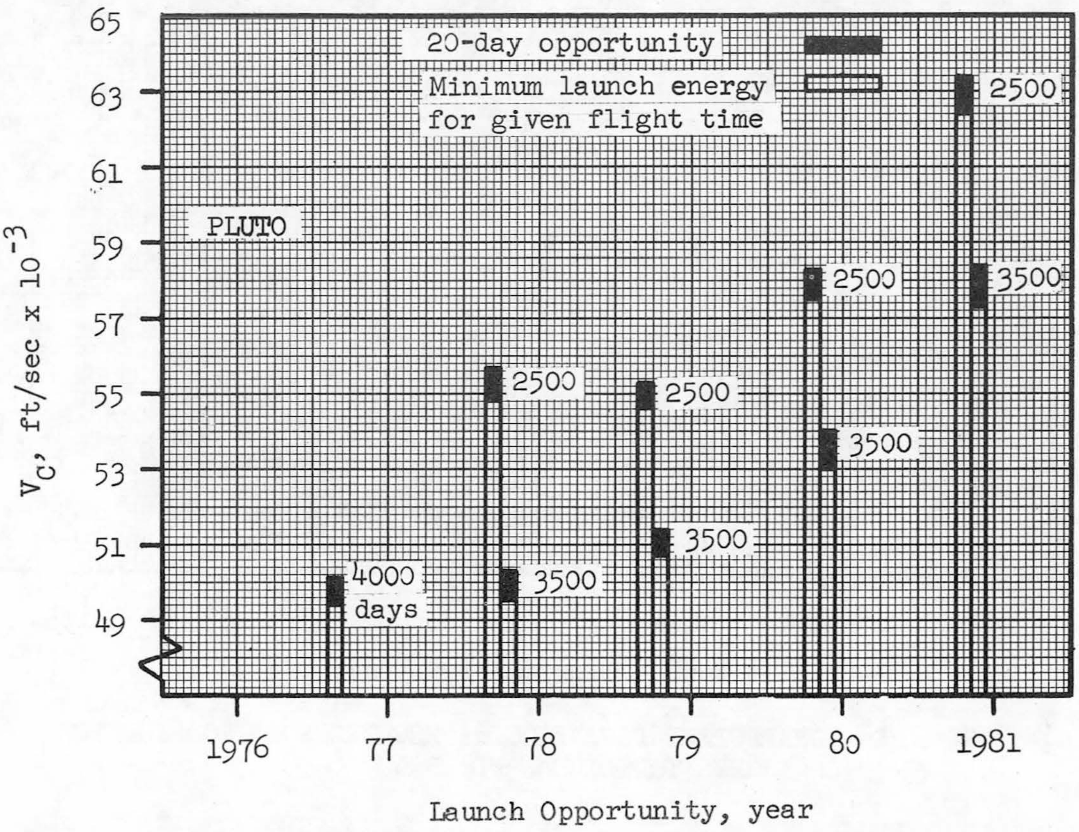


FIGURE II-17. CHARACTERISTIC VELOCITY REQUIREMENTS FOR JUPITER SWINGBYS TO NEPTUNE AND PLUTO

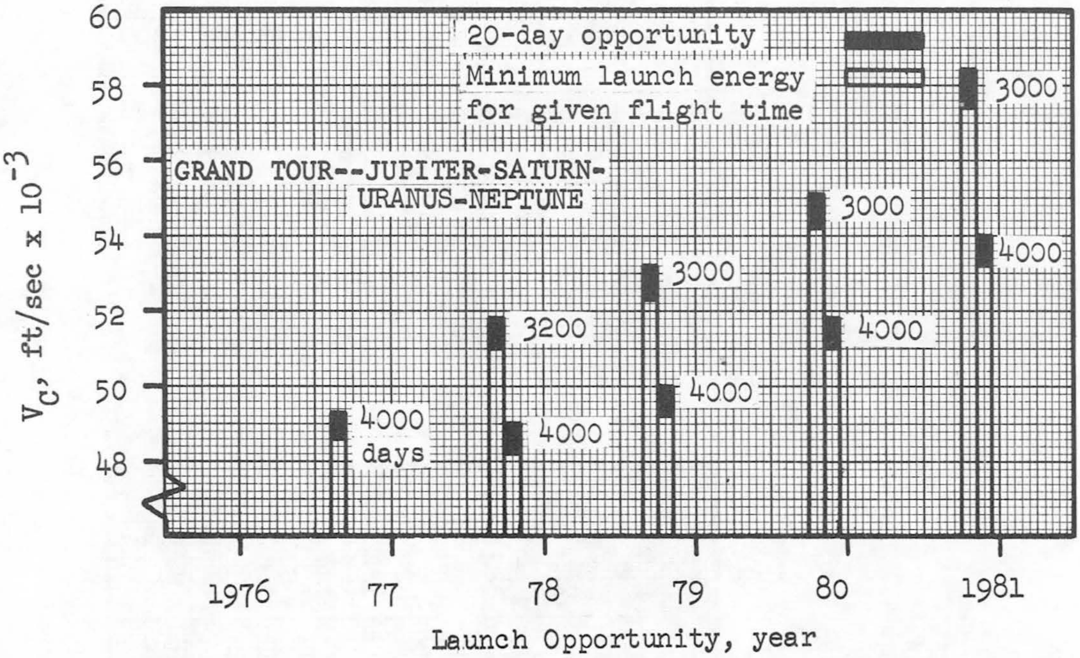
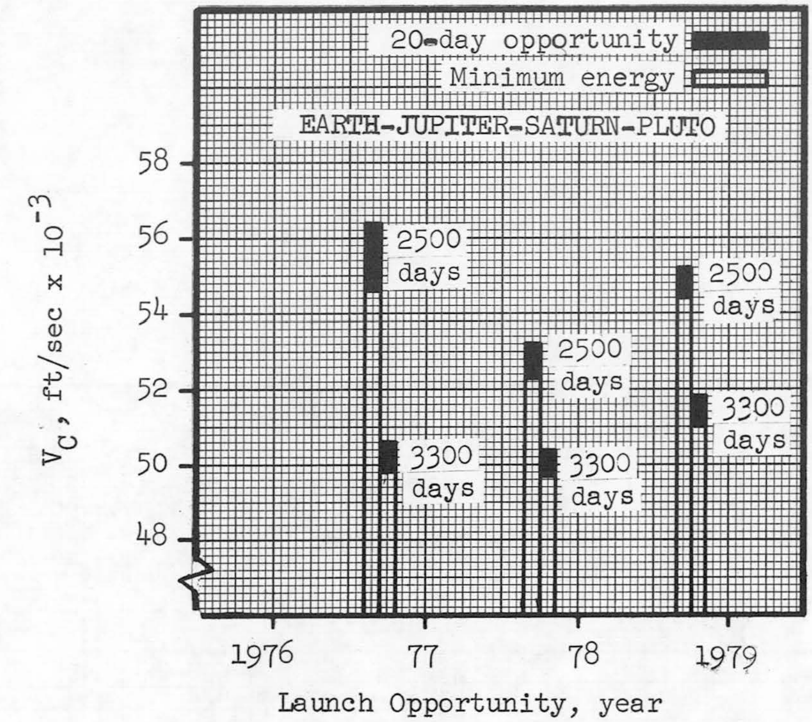


FIGURE II-18. CHARACTERISTIC VELOCITY REQUIREMENTS FOR GRAND TOUR MISSIONS



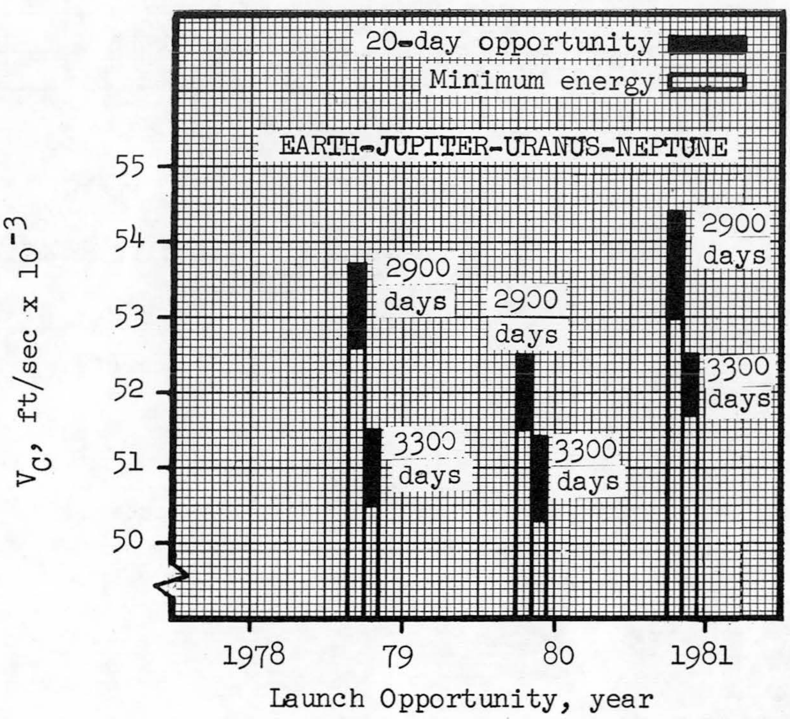


FIGURE II-19. CHARACTERISTIC VELOCITY REQUIREMENTS FOR THREE PLANET JUPITER SWINGBYS

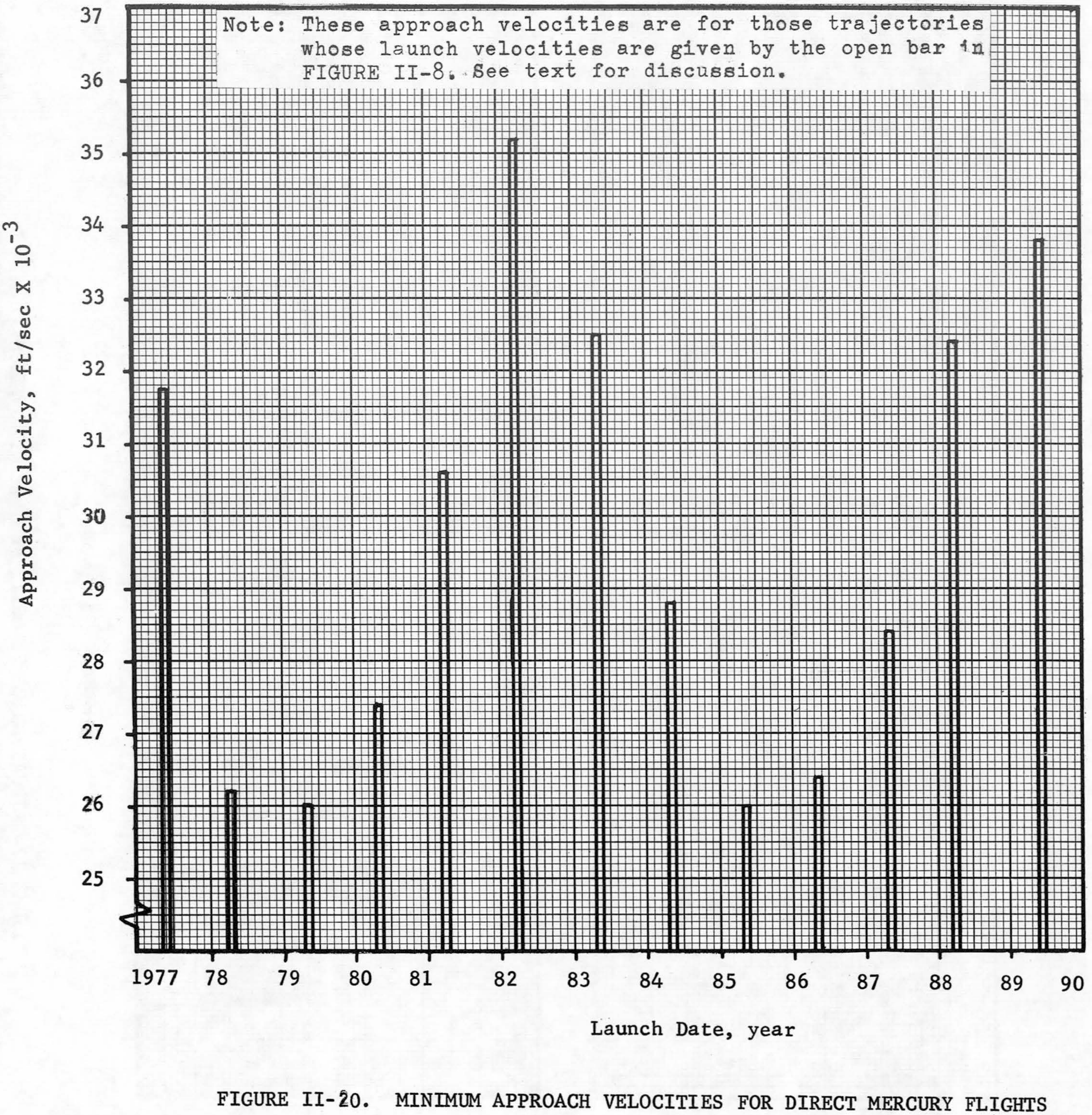


FIGURE II-20. MINIMUM APPROACH VELOCITIES FOR DIRECT MERCURY FLIGHTS

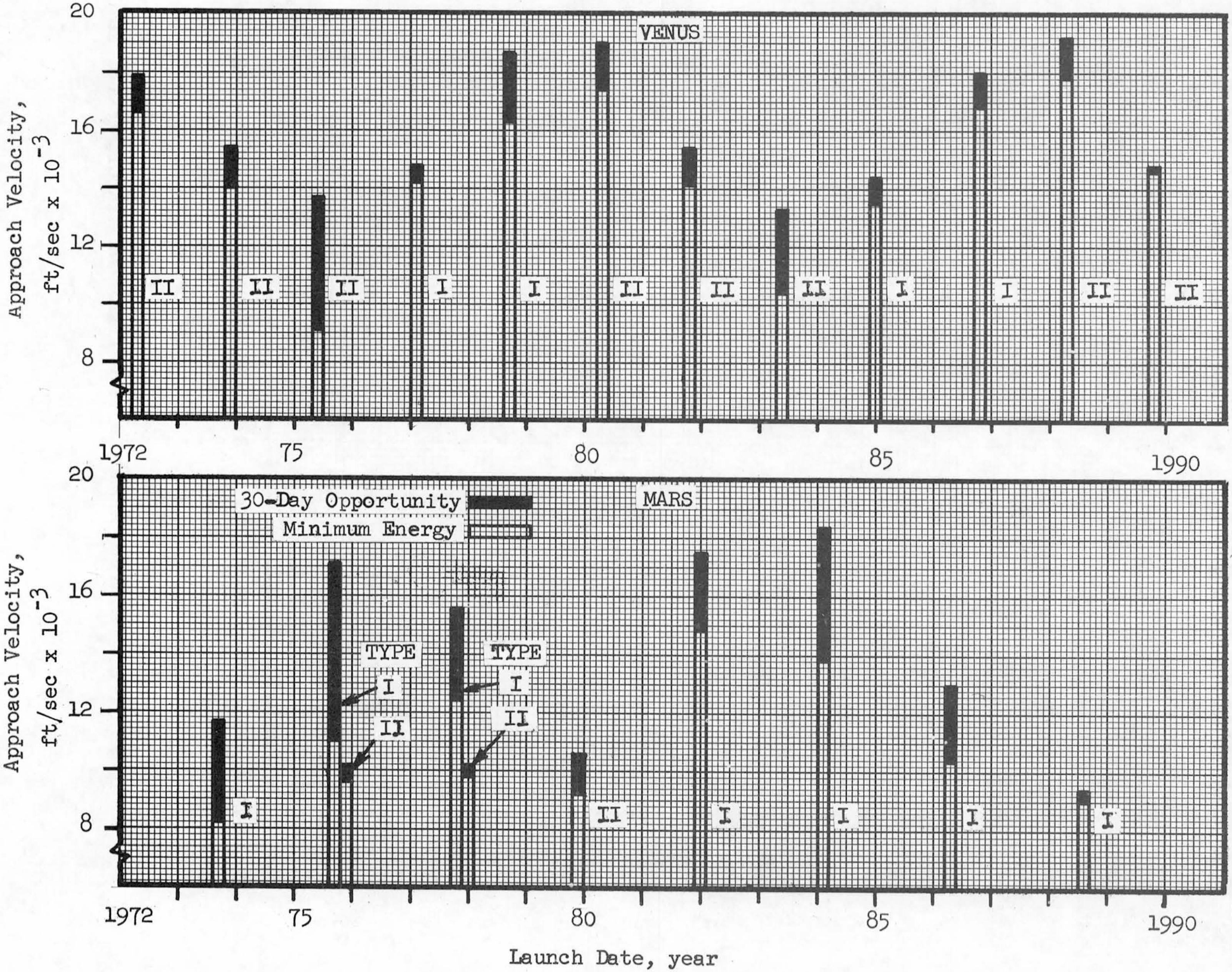


FIGURE II-21. APPROACH VELOCITY FOR MARS AND VENUS

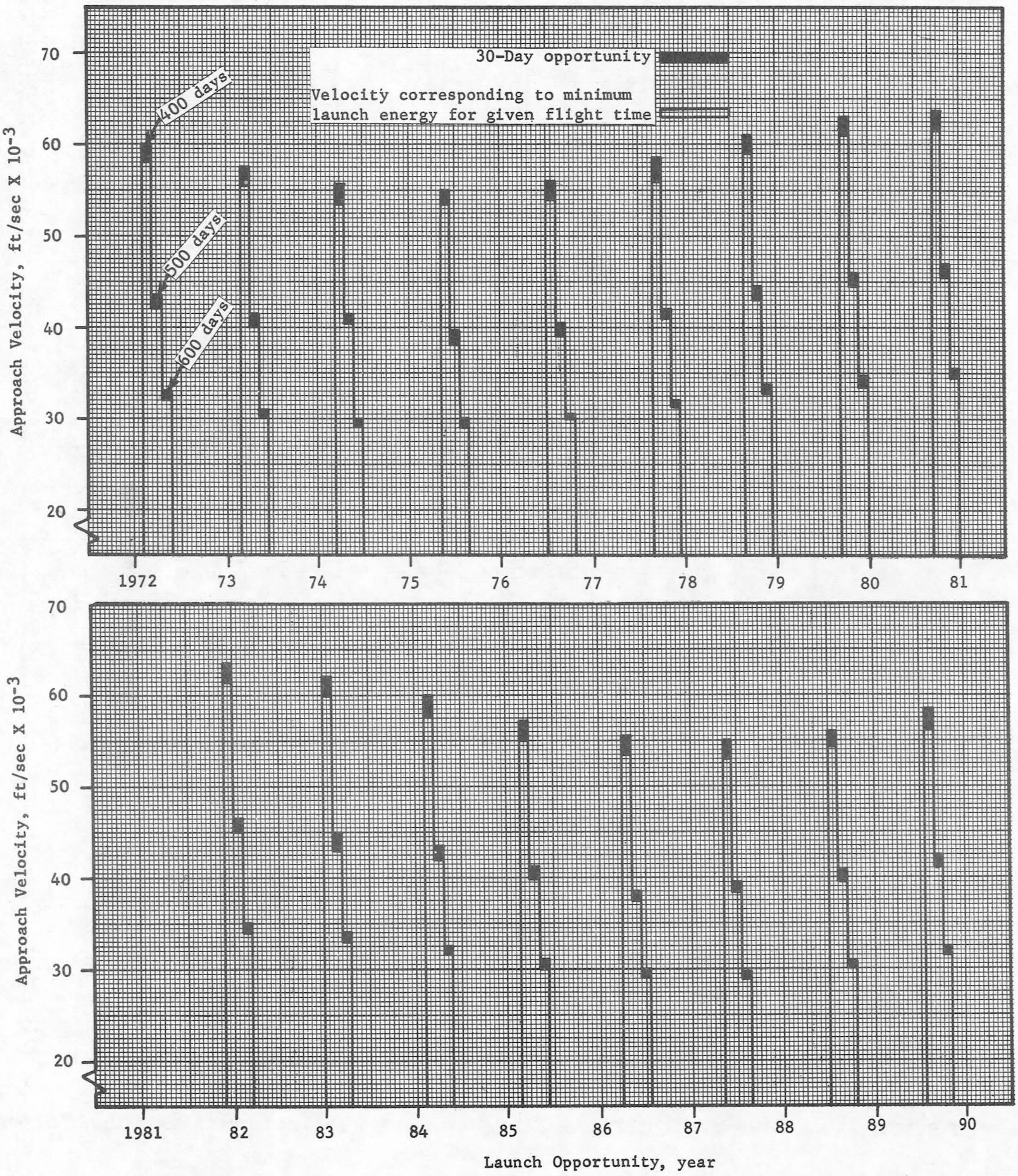


FIGURE II-22. APPROACH VELOCITY FOR JUPITER (400, 500, AND 600 DAY FLIGHT TIMES)

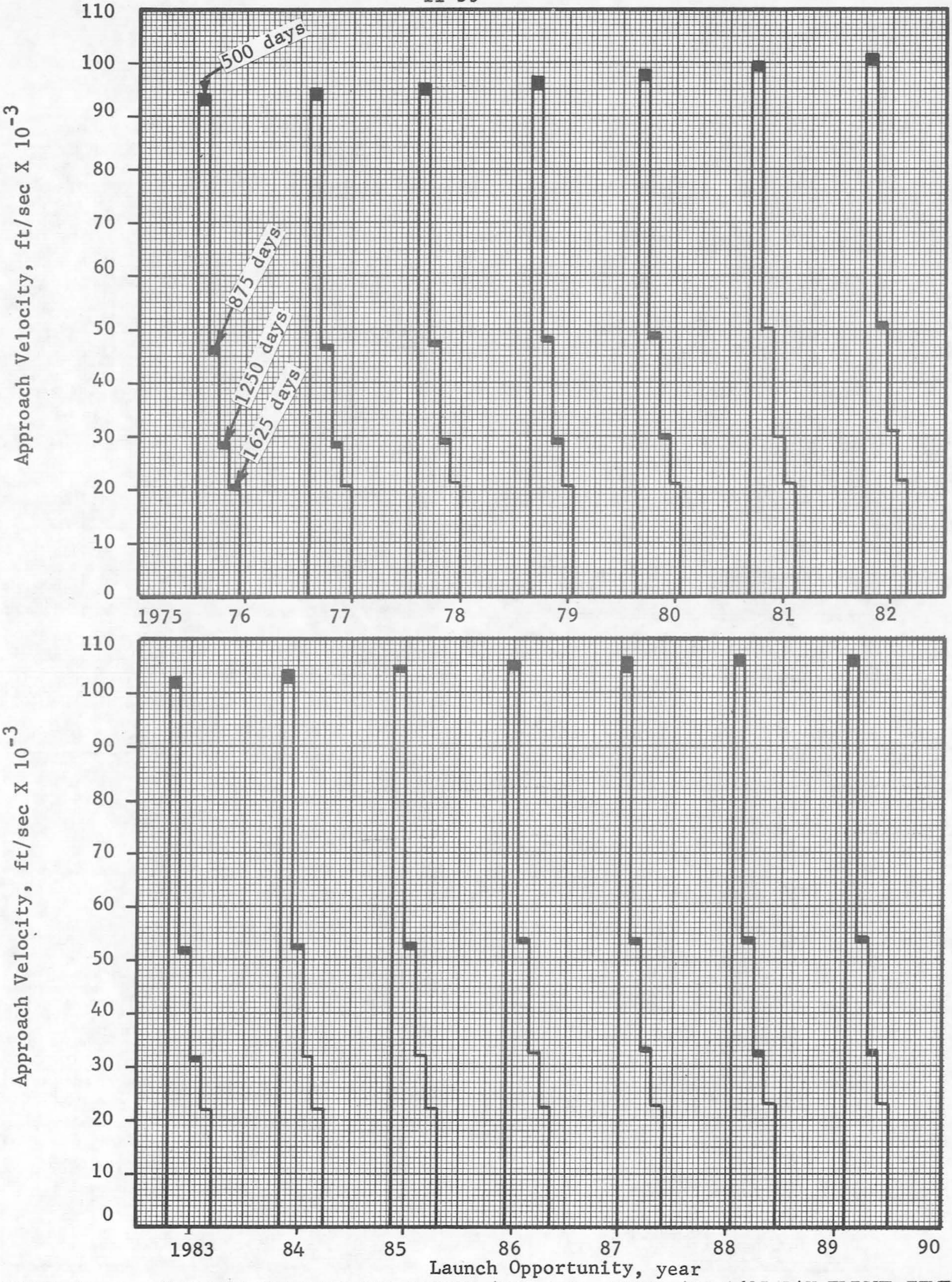


FIGURE II-23. APPROACH VELOCITY FOR SATURN (500, 875, 1250, AND 1625 DAY FLIGHT TIMES)

30-Day opportunity (when not shown, increment is negligible)

Velocity corresponding to minimum launch energy for given flight time

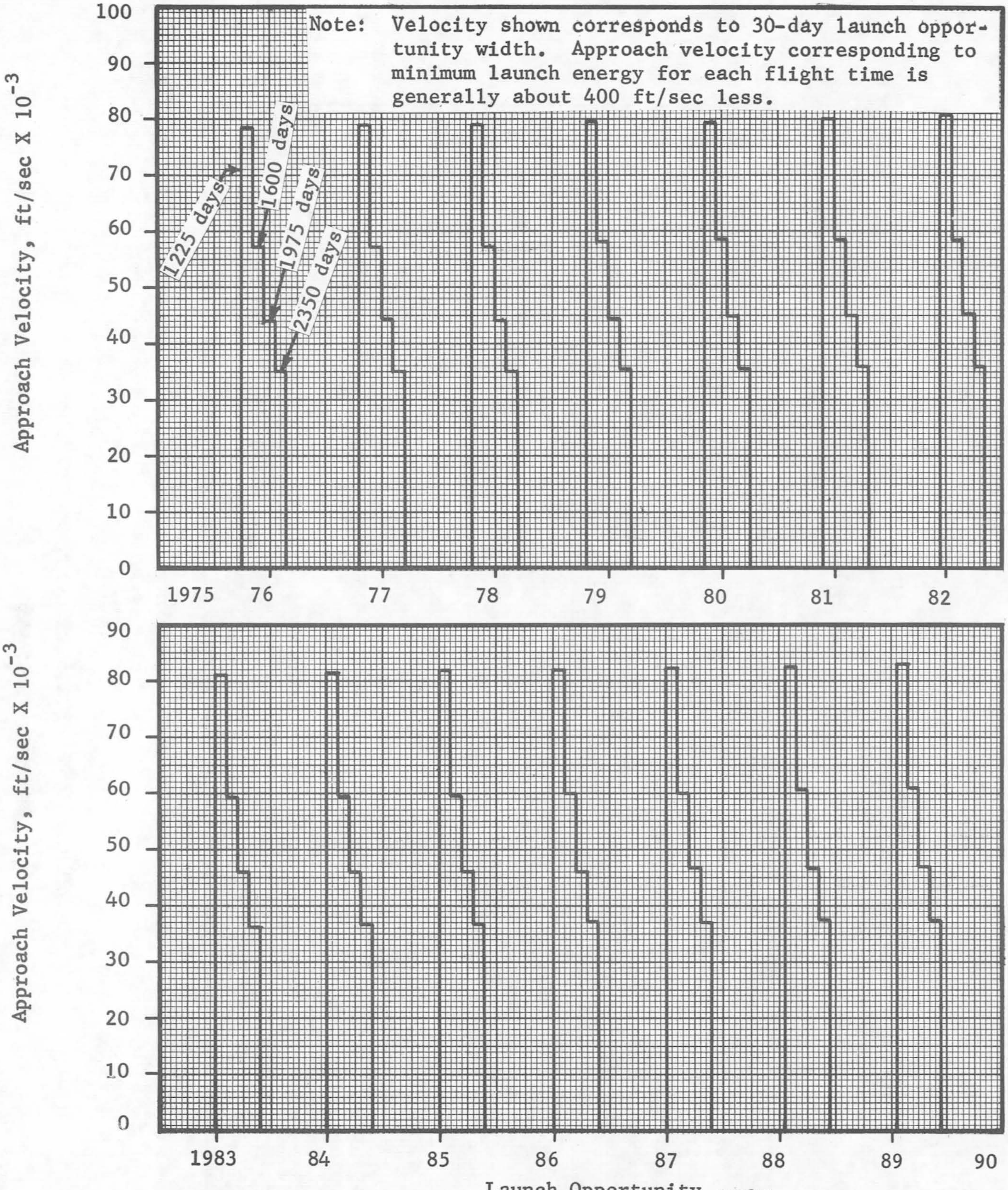


FIGURE II-24. APPROACH VELOCITY FOR URANUS (1225, 1600, 1975, AND 2350 DAY FLIGHT TIMES)

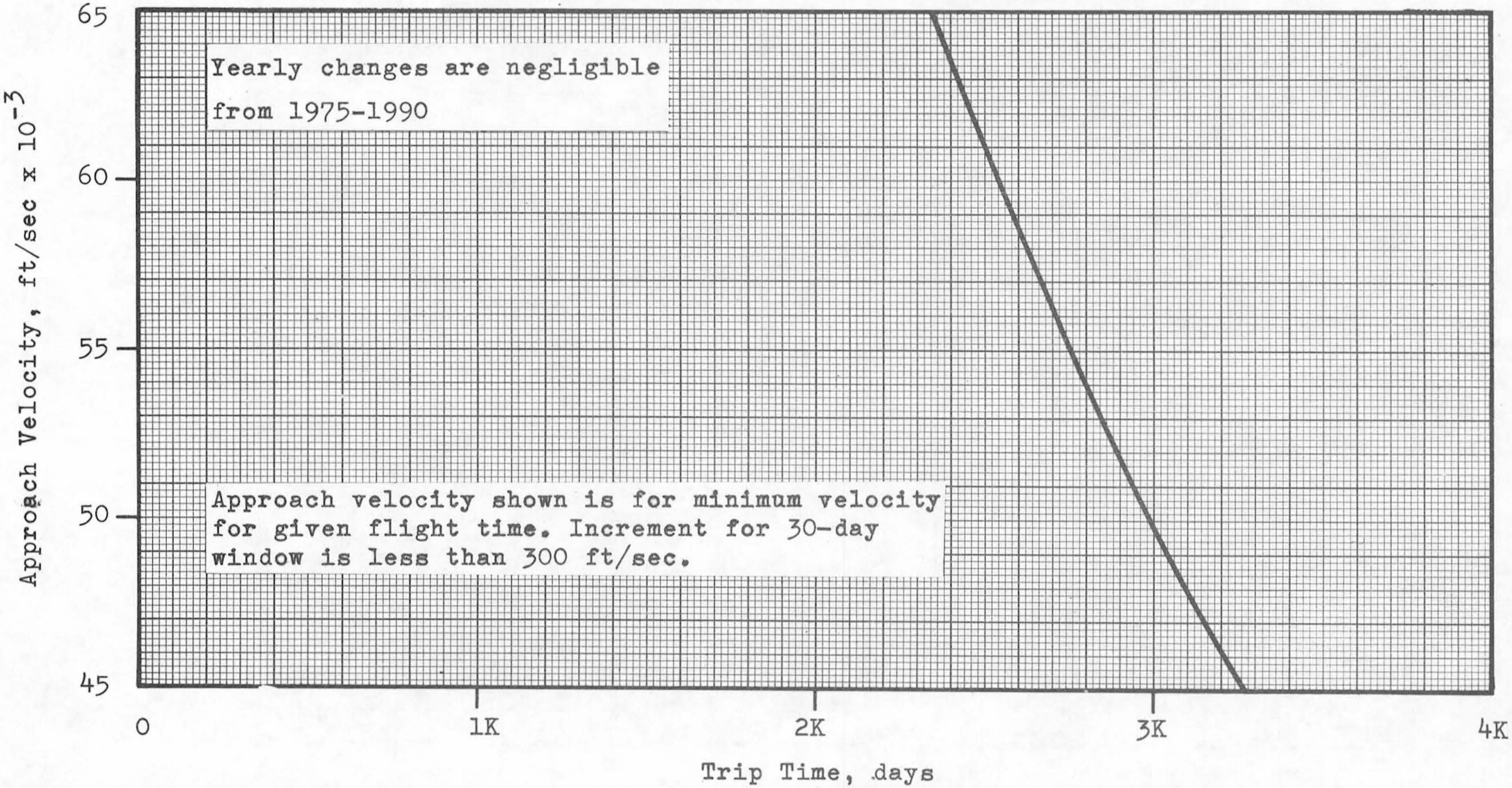


FIGURE II-25. APPROACH VELOCITY FOR NEPTUNE

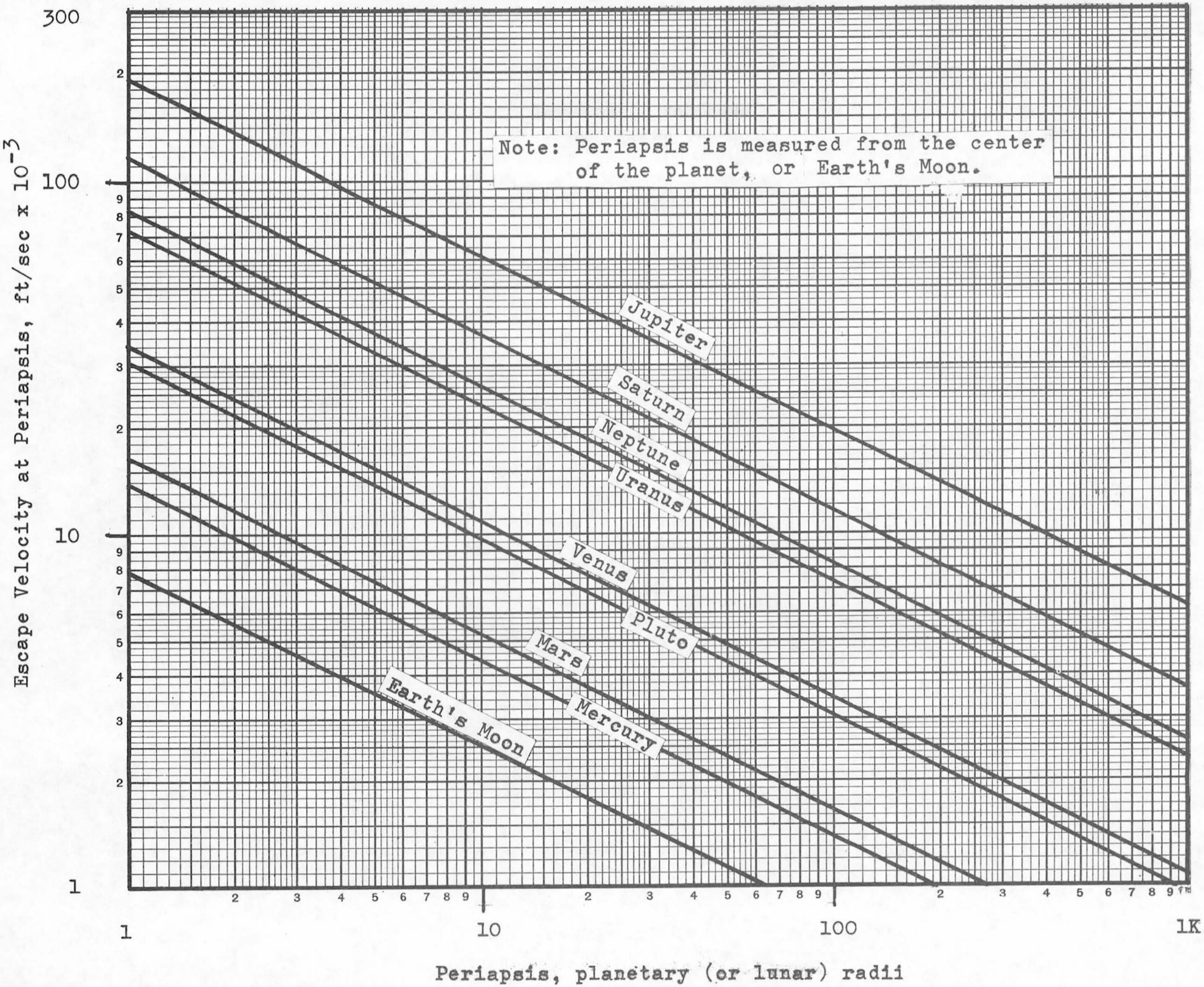


FIGURE II-26. ESCAPE VELOCITIES FOR THE PLANETS AND EARTH'S MOON

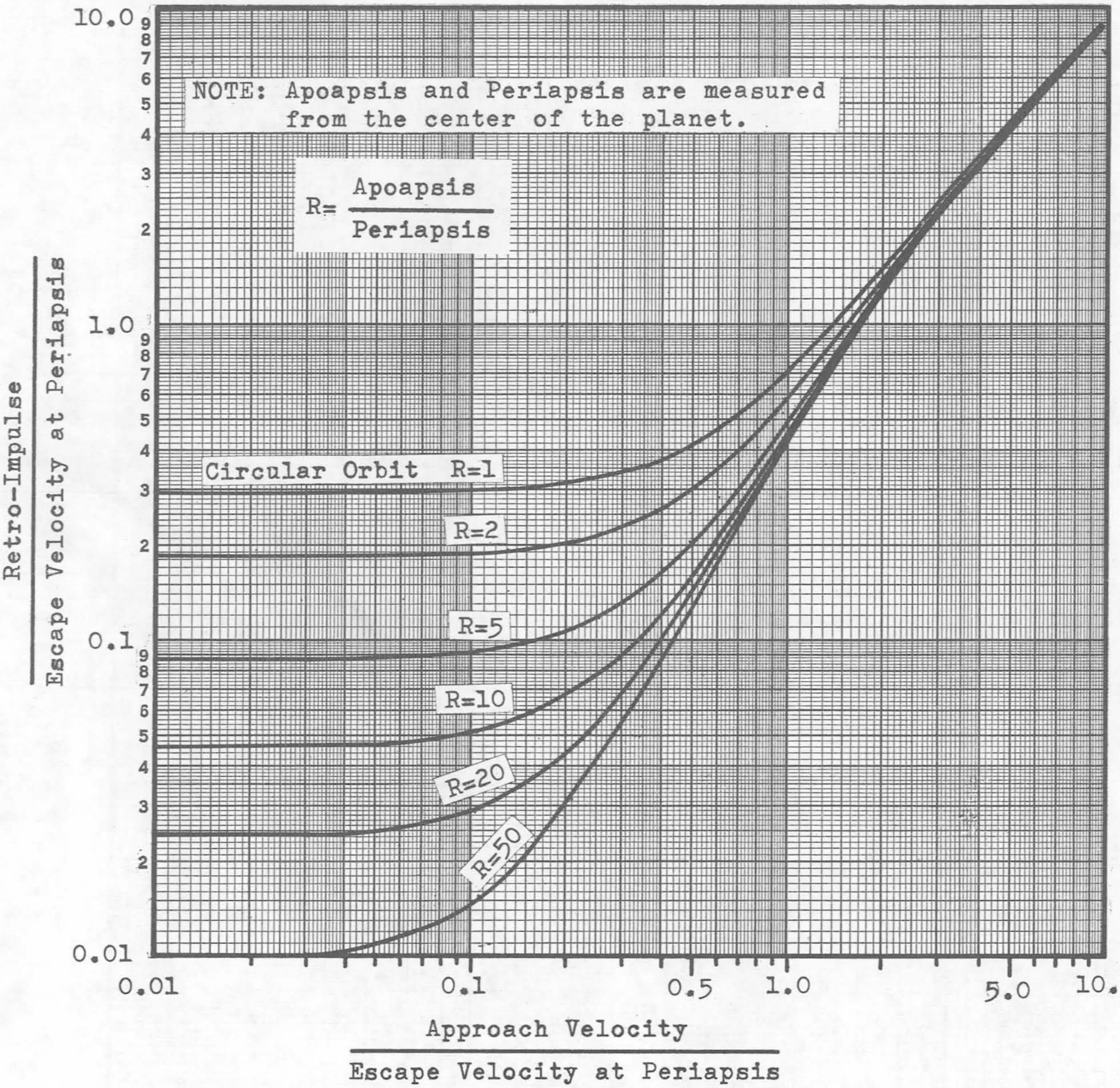


FIGURE II-27a. RETRO-IMPULSE REQUIREMENTS VERSUS APPROACH VELOCITY FOR VARIOUS SHAPED ORBITS

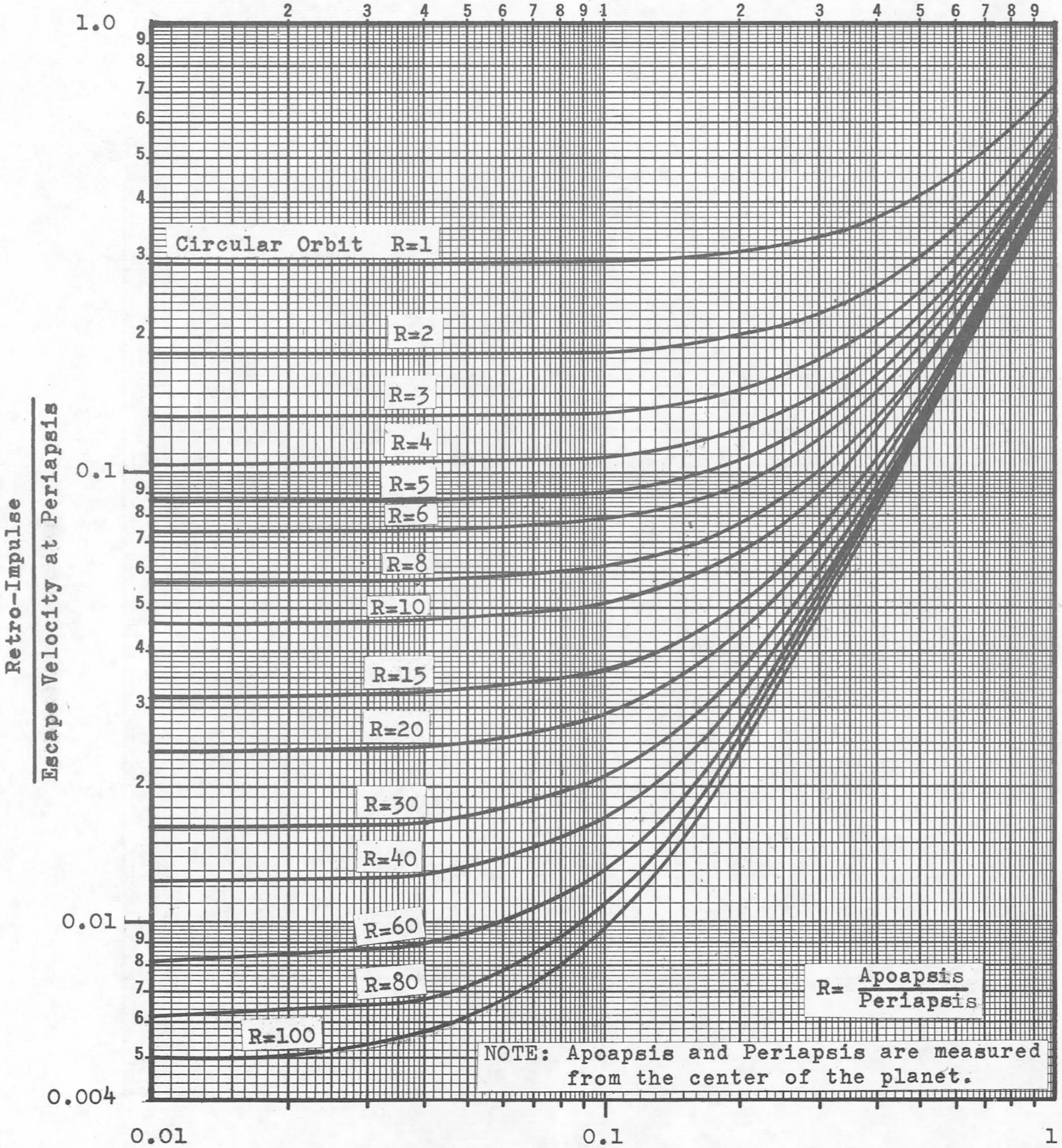


FIGURE II-27b. RETRO-IMPULSE REQUIREMENTS VERSUS APPROACH VELOCITY FOR VARIOUS SHAPED ORBITS

(An expanded scale of FIGURE II- 27a)

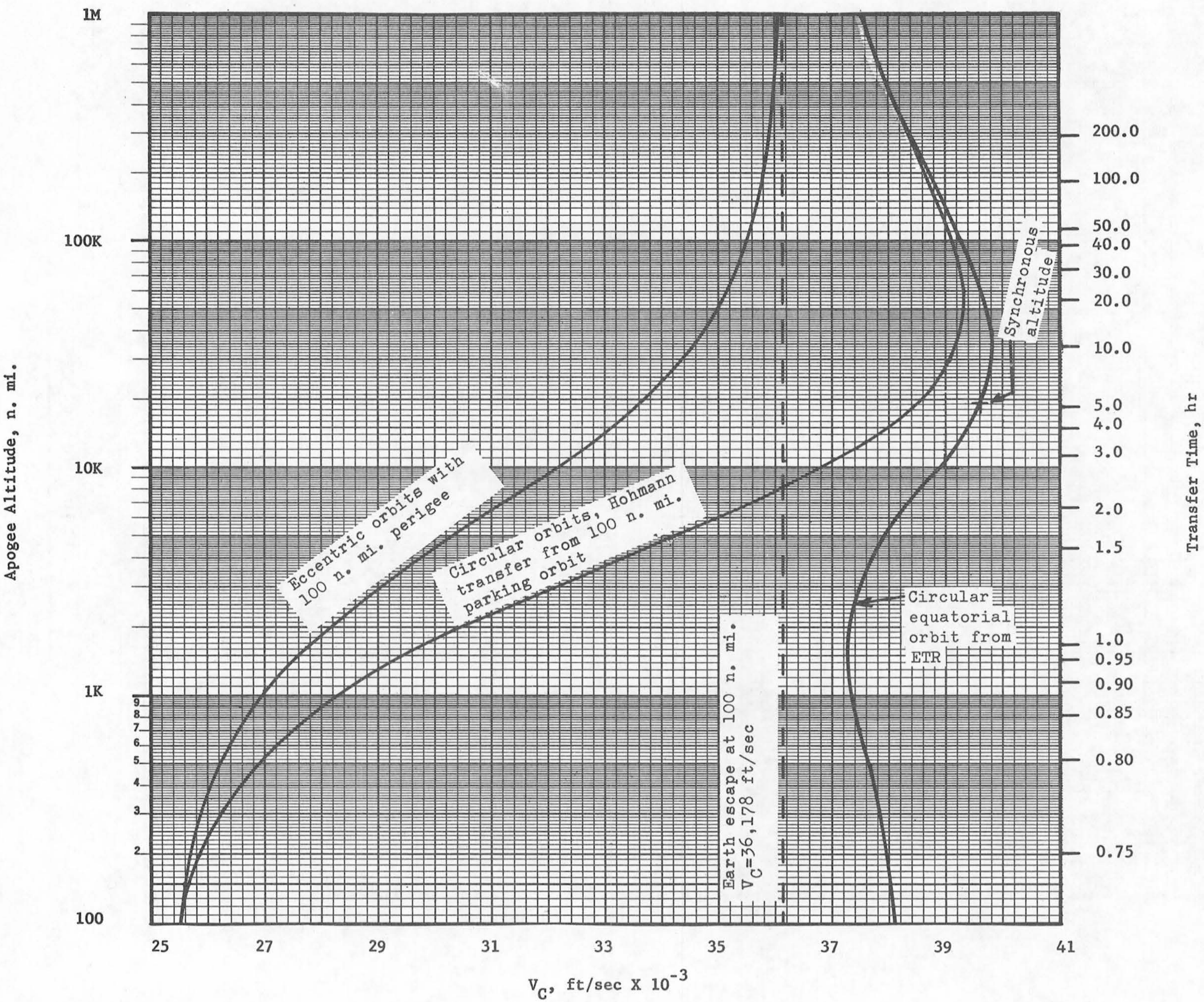


FIGURE II-28. VELOCITY REQUIRED FOR EARTH ORBITS

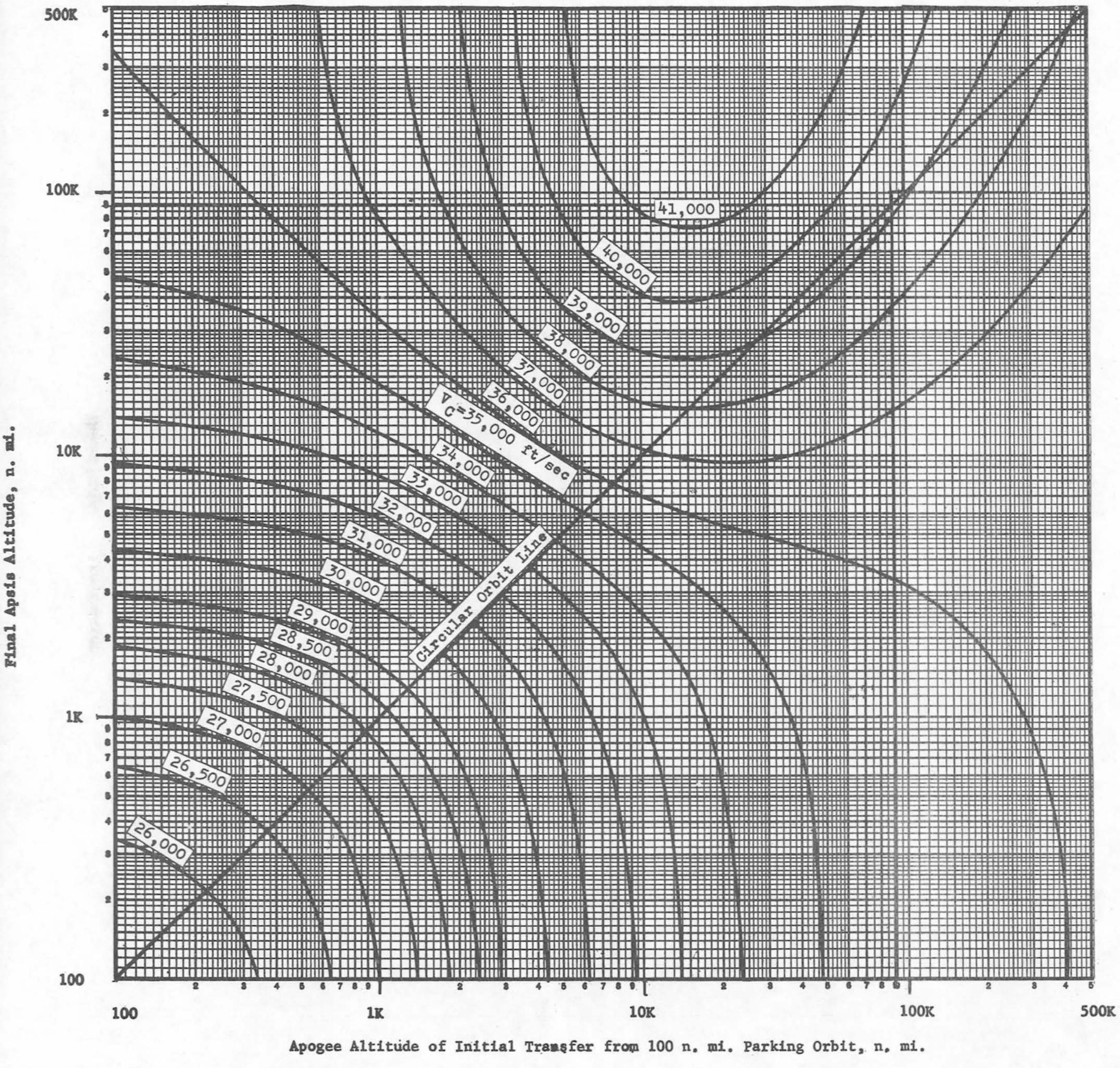


FIGURE II-29. TOTAL V_C REQUIREMENTS ASSUMING TWO-IMPULSE TRANSFER FROM 100 N. MI. ORBIT

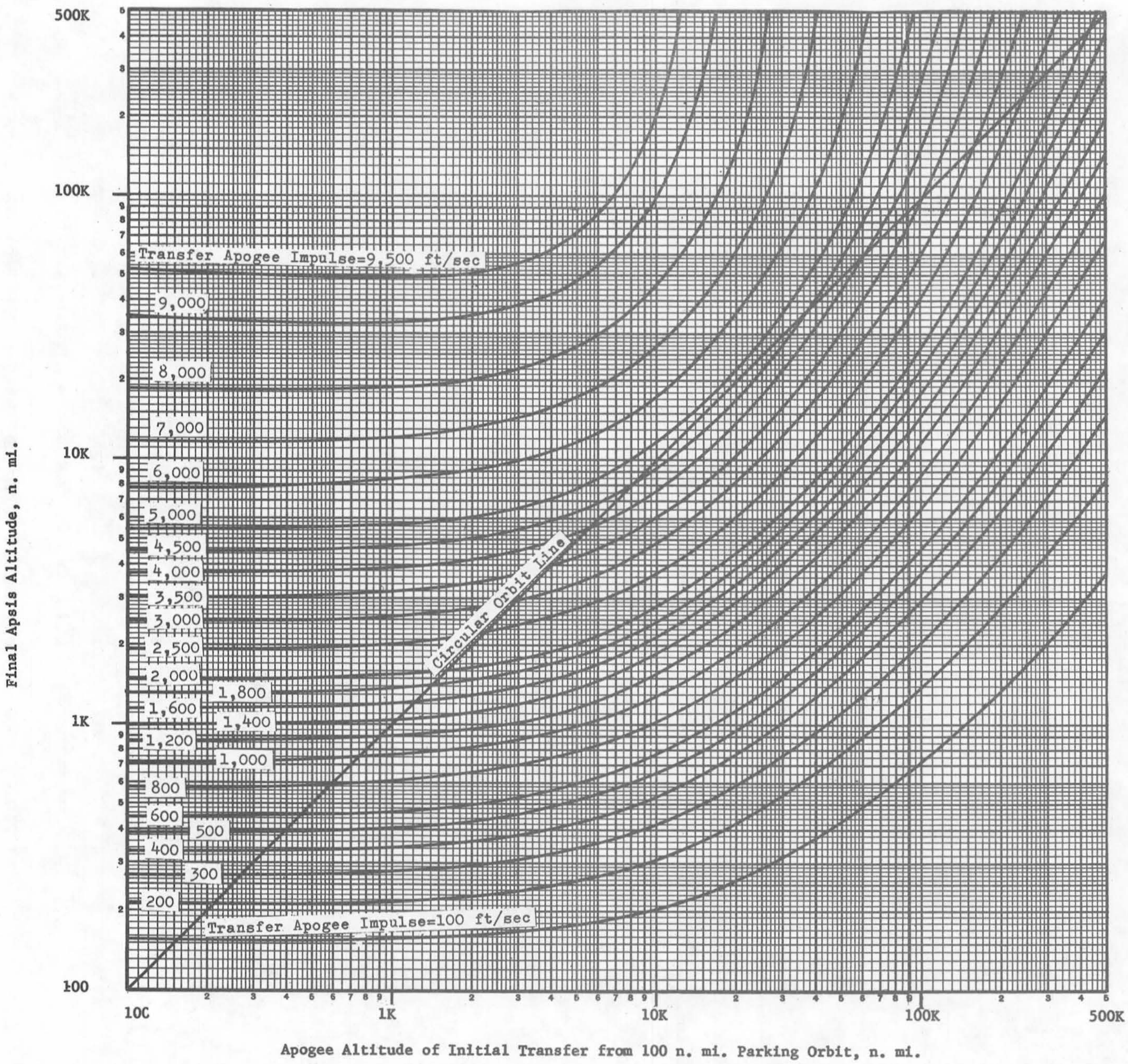


FIGURE II-30. TRANSFER APOGEE IMPULSE REQUIREMENTS AFTER TRANSFER FROM 100 N. MI. ORBIT

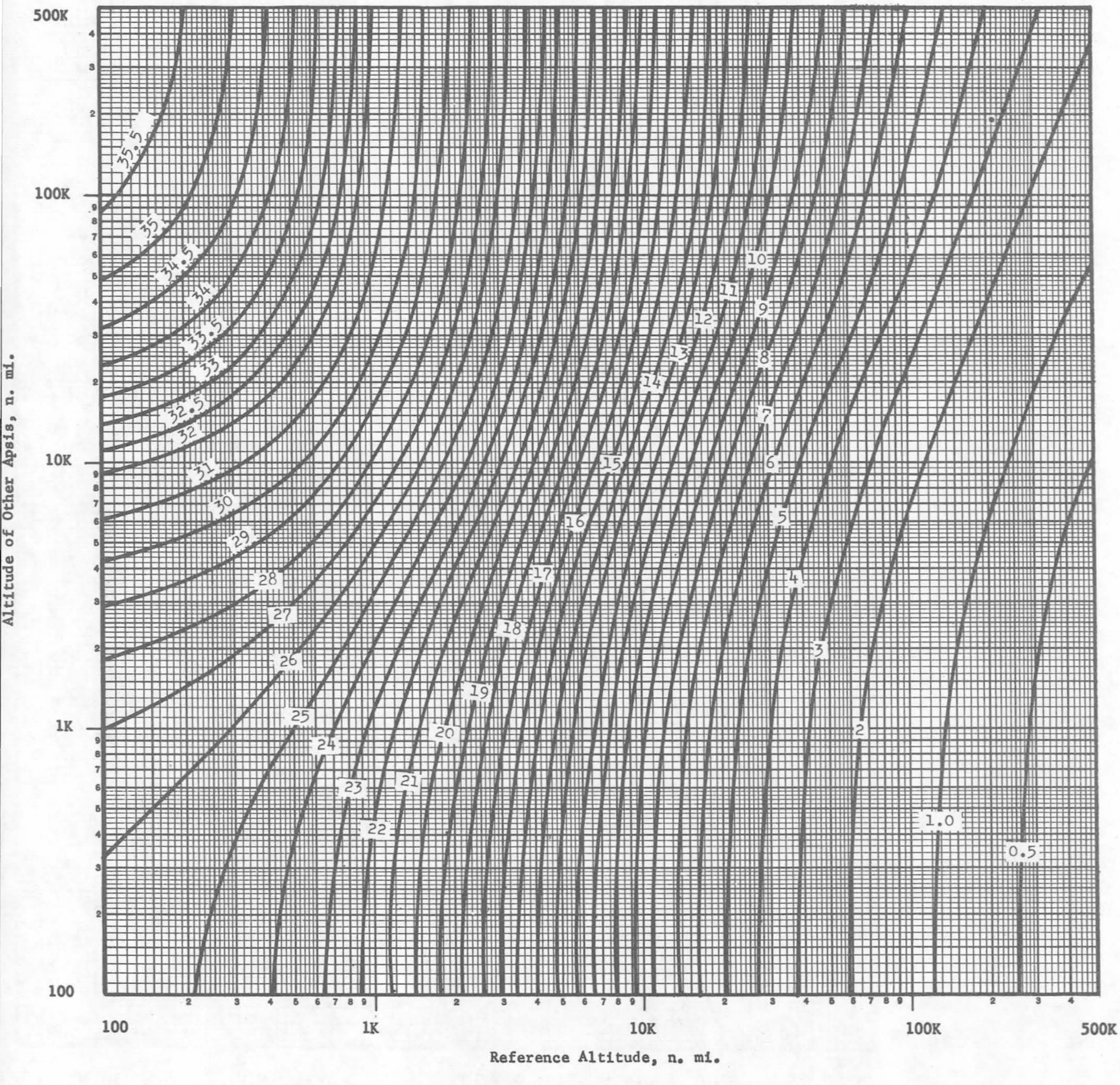


FIGURE II-31a. HORIZONTAL VELOCITY REQUIRED AT REFERENCE ALTITUDE TO ESTABLISH OTHER APSIS

Note: Numbers on curves indicate velocity in ft/sec $\times 10^{-3}$

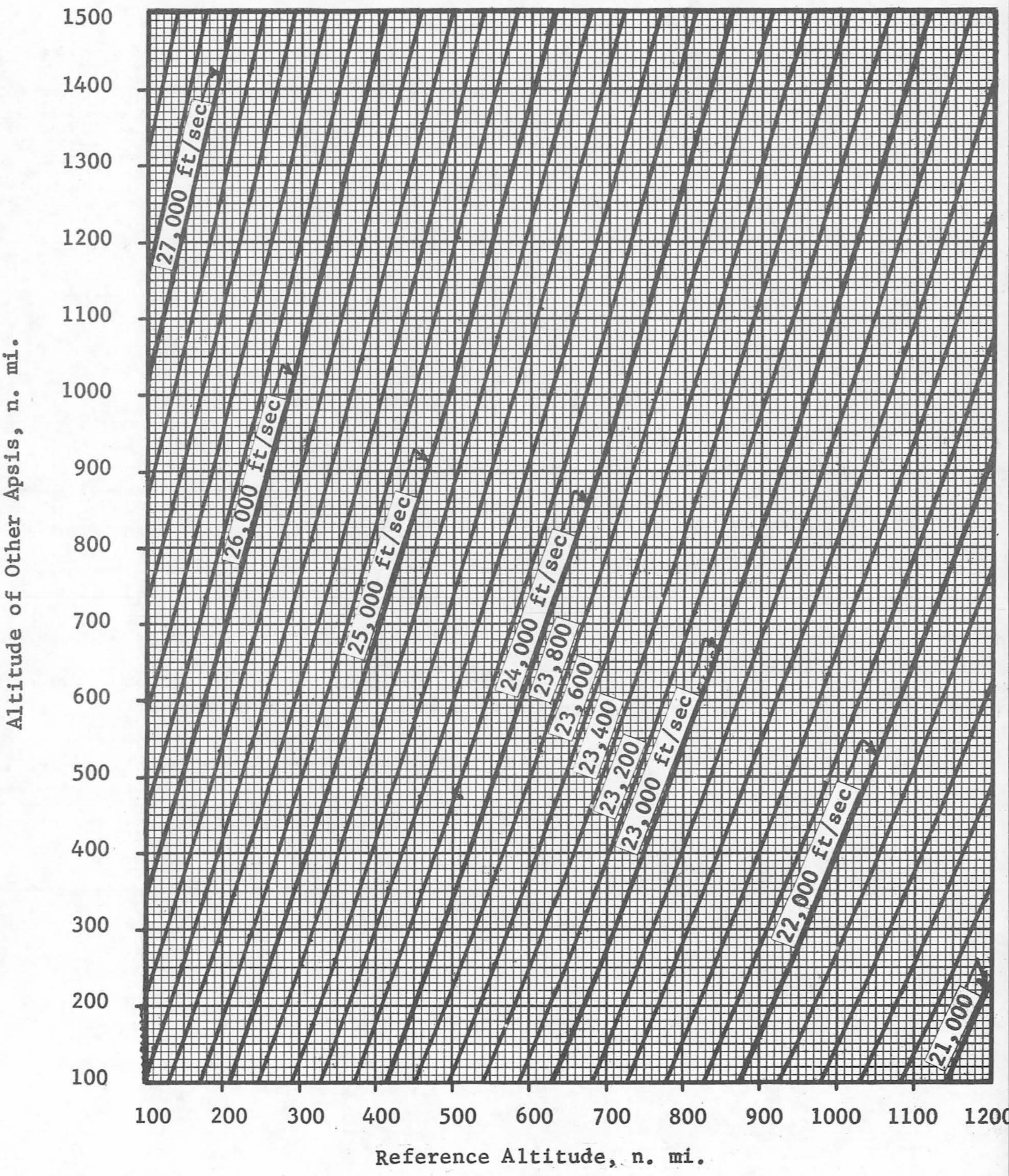


FIGURE II-31b. HORIZONTAL VELOCITY REQUIRED AT REFERENCE ALTITUDE TO ESTABLISH OTHER APSIS (LOWER ALTITUDES)

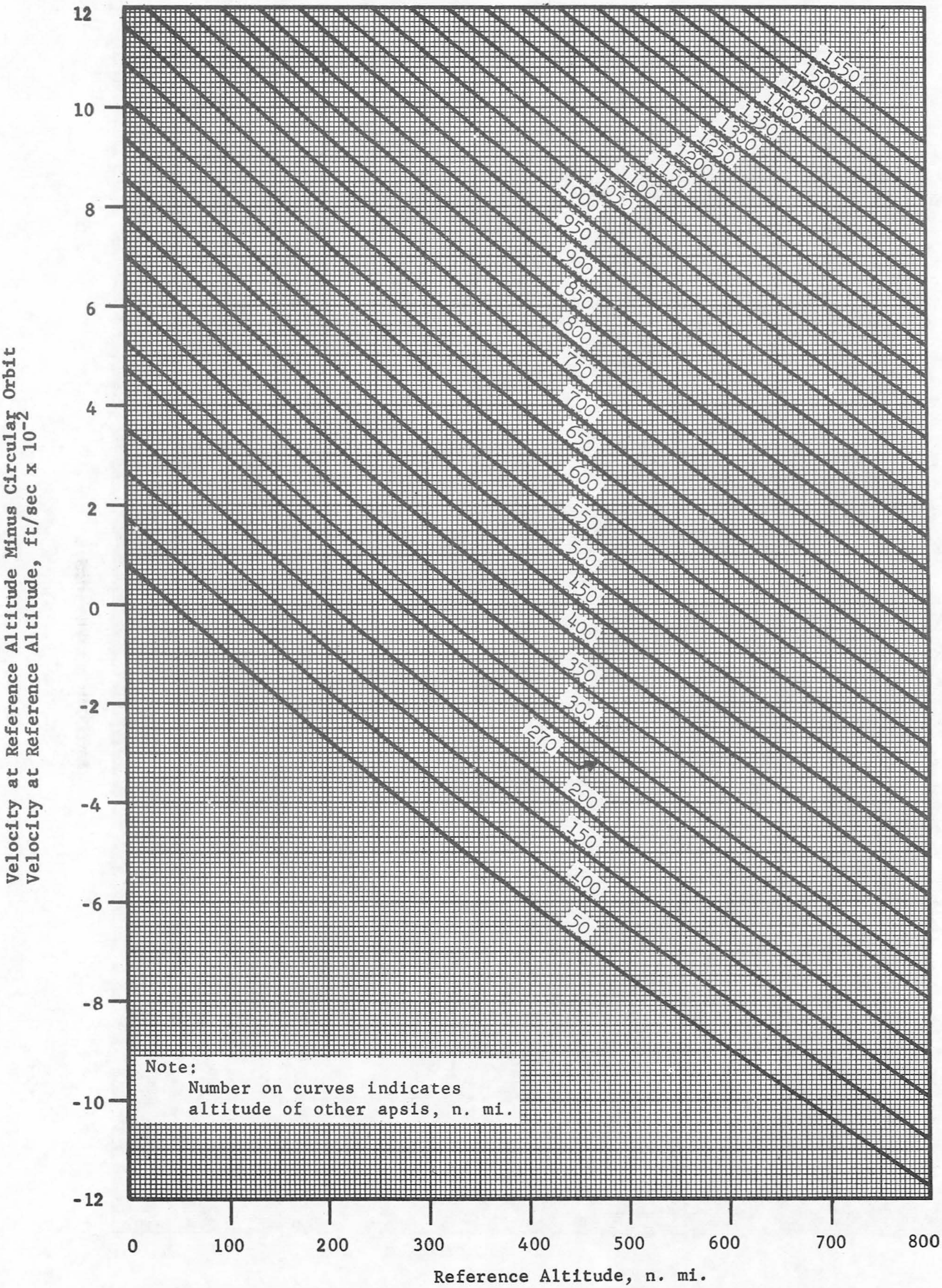


FIGURE II- 31c. HORIZONTAL VELOCITY IMPULSE AT REFERENCE ALTITUDE REQUIRED TO ESTABLISH OTHER APSIS

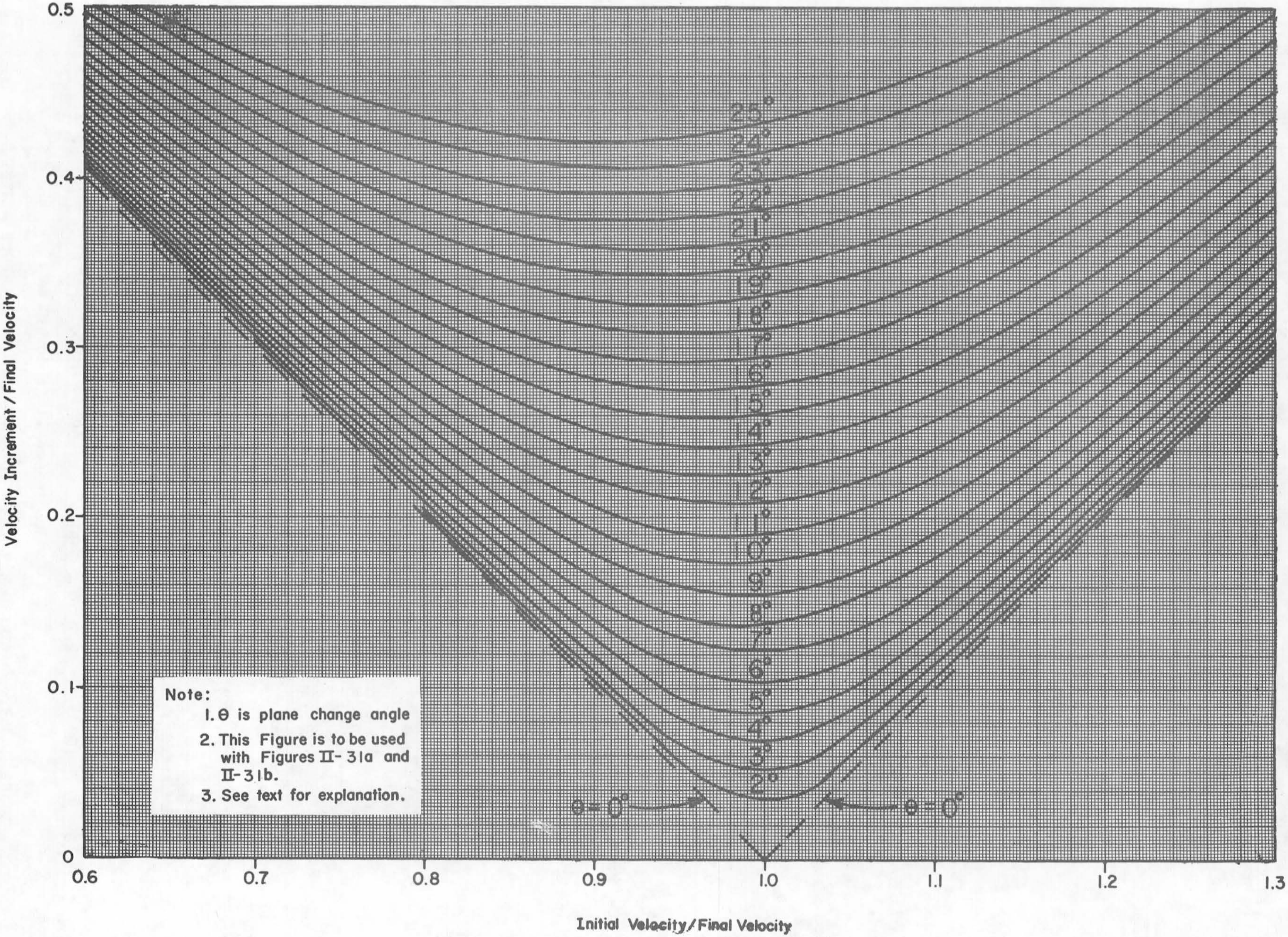


FIGURE II-32 a. NORMALIZED VELOCITY INCREMENT FOR NONCOPLANAR ORBIT TRANSFERS

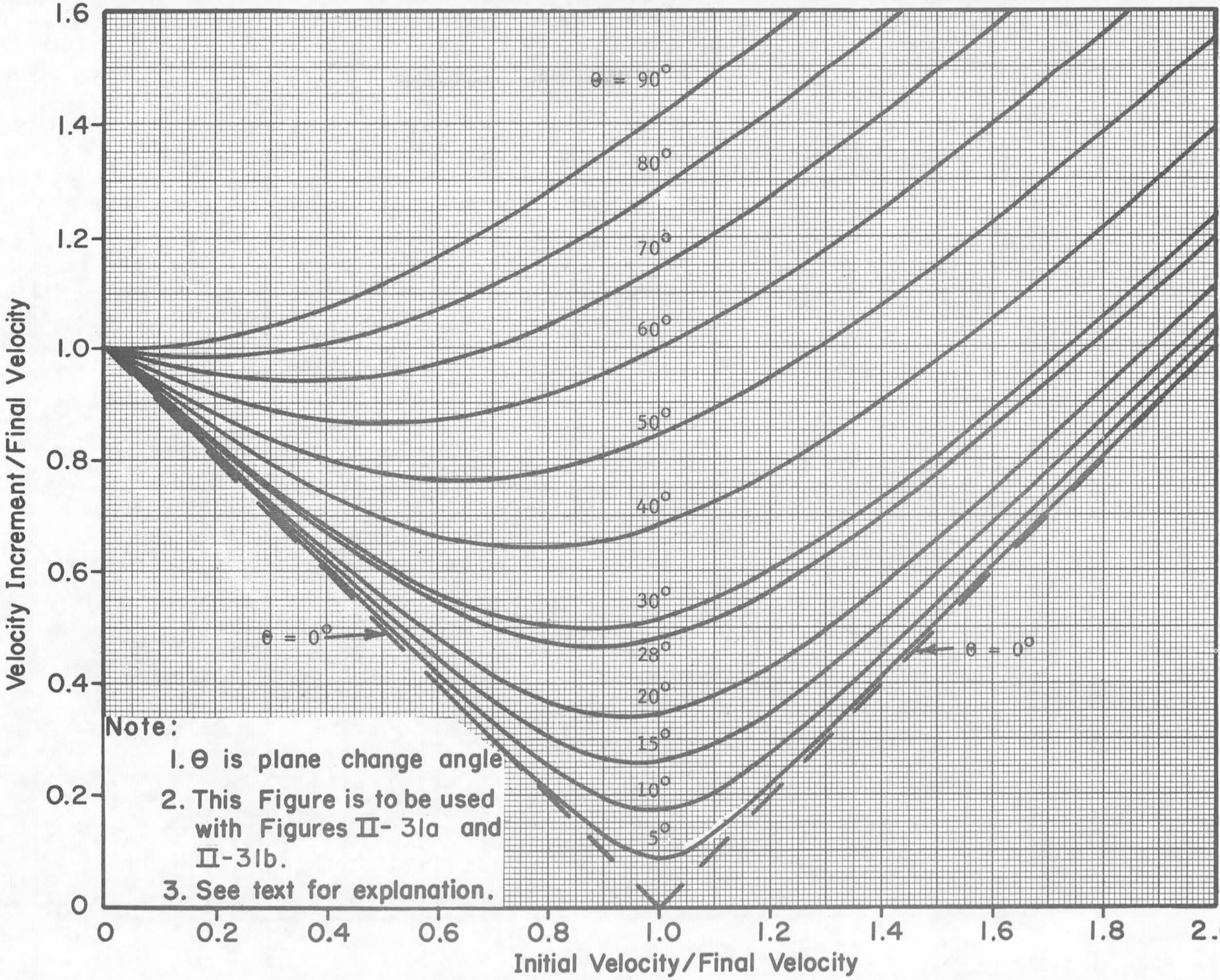


FIGURE II-32b. NORMALIZED VELOCITY INCREMENT FOR NONCOPLANAR ORBIT TRANSFERS
(EXTENDED SCALE)

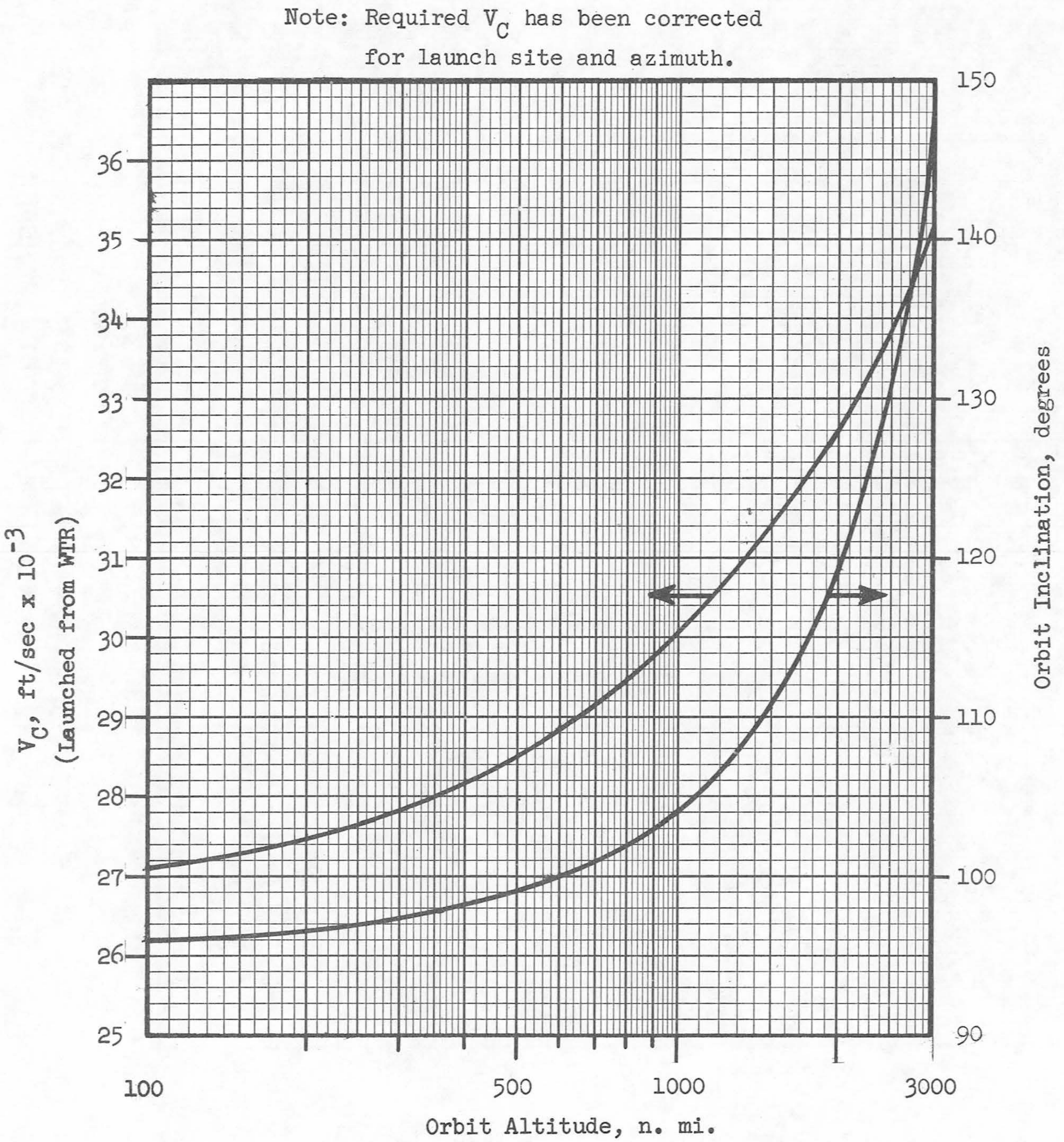


FIGURE II-33. INCLINATION AND V_C OF CIRCULAR SUN SYNCHRONOUS ORBITS

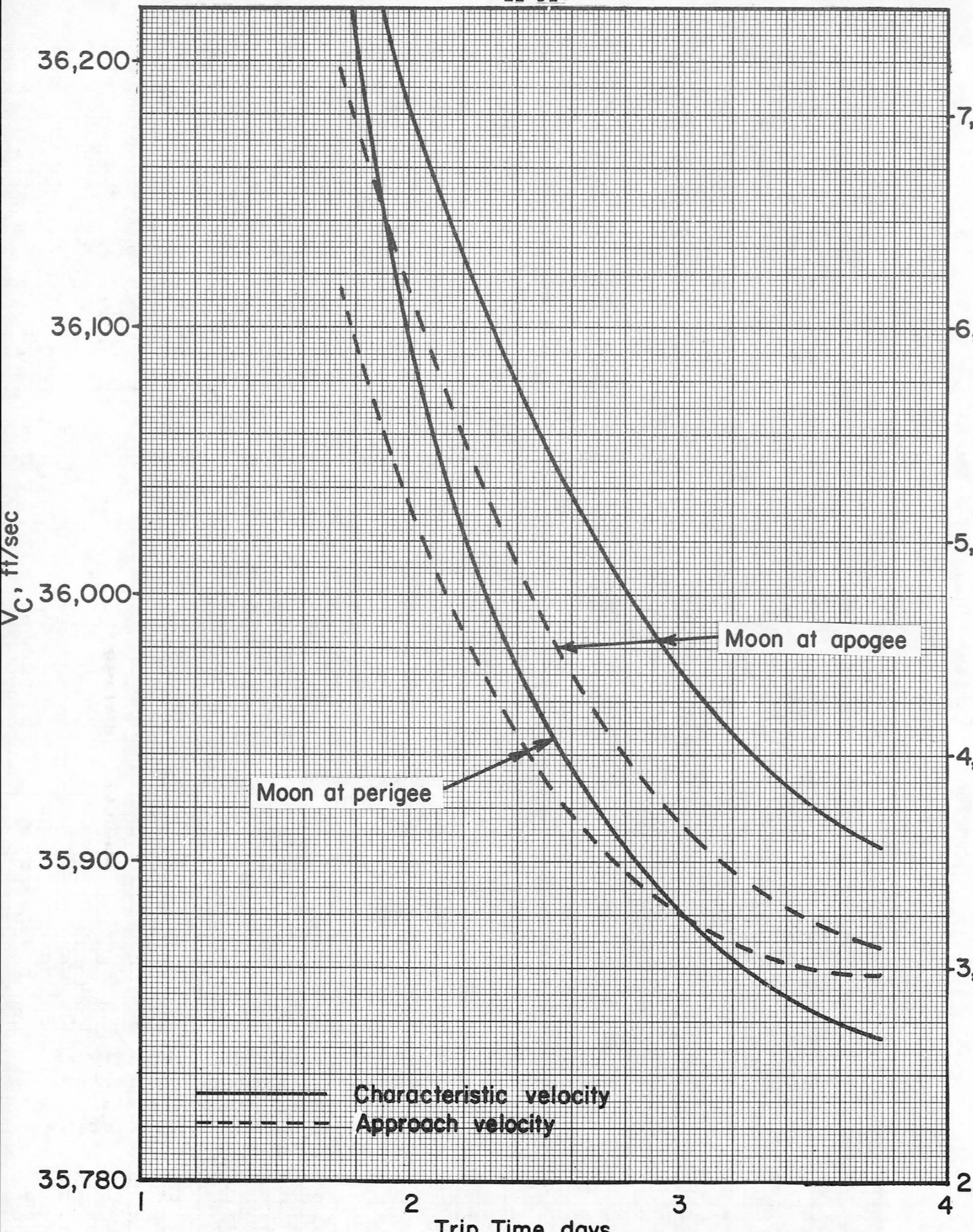


FIGURE II-34. LAUNCH CHARACTERISTIC VELOCITY REQUIREMENTS AND APPROACH VELOCITIES FOR LUNAR MISSIONS

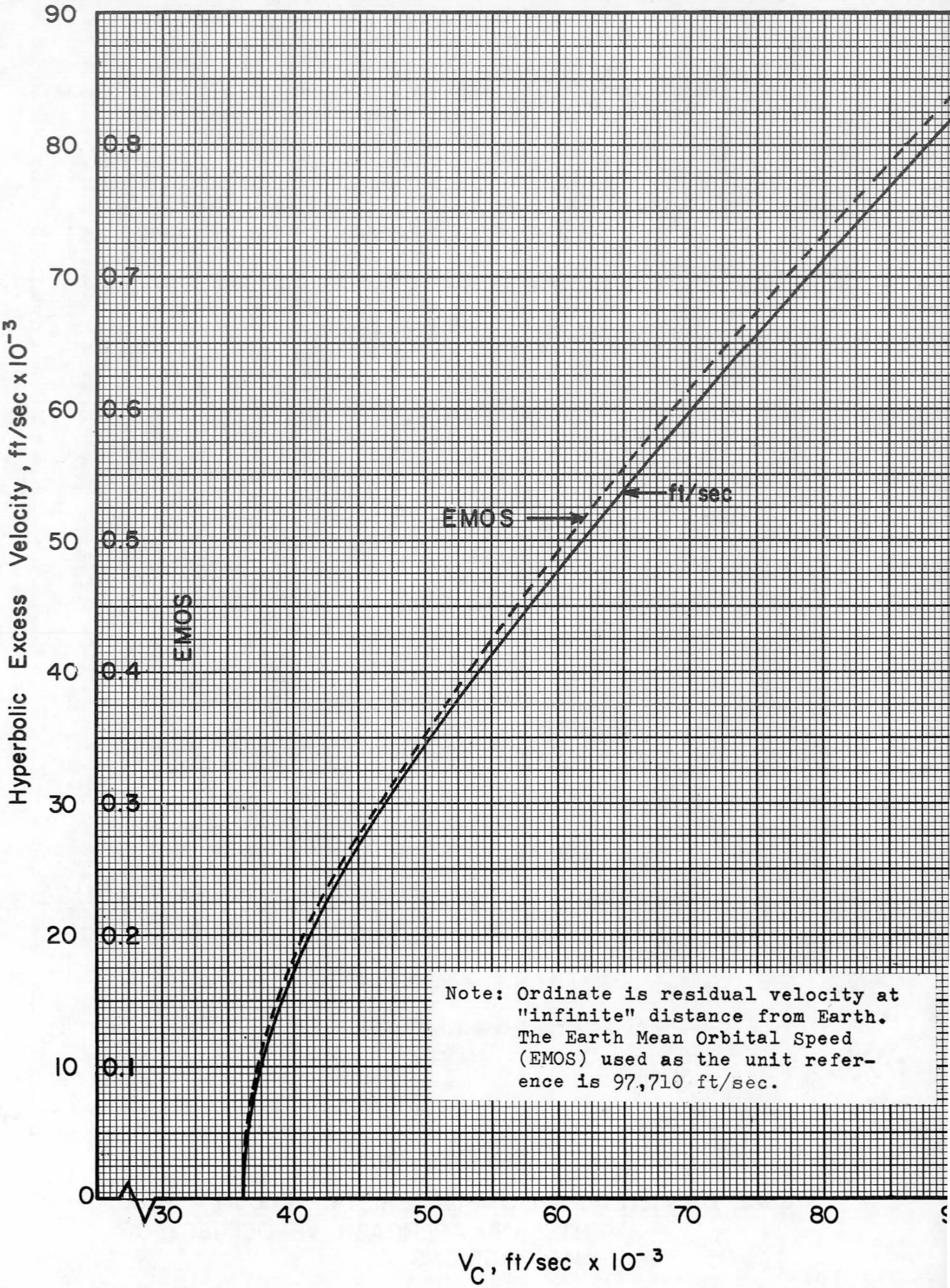


FIGURE II-35. CONVERSION CHART, V_C TO HYPERBOLIC EXCESS VELOCITY

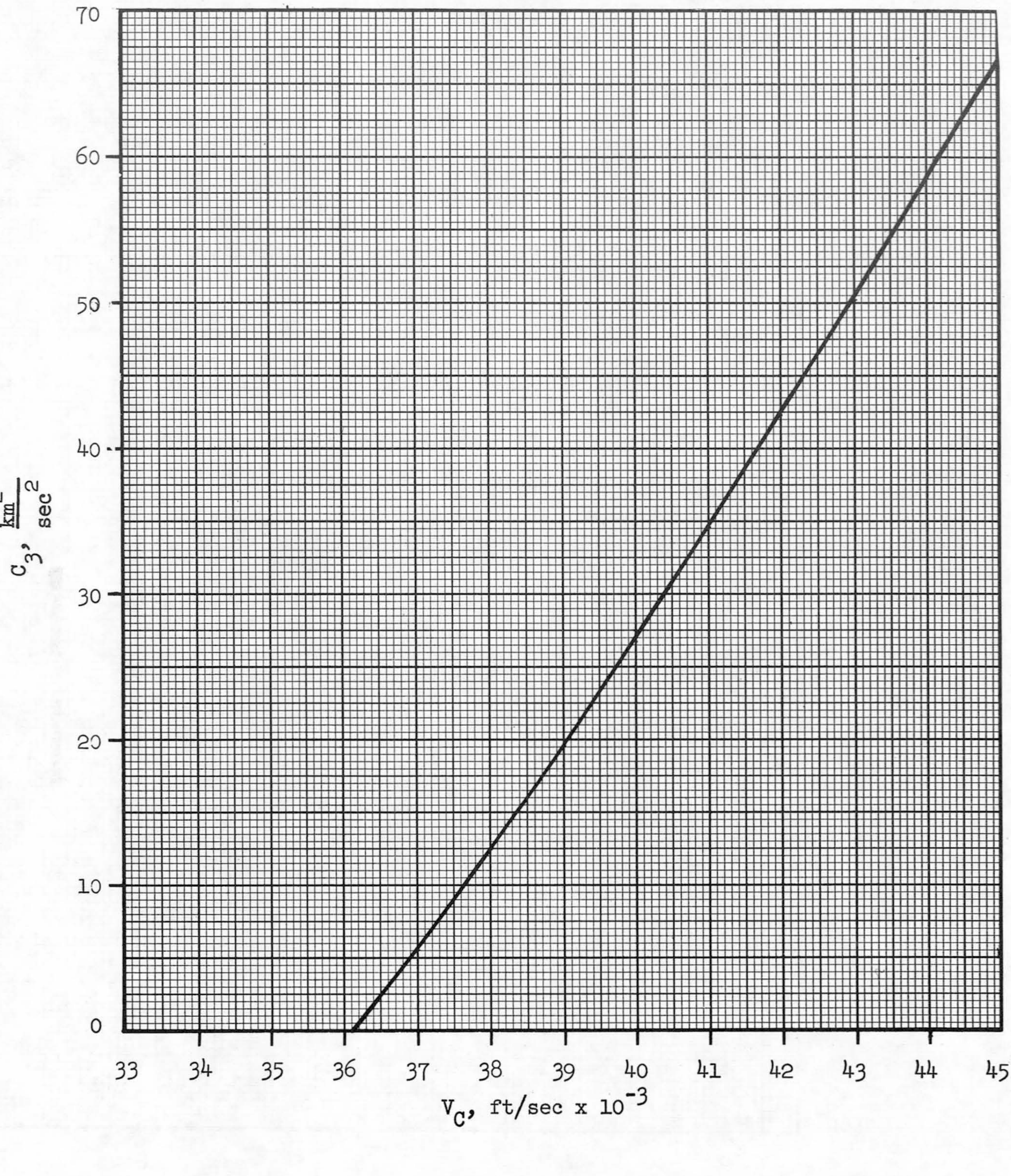


FIGURE II-36a. CONVERSION CHART, V_C TO C_3

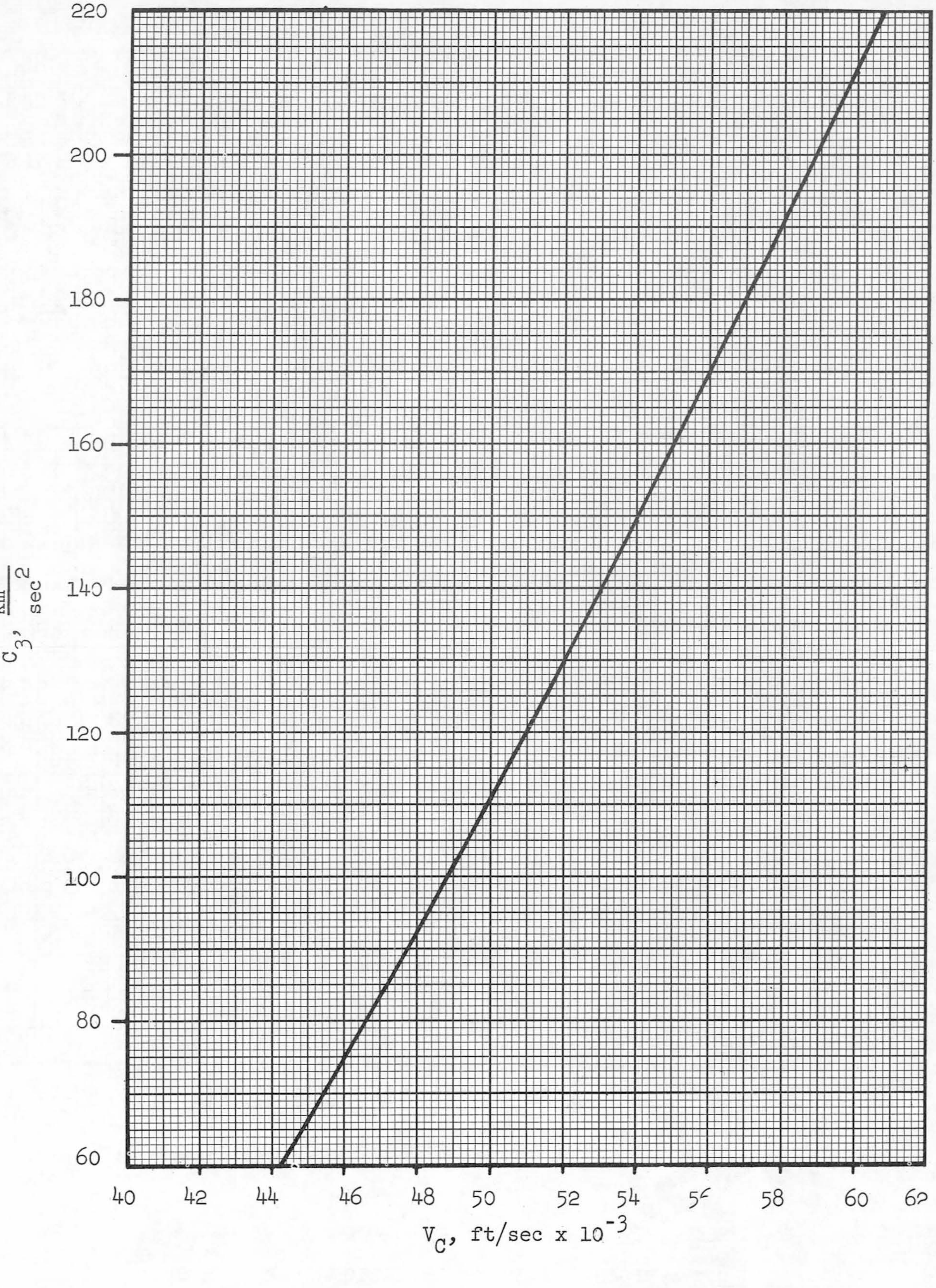


FIGURE II-36b. CONVERSION CHART, V_C TO C_3 (Cont'd)

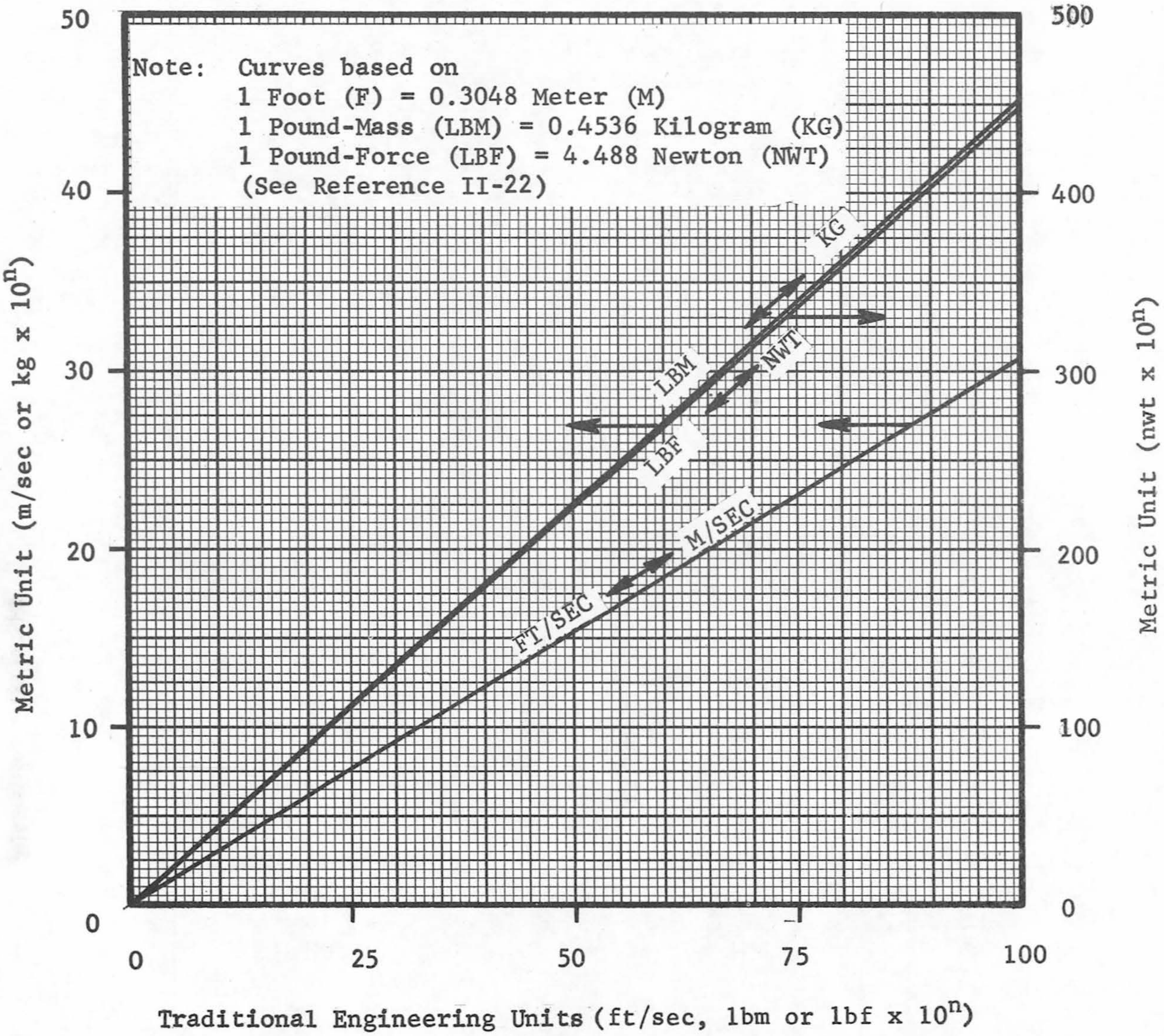


FIGURE II-37. CONVERSION CHART, TRADITIONAL ENGINEERING UNITS TO METRIC UNITS

TABLE II-1. PLANETARY CHARACTERISTICS[a]

	Mercury	Venus	Earth	Mars	Jupiter	Saturn	Uranus	Neptune	Pluto
Semi-major Axis, a. u.[b]	.387099	.723332	1.000000	1.523692	5.202803	9.538843	19.182110	30.057416	39.373641
Eccentricity	.2056203	.0068064	.0167385	.0933405	.0483865	.0548488	.0478624	.0085516	.2488033
Perihelion, a. u.	.307503	.718408	.983262	1.381469	4.951058	9.015648	18.264008	29.800377	29.577349
Aphelion, a. u.	.466694	.728255	1.016739	1.665914	5.454549	10.062038	20.100212	30.314455	49.169933
Inclination, degrees									
Orbit to Ecliptic	7.003638	3.394023	0.000000	1.850070	1.306528	2.490850	.772939	1.775675	17.169866
Equatorial to Orbit	-- (c)	-- (c)	23.449722	25.2000	3.11500	26.74500	97.5900	29.0000	-- (c)
Mean Orbital Velocity(EMOS)[d]	1.607271	1.175794	1.000000	.810124	.438411	.323782	.228324	.182399	.159367
Period (Earth years)	.240842	.615186	1.000000	1.880813	11.867413	29.460733	84.012627	164.788714	247.063373
Radius, n. mi.	1315	3268	3444	1832	38538	32614	12708	12054	3788
Mass (Earth=1)	.054972	.805097	1.000000	.106141	314.018416	93.992913	14.343655	17.076843	.181512
Gravitational Parameter[e], ft^3/sec^2	7.833653+14	1.147276+16	1.425015+16	1.512529+15	4.474810+18	1.339413+18	2.043992+17	2.433476+17	2.586575+15
Sphere of Influence, Radius, a.u.	.0007534	.0041196	.0062112	.0038586	.3222507	.3646768	.3457282	.5808836	.0928316
Period of Rotation, Earth days	58.67	242.6	1.0	1.02596	.41069	.42639	.45069	.5833	6.39
Escape Velocity at Surface, ft/sec	14004	33994	36905	16484	195500	116267	72762	81517	14991
No. of Satellites	0	0	1	2	12	9	5	3	0

(a) Data adapted from Reference II-22.
(b) 1 a.u. = 80,776,000 n. mi.
(c) Uncertain
(d) EMOS = 97,710 ft/sec.
(e) The +n following each number indicates power of 10.

SECTION III

LAUNCH SITE FACTORS

SECTION III

LAUNCH SITE FACTORS

The Eastern Test Range (ETR) is used for launches for which it is feasible to employ the rotation of Earth to increase the velocity of the vehicle - that is, for launches predominantly eastward. The Western Test Range (WTR) is used chiefly for southerly launches, often slightly retrograde, for near-polar orbits. Scout is the only launch vehicle considered in this document for which there are facilities at Wallops Island. Scout is also launched by Italy from the San Marco Platform near the equator on the east coast of Africa.

The vehicles for which launch facilities are available or planned at the Eastern and Western Test Ranges, and at San Marco and Wallops Island are as follows:

Eastern Test Range	Western Test Range	Wallops Island and San Marco
Delta (Various Configurations)	Scout	Scout
SLV3C/Centaur	Thor/Burner II*	
Titan IIIC	Delta (Various Configurations	
Titan IIID	SLV3A/Burner II	
Titan IIID/Centaur	TAT(3C)/Agena	
Saturn IB	Titan IIIB/Agena	
Saturn V	Titan IIID	

The figures in this section give an approximate velocity increment that must be either added to the mission velocity requirement or subtracted from the launch vehicle capability for any launch that is not in an eastward direction from the ETR. Figure's III-1a and III-1b show the velocity penalty and launch azimuth as functions of orbit inclination for the ETR. Figure III-1a shows the velocity penalty for the values of launch azimuths normally permitted by nominal range safety requirements. Figure III-1b shows the velocity penalty for the expanded range of launch azimuths possible where flights outside the normal azimuth limits are possible (e.g., as expected for the proposed space shuttle).

* Where convenient, the abbreviation BII will be used to designate Burner II.

Figure III-2, III-3 and III-4 present data similar to that in Figure III-1a for the WT Wallops Island, and San Marco, respectively. Azimuth angle is measured in a clockwise direction from geographical north in the horizontal plane at the launch point. Earth orbital inclination is defined as the angle between the angular momentum vector of the orbit and the North Pole. This is equivalent to the angle between the Earth equatoria plane and the plane of the orbit for prograde orbits and 180^{o} minus this angle for ret grade orbits. The corrections shown are approximated as the difference between the lo Earth surface velocity in the direction of launch and the Earth surface velocity in an eastward direction at the ETR. A more precise determination of these corrections would require consideration of other factors related to the launch.

Figures III-1a, III-2, III-3 and III-4 also display generalized limits impos on launch azimuth due to range safety considerations. There are specific range safety limits associated with each launch vehicle. These safety limits can be waived, but flights outside these limits require special clearances. Questions on the subject sho be referred to NASA OSSA Launch Vehicle and Propulsion Programs.

The launch azimuth corresponding to a given orbit inclination was computed b assuming that injection into a 100 n. mi. orbit takes place directly over the launch s The relationships between launch azimuth and orbit inclination, for the four launch si are shown for a broader range of inclinations in Figure III-5. For orbital inclinatio unattainable by direct injection, a plane change maneuver must be performed. For fina orbits other than circular, specific calculations must be performed considering apses and angles involved. Refer to Section II for the appropriate procedure.

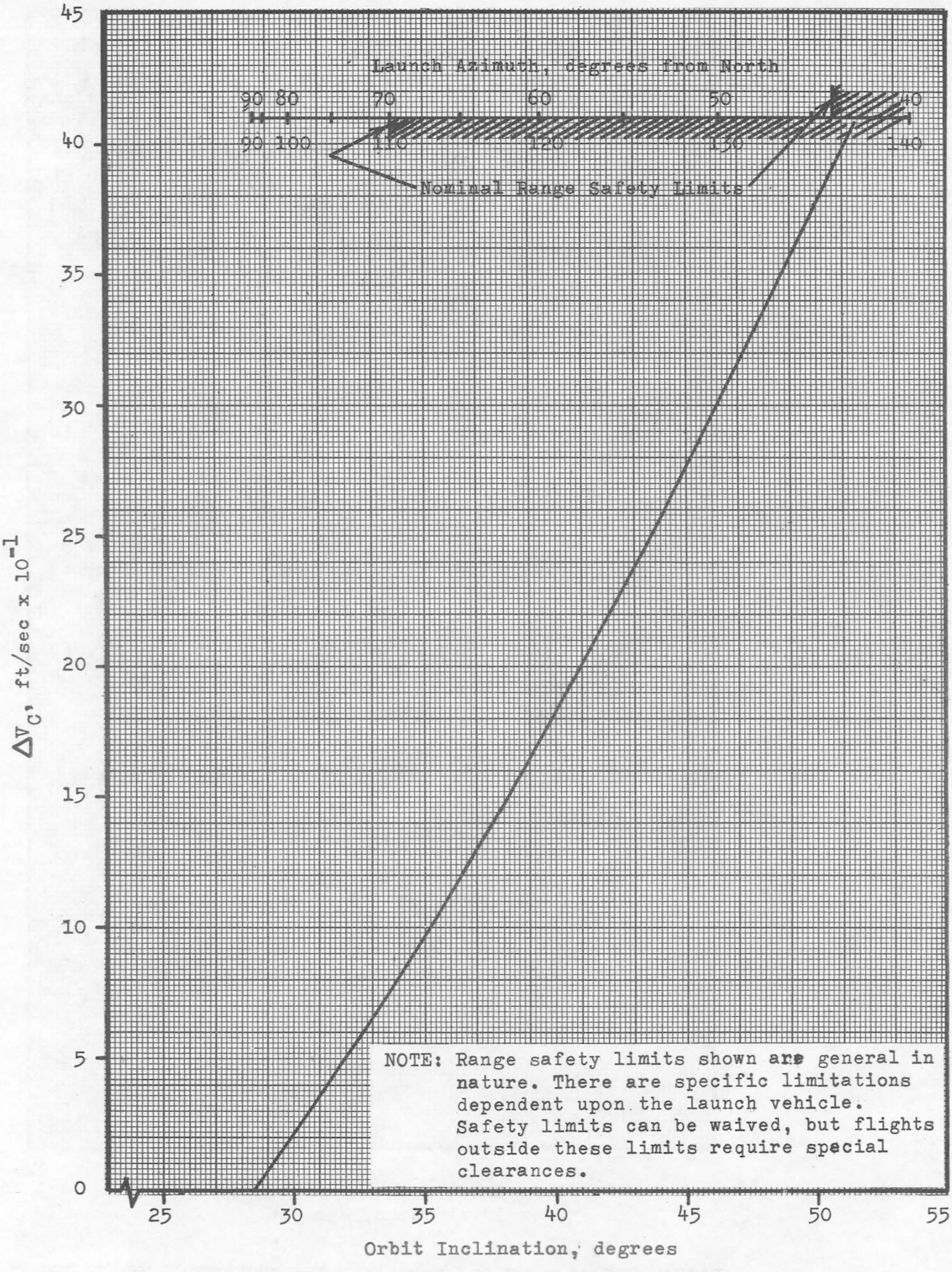


FIGURE III-1a. VELOCITY PENALTY FOR LAUNCHES FROM EASTERN TEST RANGE

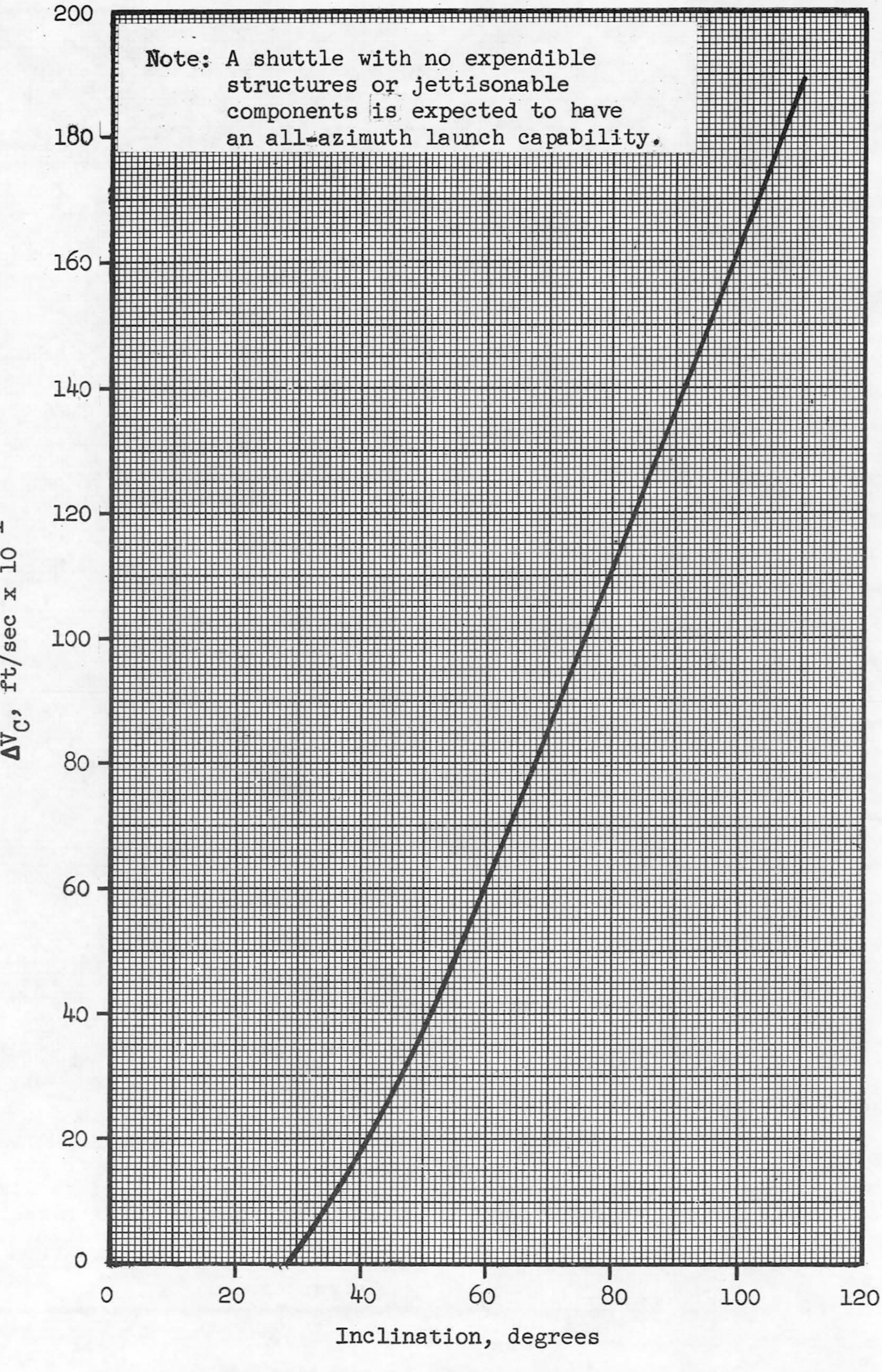


FIGURE III-1b. VELOCITY PENALTY FOR LAUNCHES FROM EASTERN TEST RANGE (HIGHER INCLINATIONS)

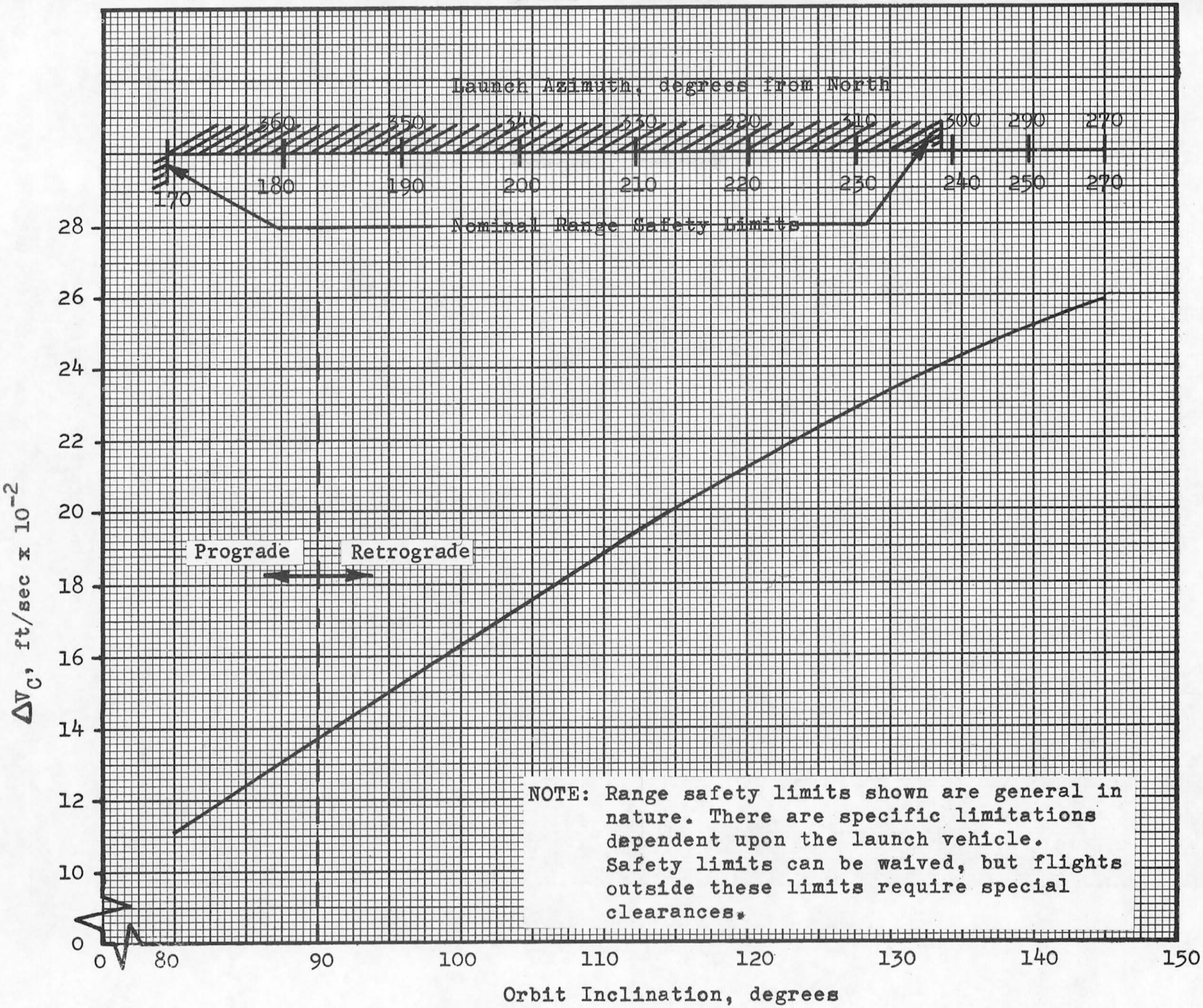


FIGURE III-2. VELOCITY PENALTY FOR LAUNCHES FROM WESTERN TEST RANGE

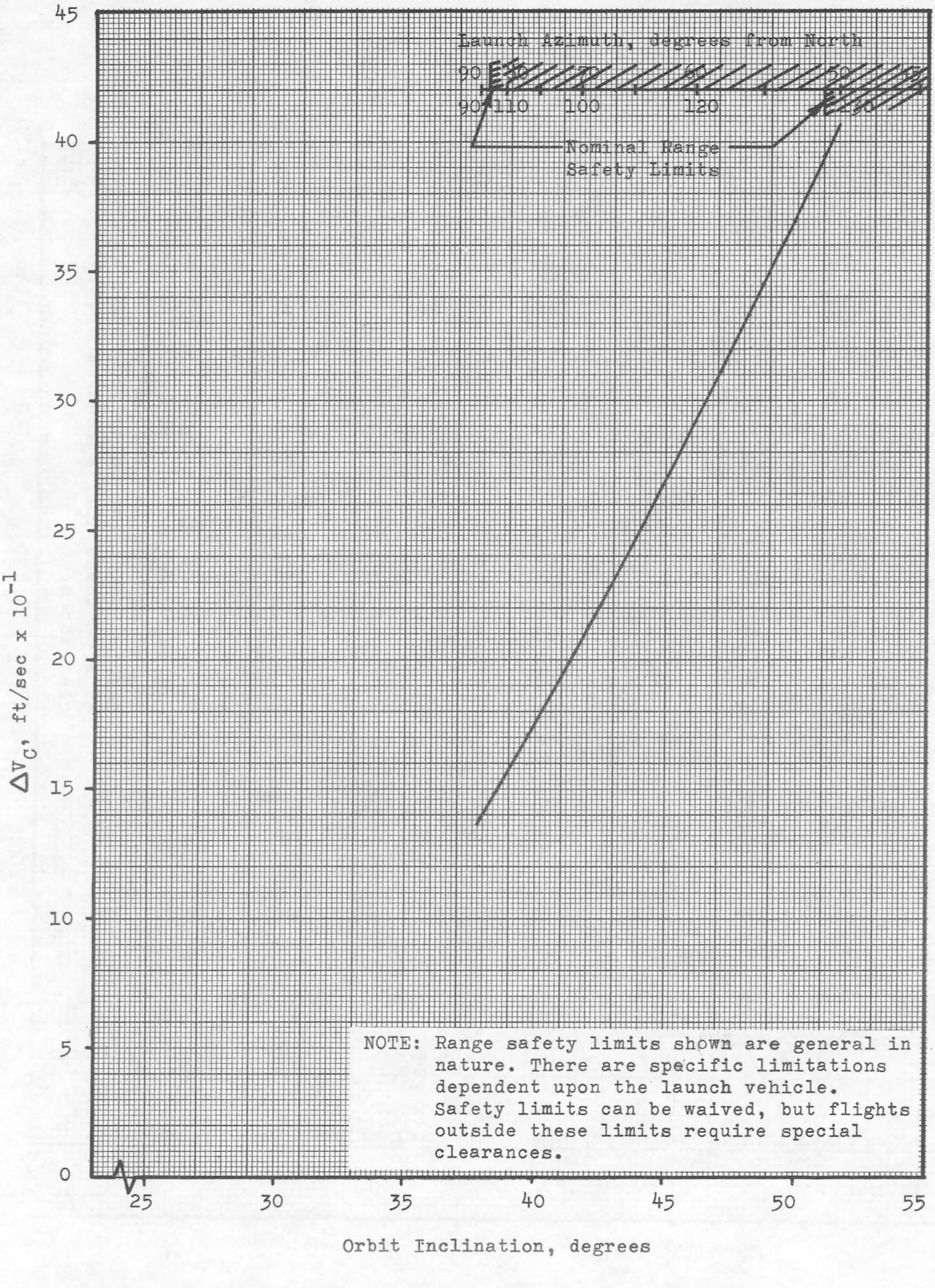


FIGURE III-3. VELOCITY PENALTY FOR LAUNCHES FROM WALLOPS ISLAND

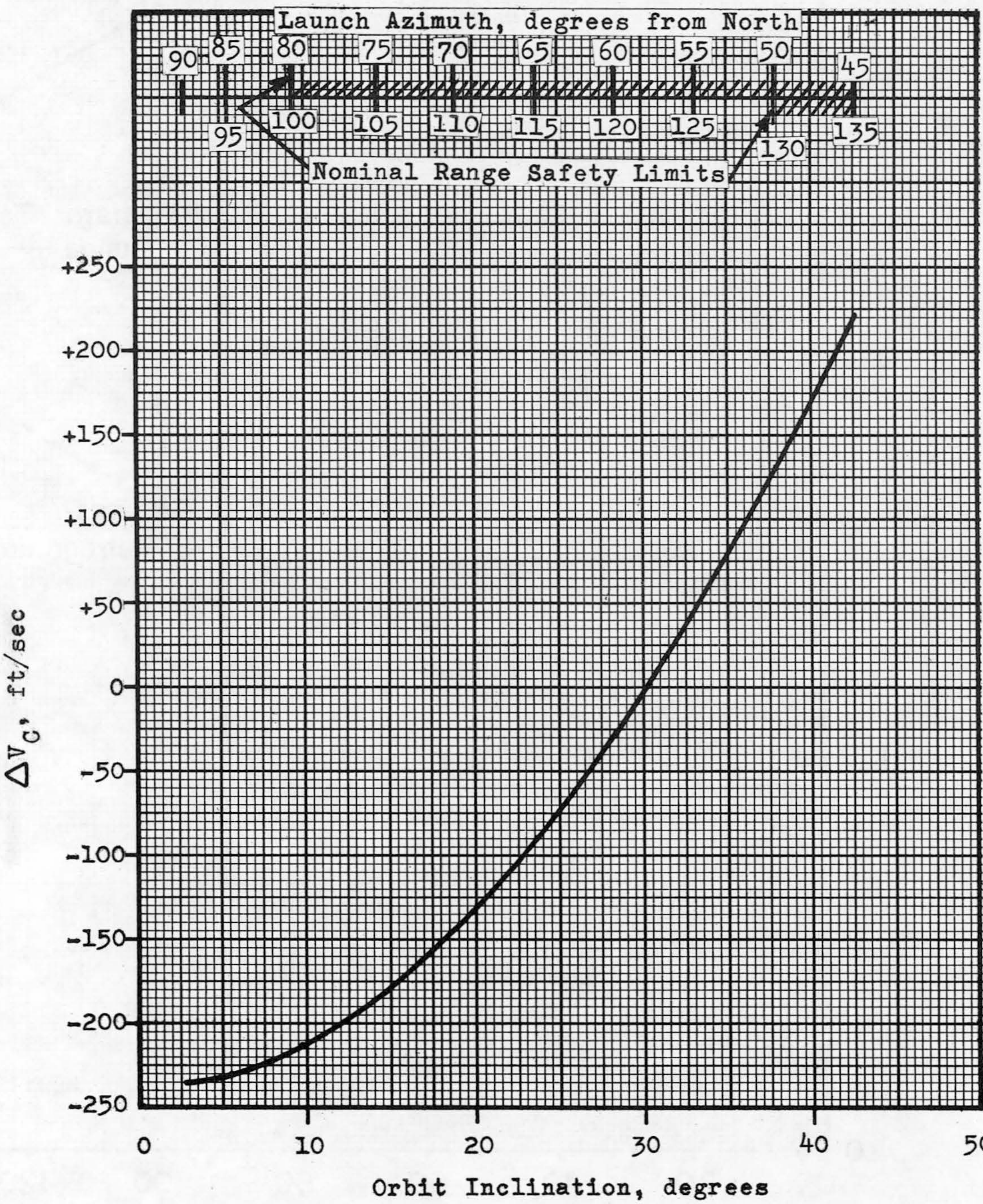


FIGURE III-4. VELOCITY CORRECTION FOR LAUNCHES FROM SAN MARCO

NOTE: Range safety limits shown are general in nature. Safety limits can be waived, but flights outside these limits require special clearances.

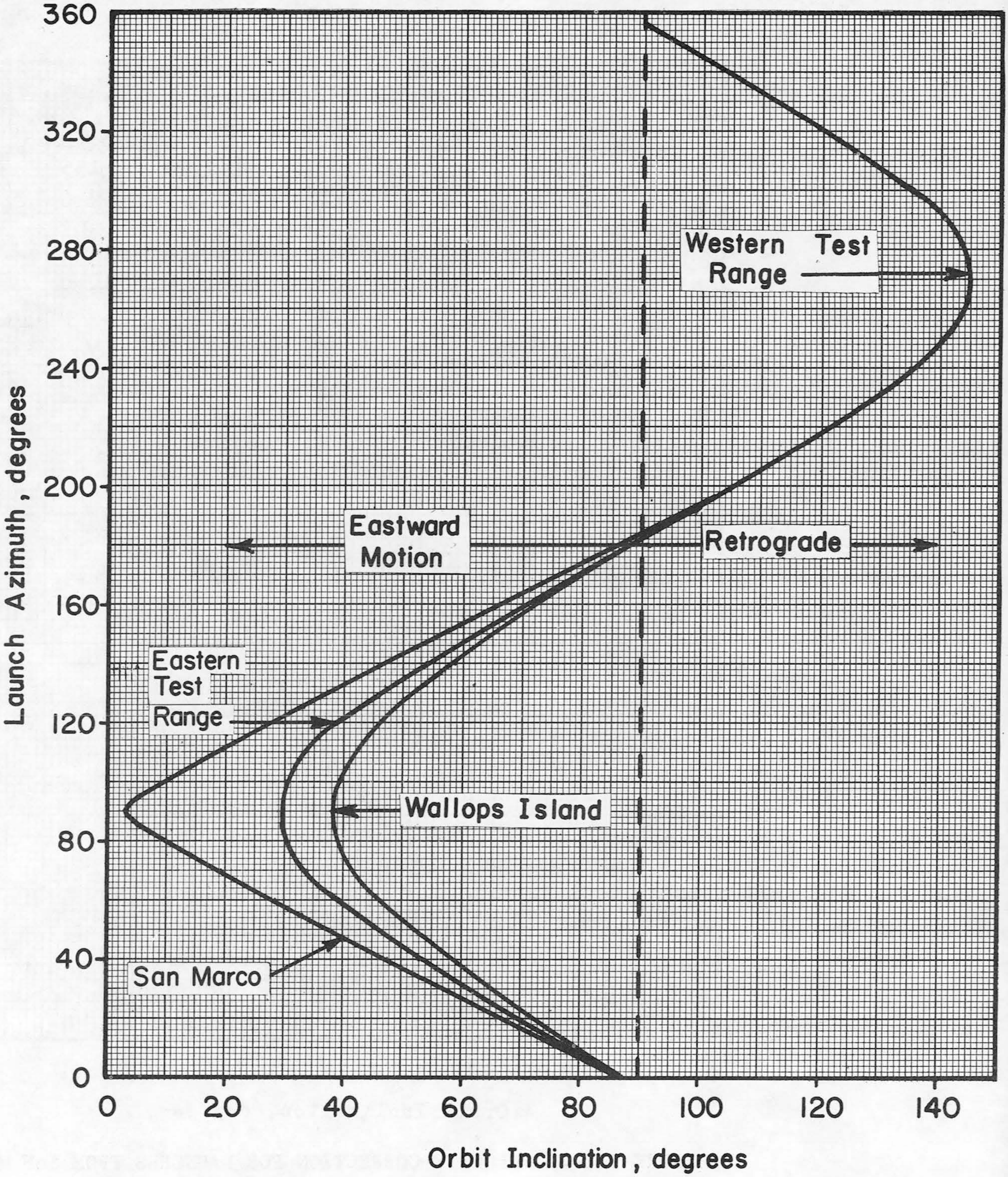


FIGURE III-5. LAUNCH AZIMUTH REQUIRED TO ACHIEVE VARIOUS ORBIT INCLINATIONS

Note: For 100 n. mi. parking orbits.

SECTION IV

LAUNCH VEHICLE PERFORMANCE DATA

SECTION IV

LAUNCH VEHICLE PERFORMANCE DATA

The following sections give performance data for existing, expected, and possible payload delivery systems. Section IV-A contains generalized performance data (payload weight versus characteristic velocity) for various expendable chemical launch vehicles. Section IV-B contains specific Earth orbital performance data for certain existing, planned and proposed expendable chemical launch vehicles. Section IV-C contains currently projected performance data for possible future space shuttle vehicles. Section IV-D presents performance data for selected velocity packages for use with small payloads. Section IV-E contains representative data on solar-electric spacecraft performance with a few launch vehicles. Section IV-F contains representative performance data for nuclear-thermal, and nuclear-electric propulsion systems. Section IV-G provides data for estimating retro-propulsion system weights, for sizing apogee kick motors and for other similar spacecraft propulsion systems.

For launch vehicles employing chemical and nuclear-thermal propulsion systems, payload is considered to include all elements normally associated with the spacecraft that must be accelerated to a required final velocity. The performance data are based on a representative adapter (except in Section IV-B where no allowance for an adapter is included) and shroud configurations and weights, as listed in Table IV-1. Performance reserves have been retained in Sections IV-A, IV-B, and IV-C, but not in IV-D through IV-G.

Vehicles are listed in Table IV-1 in order of first appearance in Section IV-A and IV-F. In Section IV-B, shrouds are identical to those given for the same vehicles in Section IV-A, but no payload adapter weight has been allowed in Section IV-B. Payload adapter weights for the velocity packages in Section IV-D are listed separately in Table IV-D-1.

For those vehicles involving solar-electric and nuclear-electric propulsion systems in Section IV-E and IV-F, adapter weights are considered to be included with the propulsion systems. Shrouds are the same as for the corresponding base vehicles in Section IV-A. Shroud and adapter data for vehicles using nuclear-thermal propulsion stages in Section IV-F are listed in Table IV-1.

Missions employing solar-electric or nuclear-electric propulsion systems are not always amenable to a clear distinction between delivery system and payload. Since a significant portion of the total mission energy is provided by the spacecraft propulsion system over an extended time period, the performance of low thrust spacecraft must be presented on a total vehicle and mission-by-mission basis. Data in Sections IV-E and IV-F are presented in such a way that payload is exclusive of the low thrust primary propulsion system; thus, the payload is compatible with the definition given for vehicles employing chemical or nuclear-thermal propulsion.

TABLE IV-1. PAYLOAD ADAPTER AND SHROUD DATA (VEHICLES LISTED IN ORDER OF APPEARANCE)

Launch Vehicle		Payload Adapter Weight* lb	Shroud Weight lb	Shroud Configuration
		Figure IV-A-1		
Scout		12	264	Scout
Scout(Algol III)		12	264	Scout
Scout(5-Stage)		3.5	280	Scout, Extended
Scout(Algol III)(5-Stage)		3.5	280	Scout, Extended
Thor/BII(1440)		20	273	Burner II
TAT(3C)/Delta(TSE)	(300)**	50	543	Delta
TAT(3C)/Delta(TSE)/TE364(1440)	(303)	20	543	Delta
TAT(6C)/Delta(TSE)	(600)	60	543	Delta
TAT(6C)/Delta(TSE)/TE364(2300)	(604)	20	543	Delta
TAT(9C)/Delta(TSE)	(900)	80	543	Delta
TAT(9C)/Delta(TSE)/TE364(2300)	(904)	20	543	Delta
		Figure IV-A-2		
TAT(3C)/Agena		100	587	Agena Long Shroud
SLV3A/BII(1440)		20	830	Atlas/BII
Titan IIIB(Str. CI)/Agena		100	2310	Martin(UPLF)
SLV3C/Centaur		116	2046	Surveyor
SLV3C/Centaur/TE364(2300)		20	2046	Surveyor
Titan IIIC		230	1967	Martin(UPLF)
Titan IIID		230	6000	Viking
Titan IIID/Centaur		116	6000	Viking
Titan IIID/Centaur/TE364(2300)		20	6000	Viking
Saturn IB		1600	5600	Voyager
Saturn V		1600	5600	Voyager
		Figure IV-A-3		
TAT(H1-3C)/Delta(TSE)		50	543	Delta
TAT(H1-9C)/Delta(TSE)		80	543	Delta
TAT(H1-3C)/Delta(TSE)/TE364(2300)		20	543	Delta
TAT(H1-9C)/Delta(TSE)/TE364(2300)		20	543	Delta
Titan IIIB(Str. CI)/Delta(TSE)		90	543	Delta
SLV3C/Delta(TSE)		80	535	Delta
		Figure IV-A-4		
Titan IIIB/Centaur		116	6000	Viking
Titan IIIB(Str. CI + 2A3)/Centaur		116	6000	Viking
Titan IIIB(Str. CI + 2A3)/Centaur/TE364(2300)		20	6000	Viking
Titan IIID(7)		230	6000	Viking
Titan IIIC(7)		230	1967	Martin(UPLF)
Titan IIID(7)/Centaur		116	6000	Viking

* Performance data in Section IV-B are given without adjustment for payload adapter.

** Vehicle contractor designations.

TABLE IV-1. PAYLOAD ADAPTER AND SHROUD DATA (VEHICLES LISTED IN ORDER OF APPEARANCE) (Continued)

Launch Vehicle	Payload Adapter Weight lb	Shroud Weight lb	Shroud Configuration
	Figure IV-A-5		
260(3.7)/SIVB(J2S)	1600	5600	Voyager
4 x 1563/SIVB(J2S)	1600	8200	Large Voyager
SIC(4F1)/SIVB	1600	5600	Voyager
SIC(4F1)/SIVB(J2S)	1600	5600	Voyager
SIC/SII	7730	5600	Voyager
Saturn IB/Centaur	300	5600	Voyager
260(3.7)/SIVB(J2S)/Centaur	300	5600	Voyager
4 x 1563/SIVB(J2S)/Centaur	300	8200	Large Voyager
SIC(4F1)/SIVB(J2S)/Centaur	300	5600	Voyager
Saturn V/Centaur	300	5600	Voyager
	Figure IV-A-6		
TAT(3C)/BII(2300)	20	273	Burner II
TAT(9C)/BII(2300)	20	273	Burner II
SLV3A/BII(2300)	20	830	Atlas/BII
SLV3A/BE BII(2-burn)	20	830	Atlas/BII
TAT(9C)/Delta(TSE)/BII(2300)	20	543	Delta
TAT(H1-9C)/Delta(TSE)/BE BII(2-burn)	20	543	Delta
SLV3C/Centaur/BII(2300)	20	2046	Surveyor
SLV3C/Centaur/Be BII(1-burn)	20	2046	Surveyor
Titan IIIB(Str. CI + 2A3)/Centaur/BE BII(1-burn)	20	6000	Viking
	Figure IV-A-7		
Titan IIIC/BII(2300)	20	1967	Martin(UPLF)
Titan IIID/Centaur/BII(2300)	20	6000	Viking
Titan IIID/Centaur(Str.)/BII(2300)	20	6000	Viking
Titan IIIC(7)/BII(2300)	20	1967	Martin(UPLF)
Titan IIID(7)/Centaur/BII(2300)	20	6000	Viking
Titan IIID(7)/Centaur(Str.)/BII(2300)	20	6000	Viking
Titan IIIC/BE BII(1-burn)	20	1967	Martin(UPLF)
Titan IIIC(7)/BE BII(1-burn)	20	1967	Martin(UPLF)
	Figure IV-A-8		
TAT(9C)/VUS	60	1000	New
TAT(9C)/VUS/TE364(2300)	20	1000	New
SLV3C/VUS	60	1000	New
Titan IIIB(Str. CI)/VUS	60	1000	New
Titan IIIB(Str. CI + 2A3)/Centaur/VUS	60	6000	Viking
SLV3C/Centaur/VUS	60	2046	Surveyor
Titan IIID/Centaur/VUS	60	6000	Viking
Titan IIID(7)/Centaur/VUS	60	6000	Viking
260(3.7)/SIVB/Centaur/VUS	60	5600	Voyager
4 x 1563/SIVB(J2S)/Centaur/VUS	60	8200	Large Voyager
SIC(4F1)/SIVB(J2S)/Centaur/VUS	60	5600	Voyager

TABLE IV-1. PAYLOAD ADAPTER AND SHROUD DATA (VEHICLES LISTED IN ORDER OF APPEARANCE) (Continued)

Launch Vehicle	Payload Adapter Weight lb	Shroud Weight lb	Shroud Configuration
Figure IV-A-9			
Titan IIIB(Str. CI + 2A3)/Centaur(F)	60	6000	Viking
Titan IIIB(Str. CI + 2A3)/Centaur(F)/VUS	60	6000	Viking
Titan IIID/Centaur(F)	60	6000	Viking
Titan IIID/Centaur(F)/VUS	60	6000	Viking
Titan IIID(7)/Centaur(F)/VUS	60	6000	Viking
2 x 1207/CI/Centaur(F-Str.)/Centaur(F)/VUS	60	6000	Viking
260(3.7)/SIVB/Centaur(F)/VUS	60	5600	Voyager
4 x 1563/SIVB(J2S)/Centaur(F)/VUS	60	8200	Large Voyager
SIC(4F1)/SIVB(J2S)/Centaur(F)/VUS	60	5600	Voyager
Figure IV-A-10			
2 x 1207/LDCI/CII	1000	6000	Viking
2 x 1207/LDCI/CRY CII	1000	6950	New
4 x 1207/LDCI/CII	1000	6000	Viking
4 x 1207/LDCI/CRY CII	1000	6950	New
2 x 1207/LDCI/CII/Centaur	300	6000	Viking
2 x 1207/LDCI/CRY CII/Centaur	300	6950	New
4 x 1207/LDCI/CII/Centaur	300	6000	Viking
4 x 1207/LDCI/CRY CII/Centaur	300	6950	New
Figure IV-F-1			
260(3.7)/SIVB(N)	--	5600	Voyager
260(3.7)/SIVB(N)/Centaur	116	5600	Voyager
SIC(4-F1)/SIVB(N)	--	5600	Voyager
SIC(4-F1)SIVB(N)/Centaur	116	5600	Voyager
4 x 1563/SIVB(N)	--	8200	Large Voyager
4 x 1563/SIVB(N)/Centaur	116	8200	Large Voyager
Figure IV-F-2			
Saturn V-SIVB(N)	--	5600	Voyager
Saturn V-SIVB(N)/Centaur	116	5600	Voyager
2 x 1207/LDCI/SIVB(N)	--	5000	New
2 x 1207/LDCI/SIVB(N)/Centaur	116	5000	New
4 x 1207/LDCI/SIVB(N)	--	5000	New
4 x 1207/LDCI/SIVB(N)/Centaur	116	5000	New

TABLE IV-1. PAYLOAD ADAPTER AND SHROUD DATA (VEHICLES LISTED IN ORDER OF APPEARANCE) (Continued)

Launch Vehicle	Payload Adapter Weight lb	Shroud Weight lb	Shroud Configuration
Figure IV-F-3			
Saturn V/Centaur(N)	--	5600	Voyager
SIC(4-F1)/SIVB(J2S)/Centaur(N)	--	5600	Voyager
260(3.7)/SIVB(J2S)/Centaur(N)	--	5600	Voyager
4 x 1563/SIVB(J2S)/Centaur(N)	--	8200	Large Voyager
2 x 1207/LDCI/LDCII/Centaur(N)	--	5000	New
4 x 1207/LDCI/LDCII/Centaur(N)	--	5000	New
Figure IV-F-4			
Saturn V-SIVB(N)/Centaur(N)	600	5600	Voyager
SIC(4-F1)/SIVB(N)/Centaur(N)	--	5600	Voyager
260(3.7)/SIVB(N)/Centaur(N)	--	5600	Voyager
4 x 1563/SIVB(N)/Centaur(N)	--	8200	Large Voyager
2 x 1207/LDCI/SIVB(N)/Centaur(N)	--	5000	New
4 x 1207/LDCI/SIVB(N)/Centaur(N)	--	5000	New

SECTION IV-A

GENERALIZED PERFORMANCE DATA FOR LAUNCH VEHICLES USING CHEMICAL PROPULSION

SECTION IV-A

GENERALIZED PERFORMANCE DATA FOR LAUNCH VEHICLES USING CHEMICAL PROPULSION

The figures in this section present performance capability data for expendable launch vehicles using chemical propulsion only. Capabilities are given in terms of plots of payload weight versus characteristic velocity (V_C). The latter quantity is defined in Section VI.

Performance reserves have been included in all data presented in this section. For vehicles having only liquid propellant upper stages this reserve was established as 1.5% of the characteristic velocity. For vehicles with solid propellant upper stages, a reserve of 0.5% of the velocity contribution of the solid propellant upper stage(s) and 1.5% of the velocity contribution of the rest of the vehicle was assumed. Scout vehicles were treated separately. For these a reserve of 0.5% of the total vehicle characteristic velocity was assumed.

Performance data for vehicles available in the 1971-1975 time period are presented in Figures IV-A-1 and IV-A-2. Figures IV-A-3, and IV-A-4 present the performance of improved launch vehicles that might possibly be available in the 1971 to 1975 time period. Figure IV-A-5 illustrates the performance of some possible near-term Saturn class launch vehicles. Performance for selected launch vehicles utilizing the Burner II(2300) and Beryllium Burner II velocity packages are shown in Figure IV-A-6. In Figure IV-A-7 performance data for various Titan vehicle options for interplanetary missions are shown. Figures IV-A-8 and IV-A-9 illustrate the performance achievable using hydrogen-fluorine (H_2/F_2) in upper stages. In Figure IV-A-8 the performance of selected launch vehicles using a possible H_2/F_2 Versatile Upper Stage (VUS) is presented. In Figure IV-A-9 the performance of selected launch vehicles using a possible future H_2/F_2 Centaur in combination with a possible H_2/F_2 VUS is shown. Finally, the performance of selected proposed large diameter core Titan vehicles is presented in Figure IV-A-10. The vehicles shown in Figures IV-A-3 through IV-A-10, or equivalent vehicles, could become available when there are sufficient mission requirements to justify the expenditure of development funds.

This subsection contains data for only a few of the possible expendable chemical propulsion launch vehicles that could be postulated. Those included were selected as reasonable alternatives on the basis of the requirements for improved performance as indicated by analysis of recently proposed missions, analysis of expected costs, and anticipated difficulty of development. The selections should be regarded as constituting the best current estimate of likely candidates from which a family of expendable launch vehicles could be chosen for OSSA applications over the next 20 years.

The data on these expendable launch vehicles are provided by the Director of Launch Vehicle and Propulsion Programs for use by OSSA mission planners. In all cases, the data are the best estimates available for the mission applications of interest in OSSA planning.

The data may be used in the following manner. For solar system missions, the curves in Section II can be used to convert the desired destination and flight time into a required characteristic velocity (V_C) for the mission. For orbital missions, the curves of Section II can be used in the same manner to convert the desired apogee and perigee into a required V_C. For all missions that are to be launched due east from the ETR, these values of V_C can be used directly with the curves of this section to obtain the payload deliverable by the various launch vehicles. If the mission is to be launched at some other launch azimuth or from one of the other launch sites, the V_C must be modified by the corrections given in Section III. This modified V_C can then be used with the curves of this section to obtain payload capabilities for various expendable launch vehicles. It should be remembered that the weight of any velocity packages, kick motors, or other systems necessary to accomplish the mission must be added to the basic spacecraft weight to determine the total payload which must be delivered by the launch vehicle.

The mission planner is warned against the indiscriminate use of the generalized performance curves in this subsection for estimating launch vehicle capabilities for Earth orbital missions. Specifically, the Earth orbital performance of vehicles which use solid propellant or other nonrestartable final stages (such as Burner II or Core II on Titan IIID) cannot be obtained by the above procedure. For Scout, Delta (various configurations), SLV3C/Centaur(single burn), TIIIC(single burn Transtage and TIIID configurations, the curves of Section IV-B must be used in estimating capability for Earth orbital missions.

Launch Vehicle Performance Available in 1971-1975 (Figures IV-A-1, Group 1, and IV-A-2, Group 2)

Group 1	Group 2
Scout	TAT(3C)/Agena
Scout(Algol III)	SLV3A/BII(1440)
Scout(5-Stage)	Titan IIIB(Str. CI)/Agena
Scout(Algol III)(5-Stage)	SLV3C/Centaur
Thor/BII(1440)	SLV3C/Centaur/TE364(2300)
TAT(3C)/Delta(TSE) (300)	Titan IIIC
TAT(3C)/Delta(TSE)/TE364(1440) (303)	Titan IIID*
TAT(6C)/Delta(TSE)(600)	Titan IIID/Centaur
TAT(6C)/Delta(TSE)/TE364(2300)(604)	Titan IIID/Centaur/TE364(2300)
TAT(9C)/Delta(TSE) (900)	Saturn IB
TAT(9C)/Delta(TSE)/TE364(2300) (904)	Saturn V

Scout (Algol III) designates an uprated version of the standard 4-stage Scout. It has a 44-inch-diameter Algol III solid propellant first stage. The 5-stage Scout vehicles use a BE-3 solid propellant motor as the fifth stage. The SLV3A and SLV3C are Atlas vehicles originally designed for use with Agena and Centaur upper stages, respectively. BII(1440) is the Burner II upper stage with a 1,440 lb propellant loading. It has been integrated with the Thor and SLV3A boosters. TAT(nC) refers to the Thrust-Augmented Thor boosters that consist of the long tank Thor(DSV2L) booster with n Castor II solid rocket motor strapons. The Delta (TSE) is an uprated version of the Delta stage using the Titan Transtage engine with a 40:1 expansion ratio nozzle. TE364(1440) and TE364(2300) are spin stabilized solid rocket motors.

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* Note that Core II does not have a restart capability.

The TAT/Delta vehicles are frequently identified by a three-digit numerical de ignation. In the above list and in Figure IV-A-1 these designations are listed in paren theses beside the vehicle names. The first digit corresponds to the number of Castor II solid rocket motors that are strapped to the Thor booster. The second digit (a zero for the vehicles shown here) identifies the second stage as being the Delta(TSE). The third digit indicates whether and/or which third stage is involved. Thus, a zero (0) indicate a two-stage vehicle, a three (3) indicates that the vehicle has a TE364(1440) third stag and a four (4) indicates that the third stage is a TE364(2300).

Titan IIIB is the basic 2-stage Titan core. (Str. CI) indicates a stretched version of Core I. The Titan IIIC employs the basic core with two 5-segment, 120-inch-diameter solid rocket motor strapons as a "zero" stage and the Titan Transtage as the third stage. The Titan IIID is a Titan IIIC without Transtage. The Saturn IB consists of the SIB booster with an SIVB upper stage, and is sometimes designated as the uprated Saturn I.

The Scout (4- and 5-stage), SLV3C/Centaur/TE364(2300), TAT(6C)/Delta(TSE)/TE364(2300), TAT(3C)/Delta(TSE)/TE364(1440), and Titan IIID/Centaur/TE364(2300) employ spin-stabilized final stages. Hence, payloads for these vehicles must be capable of withstanding the spin and must either operate while spin stabilized or have appropriate de-spin devices.

1971-1975 Improved Launch Vehicle
Performance Possibilities
(Figures IV-A-3, Group 1, and IV-A-4, Group 2)

Group 1	Group 2
TAT(H1-3C)/Delta(TSE)	Titan IIIB/Centaur
TAT(H1-9C)/Delta(TSE)	Titan IIIB(Str. CI + 2A3)/Centaur
TAT(H1-3C)/Delta(TSE)/TE364(2300)	Titan IIIB(Str. CI + 2A3)/Centaur/TE364(230
TAT(H1-9C)/Delta(TSE)/TE364(2300)	Titan IIID(7)
Titan IIIB(Str. CI)/Delta(TSE)	Titan IIIC(7)
SLV3C/Delta(TSE)	Titan IIID(7)/Centaur

In the near future a version of the TAT booster with the more powerful H1 engine, used on the Saturn IB first stage, replacing the current MB-3 Thor engine may be developed. These TAT configurations are designated above as TAT(H1-nC) with n depending on the number of Castors used. The proposed Titan IIIB(Str. CI + 2A3) consists of a stretched Core I with two Algol III solid rocket motor strapons (used for thrust augmentation) and a standard Core II. The Titan IIIC(7) and Titan IIID(7) would use the stretched Core I with 7-segment, 120-inch-diameter solid rocket motor strapons used as a "zero" stage. The Titan IIIC(7) would also have a stretched version of the Transtage, sized to provide optimal tankage for synchronous equatorial missions.

<u>Possible Near Term Saturn Class</u>
<u>Launch Vehicles (Figure IV-A-5)</u>

260(3.7)/SIVB(J2S)	Saturn IB/Centaur
4 x 1563/SIVB(J2S)	260(3.7)/SIVB(J2S)/Centaur
SIC(4F1)/SIVB	4 x 1563/SIVB(J2S)/Centaur
SIC(4F1)/SIVB(J2S)	SIC(4F1)/SIVB(J2S)/Centaur
SIC/SII	Saturn V/Centaur

The 260(3.7) booster is a 260-inch-diameter solid propellant motor with a propellant loading of approximately 3.4 million pounds. Other solid propellant boosters can also be constructed by clustering 156-inch-diameter solid rocket motors within an appropriate structure. The 4 x 1563 is a cluster of four, 3-segment, 156-inch-diameter motors. The SIC(4-F1) is a version of the SIC with four F1 engines (instead of the usual five) for use with the SIVB. The SIC(4F1)/SIVB is often called the Intermediate 20 (Int 20) and the SIC/SII is called the Intermediate 21 (Int 21). When considering the use of the Int 21 it should be remembered that the SII does not have a restart or extended coast capability as now configured. The SIVB(J2S) would be powered by the J2C engine which is a proposed modification to the standard J2 engine.

Performance of Selected Vehicles With the Burner II(2300) and Beryllium Burner II Velocity Packages (Figure IV-A-6)

TAT(3C)/BII(2300)

TAT(9C)/BII(2300)

SLV3A/BII(2300)

SLV3A/BE BII(2-burn)

TAT(9C)/Delta(TSE)/BII(2300)

TAT(H1-9C)/Delta(TSE)/BE BII(2-burn)

SLV3C/Centaur/BII(2300)

SLV3C/Centaur/BE BII(1-burn)

Titan IIIB(Str. CI + 2A3)/Centaur/BE BII(1-burn)

This Figure illustrates the performance increase available for missions involving small payloads through the use of a small upper stage as a velocity package. BII(2300) is the Burner II upper stage loaded with 2300 pounds of a conventional solid propellant. The BE BII is a possible Burner II stage built around a proposed solid rocket motor using a high energy propellant containing beryllium. This stage might be available in both 1-burn and 2-burn configurations.

Performance of Various Titan Vehicle Options for Interplanetary Missions (Figure IV-A-7)

Titan IIIC/BII(2300)

Titan IIID/Centaur/BII(2300)

Titan IIID/Centaur(Str.)/BII(2300)

Titan IIIC(7)/BII(2300)

Titan IIID(7)/Centaur/BII(2300)

Titan IIID(7)/Centaur(Str.)/BII(2300)

Titan IIIC/BE BII(1-burn)

Titan IIIC(7)/BE BII(1-burn)

This Figure shows the performance of several Titan-based vehicles with either the Burner II(2300) or Beryllium Burner II velocity package. The Centaur(Str.) is a proposed stretched Centaur that would have a propellant capacity of approximately 45,000 pounds.

Performance of Selected Launch Vehicles With a Possible H_2/F_2 Versatile Upper Stage (VUS) (Figure IV-A-8)

TAT(9C)/VUS

TAT(9C)/VUS/TE364(2300)

SLV3C/VUS

Titan IIIB(Str. CI)/VUS

Titan IIIB(Str. CI + 2A3)/Centaur/VUS

SLV3C/Centaur/VUS

Titan IIID/Centaur/VUS

Titan IIID(7)/Centaur/VUS

260(3.7)/SIVB/Centaur/VUS

4 x 1563/SIVB(J2S)/Centaur/VUS

SIC(4F1)/SIVB(J2S)/Centaur/VUS

The Versatile Upper Stage (VUS) is a concept currently being studied by NASA. The performance shown here is based on assumed VUS characteristics pending the outcome of contractor studies. These studies may lead to different VUS characteristics including selection of a different propellant.

Performance of Selected Launch Vehicles with Possible Future Centaur(F) and Possible H_2/F_2 Versatile Upper Stage (VUS) (Figure IV-A-9)

Titan IIIB(Str. CI + 2A3)/Centaur(F)

Titan IIIB(Str. CI + 2A3)/Centaur(F)/VUS

Titan IIID/Centaur(F)

Titan IIID/Centaur(F)/VUS

Titan IIID(7)/Centaur(F)/VUS

2 x 1207/CI/Centaur(F-Str.)/Centaur(F)/VUS

260(3.7)/SIVB/Centaur(F)/VUS

4 x 1563/SIVB(J2S)/Centaur(F)/VUS

SIC(4F1)/SIVB(J2S)/Centaur(F)/VUS

The VUS is as described previously. Centaur (F) is based on design evolution and concentrated development effort to employ hydrogen-fluorine propellants in the Centaur. The 2 x 1207/Core I/Centaur(F-Str.) is a Titan IIID(7) with the Core II replaced by a stretched version of the Centaur (F). This stretched Centaur (F) would carry 100,000 lb of propellant.

Post-1975 Selected Possible Large Diameter Core Titan Vehicles (Figure IV-A-10)

2 x 1207/LDCI/CII

2 x 1207/LDCI/CRY CII

4 x 1207/LDCI/CII

4 x 1207/LDCI/CRY CII

2 x 1207/LDCI/CII/Centaur

2 x 1207/LDCI/CRY CII/Centaur

4 x 1207/LDCI/CII/Centaur

4 x 1207/LDCI/CRY CII/Centaur

The large diameter Core I (LDCI) is a 180-inch-diameter stage with a propellant loading of 1,050,000 lb. The cryogenic Core II (CRY II) is a 180-inch-diameter SIVB-like stage with a liquid H_2/O_2 propellant loading of 193,000 lb. "CII" refers to the standard Titan Core II. "2 x 1207" and "4 x 1207" refer, respectively, to 2 or 4 of the 7-segment, 120-inch-diameter solid rocket motor strapons. In cases where two strapons are used, the LDCI is ignited at liftoff. In cases where four strapons are used, the LDCI is ignited approximately 30-40 sec after liftoff. The use of the 156-inch-diameter strapons has not been shown here because it is not currently considered likely that they would be used in the Titan application.

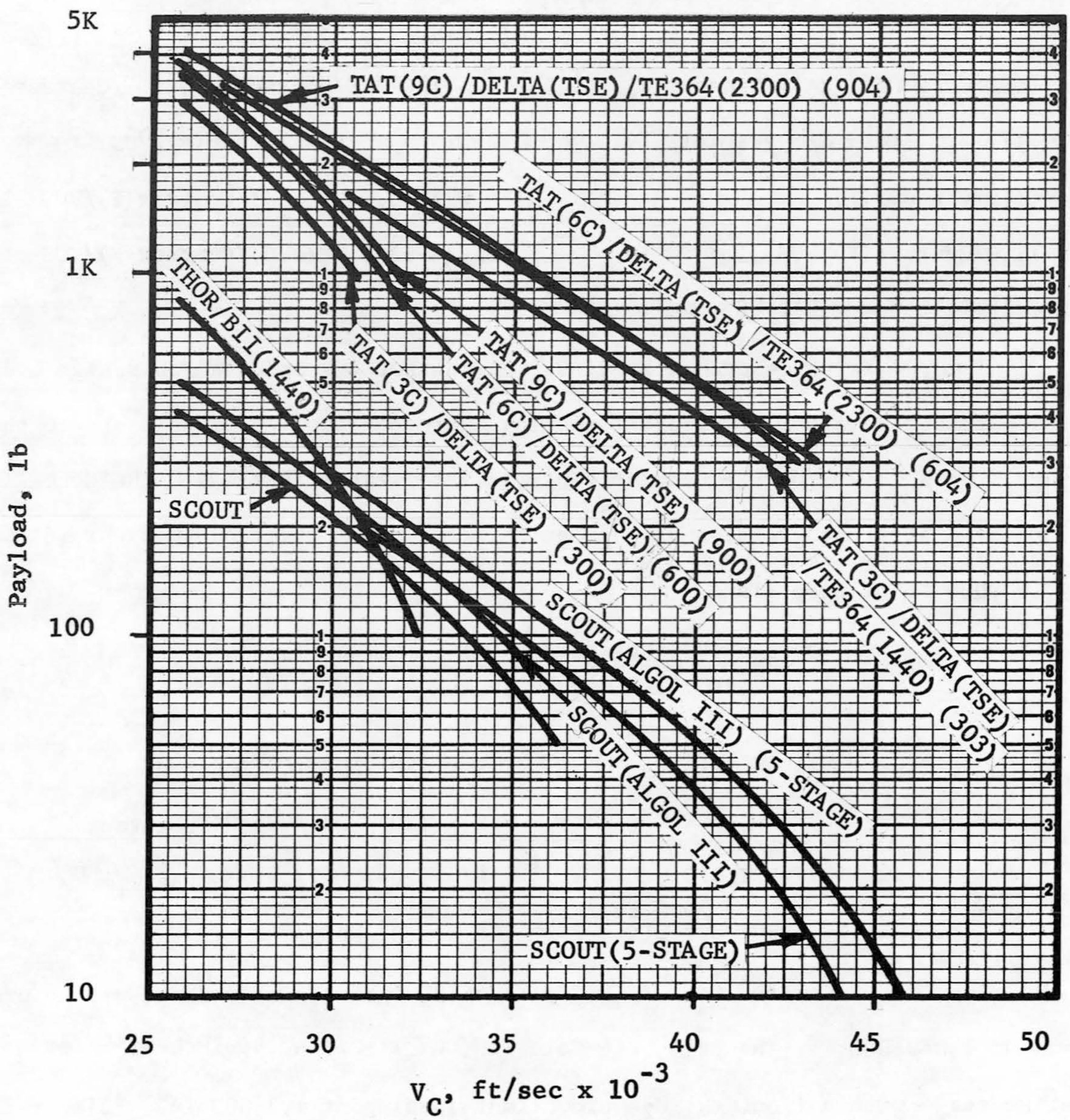


FIGURE IV-A-1. LAUNCH VEHICLE PERFORMANCE AVAILABLE IN 1971-1975 (GROUP 1)

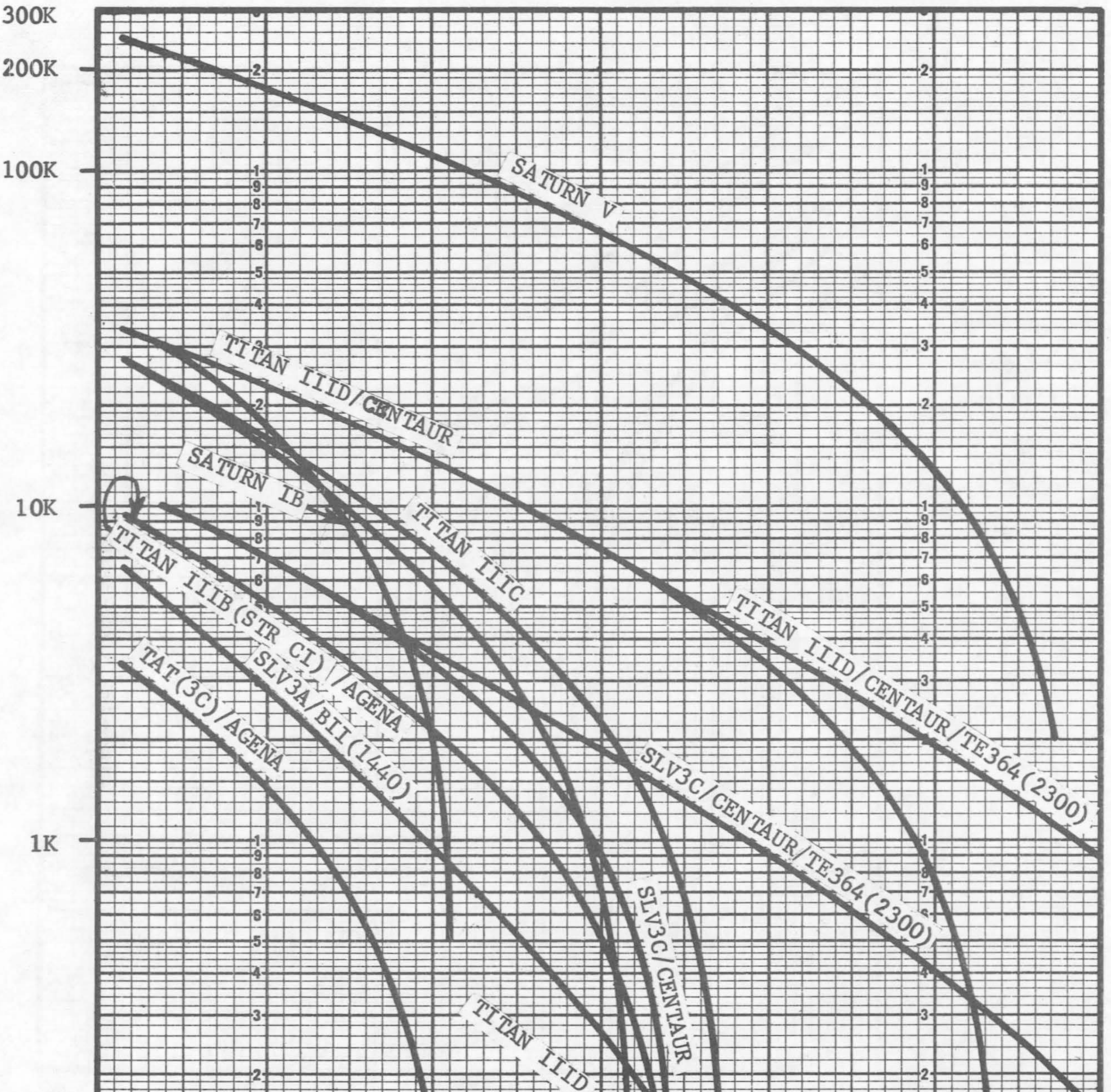


FIGURE IV-A-2. LAUNCH VEHICLE PERFORMANCE AVAILABLE IN 1971-1975

(GROUP 2)

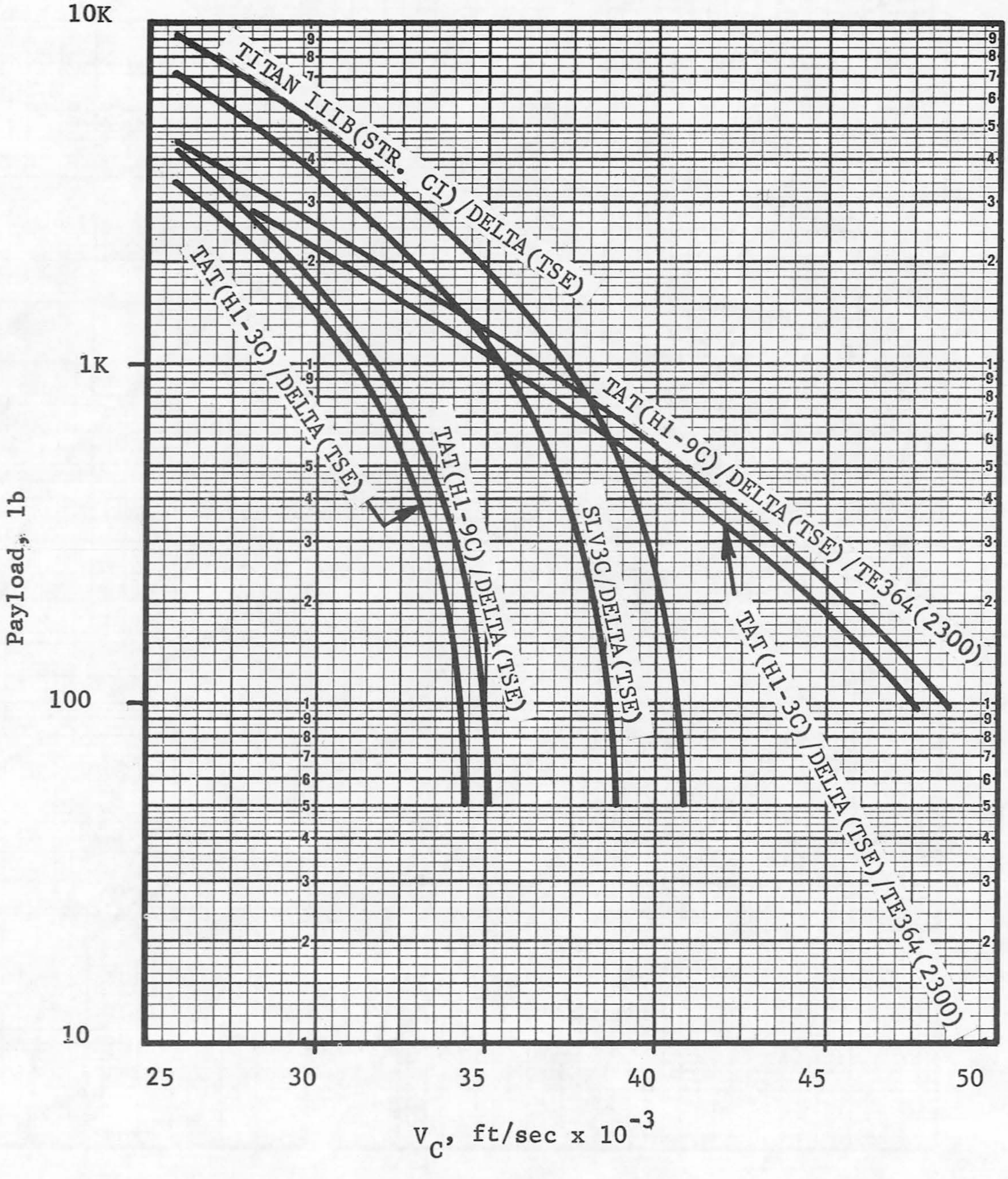


FIGURE IV-A-3. 1971-1975 IMPROVED LAUNCH VEHICLE PERFORMANCE POSSIBILITIES

(GROUP 1)

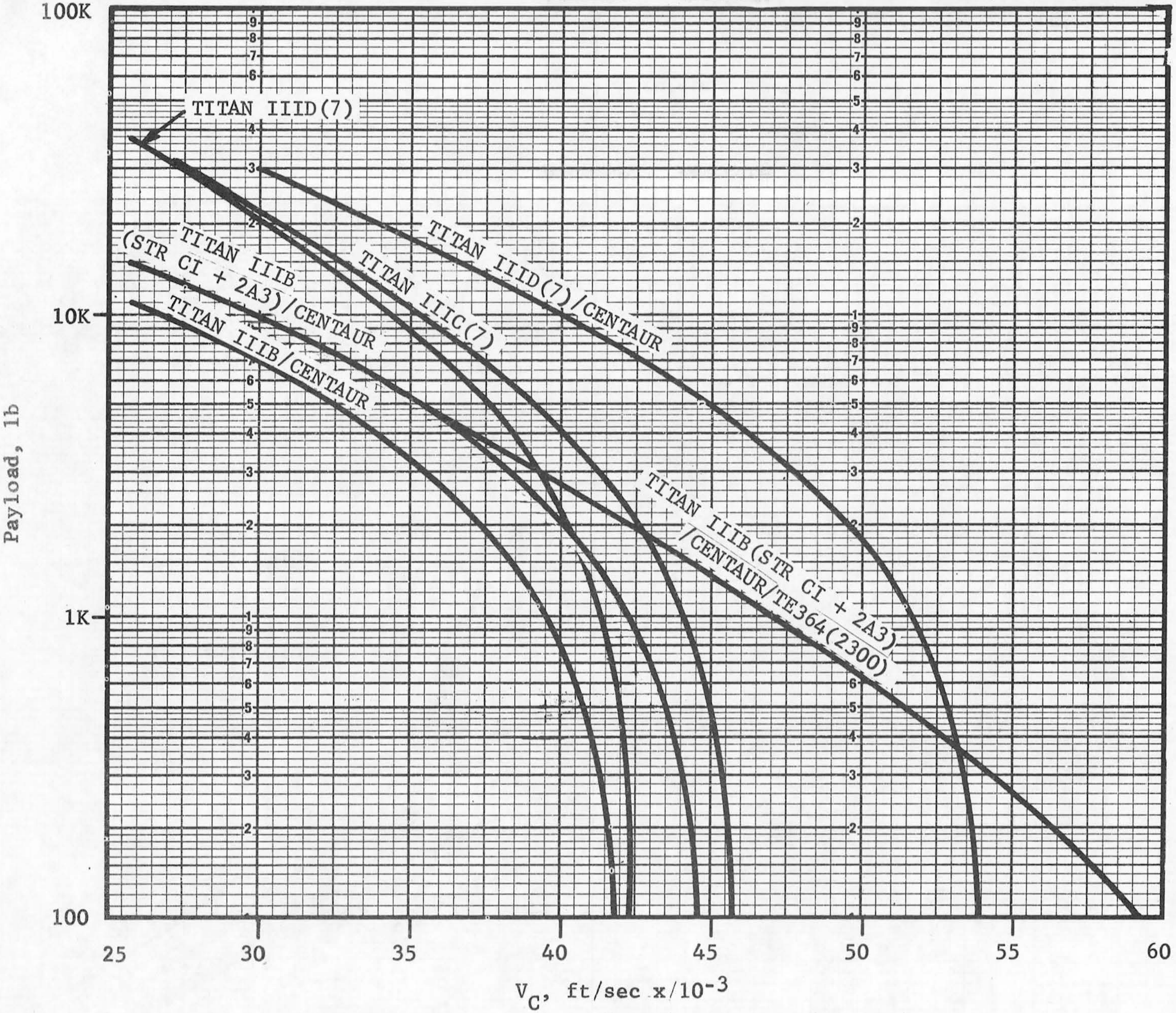


FIGURE IV-A-4. 1971-1975 IMPROVED LAUNCH VEHICLE PERFORMANCE POSSIBILITIES

(GROUP 2)

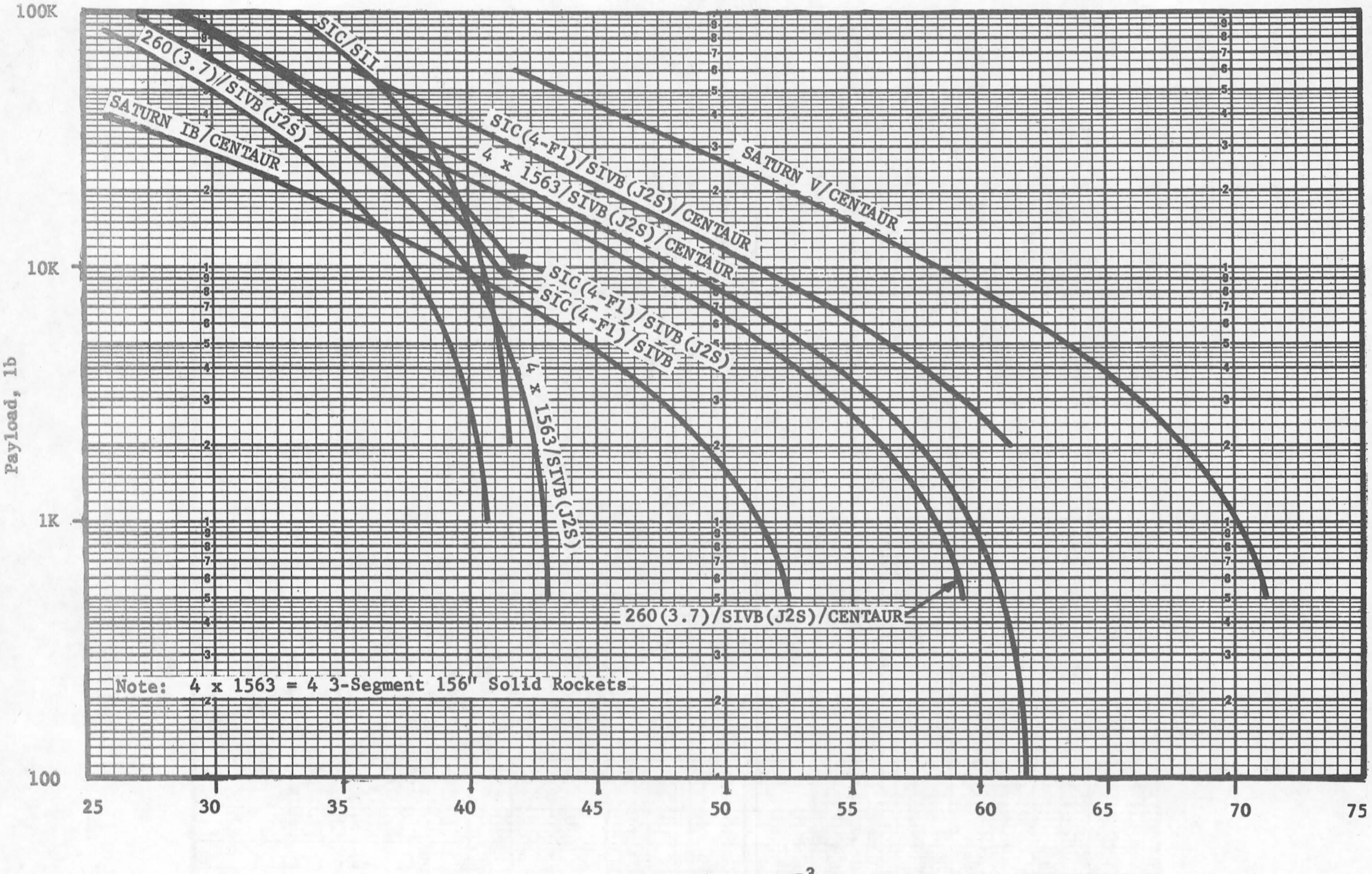


FIGURE IV-A-5. POSSIBLE NEAR-TERM SATURN CLASS LAUNCH VEHICLES

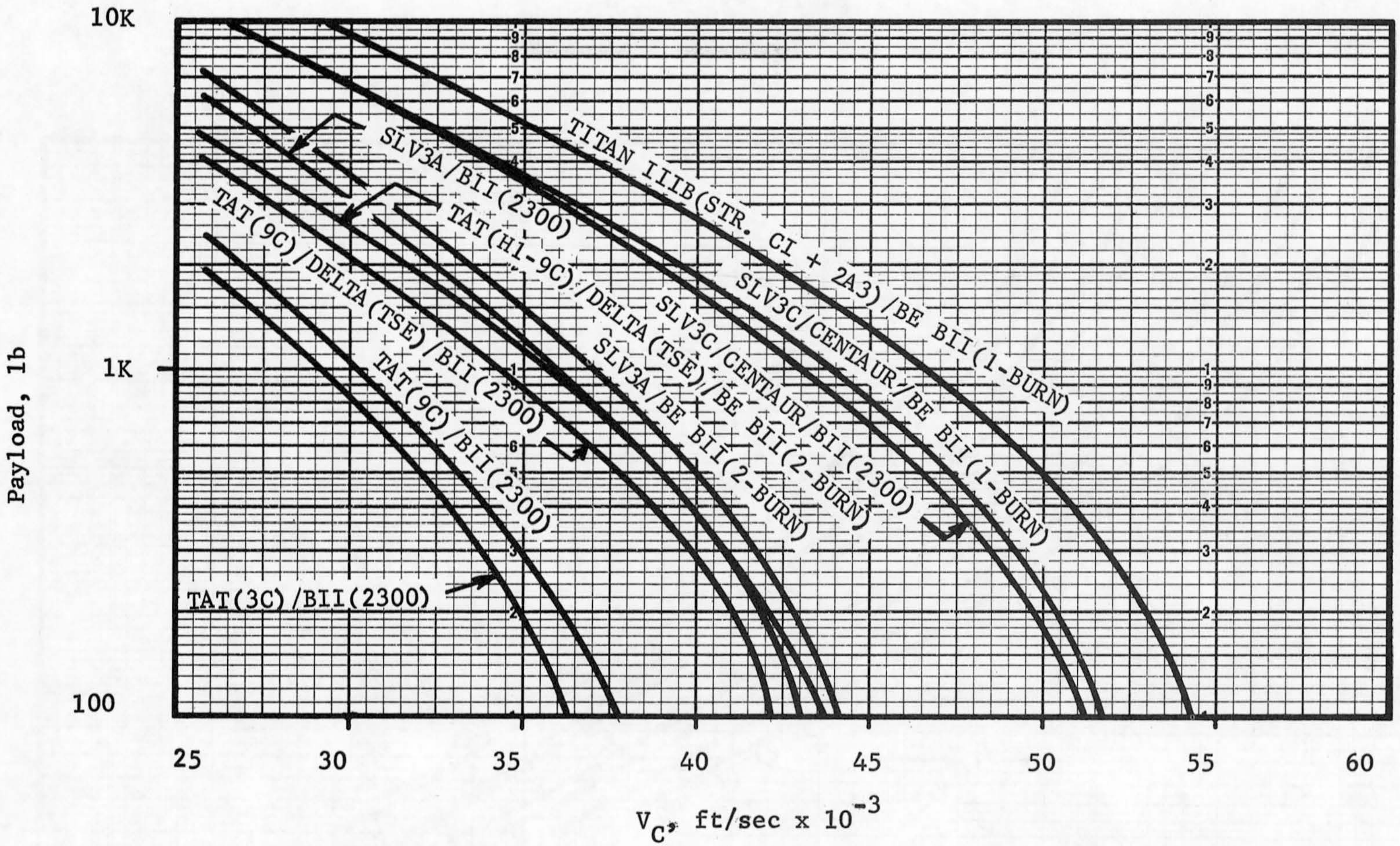


FIGURE IV-A-6. PERFORMANCE OF SELECTED VEHICLES WITH THE BURNER II(2300) AND BERYLLIUM BURNER II VELOCITY PACKAGES

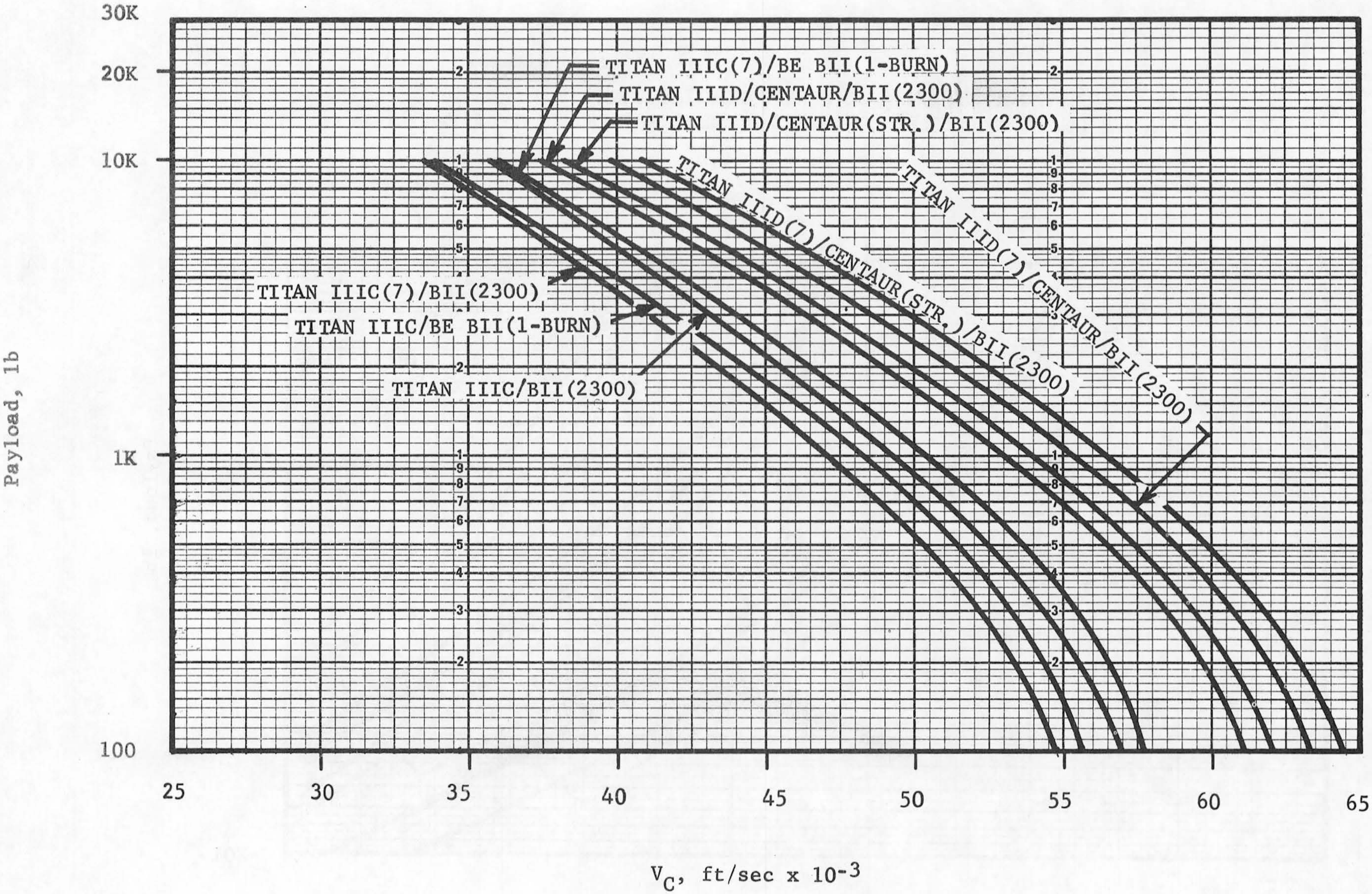


FIGURE IV-A-7. PERFORMANCE OF VARIOUS TITAN VEHICLE OPTIONS FOR INTERPLANETARY MISSIONS

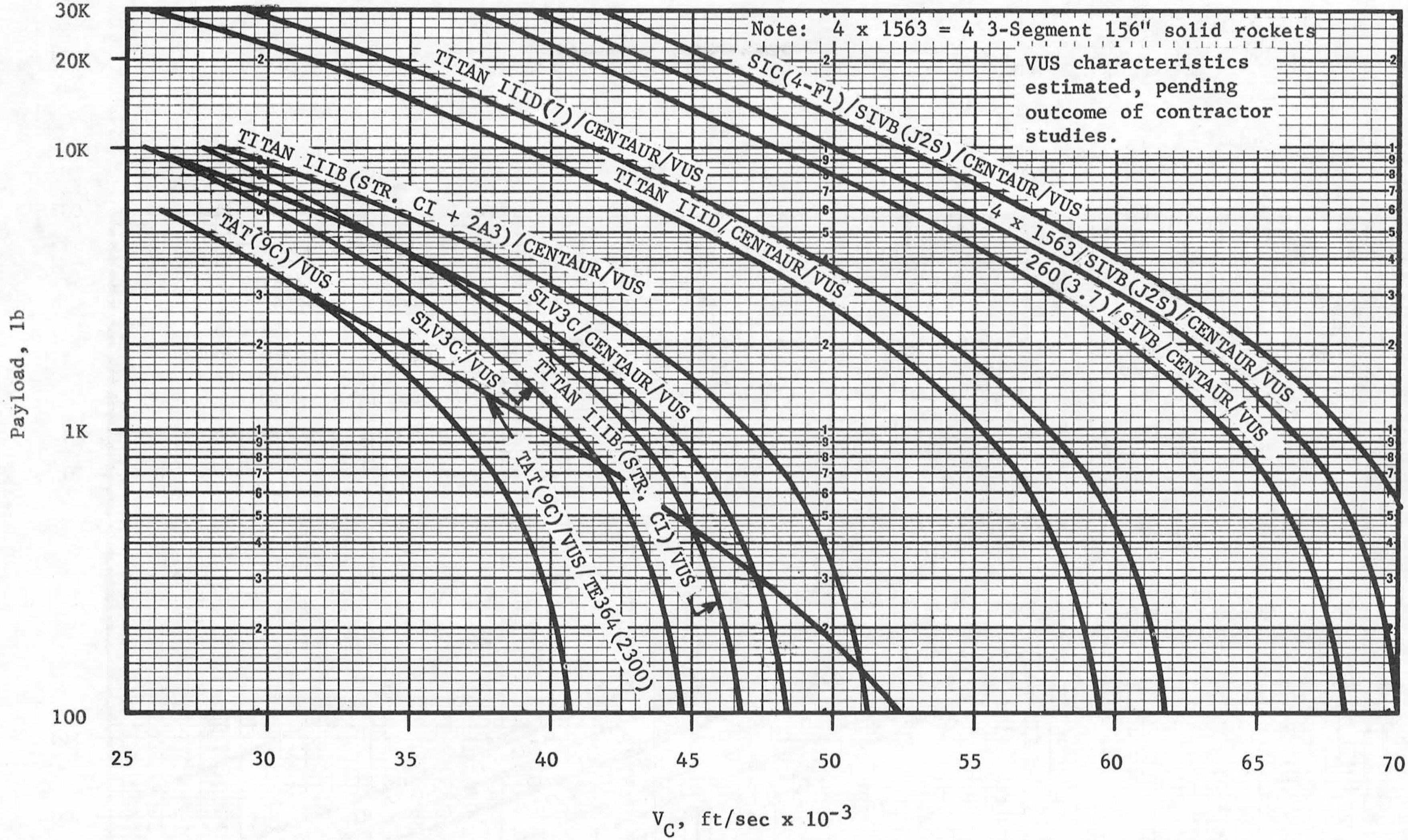


FIGURE IV-A-8. PERFORMANCE OF SELECTED LAUNCH VEHICLES WITH A POSSIBLE H_2/F_2 VERSATILE UPPER STAGE (VUS)

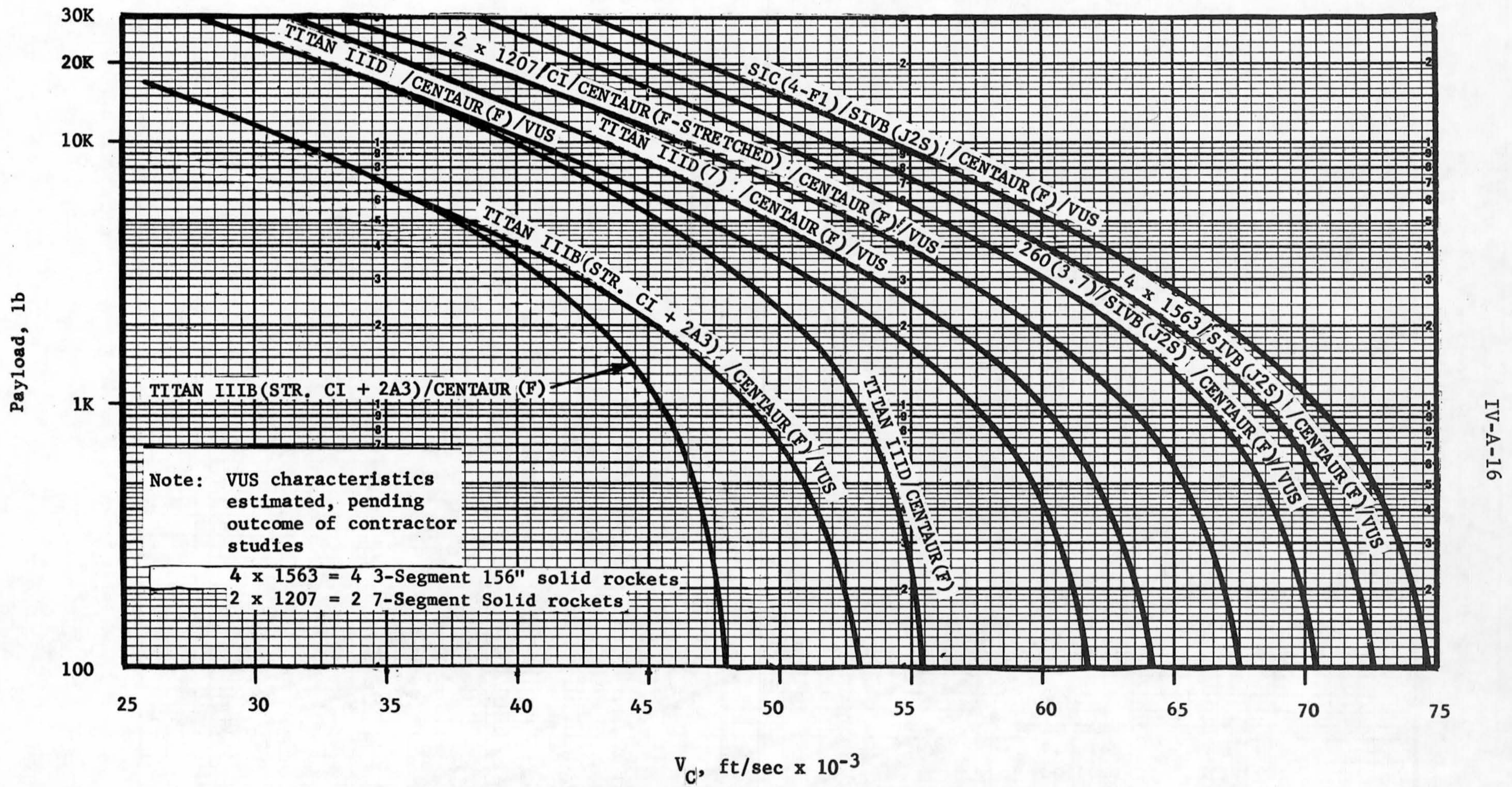


FIGURE IV-A-9. PERFORMANCE OF SELECTED LAUNCH VEHICLES WITH POSSIBLE FUTURE CENTAUR (F) AND POSSIBLE H_2/F_2 VERSATILE UPPER STAGE (VUS)

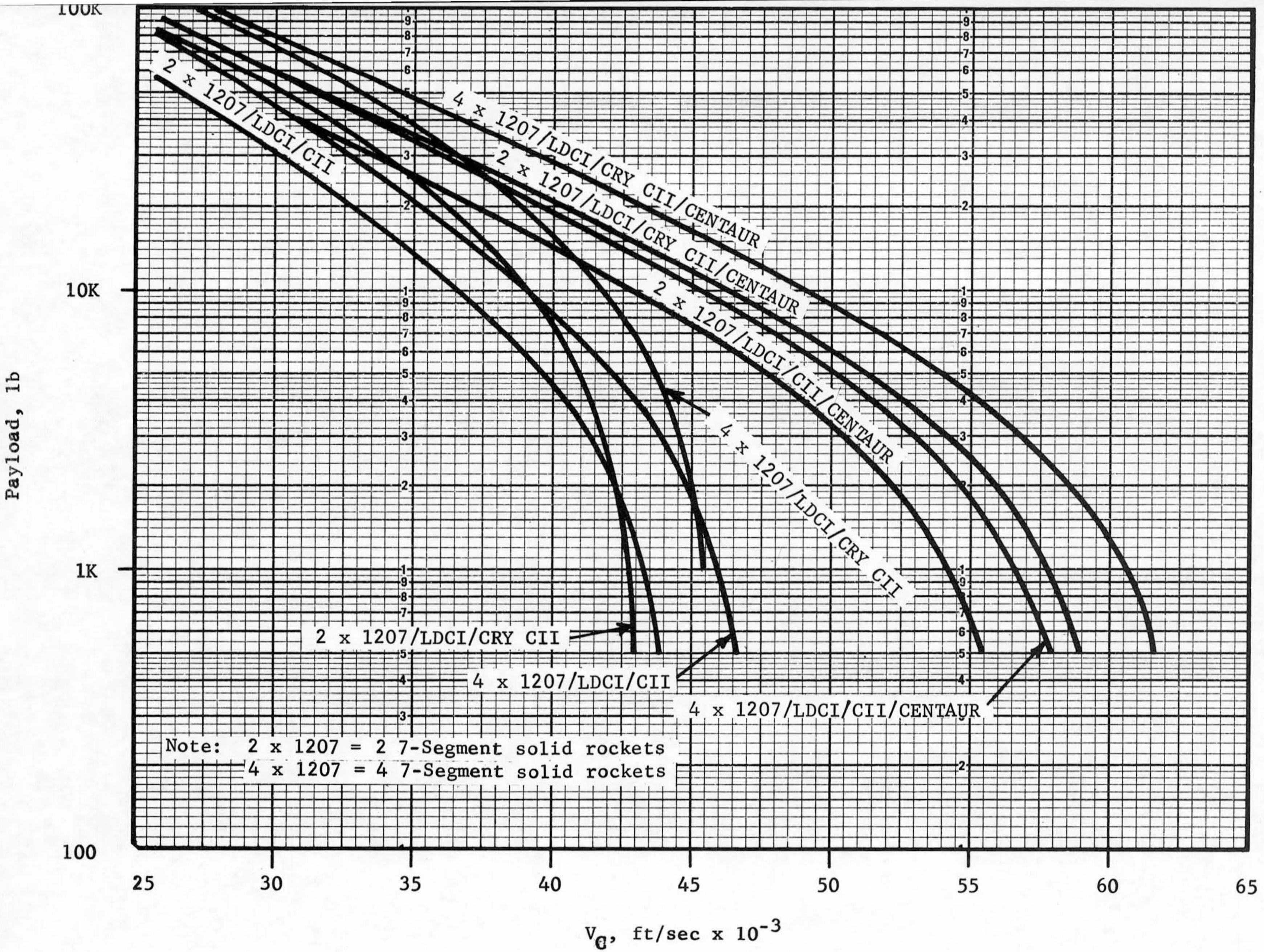


FIGURE IV-A-10. POST-1975 SELECTED POSSIBLE LARGE DIAMETER CORE TITAN VEHICLES

SECTION IV-B

EARTH ORBITAL CAPABILITY FOR
SELECTED EXPENDABLE LAUNCH VEHICLES

SECTION IV-B

EARTH ORBITAL CAPABILITY FOR SELECTED EXPENDABLE LAUNCH VEHICLES

Because of constraints imposed on stage restart, coast time, guidance, and other limitations, it is necessary to present separate data for estimating the Earth orbital performance of Scout, Delta (various configurations), SLV3C/Centaur(single burn), Titan IIIC (single burn), and Titan IIID launch vehicle configurations. Shown in Figures IV-B-1, IV-B-2, and IV-B-3, respectively, are the payload capabilities for Scout launches due east from Wallops Island, due east from San Marco, and in polar orbits from the WTR. Figures IV-B-4 and IV-B-5 show orbital capability for the current 5-stage Scout for eastward launches from Wallops Island and polar orbits from the WTR. Orbital capability of possible future uprated Scout configurations using a 44-in.-diameter Algol III first stage is presented in Figures IV-B-6 through IV-B-9. Data for the 4-stage Scout configuration with Algol III are shown in Figures IV-B-6 and IV-B-7 for launch from Wallops Island and the WTR, respectively. Data for the 5-stage uprated vehicle are shown in Figures IV-B-8 and IV-B-9 for Wallops Island and the WTR, respectively. All of the Scout performance shown is based on the use of the standard, 34-inch-diameter Scout shroud. A larger, 42-inch-diameter shroud is available (see section V, Figure V-1a and V-1b). When the larger shroud is used the payload that the Scout vehicles can place in orbit is reduced by approximately 25 pounds.

TAT(3C)/Delta(TSE)/TE364(1440), with 3 Castor II thrust augmentation, data are shown for east launches from the ETR in Figure IV-B-10 and polar orbits from the WTR in IV-B-11. Data for the TAT(6C)/Delta(TSE)/TE364(2300) are shown in Figure IV-B-12 for the ETR and in Figure IV-B-13 for the WTR. Data for the TAT(9C)/Delta(TSE)/TE364(2300) are presented in Figure IV-B-14 and Figure IV-B-15 for the ETR and WTR, respectively.

The TAT/Delta vehicles are frequently identifed by a three-digit numerical designation. In the appropriate figures in this section these designations are listed in parentheses beside the vehicle names. The first digit corresponds to the number of Castor II solid-rocket-motors that are strapped to the Thor booster for thrust augmentation. The second digit is always a zero (for the vehicles shown) and identifies the second stage as the Delta (TSE). The third digit indicates whether and/or which third is involved. Thus, a zero (0) indicates a two-stage vehicle, a three (3) indicates that the vehicle has a TE364(1440) third stage, a four (4) indicates that the third stage is a TE364(2300).

Figure IV-B-16 shows the orbital capability of the SLV3C/Centaur(single burn) for east launches from the ETR. Figures IV-B-17 and IV-B-18 show a single-burn orbital capability of the Titan IIIC with Operational Transtage for launches from the ETR and the WTR, respectively. Figure IV-B-19 shows the orbital capability for the Titan IIID for launches from the ETR and the WTR.

In all cases, the curves shown represent the best data available at the date of publication. Unlike the data in Section IV-A, no allowance has been made for payload adapters. The payload values read from the graphs in this section must include the weight of the spacecraft adapter. Range safety considerations (such as the impact point of lower stages) may cause the Earth orbital performance of some vehicles to be less than that given in this section. For information on this subject as well as for estimates or orbital payload capabilities for other possible uprated TAT-based configurations and for other vehicles with nonrestartable last stages, or other launch azimuths or perigee values, contact Mr. J. E. McGolrick, Mr. B. C. Lam, or Mr. J. W. Haughey in Launch Vehicle and Propulsion Programs of NASA OSSA.

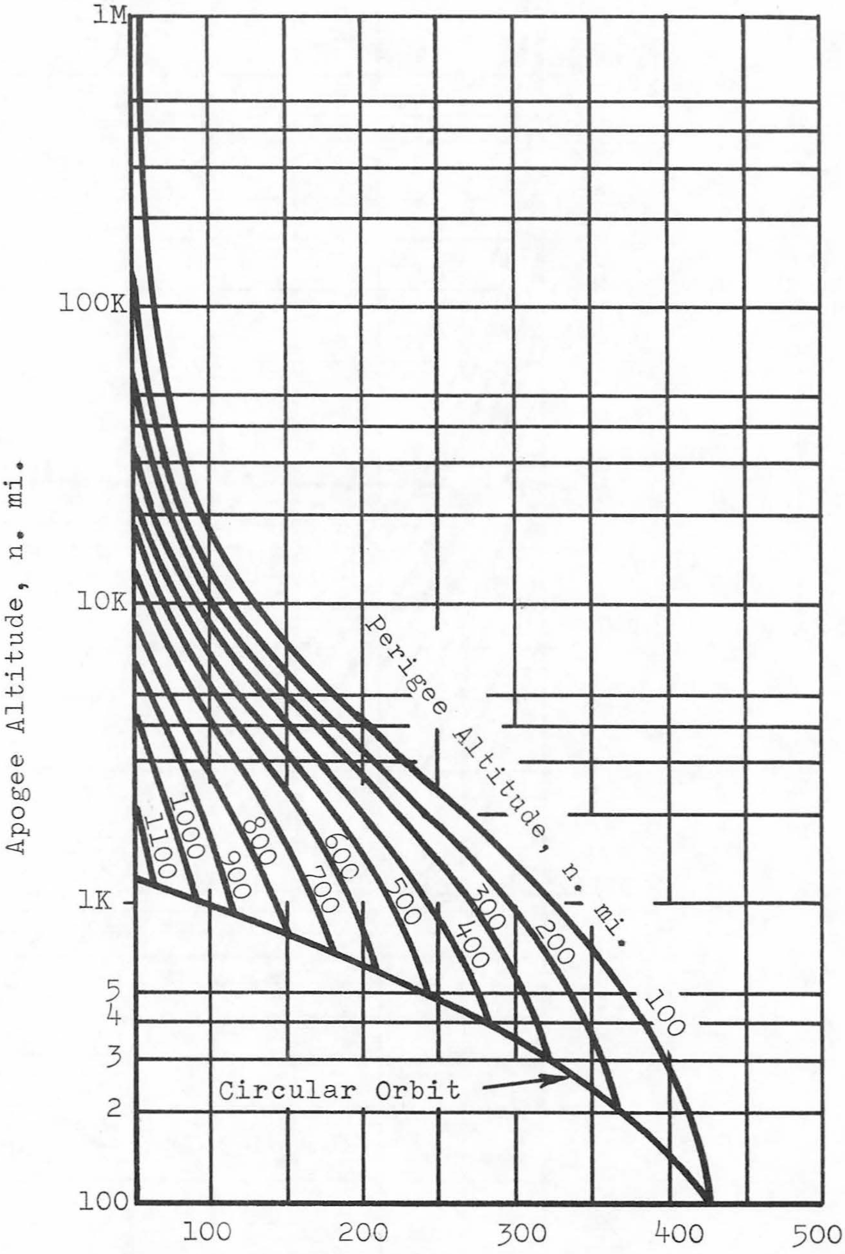


FIGURE IV-B-1. SCOUT ORBIT CAPABILITY-
DUE EAST (WALLOPS)

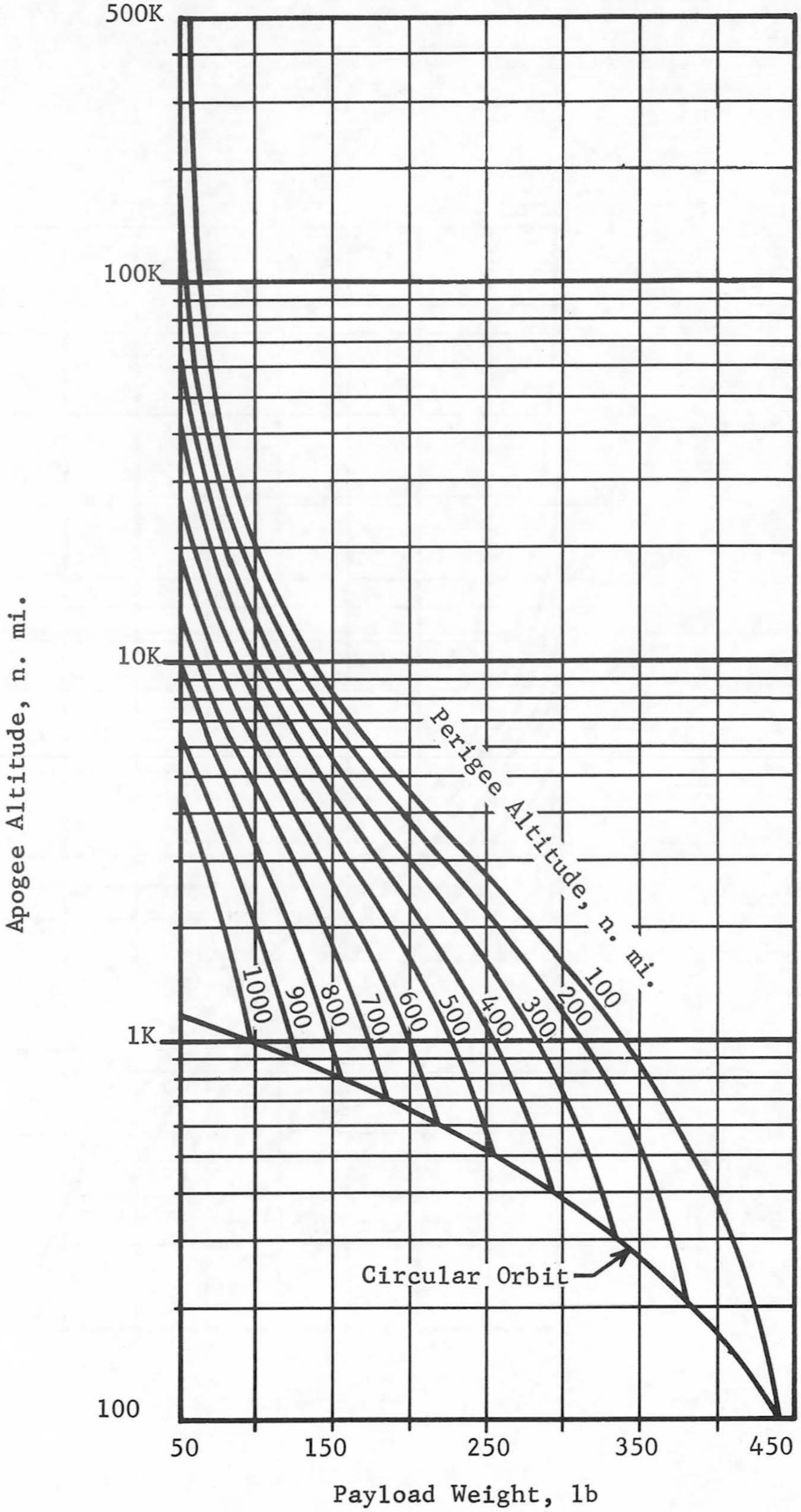


FIGURE-IV-B-2. SCOUT ORBIT CAPABILITY—
DUE EAST (SAN MARCO)

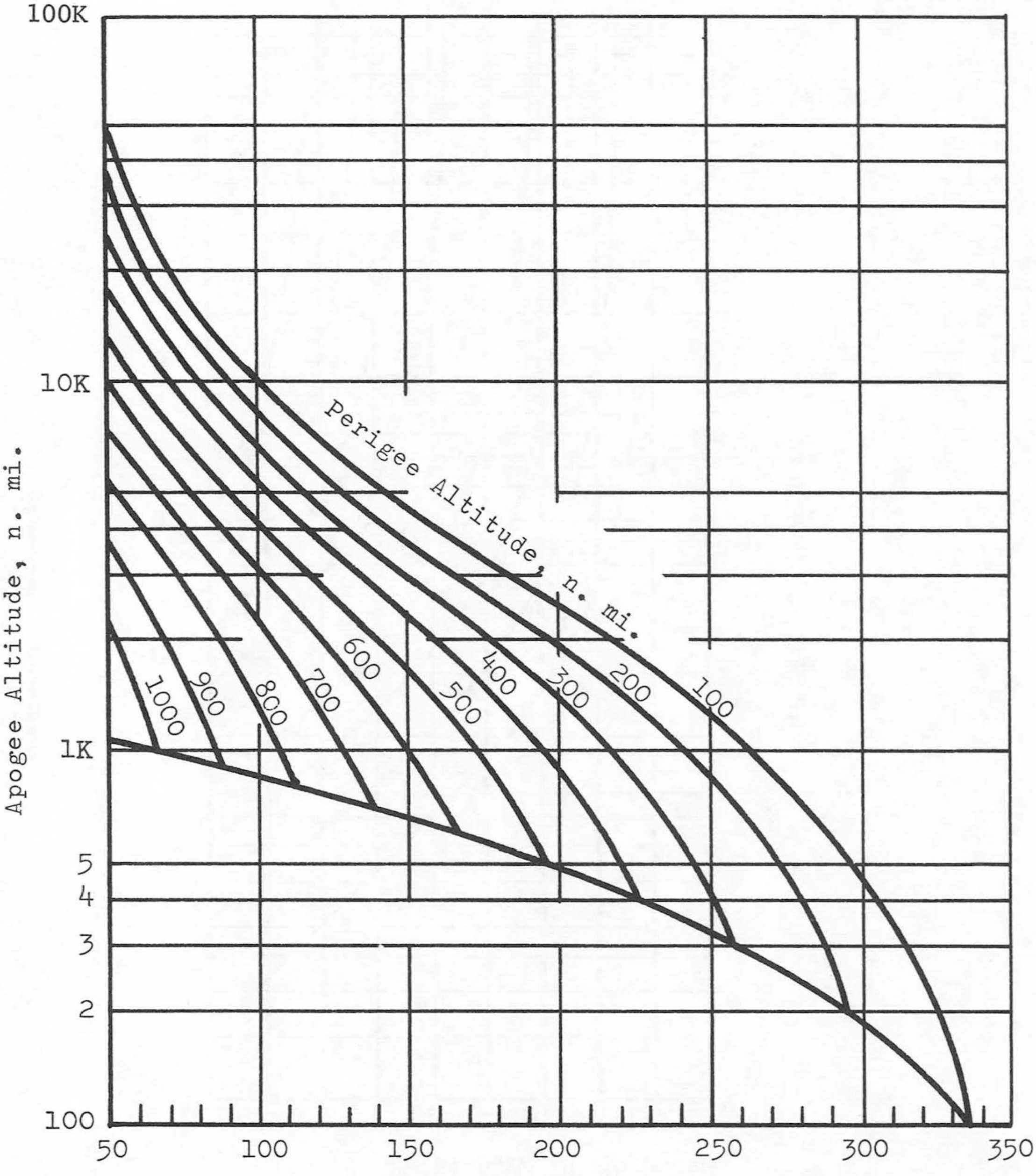


FIGURE IV-B-3. SCOUT POLAR ORBIT CAPABILITY-WTR

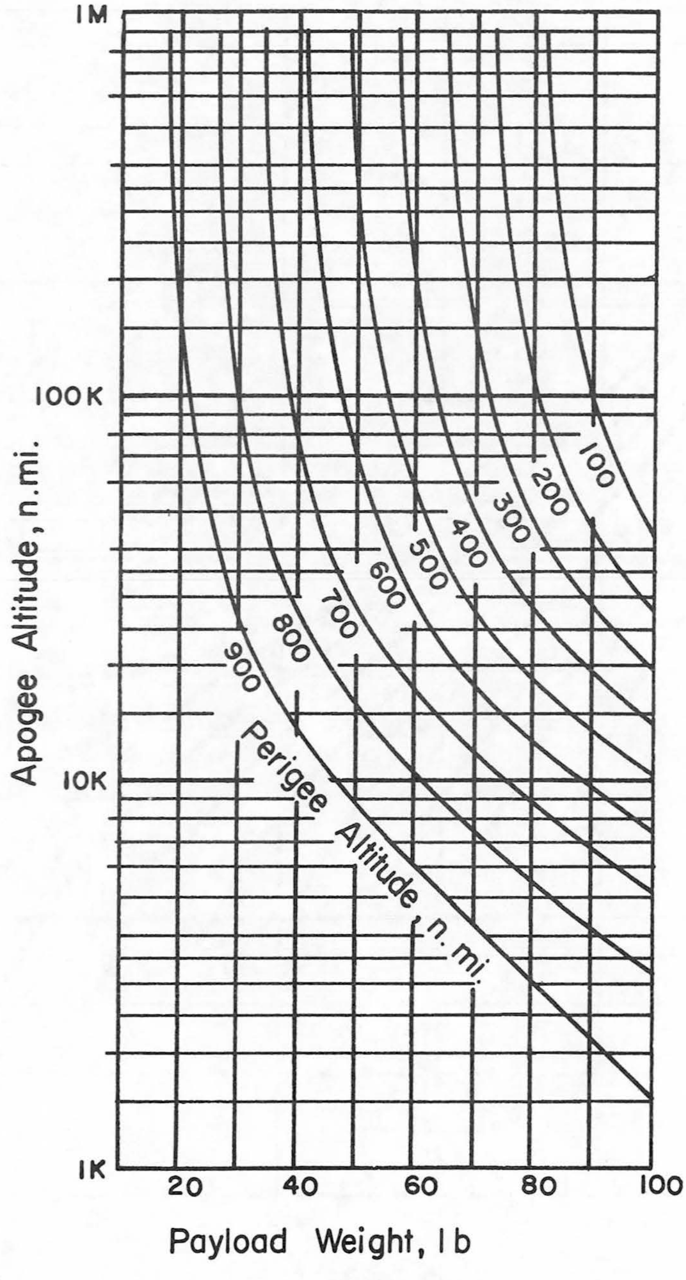


FIGURE IV-B-4. 5-STAGE SCOUT ORBITAL CAPABILITY—DUE EAST (WALLOPS)

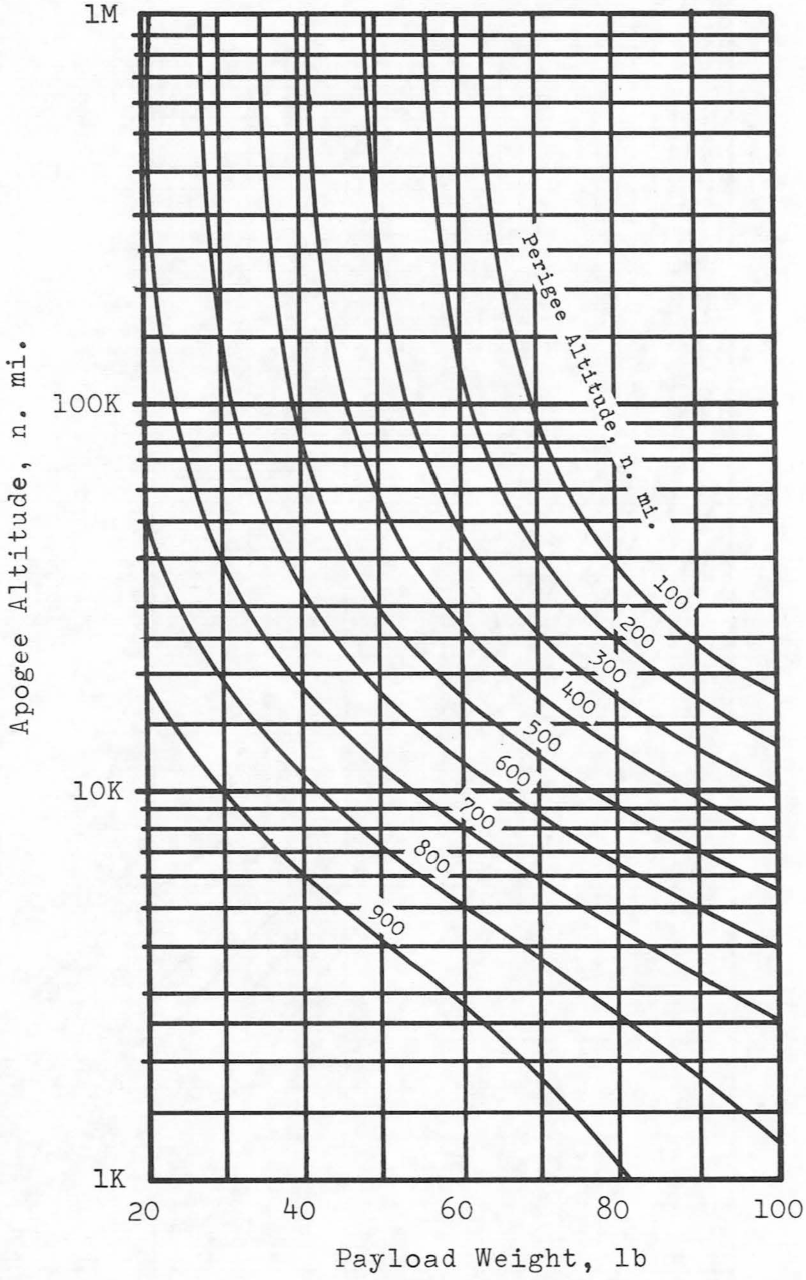


FIGURE IV-B-5. 5-STAGE SCOUT POLAR ORBIT CAPABILITY—WTR

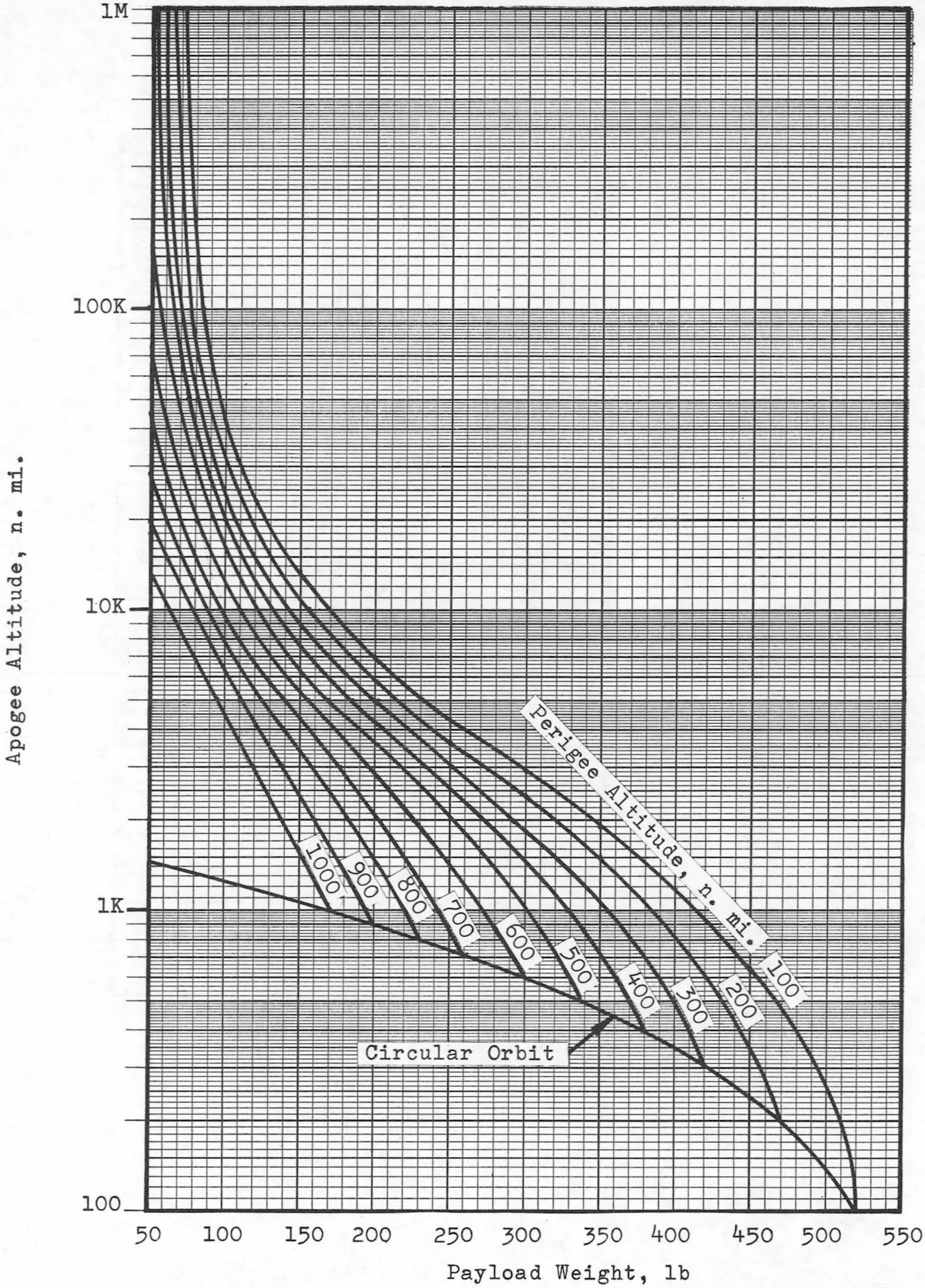


FIGURE IV-B-6. SCOUT (WITH 44-IN. ALGOL) ORBITAL CAPABILITY–
DUE EAST (WALLOPS)

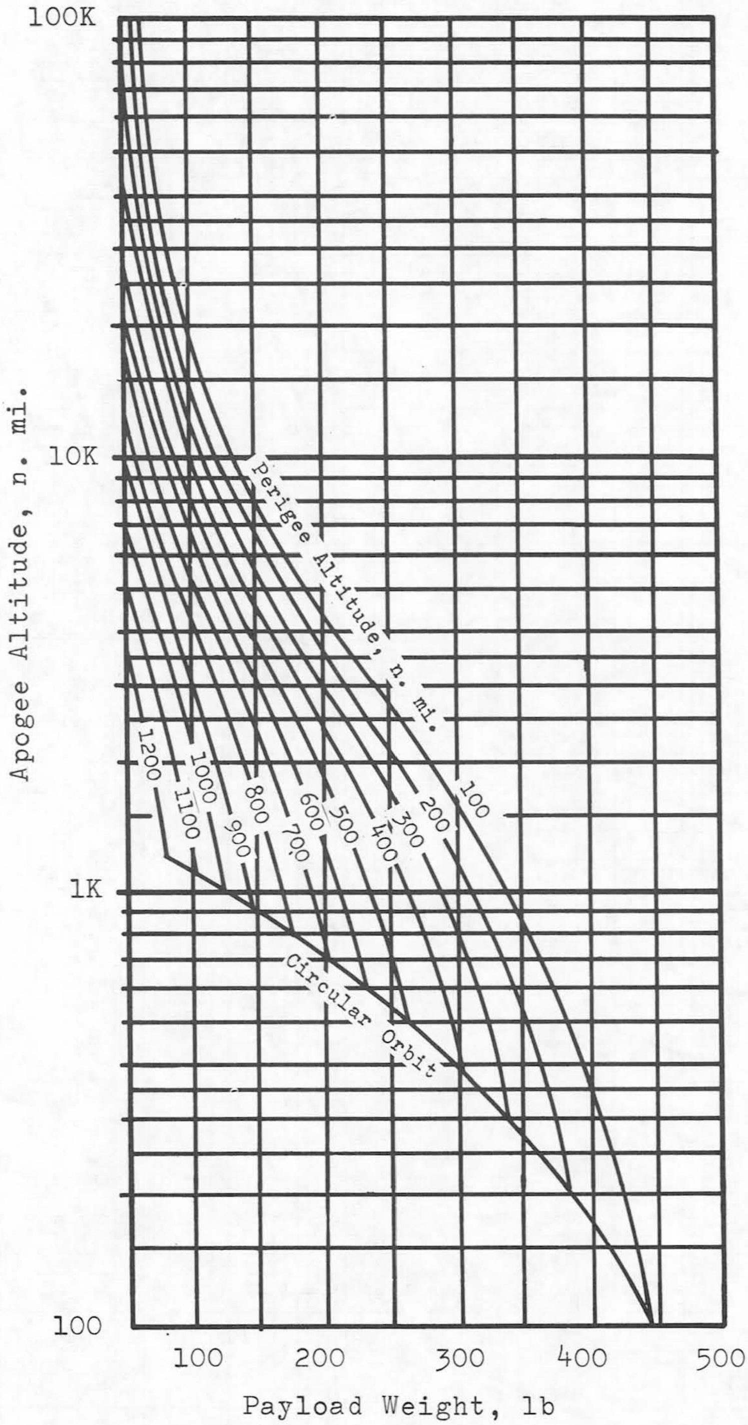


FIGURE IV-B-7. SCOUT (WITH 44-IN ALGOL) POLAR ORBIT CAPABILITY—WTR

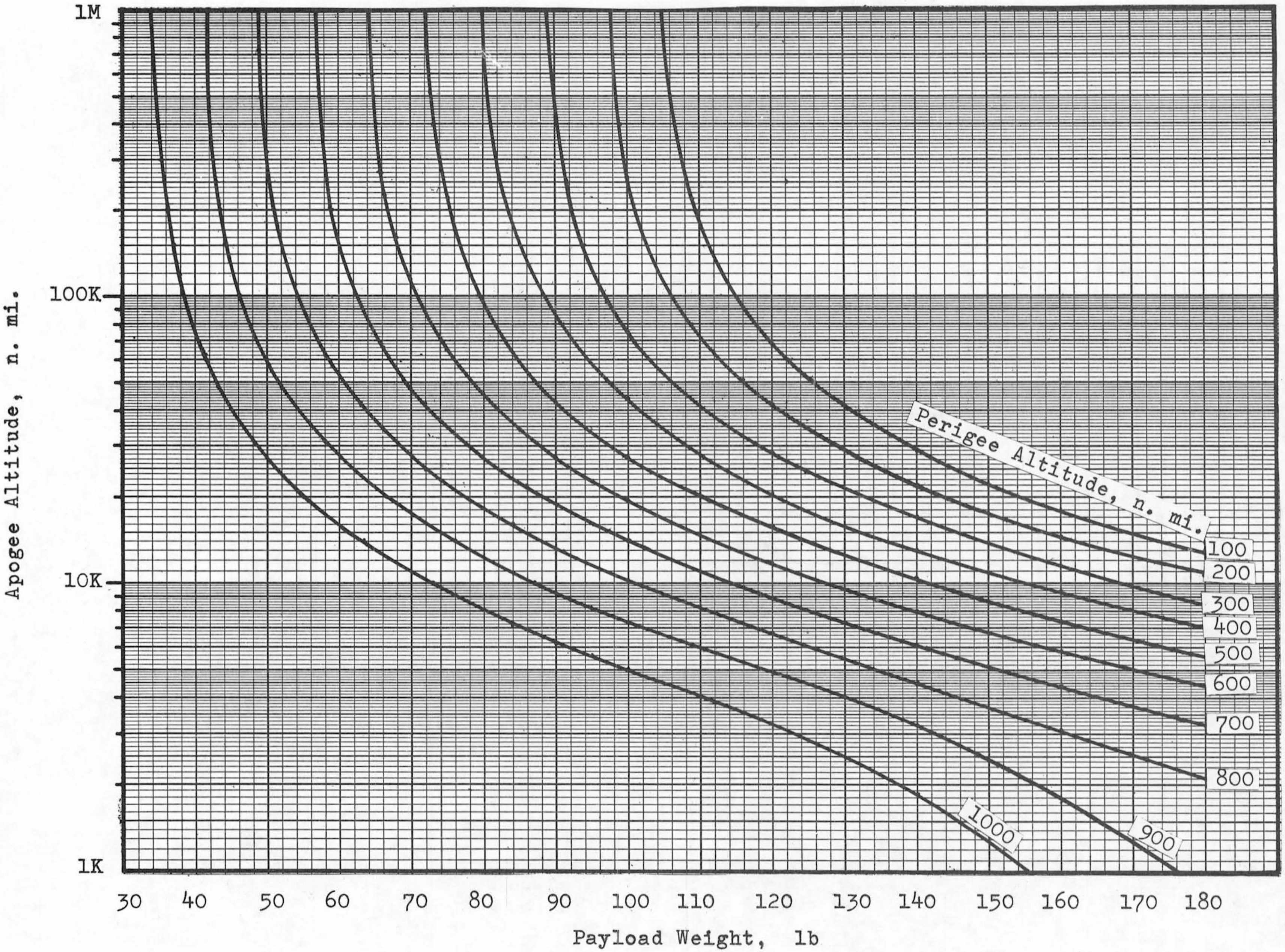


FIGURE IV-B-8. 5-STAGE SCOUT (WITH 44-IN. ALGOL) ORBITAL

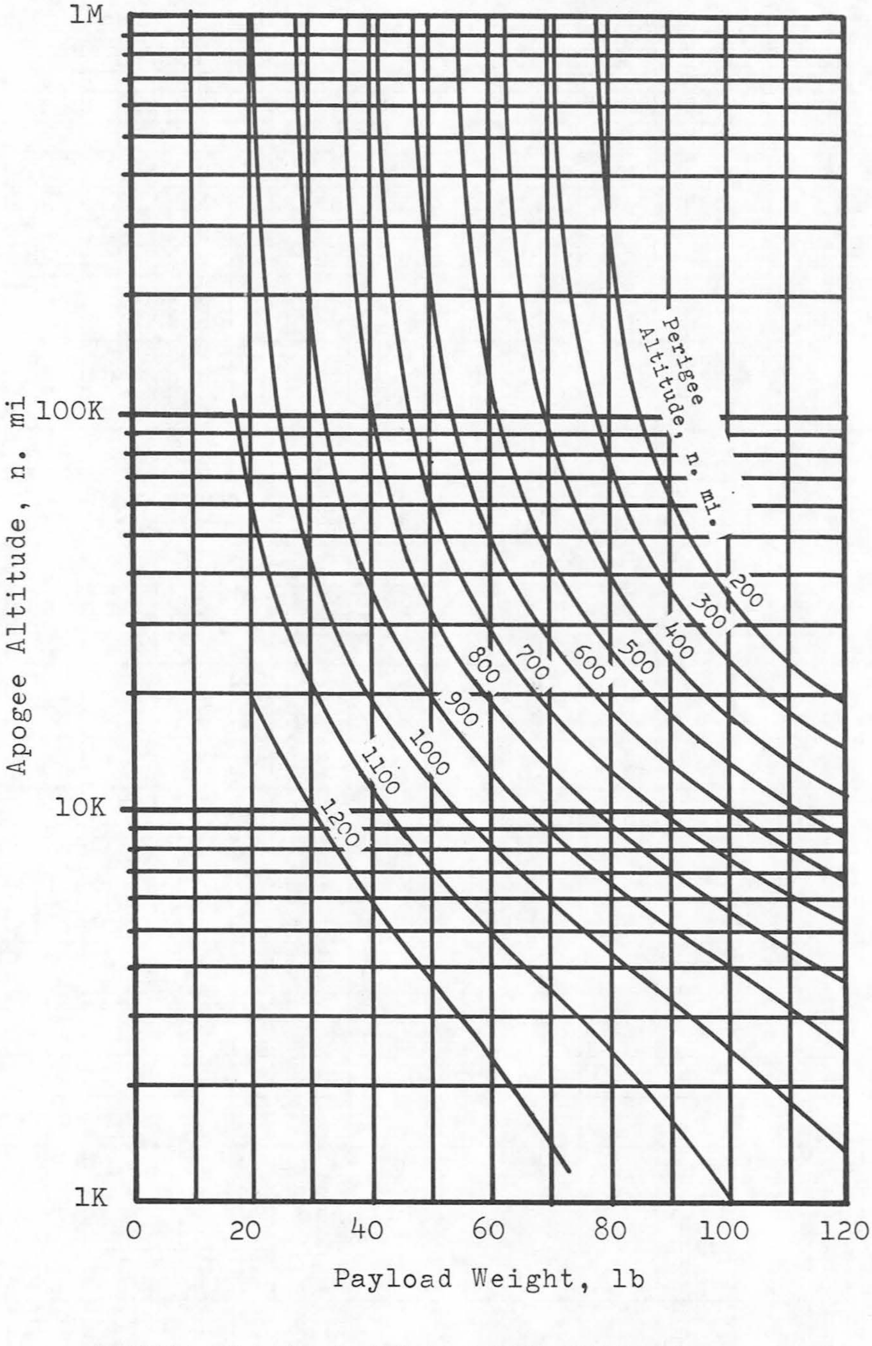


FIGURE IV-B-9. 5-STAGE SCOUT (WITH 44-IN. ALGOL) POLAR ORBIT CAPABILITY— WTR

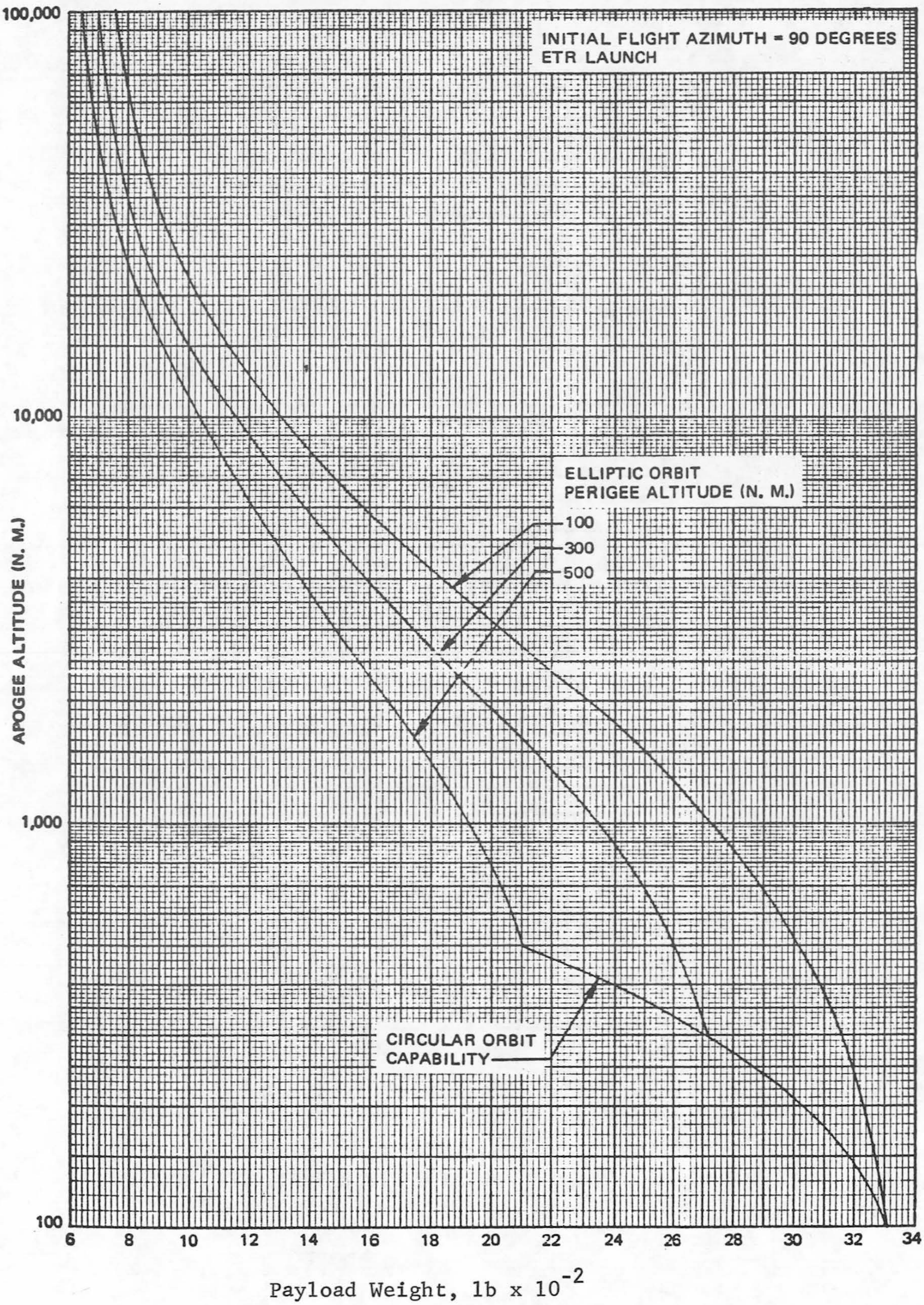
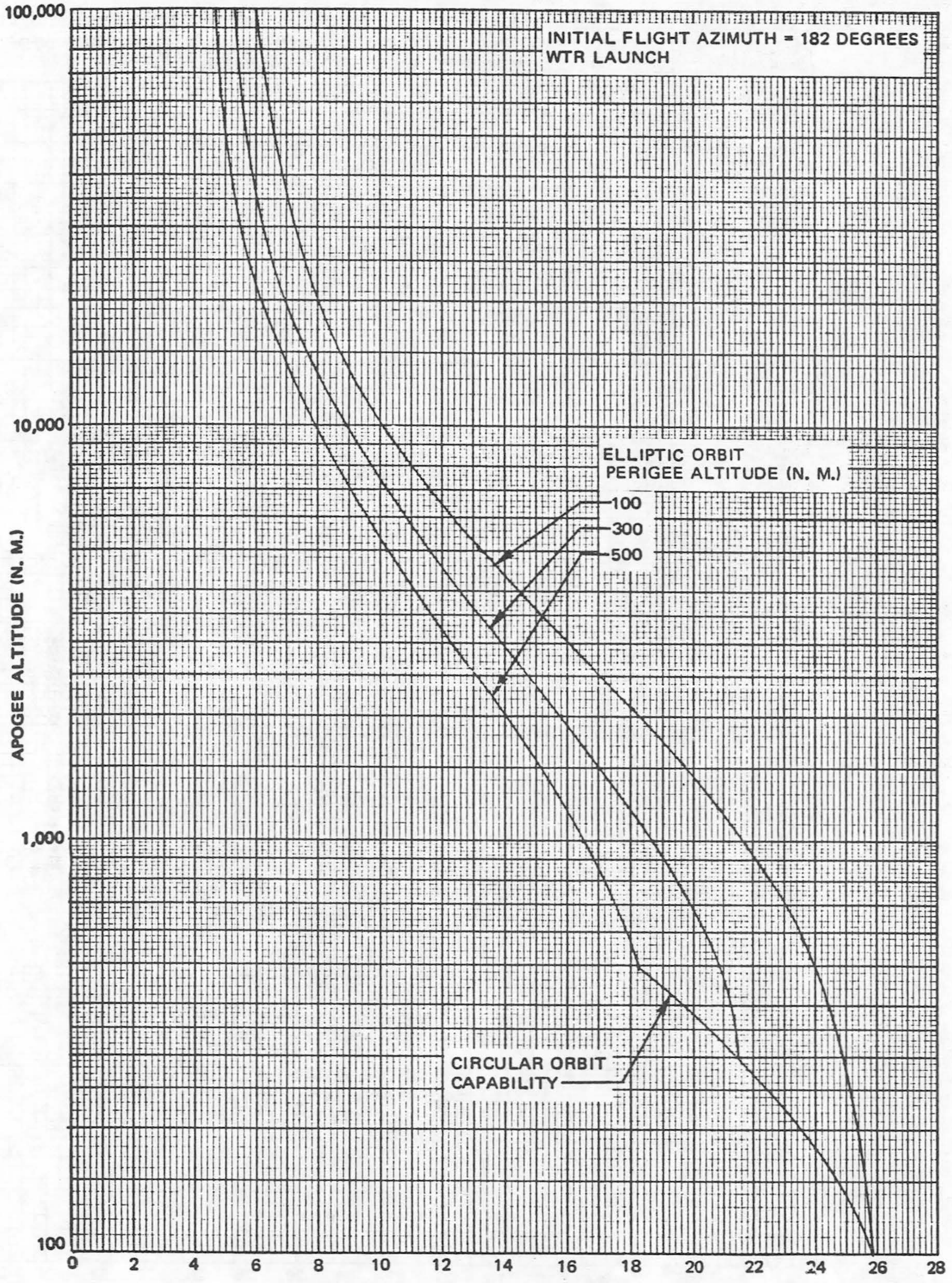
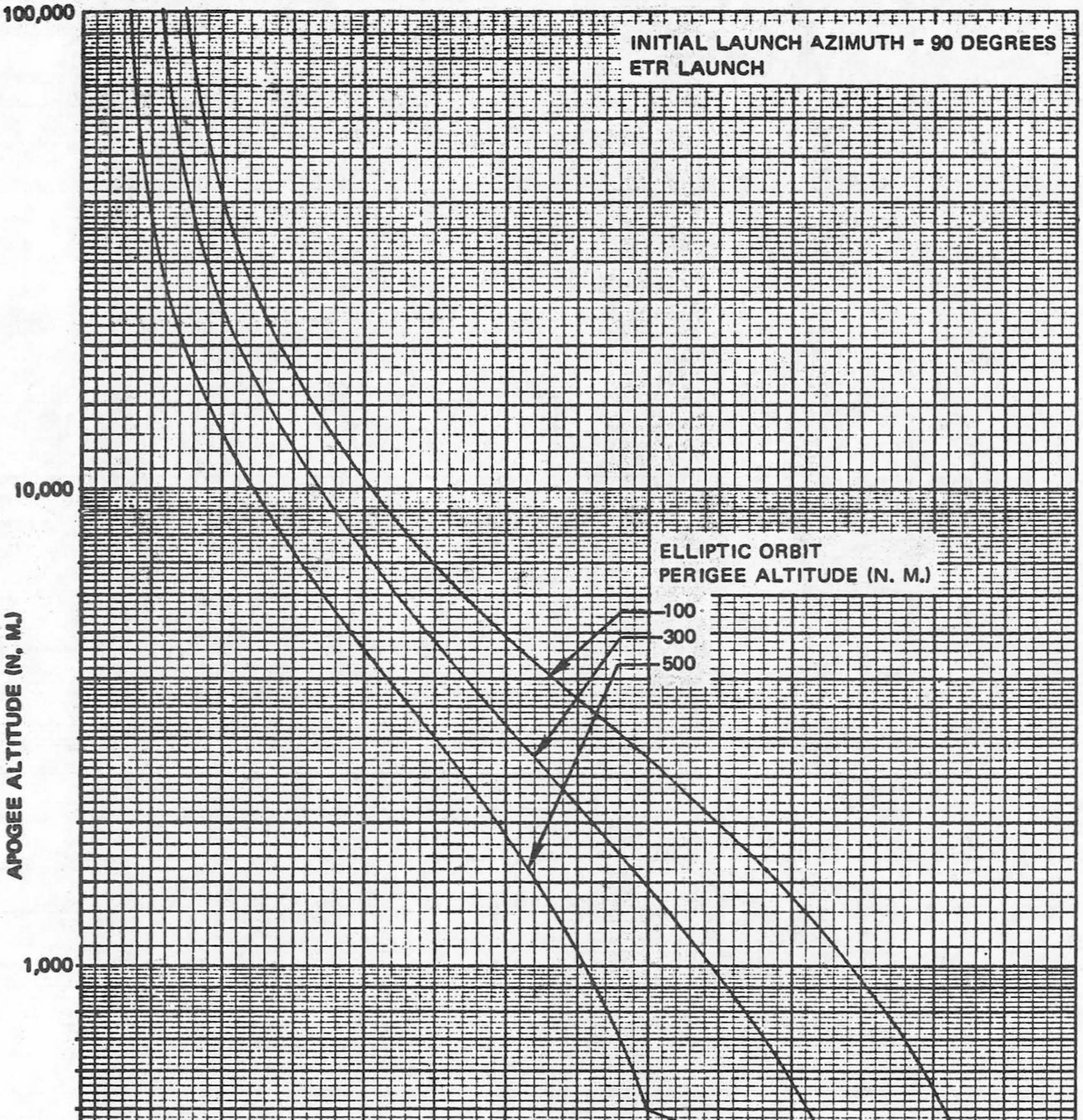


FIGURE IV-B-10. TAT(3C)/DELTA(TSE)/TE364(1440) (303) ORBITAL CAPABILITY, 90° LAUNCH FROM ETR



Payload Weight, lb x 10^{-2}

FIGURE IV-B-11. TAT(3C)/DELTA(TSE)/TE364(1440) (303) ORBITAL CAPABILITY, 182° LAUNCH FROM WTR



Payload Weight, lb x 10^{-2}

FIGURE IV-B-12. TAT(6C)/DELTA(TSE)/TE364(2300) (604) ORBITAL CAPABILITY, 90° LAUNCH FROM ETR

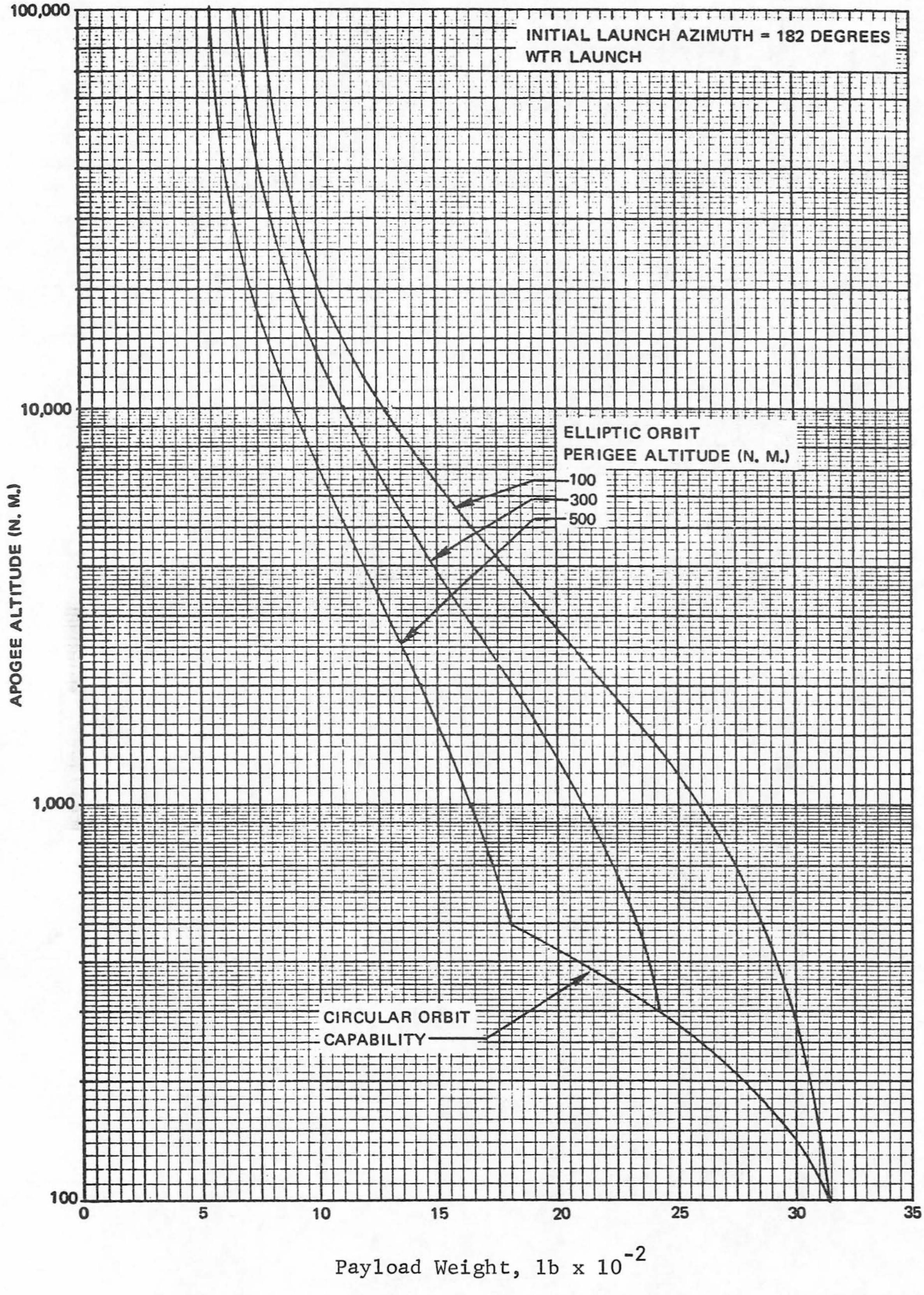
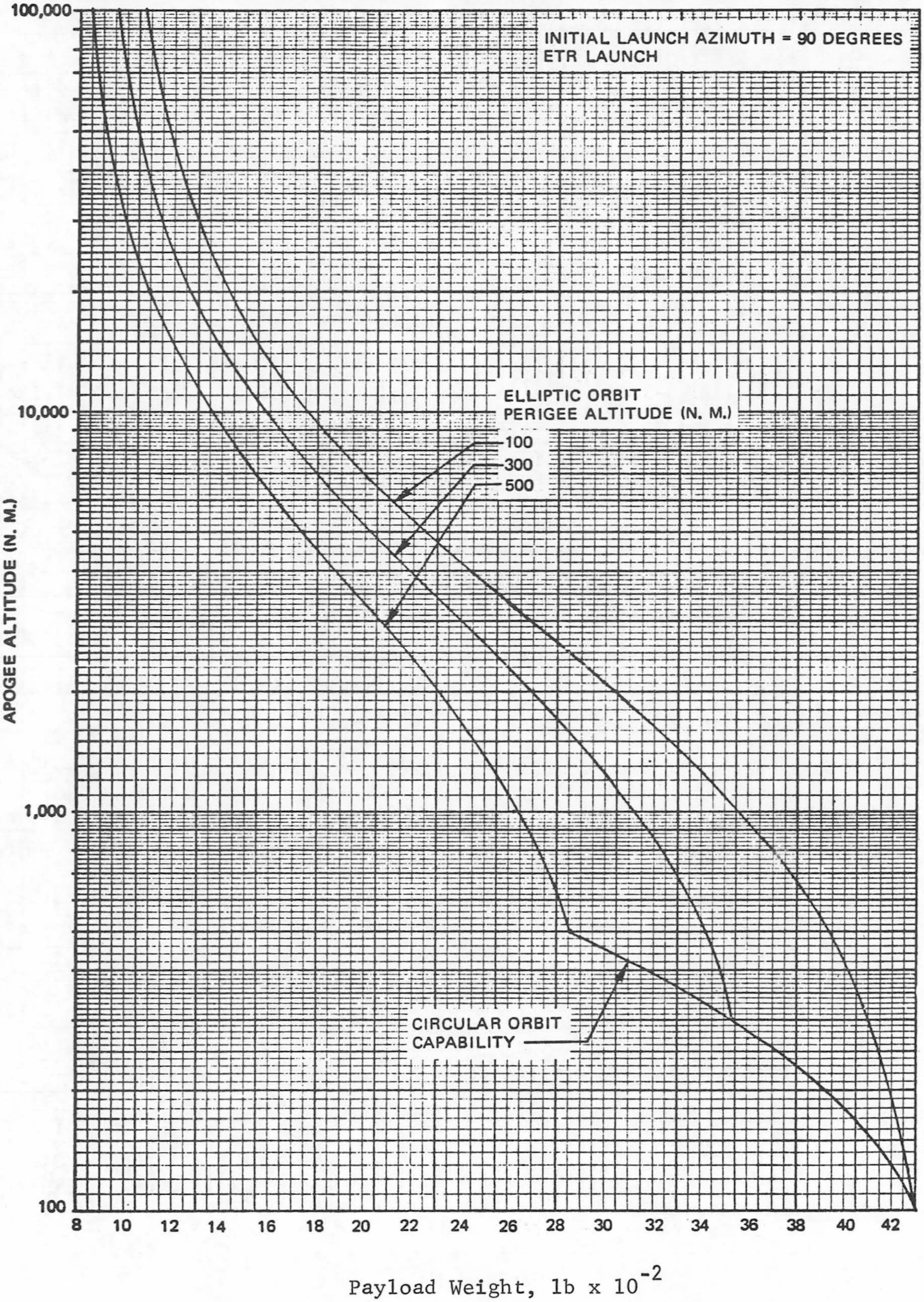
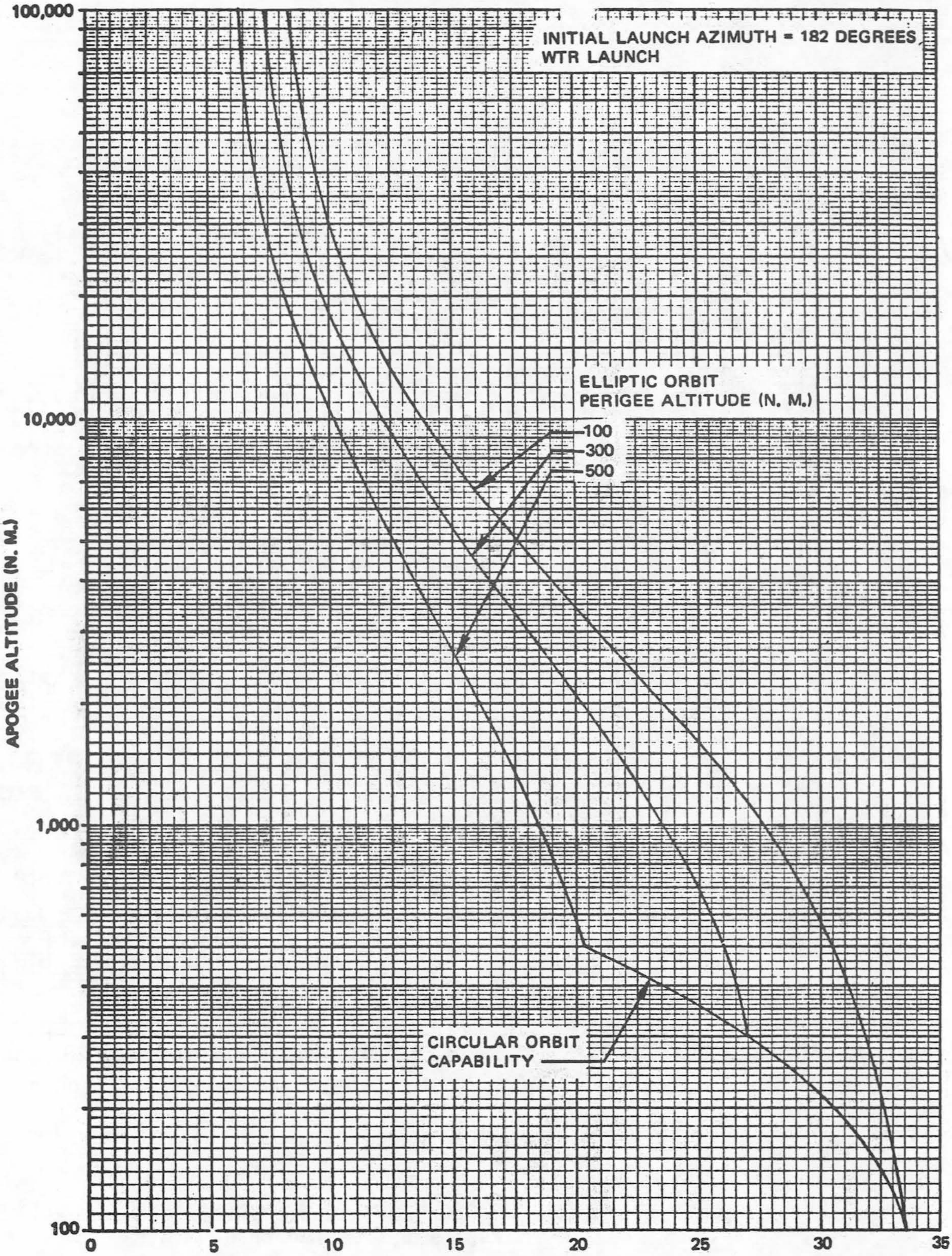


FIGURE IV-B-13. TAT(6)/DELTA(TSE)/TE364(2300) (604) ORBITAL CAPABILITY, 182° LAUNCH FROM WTR



Payload Weight, lb x 10^{-2}

FIGURE IV-B-14. TAT(9C)/DELTA(TSE)/TE364(2300) (904) ORBITAL CAPABILITY, 90° LAUNCH FROM ETR



Payload Weight, lb x 10^{-2}

FIGURE IV-B-15. TAT(9C)/DELTA(TSE)/TE364(2300) (904) ORBITAL CAPABILITY, 182° LAUNCH FROM WTR

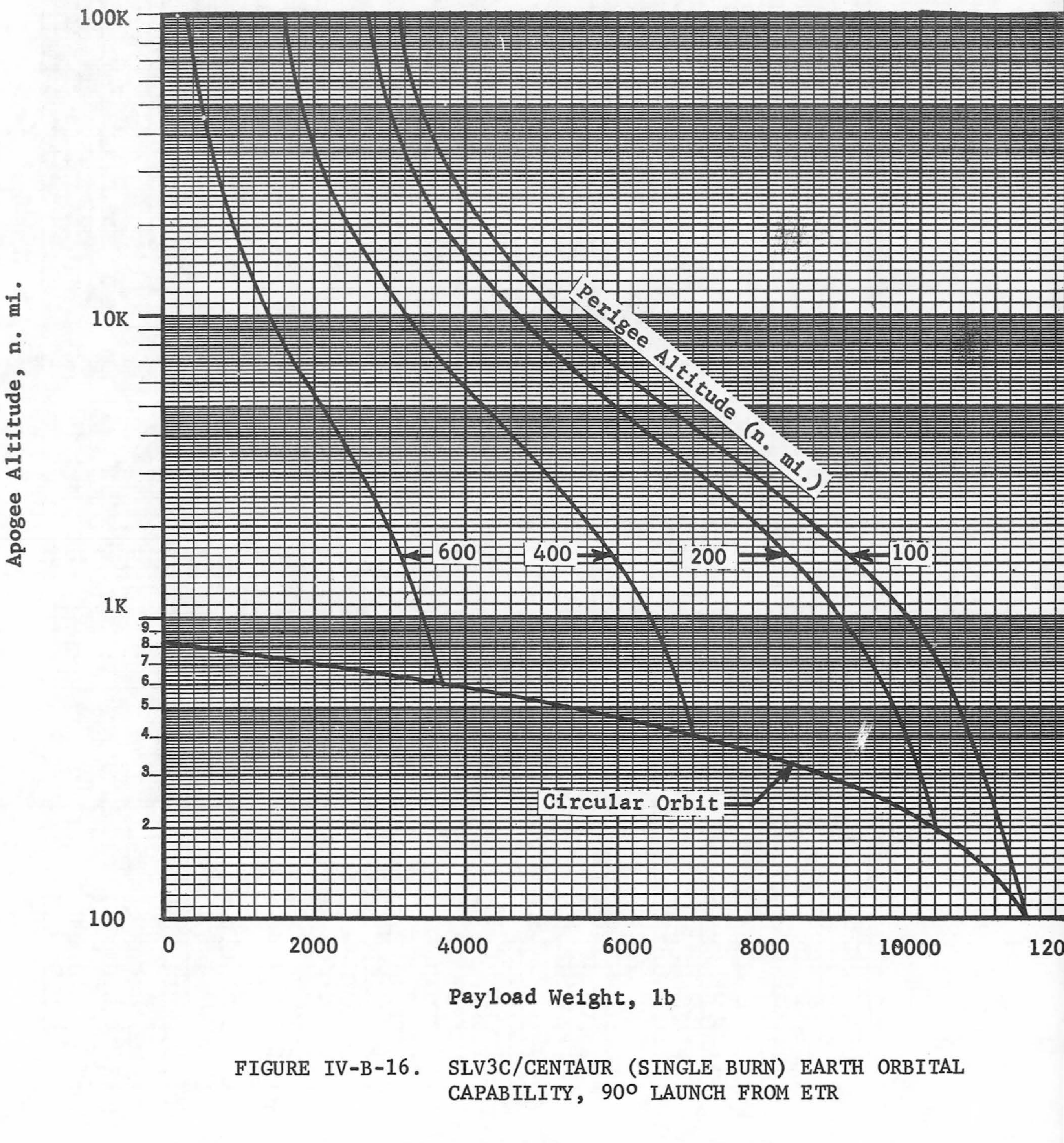


FIGURE IV-B-16. SLV3C/CENTAUR (SINGLE BURN) EARTH ORBITAL CAPABILITY, 90° LAUNCH FROM ETR

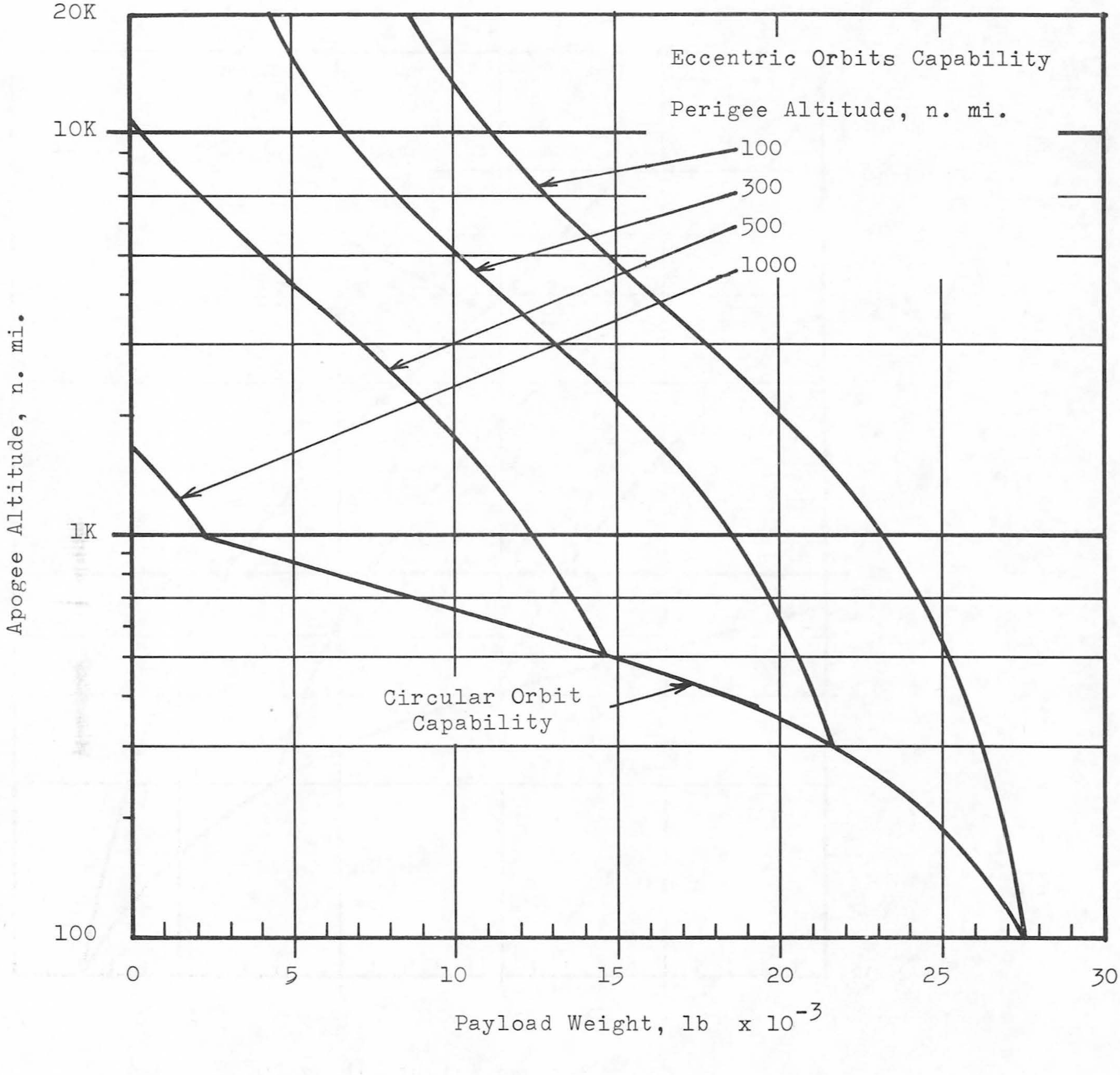


FIGURE IV-B-17. TITAN IIIC (OP. TR. ST. SINGLE BURN) EARTH ORBITAL CAPABILITY, 93° LAUNCH FROM ETR

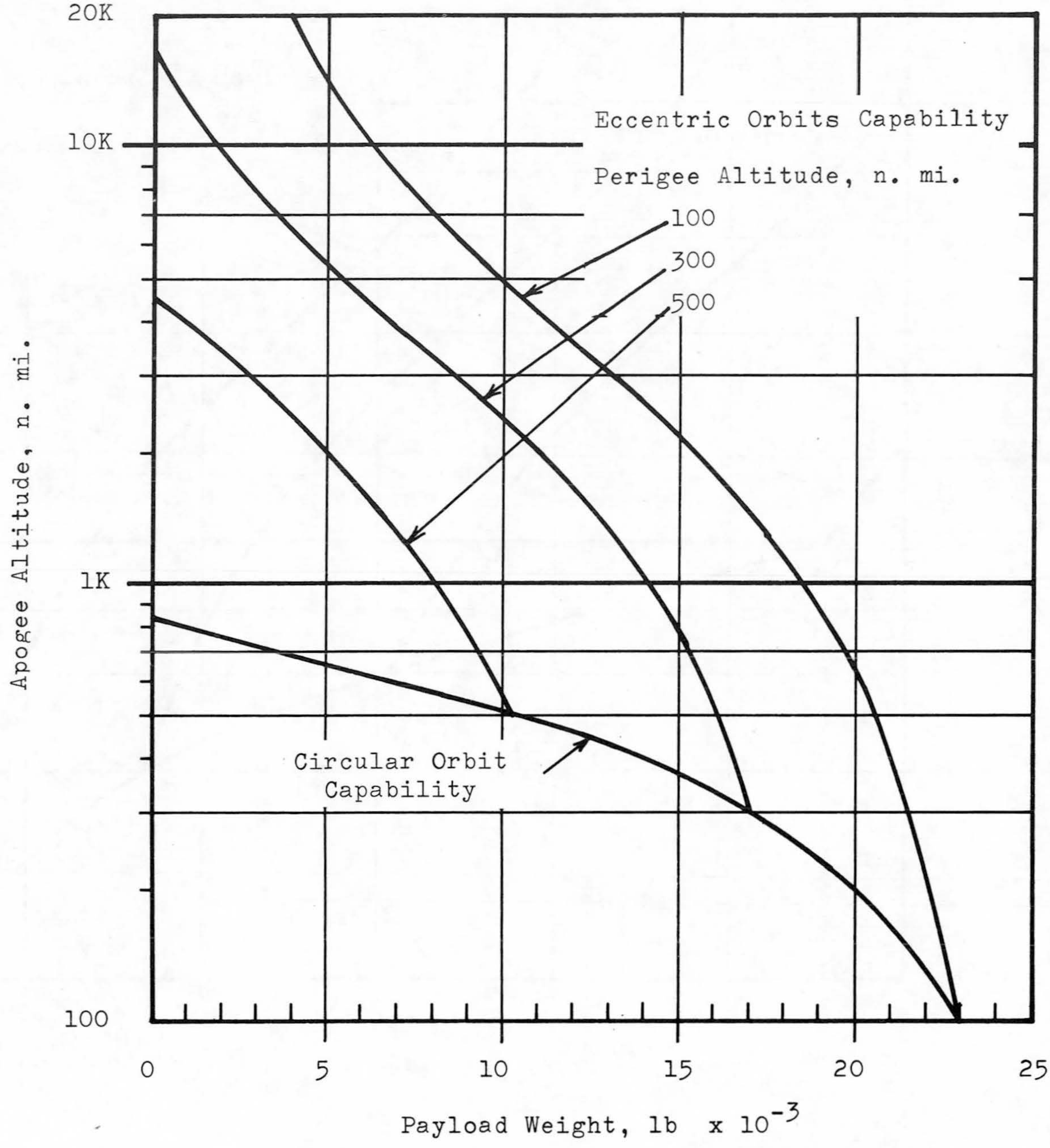


FIGURE IV-B-18. TITAN IIIC (OP. TR. ST. SINGLE BURN) EARTH ORBITAL CAPABILITY, 182° LAUNCH FROM WTR

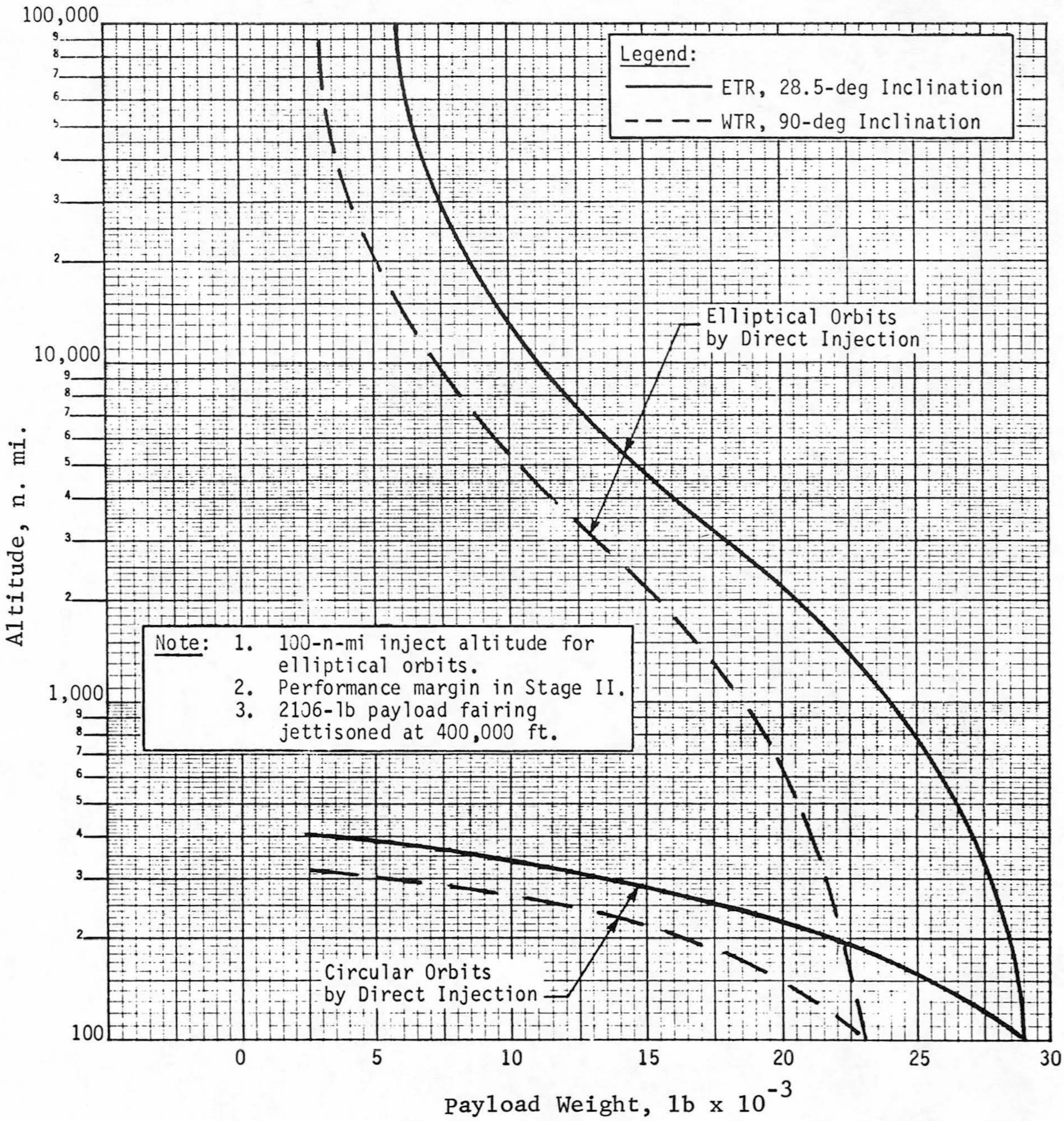


FIGURE IV-B-19. TITAN IIID, PAYLOAD WEIGHT VS ALTITUDE, ETR AND WTR

SECTION IV-C

SPACE SHUTTLE PERFORMANCE

SECTION IV-C

SPACE SHUTTLE PERFORMANCE

This subsection presents currently projected data for proposed space shuttle vehicles. The data are useful for making preliminary planning estimates for missions using space shuttles. The use of such vehicles should be considered in planning missions for the late-1970's and beyond.

Shuttle performance data presented in this subsection are based on information available up to the mid-point of Phase B of the NASA Space Shuttle Program and were obtained in cooperation with the NASA Space Shuttle Task Group Office. In Phase B, a fully reusable two-stage shuttle design is being emphasized including considerations of two shuttle orbiter configurations: a low-cross-range straight wing design and a high-cross-range delta wing design. The data presented in this subsection pertain to the straight wing orbiter. It is to have a nominal capability to deliver 25,000 pounds of payload to a 270 n. mi. circular orbit at 55 degrees inclination and return to Earth with an equivalent payload. Concurrent with Phase B, Phase A Review studies also are being conducted to investigate alternative design concepts. In view of the preliminary status of shuttle investigations, some revising and updating of design parameters and performance data are to be expected as the program progresses.

A desired goal of current space shuttle investigations is to achieve a vehicle with capabilities for serving a large number of users and a large number of mission and payload applications. While the inherent versatility of reusable space shuttle vehicles is to be recognized, the data presented in this subsection are restricted to two representative cases: (1) nominal performance capabilities and (2) maximum performance capabilities. Nominal performance data assume that airbreathing engines and cruise propellants are included in the shuttle orbiter. Maximum performance data assume off-loading the orbiter airbreathing engines and cruise propellants and trading the weight savings, approximately 22,000 lb, for additional payload.

Typically, mission requirements can be defined in terms of spacecraft or net payload weight and orbital specification or space destination. The first question to be considered in planning for the use of a shuttle is whether the mission can be accomplished by means of a shuttle alone or, if not, what type of shuttle upper stage will be required to complete the mission. In order that this question may be resolved quickly, the data in this subsection are presented in two parts -- first, data for the planning of shuttle-only missions and second, data for the planning of shuttle/shuttle upper stage missions.

Shuttle-Only Mode

It is planned that a shuttle would operate nominally at circular orbit altitudes up to 270 n. mi. or, possibly, up to about 340 n. mi. A fully reusable shuttle would have an all-azimuth launch capability so that any orbit inclination between 28.5* and about 100 degrees could be achieved via direct ascent launch. If the user's mission requirements fall within these general limits then the shuttle-only mode should be considered first.

Having defined a mission in terms of spacecraft or net payload weight and orbital specification or space destination, a potential shuttle user would normally proceed to estimate the additional weight of shuttle adapters for handling and mounting his payload module in the shuttle cargo bay. Figure IV-C-1 shows shuttle gross payload weight and shuttle adapter weight as functions of net payload weight. Gross payload is the sum of the shuttle adapter weight and the net payload weight. The net shuttle payload consists of the user's payload module including the user's payload (spacecraft or cargo) together with whatever adapters, equipment, and supplies that may be necessary to ensure its protection and/or proper functioning.

Shuttle adapters would include items such as handling rings, shuttle deployment mechanism adapters and fittings, and cargo bay mounting attachments. The external dimensions of the payload module including adapters must be equal to or less than 15 ft in diameter and 60 ft in length. The deployment of payload modules would be provided by the shuttle. Standardized deployment mechanisms with payload attachment points would be integral components of the shuttle and would not be chargable to payload weight or volume. The operation of the deployment mechanisms would be such that proper clearances would be maintained during the deployment of payload modules.

* If a direct ascent from ETR is assumed then the lowest possible inclination that could be achieved by the shuttle alone would be 28.5 degrees.

The problem of shuttle/payload integration is being studied, however, it is too early for definitive results.* Until such results become available, the potential shuttle user probably should plan for the case of minimal interface with the shuttle such as, for example, having only mechanical connections for mounting, electrical connections, and, possibly, guidance update information.

Moreover, the individual mission planner may wish to consider the matter of shuttle adapter weight in greater detail and, rather than using Figure IV-C-1, develop a weight estimate for an adapter specifically suited to his requirements. The gross payload weight obtained from Figure IV-C-1 or by some other estimating technique can be compared directly with shuttle performance capabilities to determine whether the mission can be performed using the shuttle alone.

Figure IV-C-2 and IV-C-3 shows shuttle gross payload capabilities for circular orbits as a function of inclination and altitude. Figure IV-C-2 is representative of the nominal performance mode (with orbiter airbreathing cruise engines and propellants). Figure IV-C-3 is representative of the maximum performance mode (without orbiter cruise engines and associated propellants). Allowances are made for rendezvous and docking maneuvers and for contingencies. Also, abort criteria are taken into consideration.

Abort requirements generally involve some sacrifice of performance in the interest of crew safety. For the shuttle orbiter, it is planned that the orbital maneuvering system (OMS) will serve as a back-up for the orbiter injection propulsion system. In the event of an abort situation (e.g., one injection engine out) during the last stages of the launch phase, the OMS could be required to provide velocity impulses totaling as much as 1500 ft/sec. Therefore, to satisfy abort criteria, the OMS must have the capability of providing a ΔV approximately equivalent to the nominal value of "on-orbit ΔV" for every shuttle flight.

In general, abort criteria will preclude the possibilities of off-loading substantial amounts of OMS propellant. The nominal value of "on-orbit ΔV" (1500 ft/sec) is sufficient, if the launch proceeds nominally, to perform a mission to a 270 n. mi. circular orbit and return. As a result, the shuttle payload capability would not change appreciably as a function of altitude for missions between 100 and 270 n. mi.

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* An "Interface Control Document" is in preparation.

Small incremental trade-offs may be possible depending on the abort criteria that are finally adopted. These may depend on the configuration for individual missions. If the criteria are such that a ΔV less than 1500 ft/sec is acceptable for a given mission, then it would be possible to increase the payload capability for that mission. The payload increment may be estimated by allowing 20 pounds of additional payload for each foot per second of the ΔV "decrement". The following considerations serve as guidelines for the estimation of shuttle payload increments:

(1) The nominal value of "on-orbit ΔV" includes an increment of about 360 ft/sec required for deorbit from a 270 n. mi. circular orbit. For missions to lesser altitudes that require a smaller deorbit impulse, the payload may be increased by an amount equal to 20 pounds of additional payload for each foot-per-second difference between the deorbit impulse at 270 n. mi. and that required at a lesser altitude. The procedure for estimating the ΔV "decrement" is as follows: (a) assume that the reentry altitude is 65 n. mi. and determine the ΔV required to transfer to that altitude from a given mission altitude (use Figure II-31c), (b) subtract this value of ΔV from 360 ft/sec, (c) multiply the difference by 20 lb per ft/sec, and (d) add the resulting payload increment to the payload value obtained from the uppermost curve on Figure IV-C-2 or IV-C-3. This procedure is not applicable for circular orbit altitudes greater than 270 n. mi.

(2) The nominal value of "on-orbit ΔV" includes an allotment of about 100 ft/sec for rendezvous and docking. If such a maneuver is not required then all or part of the rendezvous and docking allotment could be traded for additional payload. Allow 20 lb of additional payload for each foot per second of the ΔV decrement. Add the payload increment to the value obtained from Figure IV-C-2 or Figure IV-C-3. This procedure can be used for any mission altitude.

The mission planner should exercise caution against indiscriminate application of the preceding trade-off possibilities. Primarily, they are included here to demonstrate that the shuttle payload capabilities indicated on Figures IV-C-2 and IV-C-3 may be increased by substantial amounts, up to several thousands of pounds, under certain conditions. The extent of possible increases, however, depends on a number of factors. Questions on these matters may be addressed to one of the persons listed in the front of this document.

While the nominal value of on-orbit ΔV is to be 1500 ft/sec, it is planned that the OMS propellant tanks will be sized to hold enough propellant to provide a total ΔV capability as large as 2000 ft/sec. In this case, the shuttle could perform missions to circular orbits as high as about 340 n. mi. without the necessity of adding auxillary propellant tanks. Shuttle payload capability for circular orbit altitudes up to 340 n. mi. are shown on Figures IV-C-2 and IV-C-3. Somewhat higher altitudes may be achieved via elliptical orbits.

Figure IV-C-4 can be used together with Figures IV-C-2 and IV-C-3 for the estimation of shuttle payload capabilities for elliptical orbits assuming that a shuttle can transport equivalent payloads to orbits of equal energy. To estimate the shuttle payload capability for a given elliptical orbit; first, determine the circular orbit equivalent to the given elliptical orbit from Figure IV-C-4 and second, determine the payload for that circular orbit from Figure IV-C-2 or IV-C-3. The reverse procedure can be used to determine elliptical orbit parameters if the shuttle payload is given.

The following examples are provided to illustrate the use of Figures IV-C-1 through IV-C-4.

Example 1

Problem: Determine the net cargo weight that can be transported by a shuttle for a logistics mission to a 270 n. mi. circular orbit at 55 degrees inclination. Assume nominal performance mode.

Solution: From Figure IV-C-2, the gross shuttle payload capability for 55 degrees and 270 n. mi. altitude is 25,000 lb. The net shuttle payload capability for a gross payload of 25,000 lb, from Figure IV-C-1, is 20,000 lb.

Example 2

Problem: Deliver a 30,000 lb spacecraft to a 100 x 400 n. mi. orbit at 28.5 degrees inclination.

Solution: The gross shuttle payload for a net payload of 30,000 lb, from Figure IV-C-1 is 36,500 lb. From Figure IV-C-4, the circular orbit equivalent to a 100 x 400 n. mi. elliptical orbit is 300 n. mi. The shuttle gross payload for the nominal performance mode, a value of 33,000 lb is obtained from Figure IV-C-2, is not adequate for the mission. The gross payload capability is about 56,000 lb for the maximum performance mode (Figure IV-C-3). Therefore, the shuttle could perform the mission in the maximum performance mode.

Shuttle/Shuttle Upper Stage Mode

If mission requirements exceed shuttle-only capabilities, or if the mission destination lies outside the shuttle operating regime, then a shuttle upper stage would be required to complete the mission. In this case, the net shuttle payload would consist of a "package" that includes the users payload (spacecraft or cargo), a spacecraft/upper stage adapter, a shuttle upper stage, and whatever payload service equipment that may be required. As defined on page IV-C-2, the gross shuttle payload would be the sum of the net shuttle payload weight and the shuttle adapter weight.

For shuttle/shuttle upper stage missions (both Earth orbital and Earth escape), it has been assumed that the shuttle orbiter would be injected into a 50 x 100 n. mi. elliptical orbit (Reference Injection Orbit); that the orbiter would then transfer to an appropriate parking orbit; and, finally, that the shuttle upper stage would deliver the users payload from the parking orbit to the final orbit or space destination. All maneuvers would be coplanar if the final orbit inclination is greater than 28.5 degrees. Otherwise, the shuttle parking orbit inclination would be 28.5 degrees and the required plane change would be accomplished by means of the shuttle upper stage.

Table IV-C-1 lists the weights and characteristics of several existing, planned, and proposed propulsion stages that are among the possible candidates for shuttle upper stage applications. Additional information on solid-propellant upper stages is presented in subsection IV-D, "Velocity Packages".

TABLE IV-C-1. WEIGHTS AND CHARACTERISTICS OF REPRESENTATIVE SHUTTLE UPPER STAGES

Designation	Length(ft)	Diameter(ft)	Stage Weight (lb)[1]	Spacecraft/Stage Adapter Weight(lb)
BII(1440)	5.2	5.2	1,787	20[2]
BII(2300)	6.3	5.2	2,717	20[3]
BIIA(1440) (two stage)	7.5	5.2	2,467	20[4]
Tandem BII(1440) (two stage)	10.2	5.2	3,670	20[4]
Tandem BII(2300) (two stage)	12.4	5.2	5,530	20[5]
DELTA	9.6	4.8	12,074	50 - 80
AGENA	20.2	5.0	14,930	60 - 100
TRANSTAGE	15.	10	27,477	230
CENTAUR	30	10	34,436	116

(1) Stage weights do not include adapter weights.
(2) For payloads over 500 lb, the adapter weight must be increased by 10% of the excess payload over 500 lb.
(3) For payloads over 1000 lb, the adapter weight must be increased by 10% of the excess payload over 1000 lb.
(4) For payloads over 500 lb, the adapter/intra-stage structure must be increased by 12.5% of the excess payload over 500 lb.
(5) For payloads over 1000 lb, the adapter/intra-stage structure must be increased by 12.5% of the excess payload over 1000 lb.

Two procedures are presented in this subsection for the estimation of shuttle/shuttle upper stage performance. The first procedure is applicable to cases where a restartable liquid-propellant shuttle upper stage is used for Earth orbital or Earth escape missions or for cases where a solid propellant stage is used for Earth escape missions. In these cases, the procedure is identical to that for conventional (expendable) launch vehicles outlined on page I-1 of this document. The second procedure, discussed on page IV-C-9, is for cases where solid-propellant upper stages, comprised of one or more single-burn motors, are used for Earth orbital missions.

Figure IV-C-5a and IV-C-5b show the spacecraft payloads that could be delivered by a shuttle in combination with various solid and liquid-propellant upper stages as a function characteristic mission velocity. These performance curves represent shuttle/shuttl upper stage performance capabilities for a due east launch from ETR corresponding to an orbit inclination of 28.5 degrees.

The curves on Figure IV-C-5b show that Agena and Delta upper stages with spacecraft payloads up to about 15,500 and 18,300 lb, respectively, can be transported to orbit by means of a shuttle in the nominal performance mode. For larger payloads, the upper stage propellants must either be off-loaded (left-hand branch of the curves) or the shuttle must be operated in the maximum performance mode (right-hand branch). The Transtage and Centaur with payloads in excess of a few thousand pounds can only be transported to orbit in the maximum performance mode. For payloads exceeding 20,000 and 14,000 lb, respectively, the Transtage and Centaur must be off-loaded. The right-hand branch of the Centaur curve shows the performance capability if Centaur could be refueled in orbit.

Using the data of Figures IV-C-5a and IV-C-5b and Sections II and III, the performance of the shuttle/upper stage combinations can be estimated for a variety of Earth orbital and Earth escape missions. Example 3 illustrates the procedure for Earth orbital missions using a liquid-propellant shuttle upper stage. Example 4 illustrates the procedure for Earth escape missions using either liquid or solid-propellant shuttle upper stages.

Example 3

Problem: Deliver a 10,000 lb spacecraft to synchronous equatorial orbit by means of a shuttle/upper stage launch from the ETR.

Solution: From Figure II-28, the required characteristic mission velocity is 39,600 ft/sec. From Figure IV-C-5b, this required characteristic velocity could be provided at the required payload using the shuttle in the maximum performance mode plus the Centaur.

Example 4

Problem: Deliver a 1000 lb spacecraft to Venus in 1980.

Solution: From Figure II-10, the required characteristic velocity for Venus in 1980 (assuming a 30-day opportunity width) is 37,550 ft/sec. From Figure IV-C-5a and 5b, this mission can be flown using the space shuttle and the Delta, Agena, Transtage, Centaur or Tandem Burner II(2300).

The second procedure to be discussed is for the estimation of Earth orbital missions where solid-propellant shuttle upper stages are used. All of the solid-propellant stages shown in Table IV-C-1 incorporate one or more single-burn motors. When such stages are used to perform Earth orbital missions it is not only necessary to select a stage which in combination with the shuttle gives the proper total velocity capability (V_C), but also one that gives the proper sequence of velocity increments (ΔV's). For single-burn motors, this is equivalent to requiring the proper ΔV per motor. Figure IV-C-6 gives the incremental ΔV's per motor.

In calculating required ΔV sequences for Earth orbital missions using solid-propellant upper stages, the use of Figure II-31c in Section II is recommended for the estimation of incremental velocity impulses required to perform various coplanar maneuvers. Launch azimuth velocity corrections for inclinations other than 28.5 degrees can be obtained from Figure III-1a in Section III. Figure II-31a together with Figure II-32b may be used for the estimation of velocity impulses required for plane change maneuvers. Using these figures, a required sequence of velocity impulse corresponding to a given mission may be calculated. Using Figure IV-C-6, a shuttle/solid-propellant upper stage combination may be selected that provides adequate payload capability to perform the Earth orbital mission. Example 5 illustrates this procedure.

Example 5

Problem: Deliver a 6,500 lb spacecraft to a 800 n. mi. sun-synchronous circular orbit using a two stage solid-propellant shuttle upper stage.

Solution: From Figure II-33, the required orbital inclination for 800 n. mi Sun-synchronous orbit is 102 degrees. Using the all-launch azim capability of the shuttle, assume a shuttle orbiter circular parking orbit of 100 n. mi. altitude and 102 degrees inclination From Figure II-31c the required ΔV at 100 n. mi. to transfer to 800 n. mi. is 1120 ft/sec. From the same Figure, the required apogee ΔV to circularize from a 100 x 800 n. mi. orbit to a 800 n. mi. circular orbit is 1080 ft/sec.

From Figure IV-C-6, for a payload of 6,500 lb, the first stages of the Tandem Burner II(2300) and the Burner IIA will deliver the required initial ΔV of 1120 ft/sec, but the first stage of the Tandem BII(1440) will not. Similarly, the second stages of the Tandem BII(2300) and Tandem BII(1440) will deliver the required second ΔV of 1080 ft/sec, but the second sta of the Burner IIA will not. Therefore, only the Tandem BII(2300) can meet both ΔV requirements.

To check on the ability of the shuttle to deliver the total payload package, compute the shuttle gross payload. The spacecraft weight is 6,500 lb, the spacecraft/upper stage adapter weight (Table IV-C-1) is 20 + .125 (6500-1000) = 1145 lb and the Tandem Burner II(2300) weight is 5530 lb. Therefore, the net shuttle payload is 13,175 lb. The gross shuttle payload (from Figure IV-C-1) would be about 17,000 lb. From Figures IV-C-2 and IV-C-3, this payload is within the shuttle gross payload capability for a 102° inclination using the maximum orbiter performance mode.

Alternate Solution: In the previous solution, the Tandem BII(1440) was eliminated because the first stage velocity impulse was less, but, only slightly less, than that required. It would be possible to use the Tandem BII(1440) by tailoring the mission profile to suit the capabilities of that stage. In this case, the selection of a shuttle parking orbit higher than 100 n. mi. would be appropriate. As discussed on page IV-C-3, a circular parking orbit as high as 270 n. mi. could be assumed and the gross shuttle payload capability would not change from that at 100 n. mi. Assuming a 270 n. mi. circular parking orbit at 102 degrees inclination and using the same procedure as in the previous solution, the required velocity impulses for transfer and circularization would be 820 ft/sec and 795 ft/sec, respectively. In this case, the Tandem BII(1440) could provide both of the required impulses and could be used to perform the mission. Because of the lower stage and adapter weights, the net payload would be 10,940 lb as compared with 13, 175 lb for the Tandem BII(2300) from a 100 n. mi. parking orbit.

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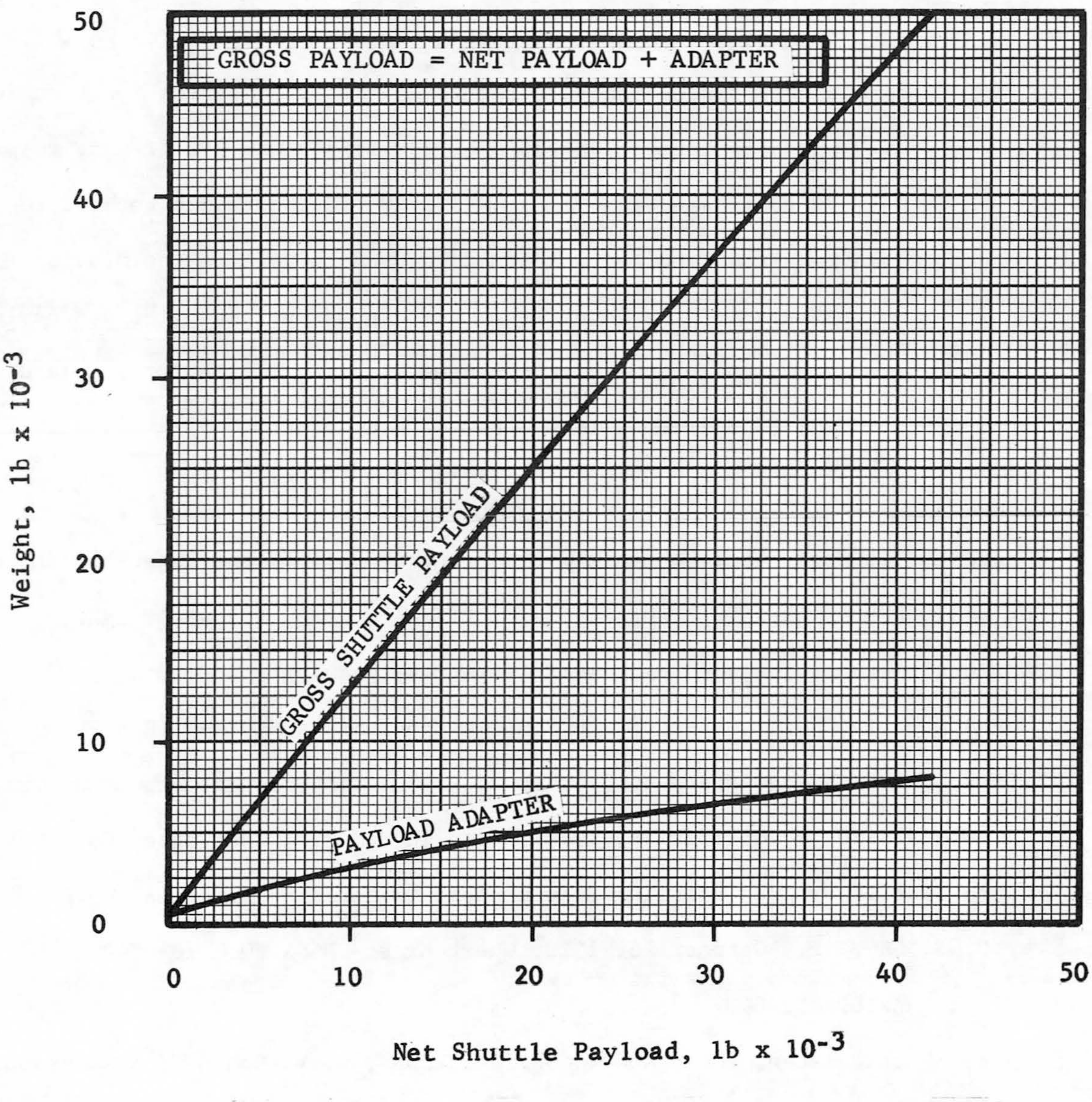

FIGURE IV-C-1. SHUTTLE PAYLOAD AND ADAPTER WEIGHTS

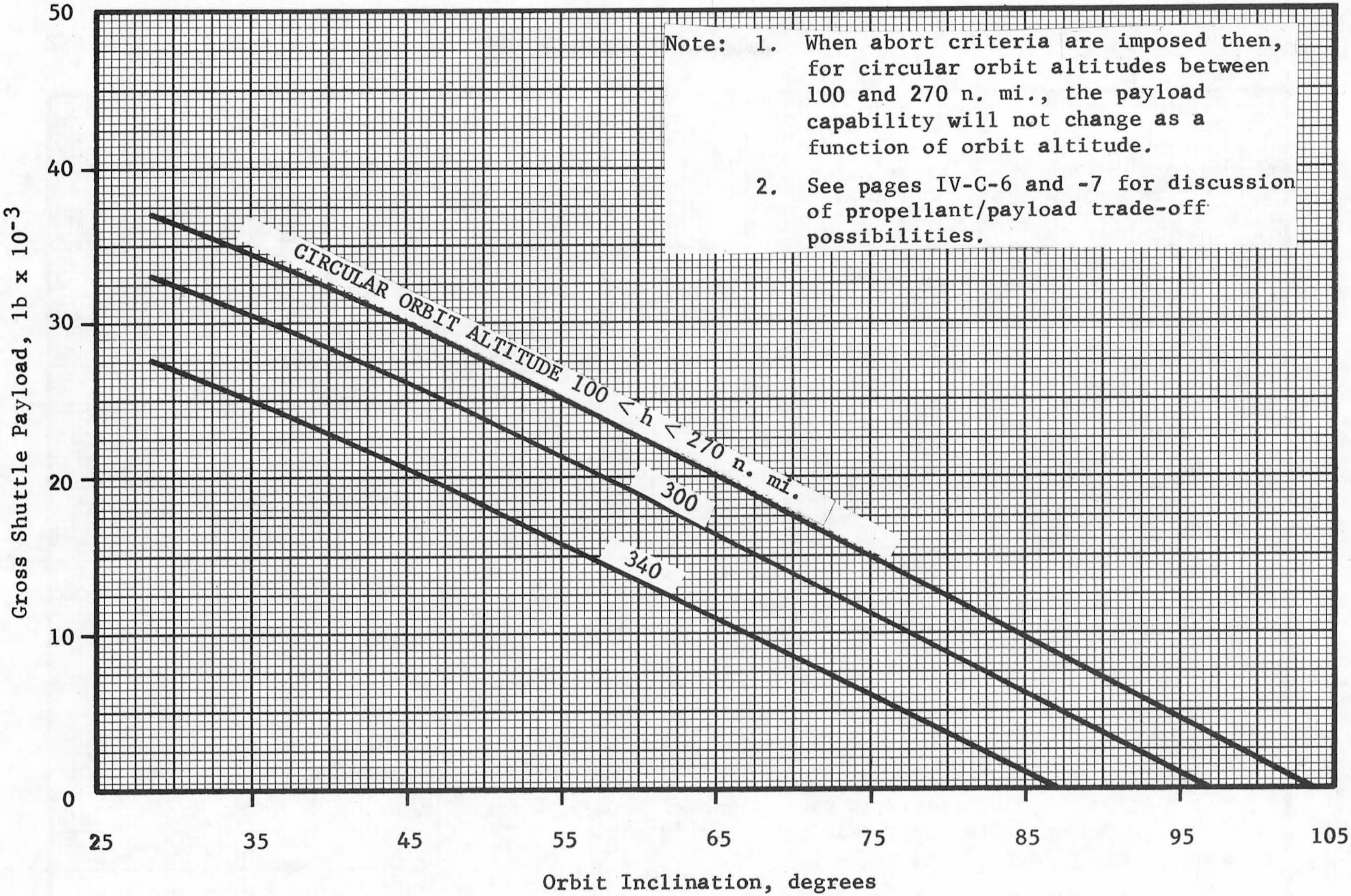

FIGURE IV-C-2. NOMINAL PERFORMANCE SPACE SHUTTLE PAYLOAD CAPABILITY
(Orbiter Air-Breathing Engines In)

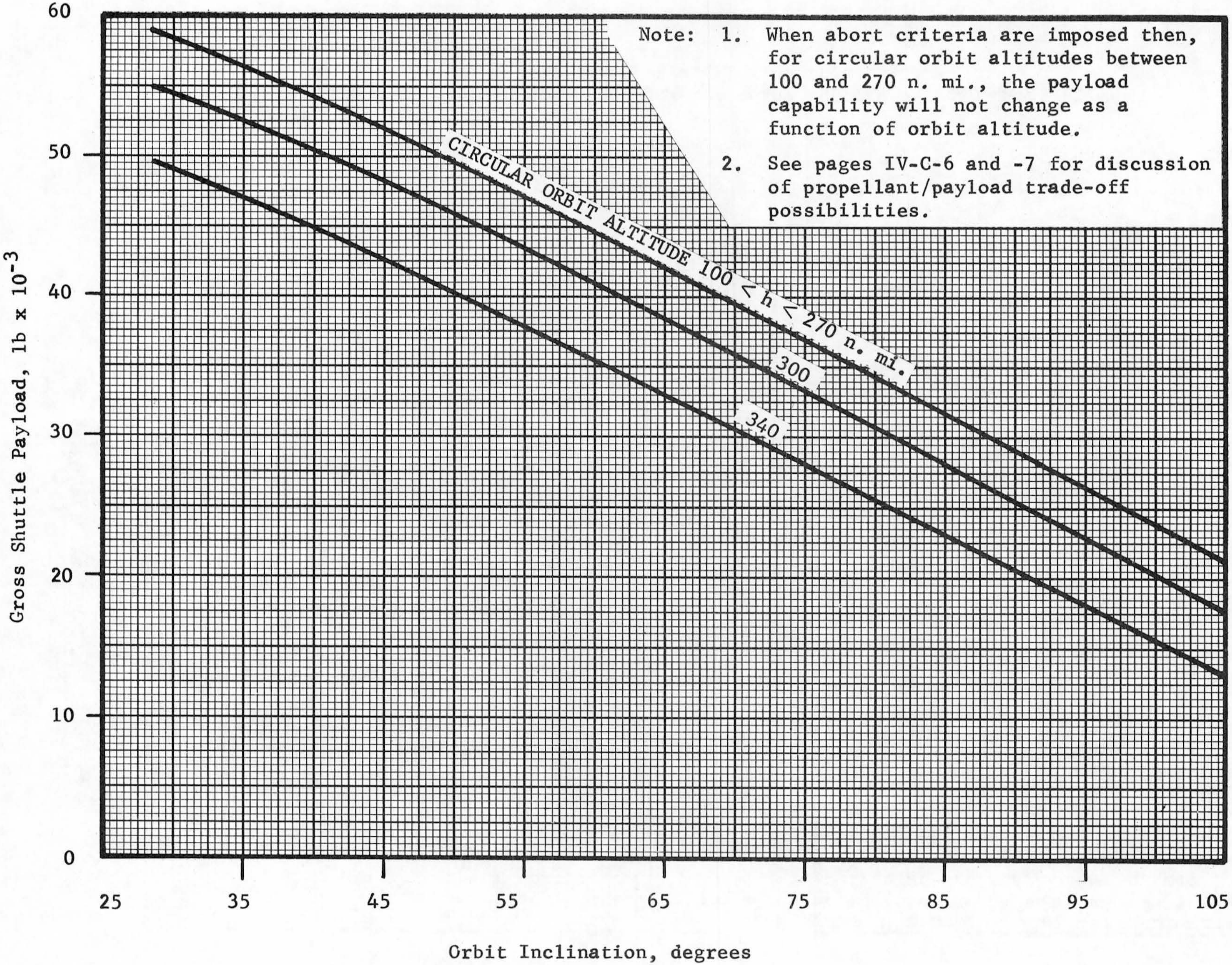

FIGURE IV-C-3. MAXIMUM PERFORMANCE SPACE SHUTTLE PAYLOAD CAPABILITY

(Orbiter Air Breathing Engines Out)

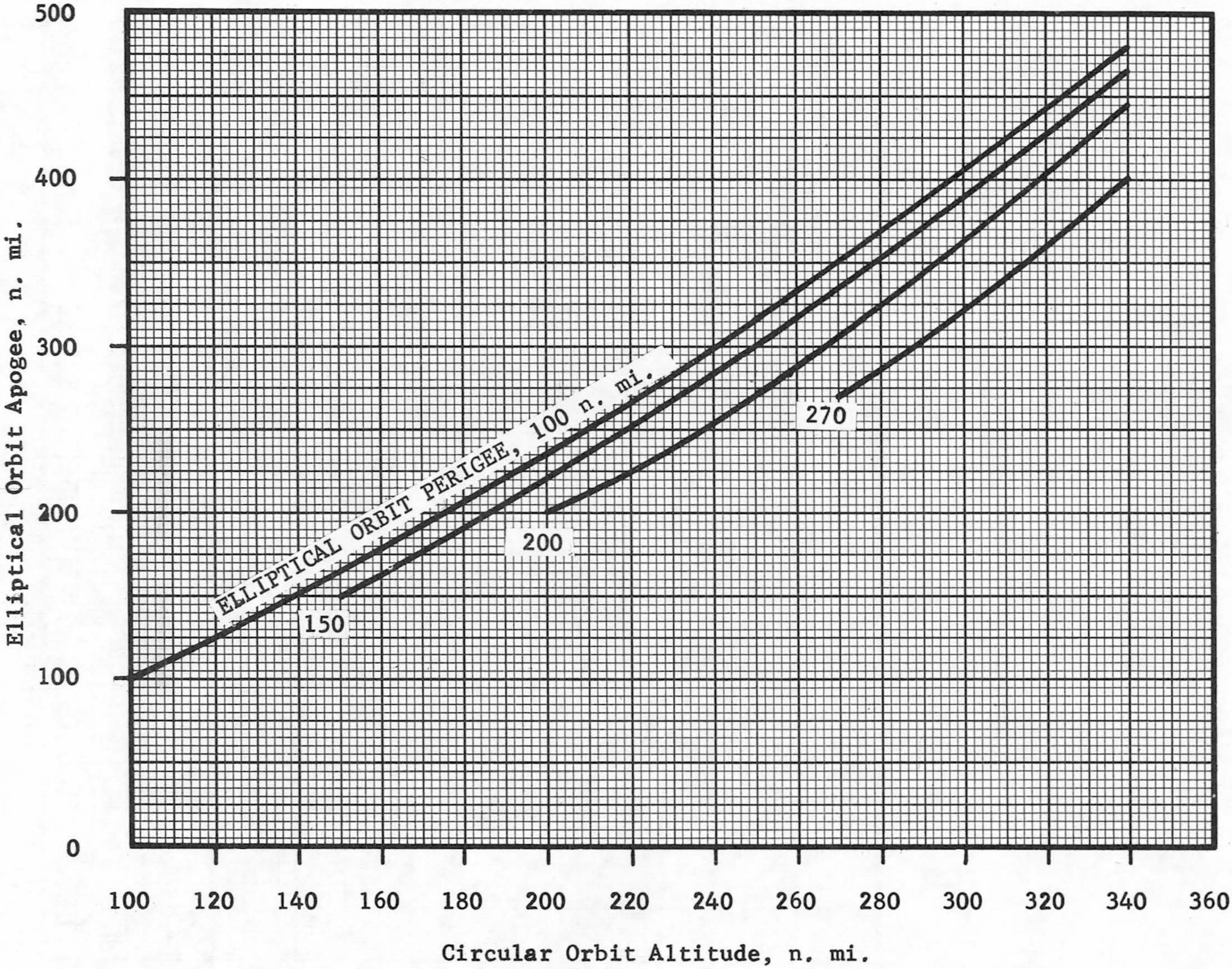

FIGURE IV-C-4. EQUIVALENT ENERGY CIRCULAR AND ELLIPTICAL ORBITS FOR SPACE SHUTTLE MISSIONS

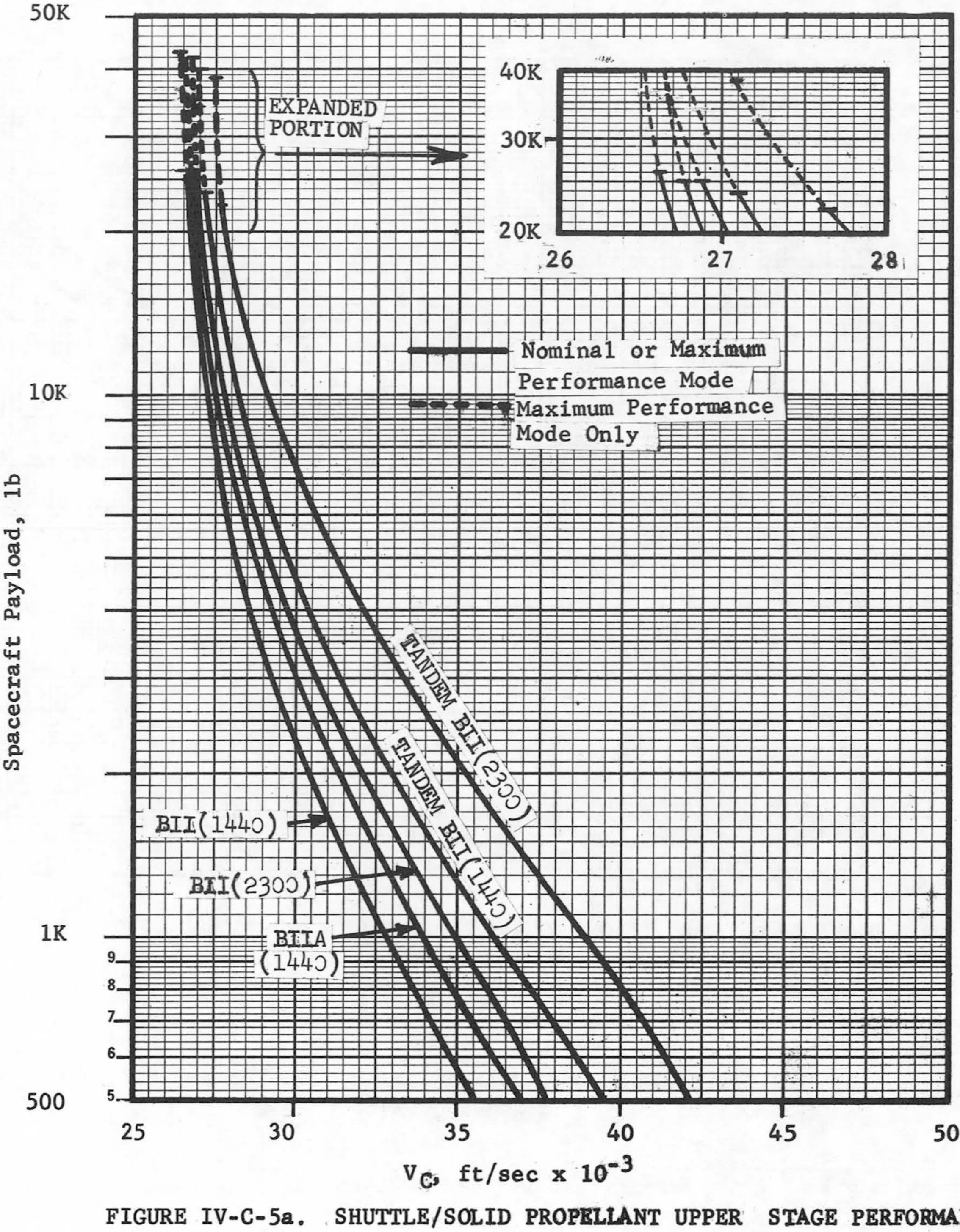

FIGURE IV-C-5a. SHUTTLE/SOLID PROPELLANT UPPER STAGE PERFORMANCE

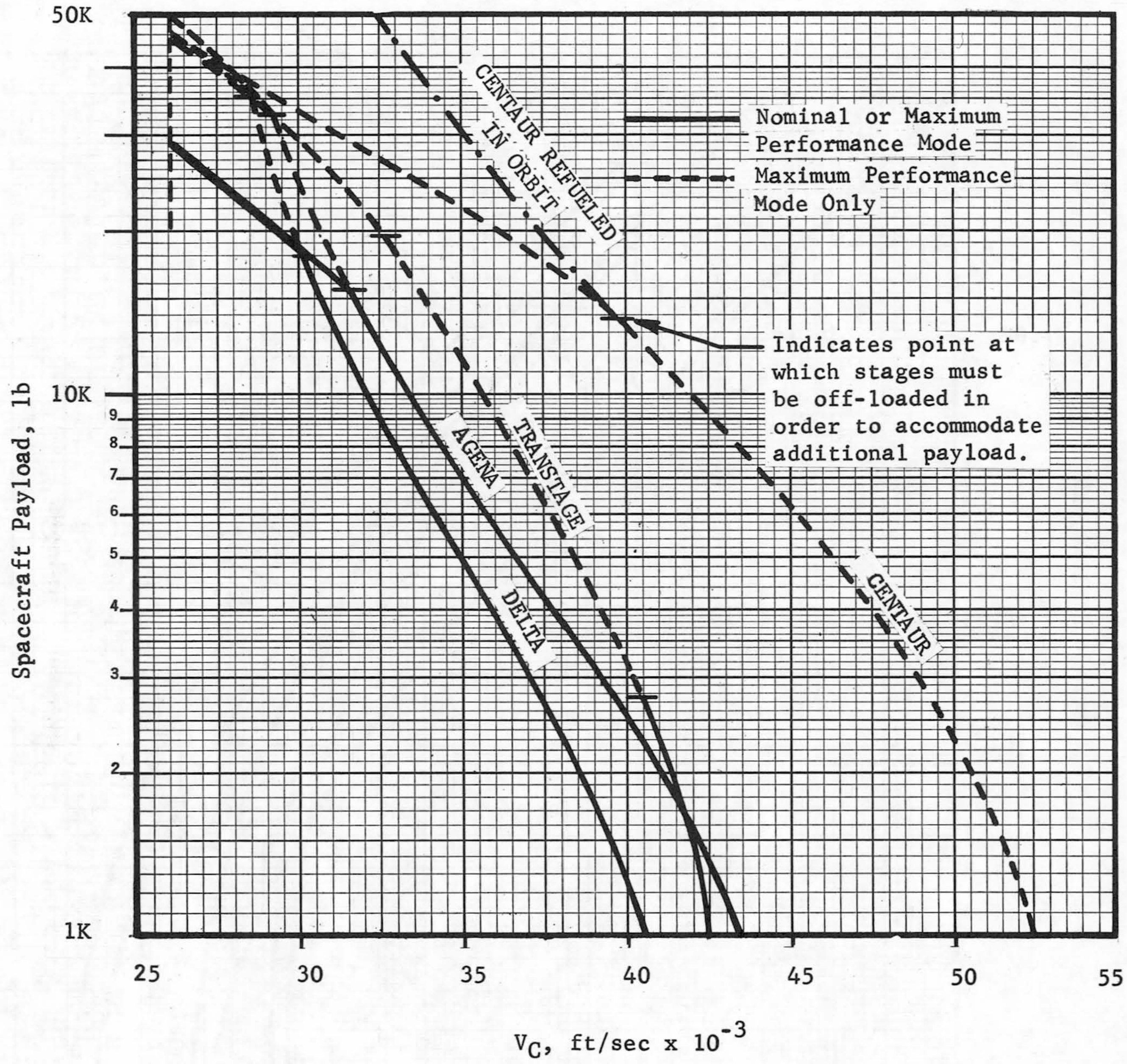

FIGURE IV-C-5b. SHUTTLE/LIQUID PROPELLANT UPPER STAGE PERFORMANCE

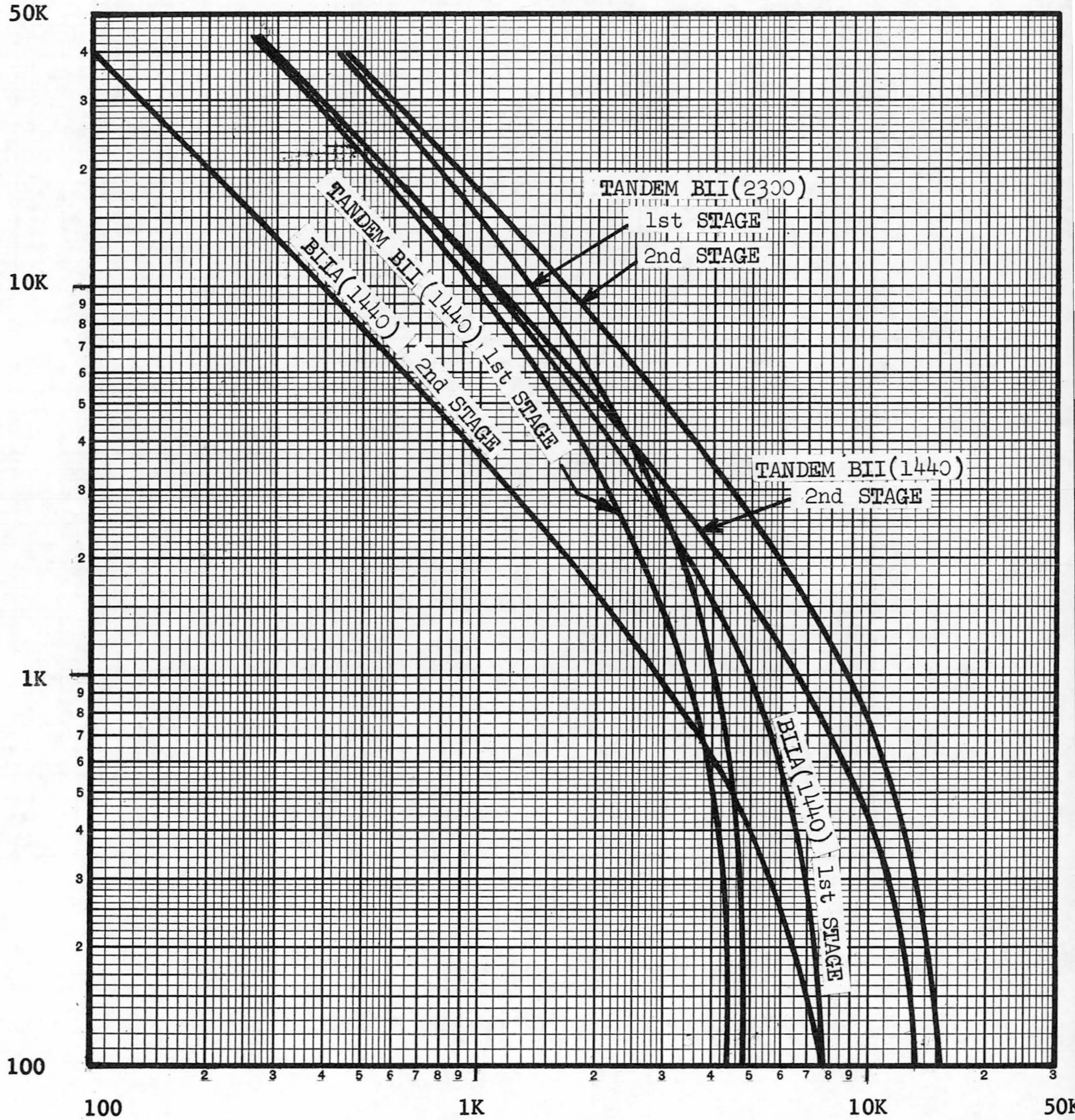

FIGURE IV-C-6. STAGE VELOCITY IMPULSES FOR SOLID PROPELLANT MOTORS

SECTION IV-D

VELOCITY PACKAGES

SECTION IV-D

VELOCITY PACKAGES

In certain circumstances it may be possible to obtain higher characteristic velocities than is provided by the vehicles shown in Section IV-A by including a small solid propellant motor as an additional stage. Such motors with their associated control systems and structure are called velocity packages. Examples of velocity packages in various states of development are the Burner II(BII) series, Fire, Fire/FW4 (a 2-stage velocity package), and the OV1 propulsion module (FW4 motor). The use of a velocity package is indicated, in general, only for payloads below the knee of the vehicle payload-characteristic velocity curve.

Table IV-D-1 gives characteristic data for several operational and proposed velocity packages. Figure IV-D-1 presents velocity increments as a function of payload for the velocity packages. To use the information presented in this section for estimating the performance of vehicles with velocity packages, the following steps should be followed:

(1) Add the weight of the proposed payload to the total weight listed in Table IV-D-1 for the velocity package. If the payload weight exceeds the limit for which stage structure is designed, it will be necessary to account for the additional weight of a strengthened structure. In this case, estimate the additional structural weight according to the procedures prescribed in the footnotes on Table IV-D-1 and include the additional weight in the summation of payload and velocity package weight.

(2) Find the characteristic velocity of the proposed base vehicle from the curves in Section IV-A for a payload equal to the total weight found in (1).

(3) Add to this characteristic velocity the velocity increment (ΔV) of the velocity package obtained from Figure IV-D-1 for the proposed payload weight. This gives total characteristic velocity available for the proposed payload using the velocity package.

The user is cautioned that this procedure is an approximation. Variations in shroud weights and interstage adapters resulting from the use of velocity packages may cause the actual vehicle performance to differ from that estimated by this generalized procedure. Also, the characteristic velocity obtained by this procedure cannot generally be used to derive Earth orbital capabilities. As discussed on page IV-C-9, the use of velocity packages consisting of one or more single-burn solid-propellant motors for Earth orbital missions requires the proper ΔV increments as well as the proper total characteristic velocity.

The TE364(1440), TE364(2300), Fire and Fire/FW4 velocity packages are spin stabilized, and this must be considered in the design of the payload. The BII series and the OV1(FW4) are attitude stabilized. The Tandem BII(1440) velocity package consists of two BII(1440) stages in tandem with the guidance removed from the lower stage. Correspondingly, the Tandem BII(2300) consists of two BII(2300) stages. The BIIA velocity packages also consist of two stages mounted in tandem. The lower stage is either a BII(1440) or a BII(2300) with the guidance removed. The upper stage, in both cases, is powered by a TE-M-442 motor with a propellant loading of 535 lb. The standard Burner II guidance and control system is installed in the upper stage. The proposed Be(beryllium) BII is a single stage velocity package that would use a propellant containing beryllium. Both one-burn and two-burn versions of the stage might be built.

TABLE IV-D-1. VELOCITY PACKAGE CHARACTERISTICS

Velocity Package	Nr of Stages	Length/Diameter (ft)	Stage Weight (lb)[a]	Adapter Weight (lb)[b]	Status
TE364(1440)	1	4.5/3.2	1,574	20[1]	Operational
BII(1440)	1	5.2/5.2	1,787	20[1]	Operational
BIIA(1440)	2	7.5/5.2	2,467	20[2]	Operational
Tandem BII(1440)	2	10.2/5.2	3,670	20[2]	Proposed
TE364(2300)	1	5.9/3.2	2,405	20[3]	Proposed
BII(2300)	1	6.3/5.2	2,717	20[3]	Proposed
BIIA(2300)	2	9.6/5.2	3,417	20[4]	Proposed
Tandem BII(2300)	2	12.4/5.2	5,530	20[4]	Proposed
Be BII(1-burn)	1	7.0/6.0	3,508	20[5]	Proposed
Be BII(2-burn)	1(1 restart)	7.0/6.0	3,508	20[5]	Proposed
OV1 (FW4)	1	6.8/2.1	860	--[6]	Operational
Fire	1	9.8/5.4	3,616	46	Semi-retired
Fire/FW4	2	16.8/5.4	4,252	15	Proposed

(a) Stage weights do not include nominal adapter weights indicated in column 5. Total velocity package weight is the sum of the stage weight and adapter weight.

(b) For greater than nominal payloads, the adapter weights must be increased as follows:
- (1) For payloads greater than 500 lb, add 10% of excess over 500 lb.
- (2) For payloads greater than 1000 lb, add 10% of excess over 1000 lb.
- (3) For payloads greater than 500 lb, add 12.5% of excess over 500 lb (2.5% is added to the interstage structure and 10% is added to payload adapter).
- (4) For payloads greater than 1000 lb, add 12.5% of excess over 1000 lb (2.5% is added to the interstage structure and 10% is added to payload adapter).
- (5) For payloads greater than 1500 lb, add 10% of excess over 1500 lb.
- (6) Adapter is integral with stage.

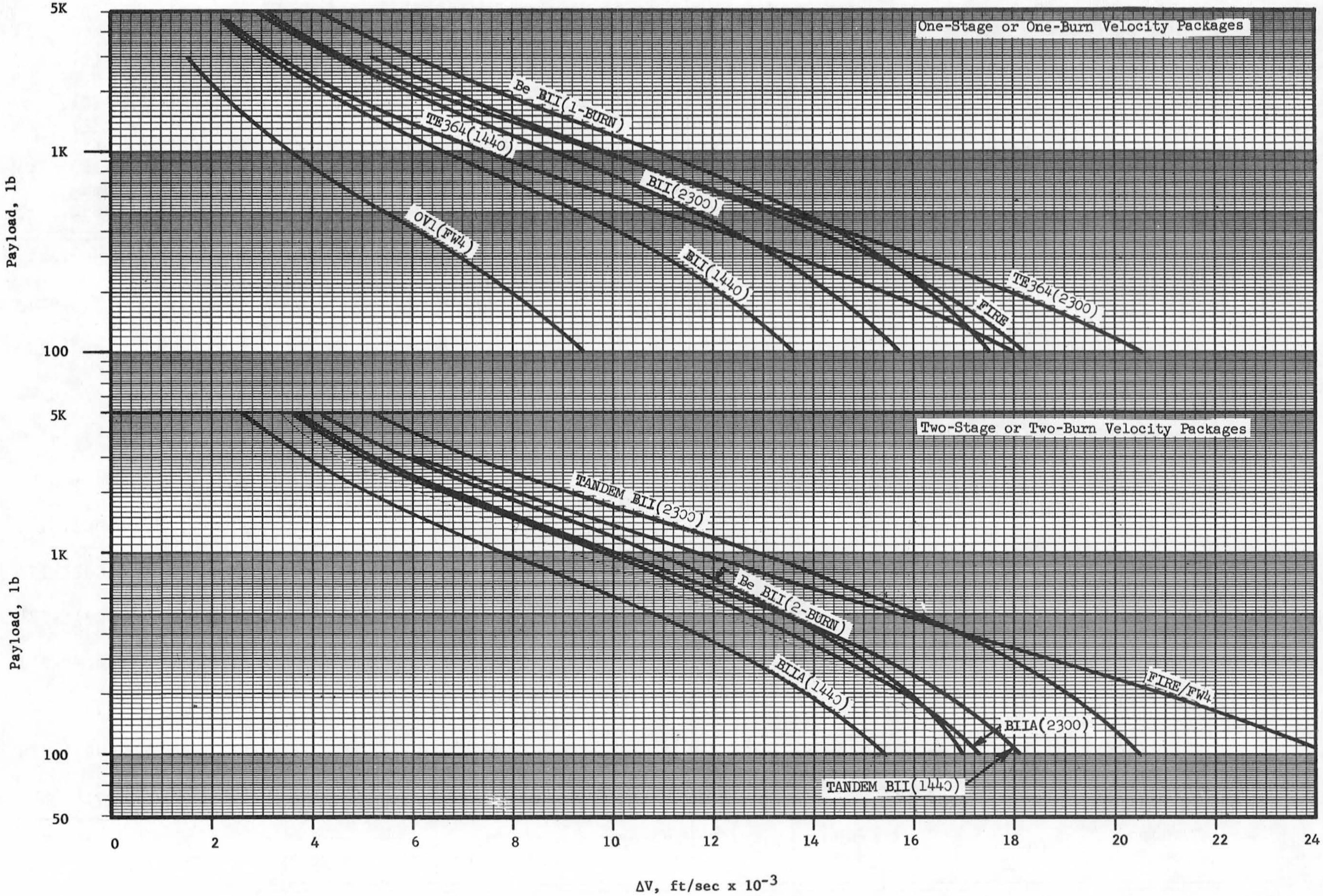

FIGURE IV-D-1. VELOCITY PACKAGE PERFORMANCE

SECTION IV-E

PERFORMANCE OF SOLAR-ELECTRIC PROPULSION SYSTEMS

SECTION IV-E

PERFORMANCE OF SOLAR-ELECTRIC PROPULSION SYSTEMS

This section contains performance data for selected launch vehicles with spacecraft containing a solar-electric propulsion system. The solar-electric propulsion concept has been separated from other advanced propulsion systems discussed in Section IV-F because it is considered to be a nearer-term possibility.

With solar-electric propulsion, energy derived from lightweight photovoltaic arrays would be used to accelerate ions to very high exhaust velocities. The system yields a very high specific impulse (I_{SP}), a factor of 10 or more above that achieved in chemical systems. This results in a significantly smaller propellant requirement to achieve a given total impulse. However, because of their low thrust, such systems would operate throughout a significant fraction of the mission time.

Where a continuous thrust is applied to a spacecraft, the distinction between the delivery system and the spacecraft/payload is no longer clear. It has not been possible to show the total vehicle mission performance in terms of plots of payload versus characteristic velocity because the spacecraft propulsion system now provides a major portion of the required final mission energy, which is no longer attained in the proximity of the Earth as is the case with ballistic missions. Rather, it is slowly attained, with the actual thrust vector history being determined by the target location relative to the Earth at launch and by desired end conditions. For this reason, no single general form has yet been derived to present performance data for low thrust spacecraft, and the performance of the low thrust spacecraft must be presented on a total vehicle and mission-by-mission basis.

To better understand the results shown in this section, some discussion of the electric propulsion system and mission design process may be helpful. With conventional chemical propulsion systems, there is a relatively limited choice of system parameters and control variable histories. Electric propulsion system and mission design studies are greatly complicated by the increased freedom of choice in system parameters and operating modes available. The design specific impulse can be varied over a wide range of values. Engine operating thrust is also variable within limits. Relatively unlimited storage capability is available because of the nature of the propellant, usually a liquid metal. The engine can be shut down and restarted with relative ease. The thrusting period is lengthy.

The performance of solar-electric propulsion systems is sensitive to the choice of thrusting history and system parameters. The analyst must determine the best combination of such parameters as boost vehicle injection energy, spacecraft thrust program, solar array size, fuel mass, and launch period while considering operational constraints such as guidanc requirements, ground coverage requirements, operational simplicity and mission reliability.

The overall region of possible mission operation is determined by assuming a "rubber spacecraft" and calculating the optimum constrained set of parameters for each launch date. This results in different values of power, exhaust velocity (I_{SP}), fuel mass, injected mass, injection energy, and thrust program at each point along the performance curve for each mission. This could never be mechanized in reality, since a spacecraft has fixed parameters once it is constructed. However, this type of calculation does indicate the limiting performance possible and the concomitant spacecraft parameters.

Figures IV-E-1 through IV-E-14 show current estimates of possible planetary flyby and orbiter mission performance capabilities with spacecraft using solar-electric propulsion and launched on three possible launch vehicles. The procedures employed to prepare the data were as described in the previous paragraph (References IV-E-1 through IV-E-3). The spacecraft propulsion systems would be based on present NASA programs oriented toward demonstrating the technology of small, solar-cell-powered, ion engine propulsion systems (Reference IV-E-4). Assuming successful completion of this effort, operational launches in the mid to late 1970's appear to be practical.

These figures are based on an assumption of circular coplanar planetary orbits and a propulsion system specific weight of 30 kg/kw at 1 a.u. (considered to include the arrays, thrustors, and power conversion equipment). The net spacecraft mass (payload) is defined as the initial spacecraft weight minus the weight of the propulsion system (as defined above), propellant, tankage, and any structure that would not be required if the spacecraft propulsion system were not used. For the orbiter missions, a highly eccentric elliptical orbit and a chemical retro-propulsion unit with an I_{SP} of 300 sec and a propellant fraction of 0.9 for establishing the orbit have been assumed. For outer planet (Jupiter and beyond) and Mercury orbiters (Figures IV-E-7 to IV-E-12), the solar-electric propulsion system is assumed to be jettisoned prior to the retro maneuver. For inner planet orbiters except Mercury (Figures IV-E-10 to IV-E-12), the solar-electric power supply is retained and goes into orbit with the spacecraft. The mass of the power supply is not included, however, in the net spacecraft mass values shown on the Figures.

It should be noted that the payloads for orbiter missions are highly dependent upon specification of the retro-propulsion system and of the capture orbit. For cases in which the base vehicle outperformed the vehicle plus solar-electric combination, the basic launch vehicle performance has been shown by a broken line (e.g., Figure IV-E-4). The curves labeled with a "D" are for direct trajectories (transfer angle less than 360°), while those labeled "I" are for indirect trajectories (transfer angle greater than 360°). The approximate value of the power in kilowatts available at 1 a.u. and (in parentheses) at the destination for each mission is also indicated on the curves.

Figures IV-E-15 through IV-E-19 provide further data on solar-electric propulsion. These are included to illustrate the effect of launch date and the effect of selecting a single spacecraft power level on solar-electric performance.

Figure IV-E-15 shows the results of trajectory calculations based on planetary ephemerides. These data were generated as a part of a study on the feasibility of a Jupiter flyby using the SLV3C/Centaur with a solar-electric spacecraft (Reference IV-E-4). As can be seen, the performance of the solar-electric system is dependent on launch date, but the dependence is less severe than for ballistic trajectories.

It should be noted that there is an alternative to the assumption of the optimized, flexible parameter ("rubberized") spacecraft used in developing data for Figures IV-E-1 through IV-E-15). This alternative involves selection of a limited number (hopefully, only one) of fixed spacecraft designs or a single stage design and investigating their performance for various missions. As an example, Figures IV-E-16 through IV-E-19 illustrate the performance of a 15 kw solar-electric spacecraft on the Titan IIID/Centaur (References IV-E-1 through IV-E-3, IV-E-5, and IV-E-6). If good basic design choices can be made so that it is feasible to use such spacecraft or stages for multiple missions, without severe performance penalties, this type of system could provide a cost-effective method for return of scientific data from a variety of solar system locations. Further definition of the multimission spacecraft and stage concept is expected in the future.

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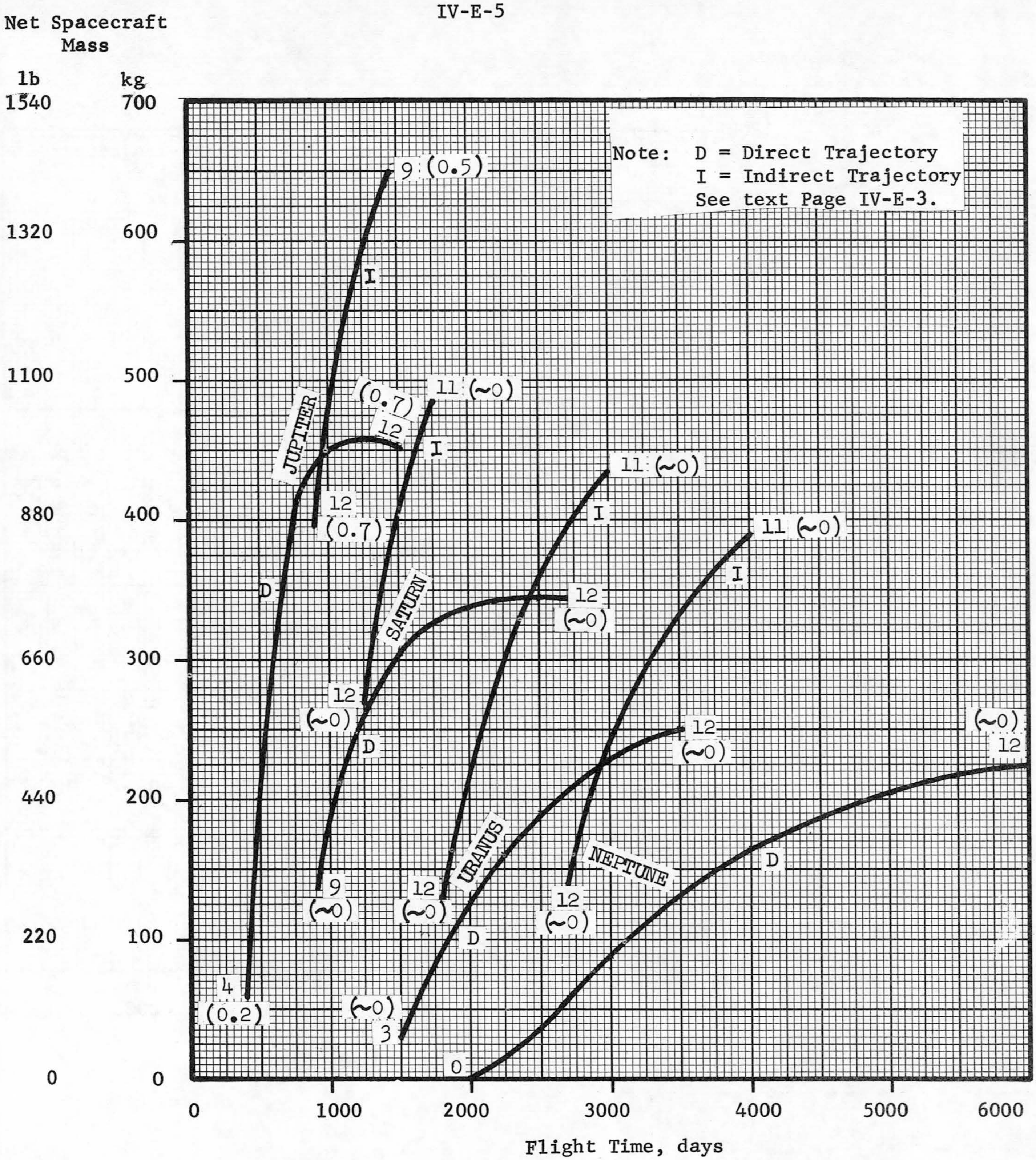


FIGURE IV-E-1. CAPABILITY OF AN SLV3C/CENTAUR SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET FLYBYS

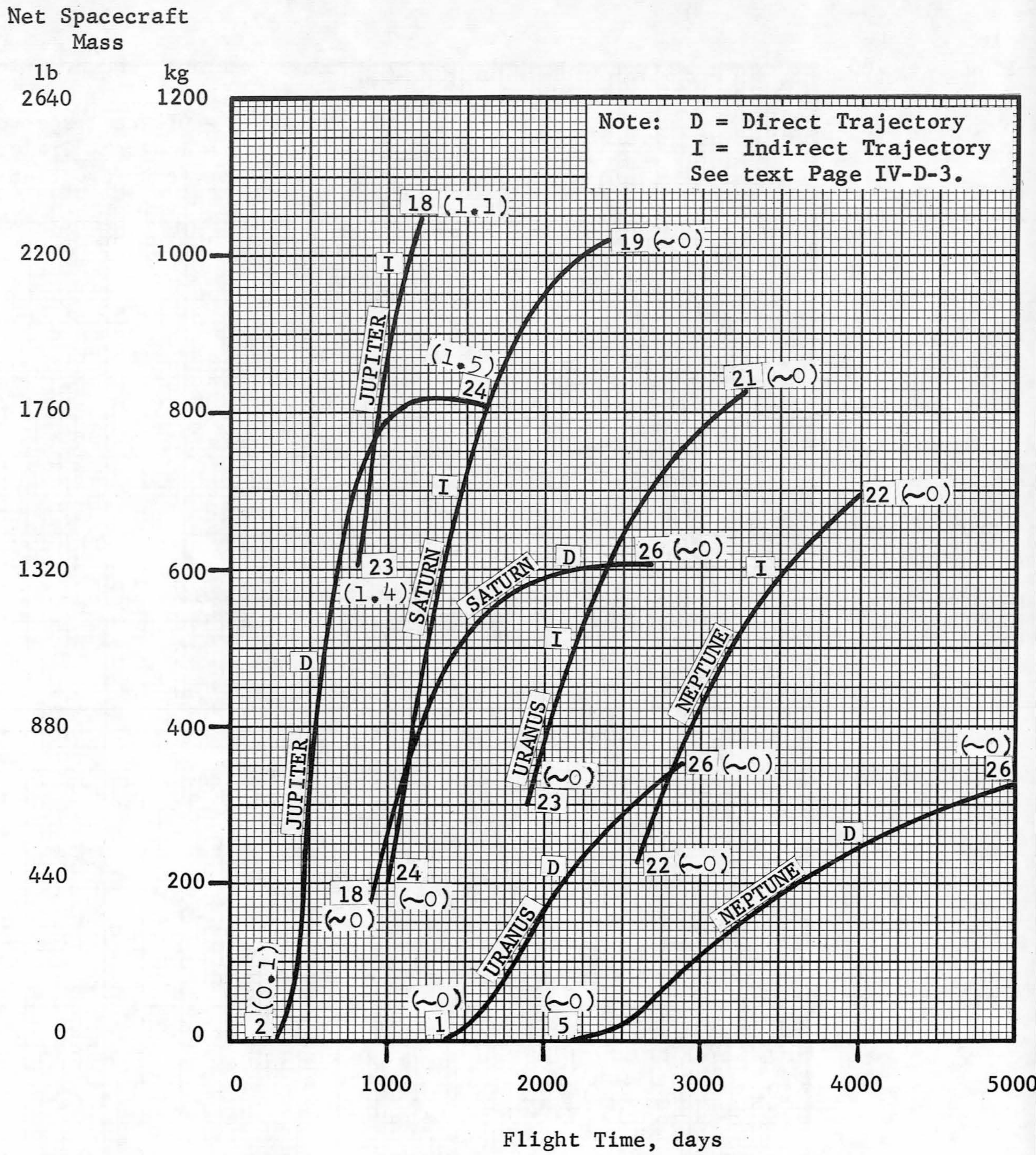


FIGURE IV-E-2. CAPABILITY OF A TITAN IIIC/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET FLYBYS

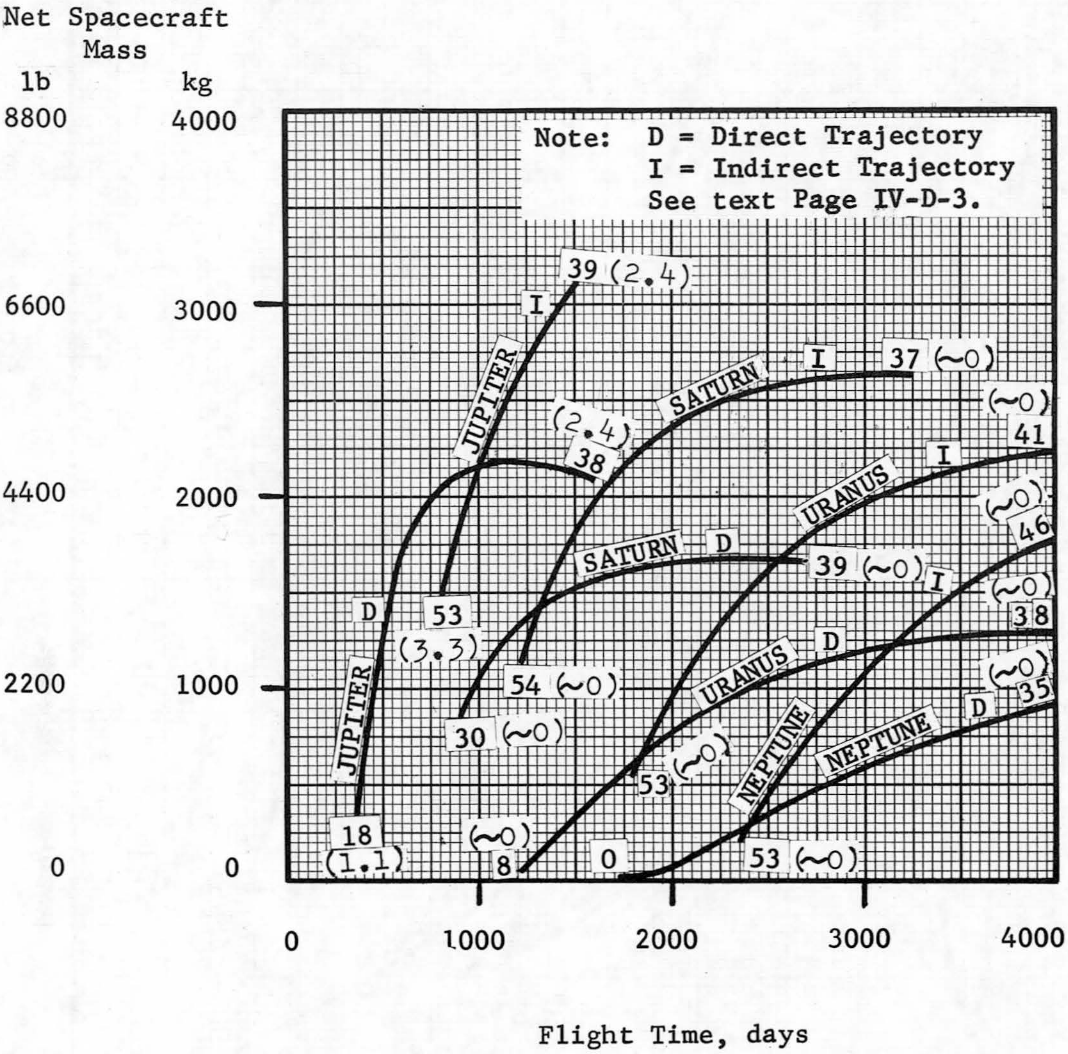


FIGURE IV-E-3. CAPABILITY OF A TITAN IIID/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET FLYBYS

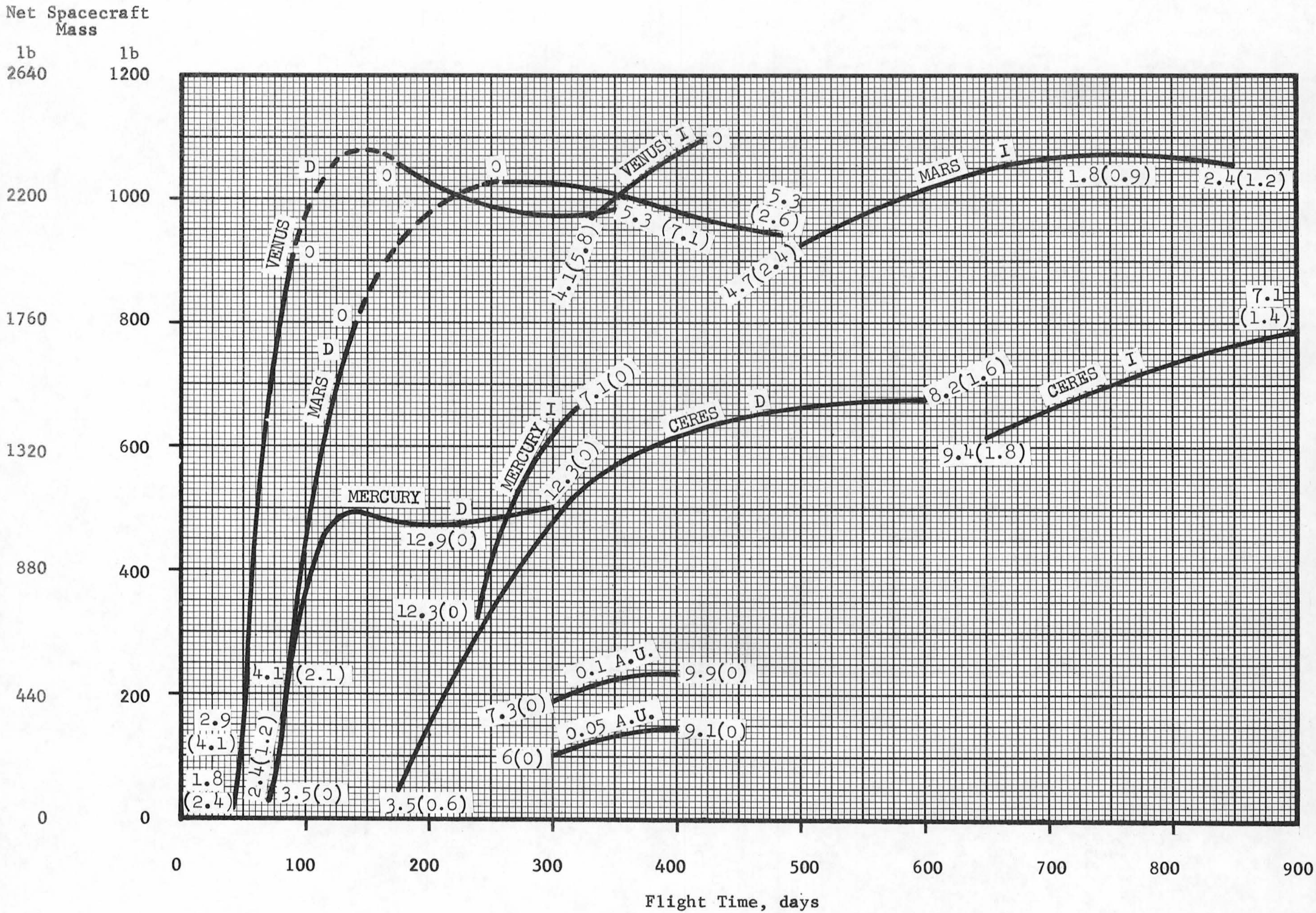


FIGURE IV-E-4. CAPABILITY OF AN SLV3C/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY AND CERES FLYBYS AND CLOSE SOLAR PROBES

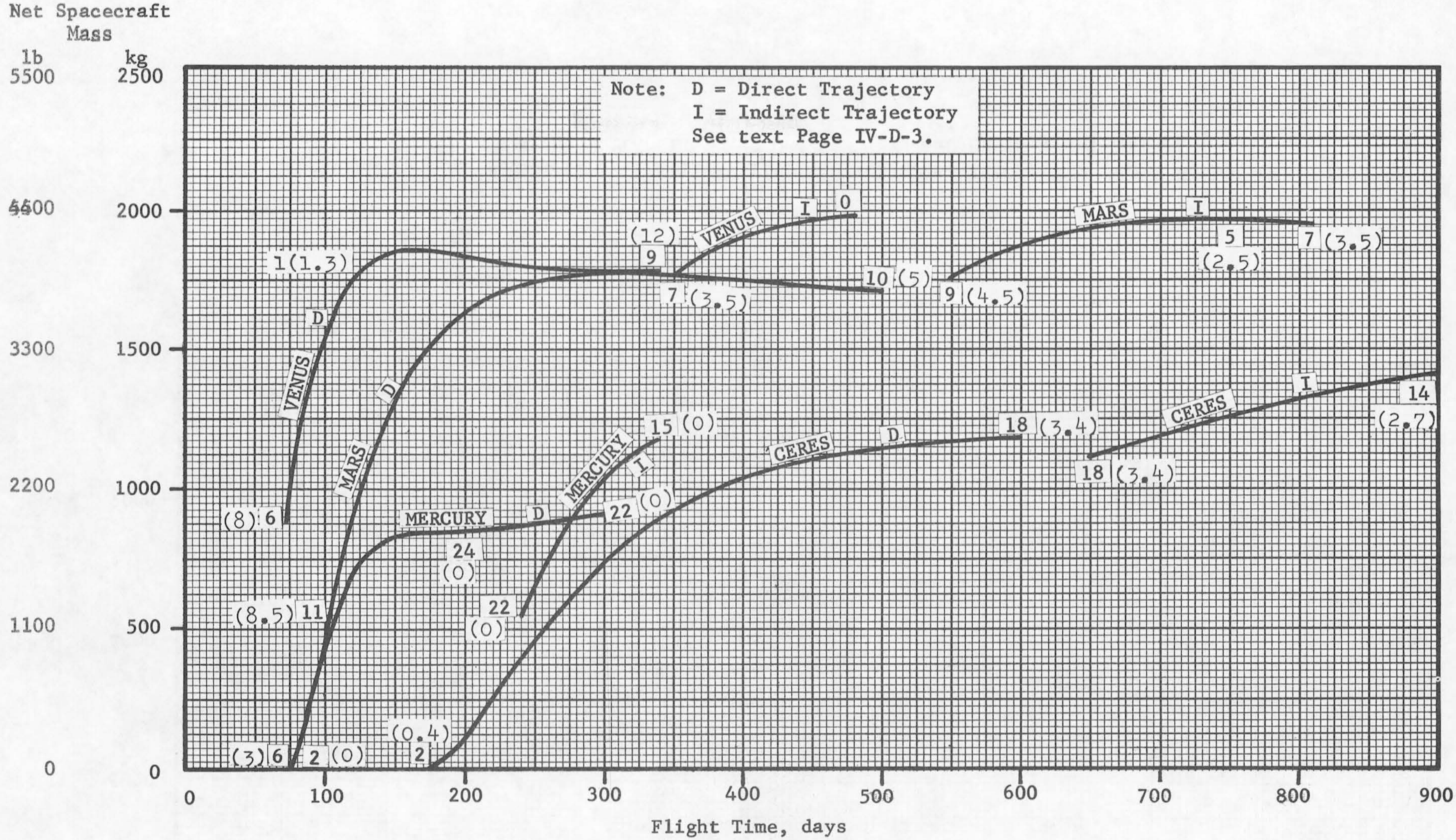


FIGURE IV-E-5. CAPABILITY OF A TITAN IIIC/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY AND CERES FLYBYS

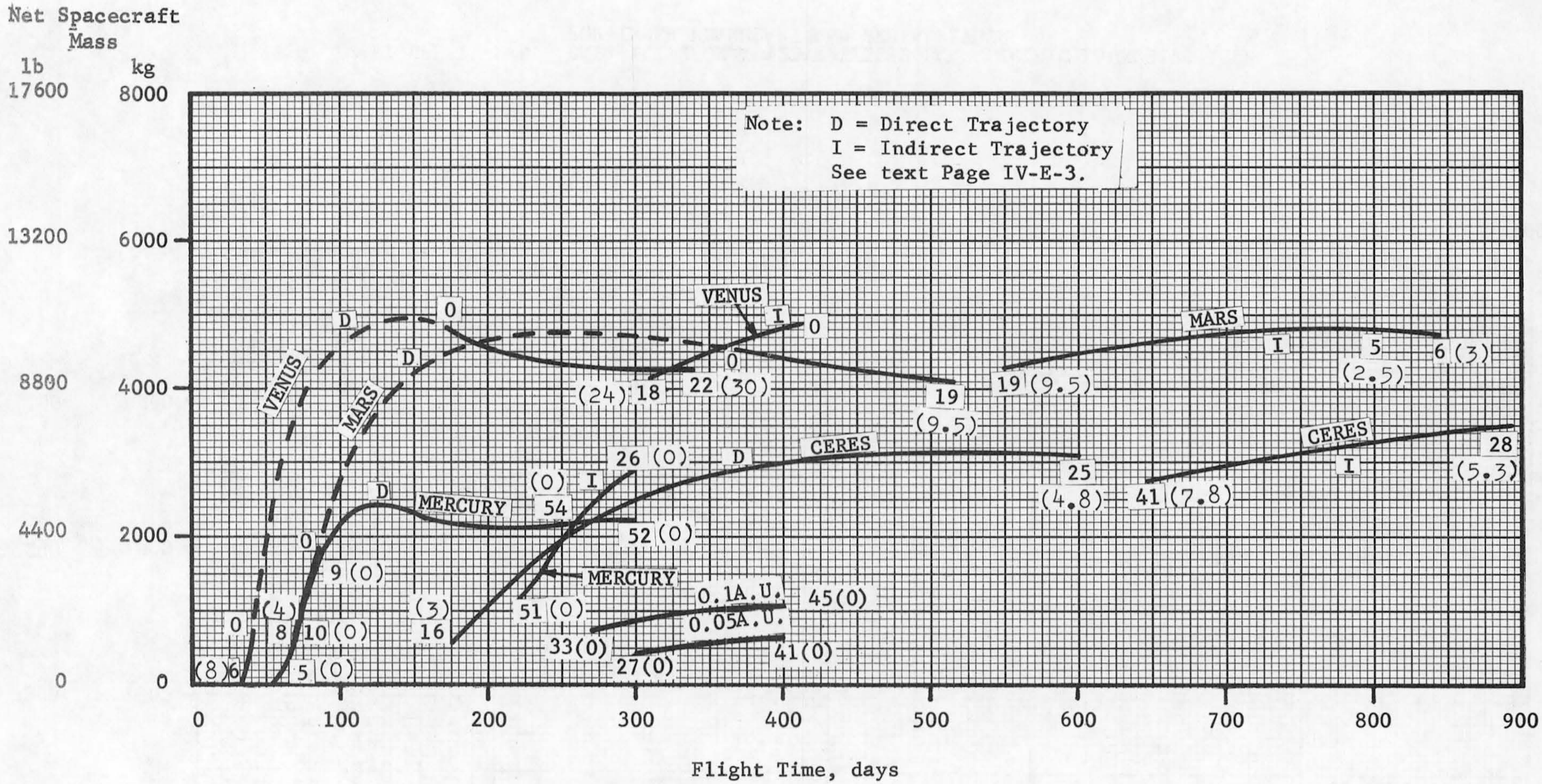


FIGURE IV-E-6. CAPABILITY OF A TITAN IIID/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY AND CERES FLYBYS AND CLOSE SOLAR PROBES

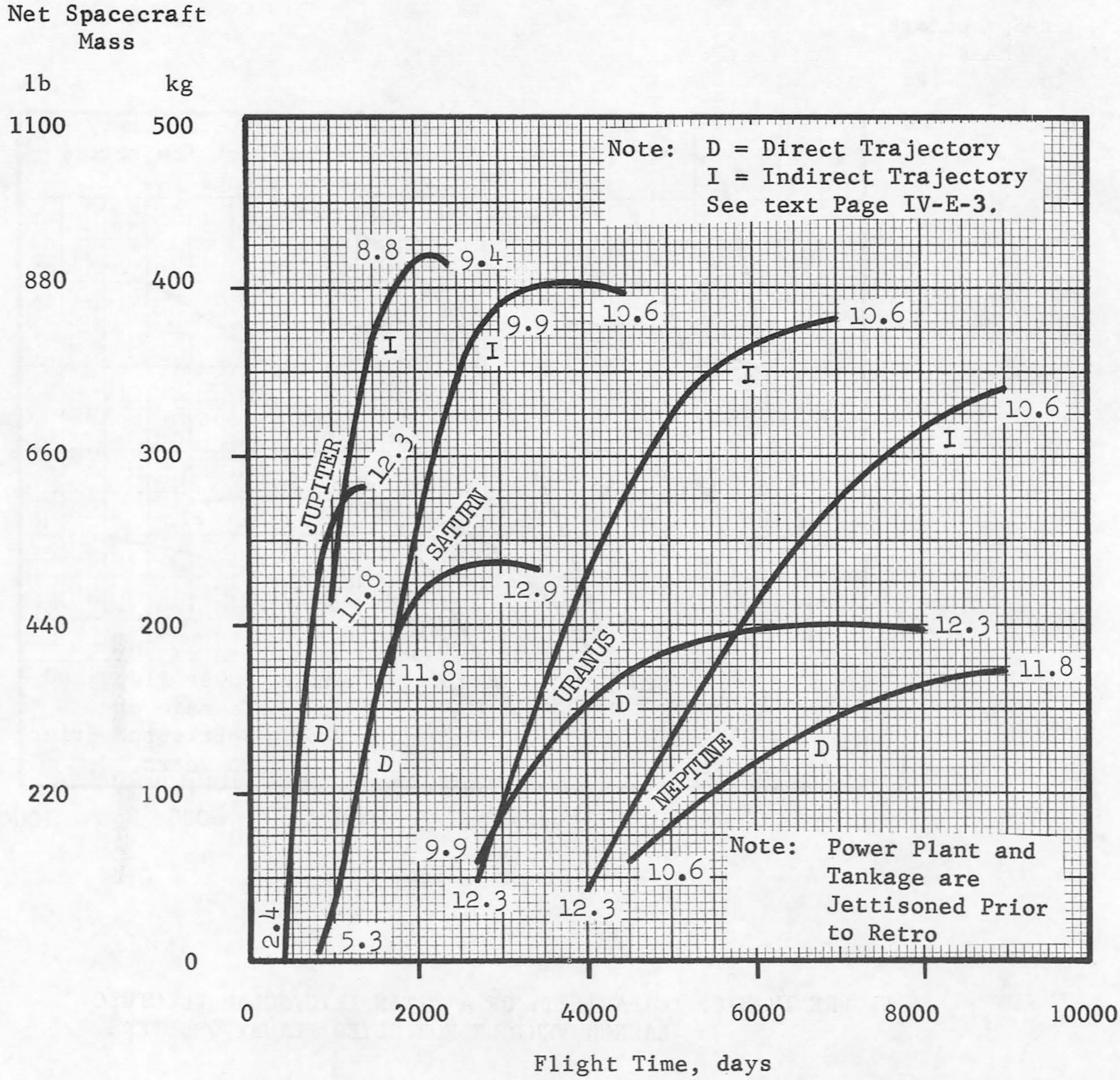


FIGURE IV-E-7. CAPABILITY OF AN SLV3C/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET ORBITERS

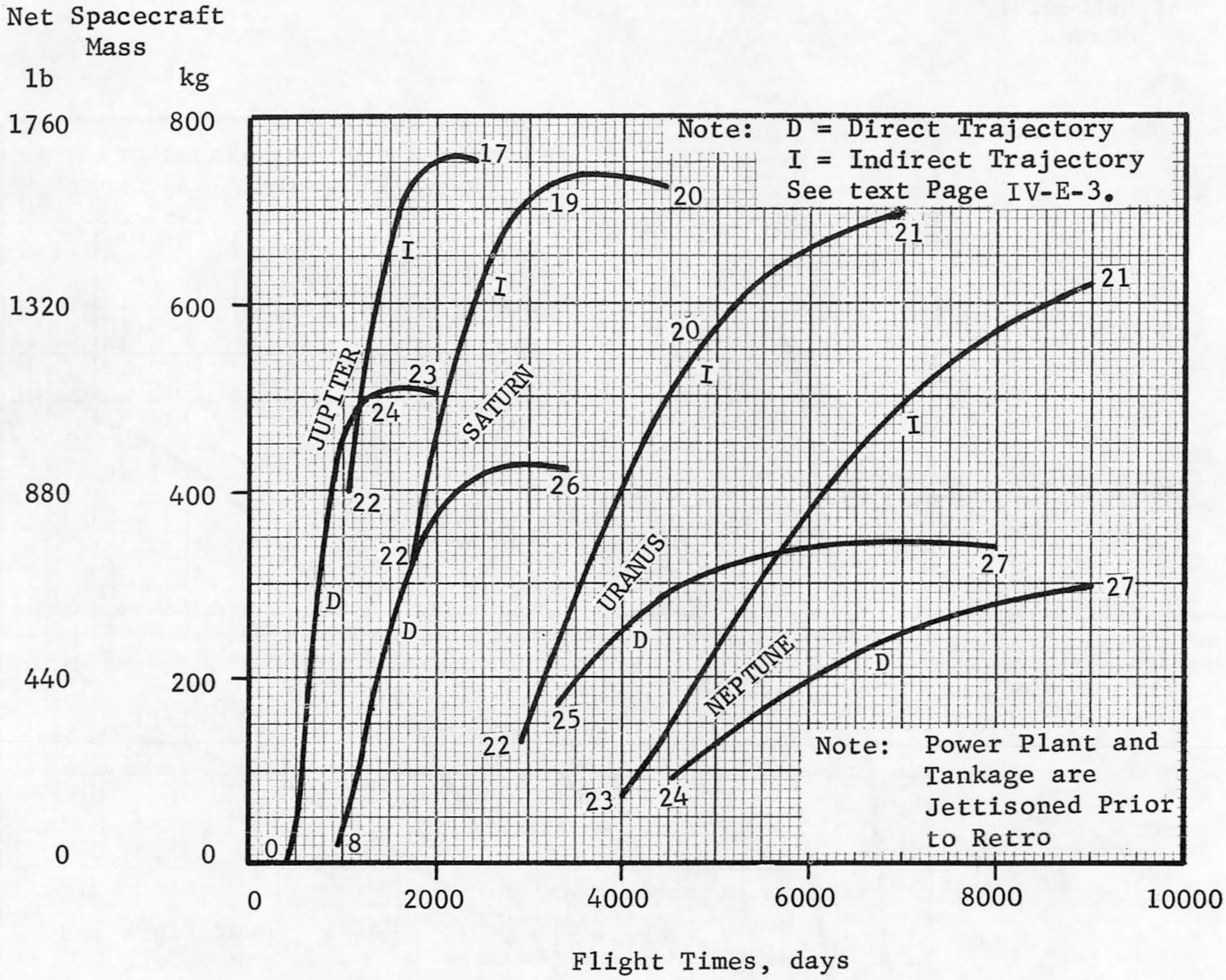


FIGURE IV-E-8. CAPABILITY OF A TITAN IIIC/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET ORBITERS

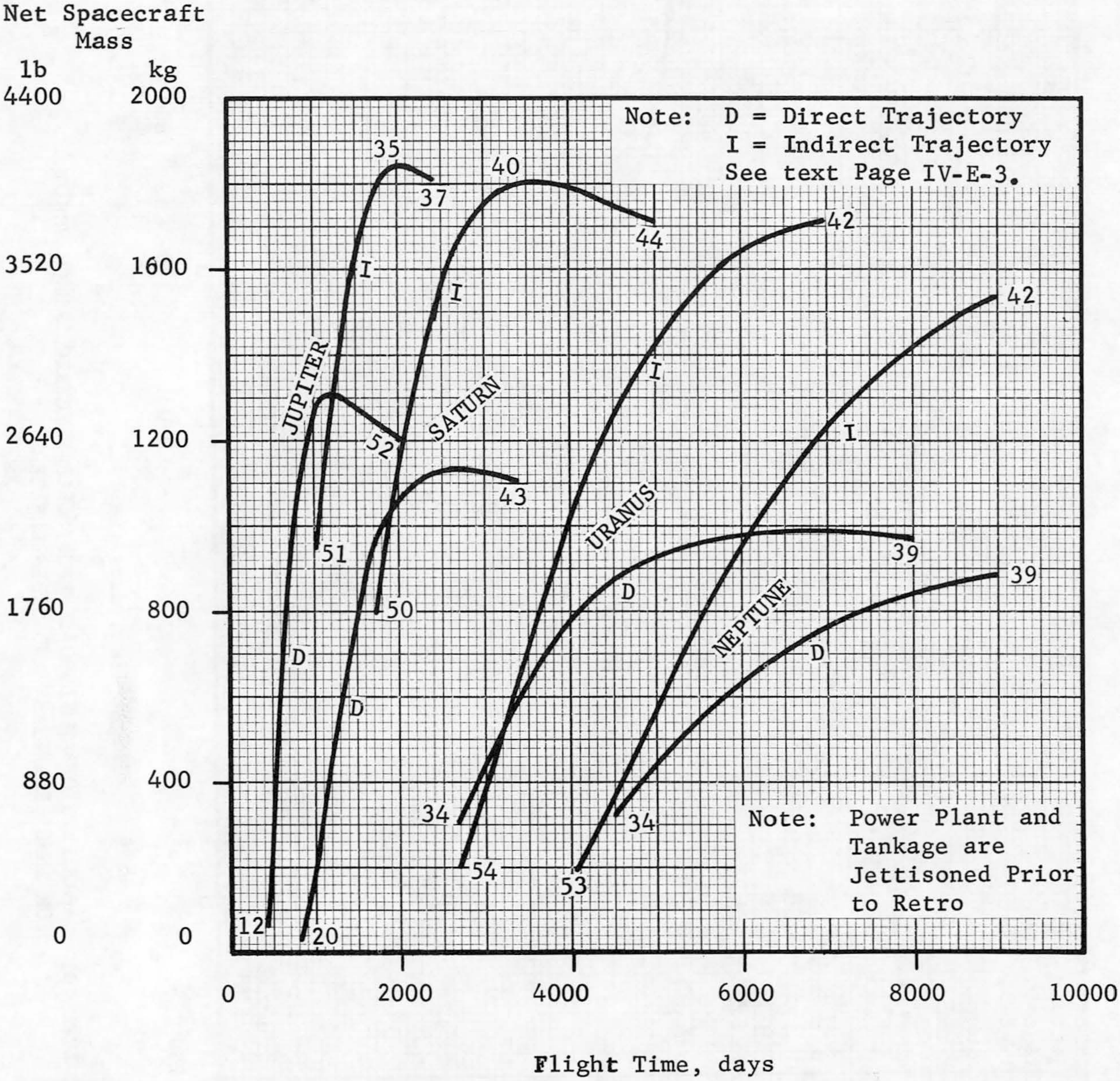
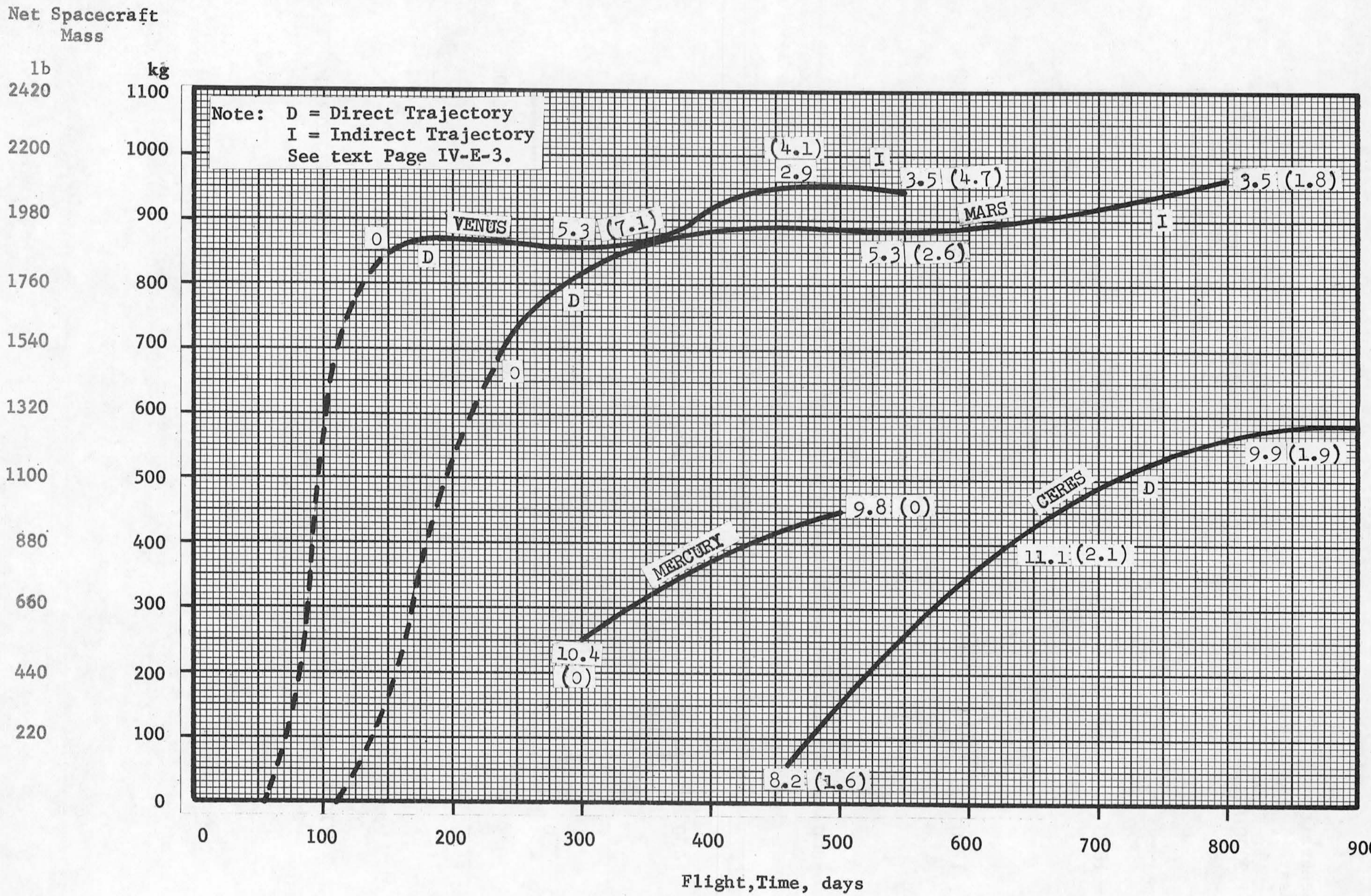


FIGURE IV-E-9. CAPABILITY OF A TITAN IIID/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET ORBITERS



FLIGHT IV-E-10. CAPABILITY OF AN SLV3C/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY AND CERES ORBITERS

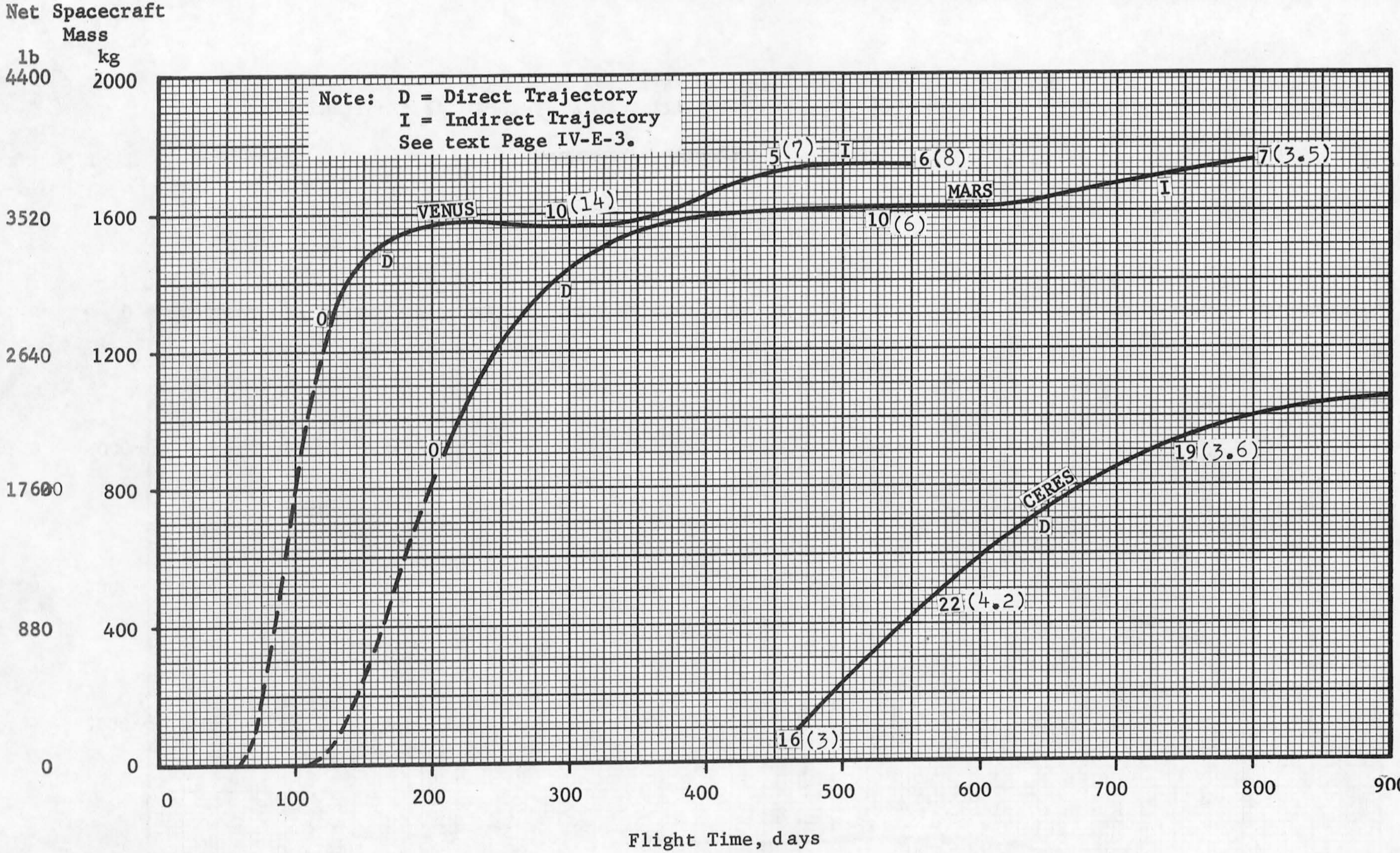


FIGURE IV-E-11. CAPABILITY OF A TITAN IIIC/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY ORBITER AND CERES RENDEZVOUS MISSIONS

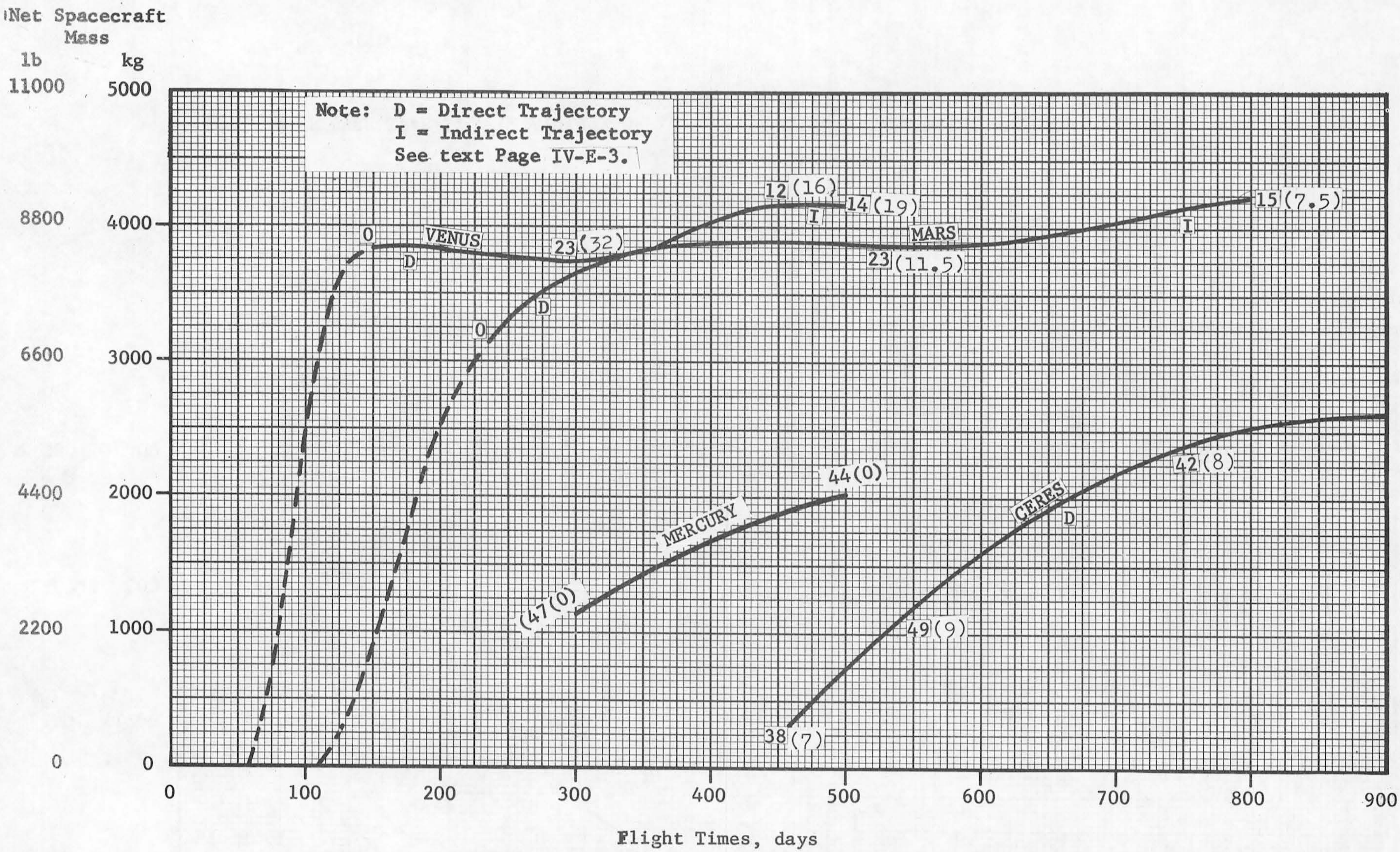


FIGURE IV-E-12. CAPABILITY OF A TITAN IIID/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY ORBITER CERES RENDEZVOUS MISSIONS

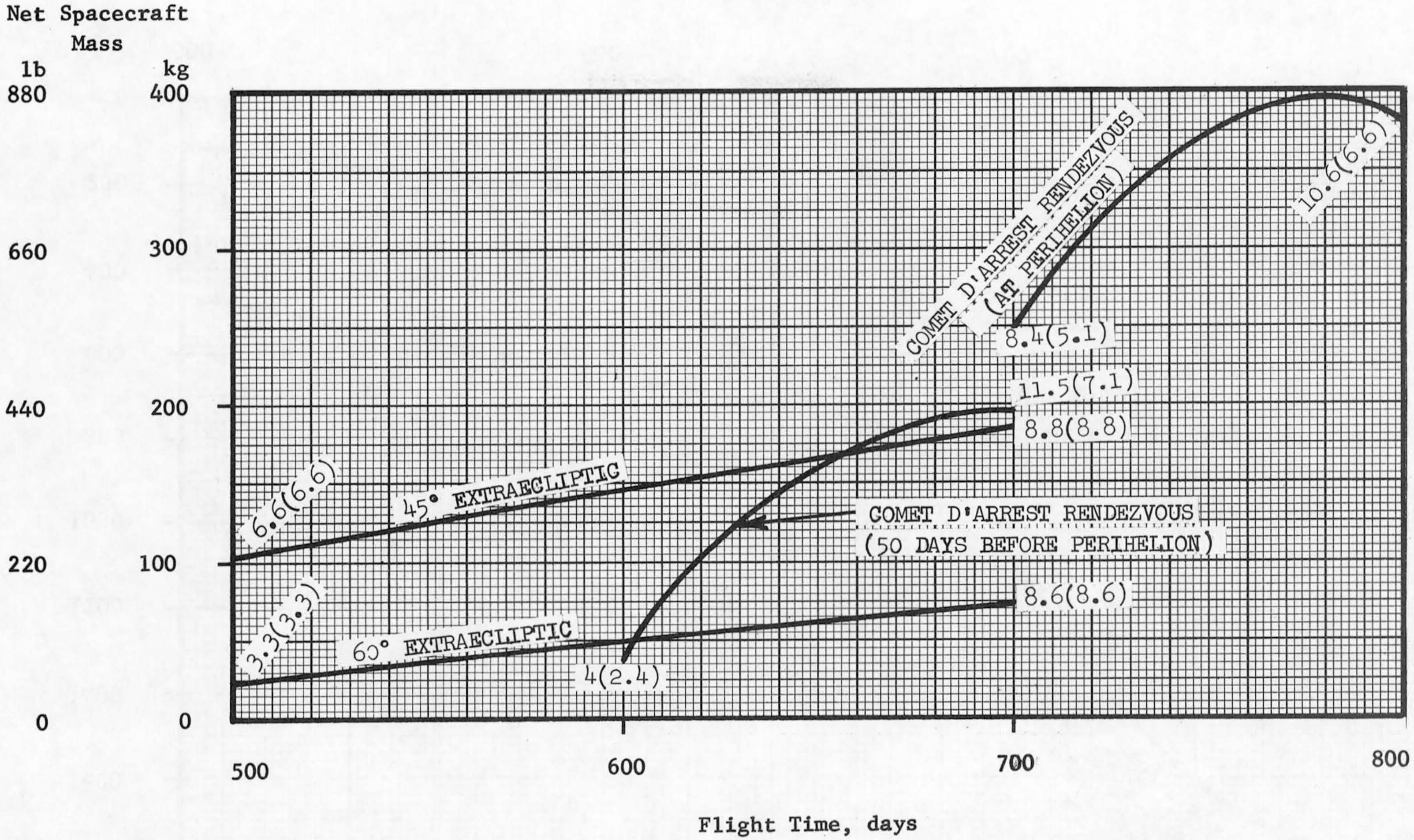


FIGURE IV-E-13. CAPABILITY OF AN SLV3C/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR COMETARY AND EXTRAECLIPTIC MISSIONS

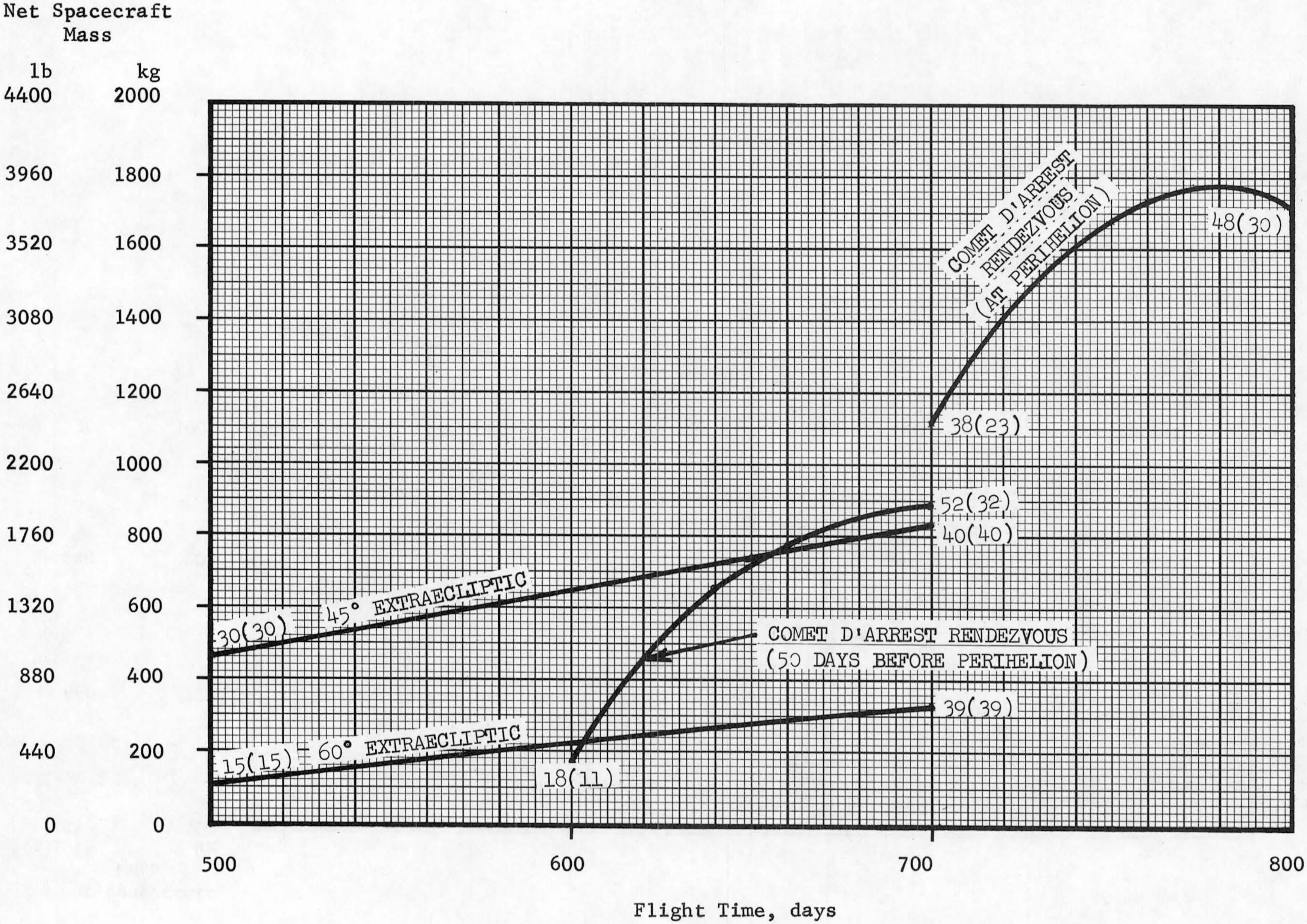


FIGURE IV-E-14. CAPABILITY OF A TITAN IIID/CENTAUR/SOLAR-ELECTRIC LAUNCH VEHICLE FOR COMETARY AND EXTRAECLIPTIC MISSIONS

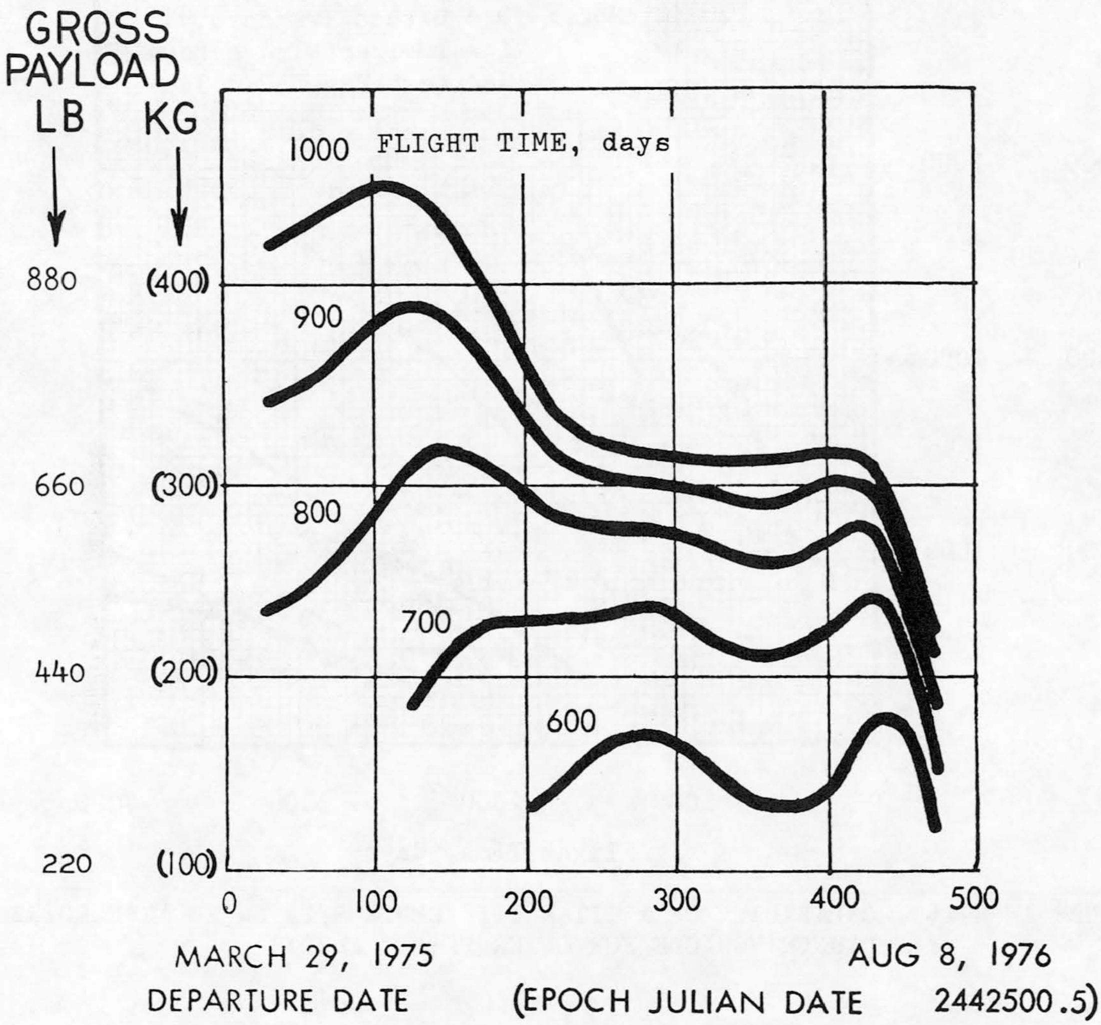


FIGURE IV-E-15. GROSS PAYLOAD VERSUS DEPARTURE DATE FOR JUPITER FLYBY WITH AN SLV3C/CENTAUR/SOLAR-ELECTRIC SPACECRAFT

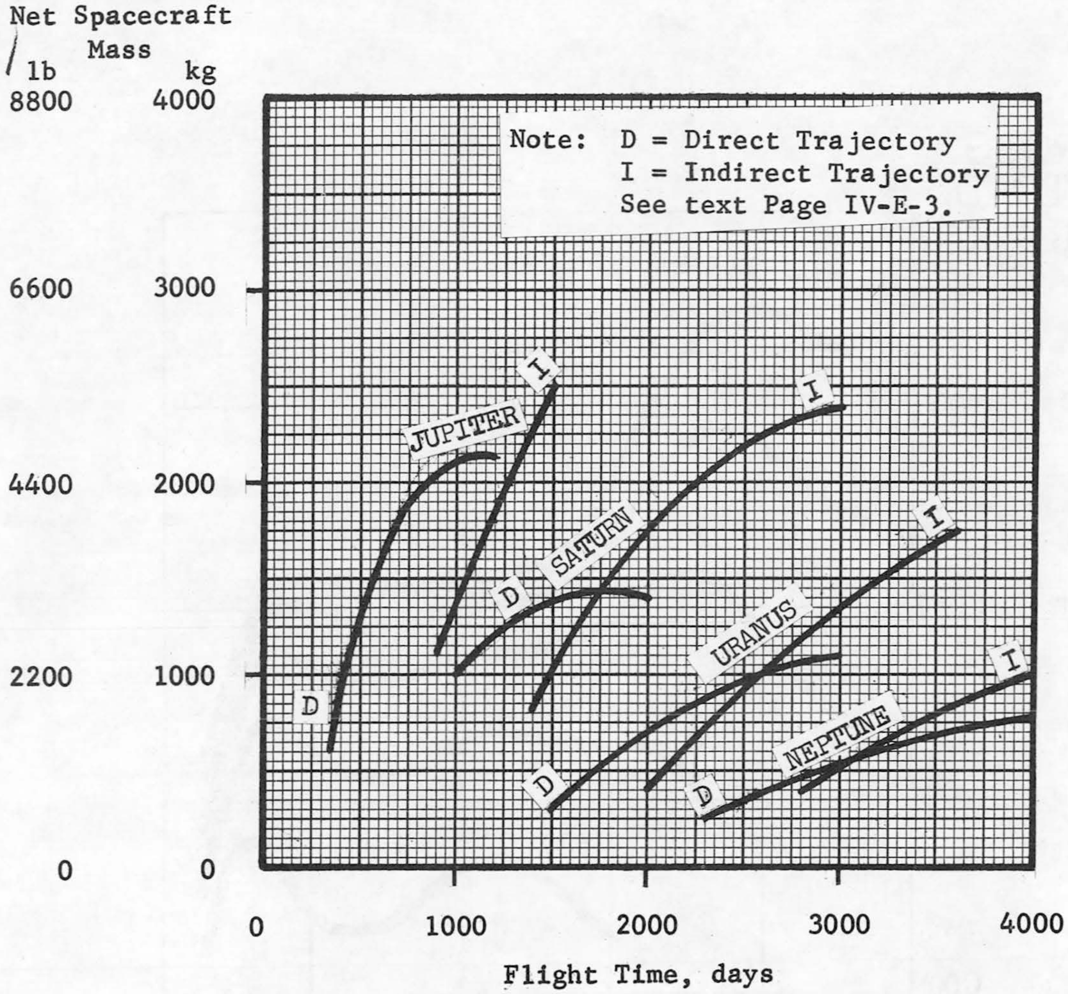


FIGURE IV-E-16. CAPABILITY OF A TITAN IIID/CENTAUR/15 KW SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET FLYBYS

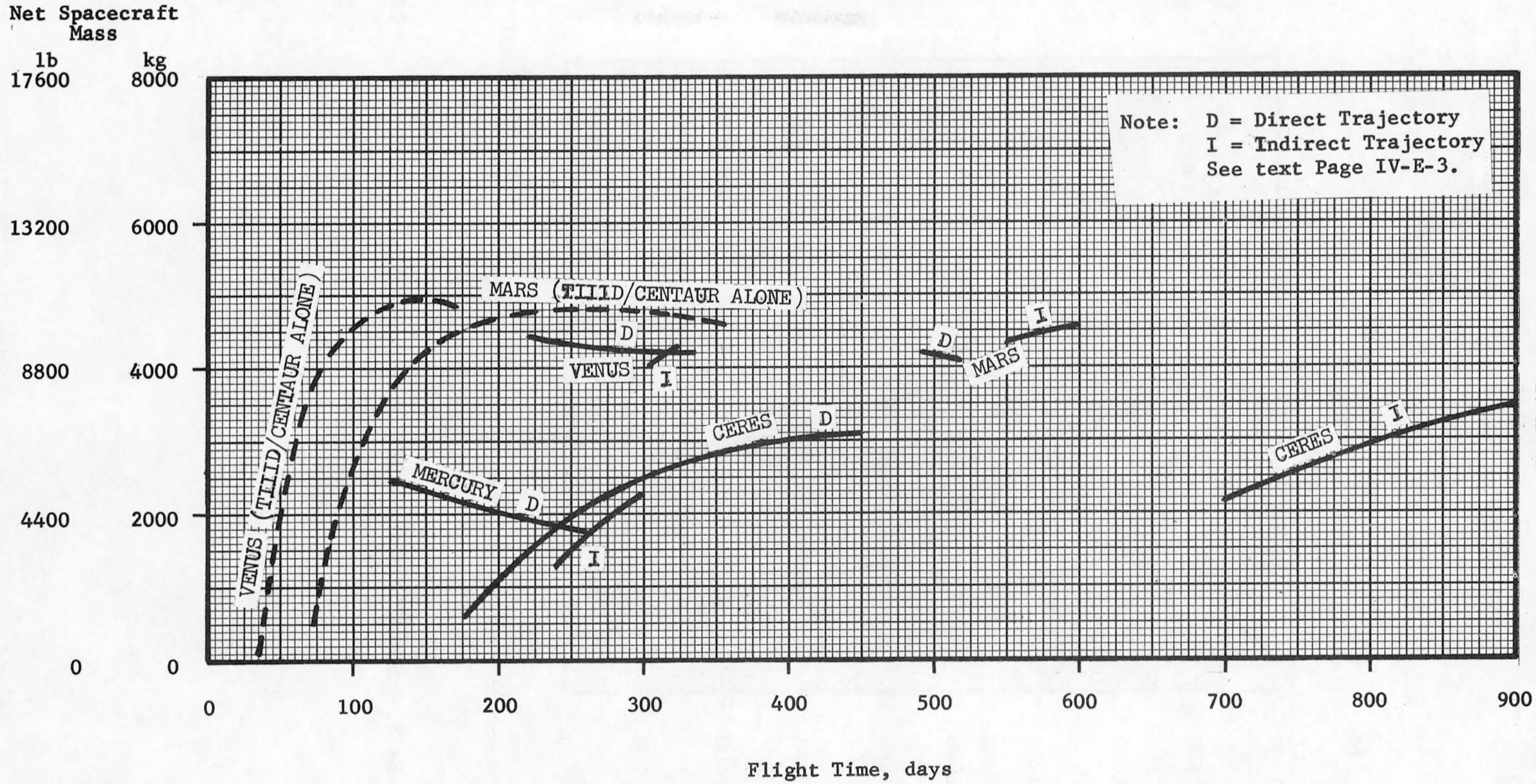


FIGURE IV-E-17. CAPABILITY OF A TITAN IIID/CENTAUR/15 KW SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY AND CERES FLYBYS

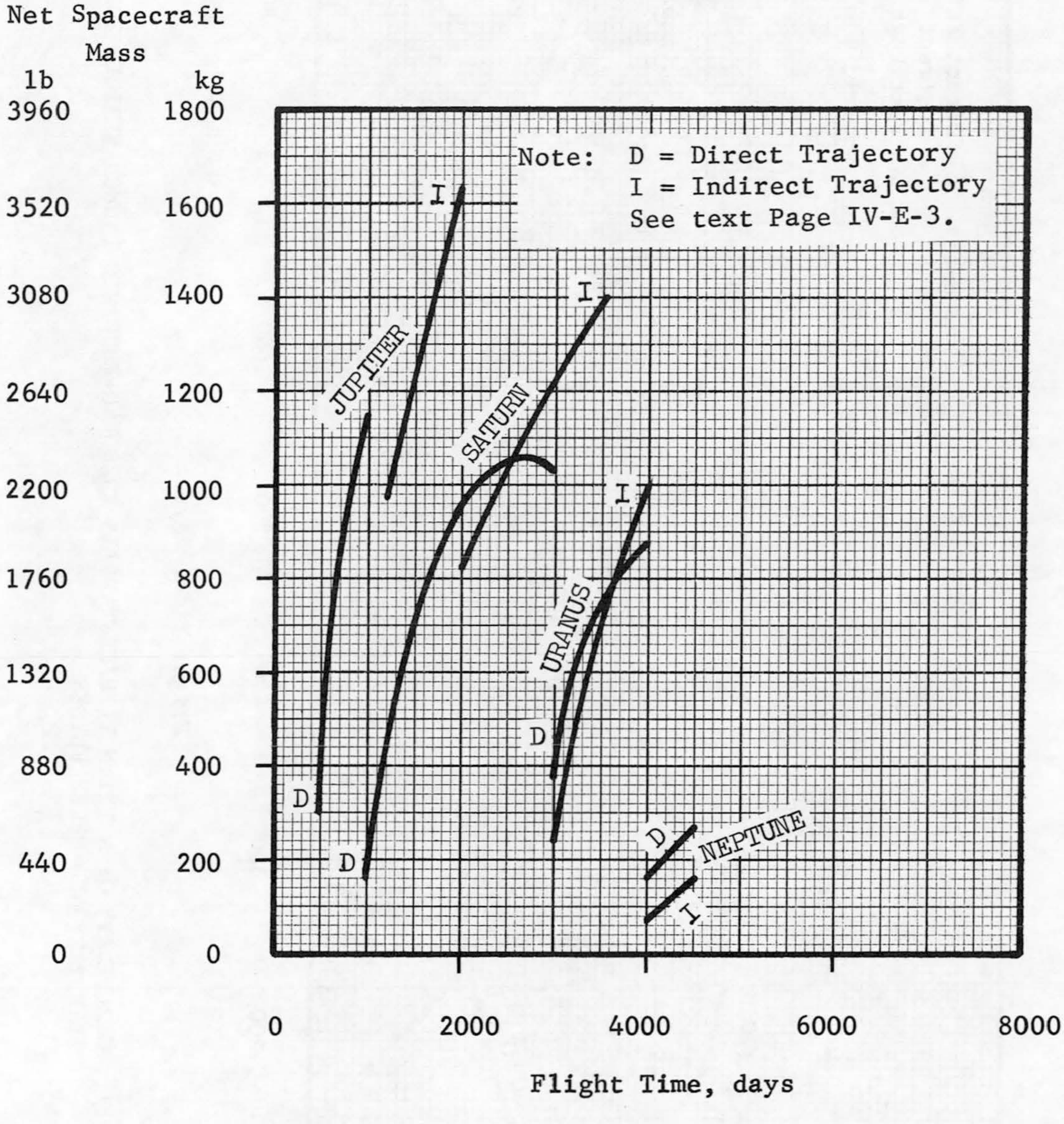


FIGURE IV-E-18. CAPABILITY OF A TITAN IIID/CENTAUR/15 KW SOLAR-ELECTRIC LAUNCH VEHICLE FOR OUTER PLANET ORBITERS

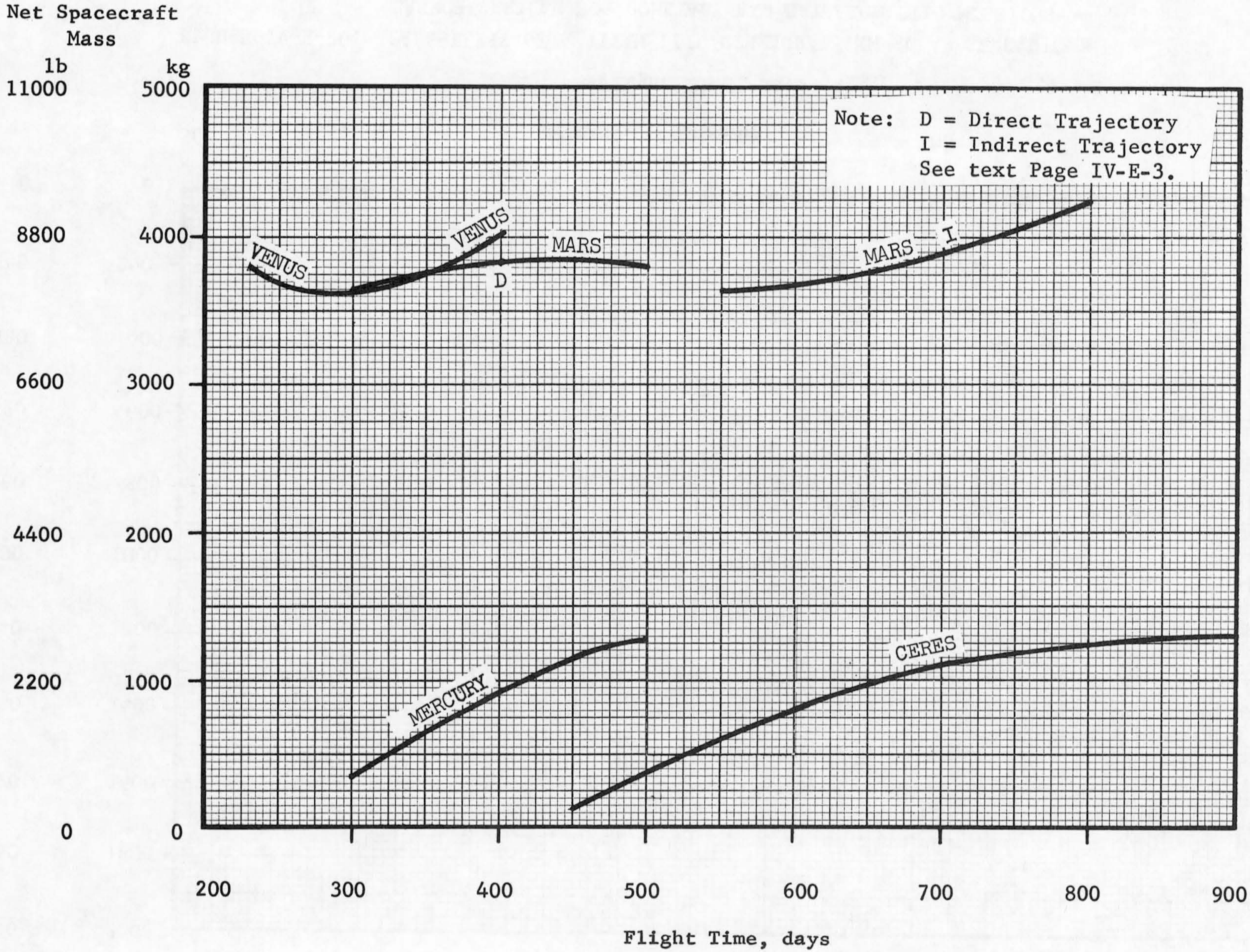


FIGURE IV-E-19. CAPABILITY OF A TITAN IIID/CENTAUR/15 KW SOLAR-ELECTRIC LAUNCH VEHICLE FOR INNER PLANETARY AND CERES ORBITERS

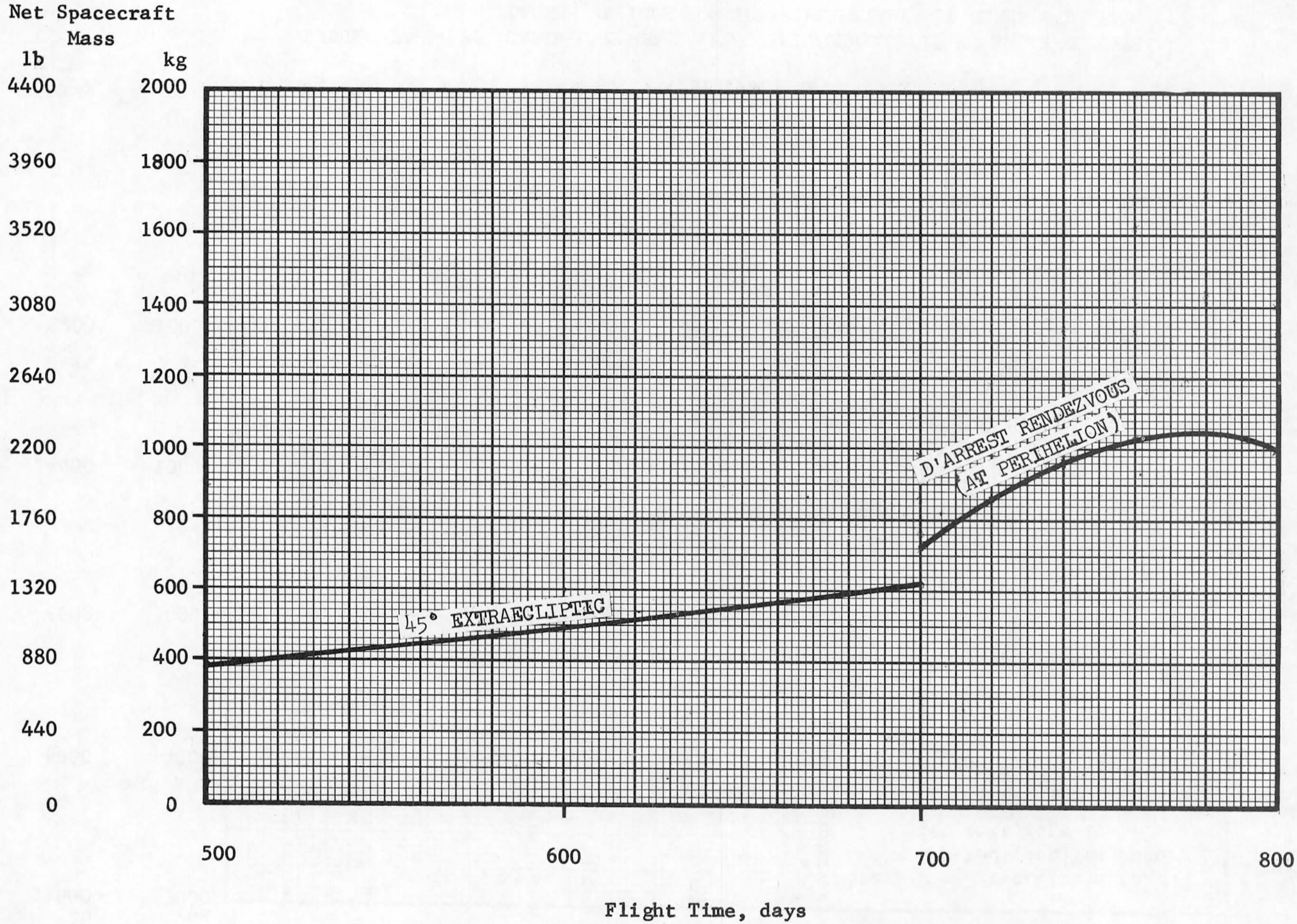


FIGURE IV-E-20. CAPABILITY OF A TITAN IIID/CENTAUR/15 KW SOLAR-ELECTRIC LAUNCH VEHICLE FOR COMETARY AND EXTRAECLIPTIC MISSIONS

SECTION IV-F

PERFORMANCE OF NUCLEAR-THERMAL AND NUCLEAR-ELECTRIC PROPULSION SYSTEMS

SECTION IV-F

PERFORMANCE OF NUCLEAR-THERMAL AND NUCLEAR-ELECTRIC PROPULSION SYSTEMS

The figures of this section present performance data for selected launch vehicles with upper stages or spacecraft using nuclear-thermal or nuclear-electric propulsion systems. The systems shown were selected as having significant potential for improving launch vehicle capability.

Figures IV-F-1 and IV-F-2 show performance estimates for vehicles with a solid-core nuclear rocket stage [identified here as SIVB(N)] substituted for the SIVB. This stage could be developed using the results of the current NERVA technology development program and could probably be available by 1978-79 (see Reference IV-F-1). The SIVB(N) would have one 75,000 lb thrust, 825 sec I_{SP} nuclear rocket. The SIVB(N) propellant loading varies with each vehicle and has been selected to maximize overall launch vehicle performance when the Centaur is used as an upper stage (see Reference IV-F-2). The performance data are based on an assumption of a suborbital start where required.

Figure IV-F-3 shows performance estimates for vehicles with a 38,000 lb weight solid-core nuclear rocket stage [identified here as Centaur(N)] substituted for the Centaur. This stage design is based on past studies of small solid-core nuclear engines and stages (see Reference IV-F-3). The stage could probably be available by 1978.

Figure IV-F-4 shows performance estimates for vehicles where the SIVB(N) and Centaur(N) solid-core nuclear rocket stages are substituted for the SIVB and Centaur, respectively. These curves indicate the upper limit of launch vehicle performance improvement potentially available through the introduction of current nuclear propulsion technology.

Figures IV-F-5 to IV-F-8 show the potential performance of a nuclear-electric propulsion system on the Titan IIID(7)/Centaur (see Reference IV-F-4 for related information). These data are based on an assumed propulsion system specific weight of 40 lb/kw (considered to include the power source, thrustors, and power conversion equipment).

Payload is defined as the initial spacecraft weight minus the weight of the propulsion system (as defined above), propellant, tankage, and any structure that would not be required if the spacecraft propulsion system were not used. For orbiter missions, a highly eccentric elliptical orbit with periapsis equal to 2 planetary radii, and a chemical retro-propulsion unit with an I_{SP} of 300 sec and a propellant fraction of 0.8 has been assumed. The data shown are optimized; i.e., for each objective and flight time, a specific set of spacecraft parameters (power level, specific impulse, booster injection velocity, etc.) and a thrusting history have been chosen that yield the maximum delivered payload.

It should be emphasized that the nuclear-electric propulsion technology is not as well established as that of solar-electric propulsion. Whereas the solar-electric system is a near-term possibility, the nuclear-electric system is not likely to be available before the mid 1980's.

References

IV-F-1. "20-Year Forecast of Nuclear Propulsion", Memorandum from D. S. Gabriel, SNPO to Director of Launch Vehicles and Propulsion Programs, June 7, 1968.

IV-F-2. "Nuclear Stage Performance Data", BMI-NLVP-ICM-68-224, D. S. Edgecombe and R. R. Teeter, Battelle Memorial Institute, Columbus Laboratories, Columbus, Ohio, December 5, 1968.

IV-F-3. "Nuclear Stage for Unmanned Planetary Probes", Missiles and Space Systems Division, Douglas Aircraft Company, 1967.

IV-F-4. "Data on Advanced Non-Chemical Propulsion for Inclusion in Estimating Factors Book for OSSA Prospectus 1969", BMI-NLVP-IM-68-16, D. S. Edgecombe, Battelle Memorial Institute, Columbus Laboratories, Columbus, Ohio, August 20, 1968.

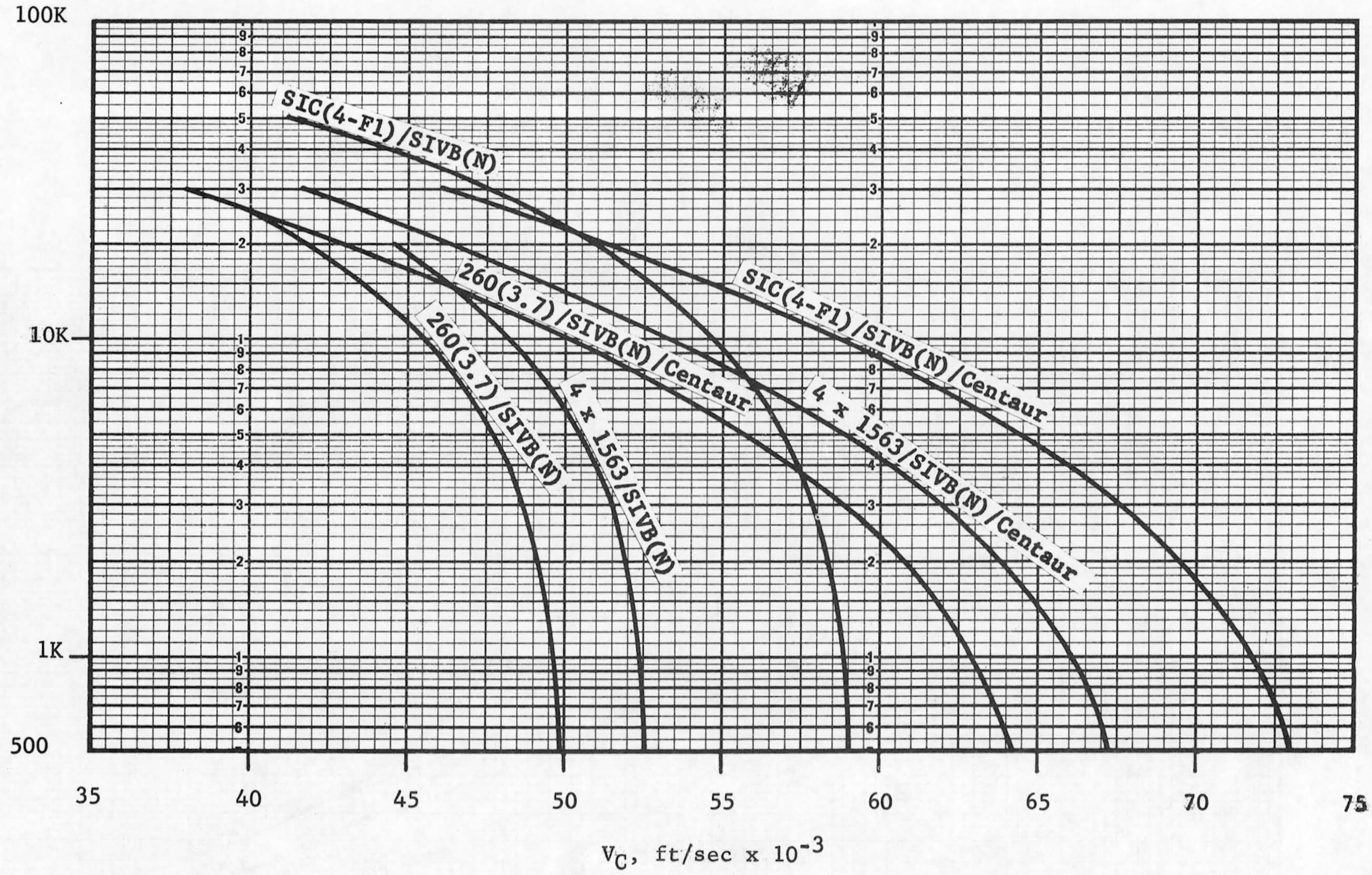


FIGURE IV-F-1. CAPABILITY OF SEVERAL POSSIBLE LAUNCH VEHICLES USING A NUCLEAR STAGE IN PLACE OF THE SIVB

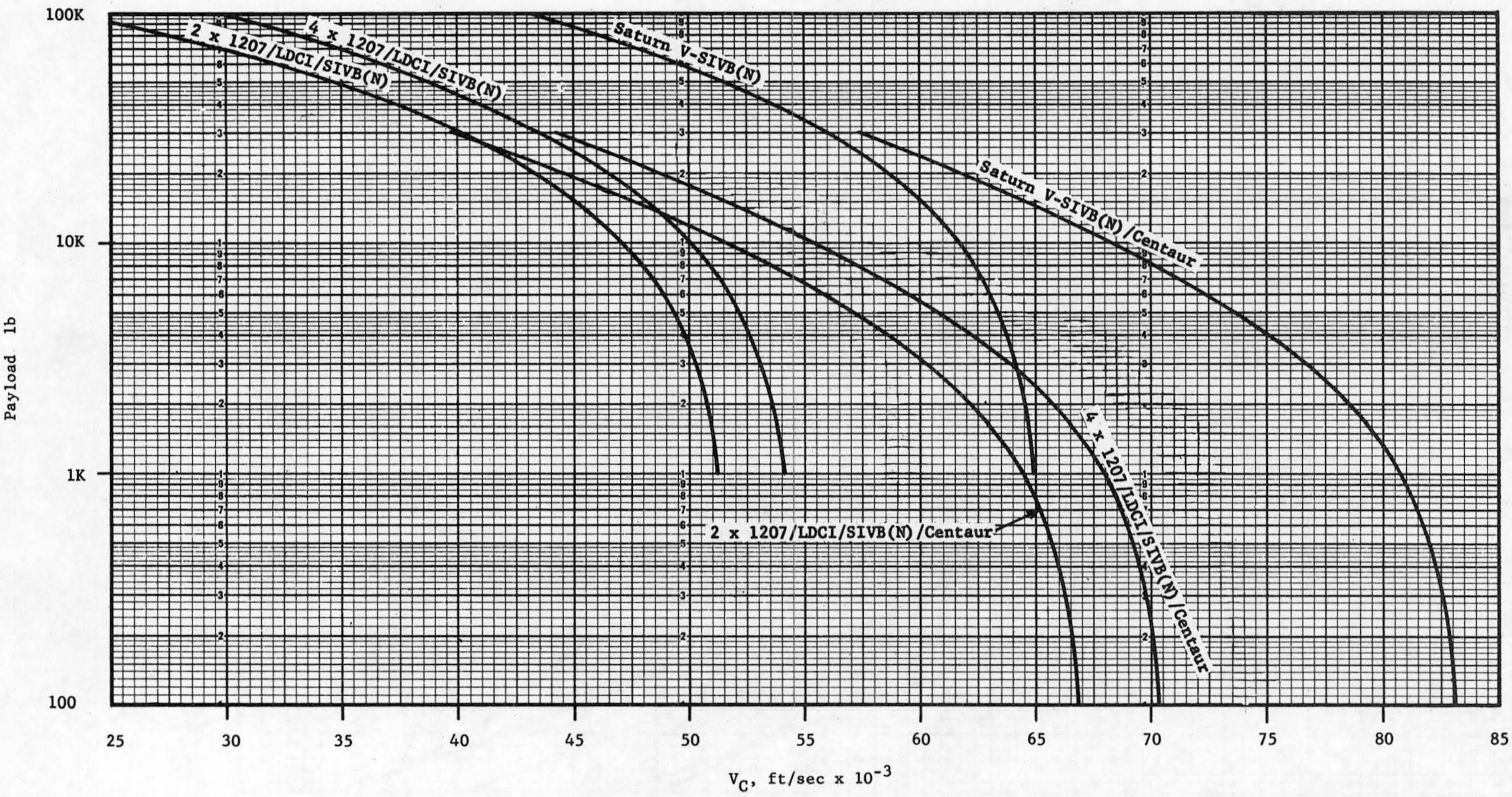


FIGURE IV-F-2. CAPABILITY OF SEVERAL POSSIBLE LAUNCH VEHICLES USING A NUCLEAR STAGE IN PLACE OF THE SIVB

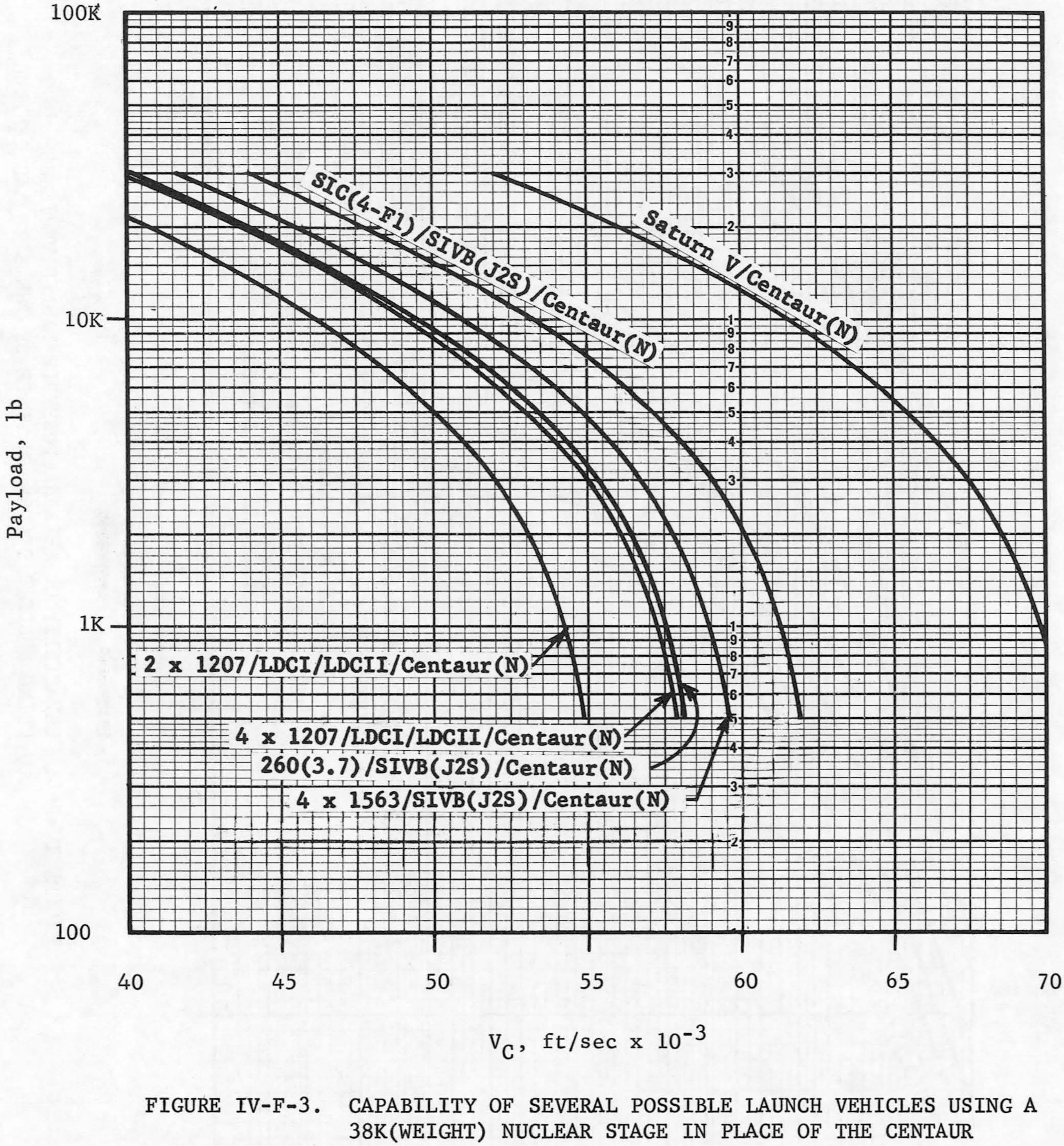


FIGURE IV-F-3. CAPABILITY OF SEVERAL POSSIBLE LAUNCH VEHICLES USING A 38K(WEIGHT) NUCLEAR STAGE IN PLACE OF THE CENTAUR

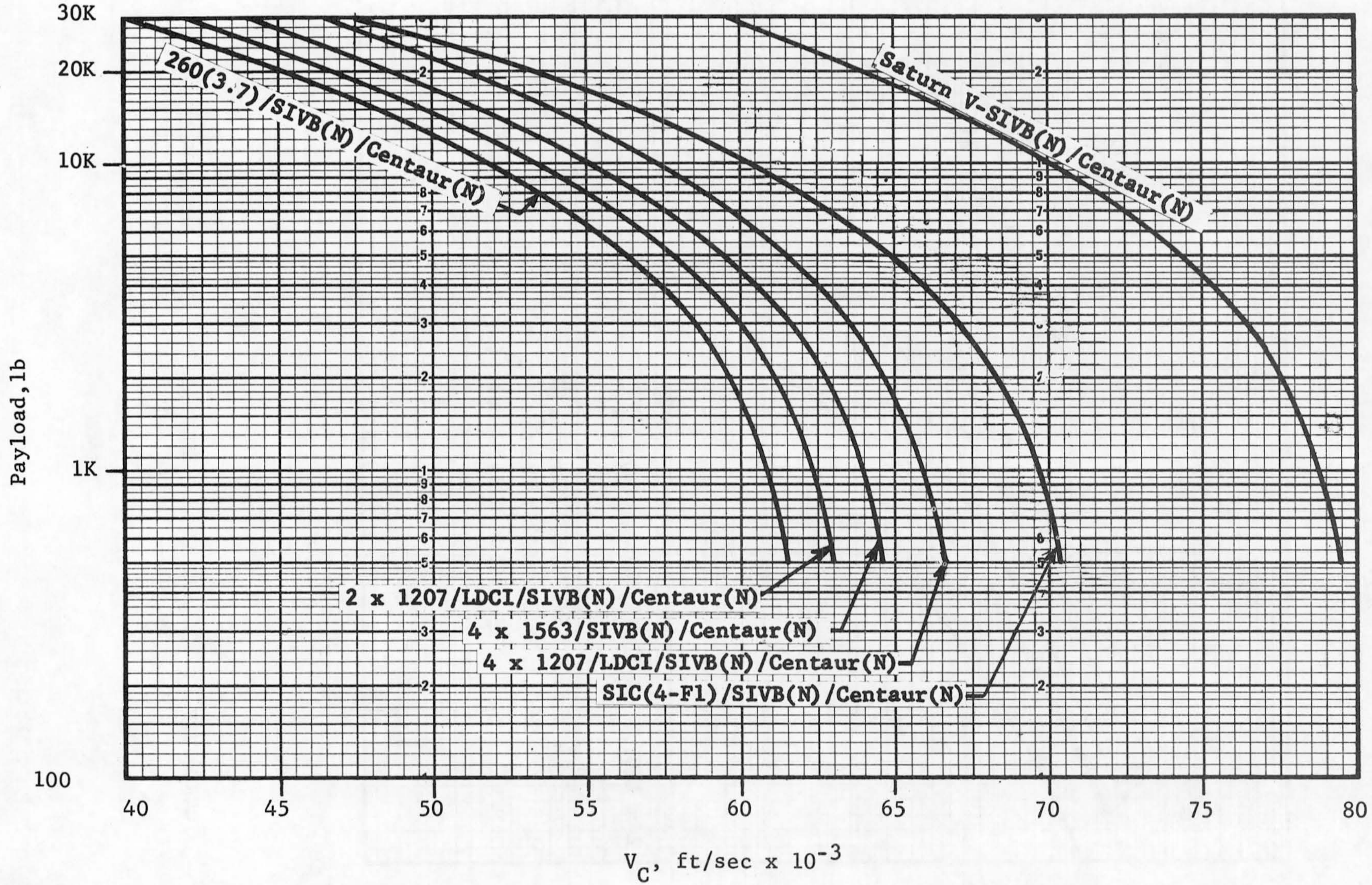


FIGURE IV-F-4. CAPABILITY OF SEVERAL POSSIBLE LAUNCH VEHICLES USING NUCLEAR STAGES IN PLACE OF THE SIVB AND CENTAUR

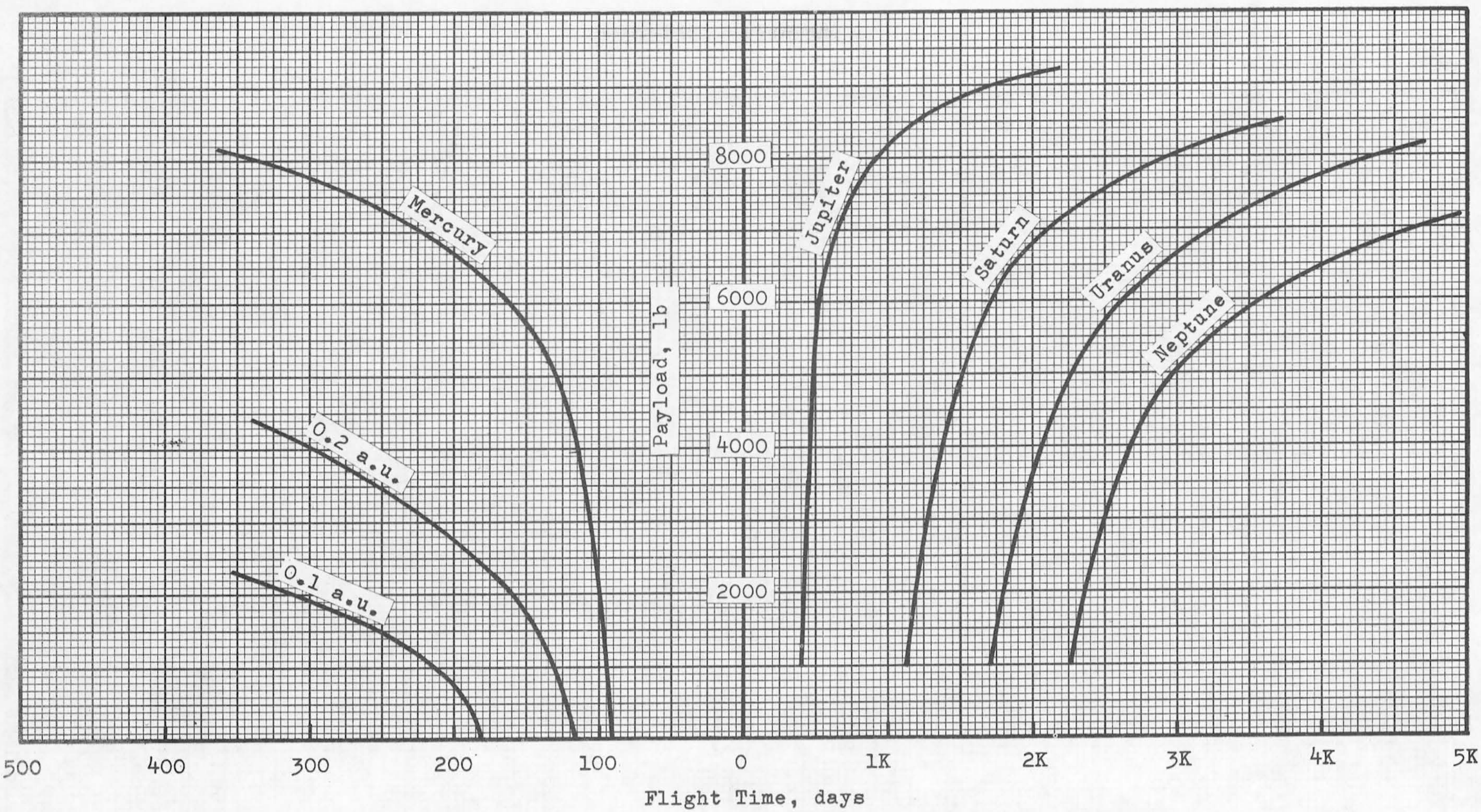


FIGURE IV-F-5. CAPABILITY OF THE TITAN IIID(7)/CENTAUR/NUCLEAR-ELECTRIC FOR DISTANT PLANETARY FLYBYS AND SOLAR PROBES

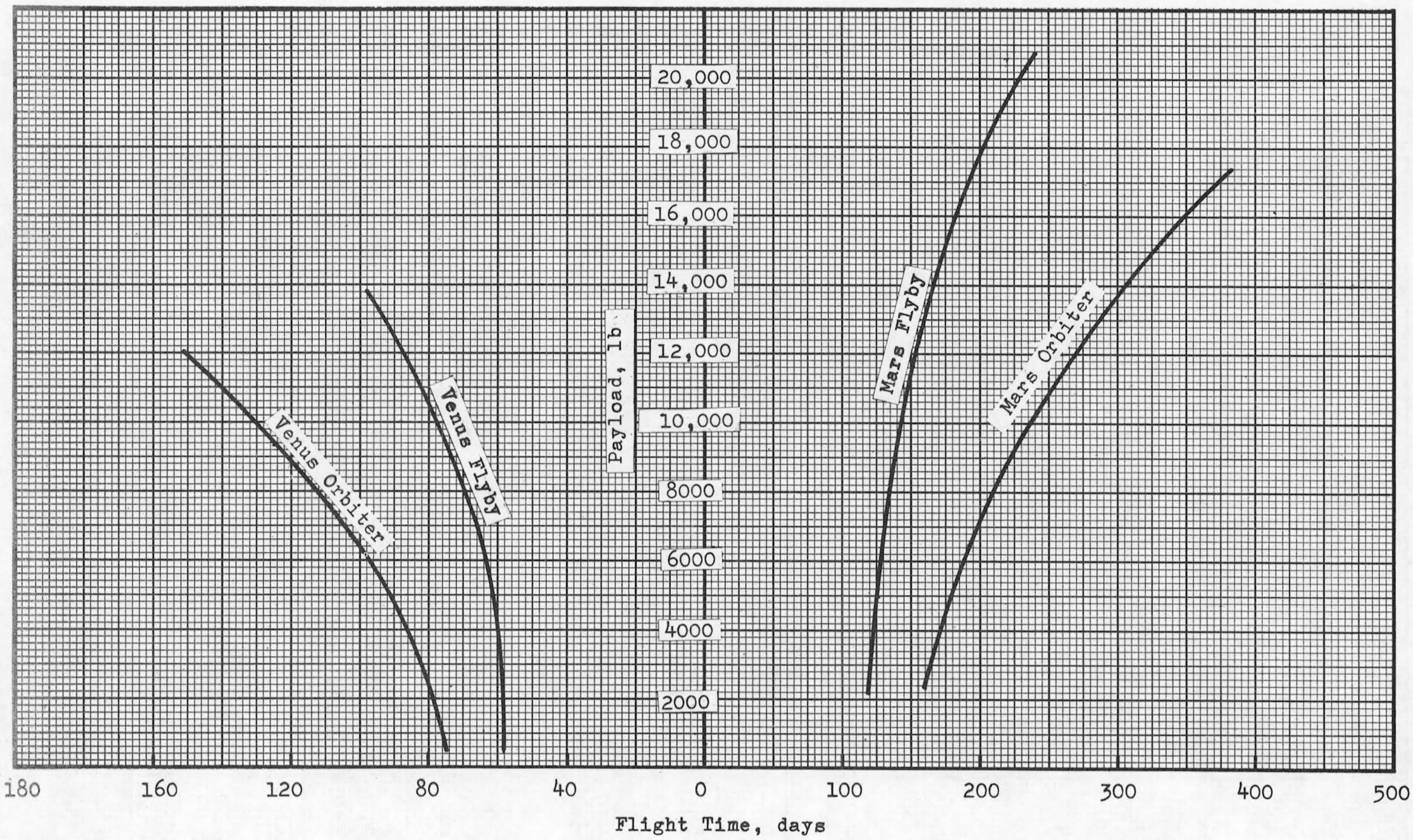


FIGURE IV-F-6. CAPABILITY OF THE TITAN IIID(7)/CENTAUR/NUCLEAR-ELECTRIC FOR NEAR PLANETARY FLYBY AND ORBITER MISSIONS

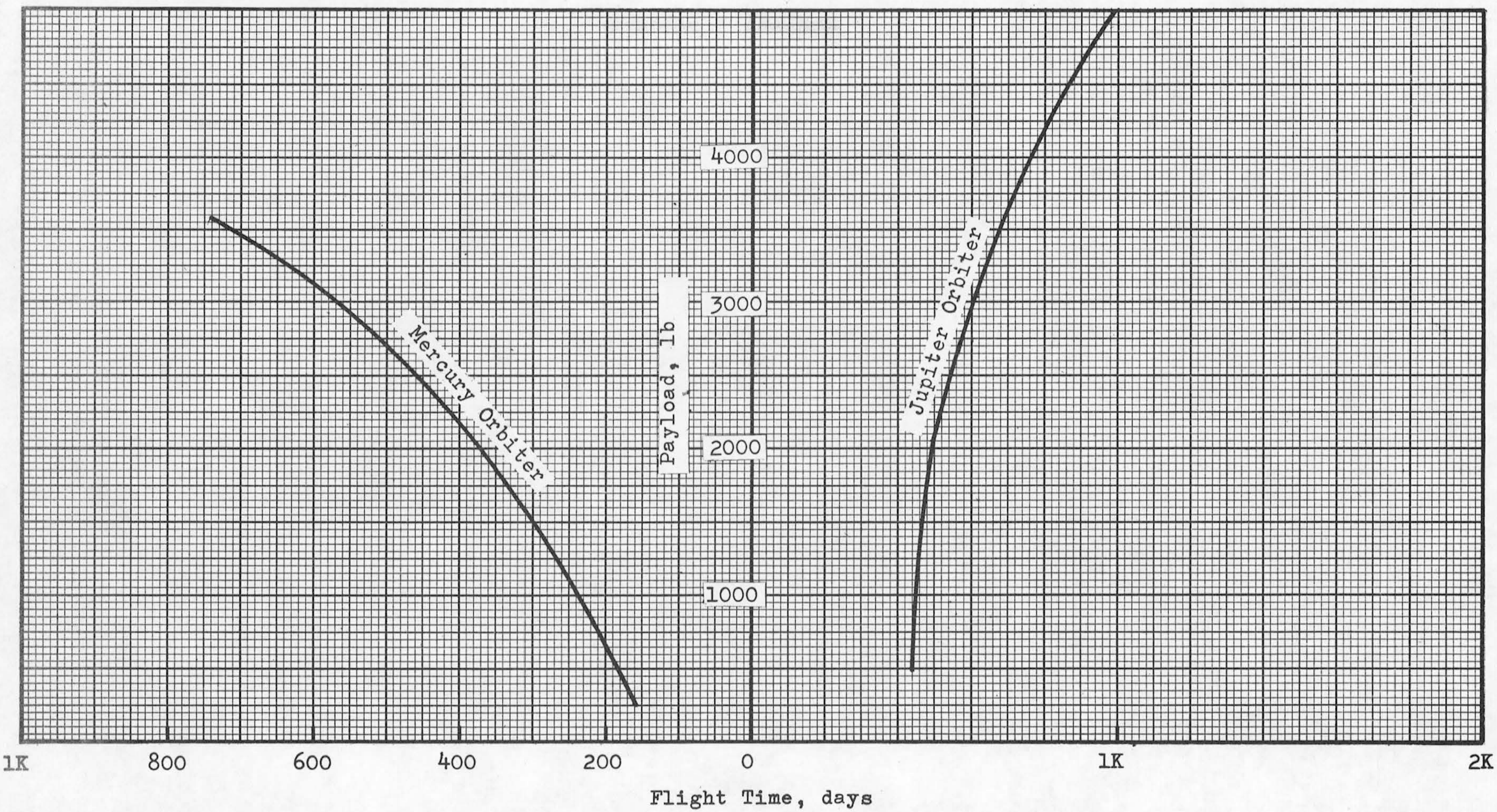


FIGURE IV-F-7. CAPABILITY OF THE TITAN IIID(7)/CENTAUR/NUCLEAR-ELECTRIC FOR DISTANT PLANETARY ORBITER MISSIONS

SECTION IV-G

RETRO-PROPULSION AND APOGEE KICK MOTOR SYSTEM DATA

SECTION IV-G

RETRO-PROPULSION AND APOGEE KICK MOTOR SYSTEM DATA

This section provides information for making planning estimates of system weights for retro-propulsion to enter planetary orbits, for apogee kick stages to enter a prescribed Earth orbit, and for other similar applications. The information contained in Figures IV-G-1, IV-G-1, IV-G-2, IV-G-3, and IV-G-4 can be used with information derived from Figures II-27a and II-27b in Section II to estimate the size of the retro-propulsion system for establishing planetary orbits. It also can be used with Figures II-30, 31a, 31b, 31c, 32a, and 32b to estimate the size of propulsion systems for satisfying requirements for transfer apogee impulse, Earth orbital plane change velocity increment, or a combination of these.

To use the Figures of section IV-G the following information is required.

ΔV - the required velocity increment, ft/sec

I - the specific impulse of the selected propellant system, sec

W_L - the spacecraft weight exclusive of the propulsion system, lb.

irst, the ratio $\Delta V/Ig_o$ is calculated, where g_o is 32.17 ft/sec^2. Then, interpolating etween the lines of constant $\Delta V/Ig_o$ as required, the weight of the required propulsion ystem (W_{RP}) can be obtained.

The curves are based on specific empirical relationships between size of a proulsion system and its mass fraction, or ratio of propellant weight to total propulsion ystem weight. Among other things, the mass fraction depends on the division of functions etween the spacecraft and the propulsion system, the acceleration level chosen or perissible for the maneuver*, and detailed mission-related factors. The relationships used n Figures IV-G-1, IV-G-2, and IV-G-3 are for liquid propulsion systems, and it is assumed hat the systems include a thrust vector control system and basic load-carrying structure

* Thrust equal to propellant weight was used in Figures IV-G-1, IV-G-2, and IV-G-3. This choice leads to reasonable values of the burn time for the types of maneuver being considered here. Some variation in thrust/weight is permissible without significant changes in the data shown.

but do not include guidance, power, or control electronics systems. Figure IV-G-1 presents information for Earth storable propellants; Figures IV-G-2 and IV-G-3 present similar data for systems using mildly cryogenic space storable propellants and deep cryogenic propellants respectively. Propulsion system weights given in Figure IV-G-4 are for application of solid propellant motors where the spacecraft is assumed to provide all structure (other than motor case) and control functions, as for example with apogee Kick motors in spin-stabilized spacecraft.

Liquid propulsion systems employing Earth storable propellants, such as N_2O_4/Aerozine 50, yield specific impulses in the range of 300 to 325 sec (Figure IV-G-1). The mildly cryogenic space storable propellants, such as OF_2/B_2H_6 or Flox/CH_4, produce specific impulses in the range of 390 to 410 sec (Figure IV-G-2). Deep cryogenic propellants, such as H_2/F_2, produce specific impulses in the range of 455 to 465 sec (Figure IV-G-3). The specific impulse may be taken as 285 to 290 sec for current solid propellant propulsion systems and up to 310 sec for future solid propellant systems (Figure IV-G-5). These may be considered as conservative and optimistic limits in using data from the figures in this section.

If the required velocity increments are large, consideration should be given to the use of two or more propulsion stages, especially when the total propulsion system weight becomes large compared to the spacecraft weight. There is obviously a trade-off between system weight and system complexity. Multistage propulsion systems should be considered when the propulsion-system weight exceeds that of the spacecraft by a factor of about 10, shown by the region above the broken lines in the figures. As a first approximation, the velocity increment may be divided equally among the propulsion stages, using the minimum number of stages required to maintain the ratio of the final, or smallest, stage weight to spacecraft weight below about 10. Each stage can then be sized from the appropriate figure, if it is remembered that, for the lower stages, W_L must include not only the spacecraft weight but all upper stages as well.

The propulsion system weight determined by the methods of this subsection should be added to the spacecraft weight exclusive of the propulsion system to obtain the total spacecraft weight. This total spacecraft weight is the launch vehicle payload to be used with the figures of Section IV-A and with those systems in Section IV-F which do not involve nuclear-electric spacecraft propulsion.

Because of the many factors which influence the choice of propellants and the design and performance of spacecraft propulsion systems, the user is cautioned that propulsion system weights obtained by the above procedures should be used only as estimates for planning purposes. Should assistance be needed, the user may contact the persons listed in the Foreword of this document.

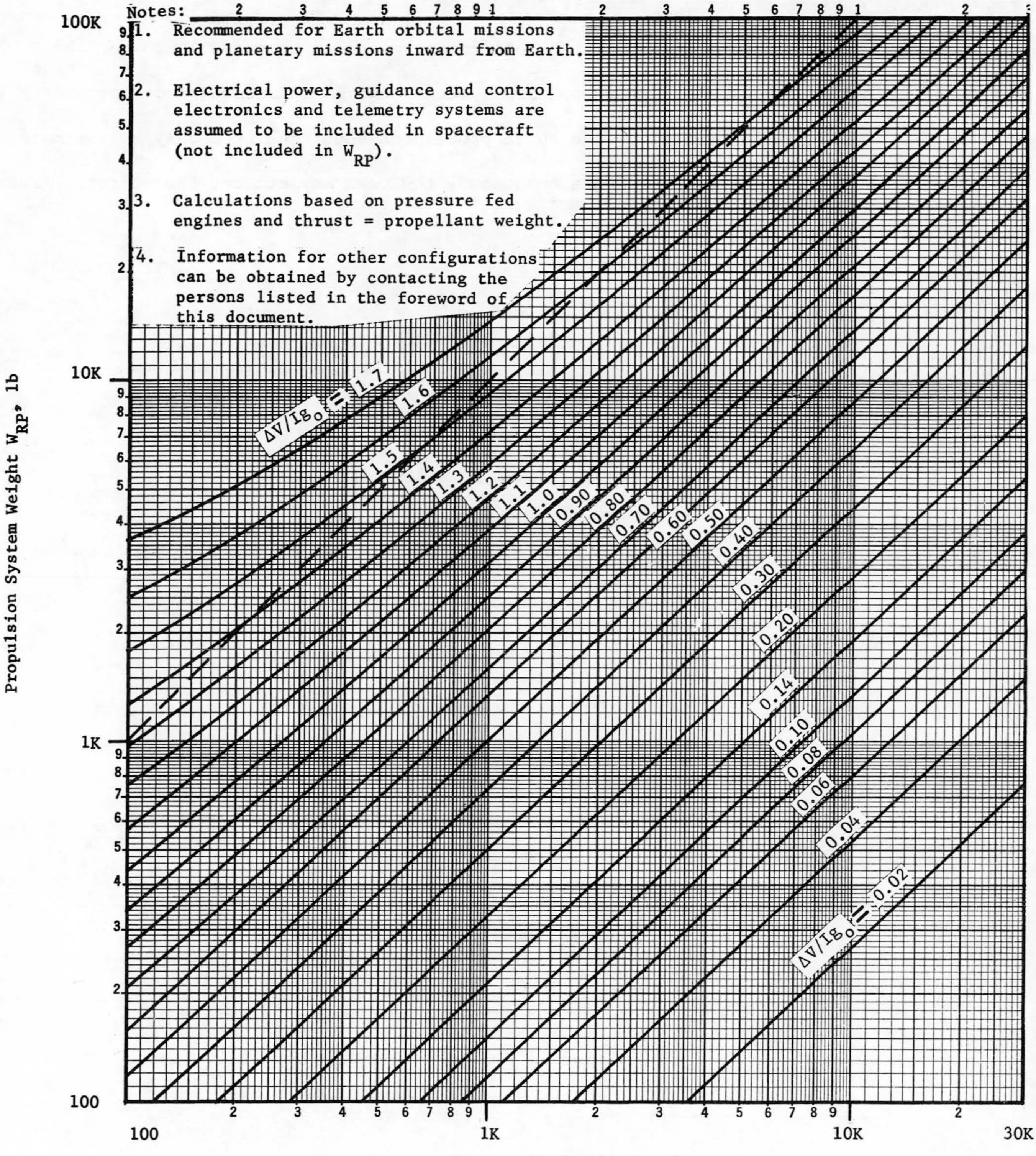


FIGURE IV-G-1 SPACECRAFT PROPULSION SYSTEM WEIGHT USING EARTH STORABLE PROPELLANTS

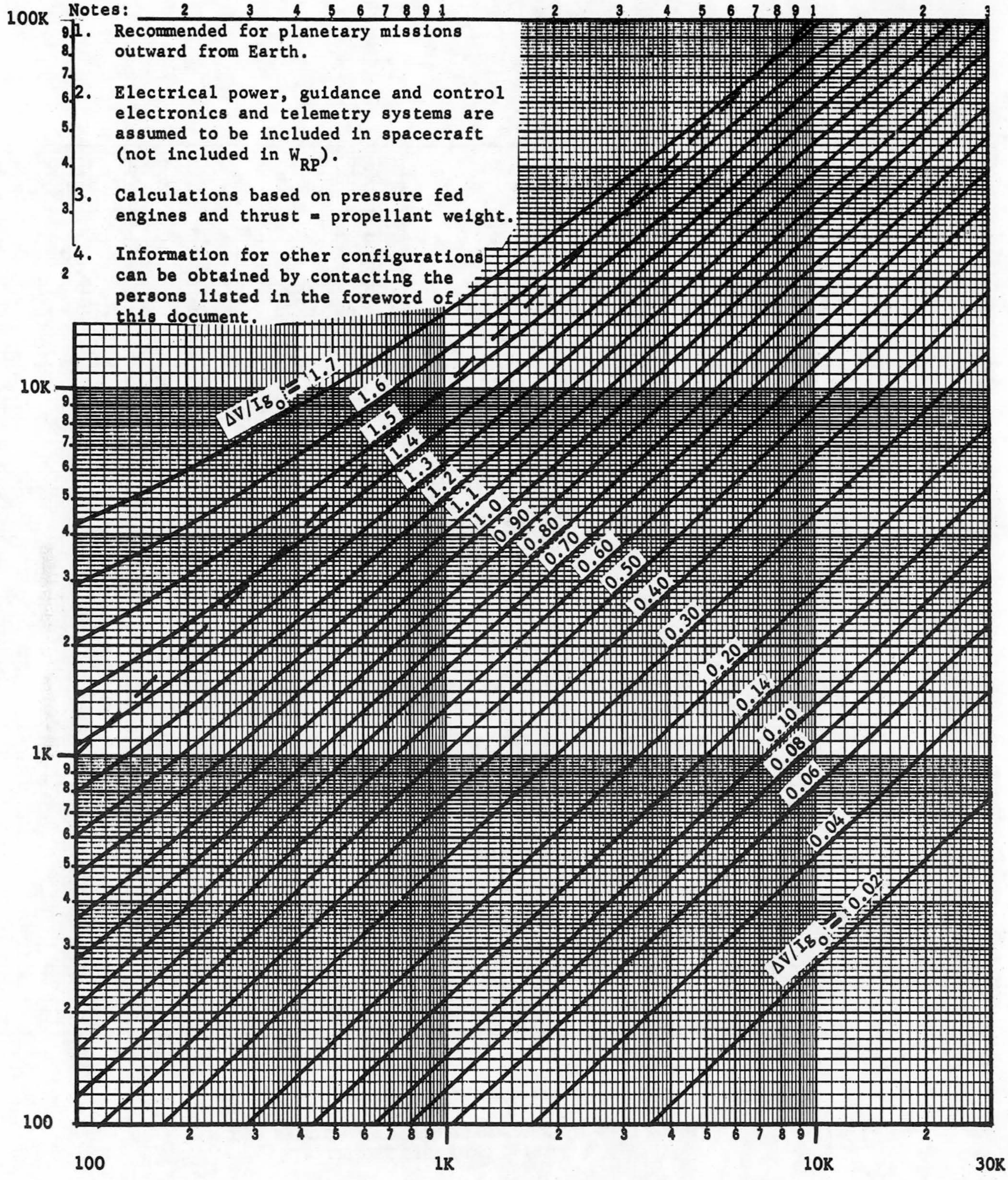


FIGURE IV-G-2. SPACECRAFT PROPULSION SYSTEM WEIGHT USING SPACE STORABLE PROPELLANTS

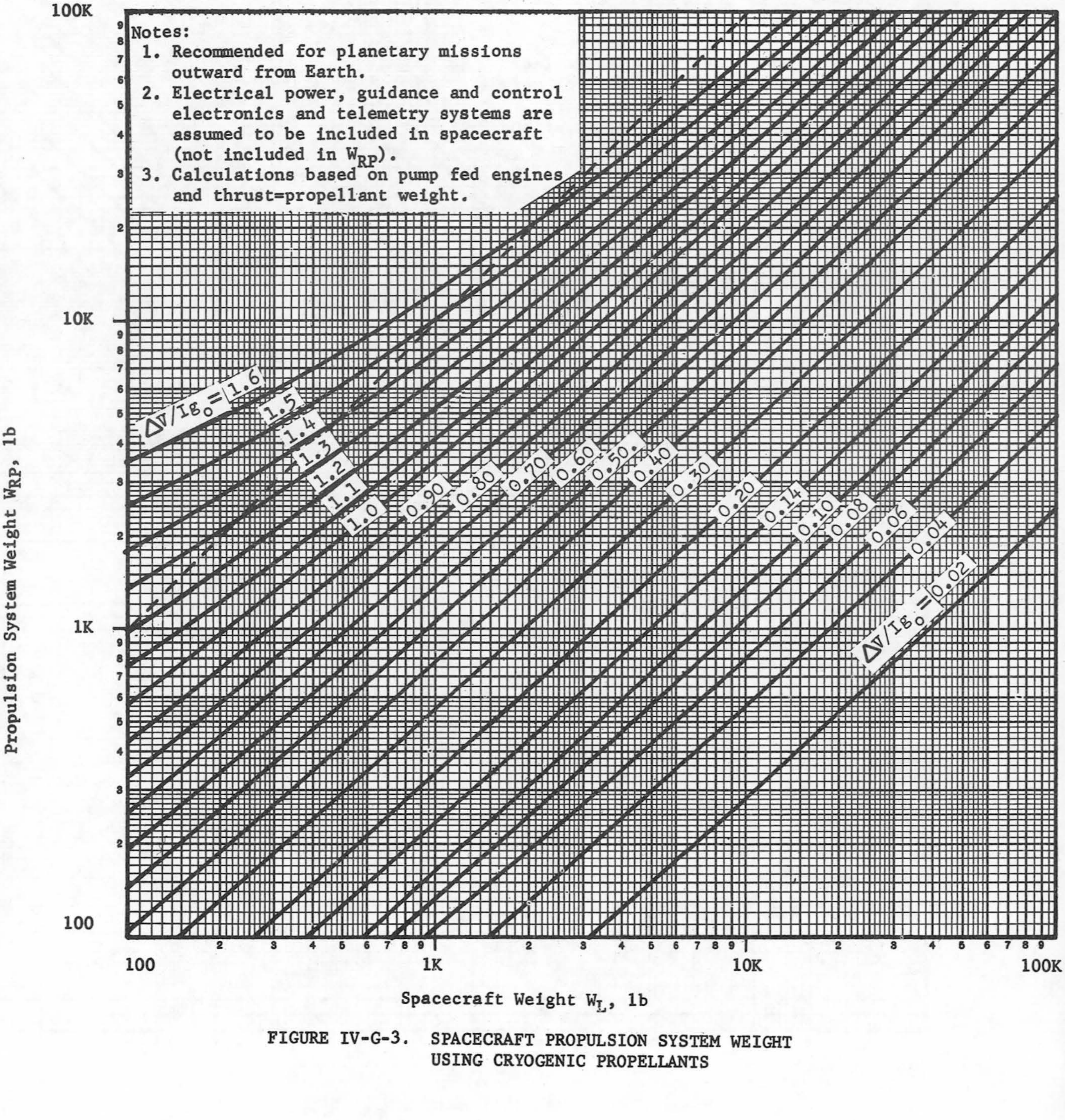


FIGURE IV-G-3. SPACECRAFT PROPULSION SYSTEM WEIGHT USING CRYOGENIC PROPELLANTS

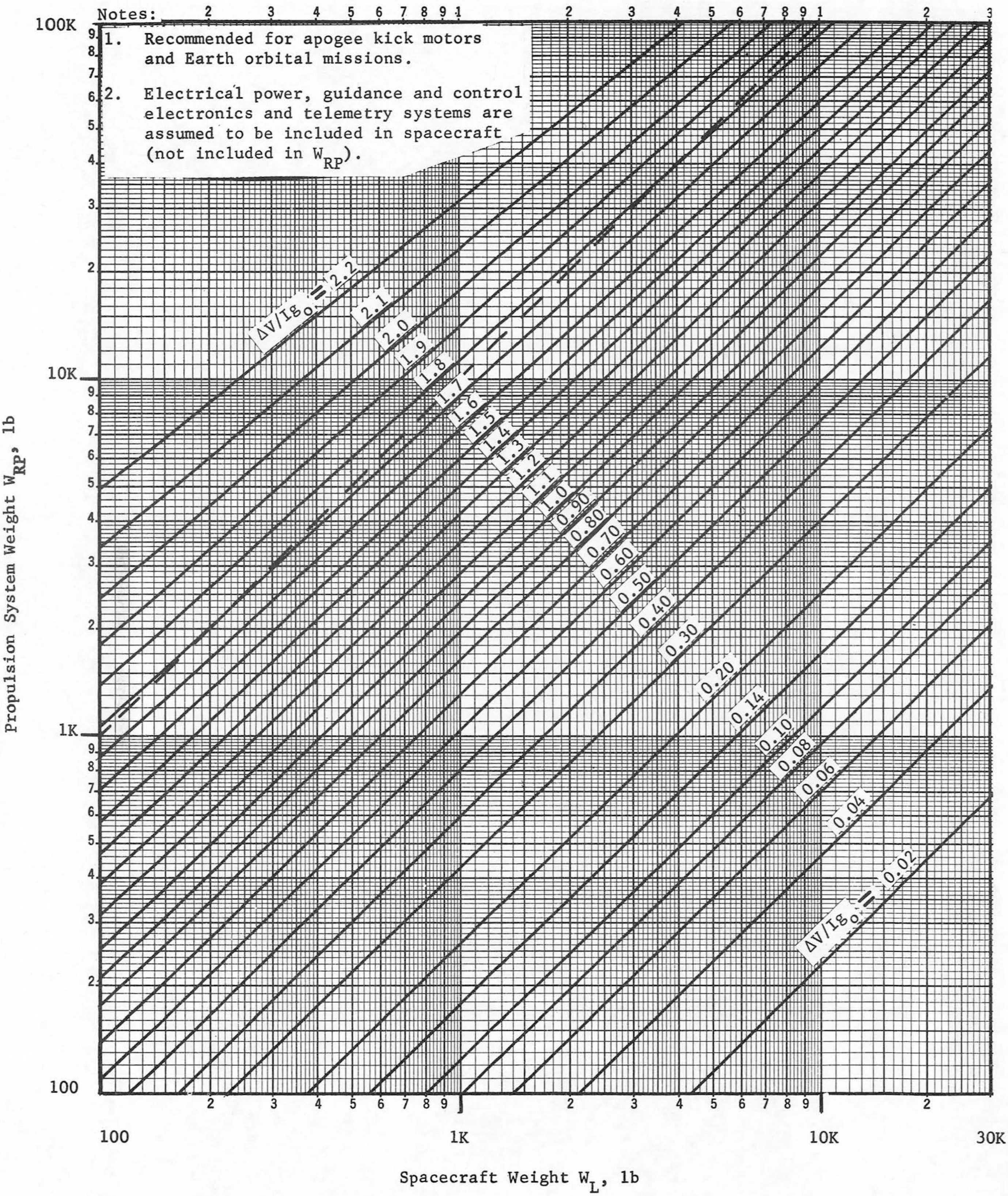


FIGURE IV-G-4. SPACECRAFT PROPULSION SYSTEM WEIGHT USING SOLID PROPELLANTS

SECTION V

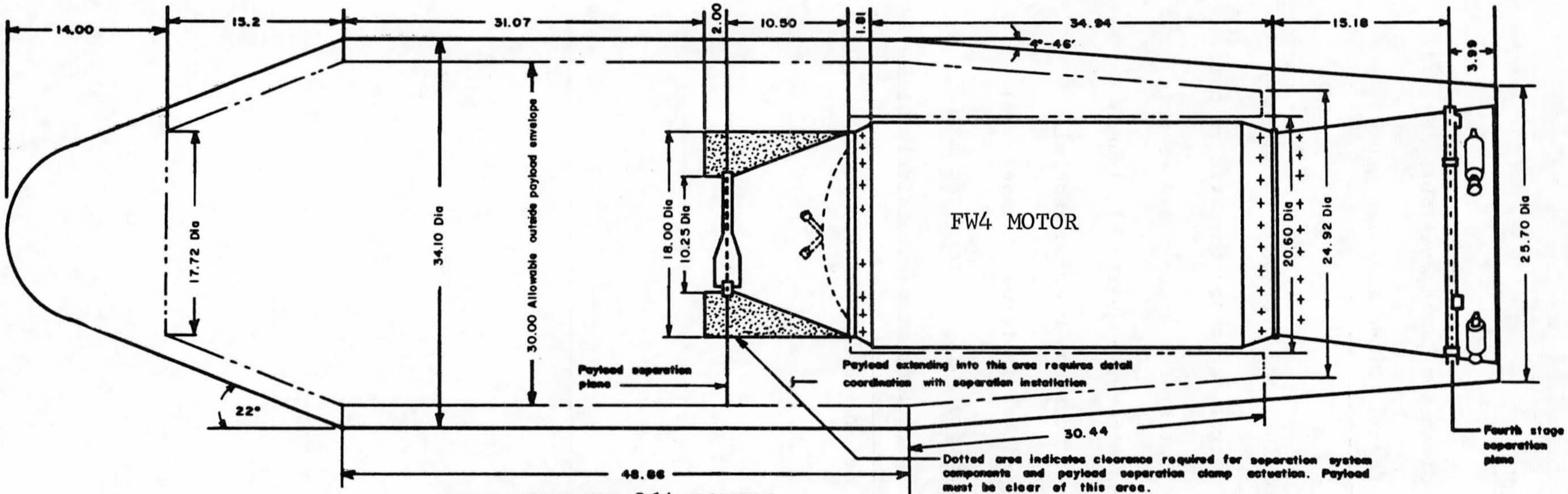
SHROUD CONFIGURATIONS

SECTION V

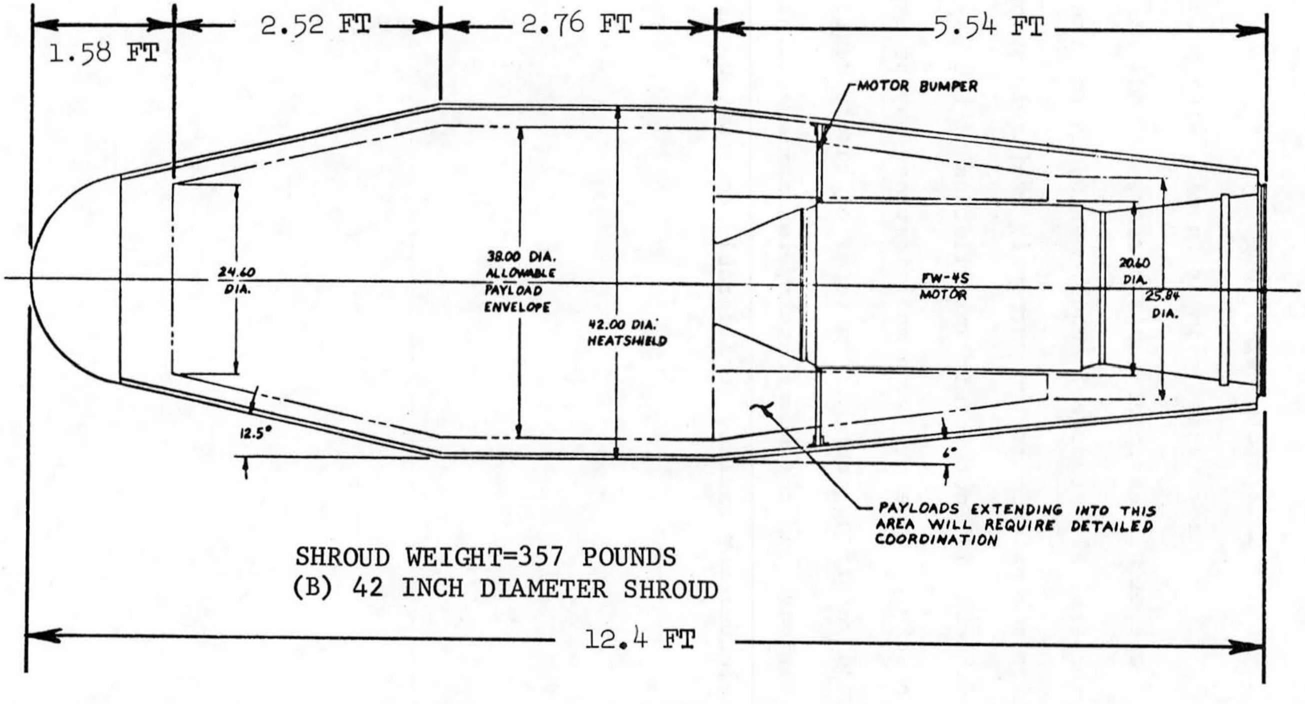
SHROUD CONFIGURATIONS

This section presents line drawings, nominal dimensions, and available payload volumes of shrouds for some launch vehicles and upper stages. Dimensions are given in inches unless otherwise noted.

The mission planner should, when possible, consider and provide data on the shape of his payload. This factor, as well as payload weight and characteristic velocity (or orbital parameter) requirements, is important in selecting the appropriate vehicle for a mission. If the currently available shroud for an otherwise applicable launch vehicle will not permit installation of the payload, modifications are, in general, possible. However, such modifications may be expensive, either in terms of development and recurring costs or in terms of launch vehicle performance. If the required modifications lead to the necessity for hammerheading or configurations with excessively high slenderness ratios, considerable work may be needed to determine feasibility.

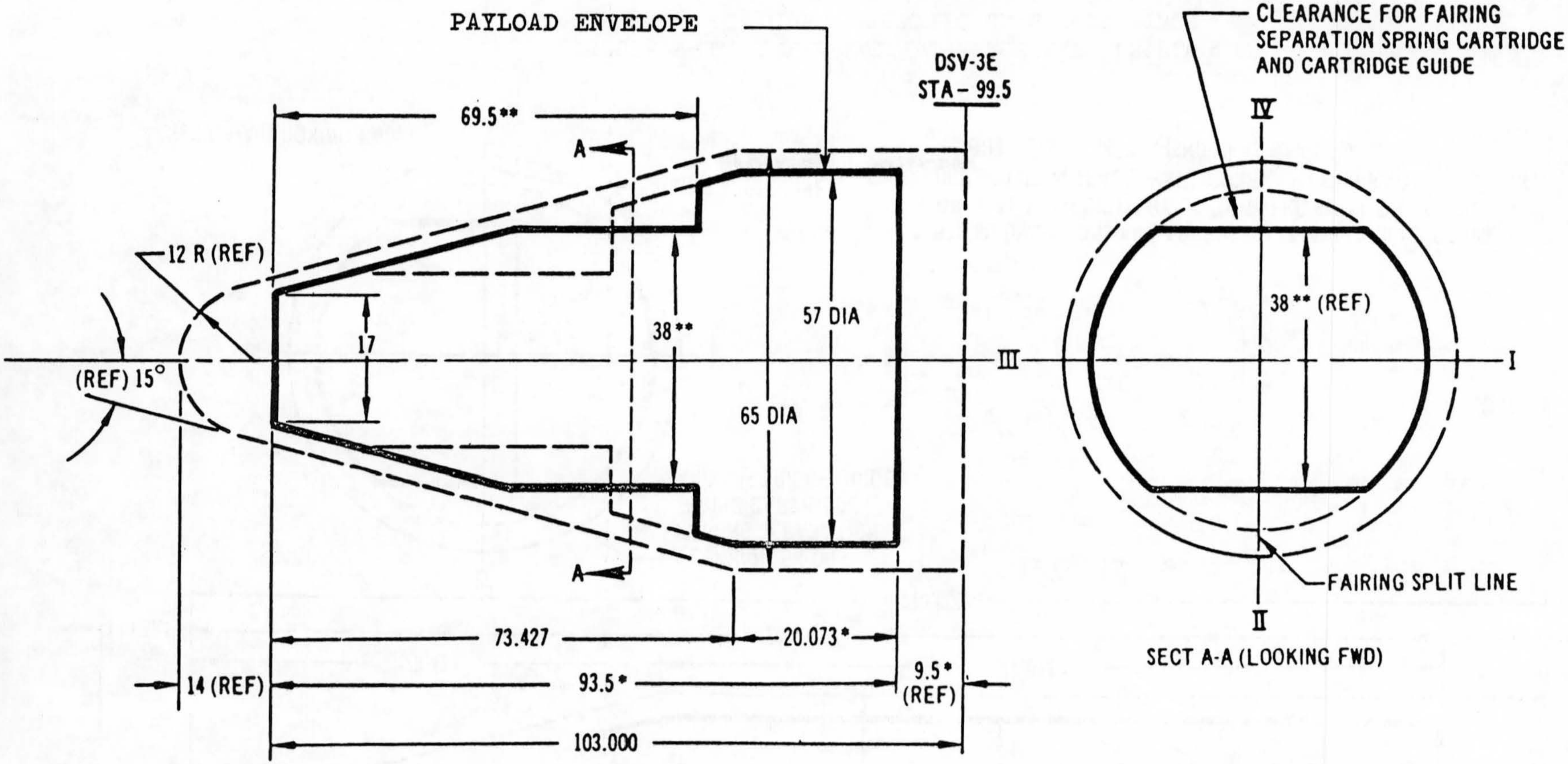


SHROUD WEIGHT=264 POUNDS
(A) 34 INCH DIAMETER SHROUD



SHROUD WEIGHT=357 POUNDS
(B) 42 INCH DIAMETER SHROUD

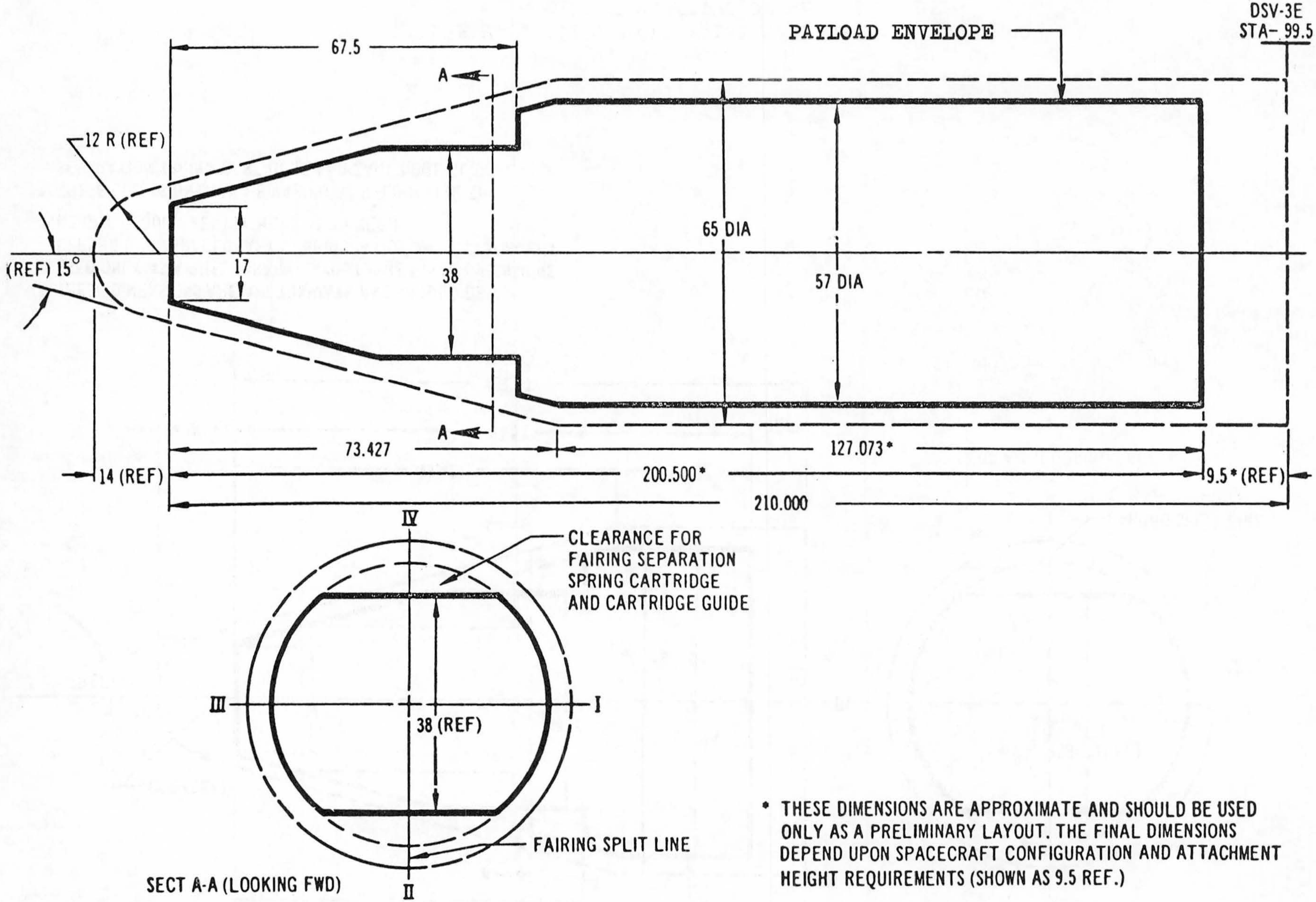
FIGURE V-1. SCOUT SHROUD



* THESE DIMENSIONS ARE APPROXIMATE AND SHOULD BE USED ONLY AS A PRELIMINARY LAYOUT. THE FINAL DIMENSIONS DEPEND UPON SPACECRAFT CONFIGURATION AND ATTACHMENT HEIGHT REQUIREMENTS (SHOWN AS 9.5 REF.)

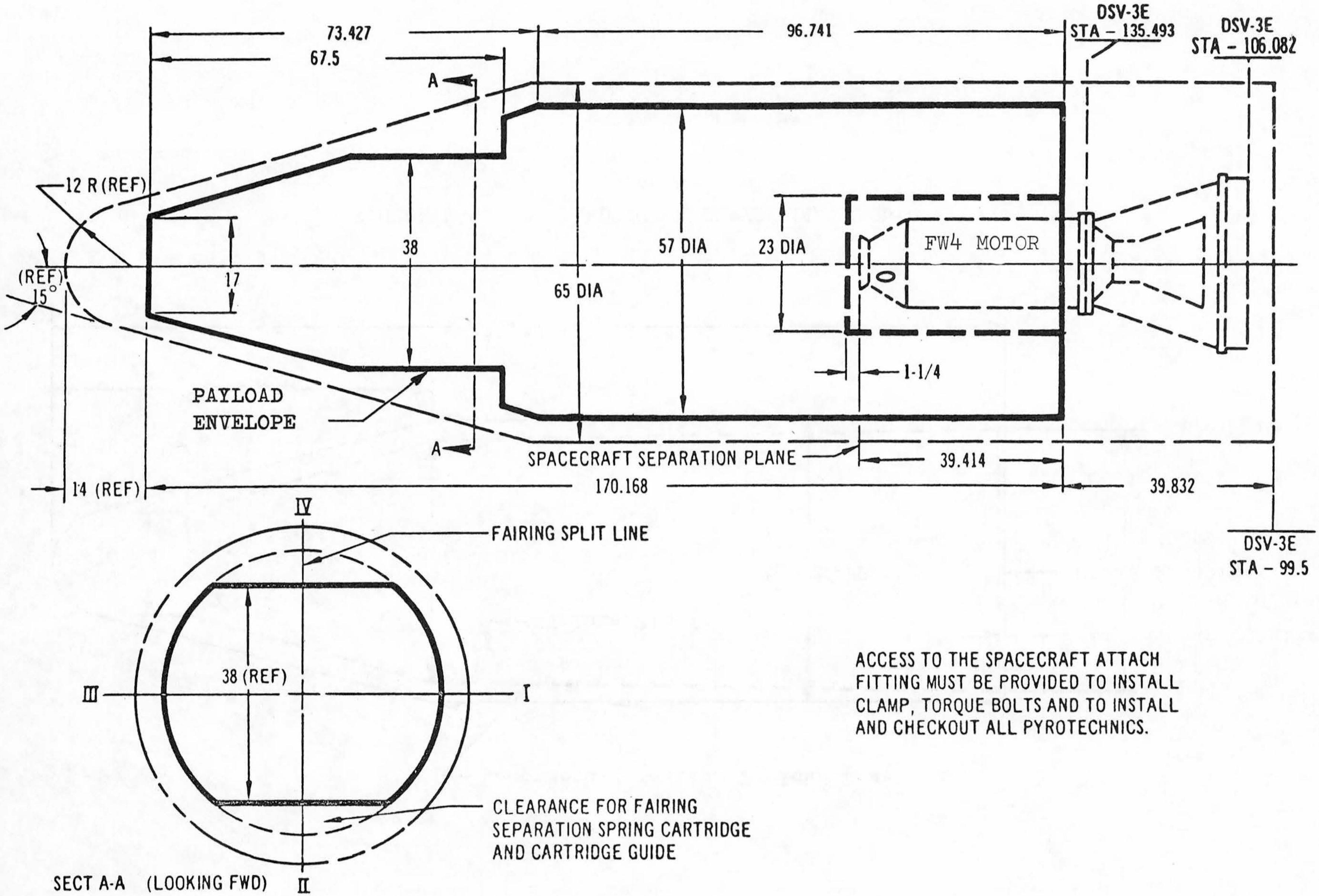
** THESE DIMENSIONS MAY BE VARIED BY RELOCATING THE SEPARATION SPRING CARTRIDGE AND CARTRIDGE GUIDE.

FIGURE V-2. DELTA A-12 FAIRING/SPACECRAFT ENVELOPE DIRECT MOUNTING ON SECOND STAGE



* THESE DIMENSIONS ARE APPROXIMATE AND SHOULD BE USED ONLY AS A PRELIMINARY LAYOUT. THE FINAL DIMENSIONS DEPEND UPON SPACECRAFT CONFIGURATION AND ATTACHMENT HEIGHT REQUIREMENTS (SHOWN AS 9.5 REF.)

FIGURE V-3. DELTA FAIRING/SPACECRAFT ENVELOPE
DIRECT MOUNTING ON SECOND STAGE



ACCESS TO THE SPACECRAFT ATTACH FITTING MUST BE PROVIDED TO INSTALL CLAMP, TORQUE BOLTS AND TO INSTALL AND CHECKOUT ALL PYROTECHNICS.

FIGURE V-4. DELTA FAIRING/SPACECRAFT ENVELOPE WITH THE FW4 THIRD STAGE MOTOR

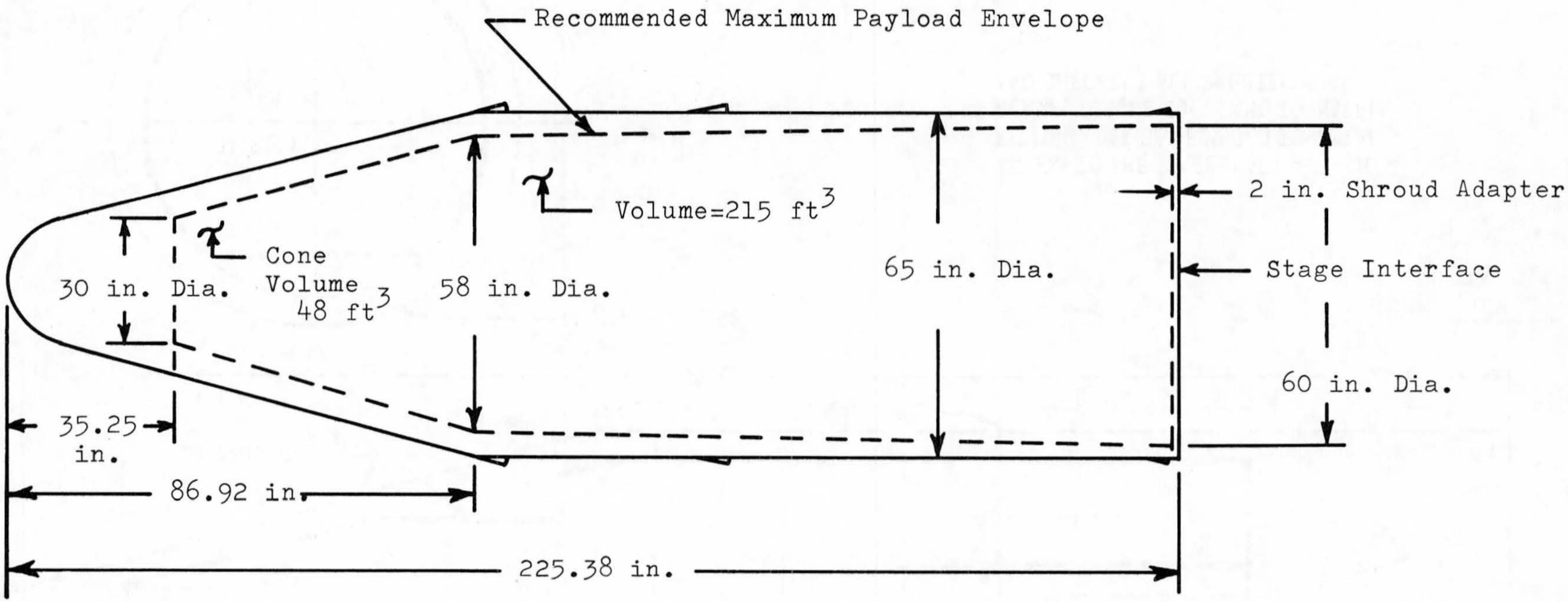


FIGURE V-5. STANDARD AGENA CLAMSHELL SHROUD

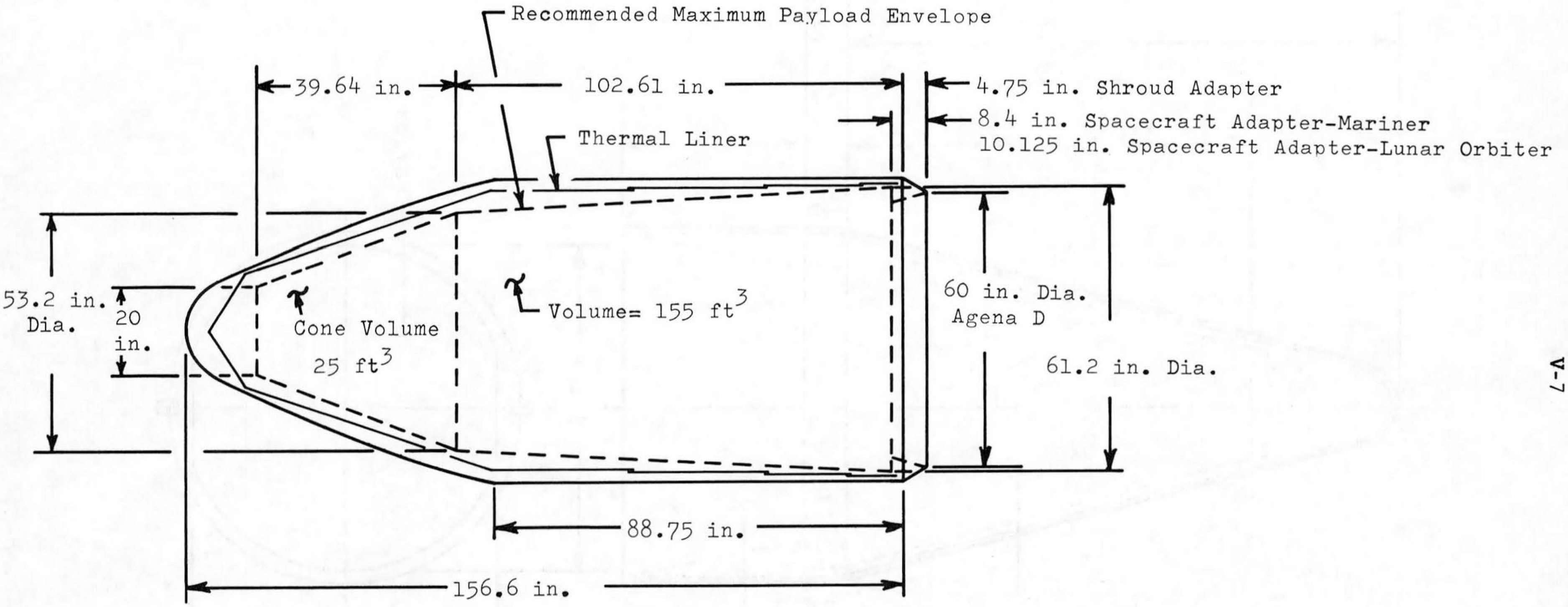


FIGURE V-6. AGENA MARINER C AND LUNAR ORBITER OVER-THE-NOSE SHROUD

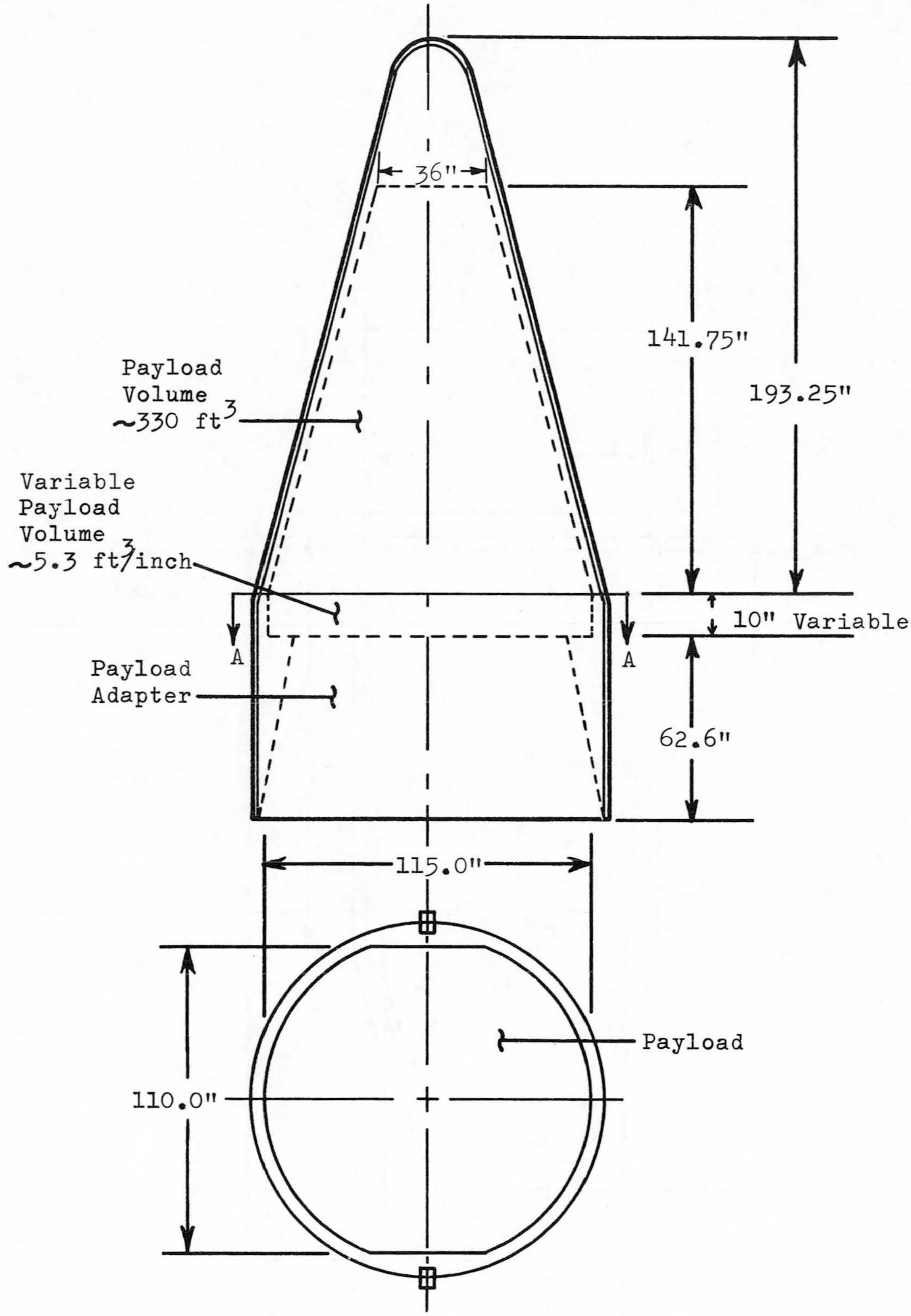


FIGURE V-7. BASIC CENTAUR NOSE FAIRING

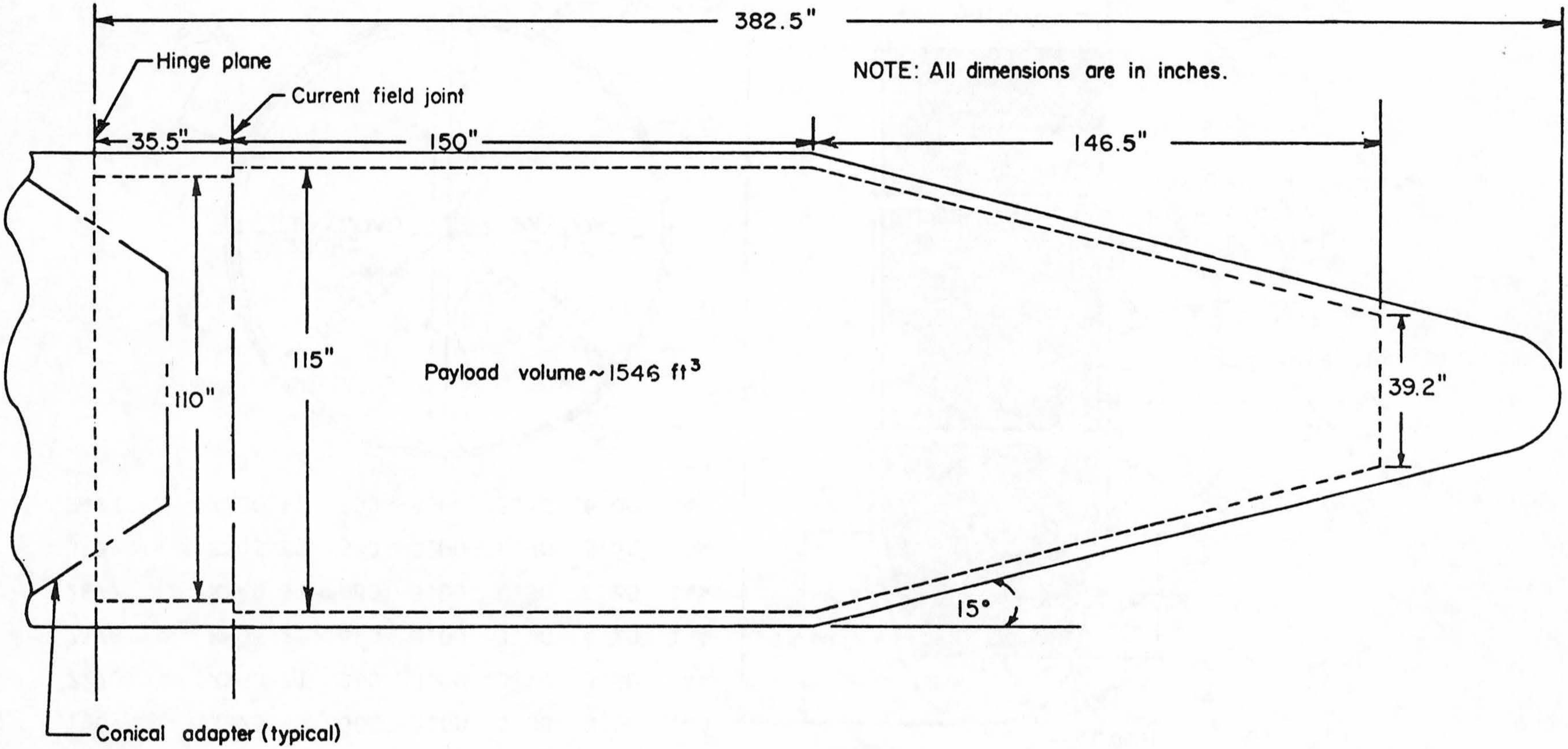
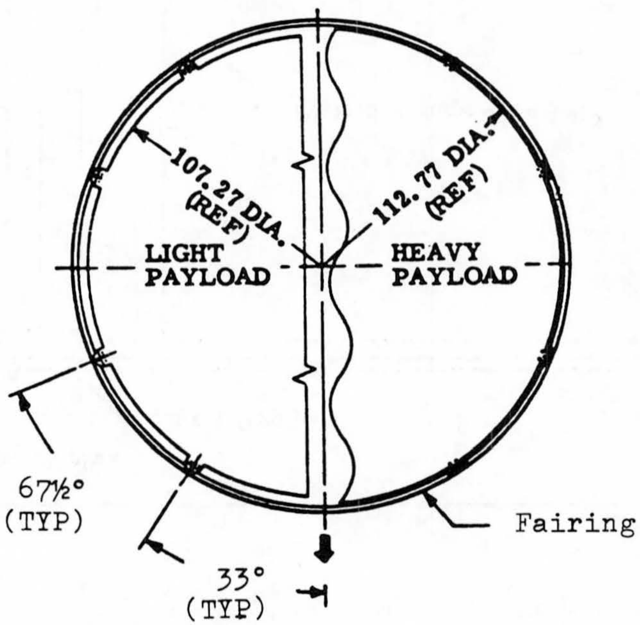


FIGURE V-8. CENTAUR OAO SHROUD

WEIGHT, LB	CONFIG NO.	L	A	B	C	D
1364	XX15 FT	180	3.00	3.00	3.00	84
1668	XX20 FT	240	3.00	3.00	3.00	144
1967	XX25 FT	300	3.00	3.00	3.00	144
2270	XX30 FT	360	3.00	3.10	3.60	144
2578	XX35 FT	420	3.00	3.70	4.20	144
2877	XX40 FT	480	3.00	4.30	4.80	144
3179	XX45 FT	540	3.00	4.90	5.40	144
3478	XX50 FT	600	3.00	5.50	6.00	144



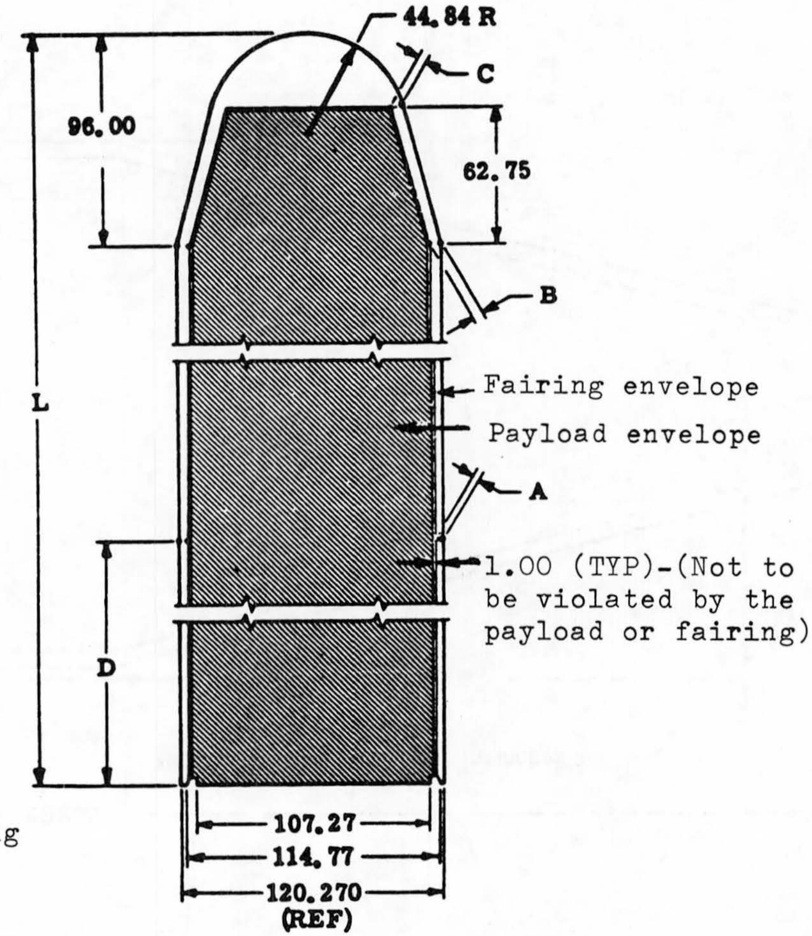
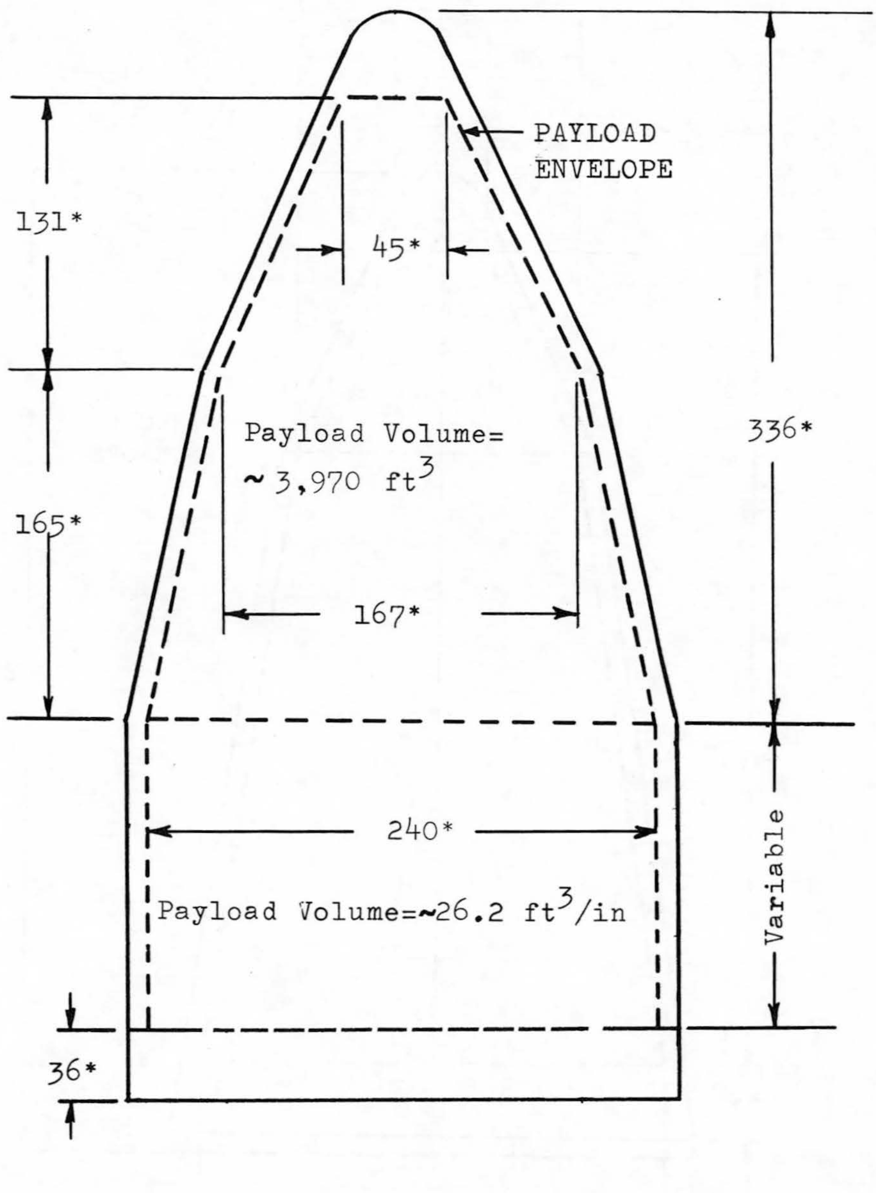
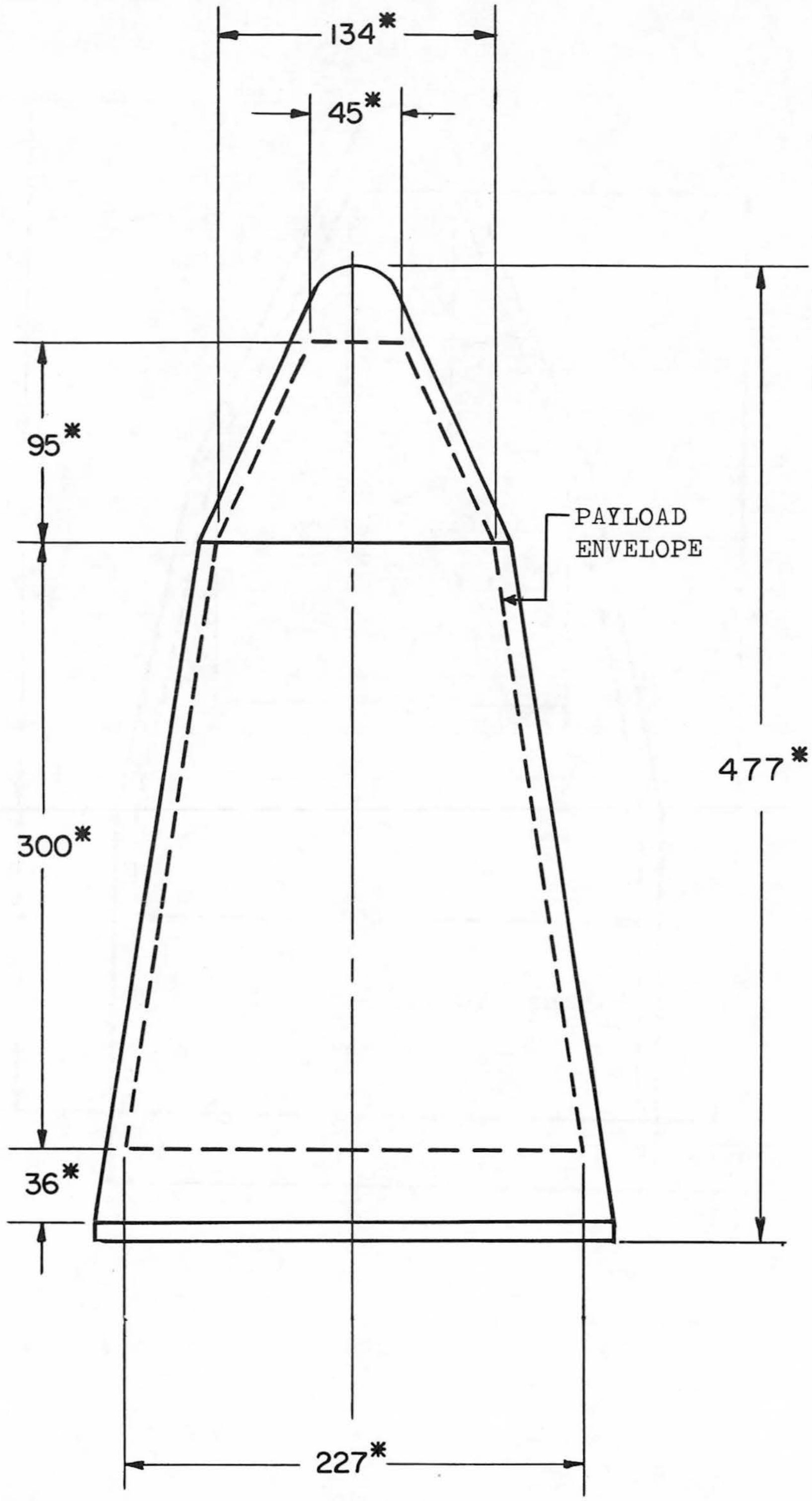


FIGURE V-9. TITAN UPLF (UNIVERSAL PAYLOAD FAIRING)



* Approximate dimensions to be used for preliminary layout only

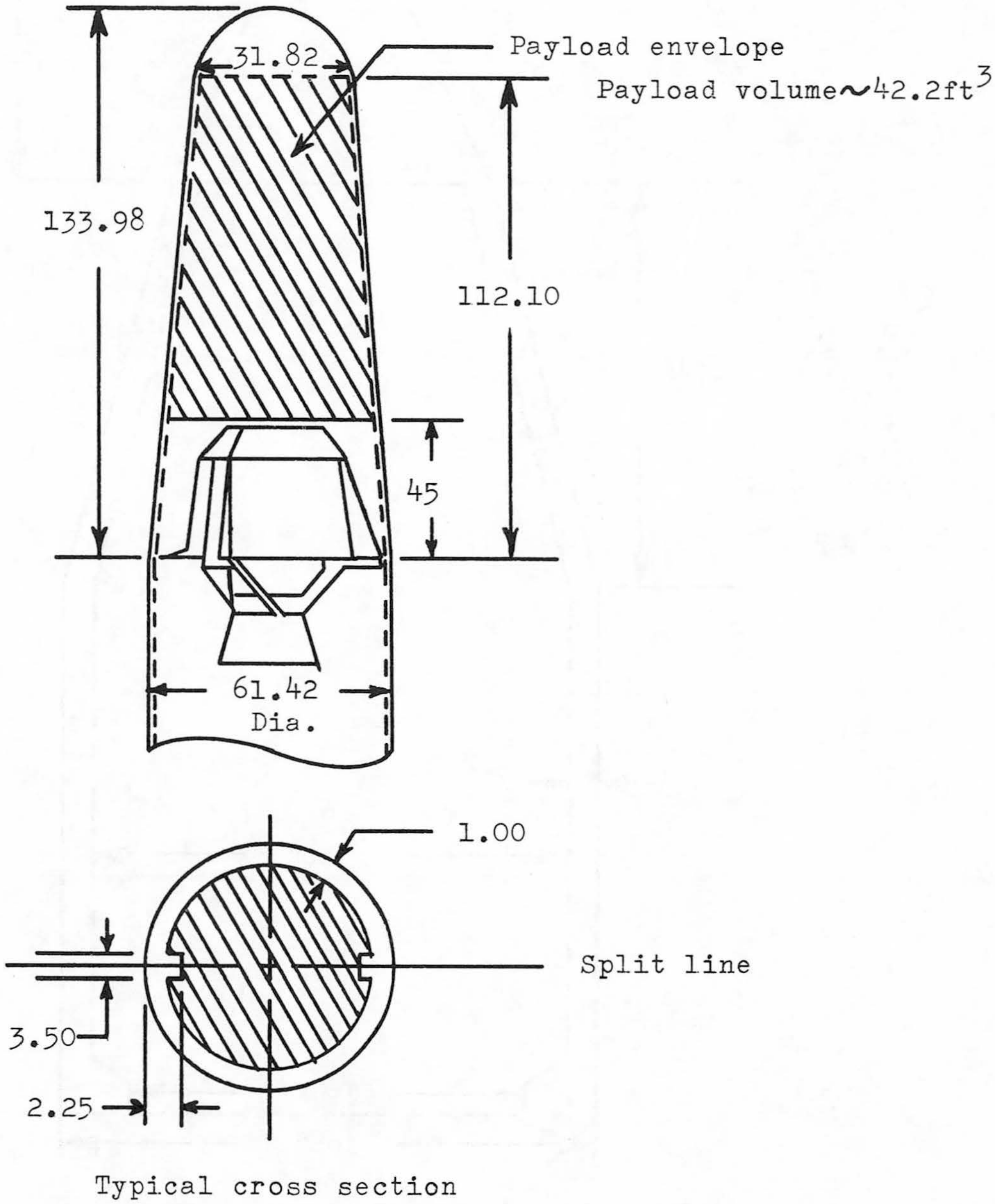
FIGURE V-10. SATURN IB AND SATURN V SHROUD (BASED ON VOYAGER NOSE CONE)



Basic payload volume is ~ 5000 ft^3

* Approximate dimensions to be used for preliminary layout only

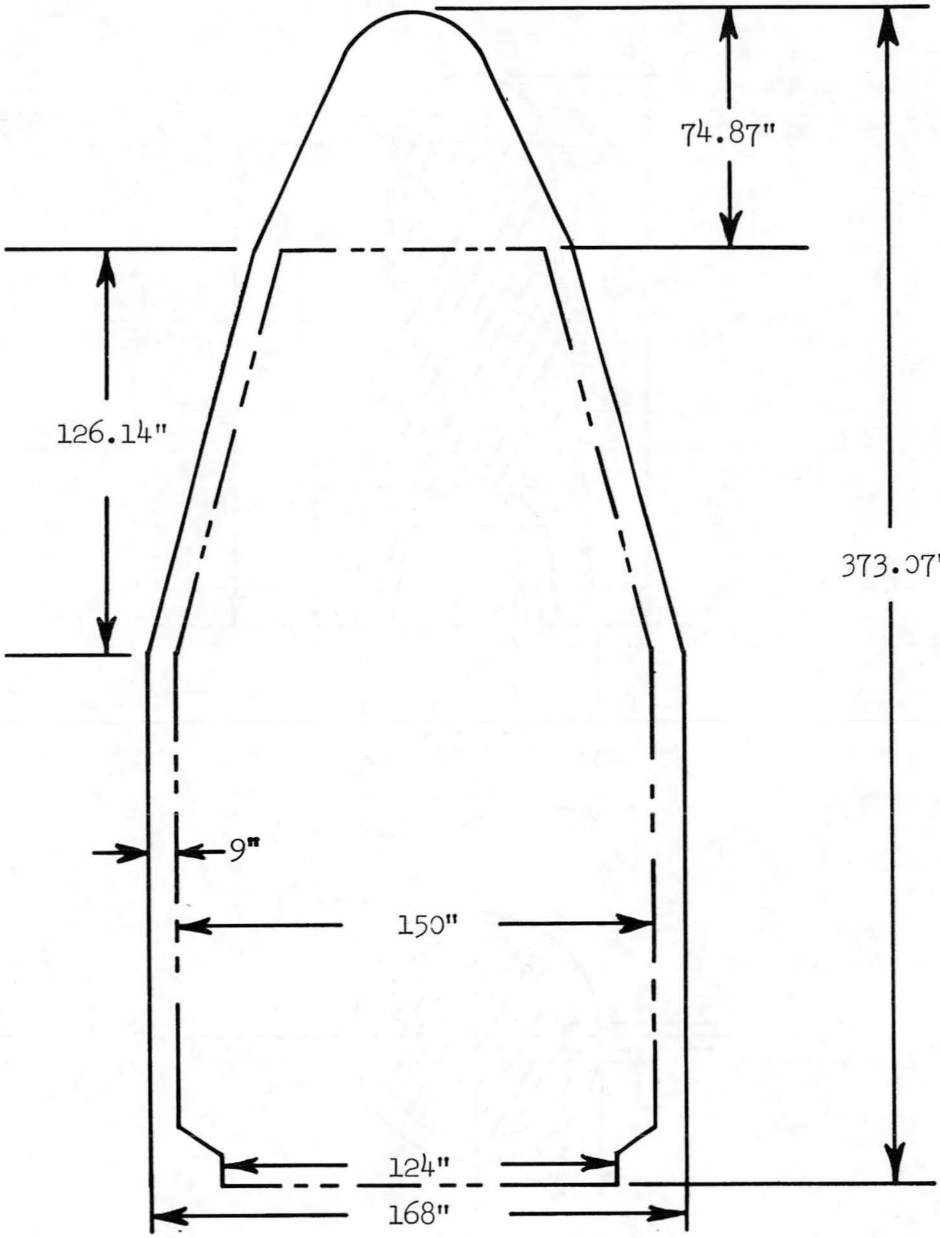
FIGURE V-11. SATURN IB AND SATURN V SHROUD (BASED ON MODIFIED LEM ADAPTER)



Note:

1. This fairing is being extended by the addition of a 266-inch cylindrical section.
2. All dimensions are in inches.

FIGURE V-12. STANDARD BURNER II FAIRING SYSTEM



Weight= 6000 lb
(Including section enclosing Centaur)
Payload Volume=~2566 ft^3

FIGURE V-13. TITAN/CENTAUR BULBOUS VIKING SHROUD PAYLOAD ENVELOPE

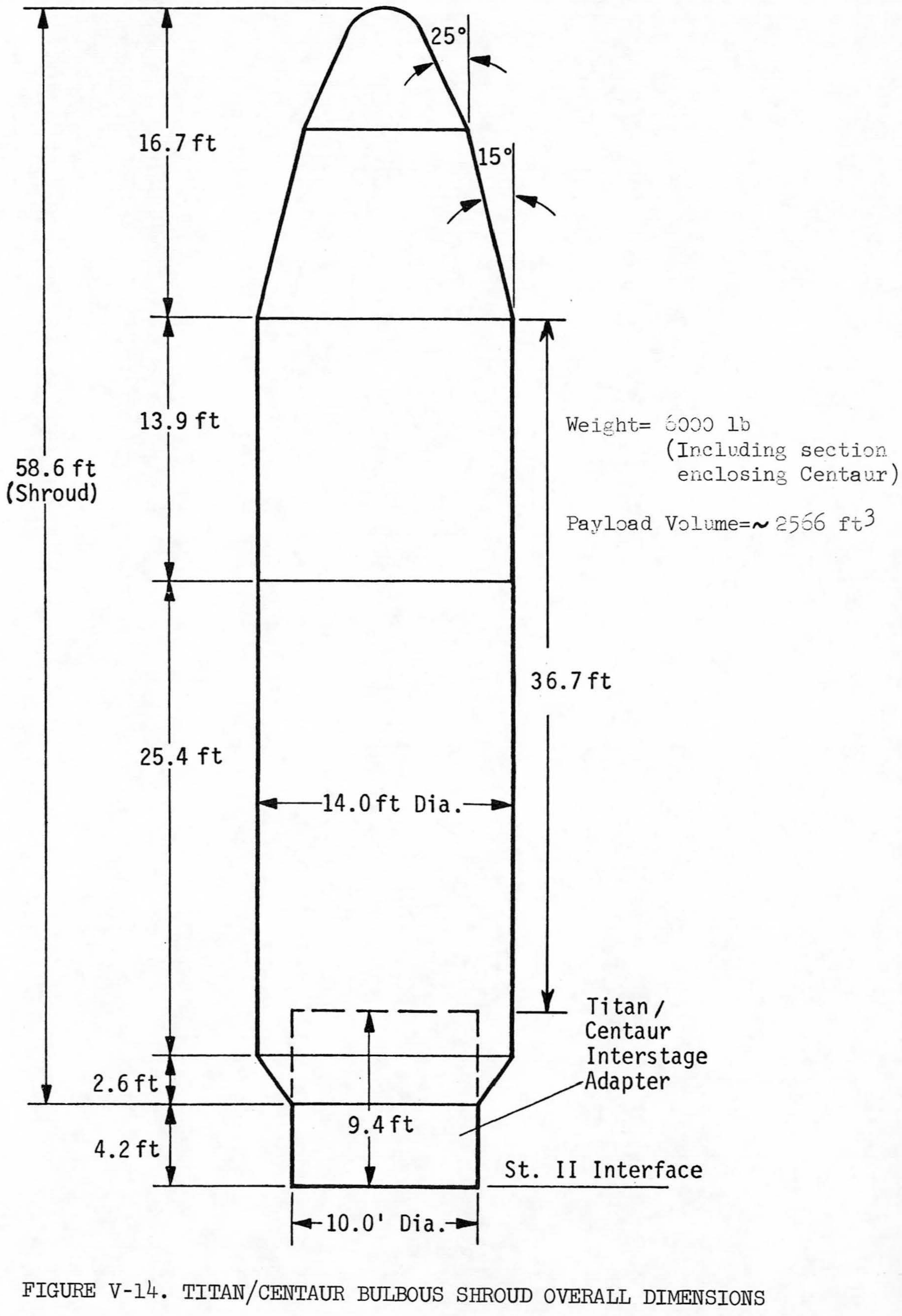


FIGURE V-14. TITAN/CENTAUR BULBOUS SHROUD OVERALL DIMENSIONS

SECTION VI

DEFINITION OF TERMS

SECTION VI

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a.u., astronomical unit:

The mean distance between Earth and the Sun.

Apsis:

The point on an orbit where the distance from the attracting body is either greatest or least. The greatest distance is the apoapsis, while the least is the periapsis.

Aphelion:

The apoapsis of an orbit about the Sun.

C_3:

A measure of kinetic energy (i.e., twice the kinetic energy/unit mass), relative to Earth, remaining after Earth escape, given in km^2/sec^2. The square of the hyperbolic excess velocity.

Declination angle:

The angle between a vector and the equatorial plane. The declination angle, as used in Section II, refers to the declination of the hyperbolic excess velocity vector with respect to the Earth's equatorial plane.

Eccentricity:

An indicator of the shape of an orbit. For elliptical orbits, the eccentricity is the difference between apoapsis and periapsis, divided by their sum. For a circular orbit, the eccentricity is zero; for elliptical orbits, the eccentricity approaches 1 as the orbit becomes more elongated. For hyperbolic orbits, the eccentricity is greater than 1.

Hyperbolic excess velocity:

In the preliminary analysis of interplanetary trajectories, both the Earth-centered escape path and the target planet-centered approach path are hyperbolic. The hyperbolic excess velocity is the velocity at an infinite distance along the asymptote of the hyperbolic path. For Earth escape paths, the hyperbolic excess velocity is approximately the velocity, relative to Earth, as the spacecraft departs the Earth's sphere of influence. Conversely, as the spacecraft approaches a target planet, the hyperbolic excess velocity of the approach hyperbola is approximately the velocity relative to the target planet, as the spacecraft enters the sphere of influence.

Payload:

Payload is considered to include all elements, normally associated with the spacecraft, which must be accelerated to a required final velocity.

Perihelion:

The periapsis of an orbit about the Sun.

Perijove:

The periapsis of an orbit about Jupiter.

Sphere of influence:

A loosely defined region about a planet within which the planet's gravitational field dominates that of the Sun.

Stable libration points:

In a system with two attracting bodies, such as Earth and Moon, rotating about a point, five positions may be found at which a third massless body could be placed in equilibrium under the combined attracting force of the two bodies and the centripetal acceleration associated with the rotation of the system. Only two of these points are stable, such that a small displacement would give rise to forces tending to return the spacecraft to its original position. These stable libration points lie at 60 degree angles on both sides of the line connecting the two attracting bodies, and are equidistant from both. This distance is equal to the distance between centers of the two primary bodies.

Synodic period:

The period between two successive conjunctions of two bodies in orbit about a central body.

V_C, characteristic velocity:

As used in this document, characteristic velocity has two meanings, one related to the mission and the other related to the launch vehicle.

As used in Section II, the characteristic velocity is the arithmetic sum of all velocity increments required to perform a given mission, assuming the mission begins from a 100 n. mi. parking orbit. The velocity in that orbit, 25,581 ft/sec, is included in the characteristic velocity.

As used in Section IV-A, the characteristic velocity is the actual total velocity deliverable for each given payload at an altitude of 100 n. mi. after an eastward launch from ETR.

V_I, ideal velocity:

Ideal velocity as related to mission requirements above escape velocity is often given as

$$V_I = \sqrt{(36{,}178)^2 + (VHL)^2} + 4{,}000 \quad ,$$

or

$$V_I = V_C + 4{,}000$$

This quantity is the total idealized vehicle velocity capability, in ft/sec, required to achieve a given hyperbolic excess velocity after Earth escape from a 100 n. mi. parking orbit, assuming all velocity losses from Earth surface to escape are equivalent to a 4,000 ft/sec velocity decrement.

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