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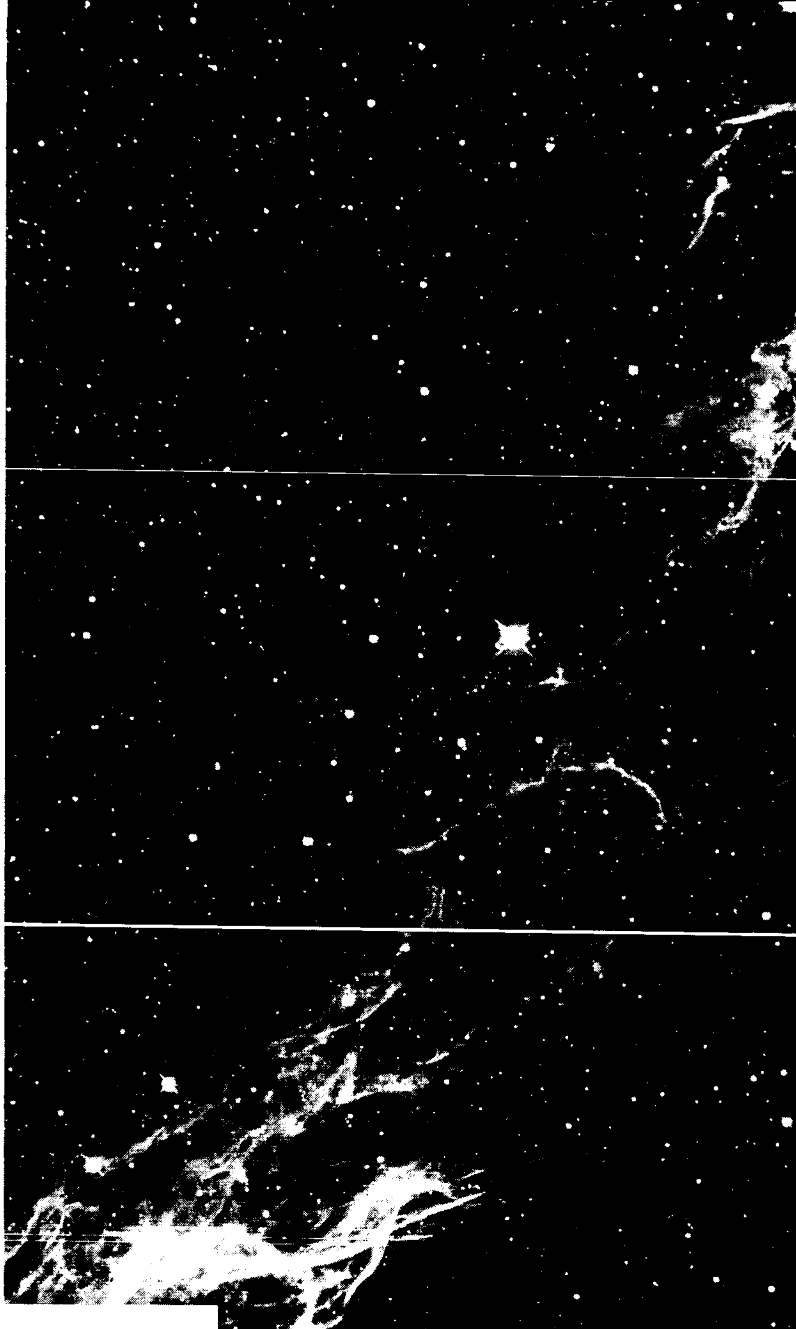

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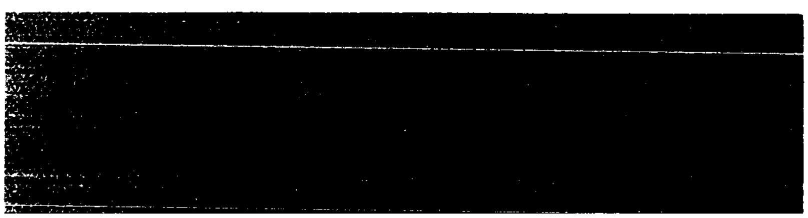
Report No. M-11

A SURVEY OF MISSIONS TO SATURN, URANUS, NEPTUNE, AND PLUTO



Report No. M-11

A SURVEY OF MISSIONS TO SATURN, URANUS, NEPTUNE, AND PLUTO

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SUMMARY

A survey has been conducted of potential missions to Saturn, Uranus, Neptune, and Pluto. Because these planets are so different from the more familiar terrestrial planets, such missions are of considerable scientific interest. The data to be obtained, particularly the atmospheric composition, heat balance, and internal structure data, are of interest per se and also will shed much light on the origin and the evolution of the planetary system. A direct ballistic mode of flight appears to be satisfactory for loose planetary orbiters, except for Pluto. Flight times range from 2.5 to 10 years for 600- to 2000-lb Saturn-V-Centaur and Saturn-1B-Centaur-Kick loose orbiters to Saturn, Uranus, and Neptune. A gravity assist mode of flight should be used, in years when it is possible, for flyby missions. The next launch opportunities for some gravity assist missions are:

Earth/Jupiter/Saturn	1976-1978, then in 1996
Earth/Jupiter/Uranus	1978-1980, then in 1992
Earth/Jupiter/Neptune	1979-1981, then in 1992
Earth/Jupiter/Pluto	1976-1978, then in 1989

In years when gravity assist is not possible or if the guidance requirements are too high for gravity-assisted flights, direct

ballistic flights will be satisfactory. For near-planet circular orbiters a nuclear electric low-thrust mission mode becomes attractive. Missions to each of the planets could be quite similar; a basic science payload of 85 lb and a total payload of the order of 1000 lb can be used. The power requirements can be satisfied with an RTG supply of about 150 watts of raw power. A bit rate of 20 bits/sec and an antenna diameter of not more than 16 feet appear satisfactory. The guidance requirements for direct flyby flights appear to be well within the state of the art; for initial flyby flights a miss distance of 3 or more planet radii, with an uncertainty of 1 planet radius, appears to be satisfactory. Guidance requirements for gravity-assisted flights may be quite stringent. Both flyby and orbiter missions should be performed, followed by atmospheric probes and perhaps landers. An extensive program of flyby flights is not recommended, because the data attainable from flyby missions are limited in comparison to these provided by orbiters.

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

LIST OF TABLES

LIST OF FIGURES

Report No. M-11

A SURVEY OF MISSIONS TO SATURN, URANUS, NEPTUNE, AND PLUTO

1. INTRODUCTION

The outer planets, Jupiter, Saturn, Uranus, Neptune, and Pluto, are intriguing subjects for exploration. The inner terrestrial planets, Mercury, Venus, Earth, and Mars, are small, rather dense, and have slow rotation rates; while the outer planets (with the probable exception of Pluto) are large low-density bodies, often with extensive satellite systems and rapid rotation rates. Their low density suggests a light element content such that their chemical composition may be close to that of the material from which the solar system formed. Thus, determination of their atmospheric and interior properties would be of particular significance to cosmogonical theories.

This report documents a preliminary survey of missions to the planets Saturn, Uranus, Neptune, and Pluto. Missions to Jupiter, the first of the 5 outer planets, are not considered here, since they have been covered in earlier Astro Sciences Center reports (Davies et al. 1964; Witting, Cann, and Owen 1965; Roberts 1964).

Section 2 of this report presents the conclusions of the survey.

Section 3 discusses in some detail the scientific mission objectives. Both the fundamental scientific measurements and the mission mode which could be used to obtain the measurements are considered. The main mission modes considered are flyby and orbiter; some attention is also given to atmospheric probes and planetary landers.

Section 4, Flight Considerations, compares direct ballistic, gravity-assisted ballistic, and nuclear electric low-thrust flight modes for flyby and orbiter flights.

Section 5 touches upon some of the constraints which would be involved in outer-planet missions. In particular, guidance, attitude control, thermal control, power supplies, and communications are discussed.

Section 6 outlines a typical payload for a ballistic flyby flight.

An extensive compilation of trajectory and payload data for Saturn, Uranus, Neptune, and Pluto is included in the appendix.

2. CONCLUSIONS

The basic conclusion of this mission survey is that scientifically interesting missions to Saturn, Uranus, Neptune, and Pluto are technically possible. Flight times range from 1 to 3 years for Saturn to a minimum of 4 years for Pluto with a nuclear electric low-thrust stage.

Space flight studies of these planets, especially Saturn, Uranus, and Neptune, which differ radically from the more familiar terrestrial planets, may provide essential information on the origin and the evolution of the planetary system. In particular, atmospheric studies can lead to an understanding of the heat balance and the level of radioactivity in the planet interiors. From this information it should be possible to determine whether the planets were formed simultaneously throughout the solar system. Furthermore, the composition of the atmospheres and the related escape temperature of the exospheres may provide important data leading toward a definition of the material from which the solar system was formed. Knowledge of the planetary magnetic fields is of particular importance in understanding the planet interiors. This knowledge, in turn, will provide data on the thermal history of the planets.

Both flyby and orbiter missions should be performed, followed by atmospheric probes and eventually perhaps landers. An extensive program of flyby flights is not recommended, because the data attainable from flyby missions are limited in comparison to those provided by orbiters.

The following conclusions have been reached from a comparison of different flight modes.

Flyby Flights

For flyby flights the preferred mode is ballistic gravity assist. This mode should be used in those years in which the planets are correctly aligned. Launch opportunities for gravity-assisted missions are the following:

Mission	Next Launch Period	Start of Following Period
Earth/Jupiter/Saturn	1976-1979	1996
Earth/Jupiter/Uranus	1978-1980	1992
Earth/Saturn/Uranus	1979-1985	2025
Earth/Jupiter/Neptune	1979-1981	1992
Earth/Saturn/Neptune	1979-1985	2015
Earth/Jupiter/Pluto	1976-1978	1989

For years when gravity assist is not available, nuclear electric low-thrust vehicles offer a significant advantage over the Saturn-1B-Centaur for Saturn, Uranus, and Neptune flights, but do not offer a significant advantage over the Saturn-V-Centaur. For Pluto, a thrusted trajectory does have a very significant flight time advantage over a Saturn-V-Centaur.

Orbiter Missions

For orbiter missions a gravity assist mode should not be used, because the high approach velocity in gravity-assisted missions actually reduce the payload in orbit to less than that obtainable from direct ballistic flights. For loose (highly eccentric) orbiters the low-thrust mode of flight is not significantly better than a ballistic flight using a Saturn-V-Centaur ballistic vehicle, except for missions to Pluto.

Circular near-planet orbits are not feasible with the Saturn-V-Centaur. For these missions a nuclear electric low-thrust vehicle is very attractive.

For initial flyby flights a miss distance uncertainty of the order of 1 planet radius and a target miss distance of 3 or more planet radii are satisfactory and within the guidance state of the art. Guidance requirements for gravity-assisted missions will certainly be far more stringent than for direct missions.

An RTG power source producing 100 to 150 watts of raw power will be required, as will an active system of thermal control. A communications bit rate of about 20 bits/sec appears to be satisfactory and could be obtained with an antenna and transmitter weight of 20 to 75 lb, and an antenna diameter of 9 to 16 ft. The total payload weight for flyby missions will be of the order of 1000 lb or less.

The communications, power supply, thermal control, and other spacecraft design parameters appear to be quite similar, at least for Saturn, Uranus, and Neptune. Thus it appears that one standard spacecraft could be used to do each type of mission to all the planets.

A reasonable sequence of missions appears to be a direct or gravity-assisted ballistic flyby mission followed by a series of direct ballistic loose planet orbiters, followed perhaps by nuclear electric low-thrust near-planet circular orbiters, and probes and landers.

3. SCIENTIFIC MISSION OBJECTIVES

3.1 Introduction

> "The exploration of the solar system bears on the three central scientific problems of our time: the origin and evolution of the Earth, Sun, and planets, the origin and evolution of life, and the dynamic processes that shape man's terrestrial environment." (Space Science Board 1965).

The outer planets,* with perhaps the exception of Pluto, present themselves as most interesting subjects for investigation of these problems because of their extreme differences from the more familiar inner terrestrial planets.

Whereas the terrestrial planets are small, rather dense, and have slow rotation rates, the outer planets are characterized as large low-density bodies, often with extensive satellite systems and more rapid rotation rates. Their low densities imply an appreciable light-element content such that their chemical composition may be much closer to that of the material from which the solar system formed rather than that of the inner planets. Thus determinations of both atmospheric and interior properties would be of particular importance to cosmogonical theories.

At the present time, very little about these planets is known with certainty. Hydrogen, methane, and ammonia have been identified in certain of their atmospheres. Various authors have constructed theoretical models for their interior

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*Jupiter, having been treated extensively in previous ASC/IITRI reports, is not explicitly included in this discussion. A separate ASC/IITRI report discusses the scientific objectives of Saturn, Uranus, Neptune, and Pluto missions (Dickerman 1966).

structures by using plausible ratios of hydrogen to helium, but these are at best tentative. The properties of these planets that are well-defined are given in Table 1. To achieve the desired knowledge, further data must be obtained from a combination of Earth-based observations, Earth orbiter observations, and flyby and planetary orbiter missions.

3.2 Bulk Characteristics of the Outer Planets

The planets, in general, appear to form three chemically distinct groups (Ramsey 1951). Group I consists of the terrestrial planets, Mercury to Mars, of which the Earth is both the densest and the most massive. The planets of Group II, Uranus and Neptune, are more than ten times as massive as the Earth, but they are less than half as dense. The planets of Group III, Jupiter and Saturn, are of the order of a hundred times as massive as the Earth and of very low mean density. The density of Pluto is as yet undetermined; however, speculation seems to favor its being classed as a terrestrial planet or an escaped satellite. Thus planets in the different groups differ widely in mass; the more massive planets contain much higher proportions of the lighter elements.

The cosmogonical significance of this grouping of the planets has been somewhat clarified by Brown (1950), Ramsey (1951), and Wildt (1947). When the temperature is sufficiently low to permit the formation of molecules, the principal constituents of solar material fall into three well-defined classes

Table 1

PLANET PROPERTIES*

Symbol	Earth	Saturn	Uranus	Neptune	Pluto
Semimajor axis of orbit (AU)	1.000000	9.540	19.18	30.07	39.44
Sidereal period (years)	1.0000	29.4577	84.013	164.79	248.4
Eccentricity	.01677	0.05606	0.0471	0.0085	0.2494
Inclination to ecliptic	0°00'00"	2°29'26"	0°46'23"	1°46'28"	17°10'
Mean diameter (km)	12,742	115,100	51,000	50,000	
Earth = 1	1.000	9.03	4.00	3.90	
Mass Earth = 1	1.000	95.3	14.58	17.26	0.93
Mean density (g/cm^3)	5.52	0.71	1.26	1.61	
Axial rotation	$23^h 56^m 4.09$	$10^h 14^m - 10^h 38^m$	$10^h.7$	$15^h.8$	
Inclination of equator to orbit	23°26'59"	26°44'	97°55'	28°48'	
Oblateness	1/297	1/9.5	1/14	1/45	
Mean surface gravity (Earth = 1)	1.00	1.17	0.91	1.12	
Albedo	0.29	0.42	0.45	0.52	
Velocity of escape (km/sec)	11.2	36	21	23	
Atmospheric constituents (observed)		NH_3**, CH_4, H_2	CH_4, H_2	CH_4, H_2	
Atmospheric temperature (spectroscopic estimates)		90-100°K	70-80°K	70°K	
Number of satellites	1	9	5	2	

* All data, with the exception of the atmospheric properties, are taken from Allen (1963).

** Tentative identification only.

Earth mass = 5.975 x 10^{24} kg

(see Table 2). Class I consists of the common terrestrial materials, the most important being metallic iron and the oxides of iron, magnesium, and silicon. These materials have high molecular weights and very high boiling points. The compounds of the second class have molecular weights ranging from 16 to about 20 and boiling points of the order of 100°K; the principal constituents in this class are water, methane, and ammonia. The third class consists of hydrogen and helium, which have the lowest molecular weights and also the lowest boiling points. The solar abundance for hydrogen and helium is about 60 times as high as the abundance of the non-hydrogenic elements composing components of the second class, which, in turn are about four times as abundant as those in the first class.

This separation of the constituents of solar material into three well-defined classes affords a possible explanation for the division of the planets into three chemically distinct groups. The terrestrial planets are unable to retain the light, volatile compounds of the second and third classes. The major planets, on the other hand, have retained large quantities of these materials. Such retention or loss of materials by planets is, in general, a strong function of the exospheric temperature. Specifically, more massive and cooler planets can retain light gases more easily than can the lighter and warmer planets. The fact that Uranus and Neptune, with lower temperatures than Saturn and Jupiter, have lost most of their Class III materials must be due, at least in part, to lower original masses.

Table 2

PLANET GROUPS

Groups	Probable Principal Constituents	Comments
Group I Terrestrial Planets	Class I materials; iron, oxides of iron, magnesium, and silicon	Relatively high mean atomic weight ($\gtrsim$ 20)
Group II Uranus and Neptune	Class II materials; water, methane and ammonia	These planets are approximately ten times as massive as Earth. Only tentative atmospheric data exist. Mean atomic weight ~ 4.
Group III Saturn and Jupiter	Class III materials; hydrogen and helium	These planets are approximately 100 times as massive as Earth. Contain large amounts of hydrogen; mean atomic weight ~ 1.

3.3 Planetary Atmospheres

Spectroscopically, Saturn appears generally much like Jupiter although our knowledge is more limited because of its greater distance from Earth. The methane bands generally appear as very strong features, and hydrogen has also been observed. A rotational temperature of 90°K was derived from the hydrogen spectra (Spinrad 1964). This is compatible with radiometric measurements in the 8-14μ infrared "window" of our atmosphere which yielded a temperature of approximately 100°K (Murray and Wildey 1963).

Intense methane bands have been observed in the spectra of Uranus and Neptune, completely altering the continuous spectra from the yellow to the infrared and perhaps concealing weaker bands of other molecules. No ammonia has been observed in the spectra of either planet; if it exists, it is undoubtedly frozen out, as are carbon dioxide and water. Spectral features attributed to the hydrogen molecule, when compared with laboratory spectra, indicate an atmospheric temperature of approximately 80°K for these planets (Rea 1962).

No atmosphere has yet been observed on Pluto. Most readily observable species would freeze out if the temperature is near the assumed value of 50°K.

3.4 Basic Scientific Measurements

There are many scientific measurements which can be planned at present for outer planet missions. It must be kept in mind, however, that with the possible exception of Pluto, conditions are such that most of the information from early missions will be primarily related to the outer layers of a very extensive, optically thick atmosphere.

Table 3 lists some basic measurements considered important for early missions, along with a brief description of the data which would be anticipated for each case. While it is not possible to establish the eventual full significance of each measurement at the present time, some general comments can be made for several of the categories.

For example, atmospheric studies, including IR measurements of the planetary dark sides, can tell at what rate energy is being supplied from internal sources, which is directly related to the planetary origin. Such information may lead to a determination of what roles gravitational, rotational and chemical energy and nuclear decay play in heating the planetary interiors. If the level of internal radioactivity can be established, it may be possible to determine whether or not the process of formation was simultaneous throughout the solar system. Atmospheric temperatures as well as the escape temperature of the exospheres may also be obtainable from certain spectroscopic observations. Since the value of this latter temperature governs the loss rate of atmospheric constituents, it is a basic parameter in the study of planetary evolution.

Determination of planetary compositions is important for a variety of reasons. Fundamentally, for example, if abundances of hydrogen, deuterium, helium, and carbon can be obtained, a

Table 3

PARTIAL LIST OF BASIC MEASUREMENTS FOR OUTER PLANET MISSIONS

Category	Type of Data	Instrumentation	Mission Mode
Magnetic fields	Measurement of the magnitude and configuration of the planetary magnetic fields	Rubidium vapor, helium vapor, and/or rotating coil magnetometer	Orbiter or Flyby
	Determination of the boundaries of the magnetosphere		Eccentric orbiter or flyby
	Orientation of the magnetic fields		Orbiter
	Determination of both short and long period variations in the planetary fields		Orbiter
	Detection of possible field variations due to surface features and anomalies		Low-altitude orbiter
	Detection of any magnetic fields associated with the satellites of the outer planets		Eccentric orbiter
Charged particles	Determination of changes in intensity with radial distance in the vicinities of the planets	Solid state detectors; shielded Geiger counters	Flyby
	Measurement of the solar wind flux in the vicinities of the planets		Flyby

Table 3 (Cont'd)

Category	Type of Data	Instrumentation	Mission Mode
Charged particles	Detailed study of particle energy spectrum, pitch-angle distribution, and time variations in any existing trapped radiation belts		Low-altitude orbiter
	Determination of possible interactions of satellites with particle fluxes		Eccentric orbiter
Dust	Determination of size and distribution of dust particles throughout the trajectory in the asteroid belt and in the vicinity of the planets	Large area microphones, crystal flash detectors, pressure cells	Flyby or orbiter
Solar system constants	Improved determinations of satellite orbits	Ground-based tracking network	Eccentric orbiter
	Precise determination of planetary masses, particularly of Pluto		Flyby or orbiter
	Improved value of the astronomical unit		Flyby
	Determination of Pluto's diameter by occultation of the Sun or a star	Onboard optical equipment	Flyby or orbiter

Table 3 (Cont'd)

Category	Type of Data	Instrumentation	Mission Mode
Planetary atmospheres and satellite systems	Definition of cloud structure of Saturn, Uranus, and Neptune	TV cameras and multicolor photometers	Orbiter
	Detection of motion and turbulence in the atmosphere		Orbiter
	Observations of time-variant prominent features and other anomalies		Orbiter
	Definition of belts, spots and other features which have been observed on Saturn		Orbiter
	More precise determination of rotational period of Saturn		Orbiter
	Observations of satellites		Eccentric orbiter
	Occultation of a star by the Saturn ring system to better determine optical thickness and distribution of matter		Flyby or orbiter
	Determination of species present in the atmospheres of the outer planets	UV and visible grating monochrometer	Flyby or orbiter
	Spatial distribution of these species in the atmosphere		Orbiter

Table 3 (Cont'd)

Category	Type of Data	Instrumentation	Mission Mode
	Atmospheric pressure determinations		Flyby or orbiter
	Aurora and airglow observations		Flyby of orbiter
	Determination of hydrogen-helium ratio on Uranus and Neptune		Flyby or orbiter
	Depth of atmosphere	Radar altimeter	Flyby or orbiter
	Spatially resolved thermal mapping of atmosphere	IR radiometer	Orbiter
	Detection of IR from dark side		Flyby or orbiter
	Detailed data concerning apparent nonthermal emission on Saturn	Microwave radiometer	Orbiter
	Improvement of accuracy of existing measurements plus spatially resolved data for all outer planets		Orbiter
	Measurements between 0.1 and 10 cm, sensitive to NH_3 concentration, for Saturn		Flyby or orbiter

Table 3 (Cont'd)

Category	Type of Data	Instrumentation	Mission Mode
	Pressure and temperature in lower atmosphere	Aerodynamic-type sensors or light-absorption instrumentation	Lander or atmospheric probe
	Optical thickness and polarizing properties of the atmosphere; fluxes of radiation as a function of height	Zenith-pointed skylight analyzer	Atmospheric probe
	Atmospheric density and density gradients	Orbiter tracking of atmospheric probe	Atmospheric probe
	Identification of lower atmospher constituents	Mass spectrometry	Atmospheric probe
	Detection of bio-molecules	Chemical tests	Atmospheric probe or lander
Planetary surfaces	Determination of surface features	Radar altimeter	Low-altitude orbiter
	Identification and location of fluid-solid interface; hardness and nature of surface	Impactometer or penetrometer	Atmospheric probe or lander

direct comparison might be made with solar cosmic-ray data for the helium-to-carbon ratio and solar model calculations on the hydrogen-to-helium ratio. These data are important for work which attempts to define the material from which the solar system was formed and for determining evaporation rates for the planets. These definitions in turn, serve as constraints for theoretical models.

Determination of the planetary magnetic fields is also of significance. As the configuration of these fields becomes known, more can be learned about the structure of the planets' interiors. It is important to determine in what way the cores of these planets (supposed at present to be essentially metallic hydrogen, a good electrical conductor) can support the electrical currents generating the magnetic fields. More may also be learned about convection in the interiors, information of importance for the thermal history of the planets. Related topics are the possible existence of radiation belts and the way in which these belts are energized.

Atmospheric circulation and meteorological effects are also of fundamental interest. If the atmospheres are very deep, it may be that circulations in the lower atmosphere could lead to magnetic-field generation. It is also important to determine why certain material in the atmosphere of Jupiter (and seemingly also Saturn) appears to rotate at several kilometers per second relative to the visible planet. In addition, of course, the nature of the planetary surfaces remains to be determined. A related topic is the Saturn ring system, the origin of which still remains to be determined. There are, in addition, several orbital characteristics (the anomalies in the Triton and Pluto orbits and the peculiar rotation of Uranus) which still remain to be explained.

Finally, it is essential to determine whether there is life or protobiotic material on any of the outer planets and whether free radicals or pre-biotic compounds are presently being formed in the upper atmosphere.

4. FLIGHT CONSIDERATIONS

4.1 Introduction

In planning missions for interplanetary exploration one of the first decisions is a choice of the flight mode. As solar system exploration extends beyond Jupiter, alternative trajectory modes begin to compare favorably with the direct ballistic flight mode, and then surpass it. For many early missions the crossover point is beyond Saturn.

The three modes of flight considered in this report are direct ballistic, gravity assisted ballistic, and nuclear electric low thrust. The direct ballistic flight, in which the spacecraft coasts to its target after burning all of its fuel in the vicinity of the Earth, is the simplest of the flight modes. Launch opportunities exist essentially once a year for each of the outer planets, and midcourse guidance and control requirements appear to be within the state of the art for initial missions. The disadvantage of direct ballistic flight is that for flights beyond Saturn, the flight times, even with a Saturn V-Centaur, are longer than 3 years.

Gravity assisted ballistic flights, in which the gravitational field of an intermediate planet (usually Jupiter) is used to speed the spacecraft to its final destination, are of significantly shorter flight times for flights beyond Saturn. However, gravity assist flights can only be performed in years when the gravity assist planet and the target planet are in favorable positions; the guidance and control requirements will be somewhat more stringent than for direct ballistic flights. For orbiters, gravity-assisted flights are not as good as direct flights because of the high approach velocities which are characteristic of gravity-assisted flights.

Nuclear electric low-thrust stages are particularly attractive for near-planet circular orbiter missions and can provide a wide range of payload weights for outer-planet missions. The main disadvantages of nuclear low-thrust stages are the current state of development and the projected costs.

4.2 Comparison of Flight Modes

Since flight time is a key parameter in missions to the outer planets, it is appropriate to use this parameter for a comparison of different flight modes.

Figure 1 shows the flight time to the outer planets for a 600-lb ballistic payload on the Saturn-1B vehicle and the approximately equivalent 150-lb communications and experiments payload on a thrusted stage. For the direct and the gravity-assisted flights, a Centaur upper stage is assumed. For the thrusted trajectory, a conceptual stage with a specific power plant mass α = 40 lb/kw (250-kw power source) is assumed. It can be seen from the figure for Saturn that the flight times for the three vehicles are in the 2 to 3 year range. From this observation we conclude that a direct ballistic flight is a satisfactory mode for the flyby missions to Saturn. For Uranus, Neptune, and Pluto, a gravity-assisted flight has very significant

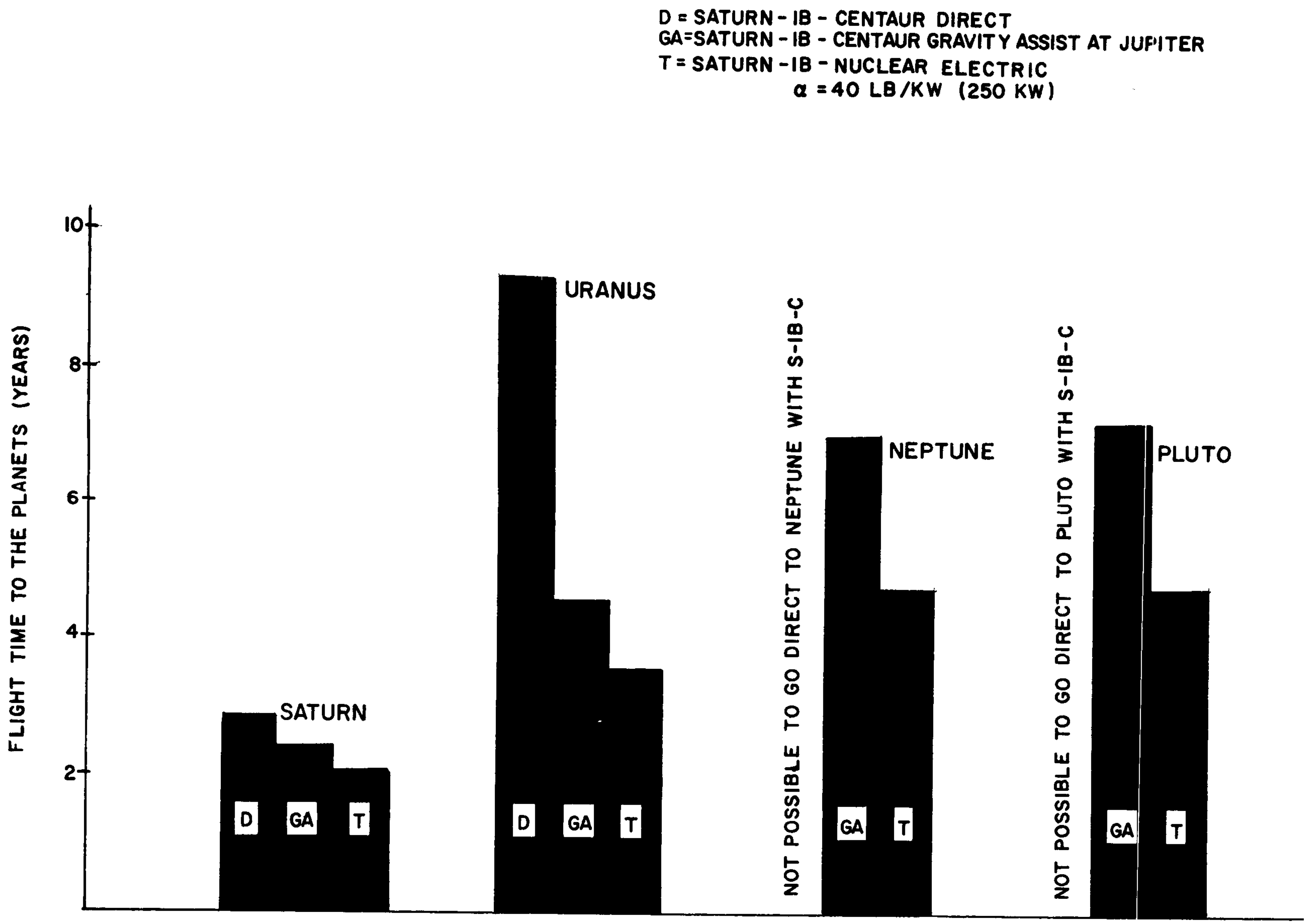

FIGURE I. COMPARISON OF FLIGHT TIME OF 600 LB. BALLISTIC PAYLOAD (OR EQUIVALENT) FLYBY FLIGHTS.

advantages over a direct ballistic flight. For Uranus, the gravity-assisted flight takes less than half the time for a direct flight. For Neptune and Pluto, the direct flight is not possible with a Saturn-1B-Centaur and 600-lb payload. In all cases the thrusted flights are somewhat shorter than the gravity-assisted flights; the flight time advantage of thrusted over gravity-assisted is significant for Neptune and Pluto.

Figure 2 is similar to Figure 1 except that a Saturn-V-Centaur vehicle is assumed for the direct and gravity-assisted flights. In this case the direct, gravity-assisted, and thrusted flights are quite comparable in flight times for Saturn, Uranus, and Neptune. The thrusted flights to Saturn and Uranus actually take longer than the ballistic flights.

From both Figures 1 and 2, we conclude that, for flyby missions to the outer planets, thrusted vehicles do not begin to offer a significant advantage until Uranus and beyond. For all the planets (except possibly Pluto), the gravity-assisted flights and the thrusted flights are not greatly different in flight times. Since it is likely that thrusted stages will be considerably more expensive, we conclude that, in years when gravity assist missions are feasible, this mode of flight will be more attractive than the thrusted flight. These conclusions would not be significantly changed for payloads up to a few thousand pounds.

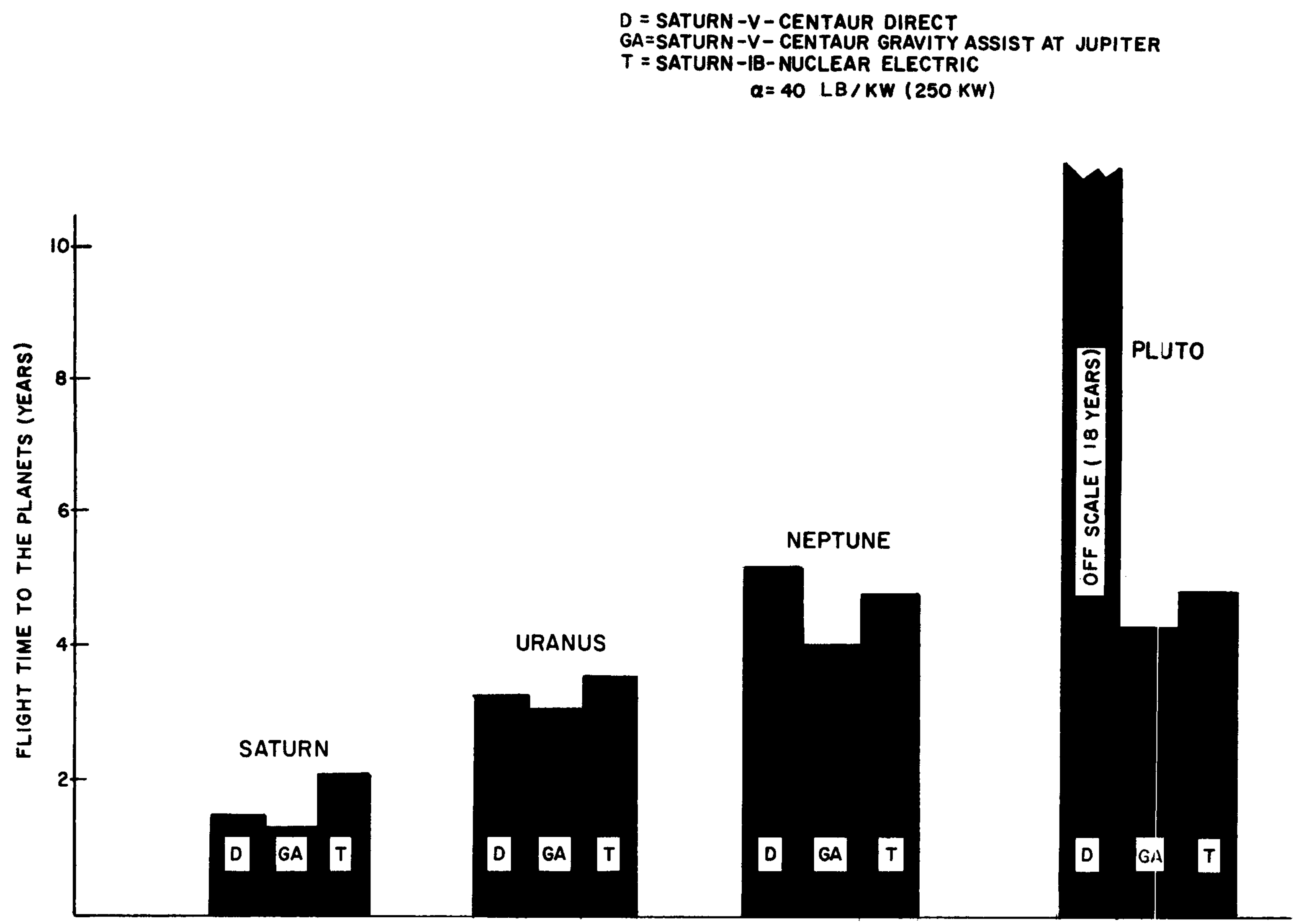

FIGURE 2. COMPARISON OF FLIGHT TIME OF 600 LB BALLISTIC PAYLOAD(OR EQUIVALENT) FLYBY FLIGHTS.

Figure 3 shows a similar comparison for 2000-lb orbiters, for which the capture orbit is parabolic* with a periapsis distance r_p = 3 radii from the planet center (for the thrusted flight any parabolic orbit is possible with essentially the same flight time). It can be seen that for Saturn the Saturn-1B-Centaur-Kick, the Saturn-V-Centaur, and the Saturn-1B-thrusted vehicle do not differ greatly in flight time. In fact, at Saturn the thrusted orbiter requires a longer flight time than the Saturn-V ballistic orbiter. For Uranus the Saturn-1B-thrusted vehicle has a slight flight time advantage over the Saturn-V-Centaur and a 6-year advantage over the Saturn-1B-Centaur-Kick. For Neptune the advantages of the thrusted vehicle are, of course, larger.

Thus it can be concluded that, for parabolic orbiters, the thrusted propulsion mode becomes particularly attractive for missions beyond Uranus.

Circular orbiters at 3 planet radii are not possible with a Saturn-V-Centaur for Saturn; for Uranus, Neptune, and Pluto the required flight times are 10 to 35 years with 2000-lb spacecraft. In contrast, the thrusted stage can put a 500-lb communications and experiments payload into a 3-planet radii circular orbit around any of these planets in 4 to 8 years. Thus thrusted propulsion is the only mode possible for near-planet circular orbiters.

Table 4 summarizes the attractive modes for outer-planet missions.

*A parabolic orbit is essentially the same as a very eccentric elliptical orbit.

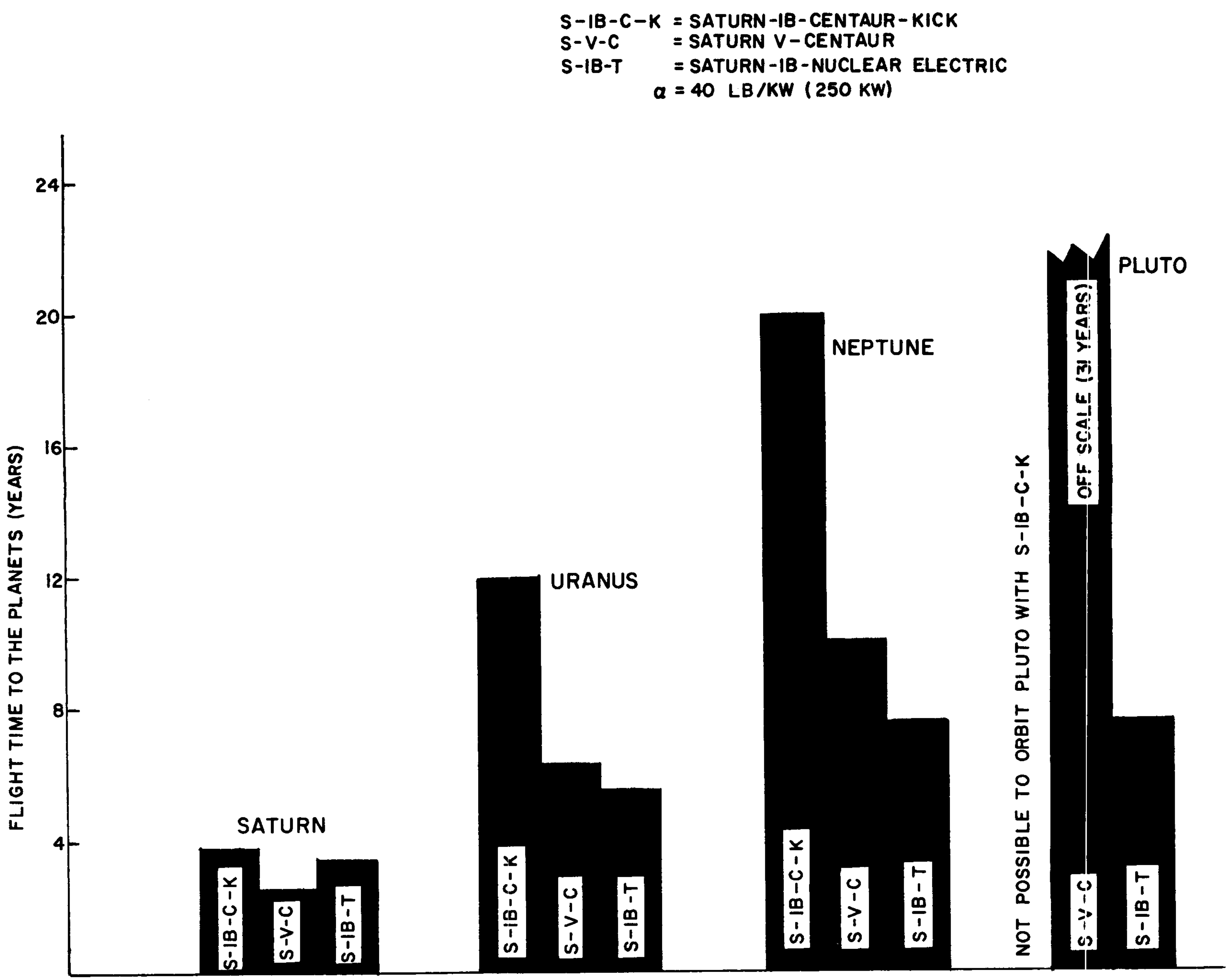

FIGURE 3. COMPARISON OF FLIGHT TIME OF 2000 LB. BALLISTIC (OR EQUIVALENT) PARABOLIC ORBITERS, R_p = 3 PLANET RADII.

Table 4

ATTRACTIVE MODES FOR OUTER PLANET MISSIONS

	Flyby Flights	Orbiter Flights	
		Highly Elliptical Orbits	Circular Orbits
Saturn	D, GA or T	D	T
Uranus	GA, D or T	D or T	T
Neptune	GA, T or D	T or D	T
Pluto	GA or T	T	T

D = Direct ballistic
GA = Gravity assist ballistic
T = Nuclear low thrust

4.3 Direct Ballistic Trajectories

Figure 4 is a side view of the solar system on a 50 AU scale and shows the traces* of the orbits of the outer planets. The dots show the positions of the planets in particular years. Since Saturn, Uranus, and Neptune all lie close to the ecliptic plane (when viewed on this scale), the trajectory requirements to these planets do not vary widely from year to year. Because these variations are slight and in order to facilitate comparison of different flight modes, all trajectory calculations for Saturn, Uranus, and Neptune assumed that the planets are in circular orbits in the ecliptic plane. Comparison with the three-dimensional cases has shown the variation in ideal velocity, ΔV, to be less than 1000 ft/sec over the next twenty years. Because Pluto's orbit is significantly inclined and non-circular, a three-dimensional analysis (with launch in 1975) was performed in this case. Figure 5 shows the flight path for a typical 3-year flight to Saturn.

The major trade-offs that are encountered within the ballistic flight mode are:

- Time of flight vs. ideal velocity (ΔV)
- Time of flight vs. approach velocity (VHP)
- Time of flight vs. retro velocity increment (DV) for planetary orbit.

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*These traces show the planets' latitude and distance from the Sun through its orbit.

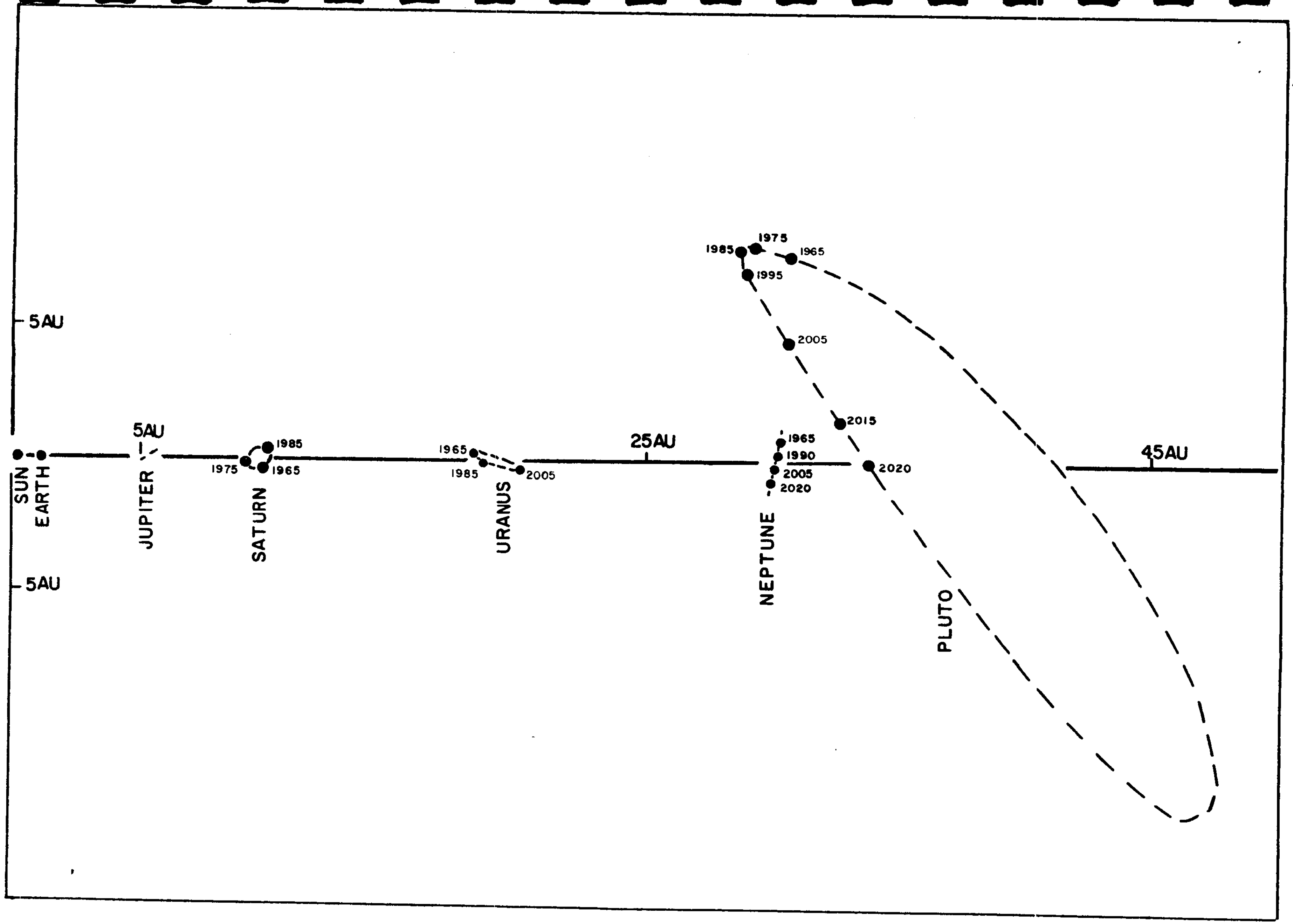

FIGURE 4. SIDE VIEW OF SOLAR SYSTEM

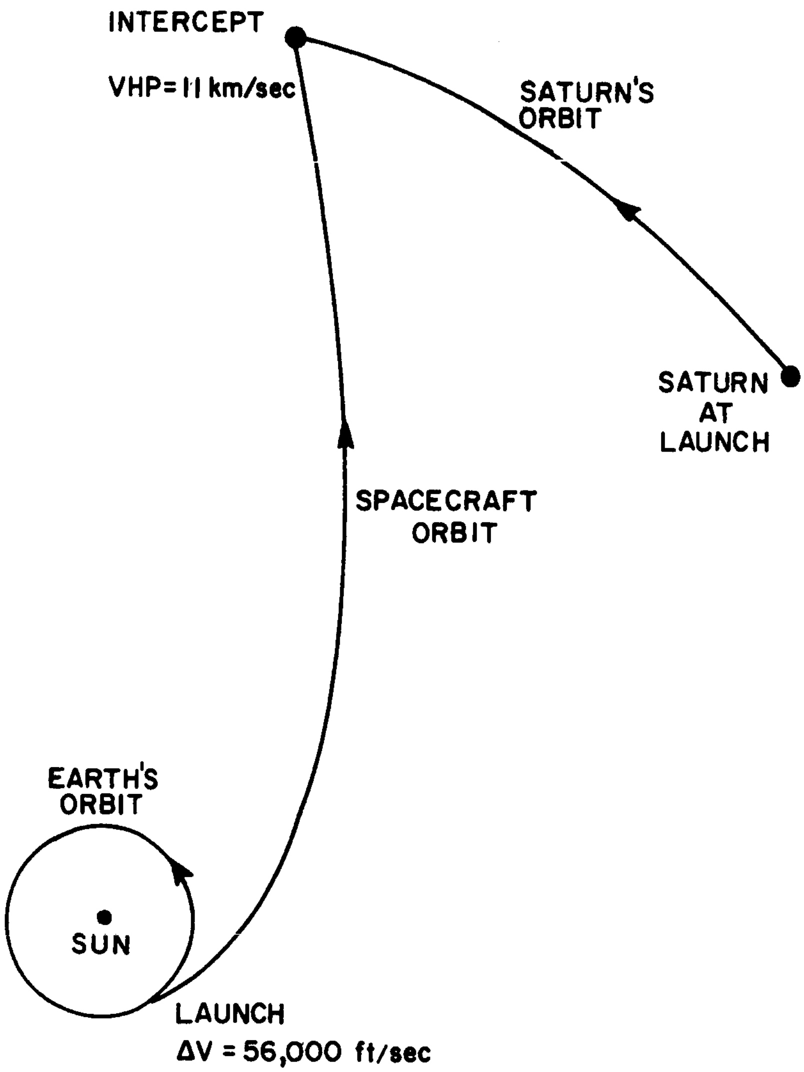

FIGURE 5. 3 YEAR DIRECT BALLISTIC FLIGHT TO SATURN

To illustrate the comparison, Table 5 presents the time-of-flight requirements and corresponding VHP for direct ballistic trajectories to the four planets, assuming 600- and 2000-lb payloads. The launch vehicles in the table were chosen to provide a wide range of comparison.

Ideal Velocity vs. Time of Flight

Ideal velocity is constrained by the launch vehicle capability, and practical times of flight are limited by reliability considerations. Because of the great distances involved, outer-planet missions are characterized by long flight times. In order to achieve a flight time savings for some given payload, a vehicle with a greater ΔV capability must be utilized. As seen from Table 5, a 600-lb payload could be launched to the outer planets within a flight range that is not unreasonable. The exception to this is Pluto, which requires 23.5 years even with the Saturn-V-Centaur. Changing the payload weight to 2000 lb has the effect of eliminating two of the four launch vehicles and lengthening the flight time for the remaining two vehicles.

VHP vs. Time of Flight

As seen from Table 5, there is an inverse relationship between flight time and VHP. Shorter, more energetic flights arrive at the planet with a larger excess velocity. For Uranus the VHP can be reduced by a factor of almost 4 (from 26 to 7 km/sec) by using a 9.4-year Saturn-1B-Centaur flight instead of a 3.2-year Saturn-V-Centaur flight. For this savings in VHP there is a corresponding penalty of 6.2 years in time of flight.

Table 5

DIRECT BALLISTIC TRAJECTORY PARAMETERS FOR 600- AND 2000-LB FLYBY PAYLOADS

Target Planet Launch Vehicle	600-lb Payload Flight Time, TF (yrs)	600-lb Payload Approach Velocity, VHP (km/sec)	2000-lb Payload Flight Time, TF (yrs)	2000-lb Payload Approach Velocity, VHP (km/sec)
Saturn				
SLV3X-Centaur-Kick	3.1	11	--	--
Saturn 1B-Centaur	2.9	12	--	--
Saturn 1B-Centaur-Kick	1.6	25	2.1	18
Saturn V-Centaur	1.4	29	1.5	27
Uranus				
SLV3X-Centaur-Kick	13	5	--	--
Saturn 1B-Centaur	9.4	7	--	--
Saturn 1B-Centaur-Kick	3.4	24.5	5.0	15.5
Saturn V-Centaur	3.2	26	3.3	25
Neptune				
SLV3X-Centaur-Kick	--	--	--	--
Saturn 1B-Centaur	--	--	--	--
Saturn 1B-Centaur-Kick	5.6	24	8.8	14
Saturn V-Centaur	5.2	26	5.6	24
Pluto				
SLV3X-Centaur-Kick	--	--	--	--
Saturn 1B-Centaur	--	--	--	--
Saturn 1B-Centaur-Kick	27	4.7	38	4.6
Saturn V-Centaur	23.5	4.8	27.5	4.7

Note: -- means that the specified launch vehicle, with the given payload, cannot perform the mission.

The point at which the magnitude of VHP becomes critical is dependent upon the individual mission objectives and experiments. Thus the trade-off must be made between time of flight and the requirements for the planetary encounter.

Figure 6 shows the near-planet profiles for typical flyby flights to Saturn, Uranus, and Neptune.

DV Vs Time of Flight

For rendezvous or orbiter missions to the planets, a certain velocity increment, DV, must be trimmed from VHP in order to effect the desired encounter. DV is related to the total weight in orbit by the rocket equation:

$$\text{Weight in orbit} = (\text{approach weight}) \cdot e^{-\left(\frac{DV}{g \cdot ISP}\right)}$$

where

$g = 9.8 \times 10^{-3}$ km/sec^2

ISP = specific impulse of retro engine (sec)

Because the equation is exponential, the payload fraction falls off quite rapidly. DV's of greater than 10 km/sec are unattainable using a high-thrust system for terminal propulsion.

Table 6 shows, for the four planets, the flight time for two spacecraft weights placed into a parabolic orbit with a periapsis distance of 3 planet radii. An ISP of 315 sec was assumed for the calculation.

The table shows that orbiter missions are possible if long times of flight can be tolerated. For example, a 2000-lb orbiter can be placed, with a Saturn V-Centaur, around Saturn

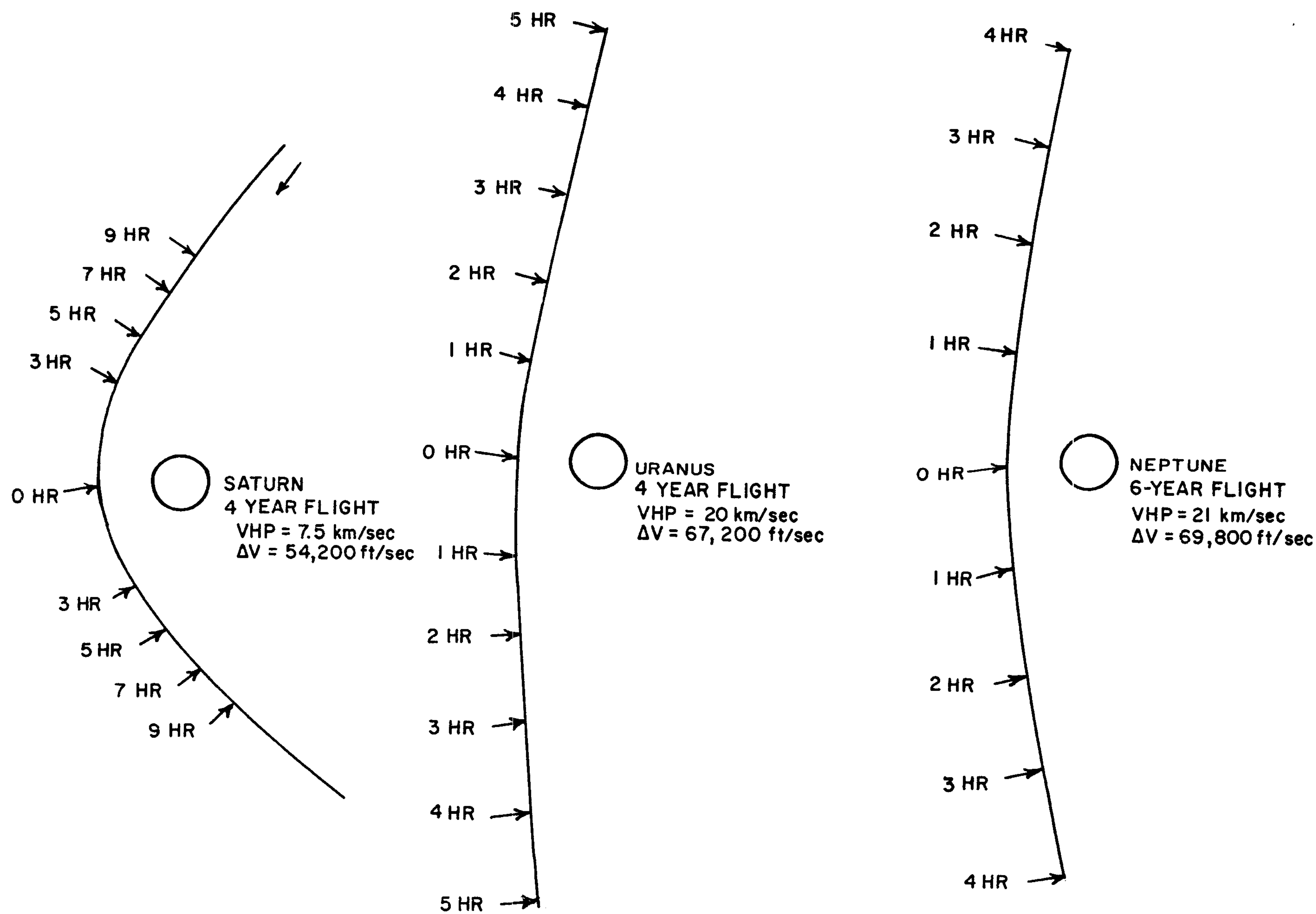

FIGURE 6. TYPICAL NEAR-PLANET TRAJECTORIES

Table 6

DIRECT BALLISTIC TRAJECTORY FLIGHT TIME
FOR 600- AND 2000-LB PARABOLIC ORBITER

Target Planet Launch Vehicle	600-lb Orbiter Flight Time, TF (yrs)	2000-lb Orbiter Flight Time, TF (yrs)
Saturn		
Saturn 1B-Centaur-Kick	2.7	3.8
Saturn V-Centaur	2.4	2.5
Uranus		
Saturn 1B-Centaur-Kick	7.1	12.0
Saturn V-Centaur	5.7	6.4
Neptune		
Saturn 1B-Centaur-Kick	11.4	20.2
Saturn V-Centaur	9.0	10.2
Pluto		
Saturn 1B-Centaur-Kick	32	--
Saturn V-Centaur	25.2	30.8

in 2.5 years, around Uranus in 6.4 years, around Neptune in 10.2 years, or around Pluto in 30.8 years.

For Saturn a lightweight orbiter payload can be accommodated without resorting to the largest vehicles. A 600-lb payload is possible with a Saturn-1B-Centaur-Kick in 2.7 years.

Direct Ballistic Conclusions

In conclusion, the trade-offs for direct ballistic flights reveal no clear-cut points at which alternative trajectory modes must be considered. However, some general conclusions do emerge and can be the groundwork for further consideration.

a) Small precursor flyby missions of 600 to 2000 lb out to and including Neptune can be accomplished by using current vehicle combinations with times of flight of less than 6 years. A 600-lb payload to Pluto, however, would require at least 23 years by using currently envisioned ballistic vehicles.

b) Orbiter missions appear feasible to the outer planets, but beyond Saturn are restricted to long times of flight (5.7-32 years).

4.4 Gravity-Assisted Ballistic Trajectories

Gravity-assisted ballistic trajectories have been discussed extensively (Niehoff 1965) and are of significant value for flyby flights to the outer planets. Figure 7 shows a typical Earth-Jupiter-Saturn mission. Note that the flight has an ideal velocity of 54,000 ft/sec, and that the total flight time is 3 years, and that the approach velocity, VHP,

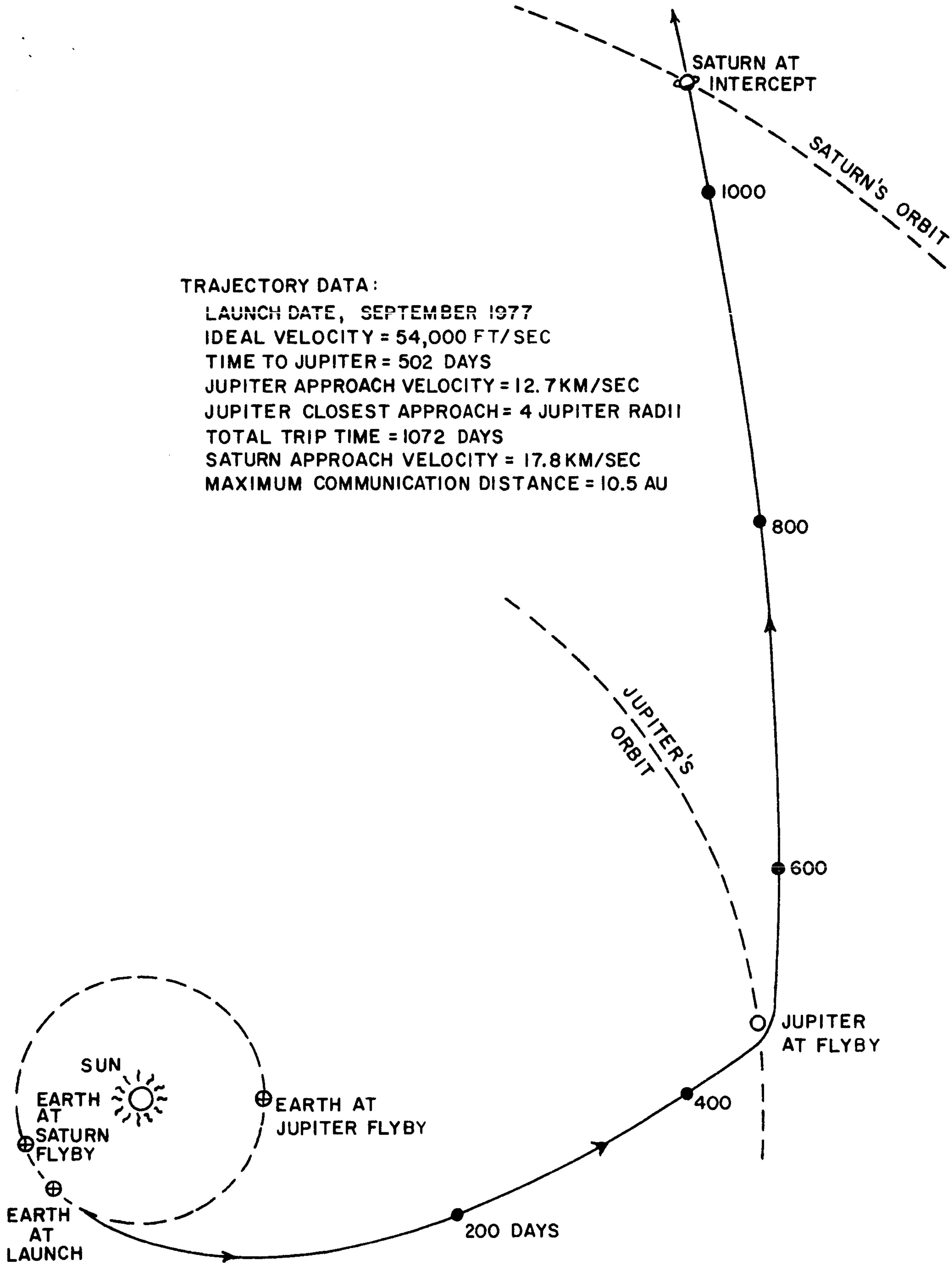

FIGURE 7. 3-YEAR JUPITER ASSISTED BALLISTIC FLIGHT TO SATURN

is 18 km/sec. A direct flight to Saturn in 3 years would require a higher ideal velocity of 56,000 ft/sec but would have a lower VHP of 15 km/sec.

Table 7 is a comparison summary between ballistic and gravity-assisted flight modes. The table shows, for a range of trip times, ideal velocity, ΔV; approach velocity, VHP; launch vehicles; and spacecraft weights.

Ideal Velocity

Consider first the change in ideal velocity and its effect on spacecraft weight for a flyby mission. Gravity-assisted flight represents a savings in ideal velocity over direct flight for all missions. A Jupiter assist always provides the smallest ideal velocity requirement, but the improvement with a Saturn assist is also notable. An 8-year Uranus mission can be used to illustrate how the ideal velocity improvement increases spacecraft weight. By using the SLV-3X-Centaur-Kick launch vehicle, the allowable spacecraft weight is 435 lb. Picking up a Saturn assist on the way increases the allowable weight to 720 lb. Finally, utilizing a Jupiter assist raises the weight capability to 1300 lb.

Approach Velocity

The second trajectory parameter considered in Table 7 is VHP, the hyperbolic approach speed at intercept with the target planet. Its value is a measure of rendezvous and orbiting difficulty. The direct flights have the lowest VHP

Table 7

COMPARISON TABLE OF DIRECT AND GRAVITY ASSIST TRAJECTORIES

	Saturn		Uranus		Neptune	
	TF=2.0 yrs	TF=3.5 yrs	TF=4.0 yrs	TF=8.0 yrs	TF=6.0 yrs	TF=10 yrs
Ideal velocity ΔV (ft/sec)						
Jupiter assist	60,900	52,000	60,000	51,000	60,000	52,200
Saturn assist	--	--	65,300	54,900	65,900	56,400
Direct	63,200	54,900	67,100	57,200	69,800	60,400
Approach velocity VHP (km/sec)						
Jupiter assist	22.4	12.8	24.8	12.0	26.2	14.9
Saturn assist	--	--	22.5	11.7	25.0	15.8
Direct	18.9	8.8	20.2	8.4	22.1	11.9
Flyby payload weight (lb)						
Launch vehicle	S-1B-C-K	SLV-3X-C-K	S1B-C-K	SLV-3X-C-K	S-1B-C-K	SLV-3X-C-K
Jupiter assist	2250	1125	2400	1300	2400	1100
Saturn assist	--	--	1500	720	1420	530
Direct	1780	720	1380	435	0	0
Orbiter spacecraft weight (lb)*						
Launch vehicle	S-V-C	S-1B-C-K	S-V-C	S-1B-C-K	S-V-C	S-V-C
Jupiter assist	0	930	0	475	0	0
Saturn assist	--	--	0	390	0	0
Direct	0	1670	0	940	0	1640

*Orbiter based on parabolic orbit, r_p = 3 planetary radii, ISP = 315 sec for retro maneuver

S-1B-C-K = Saturn-1B-Centaur-Kick
SLV-3X-C-K= SLV-3X-Centaur-Kick
S-V-C = Saturn V-Centaur

for all missions. A Jupiter assist yields the highest VHP for all missions except the 10-year Neptune flight.

The VHP can be combined with the flight time and the ideal velocity for any mission in Table 7 to determine how much payload can be placed in orbit about the target planet. This combination has been done in the last section of the table by assuming the stated retro propulsion capability, specific orbit, and launch vehicle. For short trip times the VHP for all flight modes is so high that the assumed propulsion system cannot deliver sufficient DV to obtain an orbit about the planet, even though launch vehicles as large as the Saturn-V-Centaur were considered. As expected, this situation improves with increasing flight time and concludes with the largest useful payload delivery into orbit at the long end of the flight time range for each planet.

For any given flight time, the ideal velocity requirements and VHP vary between direct and gravity-assisted trajectories. The question arises, "Is it better to reduce ideal velocity or VHP to place more payload in orbit?" Both parameters cannot be lowered simultaneously by changing the flight mode. It is apparent from Table 7 that it is better to reduce the VHP, since the direct flight mode always delivers more useful weight into orbit. In fact, in the case of a 10-year Neptune mission, for which a direct flight starting with a Saturn-V-Centaur puts 1640 lb of spacecraft into an orbit of Neptune, neither a Jupiter- nor a Saturn-assisted flight with the same launch vehicle places any useful payload into the same orbit. Hence, it is concluded that direct flights should be used to place ballistic payloads in orbits of the outer planets.

Launch Opportunities

Launch opportunities for gravity-assisted flights occur for periods of about 3 to 8 years. The launch windows are separated within these periods by one-year intervals because of the Earth's motion. The length of the period is determined by the planetary motions and by the miss distance requirements which must be imposed to ensure an intercept with the target planet. Figure 8 shows the positions of the outer planets in the next few decades. The fact that the target planets will lie in the same sector of the sky in the 1980s and 1990s accounts for the concentration of good gravity assist launch dates between 1975 and 1985.

Table 8 lists the next launch periods for all the gravity assist combinations considered as well as the beginning of the succeeding periods. Table 9 contains a list of synodic periods between all the gravity assist and target planet combinations. Notice that for any one mission, the time from the beginning of one launch period to the beginning of the next is about equal to the synodic period of the two outer planets considered in that mission. For example, launch periods start 14 years apart for the Earth-Jupiter-Uranus mission, and the synodic period between Jupiter and Uranus is 13.8 years.

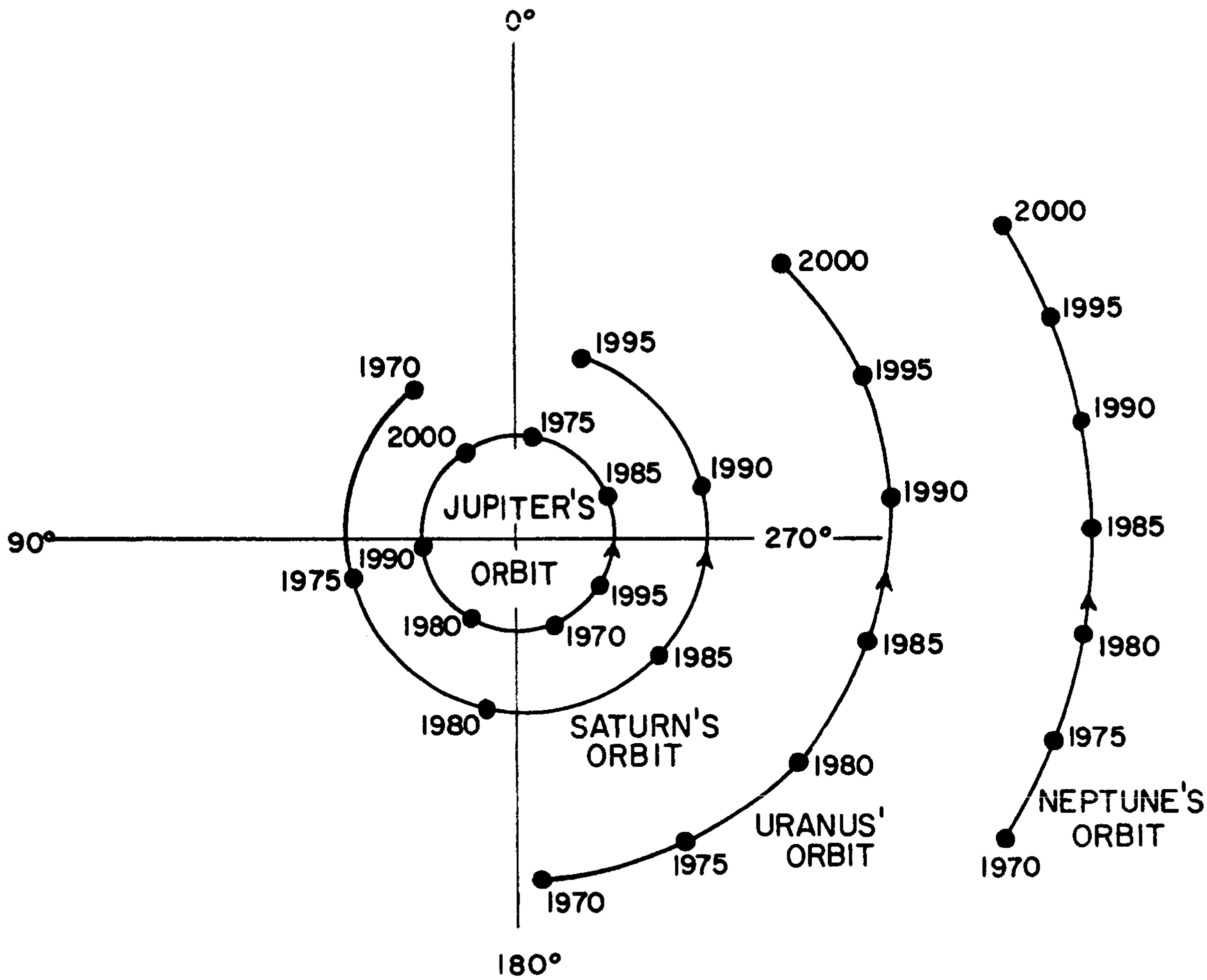

FIGURE 8. POSITIONS OF THE OUTER PLANETS PLOTTED ON THE ECLIPTIC PLANE.

Table 8

LAUNCH OPPORTUNITIES FOR GRAVITY-ASSISTED MISSIONS

Mission	Next Launch Period	Start of Following Period
Earth/Jupiter/Saturn	1976-1979	1996
Earth/Jupiter/Uranus	1978-1980	1992
Earth/Saturn/Uranus	1979-1985	2025
Earth/Jupiter/Neptune	1979-1981	1992
Earth/Saturn/Neptune	1979-1985	2015
Earth/Jupiter/Pluto	1976-1978	1989

Table 9

OUTER PLANET SYNODIC PERIODS

Planet Combination	Synodic Period (yrs)
Jupiter/Saturn	19.9
Jupiter/Uranus	13.8
Saturn/Uranus	45.4
Jupiter/Neptune	12.8
Saturn/Neptune	35.9

Gravity Assist Conclusions

It can be concluded from Table 7 that the techniques of gravity assist, within the time of flight range considered, provides significant improvement in spacecraft weight for fly-by missions.

For outer-planet orbiter missions, gravity assist should not be used. Direct flight always ensures a larger payload in orbit for equal flight times.

The gravity assist launch opportunities to the outer planets will be abundant during the period of 1976 to 1985. After 1985, a waiting period of from 10 to 38 years will exist before they again occur.

4.5 Low-Thrust Trajectories

The application of nuclear-electric propulsion systems (low thrust, high specific impulse) to upper-stage space vehicles offers a high performance potential when the mission energy requirements are very high. To measure this performance potential, a performance parameter J, somewhat analogous to the familiar ΔV of ballistic flight, has been defined (Melbourne 1961) as:

$$J = \int_0^{TF} a^2(t)\, dt \qquad (1)$$

where TF is the mission flight time and a(t) is the thrust acceleration associated with the mission trajectory.

Figure 9 illustrates payload vs. J curves for some possible nuclear-electric stages, based on a number of design studies (Beale et al. 1963, Pinkel et al. 1964). System parameters chosen for these examples are

Initial vehicle mass	M_o	= 20,000 lb
Power plant electrical rating	P_e	= 250 to 500 kw
Conversion efficiency	η	= 0.8
Specific mass of power plant	α	= 20 to 40 lb/kw
Structure and tankage*	S&T	= 1400 lb
Guidance and control	G&C	= 1000 lb

Note that 2400 pounds of the net payload (initial vehicle mass minus power plant mass) is used for S&T and G&C. The remaining net payload is available for communications, data handling, and experiments and is designated as the C&E payload.

The three types of low-thrust missions considered are

1. Flyby
2. Parabolic orbiter
3. Circular orbiter

In the flyby mission, the vehicle approaches the target planet along a hyperbolic trajectory with a specified miss distance. The approach velocity is not necessarily constrained. In the parabolic capture, or rendezvous, mission the velocity components of the vehicle and planet are matched at the time of intercept. It should be noted that from this parabolic energy

*Based approximately on S = 0.1 (mass of power plant), T = 0.05 (mass of propellant, maximum).

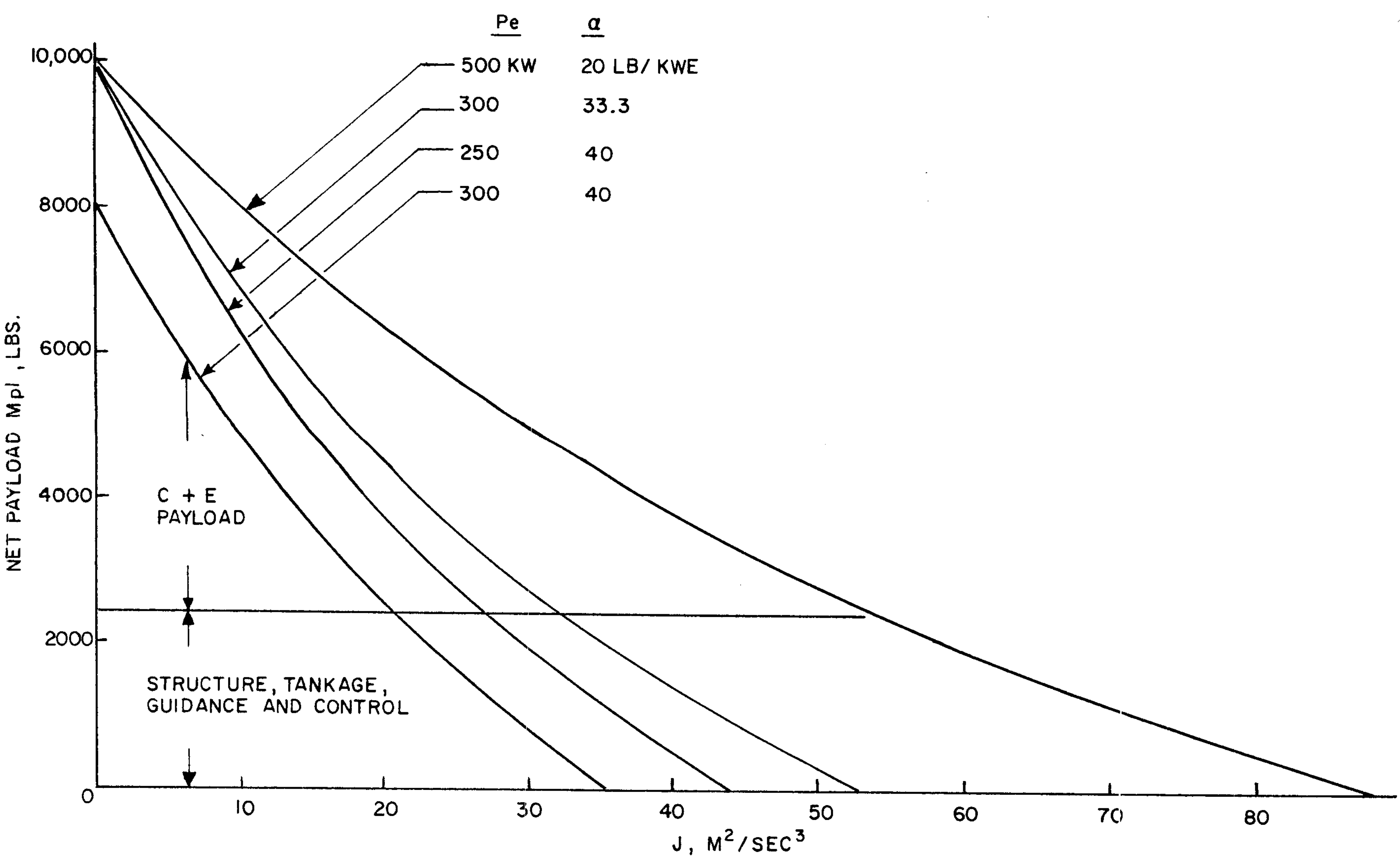

FIGURE 9. PAYLOAD VS. J CURVES FOR NUCLEAR ELECTRIC STAGE DESIGNS

condition only a very small energy increment is needed to achieve any loosely bound (highly eccentric) capture orbit. In the case of the circular orbiter mission, the vehicle continues thrusting until a circular orbit of specified radius is achieved.

It is assumed that the interplanetary vehicle begins its mission from a low-altitude (200-1000 N. mile) satellite orbit about Earth, where it has been placed by a high-thrust launch vehicle such as the Saturn 1B. The two phases common to each of the above mission types are the Earth escape phase and the heliocentric transfer phase. A planet capture phase is added in the case of the orbiter mission. In general, then, the total J requirement and flight time of a mission can be written as:

$$J = J_E + J_H + J_C \tag{2}$$

$$TF = TF_E + TF_H + TF_C \tag{3}$$

where the E, H and C subscripts refer to Earth escape, heliocentric transfer, and planetary capture, respectively. For preliminary mission analysis, it suffices to treat each phase of the mission as a separate "two-body problem," with the Earth, Sun, and target planet as successive central gravitational bodies. Overall results are then pieced together as in equations (2) and (3). Individual curves for J and for J_E, J_H, and J_C for each of the four planets are given in Appendix A.

Table 10 shows the flight time for flyby, parabolic and circular orbiters for Saturn through Pluto with 500-and 2000-lb C&E payload. Since there is abundant power available from the nuclear reactor, a 500 to 2000 lb C&E low thrust payload corresponds to a much bigger (perhaps 2000 to 8000 lb) ballistic payload. Thus a 500-lb C&E payload represents a very significant experimental capability. For the 500 lb C&E payload, the flight time range is 2 to 5 years for flyby, 3 to 8 years for parabolic orbiters, and 5 to 9 years for circular orbiters. In comparison, the Saturn V-Centaur can place 2000 lb in circular, $R_p = 3$ planet radii orbits in the following flight times.

Saturn	Not at all
Uranus	12 years
Neptune	22 years
Pluto	36 years

Table 10

FLIGHT TIMES FOR NUCLEAR ELECTRIC LOW THRUST FLIGHTS

(α = 40 lb/kw, P_e = 250 kw)

	500 lb C&E Payload			2000 lb C&E Payload		
	Flight time (yrs)			Flight time (yrs)		
	Flyby	Parabolic Orbiter	Circular Orbiter $R_p = 3$ planet radii	Flyby	Parabolic Orbiter	Circular Orbiter $R_p = 3$ planet radii
Saturn	2.15	3.45	4.95	2.55	4.05	6.05
Uranus	3.60	5.65	6.70	4.25	6.65	$>$ 8
Neptune	4.95	7.75	9.05	5.85	9.15	$>$ 10
Pluto	5.05	7.90	8.75	6.00	$>$ 9	$>$ 10

5. MISSION CONSTRAINTS

5.1 Guidance

Introduction

The direct ballistic flyby missions place a lower bound on guidance requirements for all outer-planet missions. Orbiter missions will have more stringent guidance requirements, depending on the accuracy desired in attaining a particular capture orbit. Gravity-assisted ballistic missions will have even more stringent guidance requirements because a small velocity error at the gravity assist planet can result in a large position error at the target planet. Since low-thrust stages will probably have an active guidance system operating throughout the thrusted part of the flight, it is not clear whether target intercept will add any special requirements.

The approach taken in this study was to assume reasonable values of the various sources of error and to investigate the effect of these errors on a typical mission to each of the four outer planets. This approach yields order-of-magnitude results which are useful in estimating the guidance requirements for the entire spectrum of possible missions.

It is concluded from this section that the target miss distance after a single midcourse correction can be expected to be within a few planet radii for Saturn, Uranus, and Neptune. Since this miss distance should be acceptable for flyby missions, guidance for this type of mission is not a severe problem.

Injection Errors and Midcourse Correction

A post-injection or midcourse trajectory correction is required to compensate for a target miss* resulting from an error in establishing the proper hyperbolic velocity vector $\overline{VHL}$. This error is attributed to the launch guidance system. The assumed errors are 10 m/sec along the outgoing asymptote of the escape hyperbola and 1 milliradian in both right ascension and declination of the asymptote (these errors and all further numbers given are to be taken as 1σ values). Table 11 lists the resulting target errors and midcourse correction requirements for several direct ballistic flyby missions. Flight times to Saturn, Uranus, Neptune, and Pluto are, respectively, 2.7, 6.9, 13.8, and 13.8 years. These flights represent minimum VHL launches for the given flight times. Target miss, ΔB, ranges from about 1 million kilometers for the Saturn flight to 4.4 million kilometers in the case of Pluto. The corresponding flight time errors range from 2.5 to 33 days.

Two types of midcourse velocity corrections are considered. The fixed-time-of-arrival (FTA) correction nulls both ΔB and ΔTF. The variable-time-of-arrival (VTA) correction nulls only ΔB; the extra degree of freedom is used to minimize the ΔV_c required. Assuming a single midcourse correction made 10 days after injection, Table 11 lists the ΔV_c required for each of the two correction policies. The FTA

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*Miss distance B is defined as the normal displacement of the planet approach asymptote from the planet center.

Table 11

TARGET ERRORS AND MIDCOURSE CORRECTION REQUIREMENTS FOR SEVERAL DIRECT BALLISTIC FLYBY MISSIONS

Mission Trajectory	Uncorrected target errors		Midcourse velocity correction (m/sec)	
	ΔB (10^6 km)	ΔTF (days)	ΔV_c (FTA)	ΔV_c (VTA)
Saturn				
T_L : 7/16/70 TF : 2.7 years VHL: 11.7 km/sec	0.978	2.48	18.1	13.7
Uranus				
T_L : 12/27/75 TF : 6.9 years VHL: 12.2 km/sec	1.78	8.75	18.4	13.3
Neptune				
T_L : 1/26/76 TF : 13.8 years VHL: 12.2 km/sec	2.39	25.8	18.3	15.4
Pluto				
T_L : 12/12/75 TF : 13.8 years VHL: 21 km/sec	4.43	32.9	24.4	16.7

All results are 1σ values
Assumed errors in launch asymptote; 10 m/sec, 1 mrad
Midcourse correction made 10 days after injection

correction is about 18 m/sec for the Saturn, Uranus, and Neptune flights and 24 m/sec for the Pluto flight. The minimum ΔV_c correction is seen to offer a savings of 3-8 m/sec if VTA guidance is allowable. Computer results show that the time of flight difference between FTA and VTA guidance ranges from 1.4 days for the Saturn flight to 26 days for the Neptune flight. If this time difference does not conflict with other mission constraints (such as DSIF tracking visibility during target encounter), the VTA correction is preferable. In fact, results show that ΔV_c can sometimes be reduced even further if the VTA correction is made later in flight; i.e., the optimum time to execute the VTA correction is not necessarily soon after injection, as is the case for the FTA correction.

Post-Midcourse Correction Errors

The target miss remaining after the midcourse maneuver is due to three error sources: (1) spacecraft orbit determination error due to DSIF tracking inaccuracies, (2) inaccuracy in executing the midcourse maneuver, and (3) uncertainty in the knowledge of planet position including ephemeris errors and uncertainty of the astronomical unit-kilometer conversion factor. Table 12 lists the target miss, ΔB, due to each source of error for the four trajectories previously considered. The error magnitudes assumed for this example are:

(1) DSIF Tracking Error (Clarke et al. 1963)

$\Delta V_1 = 0.1$ m/sec

Table 12

MISS DISTANCE ERRORS REMAINING AFTER MIDCOURSE MANEUVER FOR SEVERAL DIRECT BALLISTIC FLYBY MISSIONS

Mission Trajectory	Midcourse ΔV Execution Error (km)	Spacecraft Orbit Determination Error (km)	Planet Position Error: AU Uncertainty (km)	Planet Position Error: Ephemeris (km)	RSS Total ΔB (km)	RSS Total ΔB (planet radii)
Saturn						
T_L : 7/16/70 TF : 2.7 years VHL: 11.7 km/sec	9,100	3000	2500	600	9900	0.17
Uranus						
T_L: 12/27/75 TF : 6.9 years VHL: 12.2 km/sec	16,000	4600	4900	1000	17000	0.67
Neptune						
T_L : 1/26/76 TF : 13.8 years VHL: 12.2 km/sec	20,000	6900	3600	1500	22000	0.88
Pluto						
T_L : 12/12/75 TF : 13.8 years VHL: 21 km/sec	39,000	13000	20000	3000	46000	

All results are 1σ values

ΔV_2 = 0.03 m/sec

ΔV_3 = 0.003 m/sec

where $\underline{V}$ = $\underline{VHL}$ and ΔV_1, ΔV_2 and ΔV_3 are, respectively, the uncertainty components in declination, right ascension, and along $\underline{VHL}$. These errors represent several days of data gathering and statistical smoothing and are assumed to apply to all trajectories.

(2) Midcourse ΔV Execution Error

0.1 m/sec in each component

(3) AU Uncertainty (Haynes et al. 1965)

500 km

(4) Ephemeris Errors (Naqui and Levy 1963)

Saturn - 600 km
Uranus - 1000 km
Neptune- 1500 km
Pluto - 3000 km

As seen from Table 12, the largest component of ΔB is due to the error in executing the midcourse correction. This component varies from 9100 km in the case of the Saturn flight to 39,000 km for the Pluto flight. DSIF tracking errors contribute 3000-13,000 km to the target miss, and the uncertainty in the astronomical unit causes an additional error of 2500 to 20,000 km. Taking the 3σ values of the root-sum-square total ΔB as the maximum error, the target miss remaining after the midcourse correction can be expected to be within a few planet radii for missions to Saturn, Uranus, and Neptune. Pluto, having a relatively small radius, is the exception. If this accuracy is acceptable from the standpoint of mission success, it may be concluded that the guidance requirements for missions to the outer planets are easily met by a single trajectory correction made shortly after launch and a modest propulsion capability (less than 100 m/sec). An onboard terminal guidance system may be necessary only in the case of gravity assisted flyby flights and orbiter missions.

5.2 Attitude Control

Attitude control will be required on outer planet missions for the orientation of experiments, for the correct execution of midcourse maneuvers, and for the maintenance of a communications link with the Earth.

The experimental requirements for attitude control are minimal throughout the long interplanetary phases of the missions. It probably will not be necessary for the experiments to be in any fixed inertial frame, although the direction in which the instruments point during a measurement should be determinable. In the planetary intercept phase, which may last for about 12 hours, most of the scientific instruments should be oriented toward the planet. This requirement is in conflict with communication pointing if real time transmission is required unless a science platform is used or unless the spacecraft antenna has a separate pointing platform with large angular motion.

The midcourse maneuvers will require the accurate alignment of the spacecraft for a relatively short period of some ten minutes. However the attitude control system must provide and maintain the alignment throughout the maneuver. For flyby flights one midcourse correction will be necessary (see Section 5.1).

Communications between the spacecraft and Earth will probably be required for a few hours every day, and therefore the antenna should be continuously oriented towards the Earth. The beamwidth of the spacecraft antenna will determine how much attitude control will be necessary to maintain Earth orientation, and a trade-off will be required for the final spacecraft design.

Two basic types of attitude control seem possible for outer planet missions: spin stabilization and full three-axis stabilization. In the spin mode, the spin axis would probably have to be oriented toward the Earth with the communications antenna aligned with the spin axis.

The attitude control requirements are to precess the spin axis along the flight path to maintain Earth orientation This will involve a total precession angle of about 180°. Difficulties may be encountered in aligning the spacecraft for the midcourse maneuver with only a two-axis reference. One solution is to lock the spin axis for the first few days and only spin up the spacecraft after the midcourse correction. Alternatively, the midcourse correction can be done some way along the trajectory when the normal orientation of the spacecraft with respect to the target and the Earth is coincident with the direction of the required midcourse thrust vector. In general, this latter technique is the most costly in midcourse propulsion which far outweighs any saving in attitude control.

Three-axis stabilization minimizes the problems associated with the other spacecraft systems but puts an onus on the reliability of the attitude control system. This will result in a heavier attitude control system than the spin stabilization but not necessarily a heavier total spacecraft. The weight for the complete three-axis attitude control system including the control for a science platform is estimated to be 100 lb for a 1000-lb total spacecraft weight. For a 2000-lb spacecraft, 150 lb is estimated for the attitude control system. Variation in propellant weight as a function of time of flight is assumed to be a second order effect for the purposes of this study.

5.3 Power Supply

Because of the large distance of the four outer planets from the Sun, solar cell power supplies will not be adequate. Therefore nuclear power supplies will be essential, either isotopic for power levels up to approximately 1 kw or a nuclear reactor for higher levels. For Radioisotope Thermal Generation (RTG) units, it is estimated that the power source can be engineered to meet the particular power demand for each mission for a specific weight of approximately 1 lb/watt of useful power.

No attempt has been made to construct the power profiles of the missions. However, some general comments can be made. During intercept, all planetary experiments will be on and the data will probably be stored. The communication system will operate again after the intercept phase has been completed. Probably the greatest demand made on the power supply will be during planetary acquisition and during Earth acquisition after intercept. However, it should be possible to switch off nearly all the experiments at these times if it is necessary to conserve power.

The specific weight for RTG power supplies (1 lb/watt) includes an allowance for shielding to protect the spacecraft from radiation. Accurate assessment of shielding weights will only be possible after the isotope used in the RTG has been chosen and the physical location of the power source or sources has been decided. Furthermore, the tolerance of spacecraft systems and experiments to radiation is not adequately known at present. However, a gross estimate of 0.1 lb/watt can be made to cover the requirement for shadow-shielding the spacecraft.

5.4 Thermal Control

Temperature control of spacecraft on missions to the outer planets represents a significant problem. Many of the components of the spacecraft must be maintained within prescribed temperature levels, each in a manner compatible with the others and within the usual weight and size restraints.

For outer-planet missions, active temperature control (heater elements, louvers, etc.) will almost certainly be needed. The RTG unit will be a significant source of heat.

From elementary consideration of heat balance, the heat absorbed, g_A, plus the heat generated internally g_I equals the heat radiated to space, g_R, or

$$g_A + g_I = g_R.$$

The heat radiated to space is

$$g_R = \sigma\epsilon A_R T^4$$

where σ = 5.67 x 10^{-12} watts/cm^2 - °K^4 (Stefan-Boltzman constant)

ϵ = emittance of radiating surface

A_R = area of radiating surface (cm^2)

T = temperature of radiating surface (°K)

The heat absorbed from the Sun, for example, is

$$g_A = \frac{S\alpha_s A_s F}{R^2}$$

where

S = 0.135 watts/cm^2

α_s = absorption

A_s = area of absorbing surface (cm^2)

F = fraction of spacecraft effective in absorbing radiation, a geometric view factor

R = distance from the Sun (AU).

By using the above formulas, the temperature of a spacecraft is given by:

$$T = \left(\frac{S\alpha_s \, A_s \, F}{R^2 \, \sigma\epsilon \, A_R} + \frac{q_I}{\sigma\epsilon \, A_R}\right)^{1/4} .$$

Figure 10 shows T plotted against R for black ($\alpha = \epsilon = 1$) spherical and flat plate spacecraft, and with 100-watt heat sources in meter-sized spacecraft, the temperature averages -100 to -120°C. Without the heat source, the average spacecraft temperatures are 100° or more colder.

5.5 Communications

The overall problem of deep space communications is overcoming the large signal attenuation that occurs because of the extreme distances the signal must travel. There are four major limitations to the communication system: (1) the maximum available transmitter power, (2) the size and characteristics of the transmitting and receiving antennas, (3) the free space propagation loss, and (4) the sensitivity of the receiving system. The upper bound on transmitter power is dictated by weight, physical size, and available primary power plus the capability of the transmitting tubes. Antenna size depends primarily upon the weight and physical constraints of the spacecraft and launch vehicle. Once the gain-power product of the transmitting system has been determined, trade-offs between antenna size and the transmitter power can be made.

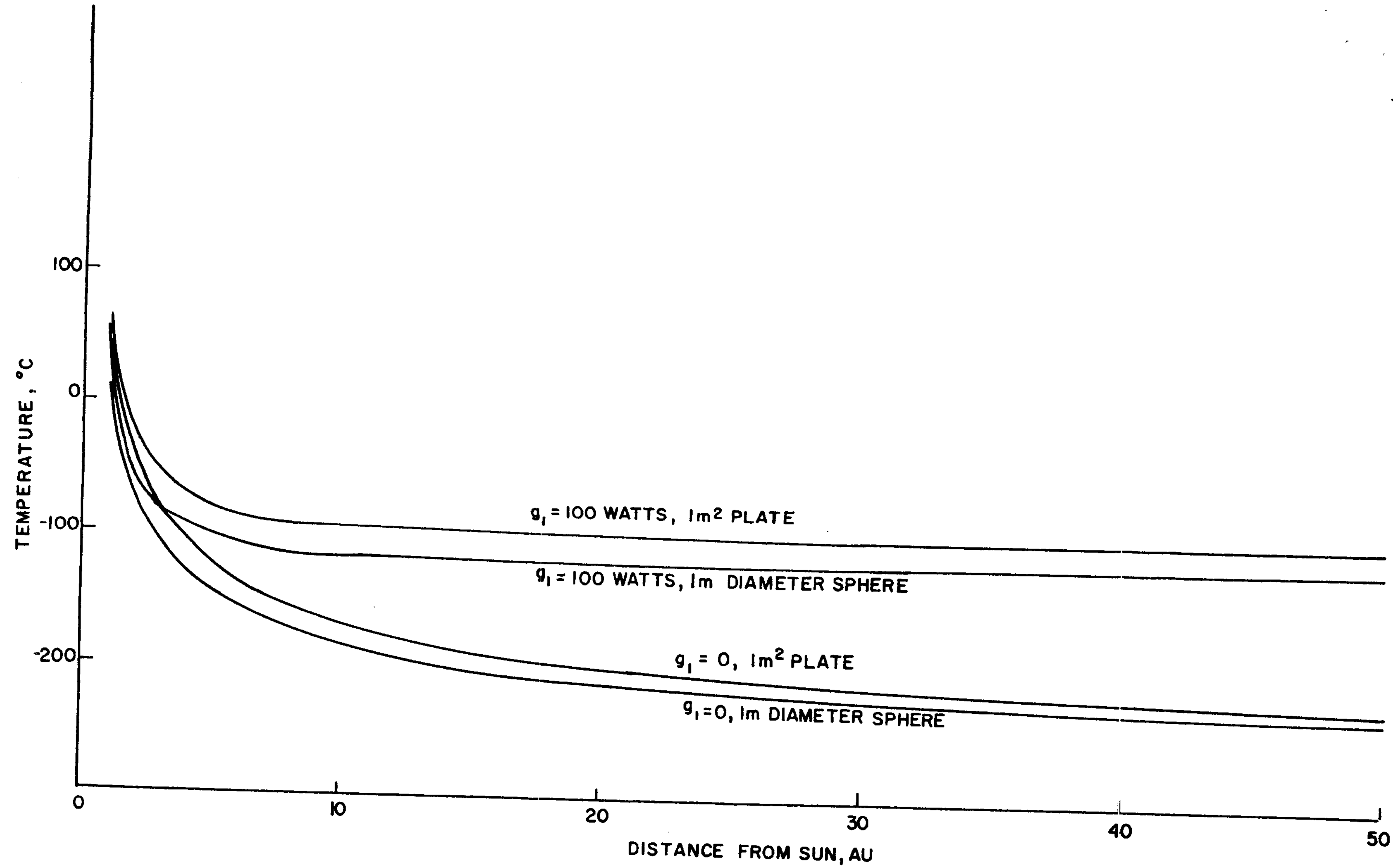


FIGURE 10. TEMPERATURE OF FLAT PLATE AND SPHERICAL BLACK, CONDUCTING SPACECRAFT WITH AND WITHOUT 100 WATT HEAT SOURCES

For a given transmitted power-antenna gain product there exists an optimum transmitter power and antenna gain from the point of view of minimizing the weight of the total transmitting system (Stein 1966). By using this methodology, Figures 11, 12, and 13 were derived. The basic assumptions, based on the projected state of the art in 1975 to 1985 are:

Miscellaneous losses = 8 db

85 ft DSIF antenna, system noise at 40°K, G_R = 60 db, ϕ_K = 212.6 db w/cps

210 ft DSIF antenna, system noise at 25°K, G_R = 52 db, ϕ_K = 214.6 db w/cps

Spacecraft antenna weight = 0.25 lb/ft^2 (parabolic antenna with aluminized phenolic honeycomb and fiberglass skin)

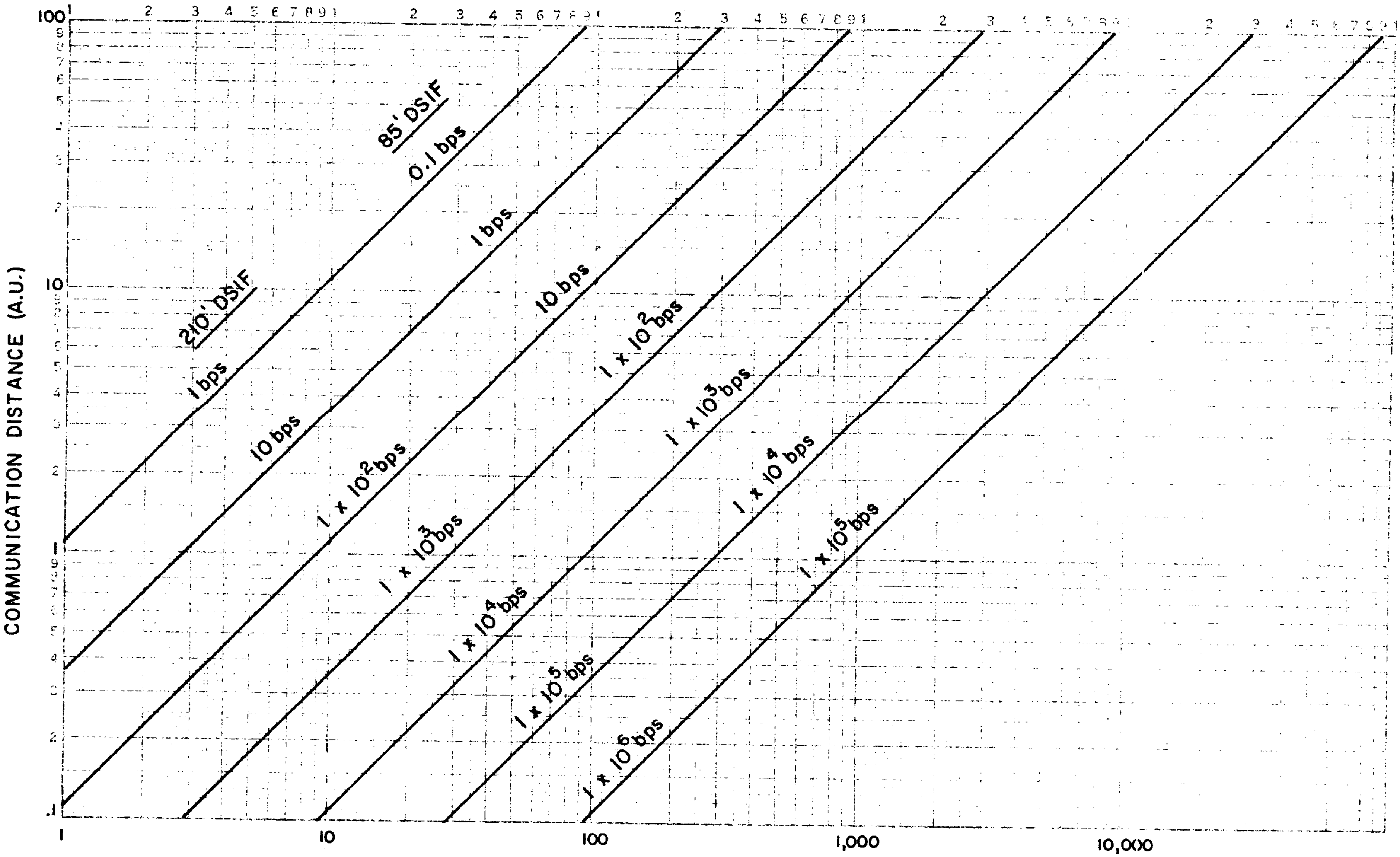
Power supply weight = 1 lb/watt of raw power with 30% efficiency

Coherent PSK with bit error probability = 1×10^{-3}

where G_R is receiver gain and ϕ_K is receiver noise spectral density.

For example, assume 20 bits/sec transmitted. Then Table 13 can be constructed for the 210 ft DSIF.

FIGURE 11: MINIMUM TRANSMITTING SYSTEM WEIGHT VS. TRANSMISSION DISTANCE AS A FUNCTION OF INFORMATION RATE



MINIMUM TRANSMITTING SYSTEM WEIGHT (W_{MIN} - LBS)
(INCLUDES ANTENNA, TRANSMITTER AND POWER SUPPLY WEIGHTS)

FIGURE 12: MINIMUM TRANSMITTER POWER VS. TRANSMISSION DISTANCE AS A FUNCTION OF INFORMATION RATE

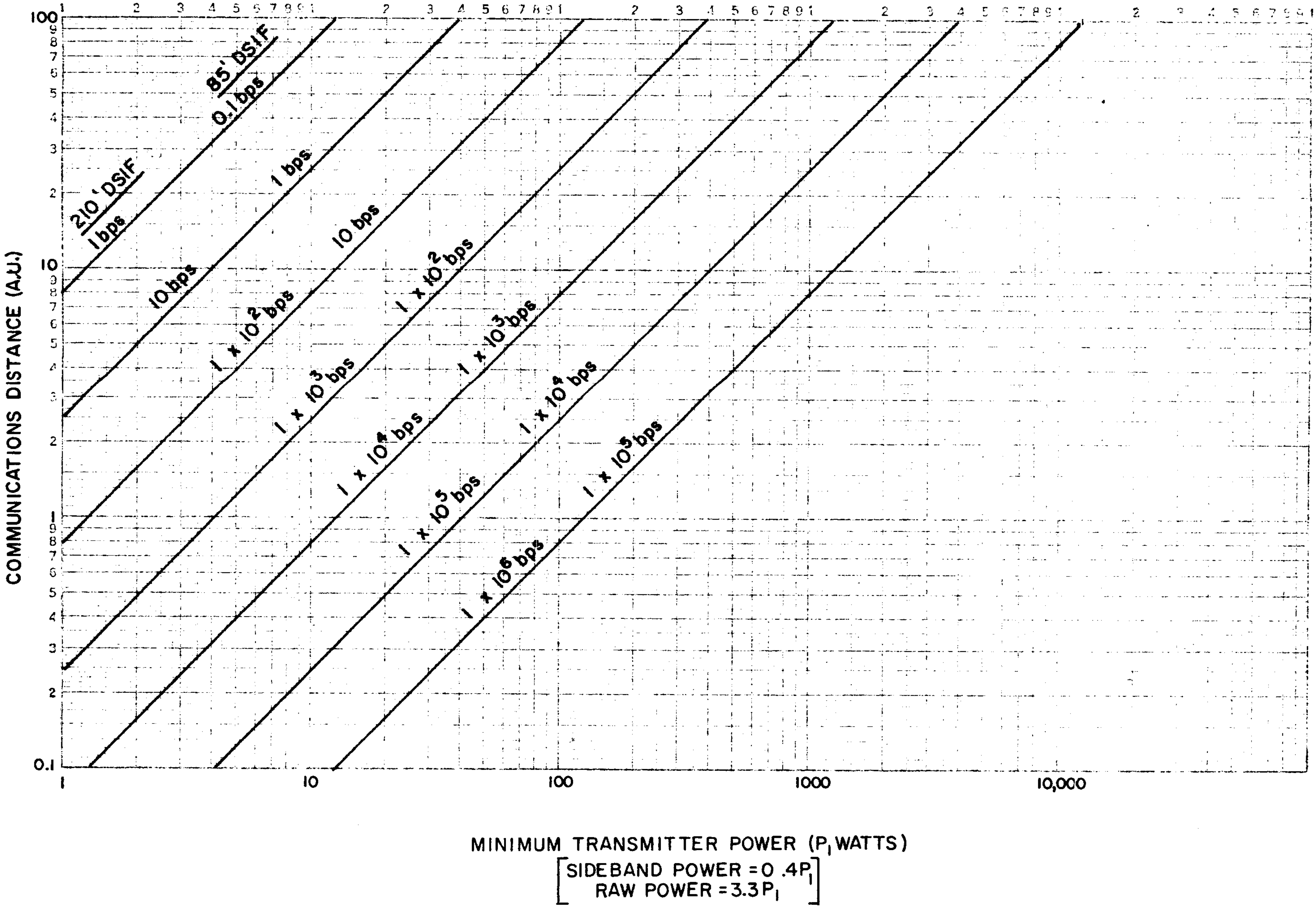


FIGURE 13: ANTENNA DIAMETER VS. TRANSMISSION DISTANCE AS A FUNCTION OF INFORMATION RATE

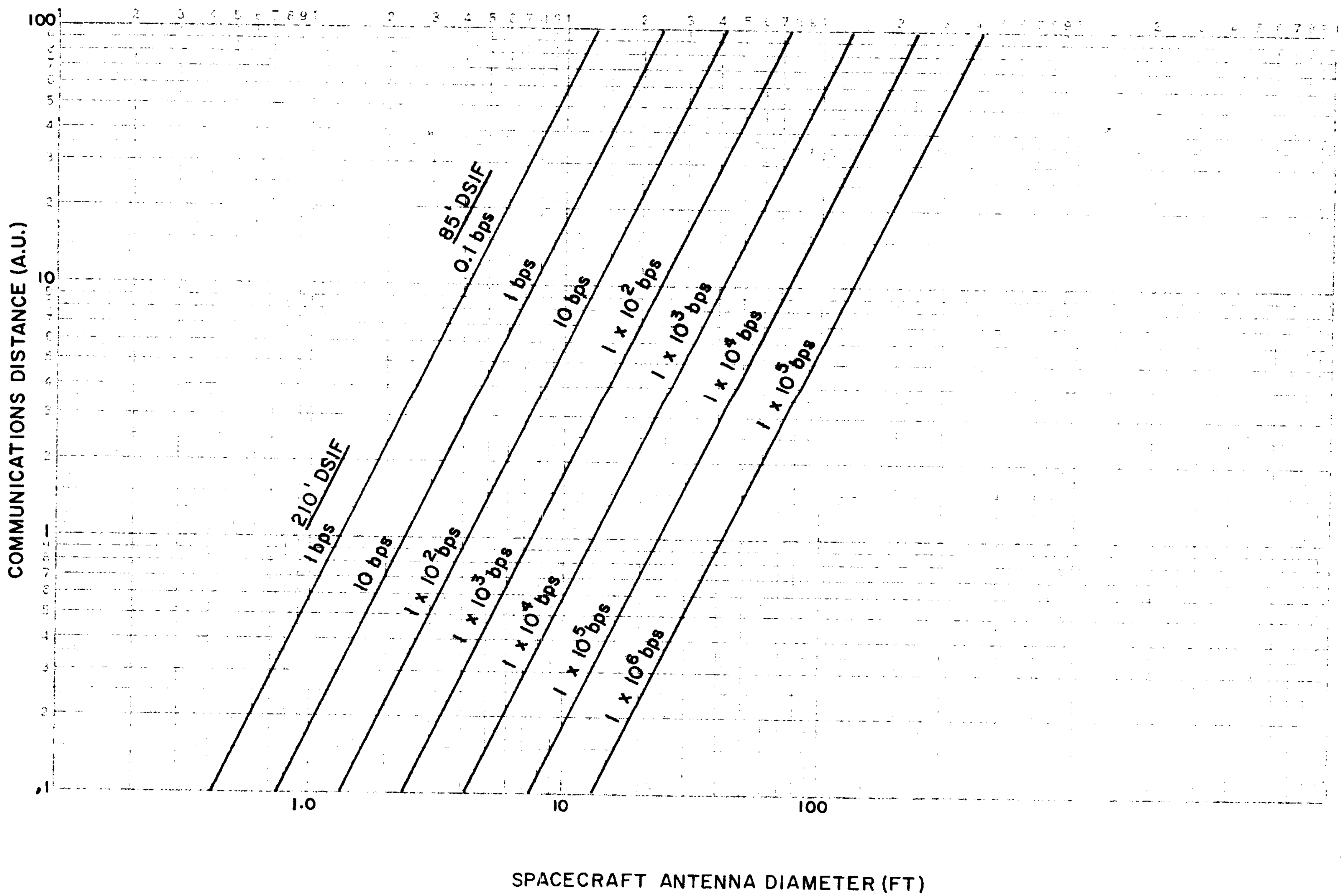


Table 13

COMMUNICATIONS DATA

	Saturn	Uranus	Neptune and Pluto
Communications distance (AU)	10	20	31
Total transmission system (lb)	40	70	130
Transmitted power (watts)	5.6	11	17
Antenna diameter (ft)	9	13	16
Power supply weight (lb)	18.5	36	56
Antenna and transmitter weight (lb)	21.5	34	74

The total transmission system weight, from Figure 11, is the weight of power supply, antenna, and transmitter. Transmitted power is taken from Figure 12, and antenna diameter is taken from Figure 13. The power supply weight is 1 lb/watt of raw power, and the antenna and transmitter weight is total transmission weight minus power supply weight.

6. PAYLOADS

6.1 Introduction

In this section some aspects of a basic payload for flyby missions are discussed. This payload could be used for Saturn, Uranus or Neptune and with some modification for Pluto also. All of the planetary instruments could operate at a distance of a few planet radii from the planets. The same class of payload could be used on an orbiter mission, which would have the advantage of obtaining far greater amounts of data than from a flyby which spends only a few hours near the target planet.

The science payload is divided into interplanetary experiments and planetary experiments. The total experimental weight is 85 lb, the power required 100 to 150 watts for experiments and communications. The data rate is 8 bits/sec for 2 hours a day interplanetary, 20 bits/sec during the encounter, and perhaps 20 bits/sec for 4 hours after encounter for each television picture transmitted. The total payload weight would be of the order of 1000 lb or less.

6.2 Interplanetary Experiment

Table 14 summarizes the interplanetary experiments. During the interplanetary phase it is assumed that transmission from the spacecraft will occupy no more than 2 hours a day. Data from the magnetometer and plasma probe will be transmitted in real time for the two hours and provide a sample of the conditions in interplanetary space. The other instruments

Table 14

INTERPLANETARY EXPERIMENTAL PAYLOAD

Experiment	Lb	Watts	Bits/sec	Remarks
Magnetometer				
Fluxgate 0.1-10γ	2	0.1	3	Real time. 1 measurement every 5 secs (3%)
Helium 0.1-100γ	6	5	0.5	Real time when required for calibration
Plasma Probe				
Faraday Cup (two) at right angles	4	1	1.5	1 energy level per 5 secs, 20 levels in all. Real time. Two detectors alternately.
Micrometeorite Detector (shield in asteroid belt) (velocity, mass and size)	5	0.1	0.1	Store data for each collision. Transmit once per day.
Cosmic Ray Telescope 10 mev-1 bev	5	1	0.1	Store data on each cosmic particle detected. Transmit once per day.
Solar Proton Detector 100 kev-10 mev	5	1	0.1	Store 24 hrs. Transmit once per day.
Ionization Chamber Integrating	3	0.2	0.1	Integrated dose transmitted once per day.
Engineering data			3	
Total Interplanetary Expts	30	8.4	8.4	

will record and store data for the full 24 hours and relay it during the 2 hour transmission period. Note that the total weight, power, and bit rate are 30 lb, 8.4 watts and 8.4 bits/sec.

6.3 Planetary Experiments

Table 15 summarizes the planetary experiments.

Table 15

PLANETARY EXPERIMENTAL PAYLOAD

Experiment	Lbs	Watts	Bits/sec	Remarks
Helium magnetometer .001-1 gauss	--	--	--	Measurement every 5 secs
Plasma probe	--	--	--	Measurement every 5 secs 20 levels
Ionization chamber	--	--	--	Integrated dose
IR spectrometer 2-50μ 1μ resolution	10	10	5	1 channel/sec; 10 mins/frame. Real time at intercept.
Visible, UV spectrometer (with polarimetry) 10,000-1000 Å	20	10	5	4500 bits/frame; 15 mins/frame. Real time at intercept.
Photometry	5	1	1	10^4 bits total
Television (Mariner type, 240 000 bits/frame)	10	10	20	4 hours transmission/frame after intercept.
Microwave radiometry (1.5 dish, 1-10 cm)	10	1	1	60 bits/min. Real time at intercept.
Planetary experiments	55	32	12	
			+20 for television	

6.4 Payload Summary

In the communications section (Section 5.5) the necessary transmitted power was shown to be 5.6, 11, and 17 watts for Saturn, Uranus, or Neptune or Pluto, respectively. Multiplying these values by 3.3 to get raw power and adding 40 watts for experiments yields the science and communications raw power requirements of:

60 watts for Saturn
75 watts for Uranus
95 watts for Neptune or Pluto.

To this should be added a small amount for other spacecraft functions to yield a total raw power requirement of approximately 100 to 150 watts.

Two recent mission studies for Jupiter missions provide some guidance for total spacecraft weight. In a General Dynamics study (Have et al. 1966) a 3-axis stabilized spacecraft with 115 lb of science weighed 1058 lb, and one with 37 lb of science weighed 717 lb. A JPL study (Haurlan 1966) considered a 3-axis stabilized spacecraft with 12 lb of science weighing 668 lb. From these and the estimates given earlier in this report, a spacecraft weight of less than 1000 lb for the 85 lb of science appears reasonable. A very minimum science mission (Advanced Planetary Probe), with 10 to 15 lb of particles and fields measurements, would save a few hundred pounds in payload weight and spacecraft complexity. However, the 85 lb of science class of mission would be far more useful scientifically.

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8. NOMENCLATURE

The following symbols are used in various sections of this report. Symbols used only in one section of the report (and defined there) are not included here.

α	Specific mass of nuclear-electric power plant (lbs/kwe)
C and E payload	Net payload available for communications and experiments on thrusted stage = M_{PL} less structure, tankage, guidance and control.
D	Direct ballistic flight mode
DV	The velocity increment required to transfer a spacecraft from a planet approach hyperbola to into an orbit around the planet
ΔV	The ideal velocity, that is, the total velocity increment which must be given to the spacecraft on leaving Earth: $\Delta V = \sqrt{(36{,}178)^2 + (VHL)^2} + 4000$ Here, 36,178 ft/sec is the characteristic velocity for Earth escape launching from Cape Kennedy and 4000 ft/sec is a correction for gravitational and frictional losses during launch
GA	Gravity assist ballistic flight mode
J	$\int_0^{TF} [a(t)]^2 dt$, where a(t) is the thrust acceleration: J is thrusted trajectory parameter analogous to ΔV
J_E, J_H, J_C	J for Earth escape, heliocentric transfer, or planetary capture
M_{PL}	Net payload for thrusted stage = initial vehicle mass less mass of power plant and propellant
R_p	Periapsis of orbit around a planet (in planet radii)

T	Nuclear electric low-thrust flight mode.
TF	The total time of flight for a mission.
TF_E, TF_H, TF_C	TF for Earth escape, heliocentric transfer, or planetary capture
VHP	The hyperbolic excess speed (approach velocity) at the planet.

Appendix to Report M-11

TRAJECTORY DATA FOR SATURN, URANUS, NEPTUNE, AND PLUTO

October 1966

LIST OF FIGURES

LIST OF FIGURES (Cont'd)

LIST OF FIGURES (Cont'd)

LIST OF FIGURES (Cont'd)

Appendix A

TRAJECTORY DATA FOR SATURN, URANUS, NEPTUNE, AND PLUTO

This appendix consists of a set of 47 curves of the key parameters for direct ballistic, gravity assist ballistic, and nuclear electric low-thrust flights to the planets Saturn through Pluto.

Figure A-1 shows payload vs. ΔV curves for the SLV-3X-Centaur-Kick, Saturn-1B-Centaur, Saturn-1B-Centaur-Kick, and Saturn-V-Centaur launch vehicles. Note that payload means total payload weight on the launch pad. Figure A-2 is a spacecraft weight/payload weight conversion curve, which plots injected spacecraft weight against payload weight. For example, a payload of 1000 lb corresponds to a spacecraft weight of 850 lb; most of the 150-lb difference is shroud and adapter.

Figure A-3 shows payload vs. J for four possible nuclear electric low-thrust stages. System parameters are:

Initial vehicle mass	M_o	= 20,000 lb
Power plant electrical rating	P_e	= 250 to 500 kw
Conversion efficiency	η	= 0.8

Specific mass of power plant	α = 20 to 40 lb/kw
Structure and tankage*	S&T = 1400 lb
Guidance and control	G&C = 1000 lb

Note that 2400 pounds of the net payload (initial vehicle mass minus power plant mass) is used for S&T and G&C; the remaining net payload is available for communications and experiments and is designated as the C&E payload.

Figure A-4 shows the J requirement J_E for Earth escape from a 1000-N. mile orbit (it is assumed that the stage has been injected into Earth orbit by a Saturn 1B class launch vehicle). The values of J_E (and later J_C) have been obtained from analytic formulas which give excellent agreement with numerical integration solutions of the escape and capture trajectories (Melbourne 1961); a tangential thrust program is assumed. Since this J_E curve (as well as the J_C curve) is only weakly dependent upon the acceleration time history, this curve can be taken as a very good approximation for all accelerations of interest. A constant thrust program with the indicated α_o ISP has been assumed.

Figure A-5 shows the J requirement J_C for capture and each of the planets, assuming a final circular orbit at 3 planet radii. These curves are similar to the J_E curves; in the case of constant thrust propulsion, the acceleration level at the beginning of the capture phase would depend upon α_o ISP and the time spent in thrusting during the heliocentric phase of

*Based approximately on S = 0.1 (mass of power plant), T = 0.05 (mass of propellant, maximum).

the flight. The figure shows that Saturn, because of its large mass, has the most severe capture requirements and that the capture times of interest range from 100 to 450 days.

Figure A-6 shows the approach velocity for minimum J flyby trajectories, utilizing the variable thrust mode of flight. The minimum J flight occurs from 2 levels of optimization.

1) It is assumed that the vehicle is launched with Earth in such a longitude as to minimize the value of J_H required to reach the given target planet in the given transfer time.
2) The thrust program is optimized to give the minimum value of J_H for each transfer time.

Figures A-7 through A-47 show a variety of parameters and payloads for various flights to Saturn, Uranus, Neptune and Pluto. For Saturn, Uranus, and Neptune, the planets were assumed to be in circular orbits in the ecliptic plane. This assumption is quite good (within 1000 ft/sec in ΔV) for these planets. For Pluto, the ballistic calculations assume a launch in 1975 and use the 3-dimensional orbital elements for the planet. For thrusted flights to Pluto, a perihelion encounter was assumed (perihelion occurs in 1989 at a latitude of about 15°).

Figure A-7 shows the ideal velocity for direct and Jupiter-assist flights to Saturn, plotted against flight time. The ideal velocity savings from using gravity assist can be seen directly from this curve.

Figure A-8 shows the payloads for direct Saturn flights, plotted against flight time for various launch vehicles. Because of the leveling of the curves, it is clear that not much is gained in stretching the Saturn flight to longer than 3 or 4 years. The next figure is similar to the previous one except that a Jupiter assist is assumed.

Figure A-10 shows approach velocity, VHP, for direct and Jupiter assisted flights to Saturn. The penalty in approach velocity in gravity assist flights can be seen in this curve.

Figure A-11 shows the velocity increment, DV, required to take a spacecraft from an approach velocity, VHP, into a parabolic* orbit around the planet, with periapsis at the given distance R_p planet radii from the planet center. For the parabolic rendezvous it is assumed that the incoming trajectory has the desired periapsis and that the velocity increment is applied at periapsis.

Figure A-12 is similar to the previous figure except that a circular final orbit rather than a parabolic final orbit is assumed. Note that the DV required for circular orbits is very much higher than that required for parabolic orbits with the same R_p.

Figure A-13 shows bus weight and parabolic orbit around Saturn, assuming an R_p of 3, an ISP of 315, and a retro-engine

*By a parabolic rendezvous we mean a very loosely bound elliptic orbit with aphelion at infinity.

and tank weight of 14% of the propellant weight. The calculation of bus weight is done in the following manner. For a given flight time, for example 4 years, the ideal velocity requirement of 54,200 ft/sec is found from Figure A-7 for a direct flight. (As was mentioned in the text, gravity-assisted flights are inferior to direct flights for orbital missions. Therefore only direct orbital missions are considered.) From Figure A-1, the launch payload is 25,000 lb for the Saturn-V-Centaur launch vehicles. By using Figure A-2, this is converted into a spacecraft weight of 22,800 lb. From Figure A-10, the 4-year flight time is found to have a VHP of 7.4 km/sec, and from A-11 this VHP is seen to correspond to a DV of 1.25. The payload fraction, f_p, is found to be 0.668 by using the rocket equation $f_p = e(-DV/g \cdot ISP)$, where g is 0.0098 and ISP is 315. The propellant weight, P, now equals the spacecraft weight$\cdot(1-f_p)$ and turns out to be 7,570 lb. The tankage and engine weight is then 14% of this value, or 1,060 lb. Finally, the bus weight is the spacecraft weight minus propellant and tanks, or 14,170 lb.

Figure A-14 shows heliocentric J requirements for Saturn flyby and parabolic capture missions using the variable thrust mode. These results were obtained from numerical integration by using the JPL low-thrust trajectory optimization computer program (Richardson 1963). Since the variable-thrust mode has the lowest possible J requirements, this curve represents the lower limit for J_H. The next figure, A-15, shows the total J requirements, that is $J_E + J_H + J_C$ for constant-thrust flights to Saturn. The constant-thrust, constant ISP mode of operation is more realistic than the variable-thrust mode for electric thrusters. The optimum choice of ISP was made for each mission; the optimum ISP range is indicated on the curve. The calculation of constant thrust, J, was performed by employing an analytical approximation method termed "Characteristic Length Correlation," which allows the constant-thrust data to be computed directly from the variable-thrust data (Zola 1964; Friedlander 1965). It should be noted that by considering each mission to each of the 4 planets, the desirable range of ISP is from 5,000 to 15,000 sec.

Figure A-16 converts the total J requirements of Figure A-15 into communications and experiments payloads, by using the payload vs. J curve from Figure A-3 for the vehicle with $\alpha = 40$, $P_e = 250$.

Figures A-17 through A-27 for Uranus correspond to A-7 through A-16 for Saturn. Similarly, A-28 through A-38 apply to Neptune and A-39 to A-47 apply to Pluto. There are no gravity assist curves for flights to Pluto; although the gravity assist mode is certainly quite attractive for flyby flights to Pluto, our currently operational gravity assist trajectory codes are limited to two dimensions and therefore are not sufficiently accurate for calculations of flights to Pluto.

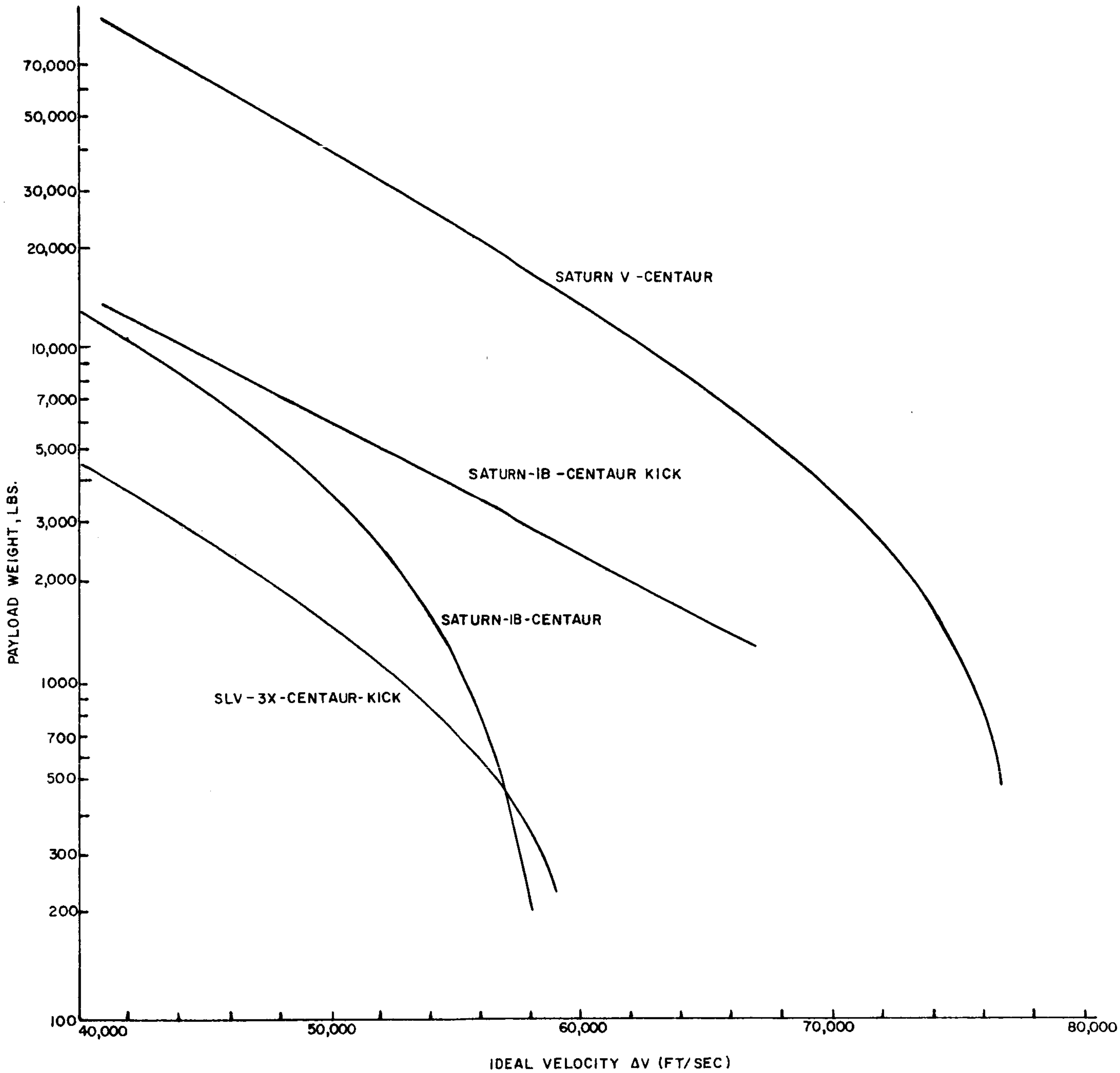

FIGURE A-1. BALLISTIC LAUNCH VEHICLE PERFORMANCE CURVE

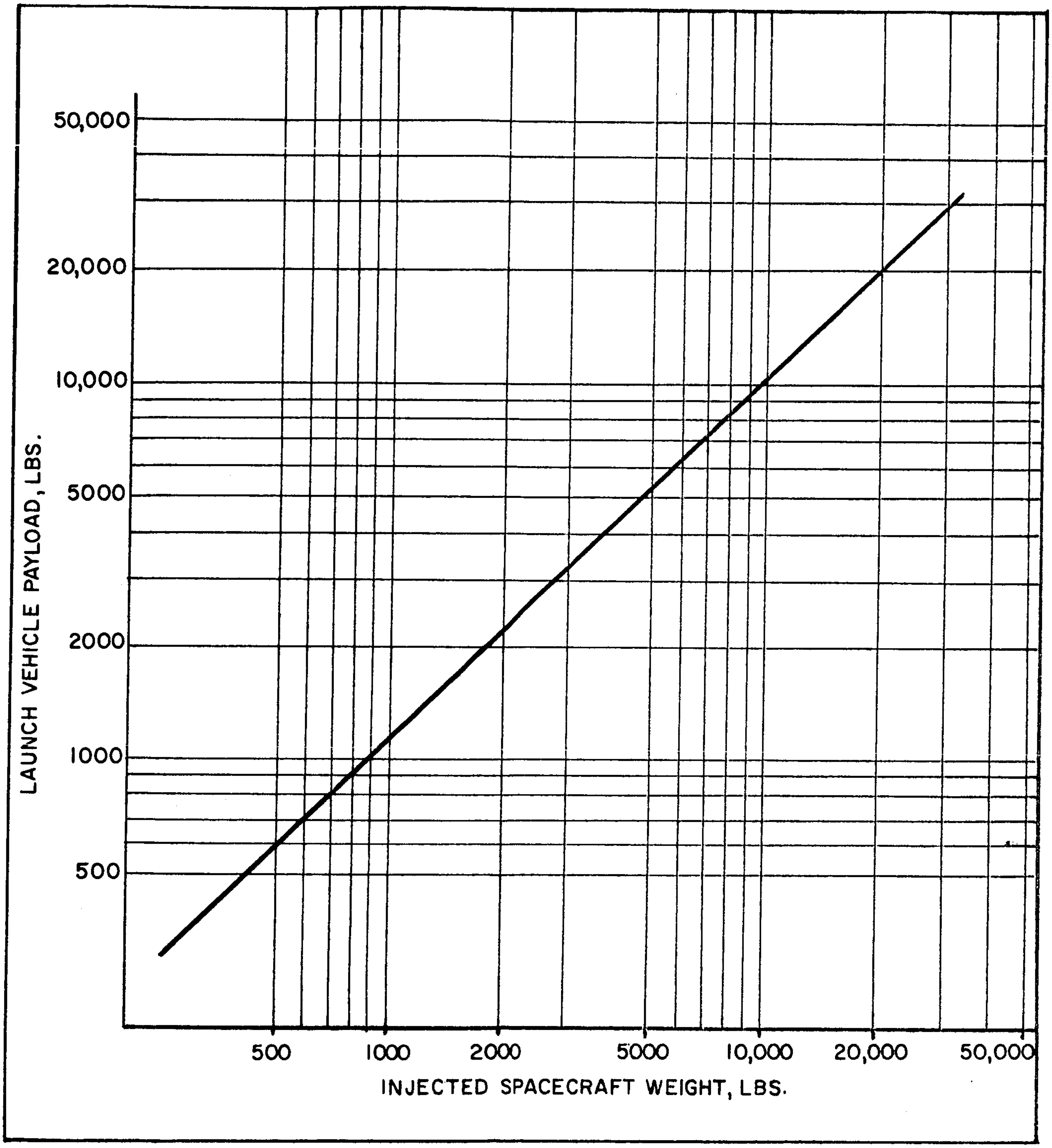

FIGURE A-2. SPACECRAFT WEIGHT/PAYLOAD WEIGHT CONVERSION CURVE

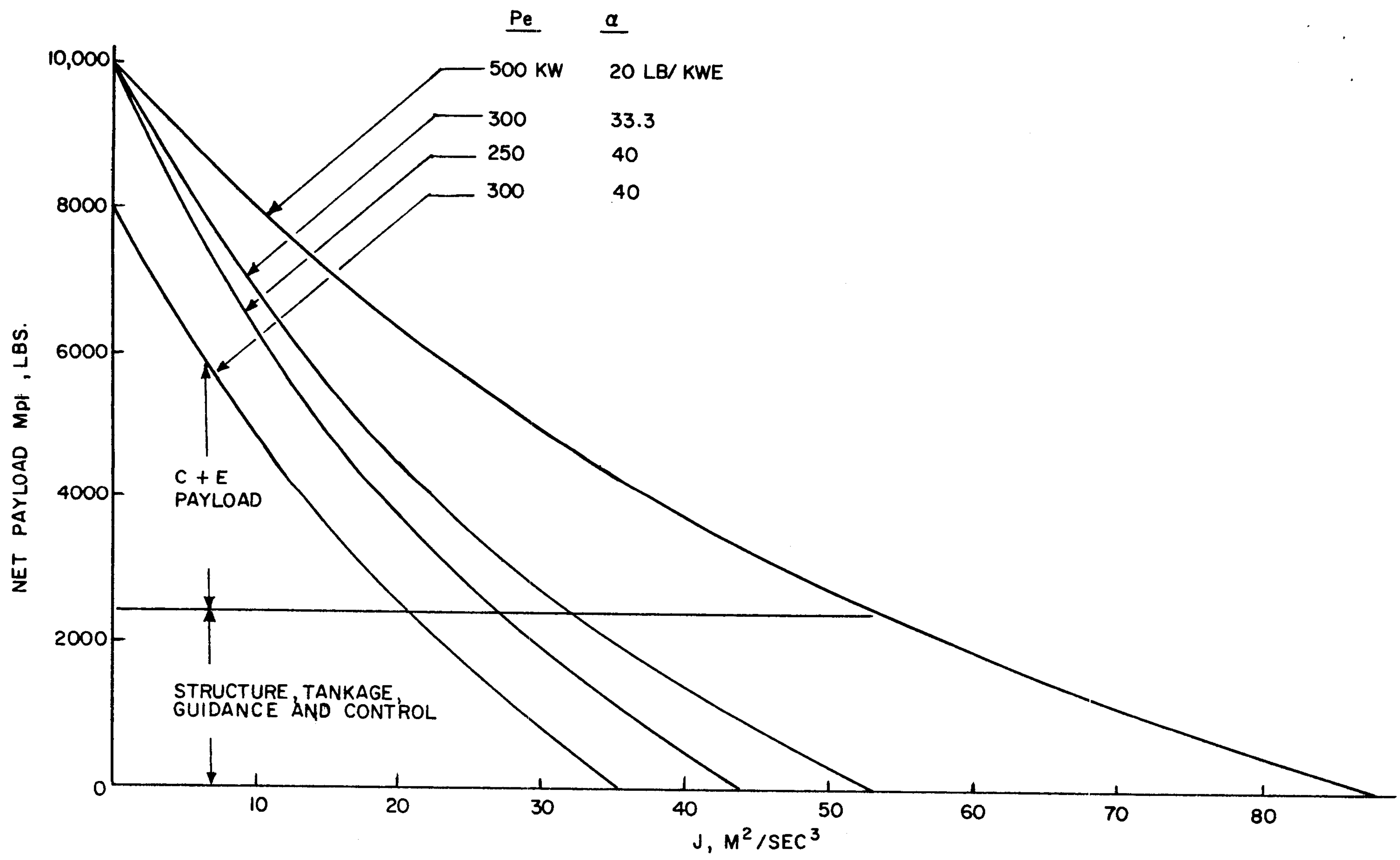

FIGURE A-3. PAYLOAD VS. J CURVES FOR NUCLEAR ELECTRIC STAGE DESIGNS

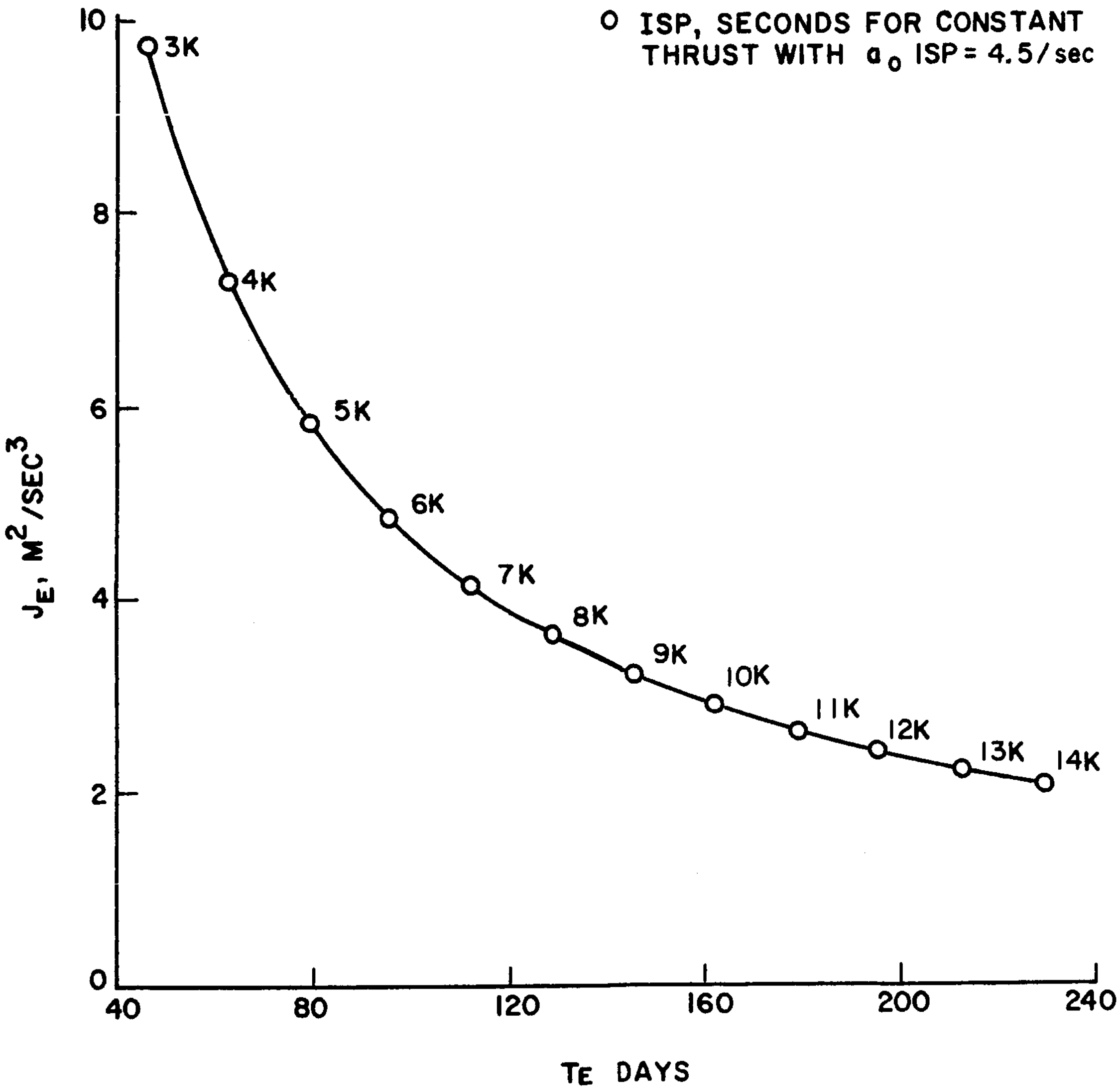

FIGURE A-4. J REQUIREMENTS FOR EARTH ESCAPE FROM 1000 n.mi ORBIT

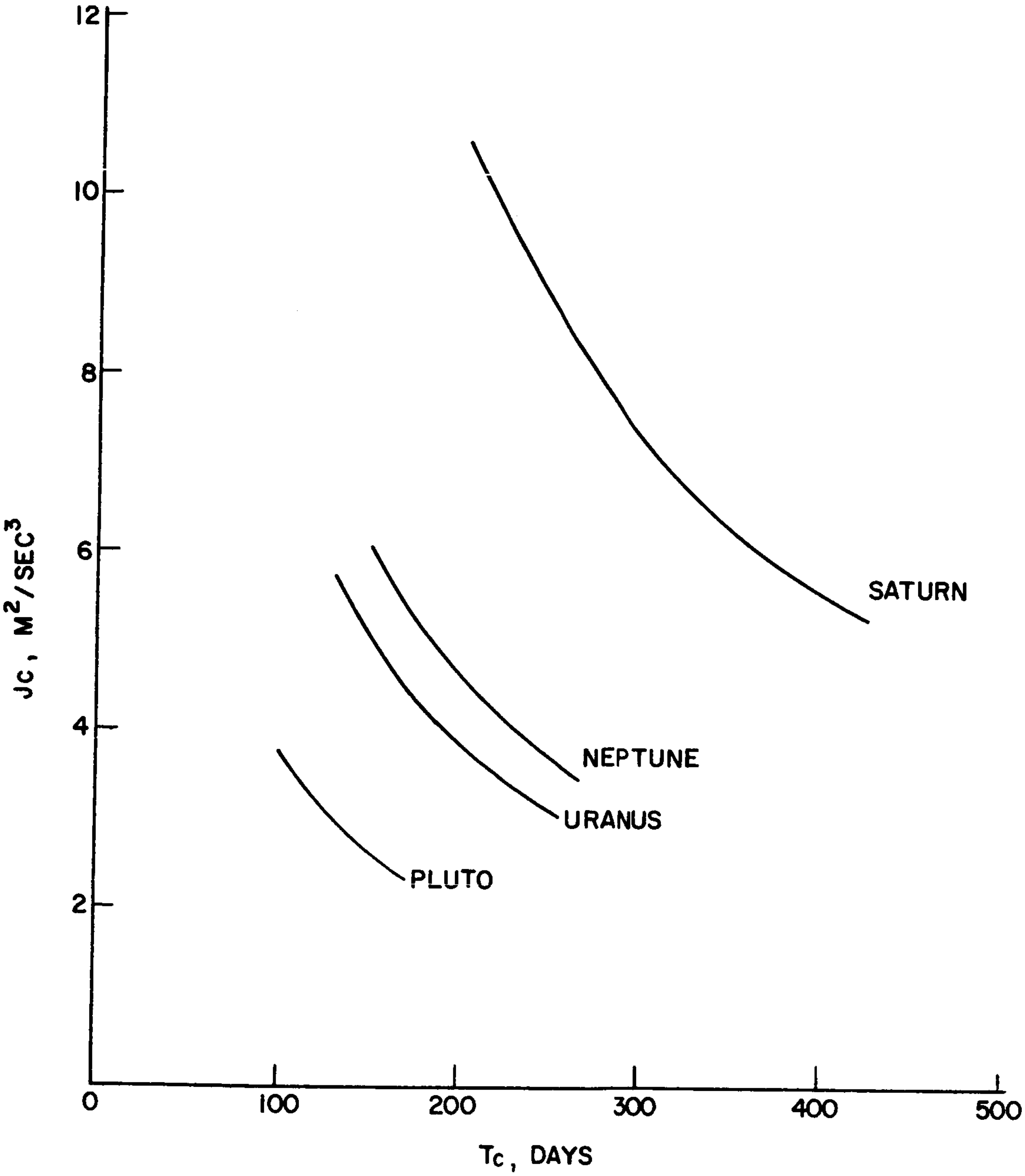

FIGURE A-5. J REQUIREMENTS FOR PLANET CAPTURE TO CIRCULAR ORBIT AT 3 PLANET RADII

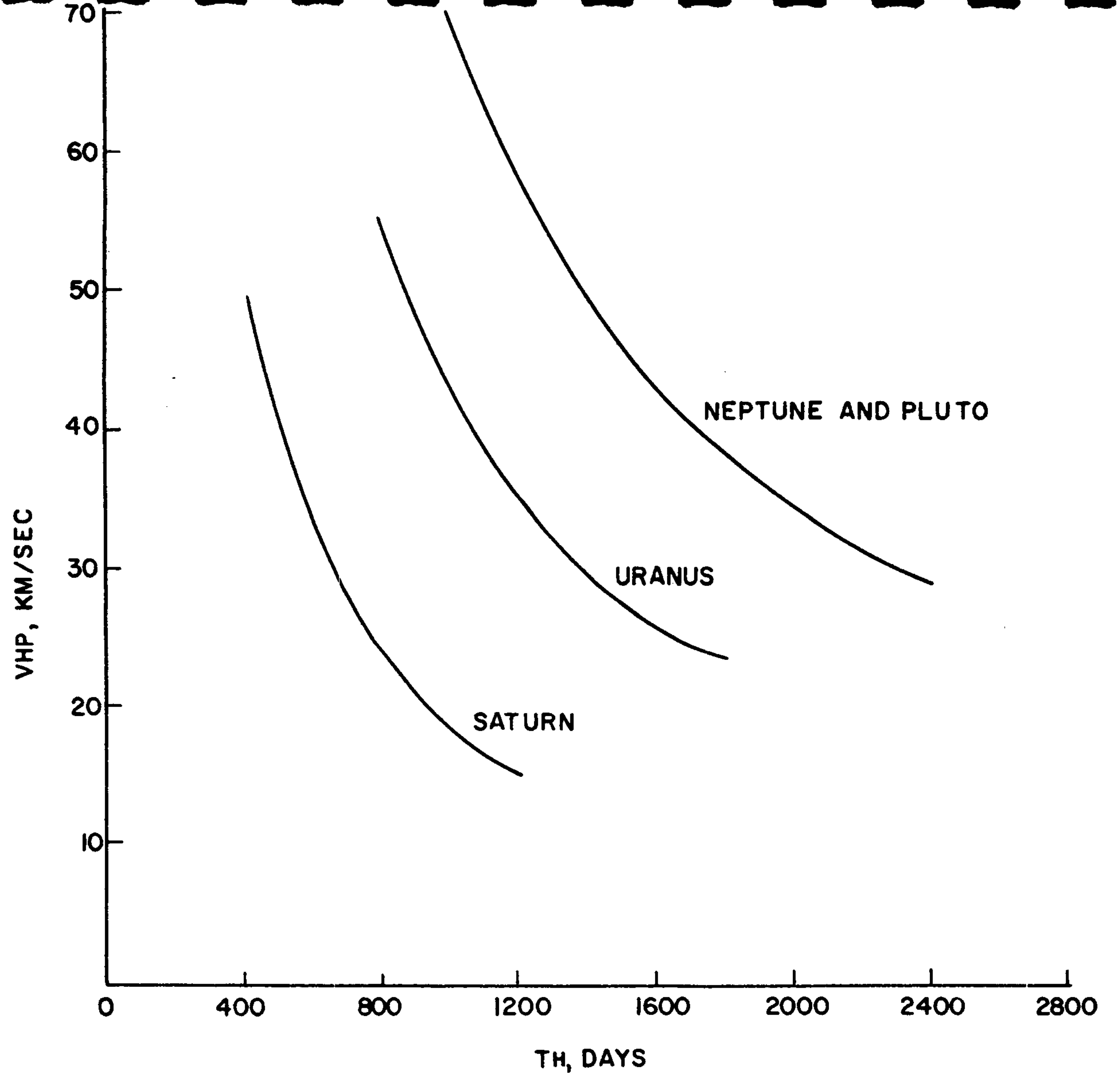

FIGURE A-6. APPROACH VELOCITIES FOR MINIMUM J FLY-BY TRAJECTORIES - VARIABLE THRUST MODE

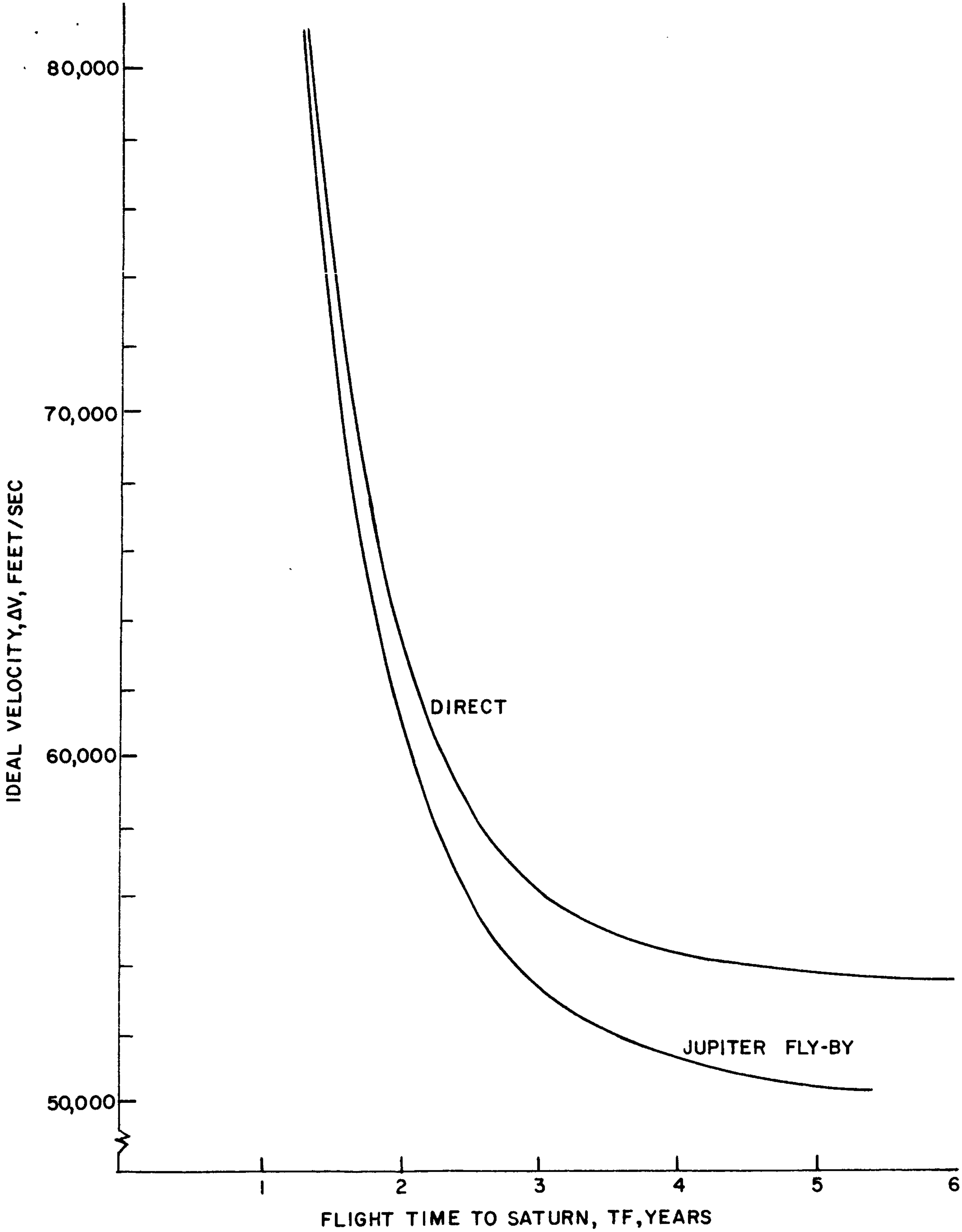

FIGURE A-7. IDEAL VELOCITY FOR FLIGHTS TO SATURN

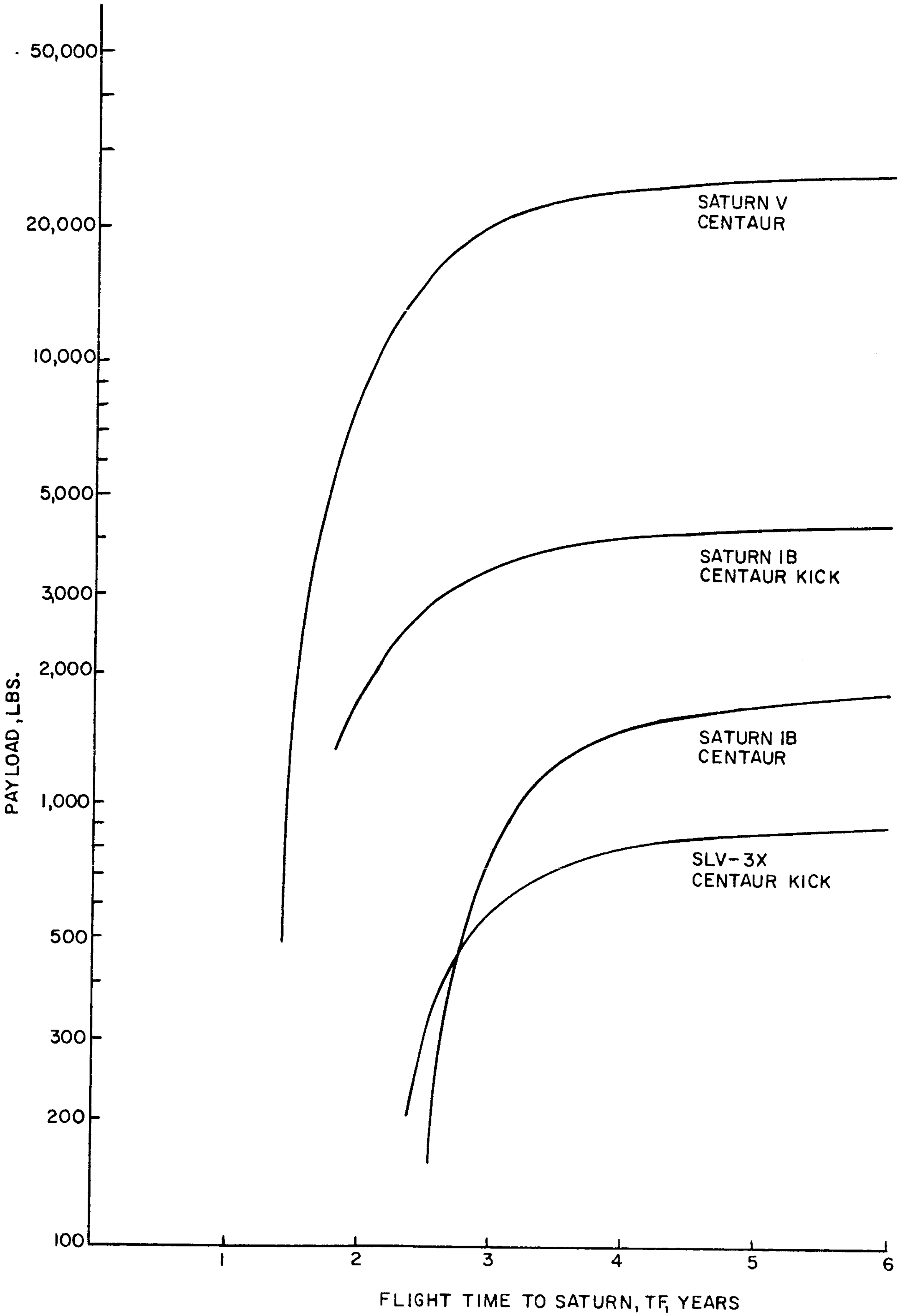

FIGURE A-8. PAYLOADS FOR DIRECT SATURN FLIGHTS

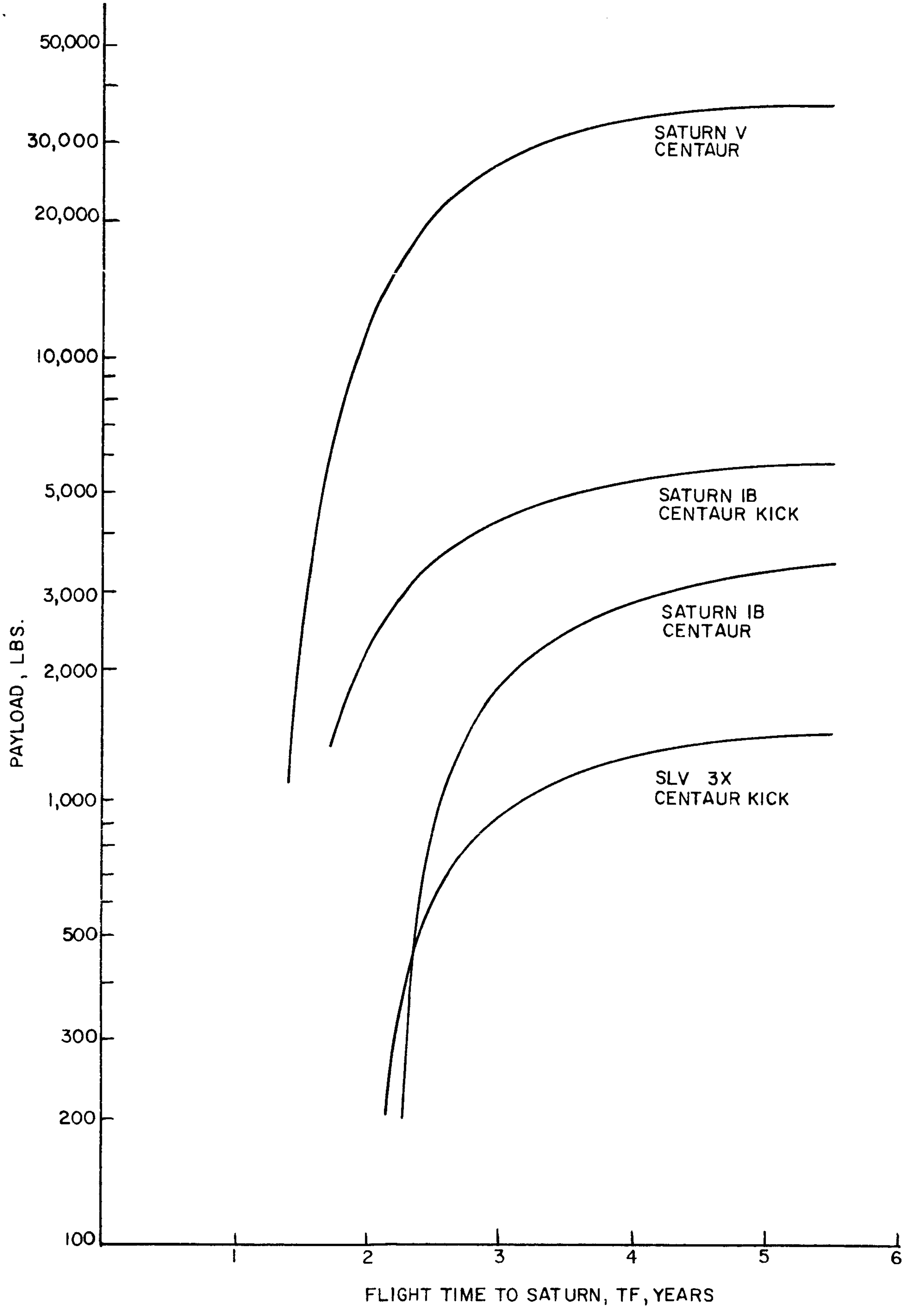

FIGURE A-9. PAYLOADS FOR JUPITER ASSIST SATURN FLIGHTS

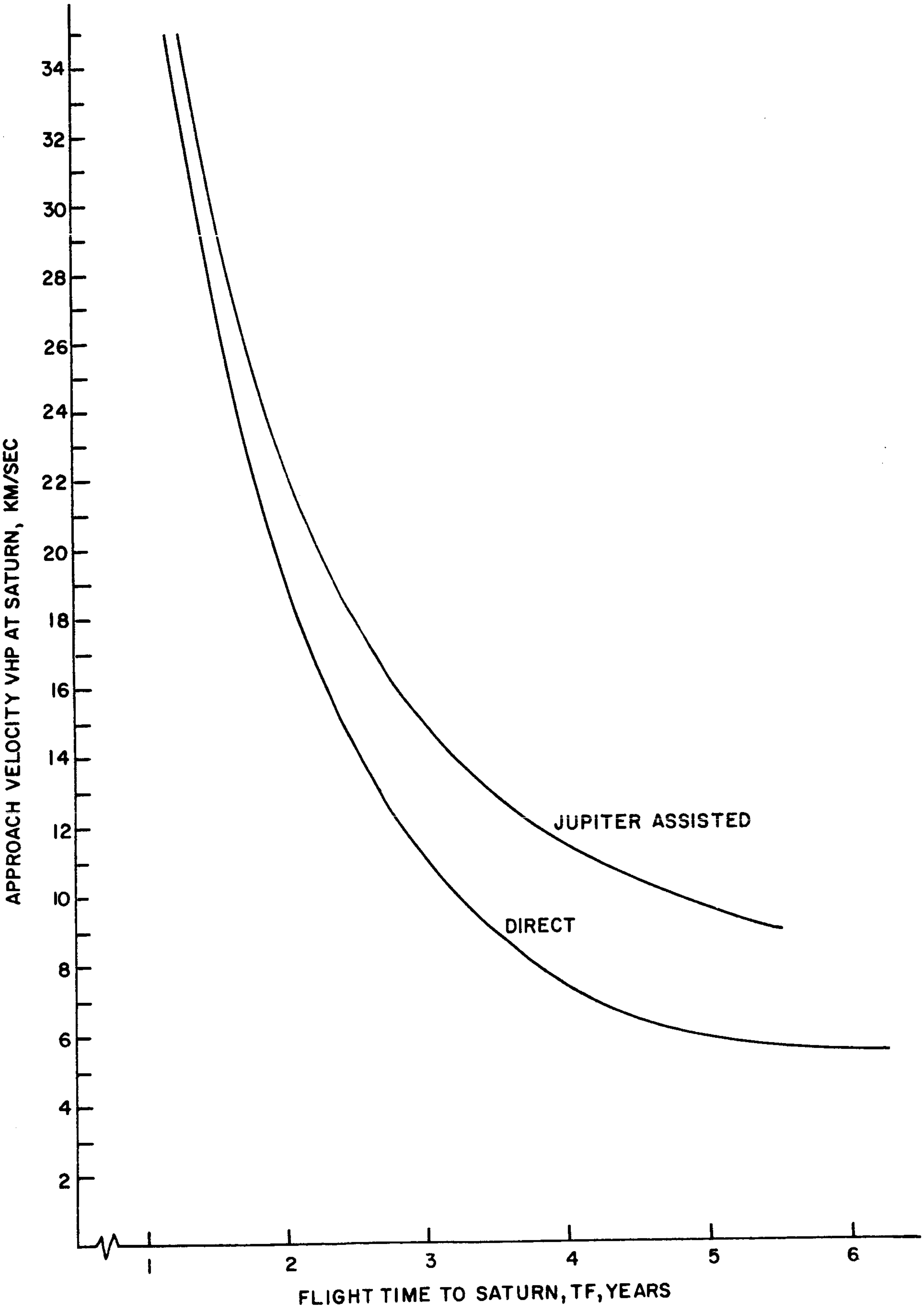

FIGURE A-10. APPROACH VELOCITY FOR FLIGHTS TO SATURN

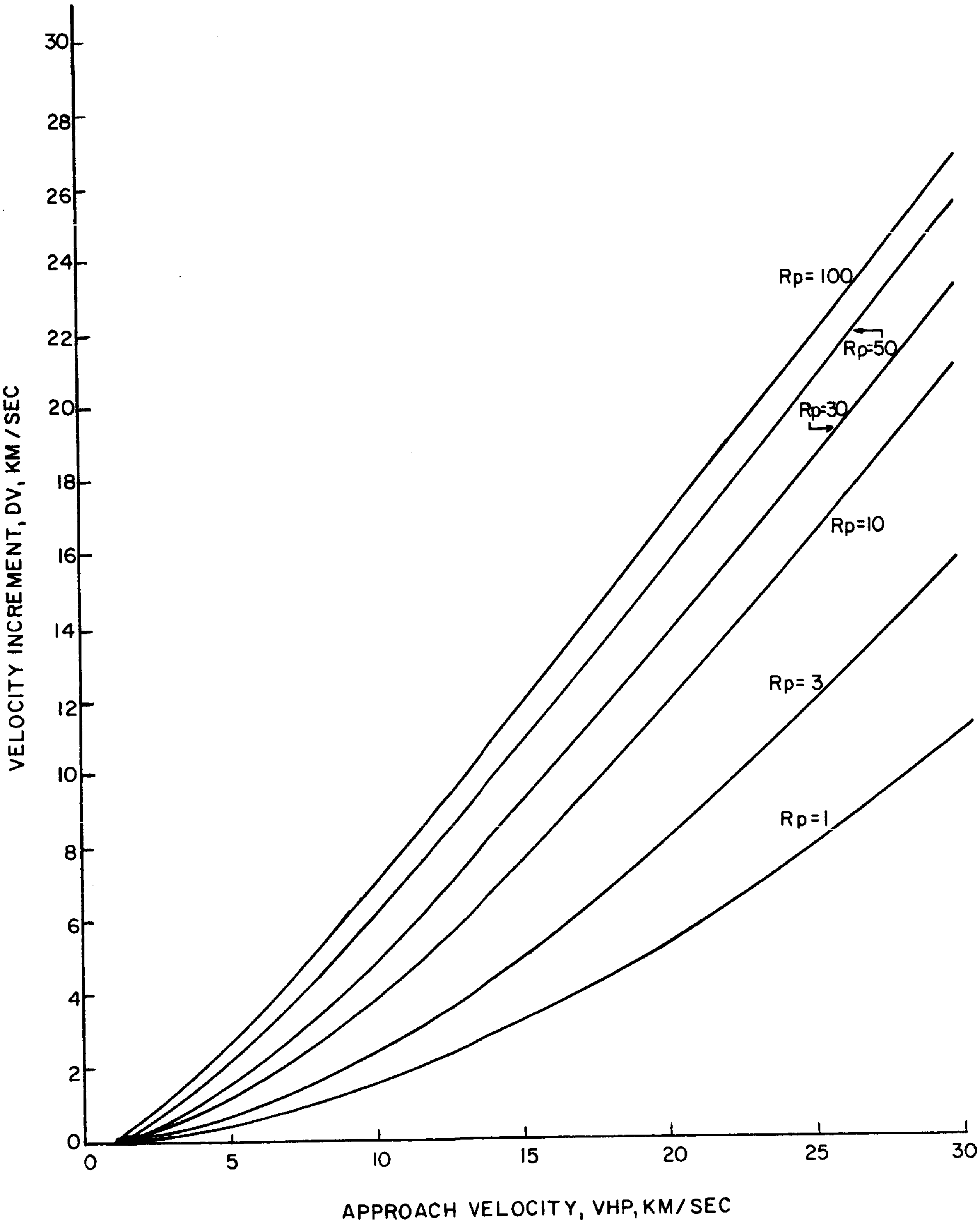

FIGURE A-11. VELOCITY INCREMENT FOR PARABOLIC RENDEZVOUS AT SATURN.

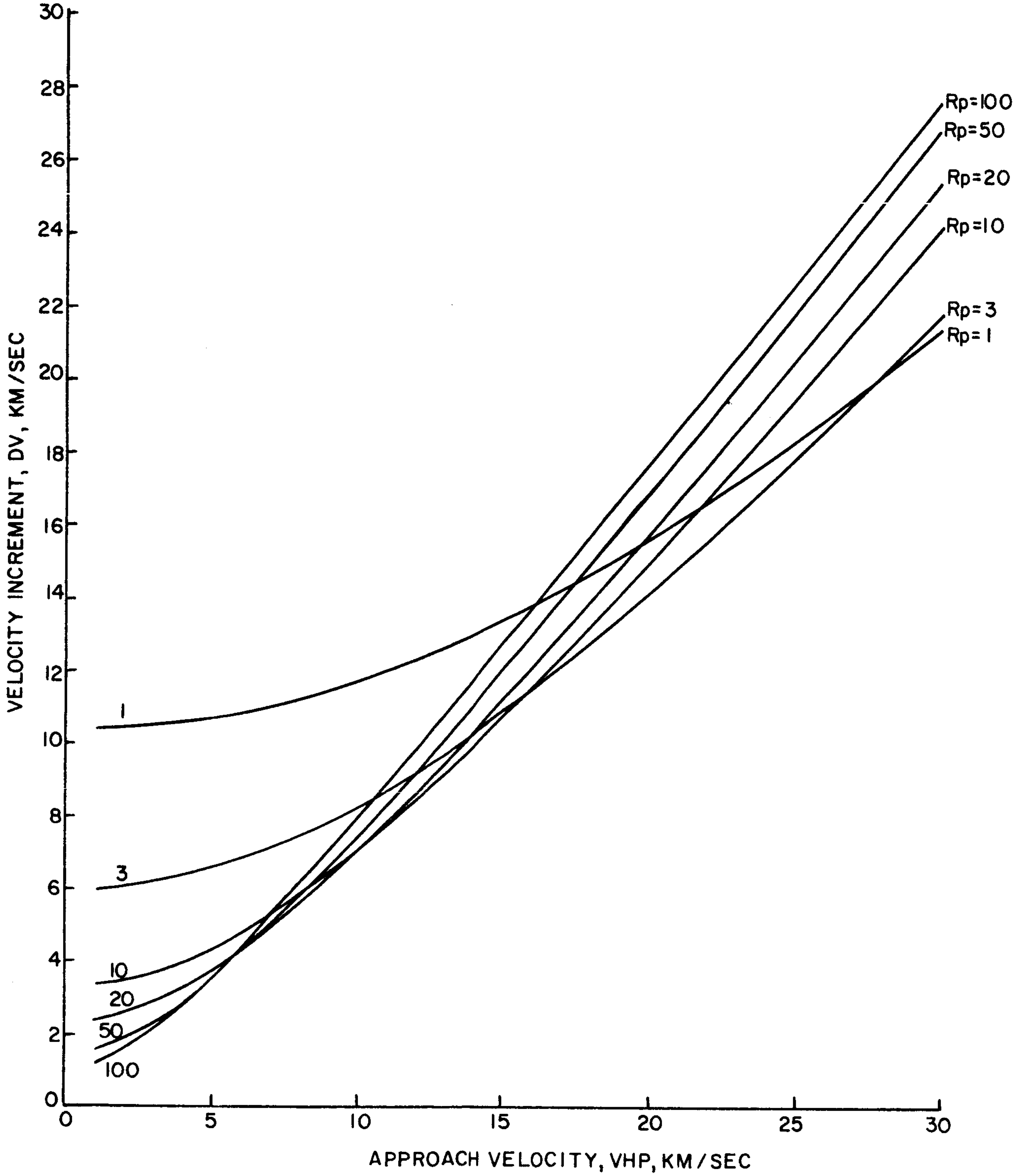

FIGURE A-12. VELOCITY INCREMENT FOR CIRCULAR RENDEZVOUS AT SATURN

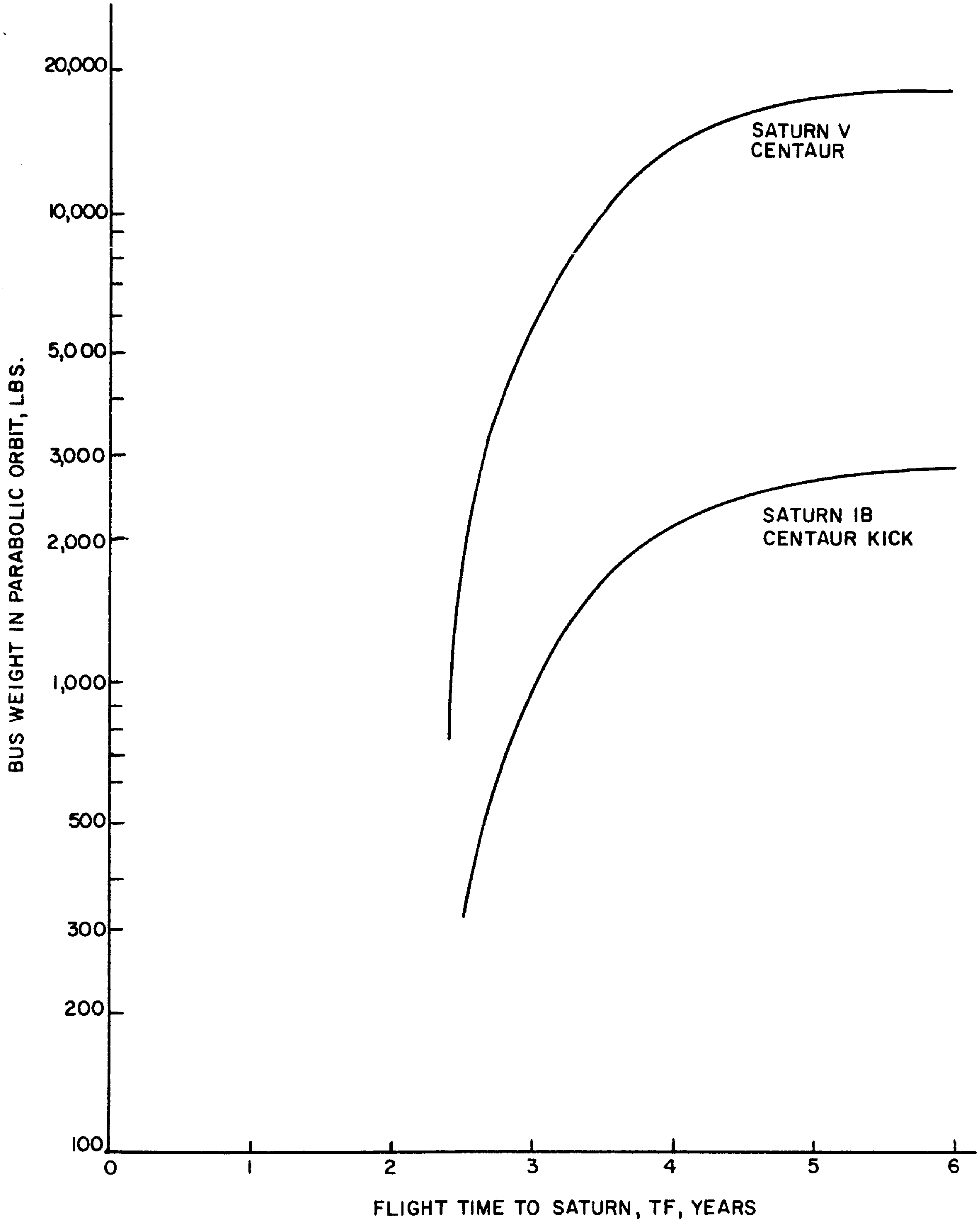

FIGURE A-13. PAYLOADS IN PARABOLIC ORBIT AROUND SATURN. ASSUMES: 3 PLANET RADII MISS. ENGINE ISP = 315 SEC.

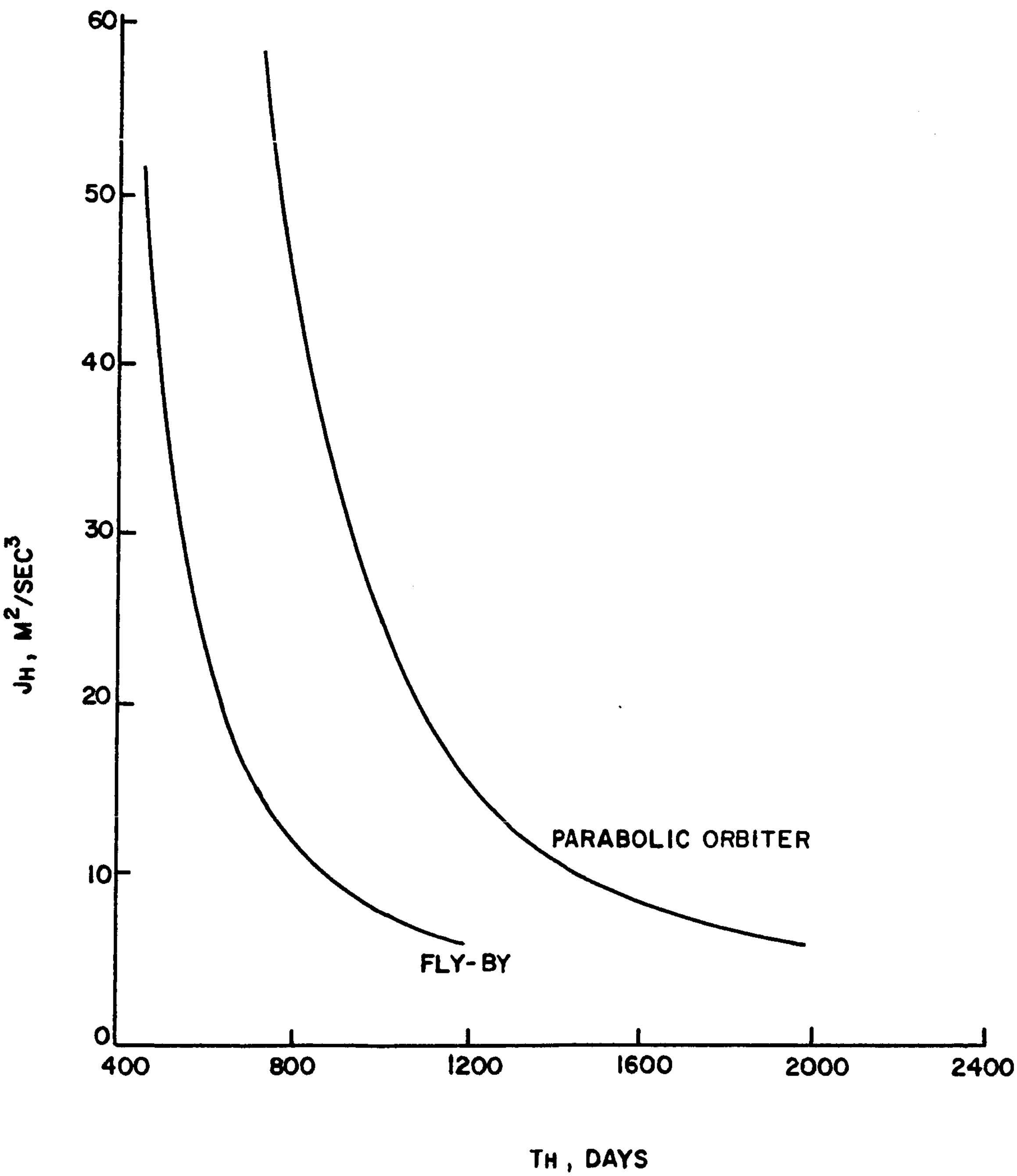

FIGURE A-14. HELIOCENTRIC J REQUIREMENTS FOR SATURN MISSIONS, VARIABLE THRUST MODE

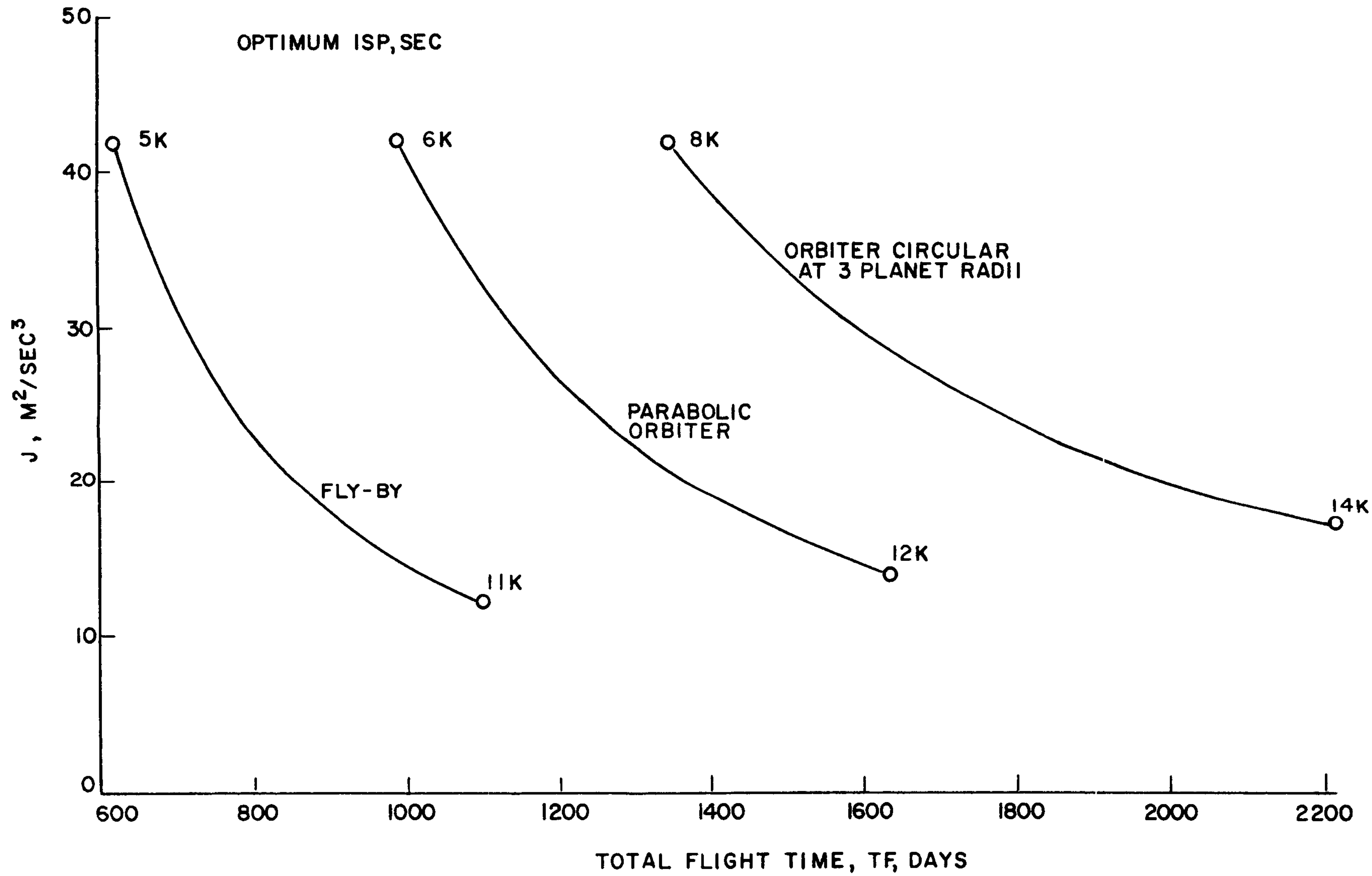

FIGURE A-15. J REQUIREMENTS FOR MISSIONS TO SATURN, CONSTANT THRUST MODE a_0 ISP = 4.5 m/sec.

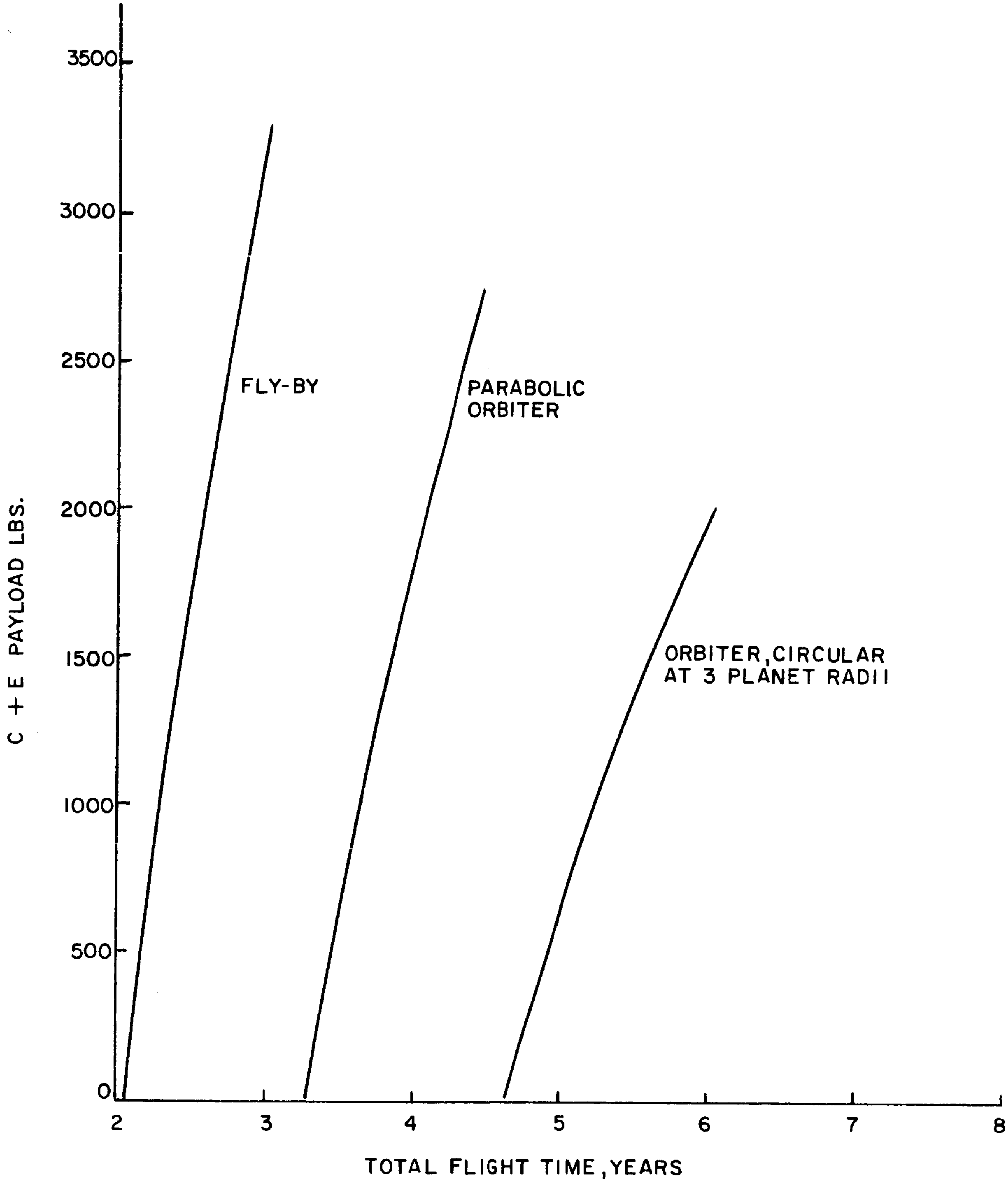

FIGURE A-16. PAYLOAD CAPABILITY FOR LOW-THRUST MISSIONS TO SATURN. SATURN-IB-THRUSTED STAGE (α = 40 LBS/KWE, P = 250 KWC)

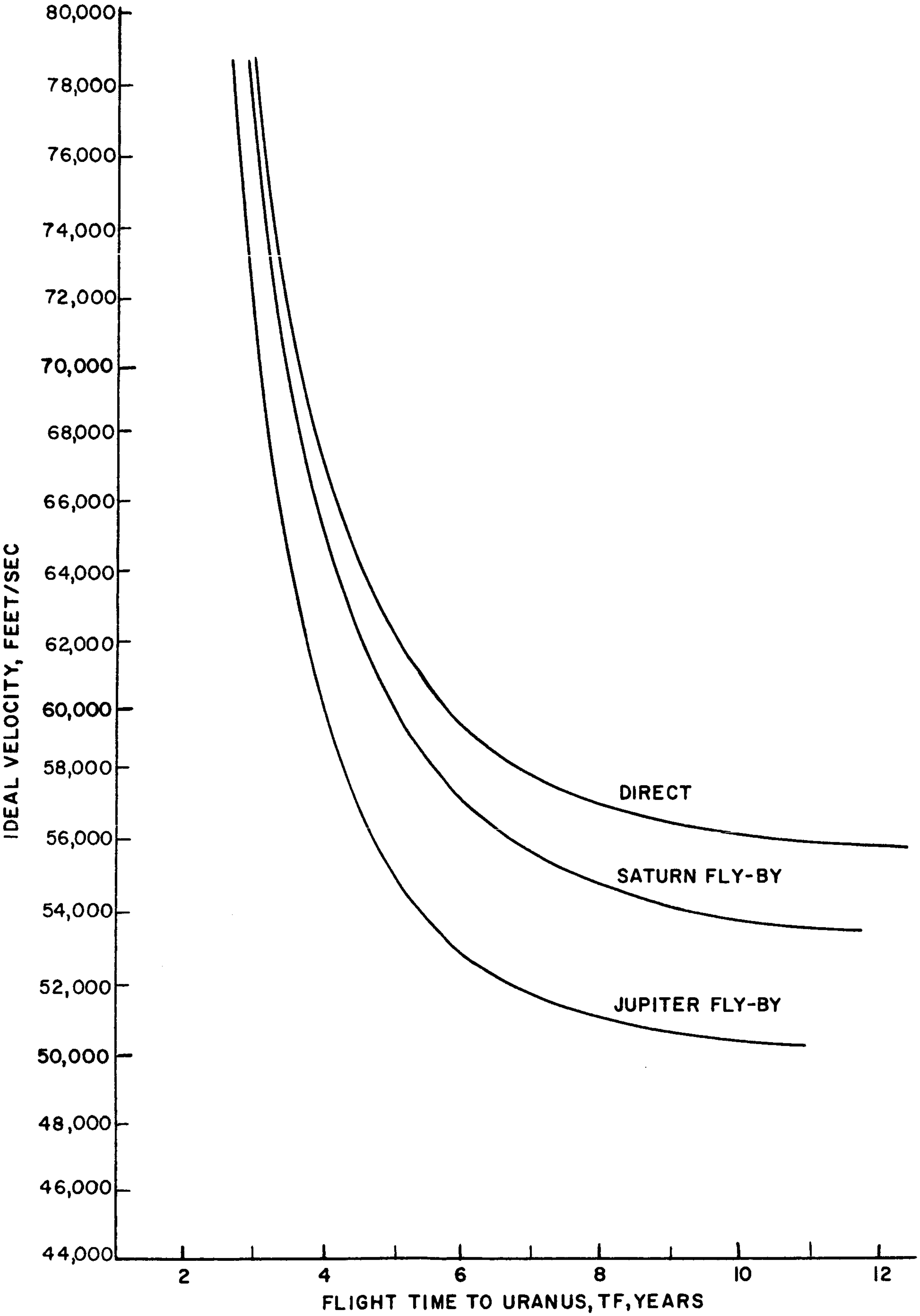

FIGURE. A-17. IDEAL VELOCITY FOR FLIGHT TO URANUS.

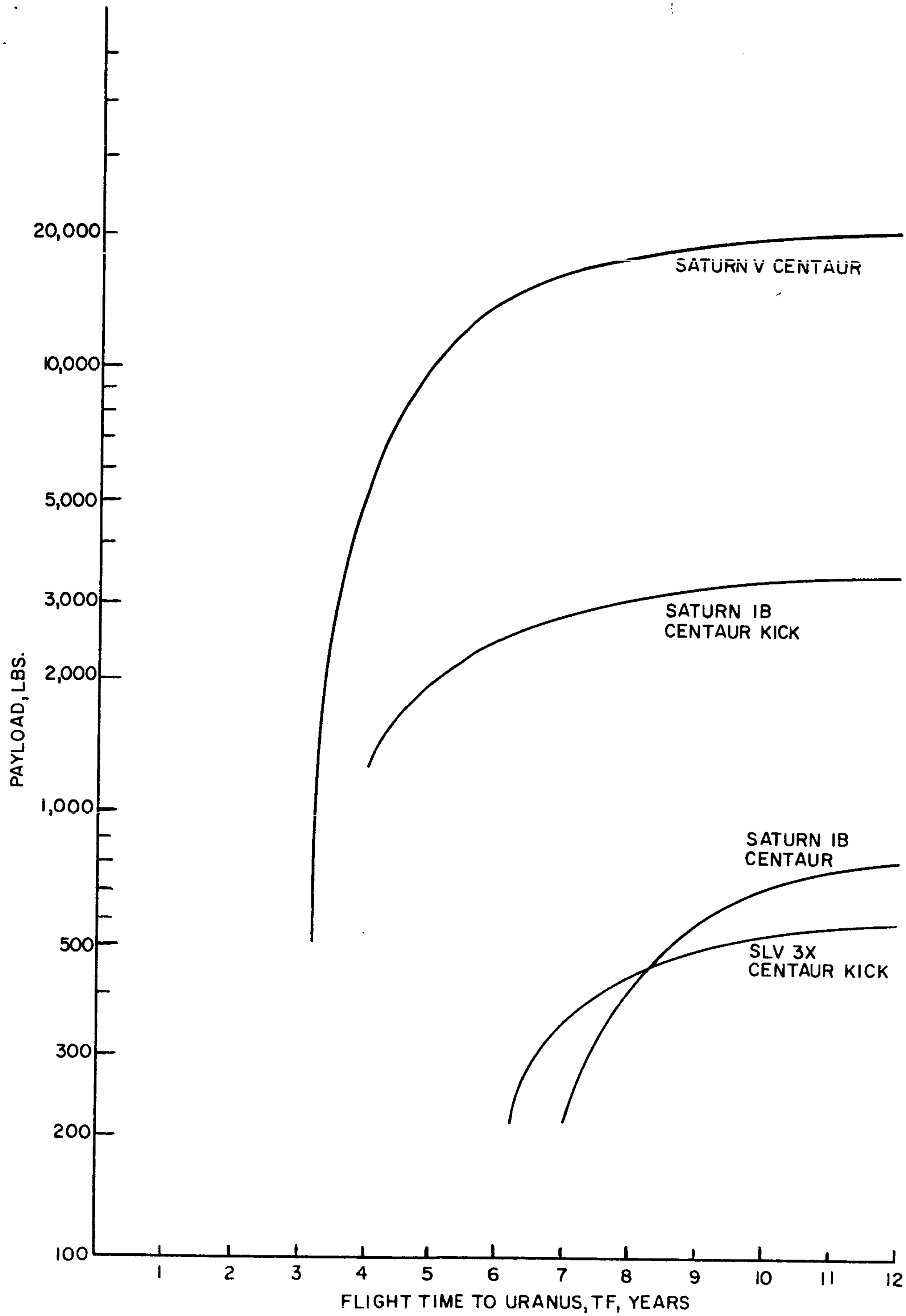

FIGURE A-18. PAYLOADS FOR DIRECT URANUS FLIGHTS

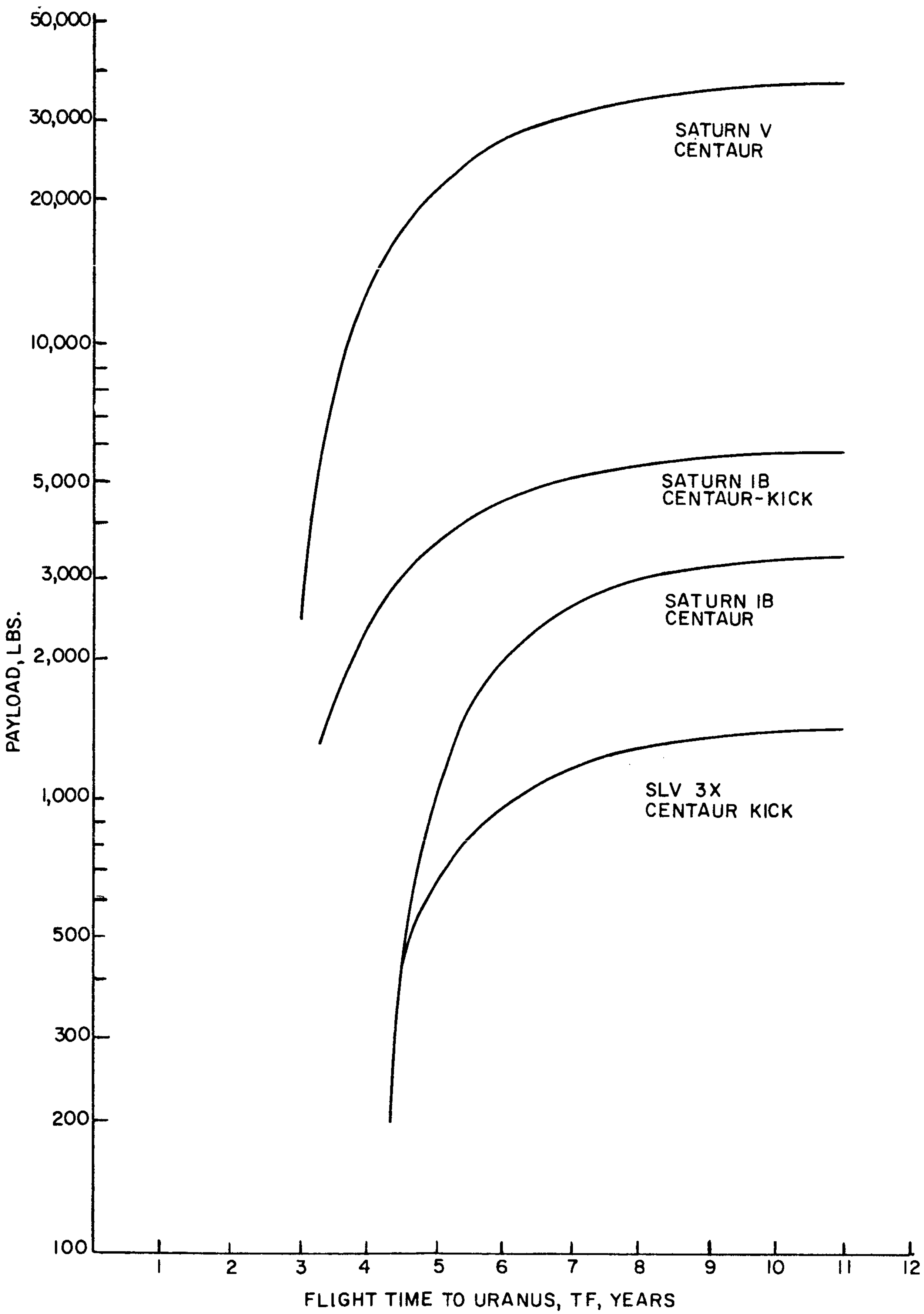

FIGURE A-19. PAYLOADS FOR JUPITER ASSIST URANUS FLIGHTS

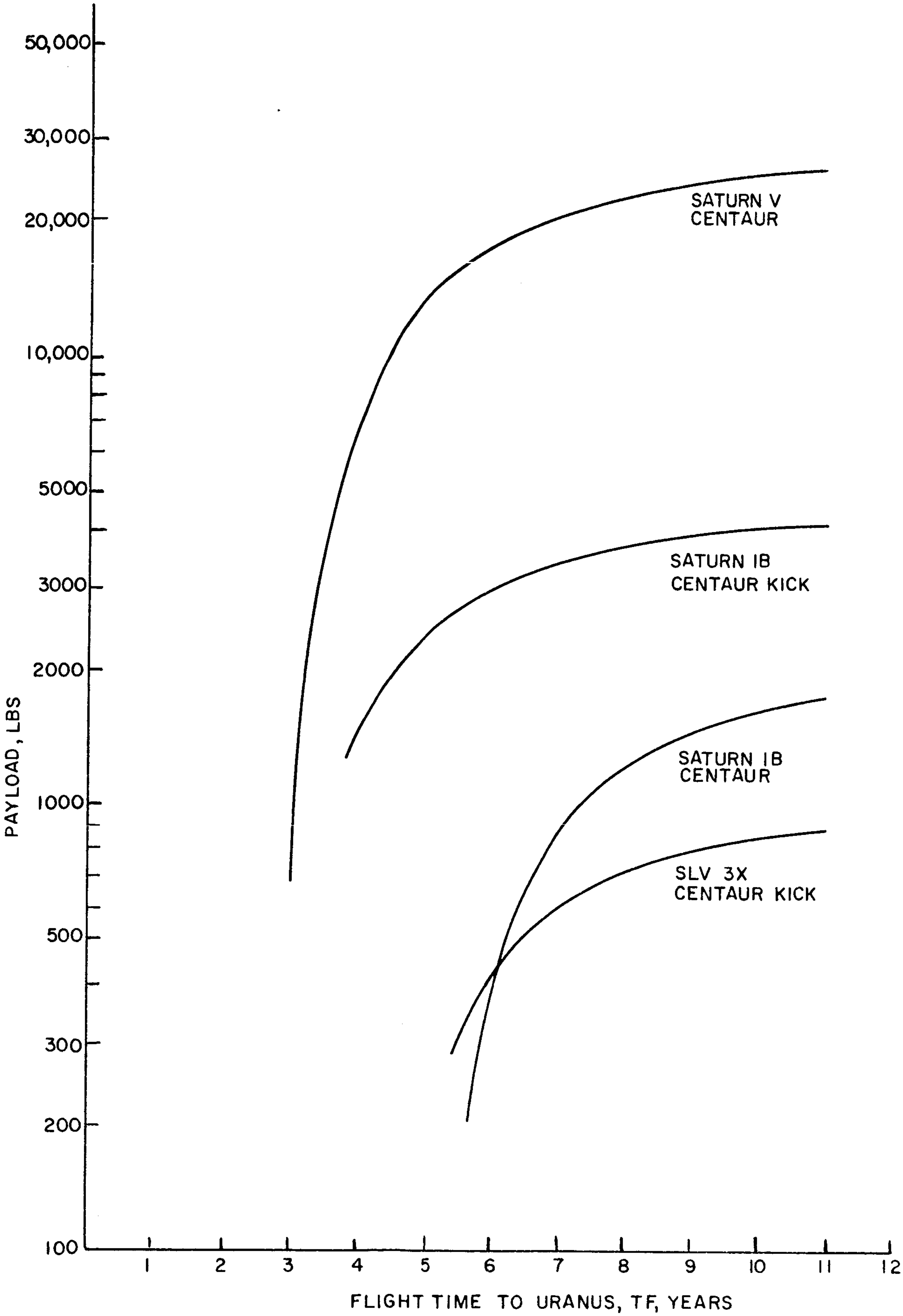

FIGURE A-20. PAYLOADS FOR SATURN ASSIST URANUS FLIGHTS.

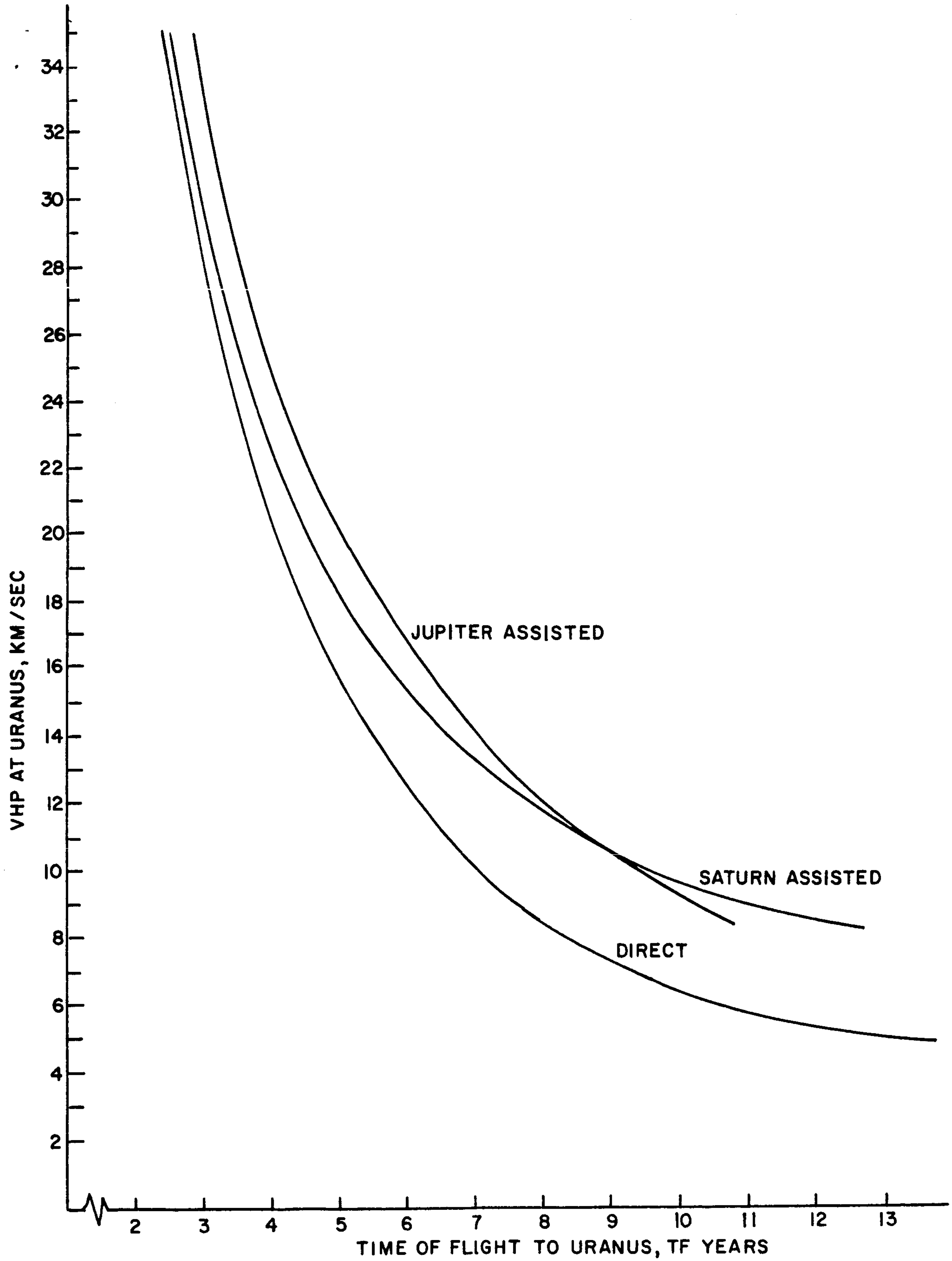

FIGURE A-21. APPROACH VELOCITY FOR FLIGHTS TO URANUS

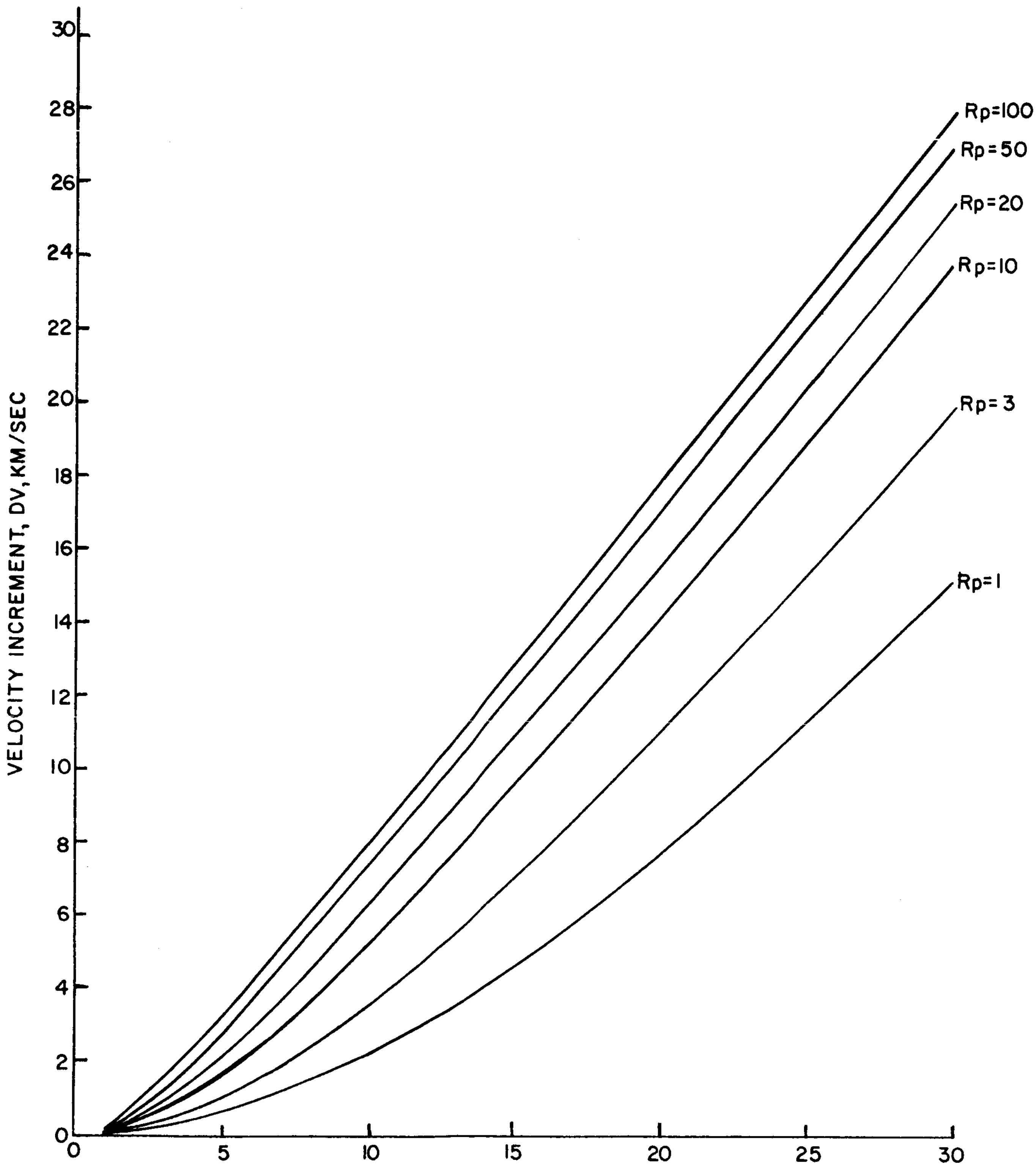

FIGURE A-22. VELOCITY INCREMENT FOR PARABOLIC RENDEZVOUS AT URANUS

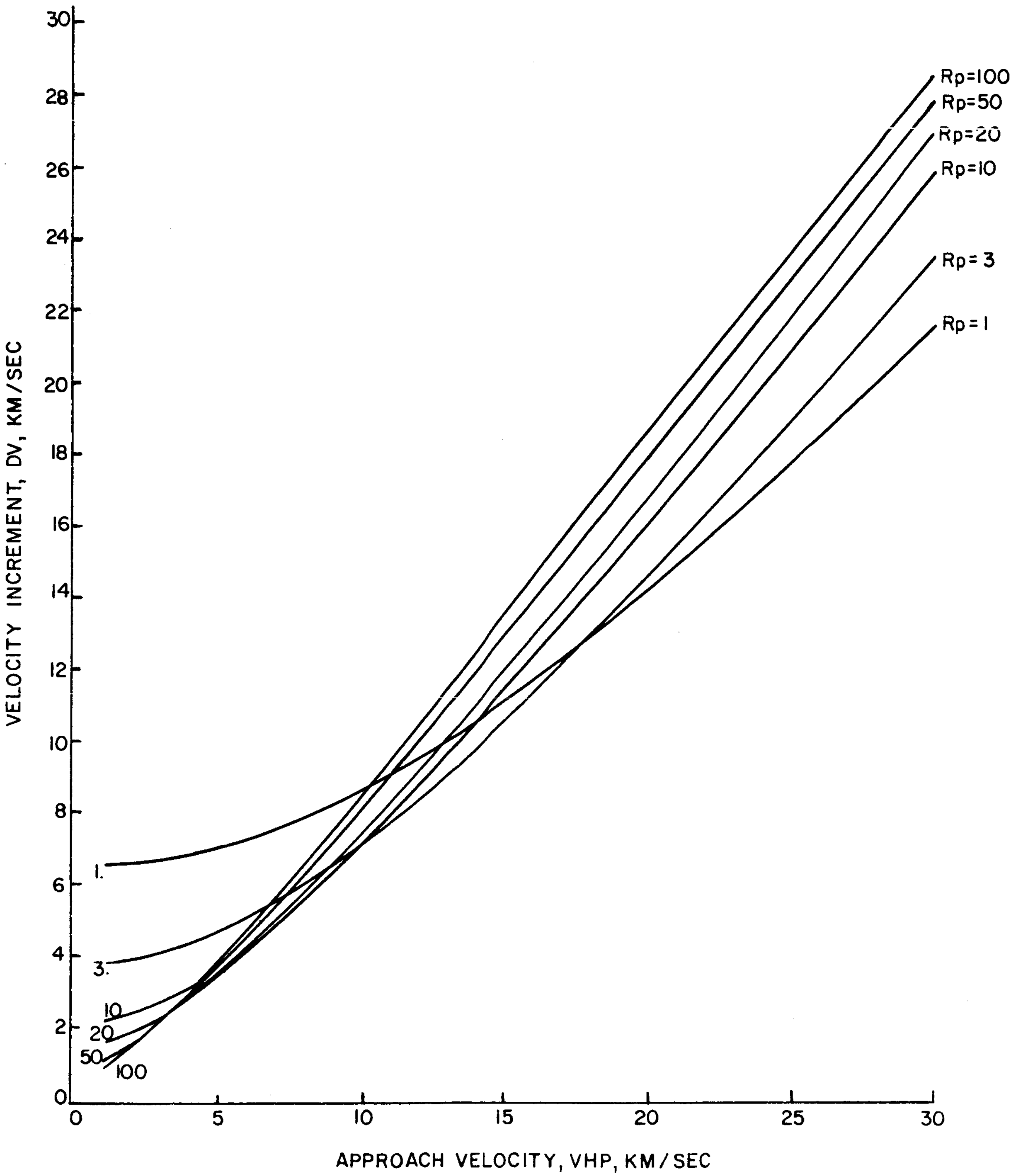

FIGURE A-23. VELOCITY INCREMENT FOR CIRCULAR RENDEZVOUS AT URANUS

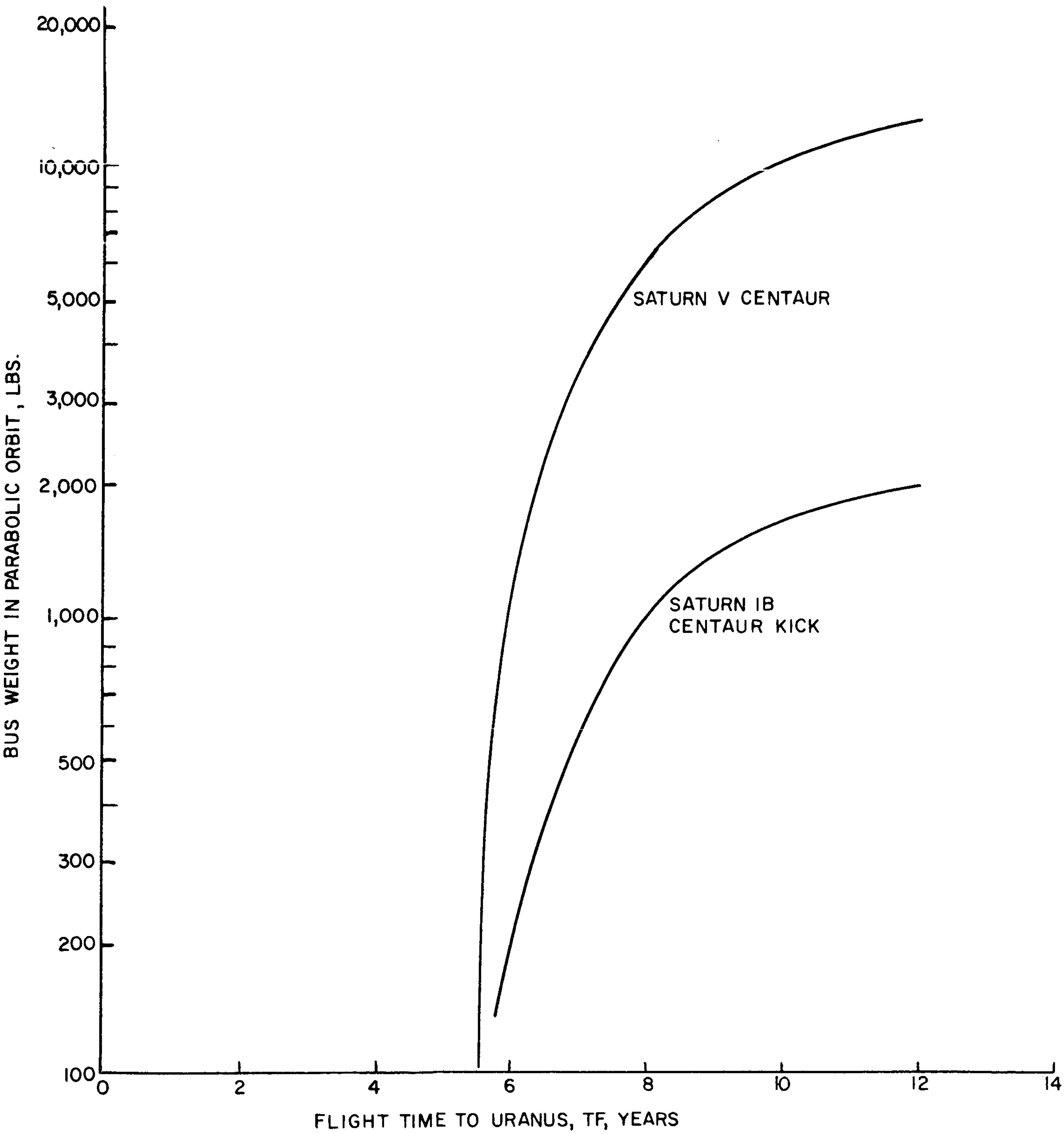

FIGURE A-24. PAYLOADS IN PARABOLIC ORBIT AROUND URANUS. ASSUMES: 3 PLANET RADII MISS. ENGINE ISP = 315 SEC.

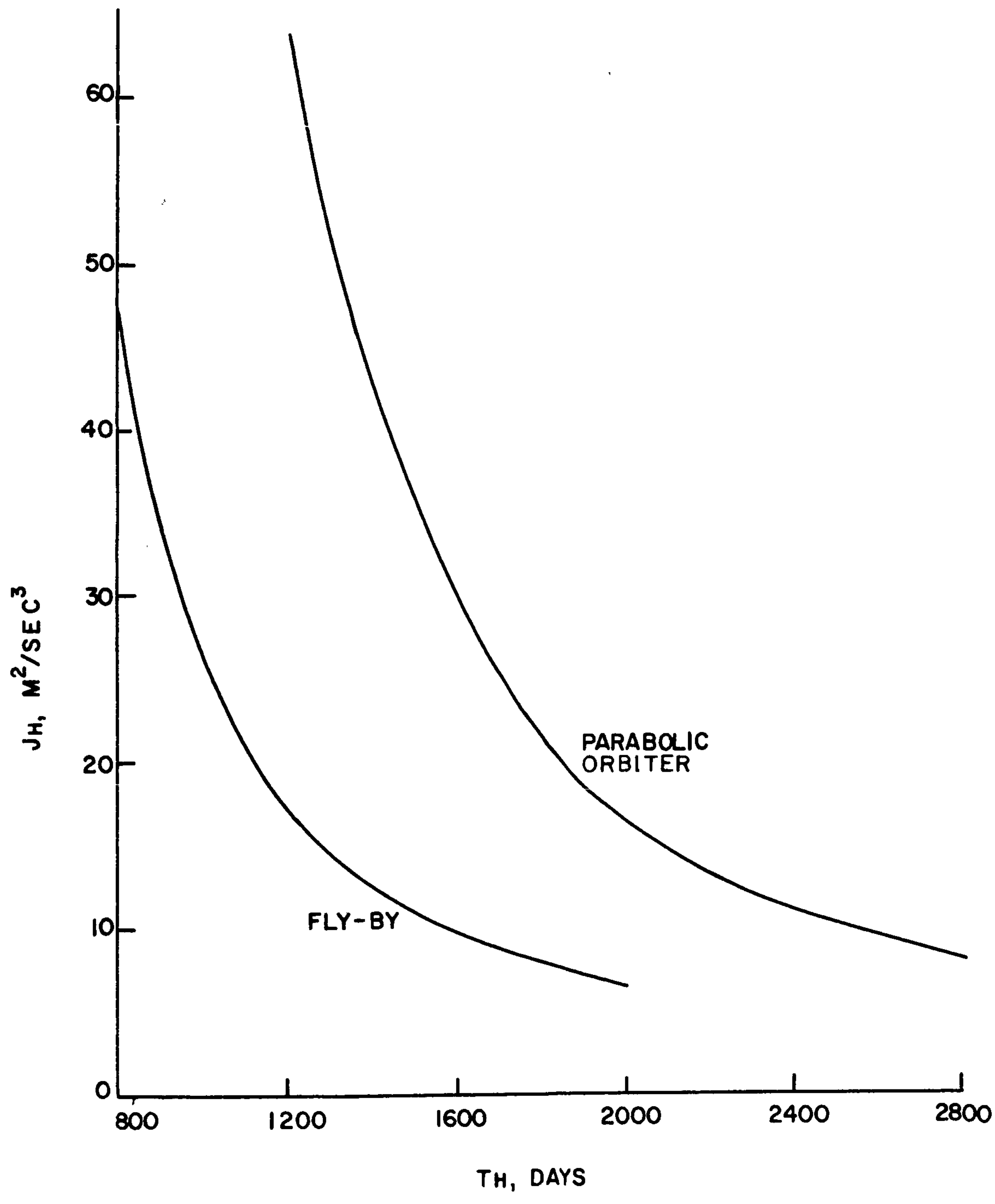

FIGURE A-25. HELIOCENTRIC J REQUIREMENTS FOR URANUS MISSIONS, VARIABLE THRUST MODE

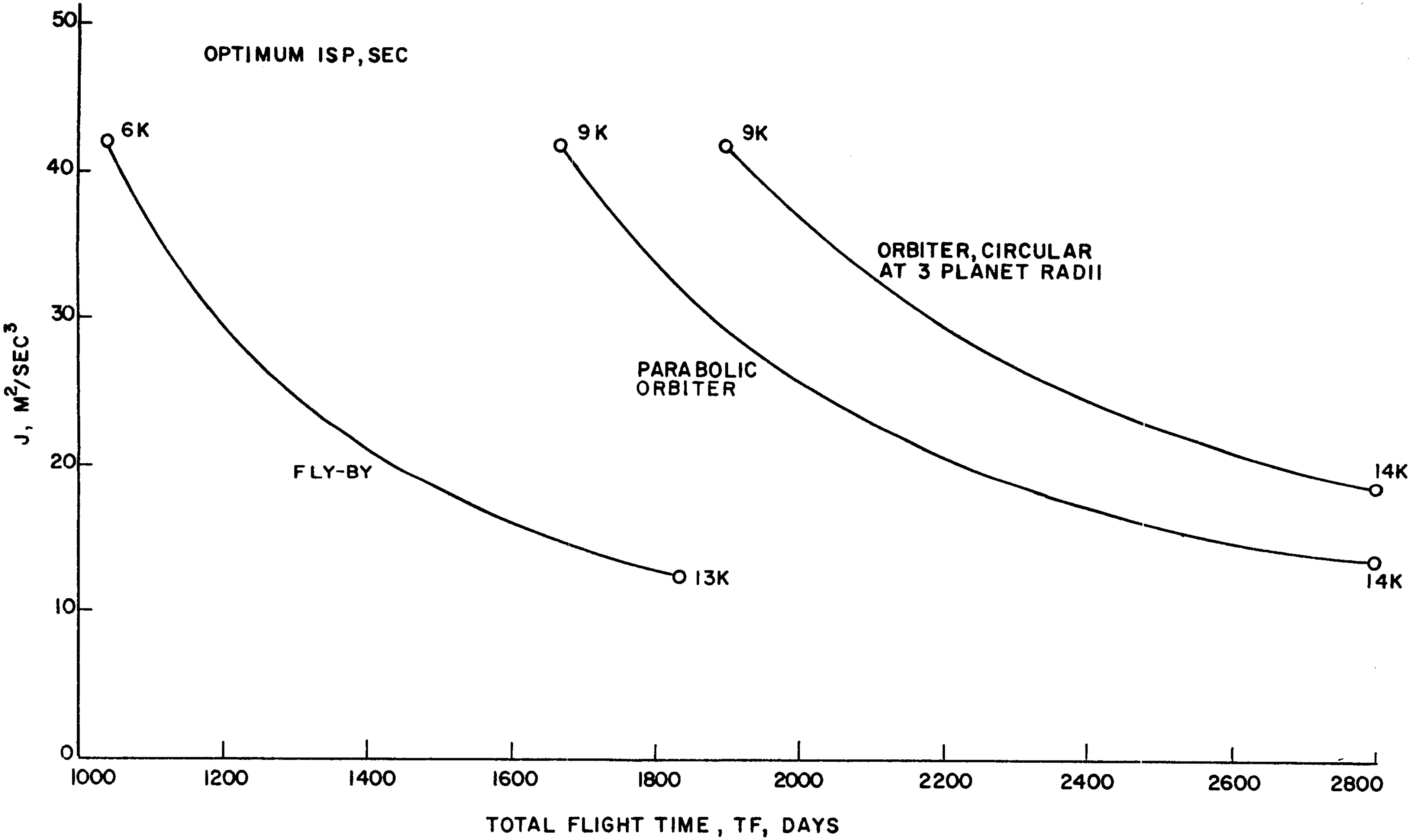

FIGURE A-26. J REQUIREMENTS FOR MISSIONS TO URANUS, CONSTANT THRUST MODE a_0 ISP = 4.5m/sec

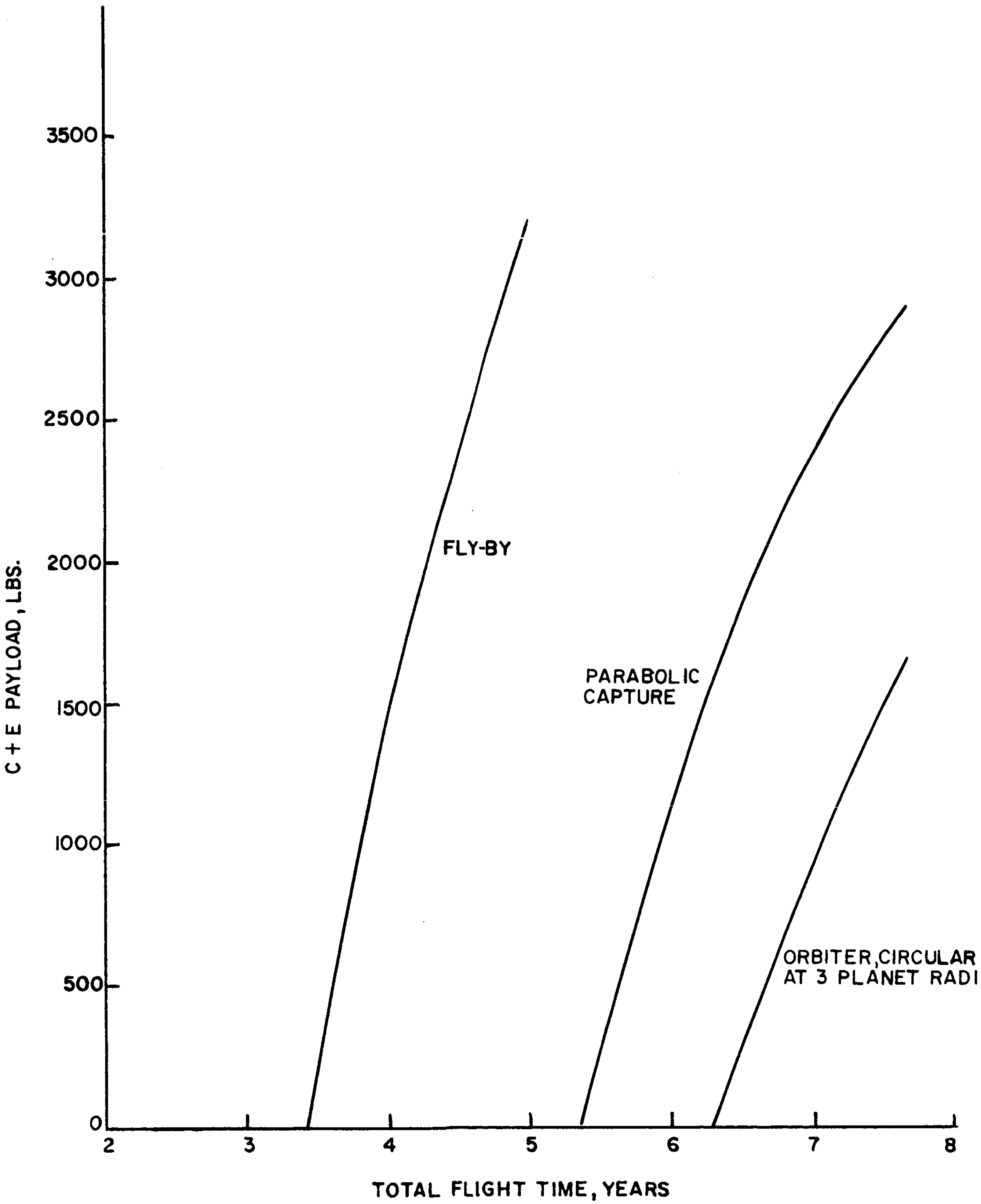

FIGURE A-27. PAYLOAD CAPABILITY FOR LOW-THRUST MISSIONS TO URANUS. SATURN-IB- THRUSTED STAGE (α = 40 LBS/KWE, P= 250 KWE)

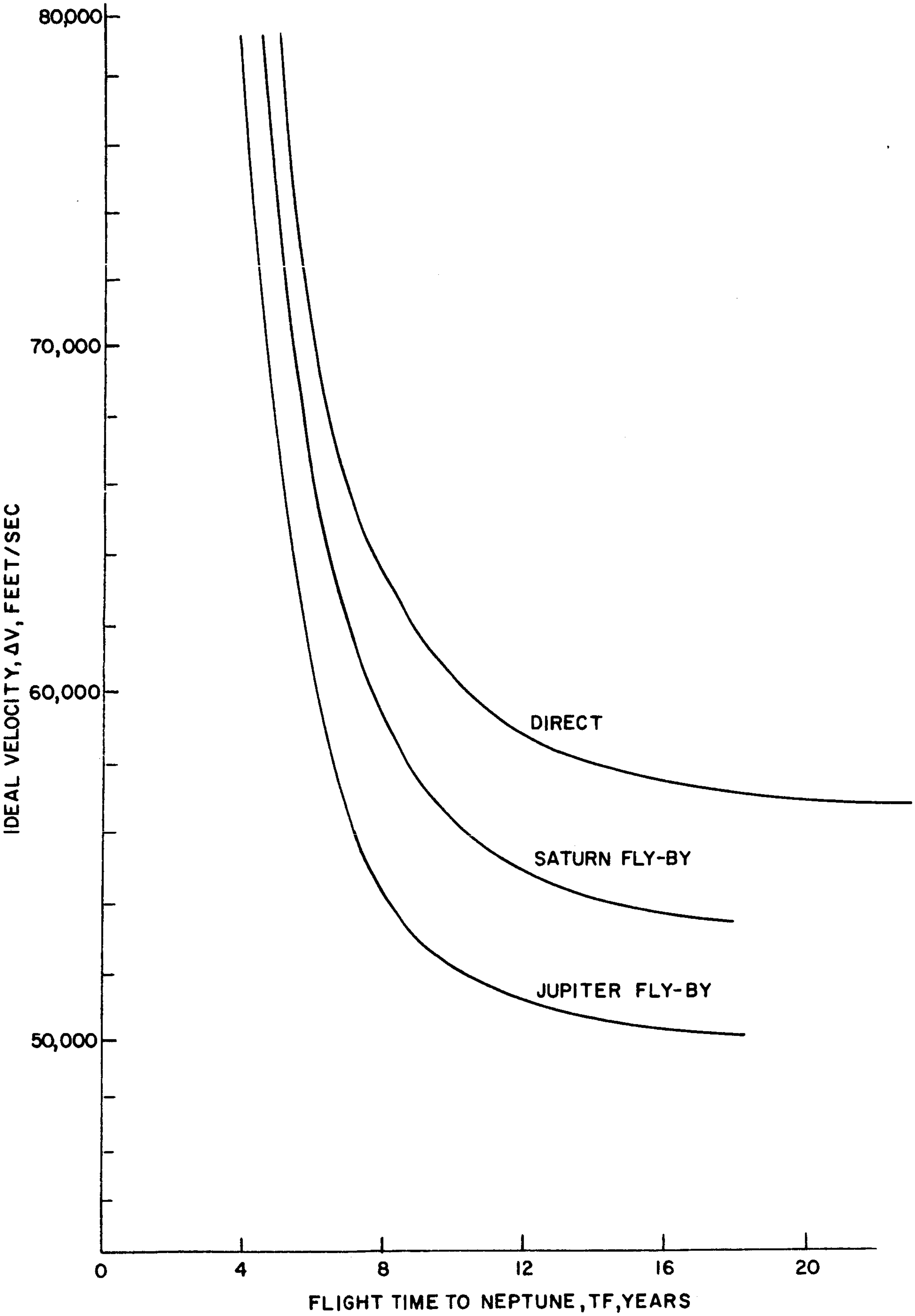

FIGURE A-28. IDEAL VELOCITY FOR FLIGHTS TO NEPTUNE

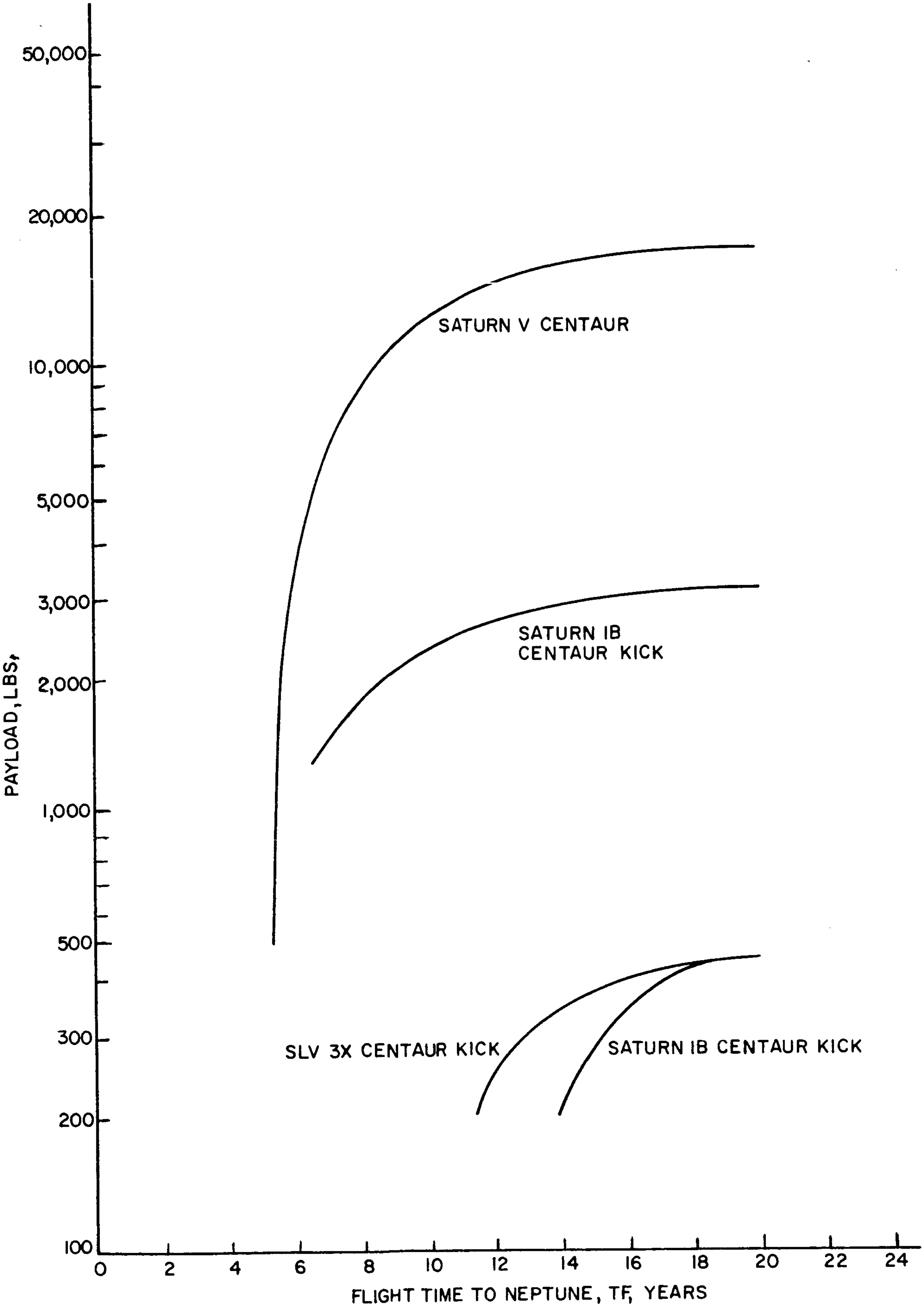

FIGURE A-29. PAYLOADS FOR DIRECT NEPTUNE FLIGHTS

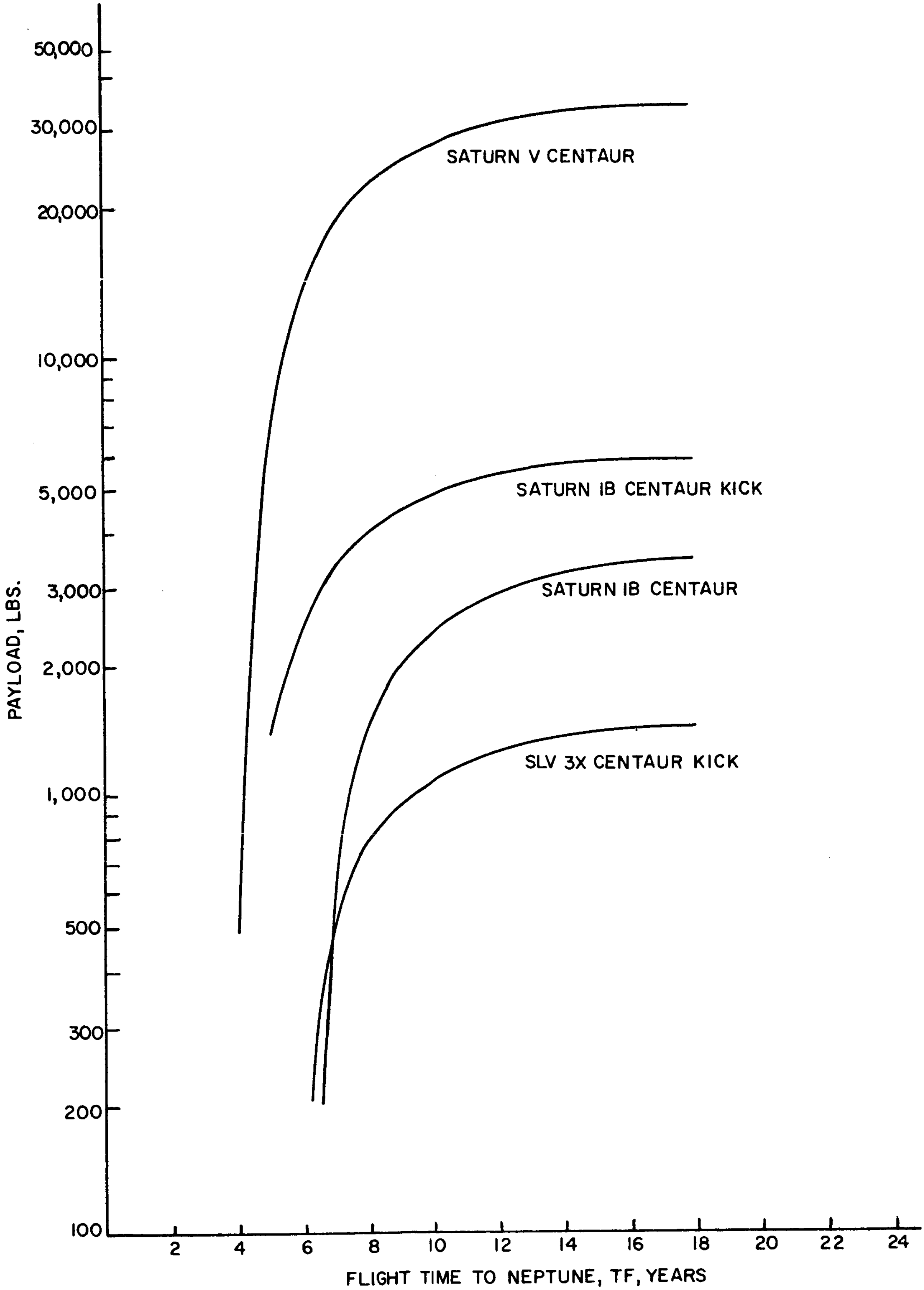

FIGURE A-30. PAYLOADS FOR JUPITER ASSIST NEPTUNE FLIGHTS

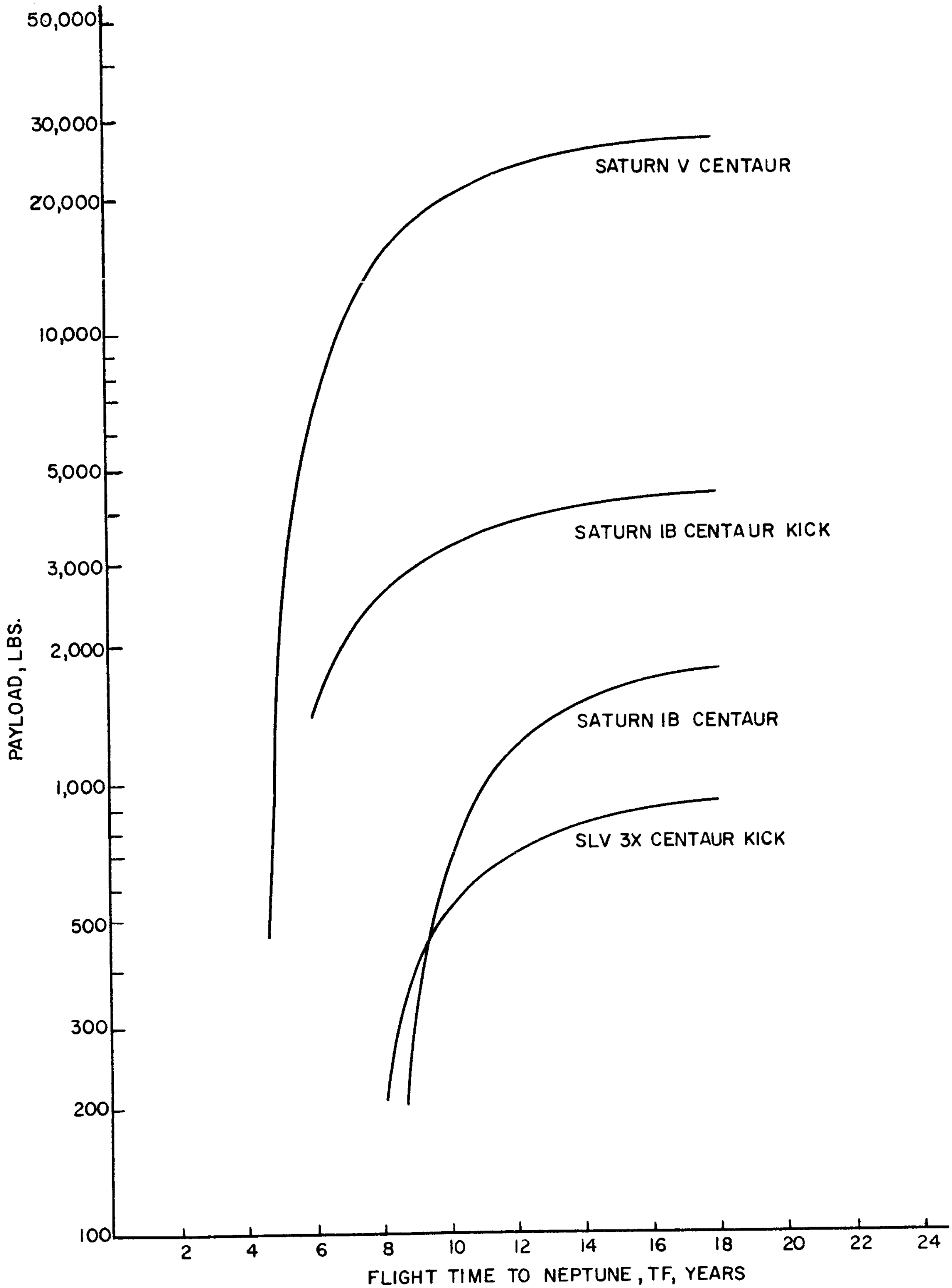

FIGURE A-31. PAYLOADS FOR SATURN ASSIST NEPTUNE FLIGHTS

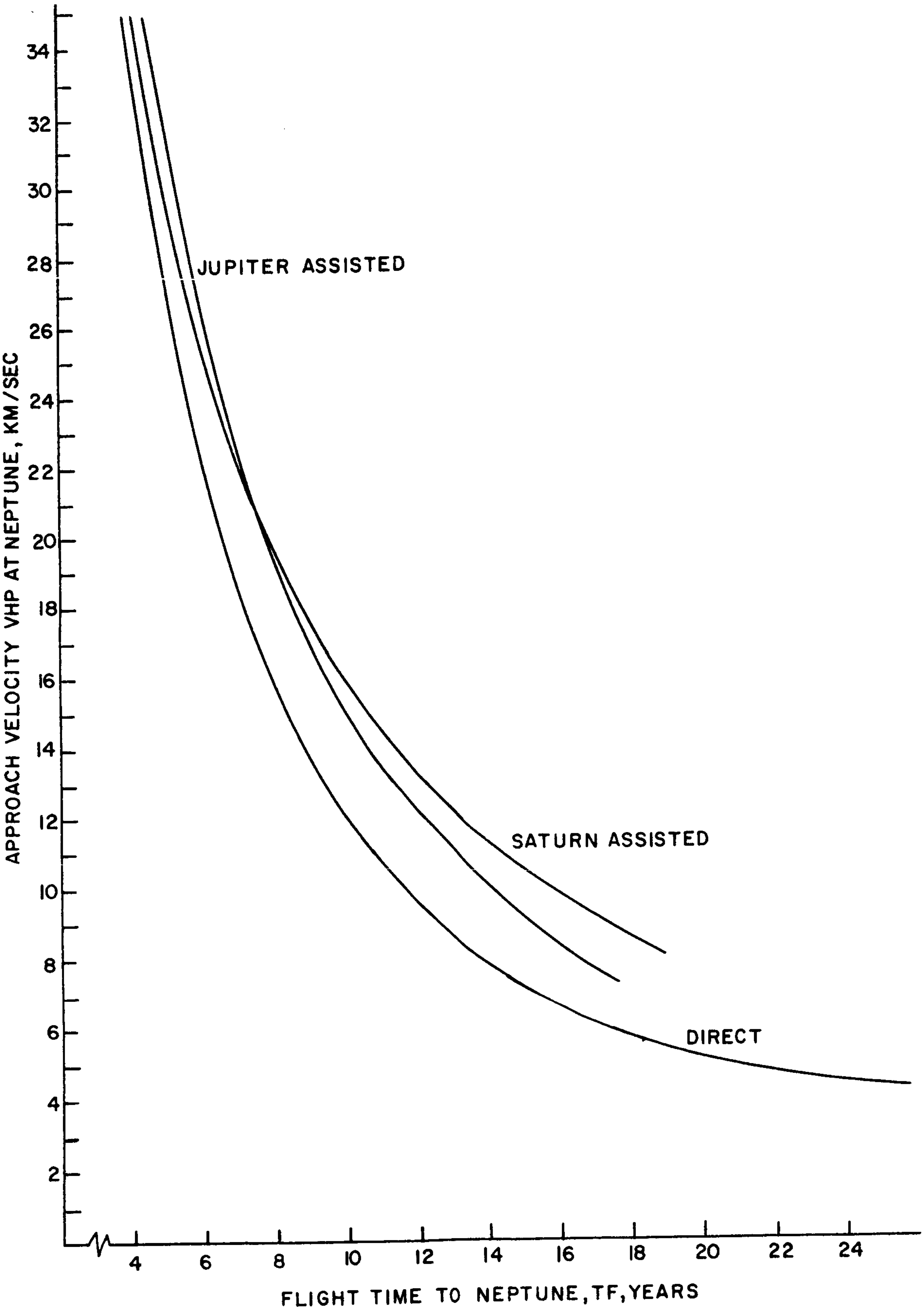

FIGURE A-32. APPROACH VELOCITY FOR FLIGHTS TO NEPTUNE

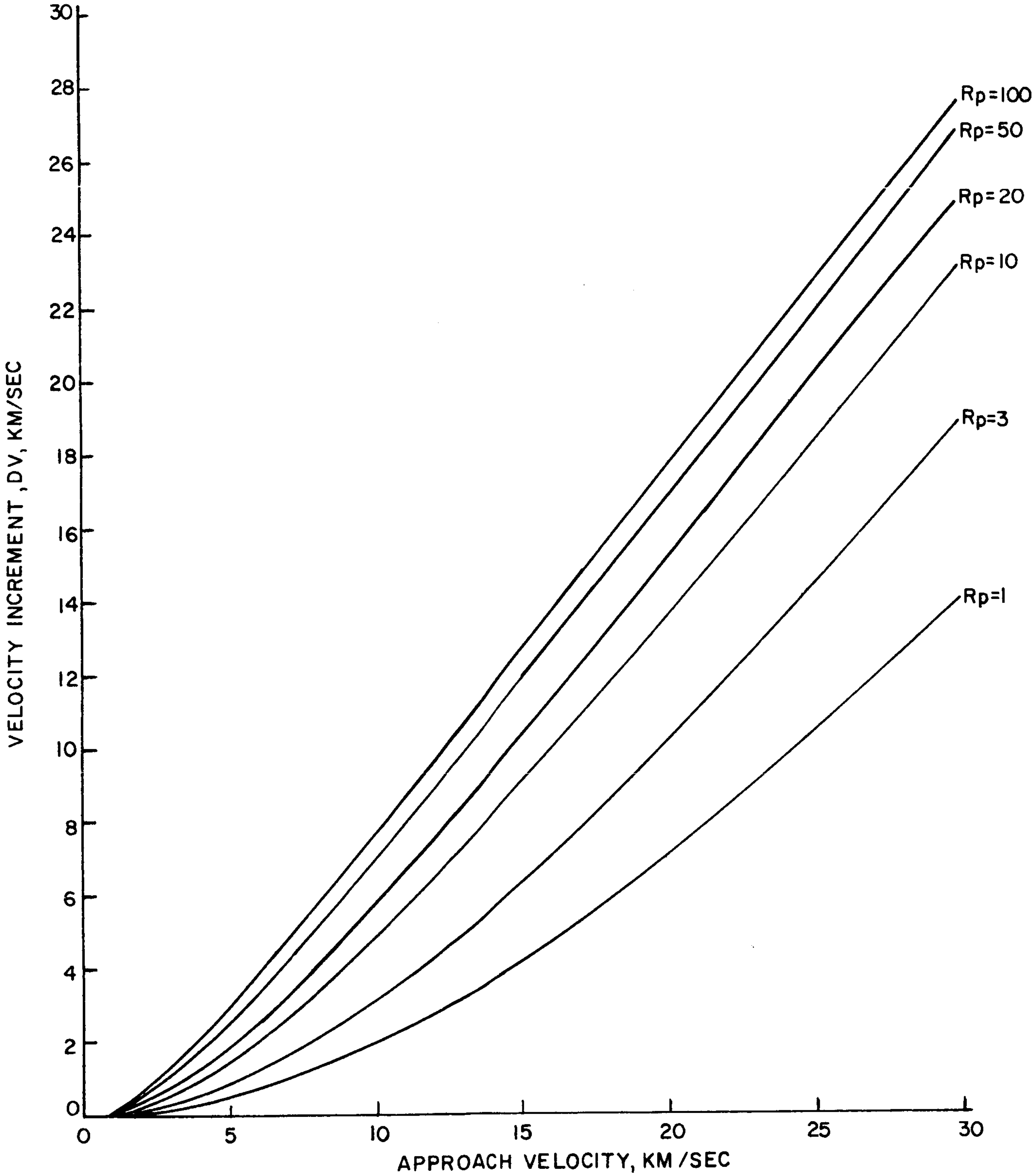

FIGURE A-33. VELOCITY INCREMENT FOR PARABOLIC RENDEZVOUS AT NEPTUNE

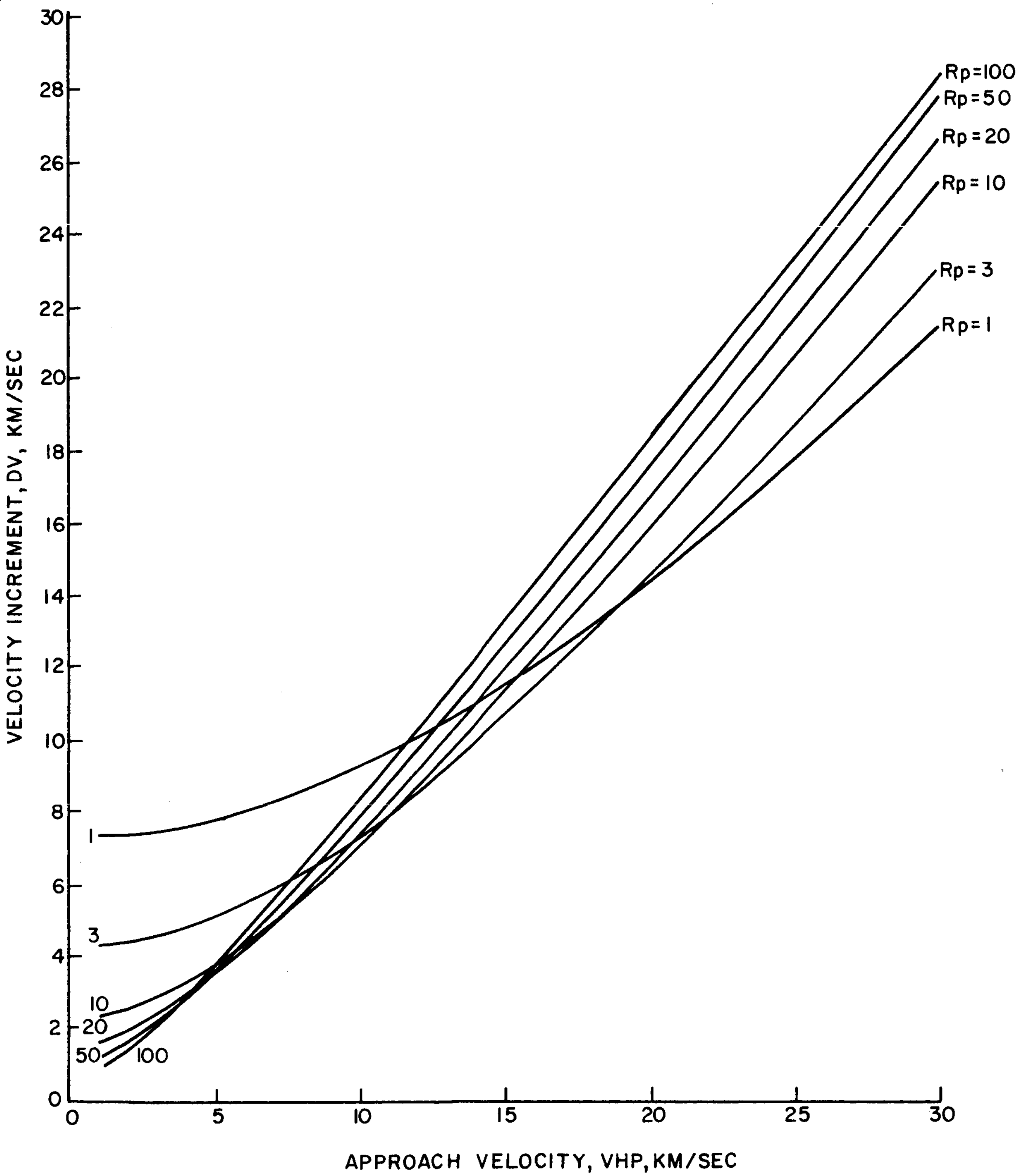

FIGURE A-34. VELOCITY INCREMENT FOR CIRCULAR RENDEZVOUS AT NEPTUNE

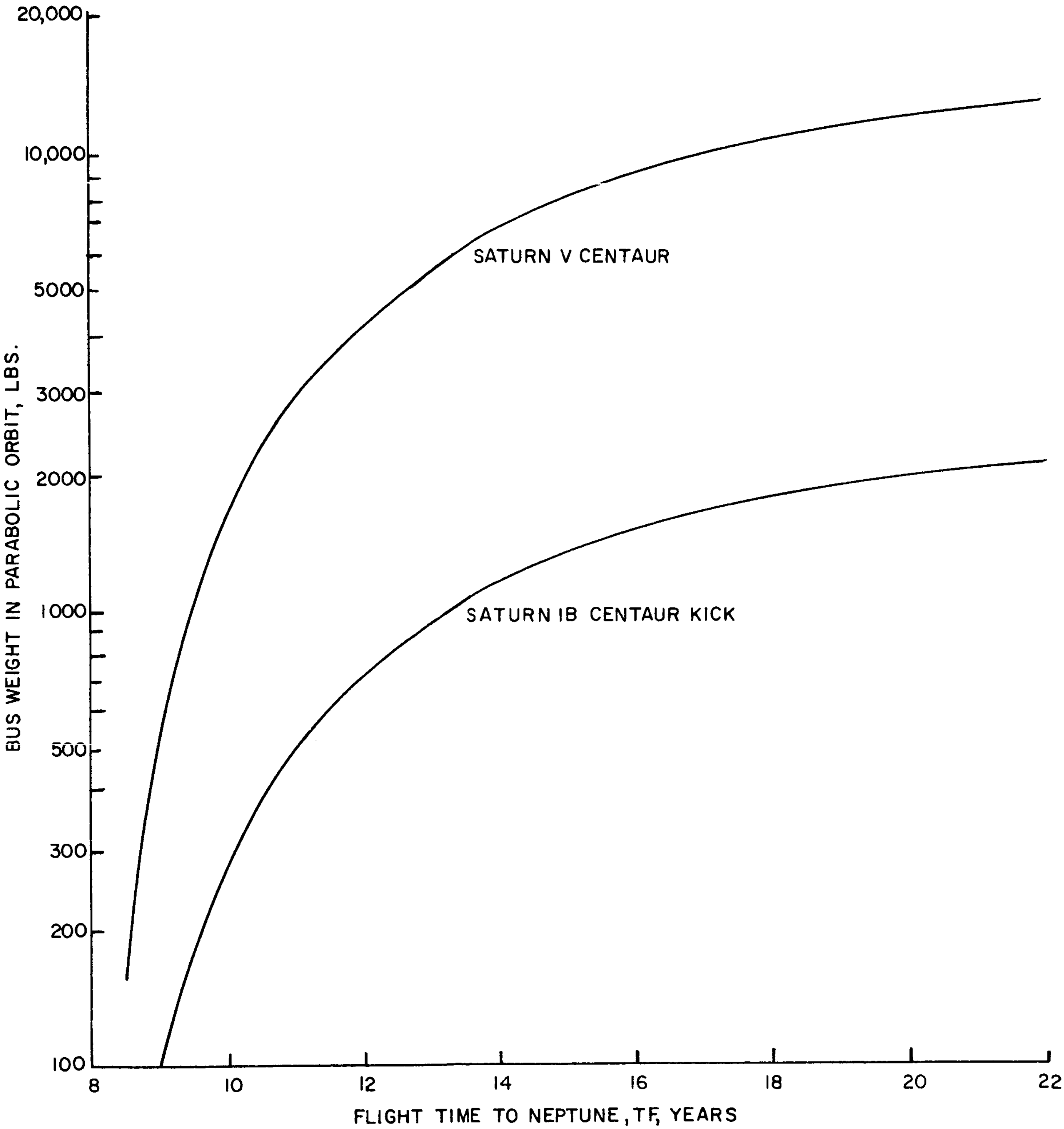

FIGURE A-35. PAYLOADS IN PARABOLIC ORBIT AROUND NEPTUNE. ASSUMES: 3 PLANET RADII MISS. ENGINE ISP = 315 SEC.

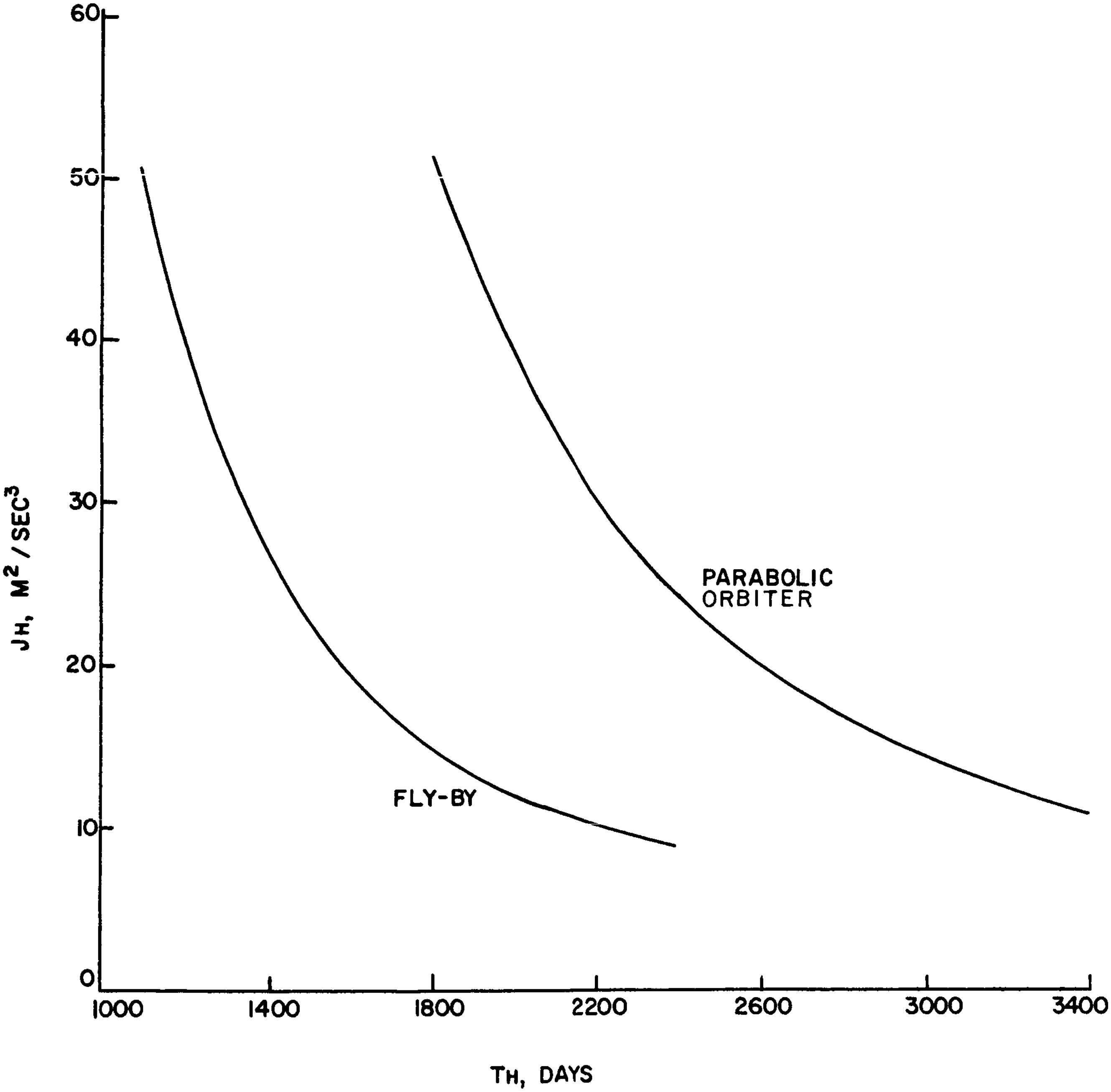

FIGURE A-36. HELIOCENTRIC J REQUIREMENTS FOR NEPTUNE MISSIONS, VARIABLE THRUST MODE

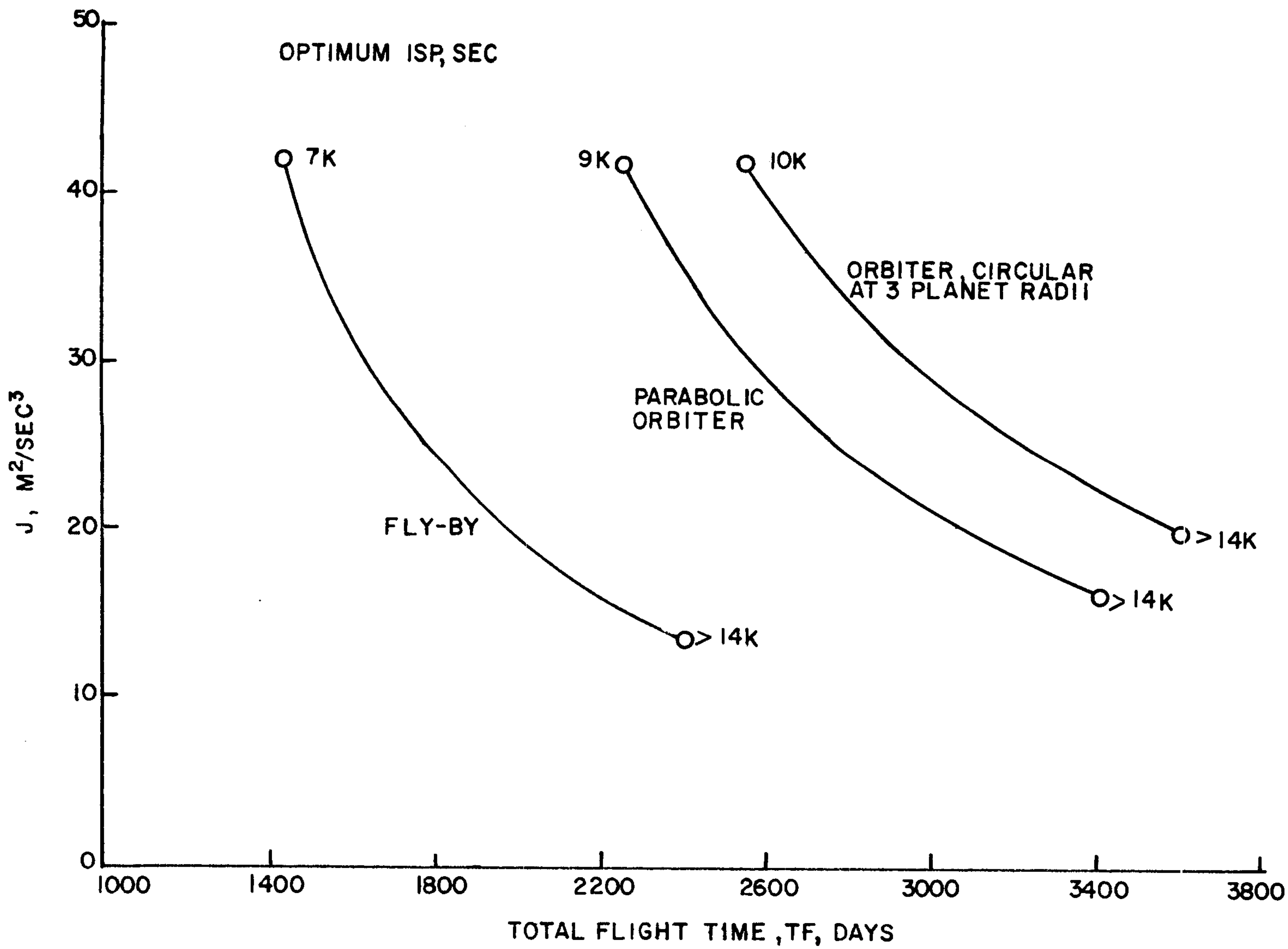

FIGURE A-37. J REQUIREMENTS FOR MISSIONS TO NEPTUNE, CONSTANT THRUST MODE
a_0 ISP = 4.5 m/sec

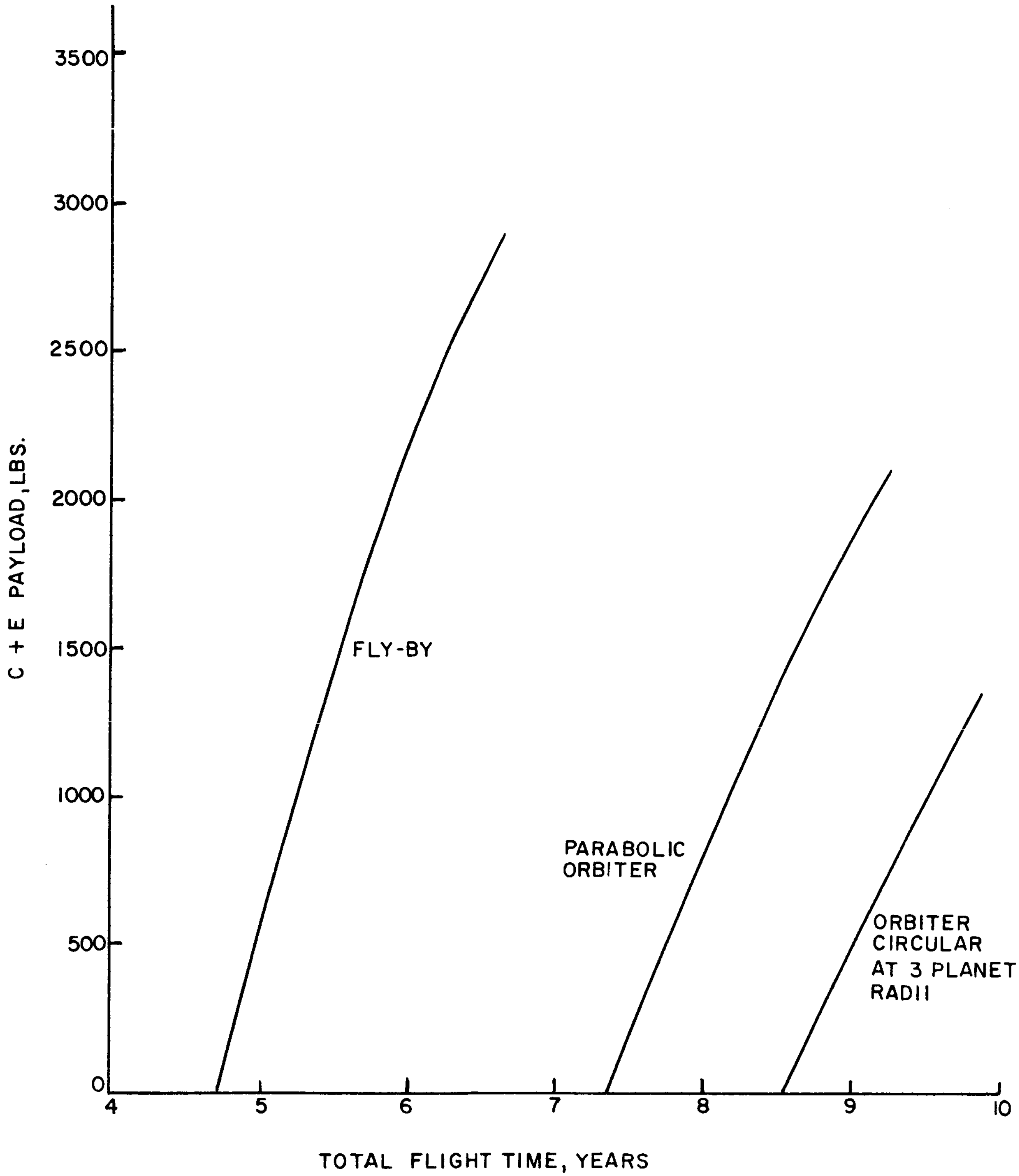

FIGURE A-38. PAYLOAD CAPABILITY FOR LOW-THRUST MISSIONS TO NEPTUNE, SATURN IB THRUSTED STAGE (α = 40 LBS/KWE, P = 250 KWE)

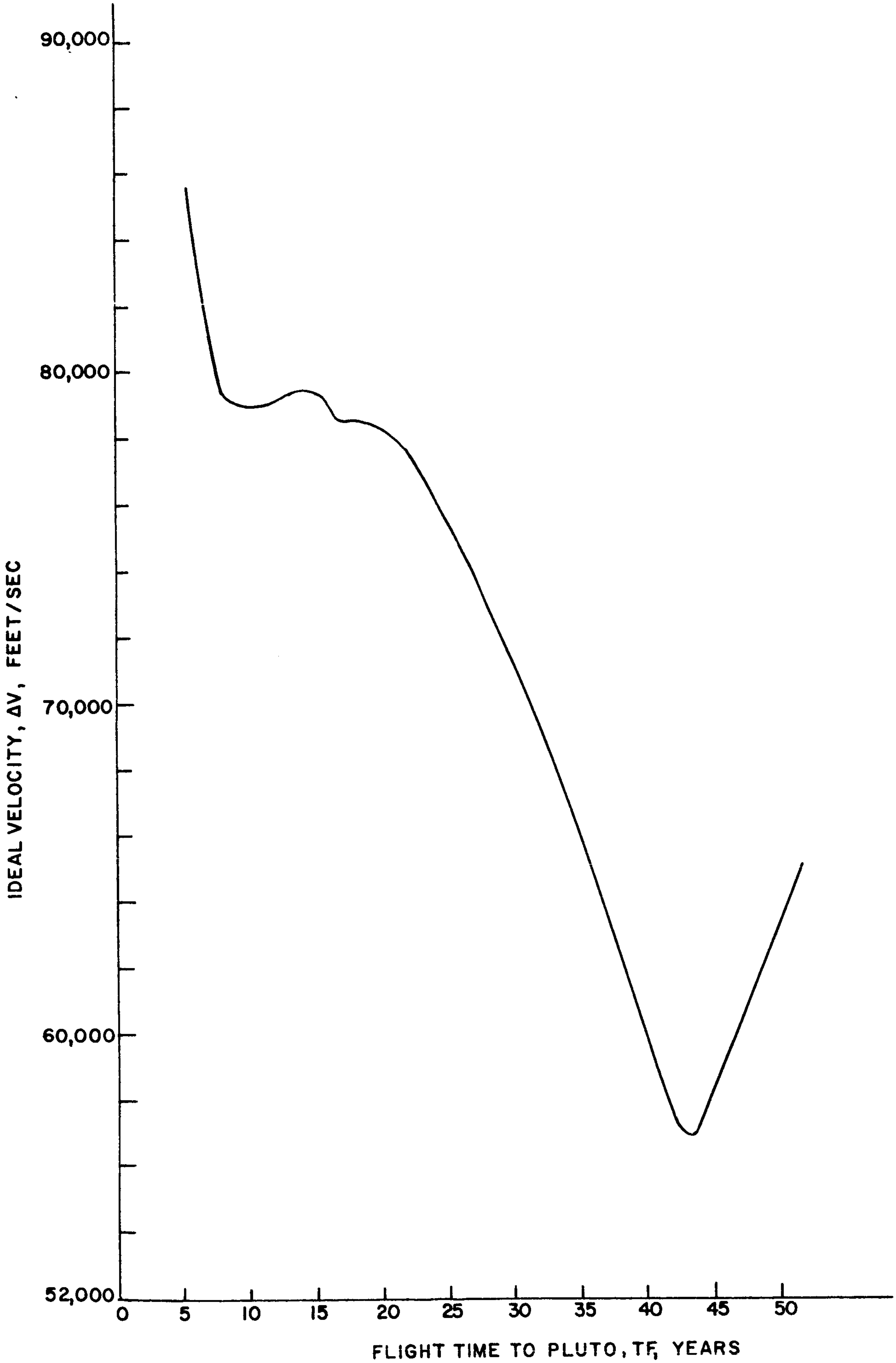

FIGURE A-39. IDEAL VELOCITY FOR FLIGHTS TO PLUTO. 1975 LAUNCHES

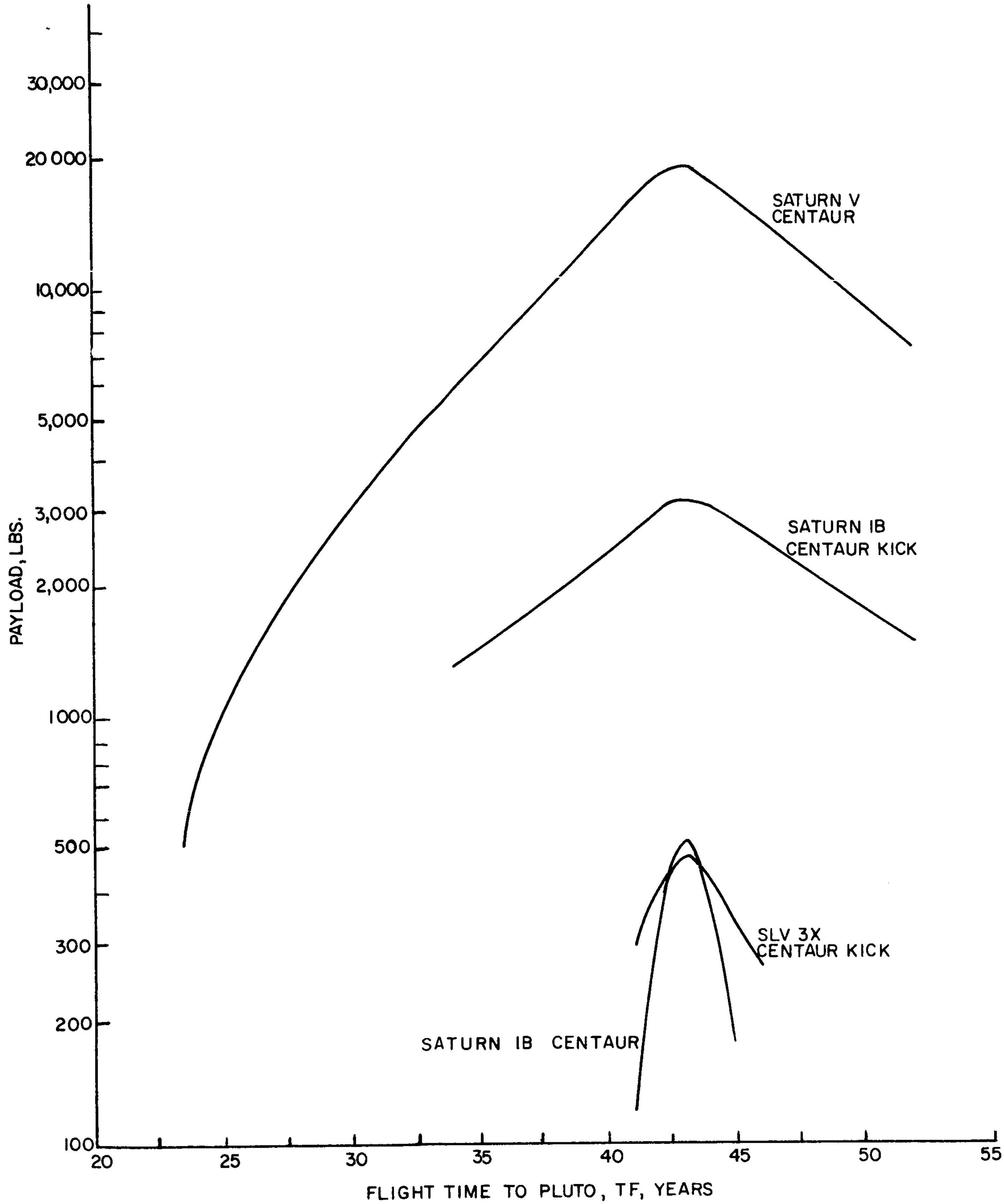

FIGURE A-40. PAYLOADS FOR DIRECT PLUTO FLIGHTS

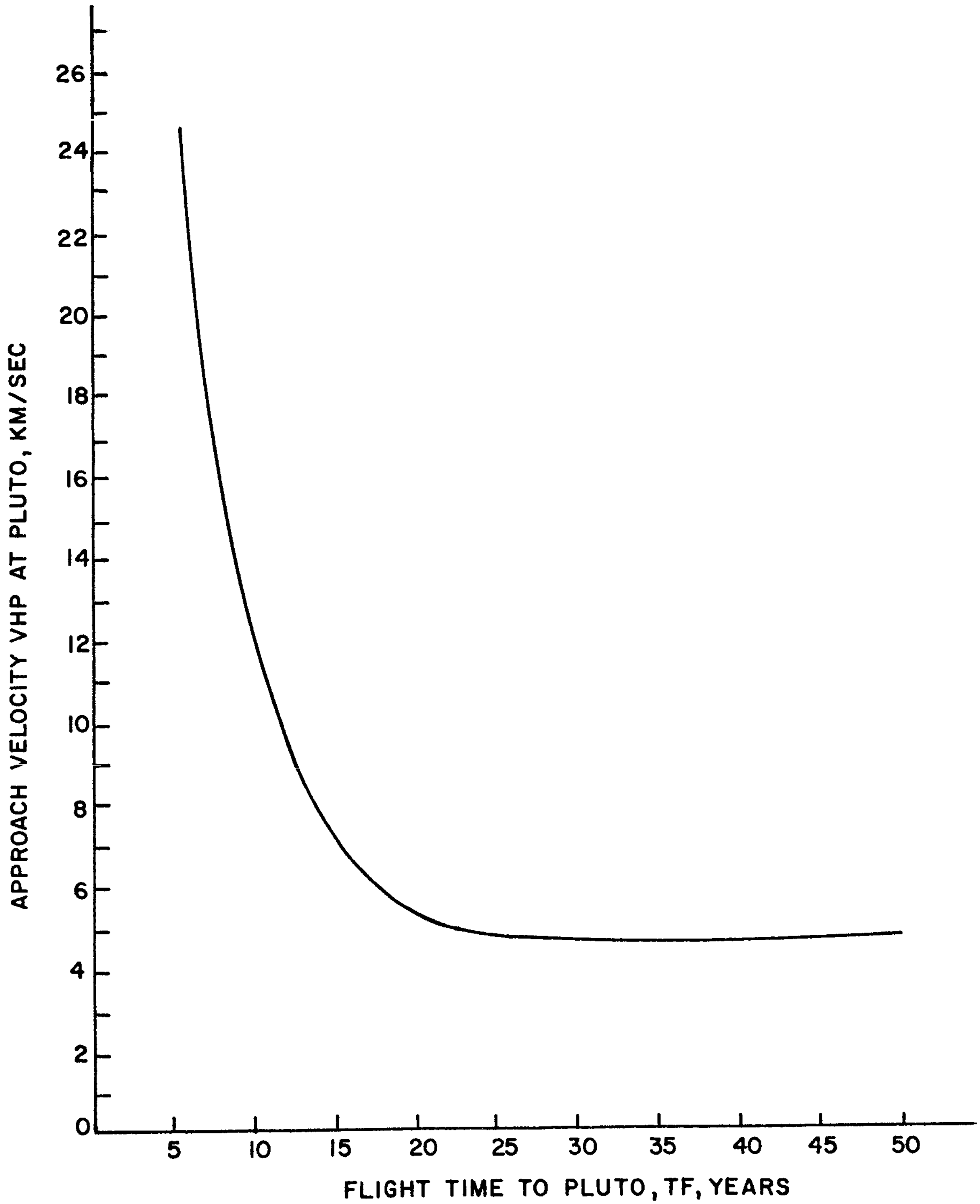

FIGURE A-41. APPROACH VELOCITY FOR FLIGHTS TO PLUTO

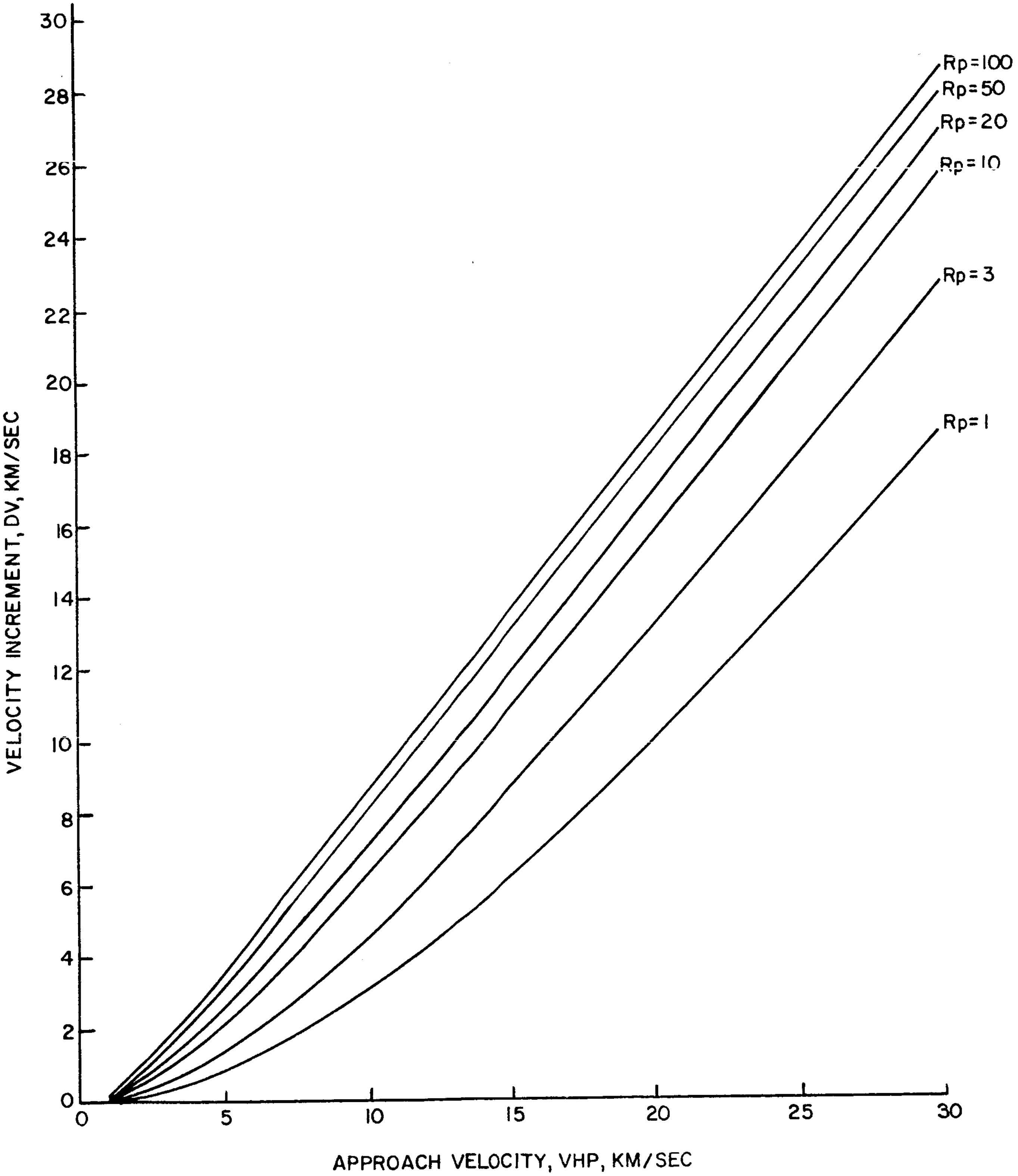

FIGURE A-42. VELOCITY INCREMENT FOR PARABOLIC RENDEZVOUS AT PLUTO

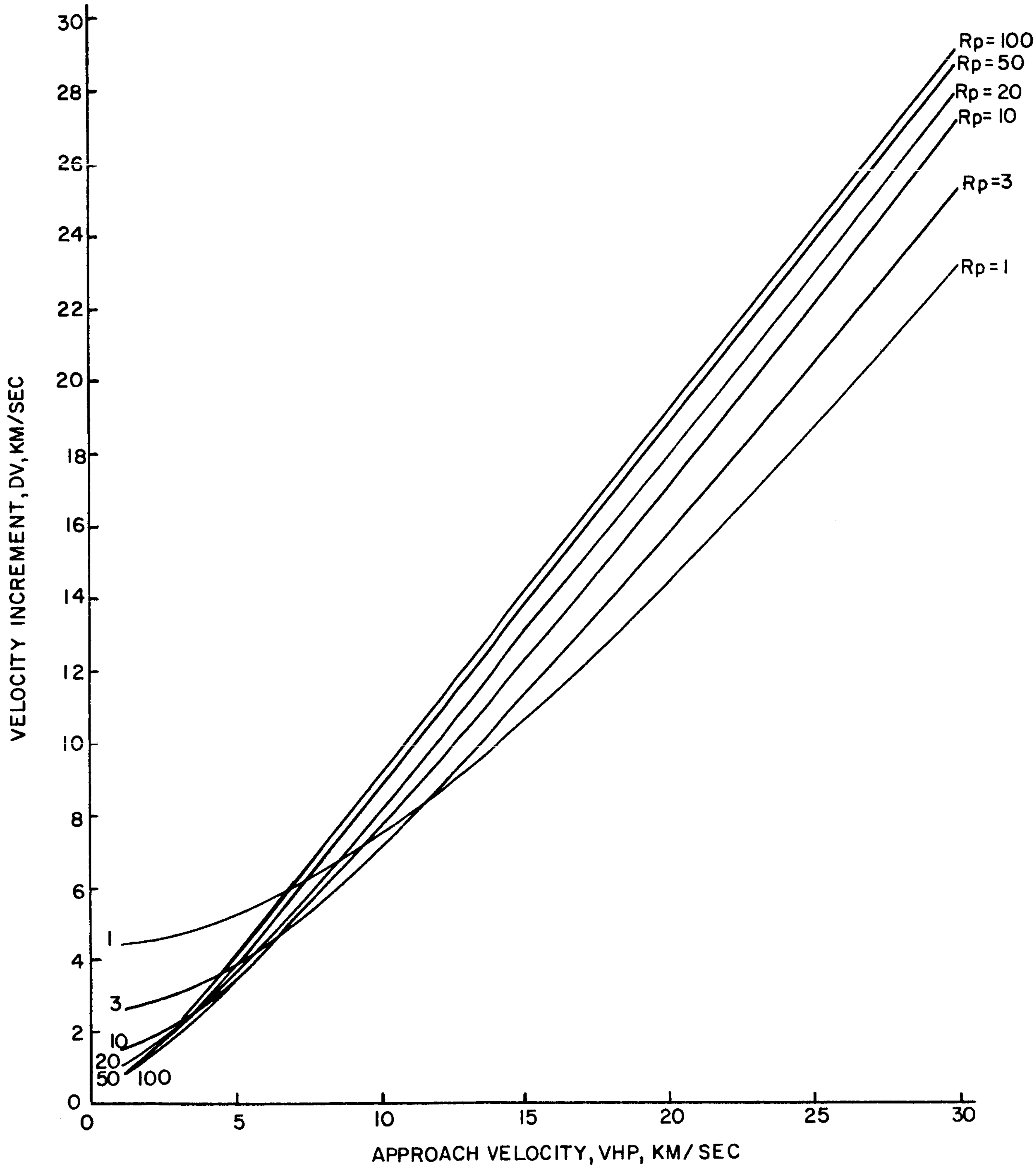

FIGURE A-43. VELOCITY INCREMENT FOR CIRCULAR RENDEZVOUS AT PLUTO

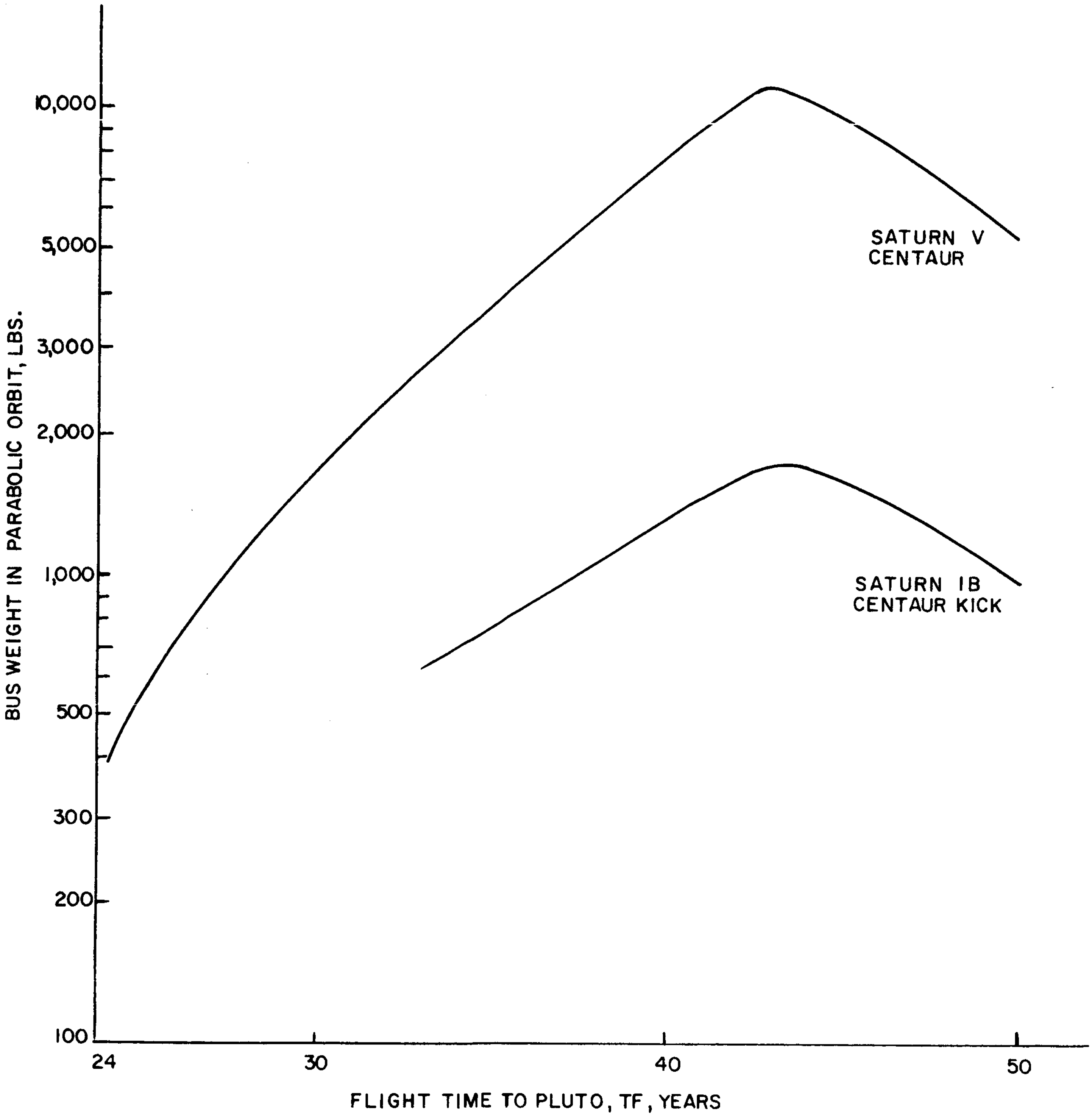

FIGURE A-44. PAYLOADS IN PARABOLIC ORBIT AROUND PLUTO. ASSUMES: 3 PLANET RADII MISS. ENGINE ISP = 315 SEC.

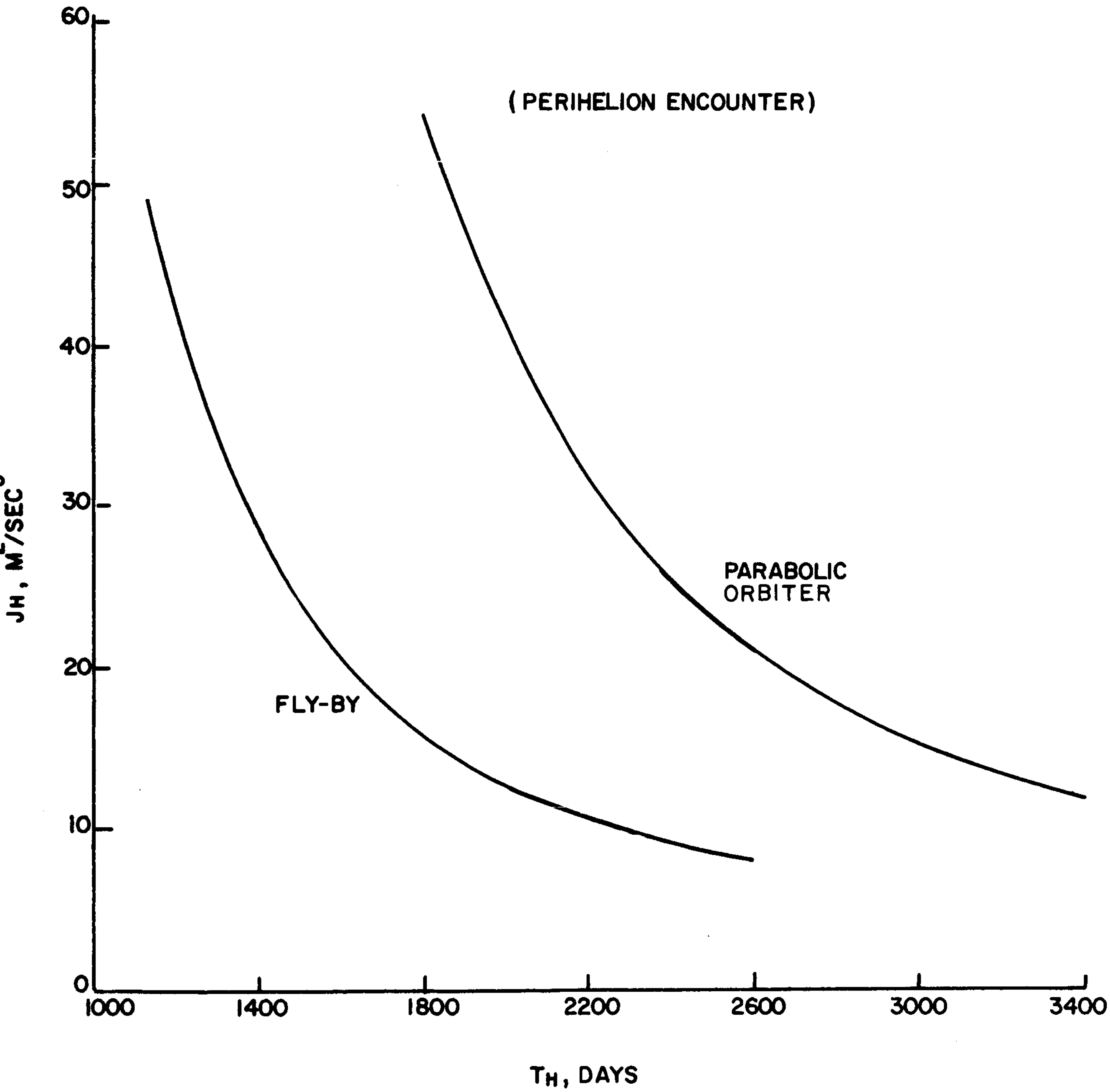

FIGURE A-45. HELIOCENTRIC J REQUIREMENTS FOR PLUTO MISSIONS, VARIABLE THRUST MODE

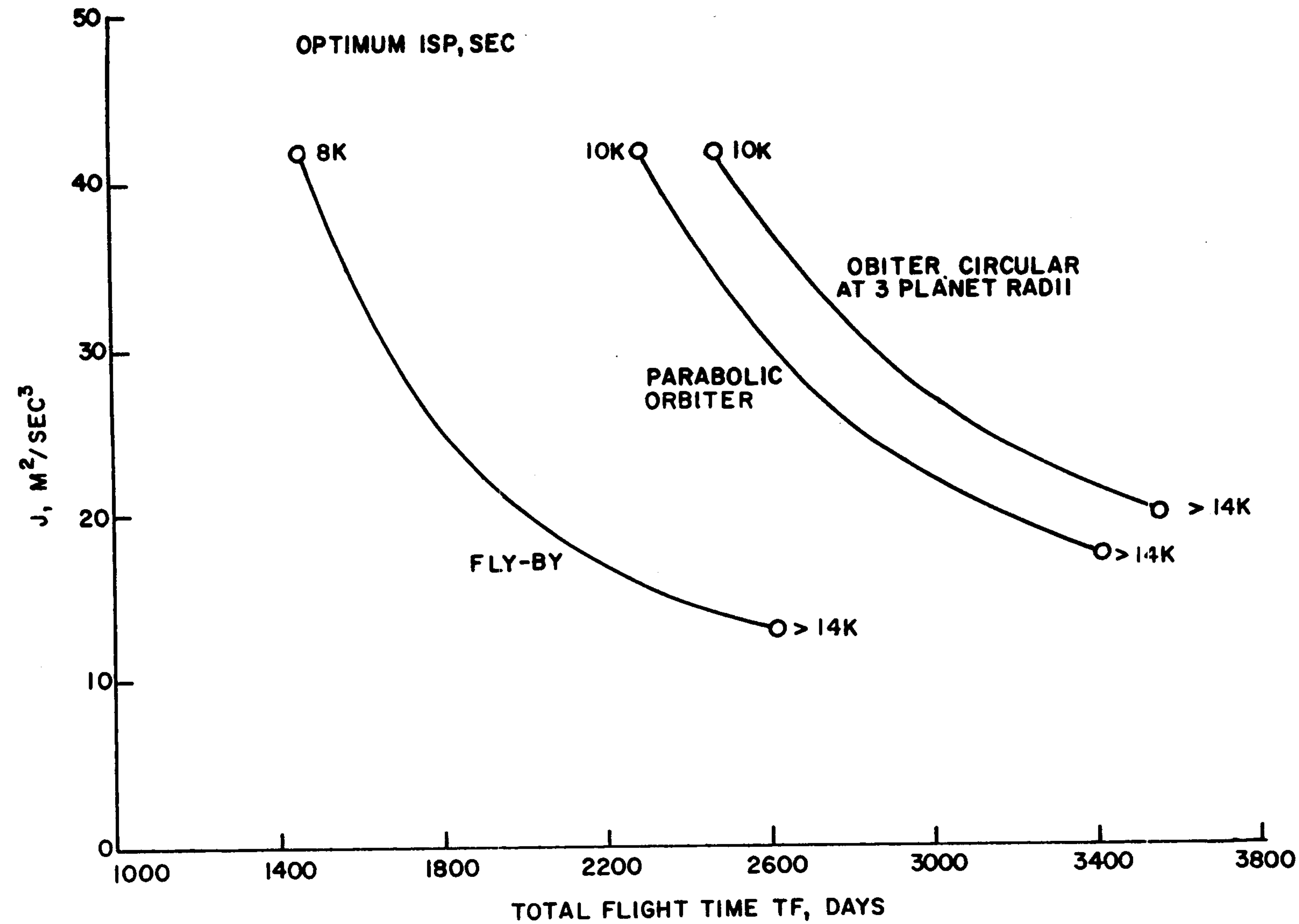

FIGURE A-46. J REQUIREMENTS FOR MISSIONS TO PLUTO, CONSTANT THRUST MODE
a_0 ISP = 4.5 m/sec

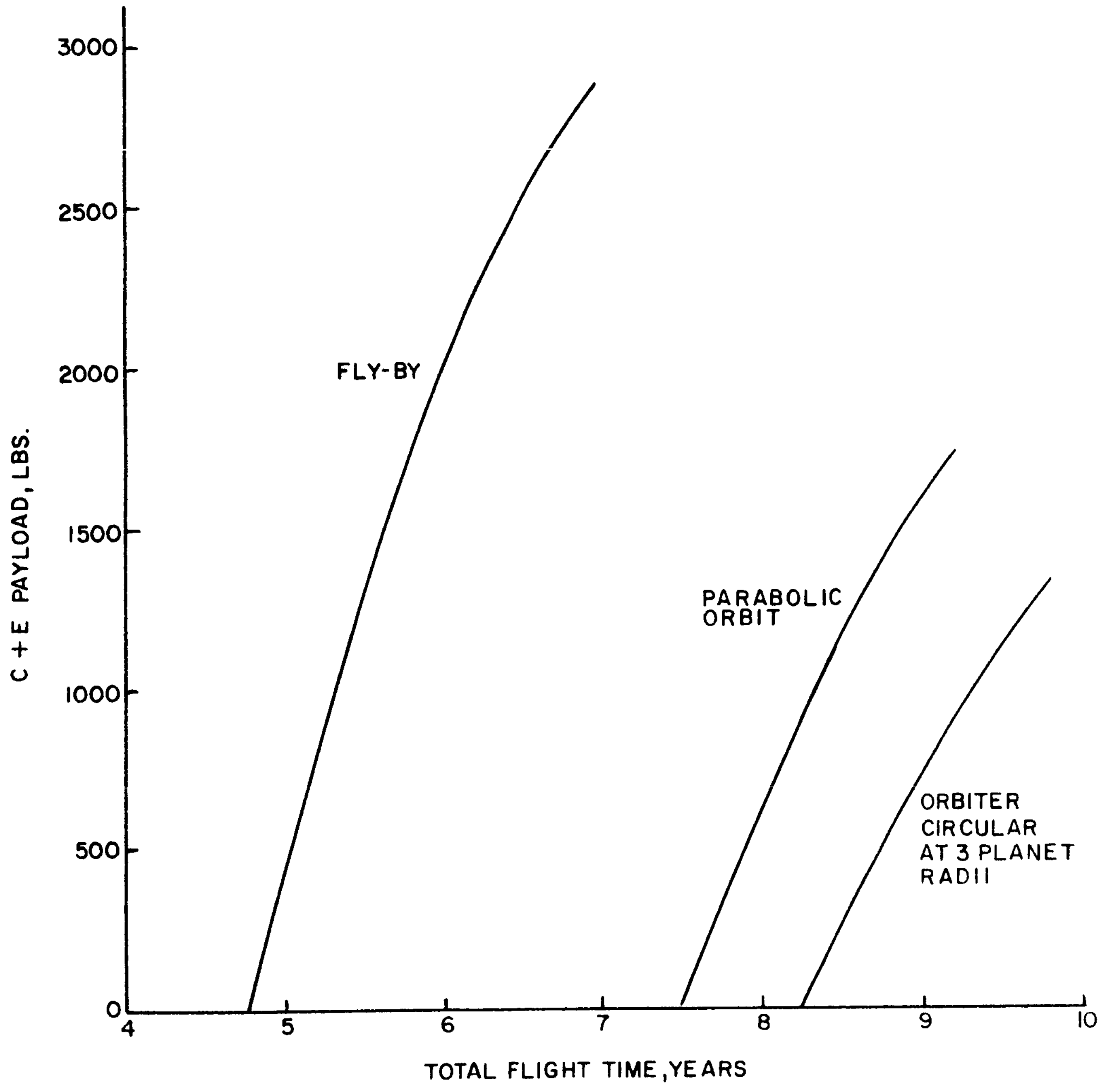

FIGURE A-47. PAYLOAD CAPABILITY FOR LOW-THRUST MISSIONS TO PLUTO, SATURN IB THRUSTED STAGE (α = 40 LBS/KWE, P = 250 KWE)