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URANUS AND NEPTUNE ORBITER MISSIONS VIA SOLAR ELECTRIC PROPULSION

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URANUS AND NEPTUNE ORBITER MISSIONS

VIA SOLAR ELECTRIC PROPULSION

by

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APPROVED:

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FOREWORD

This Technical Report is the final documentation on all data and information required by Task 9: Uranus and Neptune Orbiter Missions. The work herein represents one phase of the study, Support Analysis for Solar Electric Propulsion Data Summary and Mission Applications, conducted by IIT Research Institute for the Jet Propulsion Laboratory, California Institute of Technology, under JPL Contract No. 952701. Tasks 1 through 8 of this study have been reported under separate cover.

SUMMARY

This report describes the characteristics and capabilities of solar electric propulsion (SEP) for performing orbiter missions at the planets Uranus and Neptune. The major rationale for orbiting these planets is to complement and extend the scientific information to be obtained from early flyby and probe missions. The scope of the present study includes an assessment of the science objectives and instrumentation requirements, their relation to orbit size selection, and parametric analysis of SEP trajectory/payload performance.

Determination of atmospheric characteristics, both spatial and temporal, stands out as the main area of orbiter investigation. Since the atmospheres of both planets are unlikely to contain fine structure observables, emphasis is placed on spatial coverage rather than high resolution. The science payload selected for these missions is based largely on the instrumentation proposed by JPL for the TOPS Jupiter Orbiter. Allowing for some modifications, it is found that this payload is applicable at Uranus and Neptune. The candidate instruments include an 800 line silicon vidicon camera having short and long focal length optics, an IR imaging radiometer, a photometer-radiometer, IR and UV spectrometers, a magnetometer, and trapped radiation detectors. Total instrumentation weight and power are 67 kg and 93 watts, respectively.

Orbiter subsystems are also taken from the TOPS Jupiter Orbiter design. The spacecraft inserted into orbit weighs 632 kg. Power for interplanetary cruise and orbit operations is provided by 3 RTG units. A 1.2 x 29 polar orbit (period = 7.2 earth days) is nominally chosen for the Uranus mission. Although an equatorial orbit is best for Uranian satellite

observations, this is not generally available for small ΔV expenditure due to the 98° inclination of Uranus' pole to the ecliptic. The ideal arrival condition for this application next occurs in the year 2027 when the equatorial plane is edge-on to the Sun. The nominal orbit chosen for the Neptune mission is 1.2 x 43 planet radii, inclined 45° to Neptune's equator, and having a period of 10.4 earth days. Transmission rates for the TV picture (5 x 10^6 bits) are 1220 bits/sec from Uranus and 530 bits/sec from Neptune. If DSN reception were 12 hours per day, 78 pictures from Uranus and 47 from Neptune could be obtained per orbit. Since the other instruments require only a small fraction of the telemetry time, it is concluded that adequate science coverage of Uranus and Neptune is possible using the TOPS Jupiter orbiter systems.

Utilizing the Titan IIID/Centaur launch vehicle, minimum flight times of about 3400 days to Uranus and 5300 days to Neptune are required to place the TOPS spacecraft into the nominal orbits. However, this requires an optimum-sized SEP powerplant of 35 kw and propulsion times exceeding 600 days for Uranus and 1000 days for Neptune. Reduction in power and propulsion time to more practical values can be obtained at the expense of about 200 days additional flight time. An example baseline mission summary is presented in Table S-1. A common SEP/orbiter system design is chosen for both Uranus and Neptune. The SEP "stage" has a total weight of 1008 kg and is jettisoned after its propulsion function is accomplished. The propulsion system is rated at 20 kw (power input at 1 AU) with the ion thrusters operating at 3000 sec. specific impulse. Mercury propellant and tankage comprise 408 kg of the total SEP stage weight; this includes added propellant for an extended launch window of 20 days or more. The chemical retro stage weighs 420 kg and provides an orbit insertion ΔV of 1.5 km/sec (space-storable propellant such as Fluorine/Hydrazine is assumed for this

TABLE S-1

EXAMPLE BASELINE MISSION SUMMARY
FOR URANUS AND NEPTUNE ORBITERS

1. Common SEP/Orbiter System Weights

Solar Electric System		1008 (KG)
Propulsion System (20 kw/3000 sec)	600	
Propellant + Tankage	408	
Chemical Retro Stage (383 sec/0.25) $\Delta V = 1.5$ km/sec		420
Orbiter Systems (TOPS)		632
Earth Departure Weight		2060 (KG)

2. Mission Parameters

	URANUS	NEPTUNE
Launch Vehicle	Titan IIID/Centaur	Titan IIID(7)/Centaur
Launch V_{HL} (KM/sec)	7.2	8.2
Max. Injected Weight (KG)	2110	2200
Flight Time (days)	3600	5500
Max. Propulsion time (days)	440	455
Approach Velocity V_{HP} (KM/sec)	6.5	7.2
Orbit Size (Radii)	1.2 x 29	1.2 x 43
Orbit Period (Earth Days)	7.2	10.4

application). For the Neptune mission, the Titan IIID(7)/Centaur launch vehicle is specified since the standard 5-segment Titan is not quite adequate unless the power or flight time is increased. Direct launch opportunities to Uranus or Neptune occur once each year. The above performance results are fairly representative of any launch year since these distant planets have nearly circular orbits close to the ecliptic plane.

A comparison of SEP performance with the all-ballistic flight mode is shown in the report assuming the proposed high-energy Versatile Upper Stage (VUS) as a tradeoff against SEP. The following conclusions result: (1) for direct flights to Uranus, the Titan IIID/Centaur/SEP (20 kw) and the Titan IIID(7)/Centaur/VUS have nearly equivalent performance for TOPS-class orbiters, (2) for direct flights to Neptune, the VUS and SEP (20 kw) have nearly equivalent performance when either is the upper stage of the Titan IIID(7)/Centaur, (3) for Jupiter swingby flights to Uranus or Neptune, the VUS and SEP (15 kw) have nearly equivalent performance when either is the upper stage of the Titan IIID/Centaur, and (4) Jupiter swingby offers performance advantages relative to direct flights but launch opportunities are infrequent (13-14 year intervals); flight times to Uranus and Neptune can be reduced by about 1 and 2 years, respectively, or, for the same flight times, additional payload such as atmospheric probes can be carried.

In conclusion, this study has shown that solar electric propulsion can be used effectively to accomplish elliptical orbiter missions at Uranus and Neptune. However, because of the very long flight times required, it must be admitted that these mission profiles are not too attractive. Previous studies have shown that nuclear electric propulsion, if developed, would allow much faster trips; 5 years to Uranus and 8 years to Neptune.

CONTENTS

LIST OF FIGURES

LIST OF FIGURES (continued)

LIST OF FIGURES (continued)

LIST OF TABLES

URANUS AND NEPTUNE ORBITER MISSIONS VIA SOLAR ELECTRIC PROPULSION

1. INTRODUCTION

1.1 Study Background

The early exploration of Uranus and Neptune from flyby spacecraft is expected to take place with the JUN Grand Tour launched in 1979. Since the atmosphere is the predominant feature of these planets, atmospheric probes penetrating to at least 10 bars will be very useful in augmenting the remote sensing data obtained from the flybys (Price and Waters 1971). It may be possible to carry an atmospheric probe on the Grand Tour, but, if not, such missions will likely be planned for the early 1980's. Orbiters can provide useful scientific data to complement and extend information obtained from early flyby and probe missions. One of the main advantages of the orbiting mode is the time available for obtaining extended coverage in a dynamic planetary environment.

Although Uranus and Neptune are extremely large planets very little is known of their physical properties due to their extreme distances from the sun ($\sim$ 19 and 30 AU respectively). They appear to be cold, featureless, predominately hydrogen-methane giants without any of the spectacular eccentricities of Jupiter and Saturn. Uranus is rolled over on its side, its polar axis inclined 98° to the ecliptic, and has an unusually regular satellite system, while Neptune is "upright" with two very irregular satellites. These phenomena bring up questions of energy and momentum conservation during solar system evolution. However, with these exceptions, both planets seem conspicuously dull. This impression is perhaps due to the relatively few and difficult earth-based observations of Uranus and Neptune.

Although knowledge of Uranus and Neptune is limited at present, enough is known to design, in a preliminary fashion, orbiter missions to both planets. Such orbiters can investigate global atmospheric motions, composition, and thermal balance, detect and examine the distributions and intensities of the particle and fields environment, and at a minimum, visually image the majority of each satellite system. It must be stressed that at this point in time such orbiter payloads and missions are merely preliminary designs. The Grand Tour flybys at Uranus and Neptune in the late 1980's will provide more detailed information upon which orbiter mission can be concretely based.

Solar electric propulsion (SEP) has been shown to offer performance advantages over the contemporary ballistic flight mode for many missions throughout the solar system, including those to the outer planets (Friedlander 1970). A recent study conducted under the present contract has examined SEP application for Jupiter and Saturn orbiter missions (Friedlander and Brandenburg 1970). This new technology is under active development and is expected to be fully flight-proven by the late 1970's.

For outer planet missions, the SEP spacecraft may properly be considered as an additional stage above the launch vehicle. The required mission velocity is attained gradually over a relatively long period of time as a consequence of the low thrust acceleration and high specific impulse operation. Jettisoned after attaining this velocity, the stage delivers the combined mass of the orbiting spacecraft and the chemical retro-propulsion system needed to achieve the desired orbit.

1.2 Study Objectives and Approach

This study is undertaken to determine the performance capability and characteristics of solar electric propulsion for accomplishing elliptical orbit missions of Uranus and Neptune, and to compare the SEP performance with the all-ballistic flight mode. The study includes a delineation of science objectives, measurements and instrumentation requirements, and orbit selections. Specific study guidelines are listed below.

1. Relate orbit and payload selections to useful science goals (particle and fields, and planetology class missions to be considered).

2. Examine the TOPS orbiter (preliminary design) proposed by JPL for its applicability to Uranus and Neptune missions.

3. Consider the Titan IIID/Centaur as the baseline launch vehicle for the SEP missions, if possible.

4. Make maximum use of previously generated trajectory data for this study.

5. Use simplified performance scaling relationships to develop parametric data for other launch vehicle candidates and orbit size tradeoffs.

A weight breakdown of the TOPS orbiter system is given in Table 1-1. The spacecraft inserted into orbit weighs 632 kg of which about 10 percent comprises the science instruments. The nominal retro system weighs 500 kg and provides an

TABLE 1-1

TOPS (JUPITER ORBITER) WEIGHT ESTIMATES
(Data provided by JPL)

Orbiting Spacecraft		632 (KG)
Science	67	
RTG (3)	127	
RTG Shielding	5	
Power Conditioning	35	
Control/Conditioning Logic	6	
Control Computer	16	
Timing Synchronizer	2	
Measurement Processor	2	
Data Storage	35	
Attitude Control	40	
Attitude Propulsion	10	
Temperature Control	21	
Pyrotechnics	5	
Devices	39	
Cabling	22	
Radio	33	
Antennas	29	
Meteoroid Protection	6	
Structure	133	
Retro Propulsion		500
ΔV = 1.64 km/sec		1132 (KG)
I_{sp} = 383 sec		
Inerts = 25% of propellant		

orbit insertion ΔV of 1.64 km/sec. An advanced space-storable liquid propulsion system is assumed for this application. As stated above, the TOPS design will be used as a reference point for the present study, but will not preclude such modifications as are thought to be necessary.

The report is organized as follows: Section 2 discusses the current scientific knowledge of Uranus and Neptune, the orbiter mission objectives and experiment requirements, and relates these to the capability of the TOPS science payload and engineering support subsystems. Also included is a discussion of orbit selection and corresponding profile data. Section 3 describes the SEP trajectory analysis and payload performance for these missions showing the effects of such mission/system parameters as flight time, launch vehicle, SEP power rating and propulsion on-time, and orbit size. Also included is a comparison with the all-ballistic flight performance for both direct and Jupiter swingby missions. Section 4 presents data for an example baseline mission to each planet wherein a common SEP/orbiter system design is assumed for both Uranus and Neptune applications.

2. SCIENCE PAYLOAD AND ORBIT SELECTION

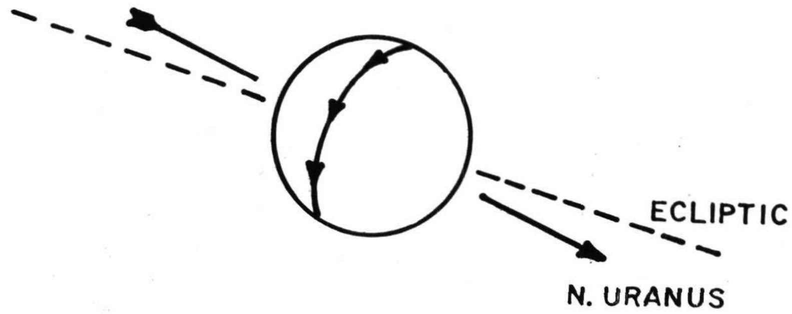
2.1 Present Knowledge of Uranus and Neptune

Currently very little is known of the physical properties of Uranus and Neptune, due primarily to their great distances from earth. Both Uranus and Neptune appear as featureless, greenish disks with some limb darkening. Past observers claimed to have seen two faint belts inclined 25° to Uranus' equator but the recent stratoscope photos of Uranus, one* of which is shown in Figure 2-1, includes no indication of the belts' presence. There have also been no clouds or belts sighted on Neptune.

Hydrogen has been identified as the major constituent of the atmospheres of these two planets. However, the few spectroscopic observations made to date and the uncertainties about the conditions in either planet's atmosphere make it difficult to calculate the actual hydrogen abundances. Belton et al (1971) arrives at a 480 km. atm. column abundance of hydrogen based on a model which may not be appropriate at such a high hydrogen abundance (Newburn 1971). The planets' greenish appearance is due to the strong absorption in the red and infrared of atmospheric methane. The amount of atmospheric methane probably lies between 3 and 7 km-atm for both planets. Ammonia has not been found but may exist within the atmospheres. Helium has not been spectroscopically identified and nothing can be said about it (McElroy 1969), except that it probably is present at the cosmic abundance level.

The effective brightness temperatures of Uranus and Neptune at 20 μ are about 55° K and 44° k, respectively (Newburn 1971).

* This photo is the average of 17 Uranus photographs made with Stratoscope II, Flight Number 7. Courtesy of Dr. Martin Tomasko and Princeton University.



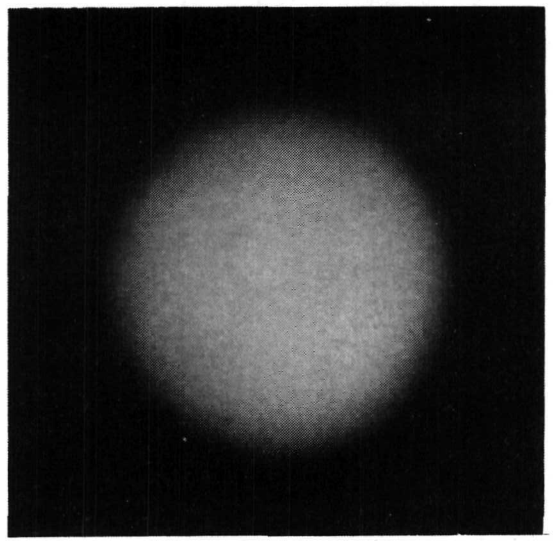


FIGURE 2-I. URANUS (STRATOSCOPE II PHOTO)

Although the Bolometric albedos of both planets are poorly known due to their small range in phase as seen from earth, neither appears to have an internal heat source, as Jupiter and Saturn are suspected of having. However this cannot be conclusively stated as both Uranus and Neptune have radio brightness temperatures considerably larger than expected, as do Jupiter and Saturn, although this may be due to saturated ammonia vapor in their upper atmospheres.

The thermal balance of the Uranian atmosphere is unusual due to the tilt (98°) of the polar axis with respect to the ecliptic. In 1966 when Uranus was facing "side-on" to the Sun the entire planet was exposed to sunlight in one 10.8 hour rotation period. In 1985 a pole will point directly at the Sun and only one hemisphere will be heated directly. This will cause a temperature differential with the sunlite hemisphere's effective temperature being about 20 percent higher ($\sim$ 67°K) than the "equal heating" value.

The rotational periods of Uranus and Neptune are 10.8 $\pm$ 0.5 and 15.8 $\pm$ 1.0 hours, respectively. These rates, determined using Doppler shifts (since there are no features to time), are not very accurate but are the best to date. The physical properties of Uranus and Neptune are summarized in Table 2-1.

While the properties of Uranus and Neptune are very similar their satellite systems are very dissimilar. The satellite systems are listed in Table 2-2. The Uranian satellites are all in the equatorial plane in nearly circular, posigrade orbits. Several satellites, Titania, Oberon and possibly Ariel, have variable brightness curves when their orbit planes are normal to the line of sight, indicating that their polar axis' must be highly inclined to their orbits. The

TABLE 2-1

THE PHYSICAL AND ORBITAL PROPERTIES OF URANUS AND NEPTUNE

	URANUS	NEPTUNE
GRAVITATIONAL MASS, GMp	5.788×10^6 km^3/sec^2	6.891×10^6 km^3/sec^2
MASS (earth = 1)	14.5	17.2
MEAN RADIUS	~ 25,300 km	~ 23,400 km
MEAN DENSITY	1.33 g/cm^3	1.92 g/cm^3
COLOR INDEX B-V (Sun - 0.63)	0.65	0.41
BRIGHTNESS TEMPERATURE (17.5 μ - 25 μ)	55 + 3°K	44° (Estimate)
MEAN DISTANCE	19.18 AU	30.06 AU
INCLINATION	0.7732°	1.7719°
ORBIT ECCENTRICITY	0.04726	0.00859
ROTATIONAL PERIOD	10.8 ± 0.5 h	15.8 ± 1.0 h

TABLE 2-2

SATELLITE PARAMETERS

SATELLITE SYSTEM	ORBIT SEMI-MAJOR AXIS X 10^6 km	PR	ECCENTRICITY	INCLINATION	SIDEREAL PERIOD	MASS (10^{-4} mass earth)	RADIUS (km)
URANUS							
MIRANDA (V)	0.130	5.15	<0.01°	0°	1.414	0.15	~100
ARIEL (I)	0.191	7.50	0.0028°	0°	2.520	2.2	~300
UMBRIEL (II)	0.266	10.5	0.0035°	0°	4.144	0.9	~200
TITANIA (III)	0.436	17.2	0.0024°	0°	8.706	7.3	~500
OBERON (IV)	0.583	23.0	0.0007°	0°	13.463	4.2	~400
NEPTUNE							
TRITON (I)	0.356	15.2	0.0	159.945°	5.877	227	1885 ± 650
NEREID (II)	5.567	237	0.749	27.7°	359.88	-	~100

truly unusual aspect of the Uranian system is, however, the distribution of angular momentum, since the satellite orbits and Uranus' equator are at nearly right angles to the ecliptic.

Neptune's system is very irregular. Both Triton and Nereid are inclined to the equatorial plane, with Triton in a close, circular retrograde orbit and Nereid in a highly eccentric posigrade orbit, with a semi-major axis of 237 Neptune radii. Triton, the largest of the Uranus/Neptune satellites is massive enough to retain an atmosphere, but as yet none has been detected.

Nothing is known about the interaction of Uranus and Neptune with the solar wind, or if indeed the solar wind extends that far from the sun. Probably only after the "Grand Tour" flyby will anything be known of the particle and fields environment of these two planets.

2.2 Science Objectives, Measurements and Instruments

For this study the Uranus/Neptune orbiter science payload was chosen directly from the JPL Jupiter Orbiter science payload (JPL 1971) and modified where necessary. The Jupiter Orbiter Science payload consists of:

- Imaging TV (800 line Silicon Vidicon)
- Absolute Photometer-Radiometer
- IR Imaging Radiometer
- Infrared Spectrometer
- UV Spectrometer
- Magnetometer
- Radio Emission Receiver

- Plasma Wave Detector
- Trapped Radiation Detectors
- Ionosphere Skimmer Package

The payload chosen for the Uranus/Neptune orbiters is listed in Table 2-3. The Radio Emission Receiver and the Plasma Wave Detector, were designed to detect radio noise bursts in the 20 KHz to 100 MHz region and measure 1 Hz to 200 Hz magnetospheric noise, respectively, near Jupiter. These were deleted because such phenomena are not expected at Uranus and Neptune. The Ionospheric Skimmer Package was left out because of the potential hazards of the extremely low periapse (1.01 planet radii) orbits required by the package. The remainder of the science instruments, with the exception of the TV and the absolute photometer-radiometer, are essentially the same as those in the Jupiter orbiter package. A shorter focal length lens was added to the Silicon Vidicon TV optics to provide the option of viewing global or regional scale areas over a greater portion of the orbit than allowed by the 450 mm lens. The long focal length lens was retained to provide a high resolution capability. The absolute photometer-radiometer has an increased sensitivity and longer focal length optics than the one specified for the Jupiter orbiter to accommodate the lower reflected light levels and larger approach distances of the satellites of Uranus and Neptune.

Table 2-4 lists the science objectives for the exploration of Uranus and Neptune. Each of the four major objectives; determination of atmospheric characteristics, mapping planetary particles and fields, probing internal structure and processes, and investigations of the satellite systems, have been broken down into 39 specific objectives which may be fulfilled by a set of measurements. A much more detailed understanding of Uranus and Neptune than currently exists will be achievable once these 39 objectives have been satisfied.

TABLE 2-3

URANUS/NEPTUNE ORBITER SCIENCE PAYLOAD

	WEIGHT kg	POWER watts	DATA RATE
TV SYSTEM (Silicon Vidicon, 450 mm & 200 mm Optics)	28.2	28	262 kbps
ABSOLUTE PHOTOMETER-RADIOMETER	3.3	5	16 bps
IR IMAGING RADIOMETER	9.1	20	460 bps
INFRARED SPECTROMETER	9.1	12	200 bps
ULTRAVIOLET SPECTROMETER	5.5	10	640 bps
MAGNETOMETER (Flux Gate)	1.8	4	130 bps
TRAPPED RADIATION DETECTORS	10.0	14	1200 bps
TOTAL	67.0	93	

SCIENCE OBJECTIVES			IMAGERY (TV)	ABSOLUTE PHOTOMETER-RADIOMETER	IR IMAGING RADIOMETER	INFRARED SPECTROMETER	ULTRAVIOLET SPECTROMETER	MAGNETOMETER	TRAPPED RADIATION DETECTORS	SOLAR OCCULTATION	OCCULTATION AND CELESTIAL MECHANICS
			URANUS/NEPTUNE ORBITER EXPERIMENTS								
ATMOSPHERE	PARTICULATE CLOUD MATTER	DUST		X						X	
		DROPLETS		X		X				X	
		CRYSTALS		X		X				X	
	COMPOSITION	H_2/He			X		X				
		ISOTOPIC ABUNDANCES				X					
		TRACE CONSTITUENTS				X	X			X	
	GLOBAL CIRCULATION	CIRCULATION CYCLES	X		X						
		GLOBAL WIND VELOCITIES	X		X						
		BELT VELOCITIES	X		X						
		ANOMALOUS ACTIVITY	X		X						
	LOCAL PHENOMENA	CLOUD FORMATION	X		X						
		CYCLONE FORMATION	X		X						
		SPOTS	X		X						
		LIGHTNING									
	THERMODYNAMIC STATE	TEMPERATURE PROFILE				X	X			X	X
		DENSITY PROFILE				X	X			X	X
		PRESSURE PROFILE				X	X			X	X
		HUMIDITY PROFILE				X					
		THERMAL BALANCE			X						
		THERMAL ANOMALIES			X						
	CLOUDS	MORPHOLOGY OF CLOUDS	X		X						
		HOR./VERT. DISTRIBUTIONS	X		X					X	
PLANETARY PARTICLES AND FIELDS	FIELDS	MAGNETIC FIELD						X			
		GRAVITY POTENTIAL									X
	PARTICLES	RADIATION BELT SPECIES							X		
		PARTICLE DISTRIBUTION							X		
		PARTICLE ENERGY							X		
		MICROMETEORITES									
	SOLAR WIND	SOLAR WIND INTERACTION						X			
	PLANETARY RADIATION	EMITTED IR RADIATION			X	X					
		AIRGLOW					X				
		AURORA	X				X				
INTERNAL STRUCTURE AND PROCESSES	INTERNAL PROPERTIES	SURFACE EXISTANCE									
		PHYSICAL SURFACE STATE									
		SURFACE RADIUS									
	INTERNAL ACTIVE PROCESSES	SURFACE ROTATION PERIOD									
		HEAT FLUX			X						
		MAGNETIC FIELD						X			
SATELLITES		SATELLITE CHARACTERISTICS	X	X	X	X		X		X	X

URANUS/NEPTUNE SCIENCE OBJECTIVES AND EXPERIMENTS

TABLE 2-4.

The right side of Table 2-4 lists the instruments specified for the Uranus/Neptune orbiters and two specific experiments of which the orbiter is capable. The solar occultation experiment, observing the sun with a photometer through the planets atmosphere, provides data on a considerable number of atmospheric parameters. The occultation and celestial mechanics experiments provide data on the atmosphere and gravity field using the radio communication system and tracking, respectively. The capability of each instrument or experiment of fulfilling one of the 39 specific objectives is indicated by a small circle at the intersection points in Table 2-4. Thirty-three of the specific objectives will be satisfied to some degree using the chosen scientific payload. Of the six that are not fulfilled, four are surface related. These can probably only be achieved using a high energy radar system. Such a radar would not be compatable with the relatively low output power system of the orbiter. Lightning is not expected at Uranus and Neptune, but its detection would require the addition of a radio receiver system. Data on the micrometeorite environment at both Uranus and Neptune will be available from the Grand Tour missions. If a significant meteoroid population is indicated from this data a micrometeoroid detector may be added for little weight penalty.

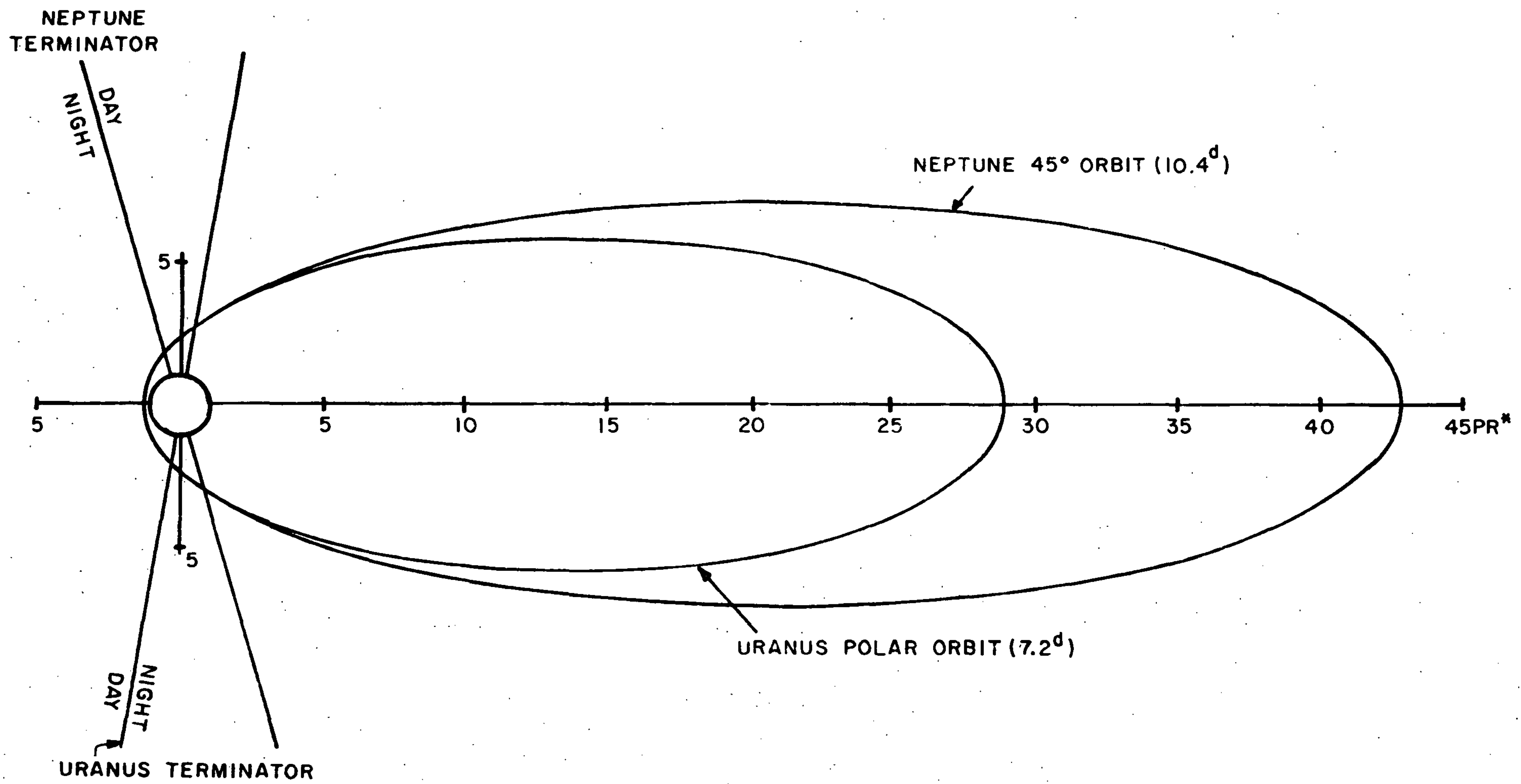
The visual imaging of Uranus and Neptune must emphasize coverage rather than resolution. The atmospheres of both planets are unlikely to contain any fine structure requiring high resolution imagery. The TV's main role will be to determine the large scale motions of each planet's atmosphere. This will be aided by the IR imaging radiometer which will map the thermal structure and detect upwelling material. These two instruments together should give a global picture of the weather at each planet.

Most of the science instruments may also be used extensively in studying the seven satellites in the Uranus/ Neptune systems if orbital conditions permit fairly close encounters. The absolute photometer-radiometer, IR imaging radiometer, and TV images will be particularly useful in providing data on satellite size, shape, and surface properties. It is of interest to compare the properties of these satellites with those of the satellites of Jupiter and Saturn (especially comparing Triton with the Galilean satellites), Pluto (from the Grand Tour flyby), and available information on the asteroids.

2.3 Candidate Missions

Two missions have been chosen to illustrate the Uranus and Neptune orbiters. A 7.2 day, polar orbit was chosen for Uranus and a 10.4 day, 45° inclination orbit for Neptune. The two orbits are shown, with the position of the terminator during the first passage, in Figure 2-2. The 1.2 x 29 Uranus radii and 1.2 x 43 Neptune radii orbits were chosen based on the retro propulsion capability of the TOPS orbiter and flight-time constraints, which will be discussed in Section 3. Sufficient scientific data will be collected using these orbits within a two year mission time frame. These two orbits are basically designed for planetology investigation. Not enough data exists at present to design a particle and fields exploration mission, but a highly elliptical orbit would be useful.

Figure 2-3 shows the variation in orbit altitude over the two orbits. Throughout 90 percent or more of each orbit period the spacecraft is ten planet radii or further from the visible atmosphere. This allows for TV and IR imagery of large portions of the planets area; essentially a constant view of the time dependent fluctuations and motions of the atmosphere. High



URANUS AND NEPTUNE ORBITS

FIGURE 2-2.

(PR = Planet Radii)

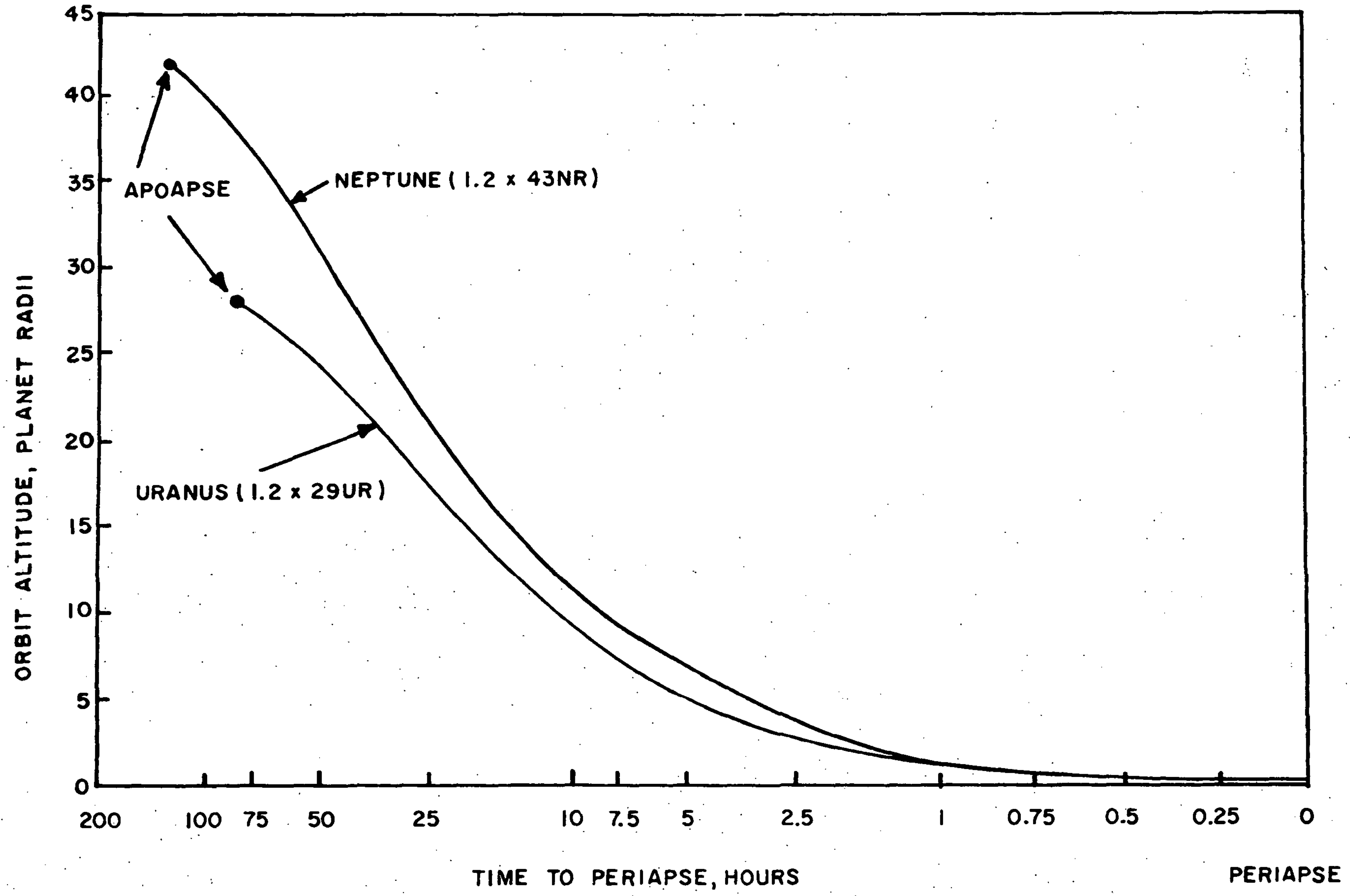


FIGURE 2-3. URANUS/NEPTUNE ORBIT ALTITUDE VARIATIONS

resolution (1 km or less) TV imagery is possible near periapse but, as was pointed out previously, such imagery is not as important as wide-angle coverage.

Figure 2-4 illustrates the Uranus orbiting spacecraft's ground track during its first orbit (1986 launch, 1995 arrival) from about 10 hours before periapse to about 14 hours after. The spacecraft passes over the lighted* pole shortly after periapse and heads into the darkened hemisphere. At apoapse the subspacecraft point is at 64° latitude, on the nightside. Figure 2-5 shows the illumination sequence during the first orbit. Each of the 21 views shows the planet with the sub-spacecraft point at dead center, the solar terminator, and a 2.3° circular field of view (corresponding to the 450 mm TV camera lens) superimposed. Starting at the top left, at apoapse, and moving to the right, the spacecraft passes over the dark pole between views 3 and 4 ($\sim -14^h$) and heads into the sun-light hemisphere. As it approaches periapse the 2.3° FOV becomes a point on the scale of the figure. During most of the orbit portions of the planet can be imaged, and atmospheric circulation plotted.

While Uranus can be studied satisfactorily from a polar orbit, its satellites cannot. In the 7.2^d candidate orbit the closest the spacecraft will come to any of the Uranian satellites is ~ 70,000 km when it passes Miranda at nearly a right angle. The angular sizes of the five satellites, at closest approach, will be between 2.5 and 7.5 minutes of arc, making for very poor study.

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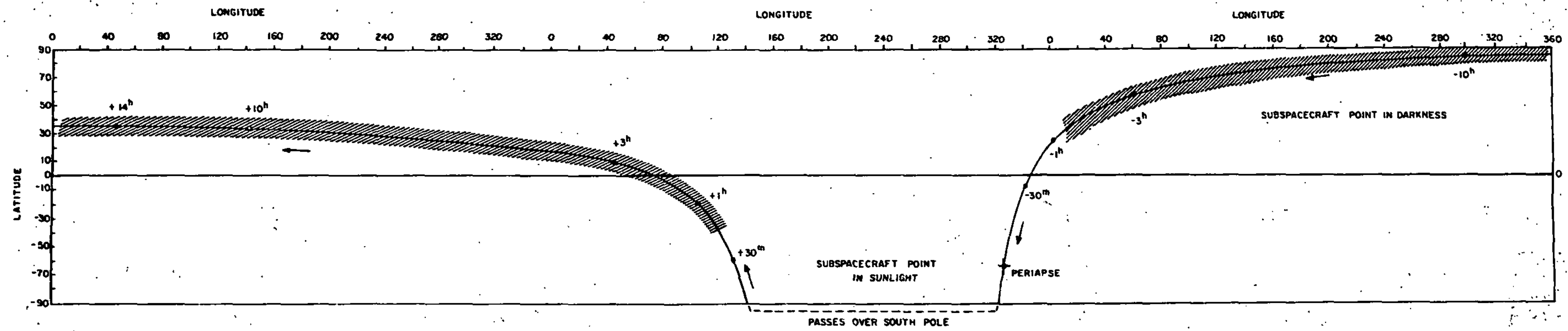
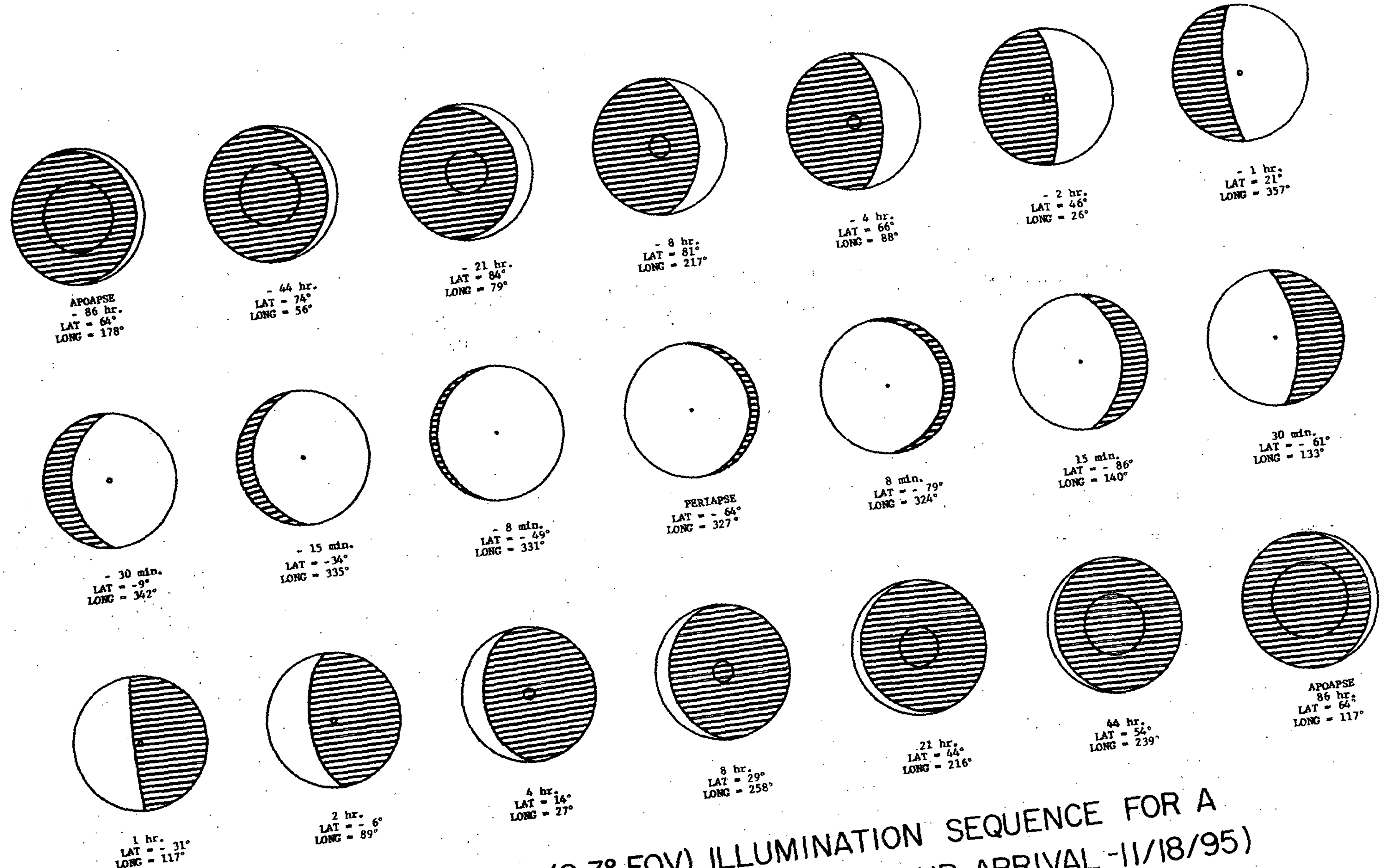
* There is some ambiguity in designating the North and South poles at Uranus. The computer program used to generate groundtrack data assigned South to the sun-pointing pole while Newburn (1971) designates this the North pole. For this study negative latitudes refer to the hemisphere with the lighted pole.



SPACECRAFT GROUND-TRACK NEAR PERIAPSE-POLAR URANUS ORBIT
(7.2^d, 1.2 x 29UR, LAUNCH-1/9/86, ARRIVAL-11/18/95)

FIGURE 2-4.



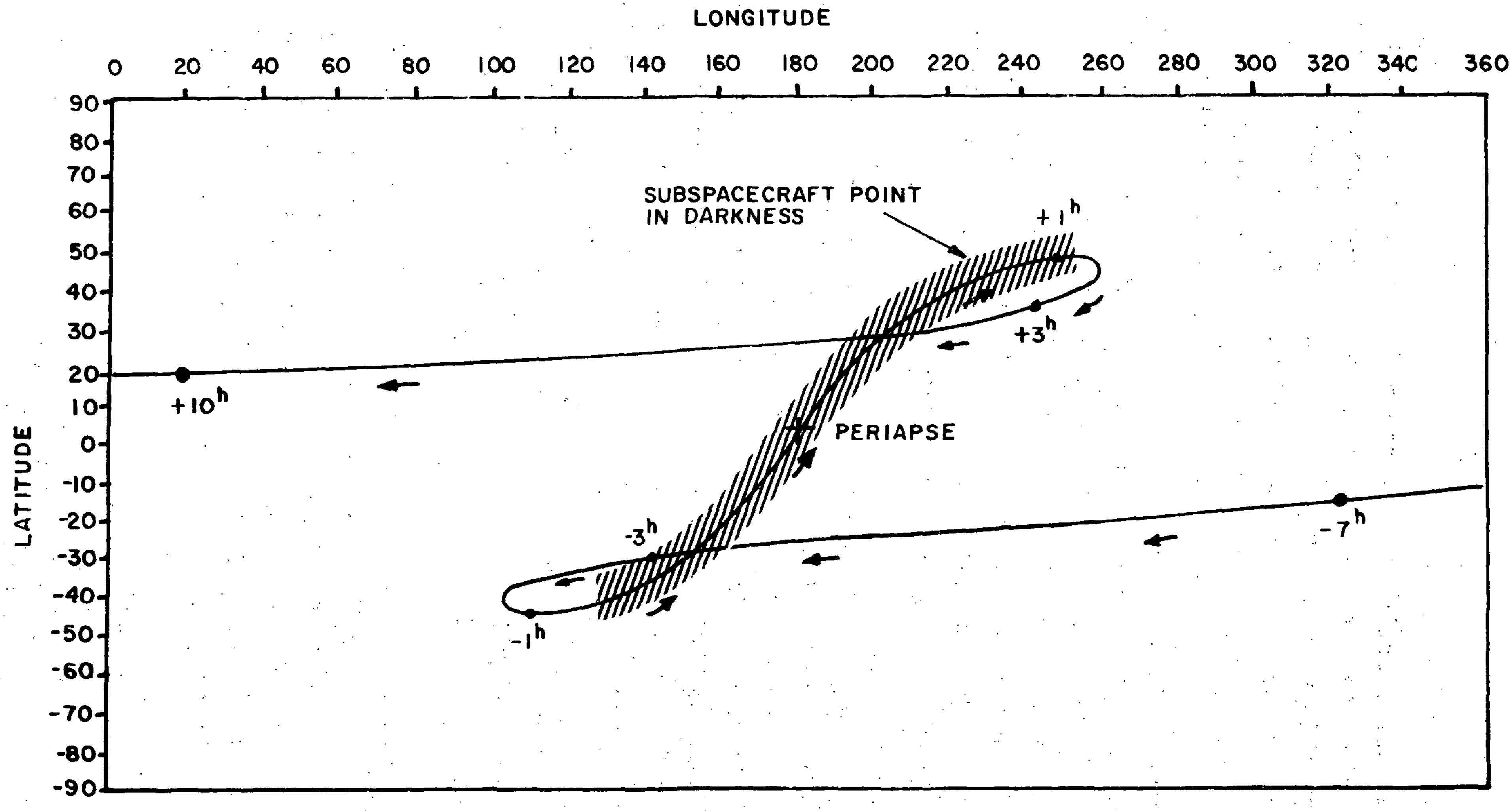
VISUAL IMAGER (2.3° FOV) ILLUMINATION SEQUENCE FOR A POLAR URANUS ORBIT (7.2^d, 1.2 x 29 UR, ARRIVAL -11/18/95)

FIGURE 2-5.

An equatorial orbit is preferable for Uranian satellite observations. Unless arrival conditions are timed carefully an equatorial orbit is very difficult to get into due to the 98° inclination of Uranus' pole to the ecliptic. Ideal arrival conditions exist when the equatorial plane is nearly edge on to the Sun and only a small spacecraft ΔV is necessary for plane change maneuvers. Such a condition occurs twice a Uranian year, or every 42 years. Unfortunately Uranus is pointing pole-on to the Sun in 1985 and conditions will have relaxed to the point where, at the candidate mission's arrival in 1995, the minimum orbit inclination possible is only 82°. Delaying the mission until the early 2000's or significantly increasing the spacecraft ΔV capability would alleviate this problem.

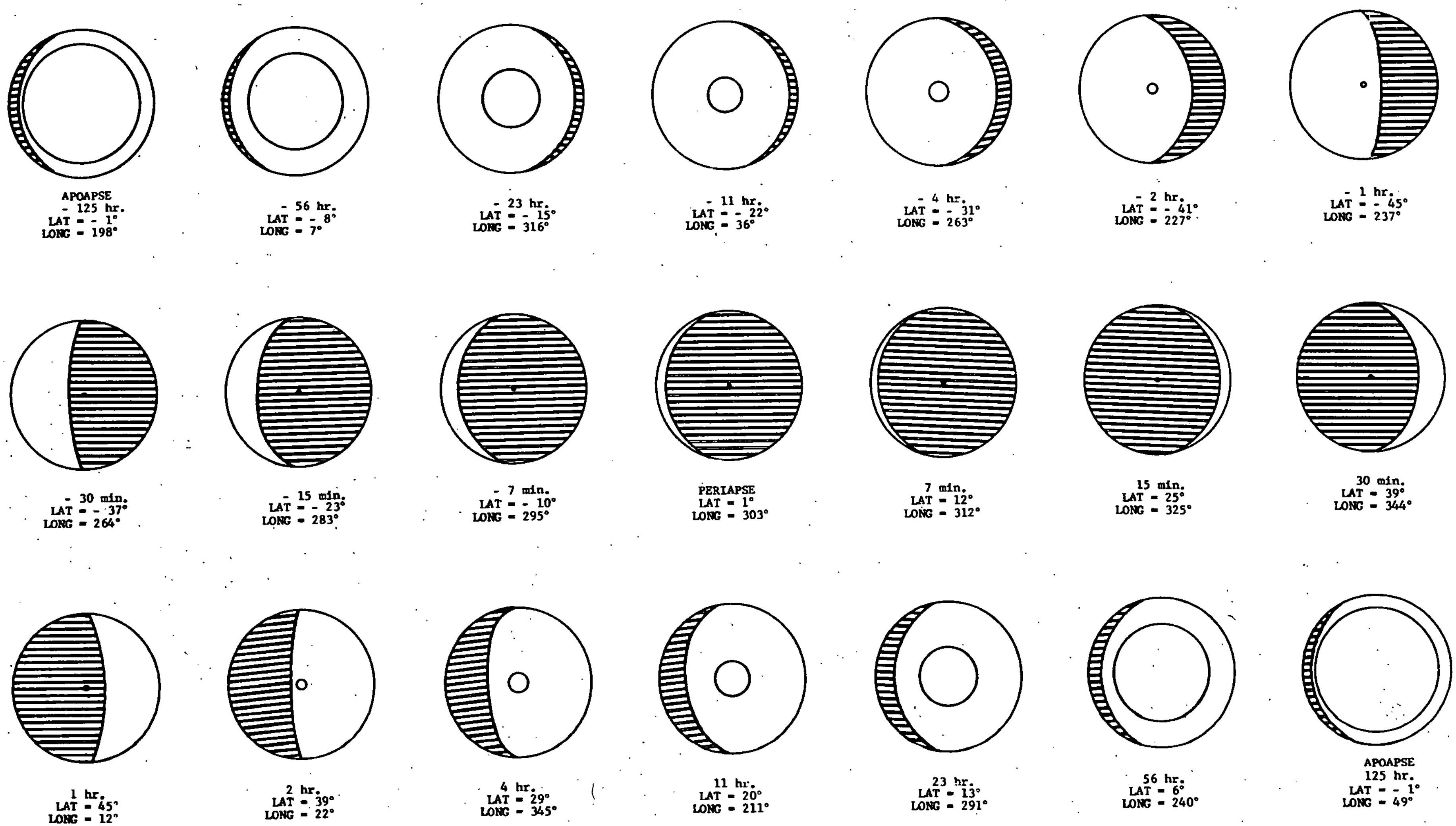
The spacecraft groundtrack for the 10.4 day, 45° inclination, Neptune orbiter is shown in Figure 2.6 from about 7 hours before periapse to 10 hours after. The $\mathscr{L}$- shaped curve near periapse is caused by the relatively high rotational velocity of the planet. Figure 2-7 shows the illumination sequence for this Neptune orbit, in exactly the same manner as Figure 2-5 did for Uranus. Notice, however, that the periapse is located on the dark side of Neptune. This provides a better opportunity for the Neptune orbiter to image a greater portion of the planet than the Uranus orbiter can, but with a loss of the high resolution imaging opportunities near periapse.

The candidate Neptune orbit does not optimize satellite exploration. Triton, the largest and most interesting satellite, can be observed from a retrograde, 21° inclination orbit. Such an orbit has a latitude coverage of Neptune's surface of only ± 21°, which decreases the planetology scientific return. An orbit constructed to observe Nereid suffers the same decreased latitude coverage.



SPACECRAFT GROUND TRACK NEAR PERIAPSE
45° NEPTUNE ORBIT (10.9° 1.2 x 43NR, ARRIVAL-1/20/01).

FIGURE 2-6.



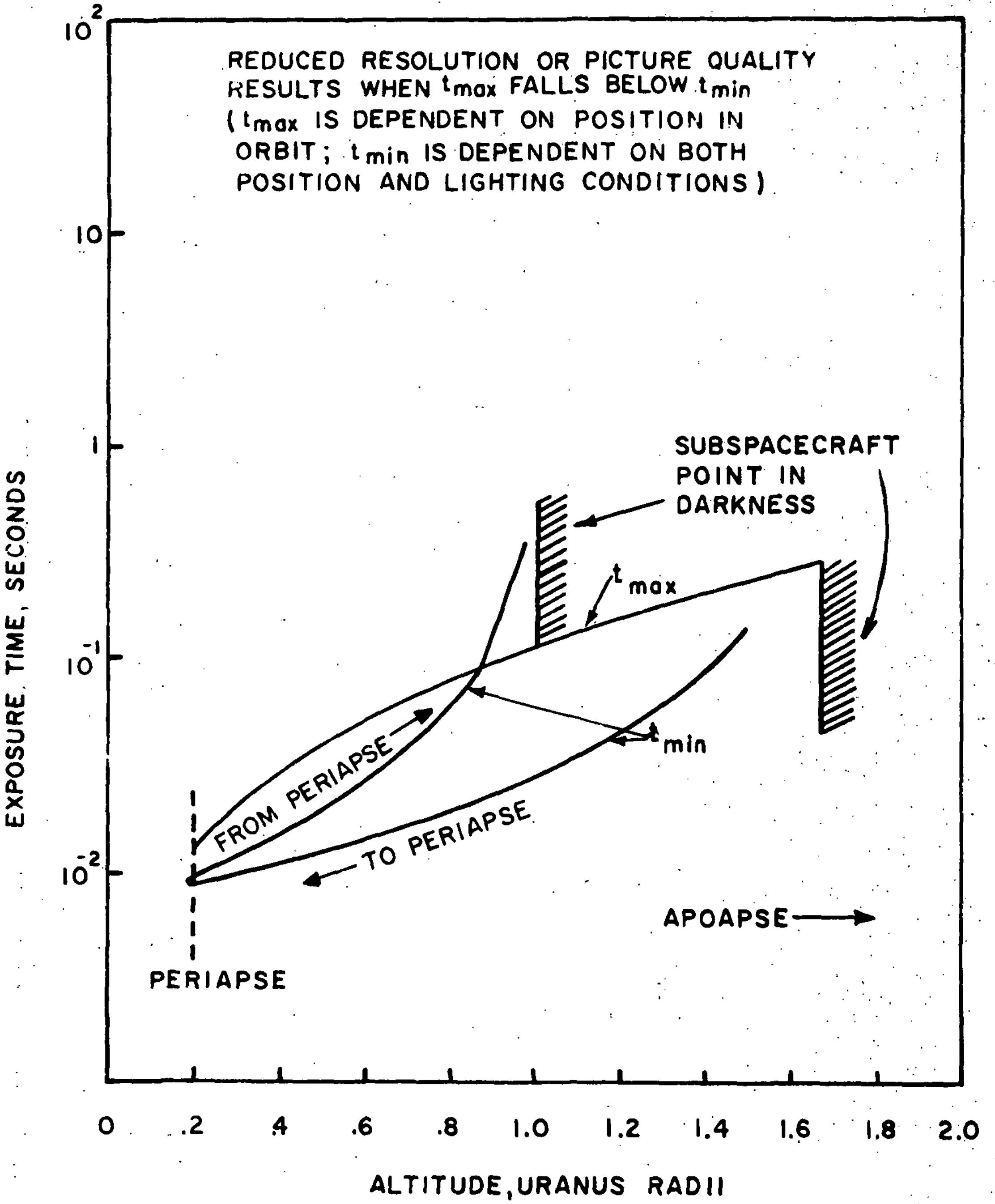
VISUAL IMAGER (2.3° FOV) ILLUMINATION SEQUENCE FOR A 45° NEPTUNE ORBIT (10.4^d, 1.2 x 43NR, ARRIVAL-1/20/01)

FIGURE 2-7.

2.4 Other Considerations

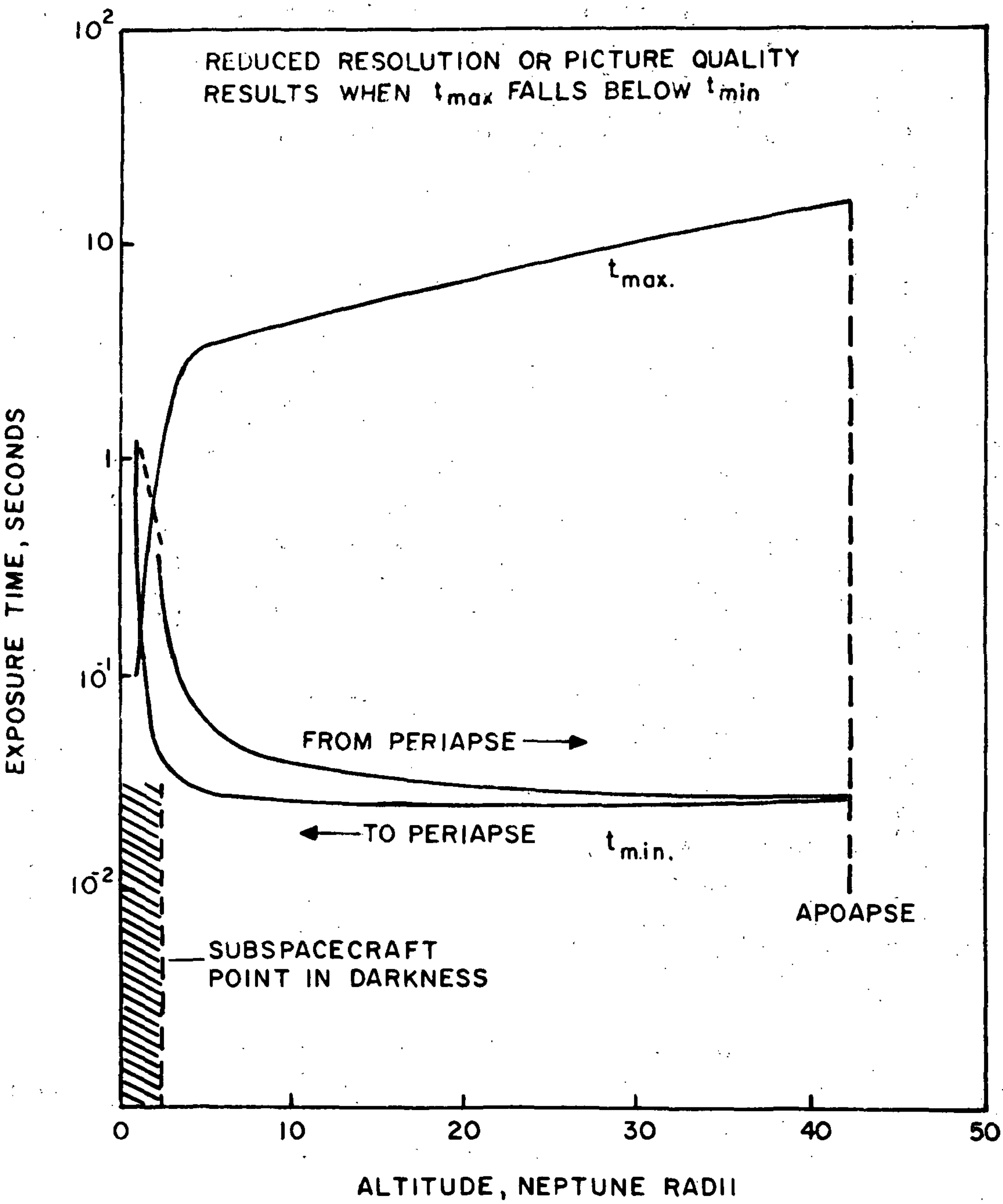
When the Jupiter orbiter payload is converted to a Uranus/Neptune orbiter payload each spacecraft subsystem must function adequately in the Uranus/Neptune environment. Fortunately the Jupiter orbiter spacecraft, based on the TOPS design, is easily convertible, with little or no weight or power penalty. Only the Silicon Vidicon TV and the communications and data storage systems may prove inadequate.

The main concern for the TV system is whether an exposure can be made without an intolerable amount of image smearing in the low light levels present at Uranus and Neptune. Using a maximum image smear of 1/2 resolution element on the 800 line Si-Vidicon tube, maximum permissible exposure times were computed for the Uranus and Neptune candidate orbits, for the portions where the subspacecraft point was in sunlight during the first orbital passage. These times are shown as the upper curves, labeled t_{max}, in Figures 2-8 and 2-9. Minimum permissible exposure times were computed using a minimum TV tube response of 3.5×10^{-2} ergs/cm^2 (signal to noise ratio of 10/1, an f4 lens, a 0.9 optical transmission factor, and a planet (sun elevation dependent) photometric function. These minimum exposure times are also shown on Figures 2-8 and 2-9, labeled as t_{min}. Wherever t_{min} exceeds t_{max} on these figures off-optimum TV pictures will result. Either a poorer signal to noise ratio will have to be accepted (t_{min} lowered) or a lower resolution tolerated (t_{max} increased or wider angle, lower f number lens used). For the Uranus orbiter, Figure 2-8, lower resolution can easily be tolerated at altitudes above 1 planet radii. It is consistent with the science objectives that resolution be sacrificed in favor of coverage. For Neptune, no problem arises becaused the critical exposure levels occur near periapse, on the dark side.



MAXIMUM AND MINIMUM TV TUBE EXPOSURE TIMES - 7.2d, POLAR URANUS ORBITER (1.2 x 29UR) (SILICON VIDICON TUBE, SIGNAL: NOISE = 10:1)

FIGURE 2-8.



MAXIMUM AND MINIMUM TV TUBE EXPOSURE TIMES-10.4^d, 45° NEPTUNE ORBITER (1.2 x 43NR)
(Silicon Vidicon Tube, Signal: Noise = 10:1)

FIGURE 2-9.

The TOPS Jupiter orbiter has a 9 foot diameter high gain, X or S band antenna with 10/20 watt X band and S band transmitters. This is capable of about 27 kilobits/sec from Jupiter under optimum conditions. The data storage system has a capacity of 1×10^9 bits. Each TV picture, the main data contributor, contains 5×10^6 bits, read out at 262 kbps. Correcting for the increased distances, the orbiter will be able to (X-band) transmit data at 1220 bps from Uranus and at 530 bps from Neptune. This will allow the orbiter to transmit one TV picture every 66 minutes from Uranus, and one every 157 minutes from Neptune, or using 12 hours per day receiving time, 78 pictures from Uranus and 47 from Neptune per orbit. Thus, since the remaining instruments require only a small fraction of the telemetry time, adequate coverage of Uranus and Neptune with all the science instruments is possible using the TOPS Jupiter orbiter systems.

3. SOLAR ELECTRIC TRAJECTORY ANALYSIS

This section will describe the SEP characteristics and payload capability for performing Uranus and Neptune orbiter missions. For a given launch vehicle, this capability is measured by the net mass delivered into orbit as a function of the interplanetary flight time. Net mass consists of the science payload and spacecraft support subsystems; it does not include either the SEP propulsion system or the chemical retro stage needed to insert the spacecraft into orbit. In fact, the SEP elements will be jettisoned well before planet approach since, at large distances, insufficient solar power is available for either propulsion or spacecraft operations. Thrust cutoff generally occurs between 3 and 5 AU for missions to Uranus or Neptune.

The basic data for the trajectory analysis is taken from previously published results (Horsewood and Mann 1969). Assuming the Titan IIID/Centaur launch vehicle, this data corresponds to a complete optimization* of SEP power and specific impulse, launch hyperbolic velocity, thrust direction, and propulsion time. Results for other candidate launch vehicles and orbit size parameters can be obtained by simple scaling relationships (Friedlander 1970). Horsewood and Mann's trajectory data assume that the planets revolve about the Sun in circular orbits lying in the ecliptic plane. This is a good approximation for Uranus and Neptune missions; trajectory requirements vary only slightly between successive launch opportunities which occur about every 12 months. Since optimum values of power, specific impulse and propulsion time are often impractical from a design standpoint, it was necessary to generate additional trajectory data to show the effect of off-optimum design. These results are based on the actual planet ephemerides and are obtained for a typical launch opportunity (1985-86).

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* Optimum performance is defined here as maximum net mass.

A point of trajectory design philosophy could be made at this time. Two types of SEP interplanetary transfers may be identified as "flyby" and "orbiter". In the case of a flyby trajectory, the hyperbolic excess velocity (V_{HP}) at planet approach is optimized to yield the maximum net mass at planet encounter. For the same time of flight, the orbiter trajectory would optimize V_{HP} (at a value lower than the flyby case) such that the maximum net mass in orbit is obtained for a specified orbit size. Generally, the flyby solution would provide a smaller net mass in the same orbit. However, for outer planet orbiters, the difference between the two solutions is practically insignificant for all typical orbits. The gain of "several kilograms" for the orbiter solution is obtained at the expense of much longer propulsion times in order to provide but a small reduction in approach velocity. Examination of the propulsion time history for orbiter trajectories shows that the thrusters are operating at large solar distances (even beyond 10 AU) where only very small increments of thrust acceleration can be attained. Such a solution is both impractical and unnecessary. In contrast, the flyby propulsion history is quite regular with thrust cutoff occurring after several hundred days (relative to flight times of several thousand days). Therefore, we propose that only flyby trajectories should be computed for SEP missions to the outer planets - even for purposes of preliminary analysis. All of the trajectory data presented in this report are obtained from optimum flyby solutions.

The nominal system parameters (launch vehicles, SEP stage and retro stage) assumed in the analysis will be described first. Trajectory characteristics of the Uranus and Neptune orbiter missions will be discussed separately. A final item of consideration is a comparison of SEP and ballistic flight modes, both direct and via a Jupiter swingby.

3.1 Nominal System Parameters

Initial spacecraft mass at Earth departure is equivalent to the injected mass of the launch vehicle. The three launch vehicles considered in this study are the Titan IIID/Centaur*, the Titan IIID(7)/Centaur*, and the Shuttle/Centaur.** Only the Titan IIID/Centaur is an actual programmed vehicle and, hence, will be taken as our baseline choice provided that sufficient net mass capability is available. The proposed Shuttle/Centaur vehicle may be a logical choice for missions in the post-1980 time period. Figure 3-1 shows the maximum injected mass of these three vehicles as a function of hyperbolic launch velocity (V_{HL}).

SEP system parameters used in this study are representative of current technology and design goals. Baseline values are a propulsion system specific mass of 30 kg/kw, a thruster specific impulse of 3000 seconds, and a propellant tankage factor of 3 percent. Overall propulsion efficiency at I_{sp} = 3000 sec is 62 percent. Although current design trends are for a SEP powerplant in the range 10-20 kw, it will be shown subsequently that optimum power requirements are much higher for the outer planet missions.

The chemical retro needed for orbit capture is assumed to be a space-storable liquid propulsion system (e.g., Fluorine/Hydrazine) having a specific impulse of 383 seconds and an inert fraction equal to 25 percent of the propellant loading. This type of retro stage has been proposed for the TOPS Jupiter orbiter application. Figure 3-2 describes the orbit insertion mass

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* Performance data provided by JPL, from Martin Marietta Manual M-70-7.

** Performance data taken from NASA Launch Vehicle Estimating Factors Handbook.

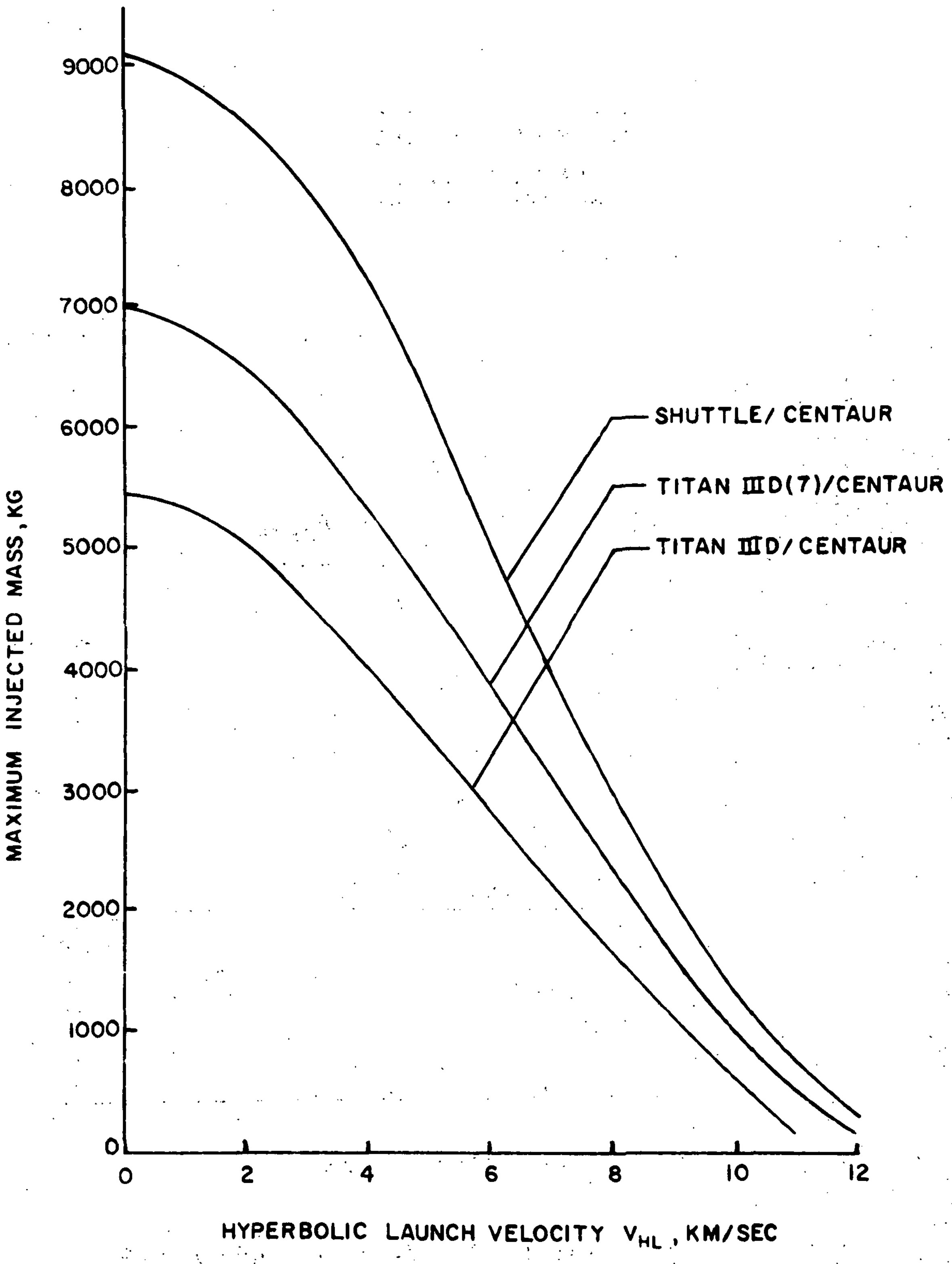

FIGURE 3-1. LAUNCH VEHICLE PERFORMANCE CURVES.

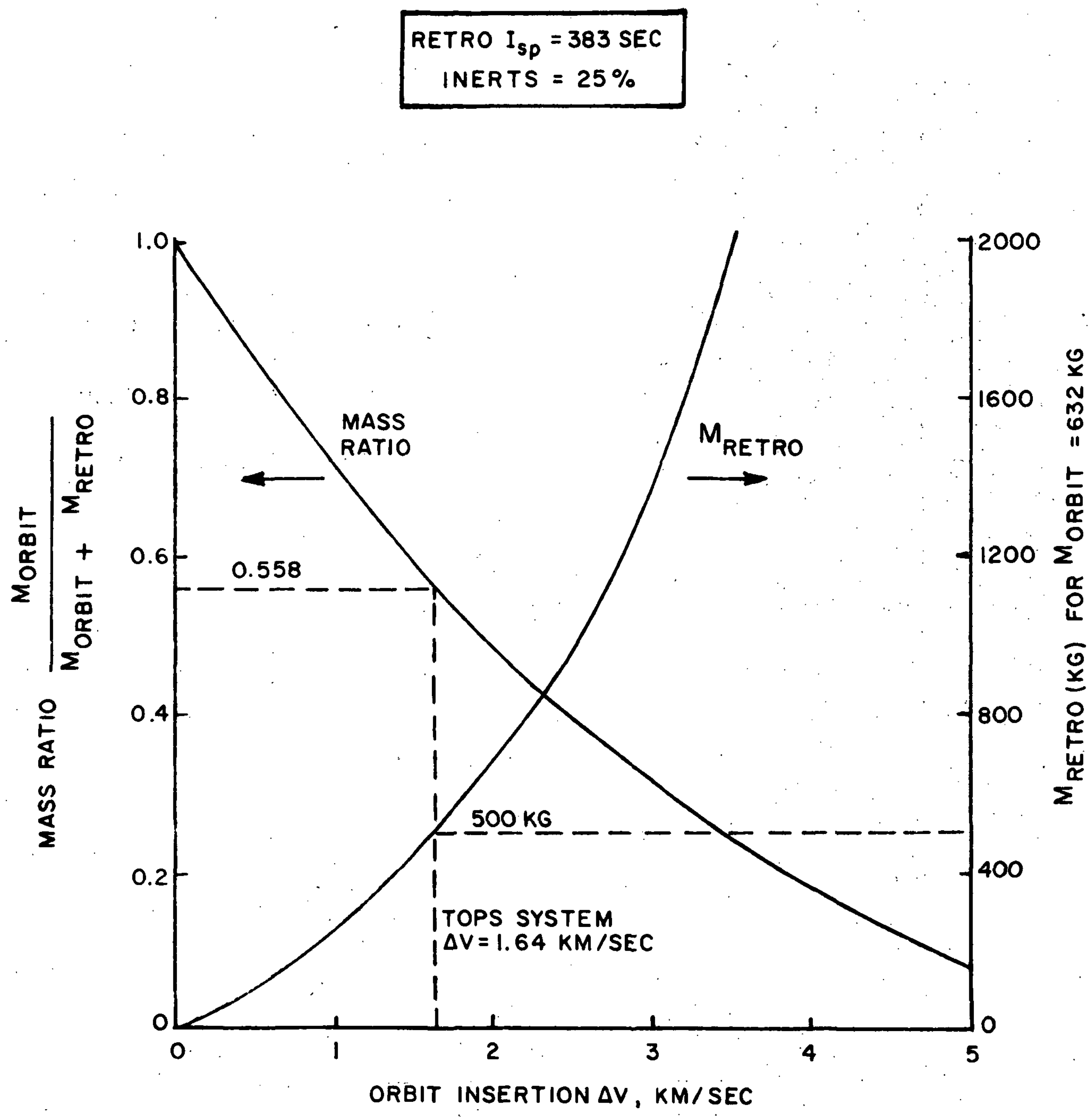

FIGURE 3-2. ORBIT INSERTION MASS REQUIREMENTS.

requirements as a generalized orbiter mass ratio as a function of ΔV, and also as the retro stage mass needed for the TOPS orbiter (632 kg). It is recalled that the proposed TOPS retro system provides a ΔV of 1.64 km/sec and weighs 500 kg. The TOPS system will be used as a reference point in the analysis, but we will not restrict ourselves to this particular design.

3.2 Uranus Orbiter Missions

Figure 3-3(a) shows the orbit insertion ΔV requirements as a function of Uranus approach velocity and orbit apoapse (maximum) distance. The orbit periapse (minimum) distance is 1.2 Uranus radii which is about 5000 km altitude above the visible "surface". This relatively low periapse will be taken as our baseline value; it should be achievable from a guidance standpoint, and is desirable in terms of both science and mass-in-orbit capability. Note that the ΔV difference between an apoapse distance of 100 and 50 radii is rather small, but that tighter orbits ($R_A < 20$) require increasingly greater velocity increments. An important way to interpret this figure is to examine the limitation imposed on approach velocity by a given ΔV capability. It is seen that flight time and V_{HP} are strongly related and that a low value of V_{HP} implies a long flight time. Assuming that the apoapse distance should be no greater than 100 planet radii, the maximum V_{HP} is 7.9 km/sec if the TOPS retro system is utilized. The minimum flight time is then about 3100 days. If, instead, the retro ΔV capability were increased to 2.5 km/sec, the maximum V_{HP} and minimum flight time are, respectively, 10 km/sec and 2650 days. Figure 3-3(b) shows similar parametric data for a periapse distance of 2.0 Uranus radii. Note that a significantly larger ΔV is now required for given values of R_A and V_{HP}. Equivalently, flight time would have to increase by nearly one year compared to $R_p = 1.2$ if the TOPS retro system were employed.

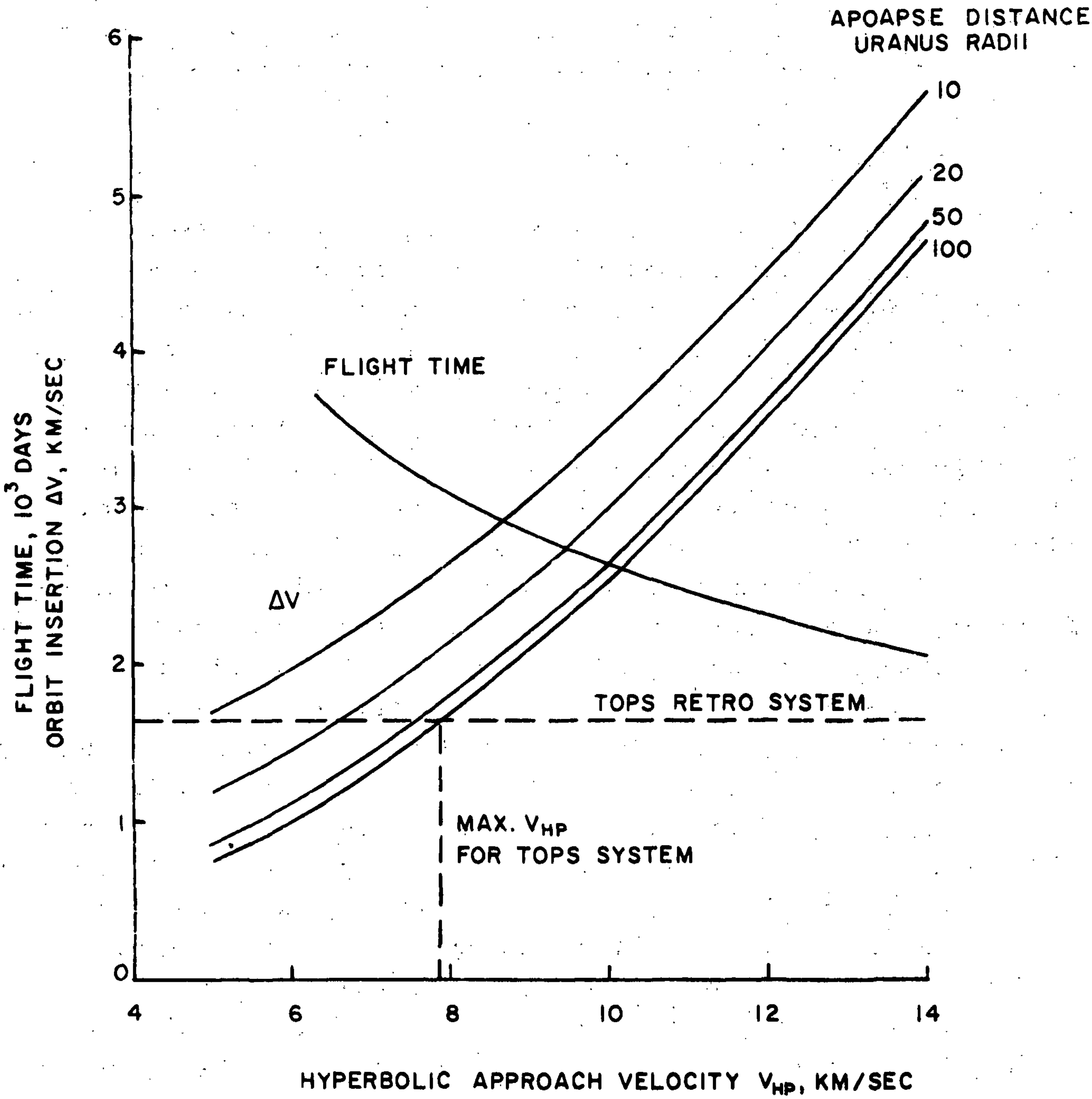

FIGURE 3-3. URANUS ORBIT INSERTION CHARACTERISTICS
a) PERIAPSE DISTANCE = 1.2 URANUS RADII.

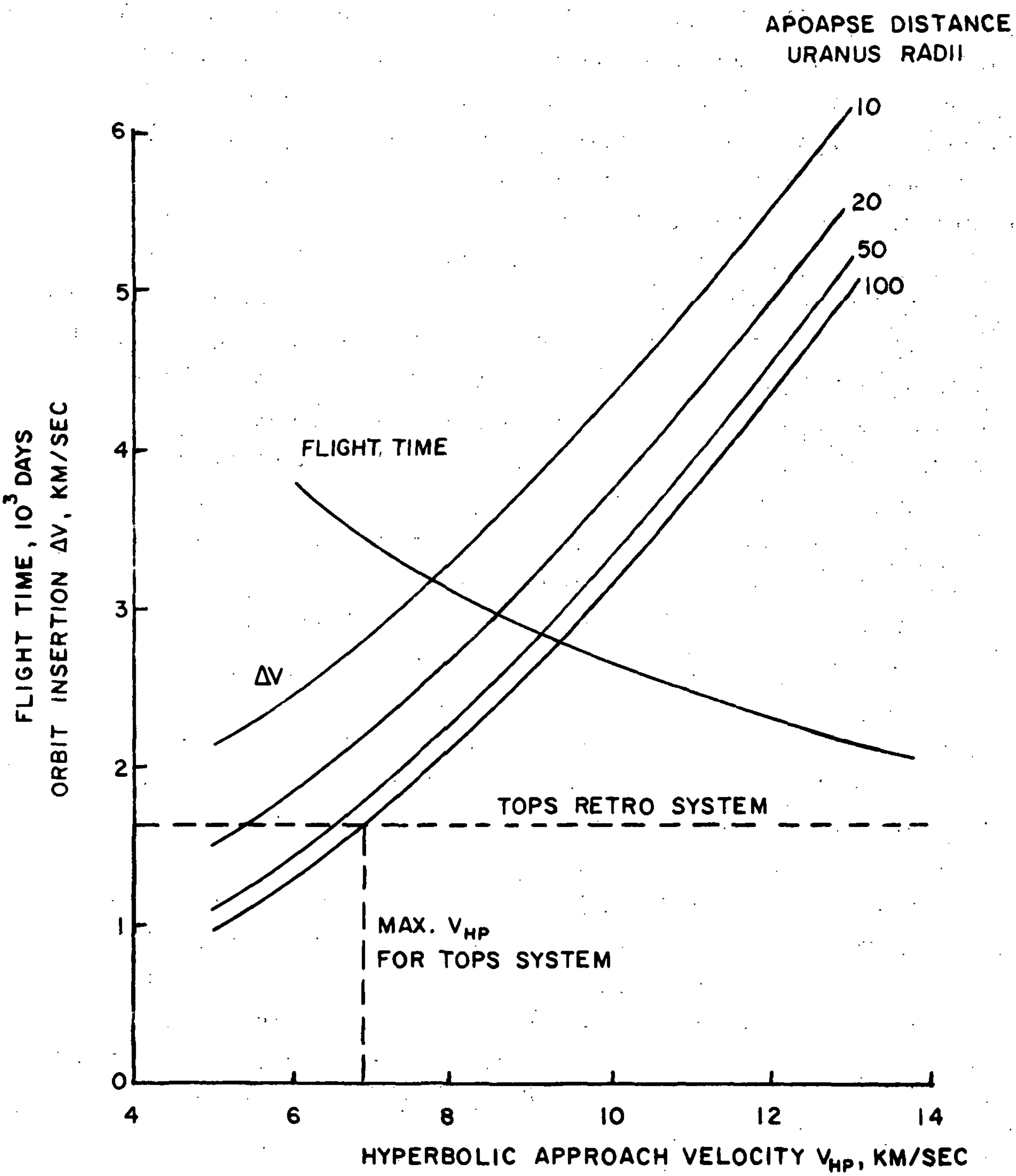

FIGURE 3-3. URANUS ORBIT INSERTION CHARACTERISTICS
b.) PERIAPSE DISTANCE = 2 URANUS RADII.

Figures 3-4 and 3-5 show the variation of optimum values of SEP power, specific impulse, and hyperbolic launch and approach velocities with flight time to Uranus over the range 2000-3600 days. The direct flight mode is assumed (heliocentric travel angle less than 360°). Although the specific optimum data shown is for the Titan IIID/Centaur launch vehicle, values of I_{sp}, V_{HL} and V_{HP} are near-optimum for the other launch vehicles considered here. Optimum power would of course scale proportionately with the injected mass capability of these other vehicles. A general characteristic of direct mode transfers is that optimum values of power and specific impulse both increase with longer flight times. Correspondingly, the launch and approach velocities decrease with flight time in a manner similar to ballistic trajectories. Note that, over the range of flight times, I_{sp} varies between 2600 and 2800 seconds, and P_o for the Titan IIID/Centaur varies between 29 and 35 kw. Current electron bombardment thruster technology has been concentrated in the I_{sp} region 2700-4000 seconds. Our baseline choice of 3000 seconds should incur but a small net mass penalty.

Figure 3-6 compares the net mass/flight time performance of three launch vehicles and orbit apoapse distance in the range 20-100 Uranus radii. Optimum system parameters are assumed; the SEP power range is indicated for each vehicle. If a small Pioneer-type orbiter (e.g., 300 kg) is acceptable, the flight time requirement is less than 2600 days (7 years) using the Titan IIID/Centaur. Note, however, that the orbit insertion ΔV is about 3.1 km/sec and the retro stage mass is 700 kg - this is greater than the nominal TOPS retro system capability. Moving up to the TOPS orbiter (632 kg) effects an increase in flight time to the range 3140-3600 days (8.5-10 years) with utilization of the Titan IIID/Centaur. In this case the TOPS retro system is adequate to attain an orbit apoapse distance as low as 20 Uranus radii. The more energetic launch vehicles are seen to provide performance advantages in terms of net mass

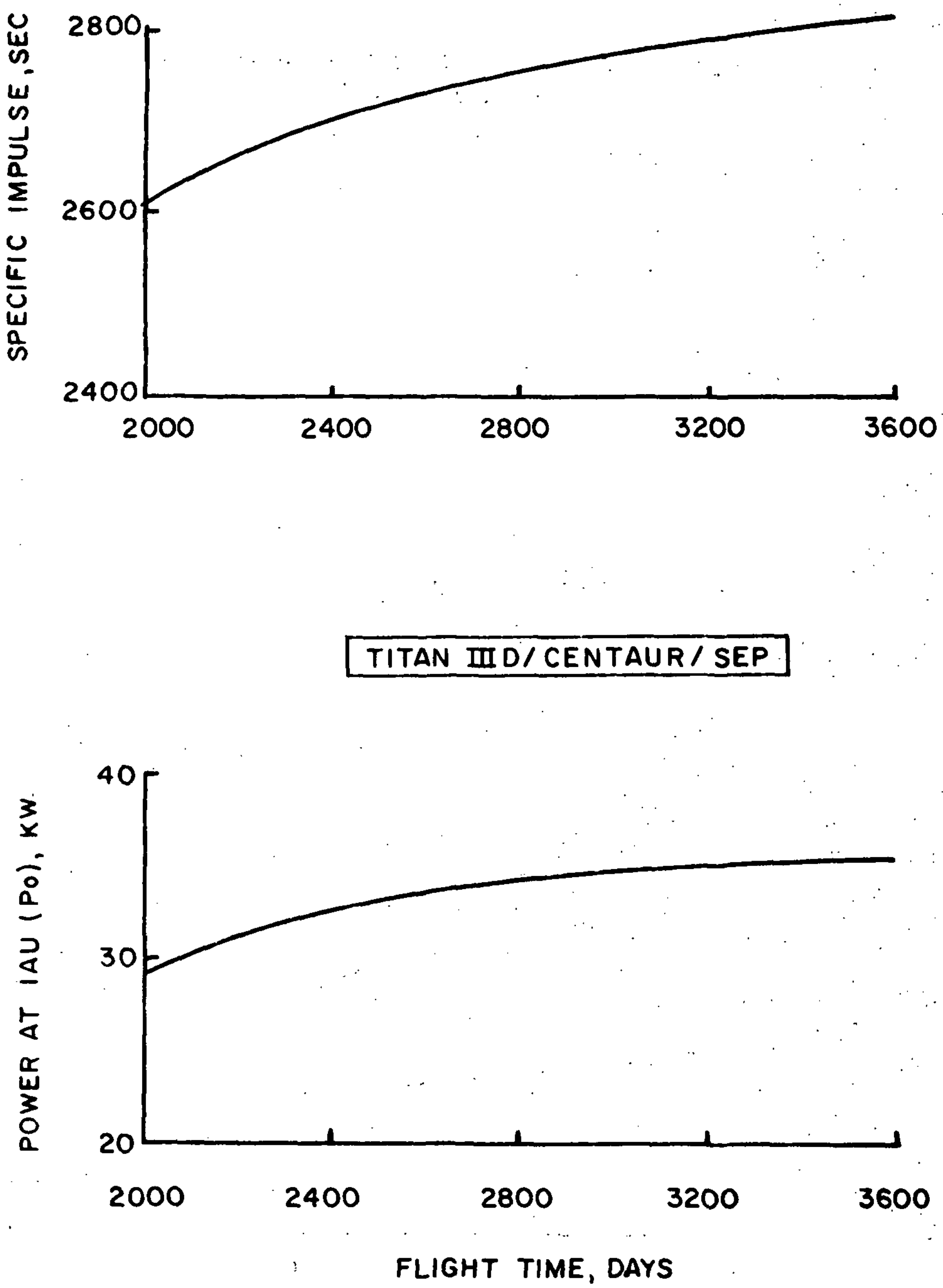

FIGURE 3-4. OPTIMUM POWER RATING AND SPECIFIC IMPULSE FOR SOLAR ELECTRIC URANUS MISSIONS.

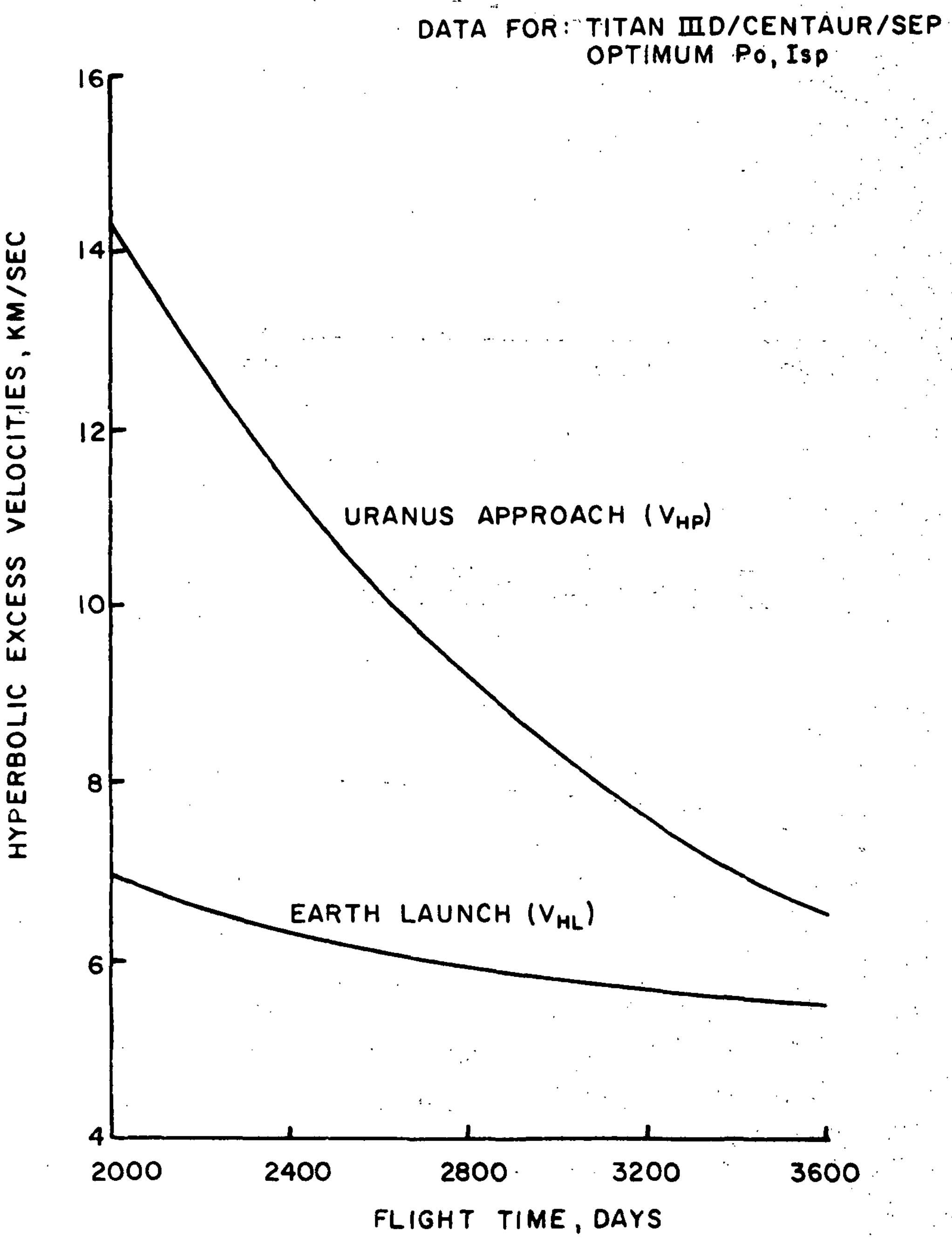

FIGURE 3-5. OPTIMUM LAUNCH AND APPROACH VELOCITIES FOR SOLAR ELECTRIC URANUS MISSIONS.

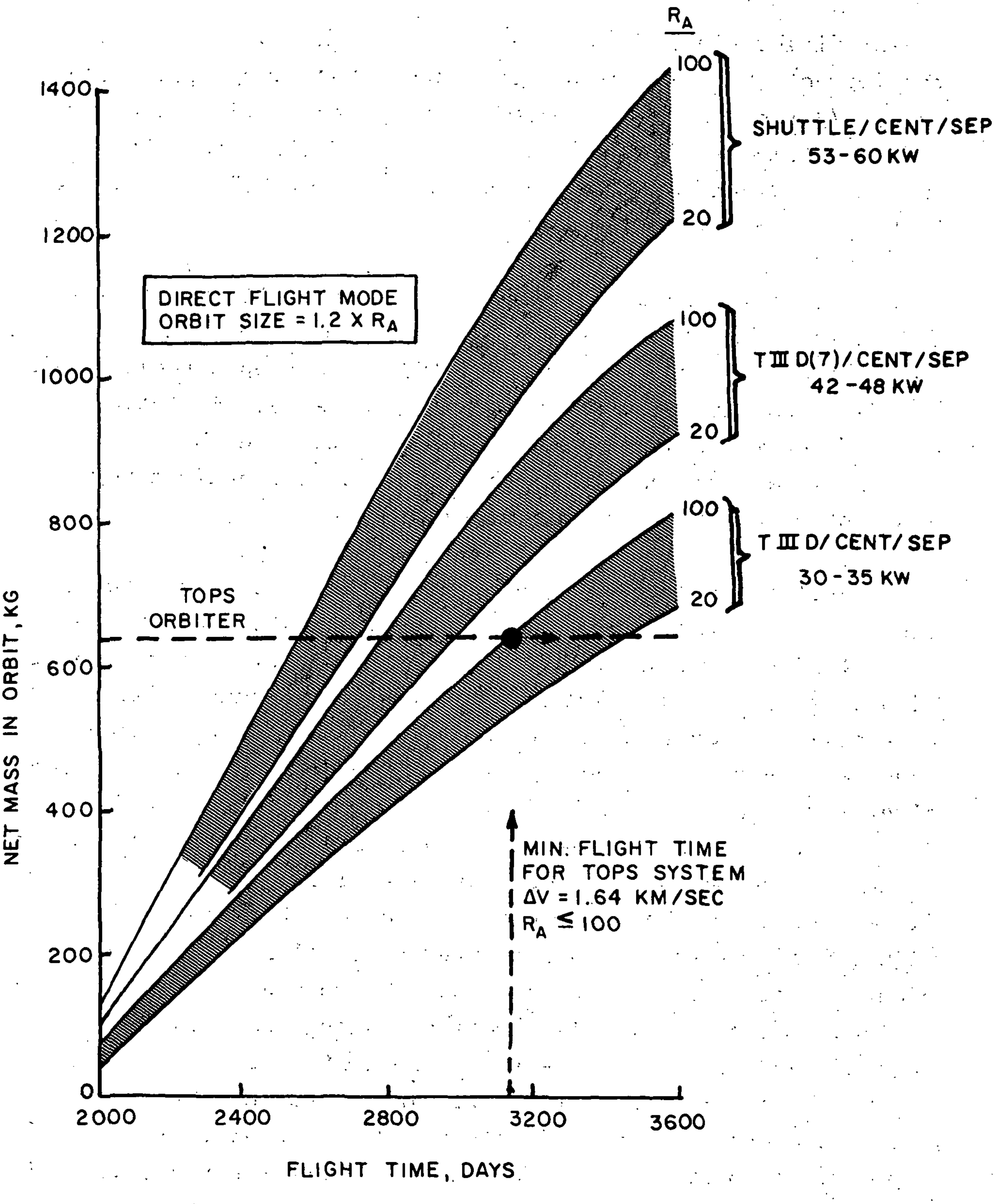

FIGURE 3-6. SOLAR ELECTRIC PROPULSION CAPABILITY FOR URANUS ORBITER MISSIONS, AVERAGE LAUNCH OPPORTUNITY.

and/or flight time. Note, however, that the optimum power requirements become exceedingly large. For example, the Shuttle/Centaur/SEP can deliver the TOPS orbiter to a 1.2 x 20 orbit in a flight time of 2720 days (7.5 years). The optimum SEP power, insertion ΔV and retro weight would be 60 kw, 2.75 km/sec and 1160 kg, respectively.

We will take 3600 days as our upper limit on acceptable flight time for the Uranus orbiter mission. Figure 3-7 illustrates the heliocentric trajectory for a 1986 launch opportunity with Uranus encounter occurring in 1995. Typically, SEP thrust cut-off occurs at about 4 AU, 400 days after launch. The following paragraphs discuss the effect of off-optimum power and propulsion time for the 10 year mission.

Net spacecraft mass in orbit is shown as a function of SEP power in Figure 3-8. The solid-line curves are for the Titan IIID/Centaur and assume the conditions of optimum propulsion time, I_{sp} = 3000 seconds, and R_P = 1.2 Uranus radii. Hyperbolic launch velocity is indicated over the power range considered - these values apply to each of the R_A curves. Consider first the limiting orbit size of 1.2 x 100. The TOPS spacecraft (632 kg) can be inserted into this orbit for a power rating as low as 10 kw. Net mass capability at 10 kw is about 21 percent lower than the optimum power (30 kw) capability. TOPS can be inserted into a tighter orbit (1.2 x 20) if the power rating is increased to 18 kw. It should be restated that the chemical retro size is a variable along these curves. However, as a point of reference, the TOPS retro system capability is indicated for each of the three launch vehicles where the 632 kg spacecraft is placed into a 1.2 x 19 orbit. The minimum power requirement is 18 kw for the Titan IIID/Centaur, 10 kw for the Titan IIID(7)/Centaur, and 7.5 kw for the Shuttle/Centaur.

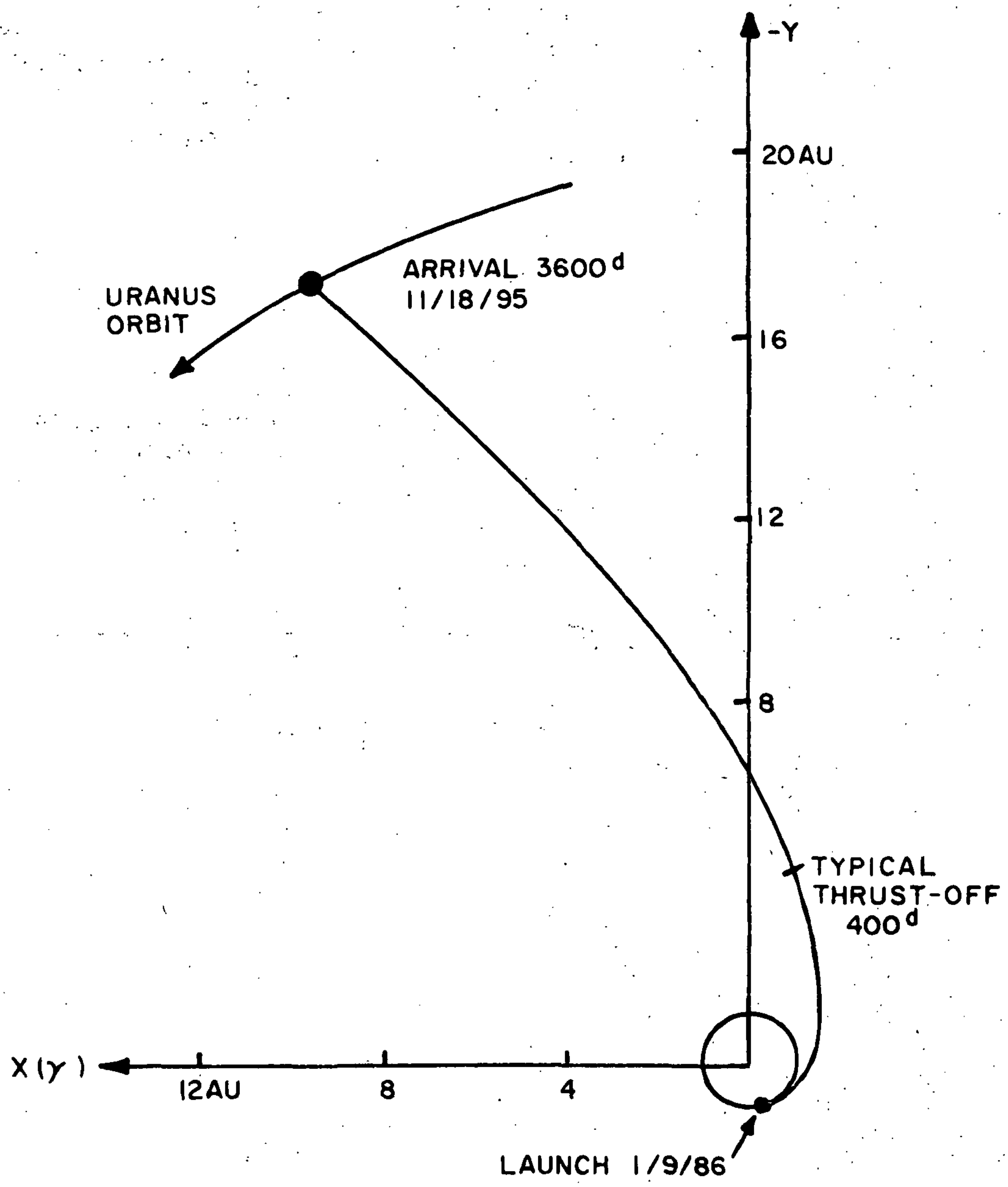

FIGURE 3-7. SOLAR ELECTRIC TRAJECTORY TO URANUS.

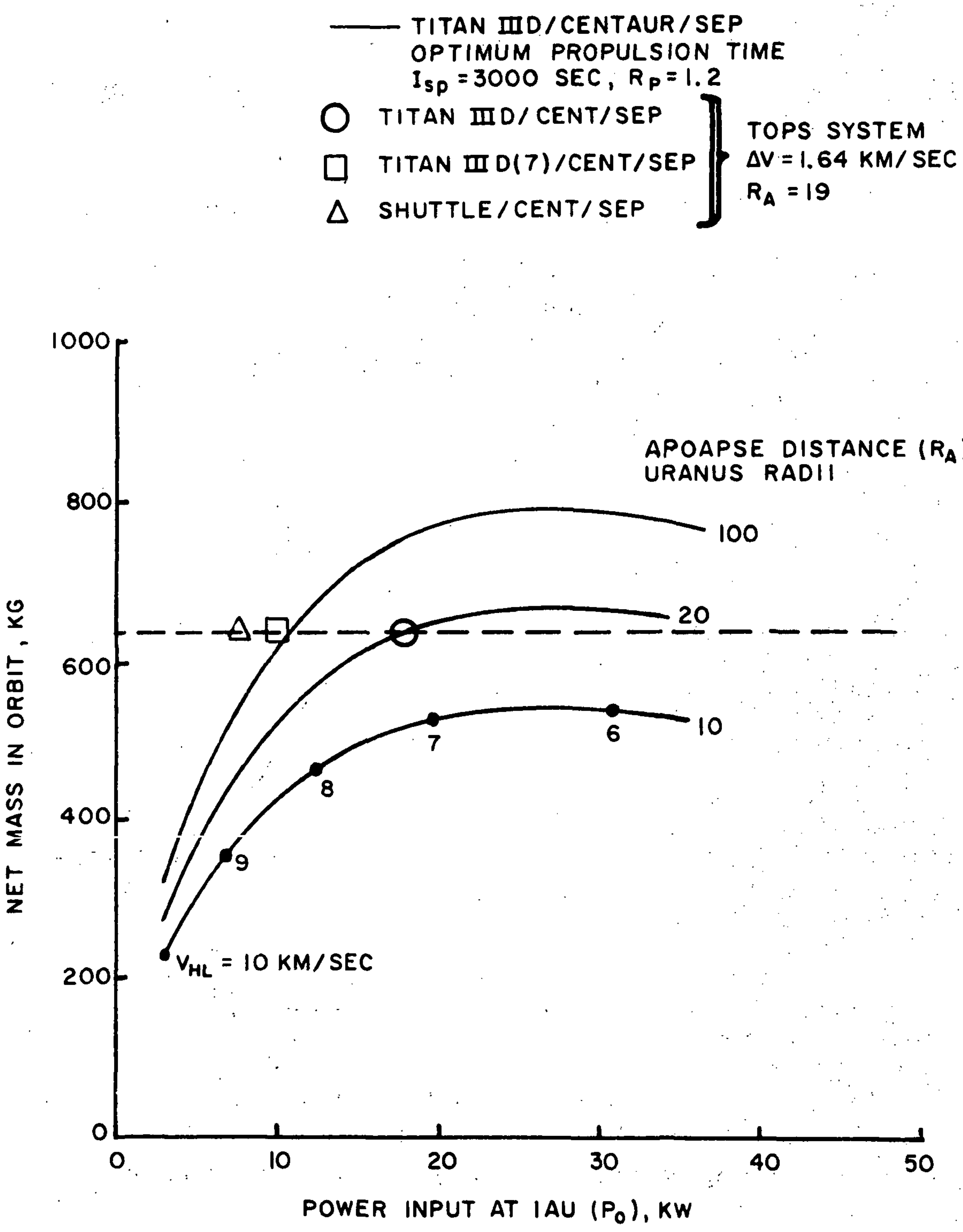

FIGURE 3-8. NET MASS IN ORBIT VERSUS SEP POWER FOR 3600-DAY URANUS ORBITER MISSION, LAUNCH 1/9/86.

We may conclude that practical sized SEP powerplants can be used for the Uranus mission provided that a flight time of about 10 years is acceptable.

Optimum propulsion on-time for the Uranus mission is in excess of 600 days. The next step in off-optimum design analysis is to examine how a desirable reduction in operating time will affect mission capability. Figure 3-9(a) shows curves of net mass versus propulsion time for several values of P_o. The Titan IIID/Centaur and the TOPS retro system are assumed for this illustration. In terms of a general characteristic it is seen that a relatively small net mass penalty is incurred for a large reduction in propulsion time. For the specific example of the TOPS net mass, propulsion time is about 600 days for a 20 kw powerplant, and only 270 days if the power is increased to 30 kw. The broken-line curve shows that a 15 kw powerplant is adequate for the TOPS orbiter if the orbit size is increased and a smaller retro than the TOPS system is utilized. An orbit selection of 1.2 x 50 requires an insertion ΔV of 1.28 km/sec and a retro stage weight of 360 kg. Propulsion time is reduced to 350 days. Figure 3-9(b) presents similar data for the Titan IIID(7)/Centaur application. In this case a 15 kw powerplant operating for only 200 days would allow the TOPS orbiter to be inserted into a 1.2 x 19 orbit.

3.3 Neptune Orbiter Missions

Only direct mode trajectories were examined for the Uranus mission. In the case of Neptune, earlier published results have apparently indicated that indirect mode trajectories offer significant performance advantages over the direct mode; i.e., larger net mass for the same flight time. By definition, indirect trajectories have heliocentric travel angles greater than 360°.

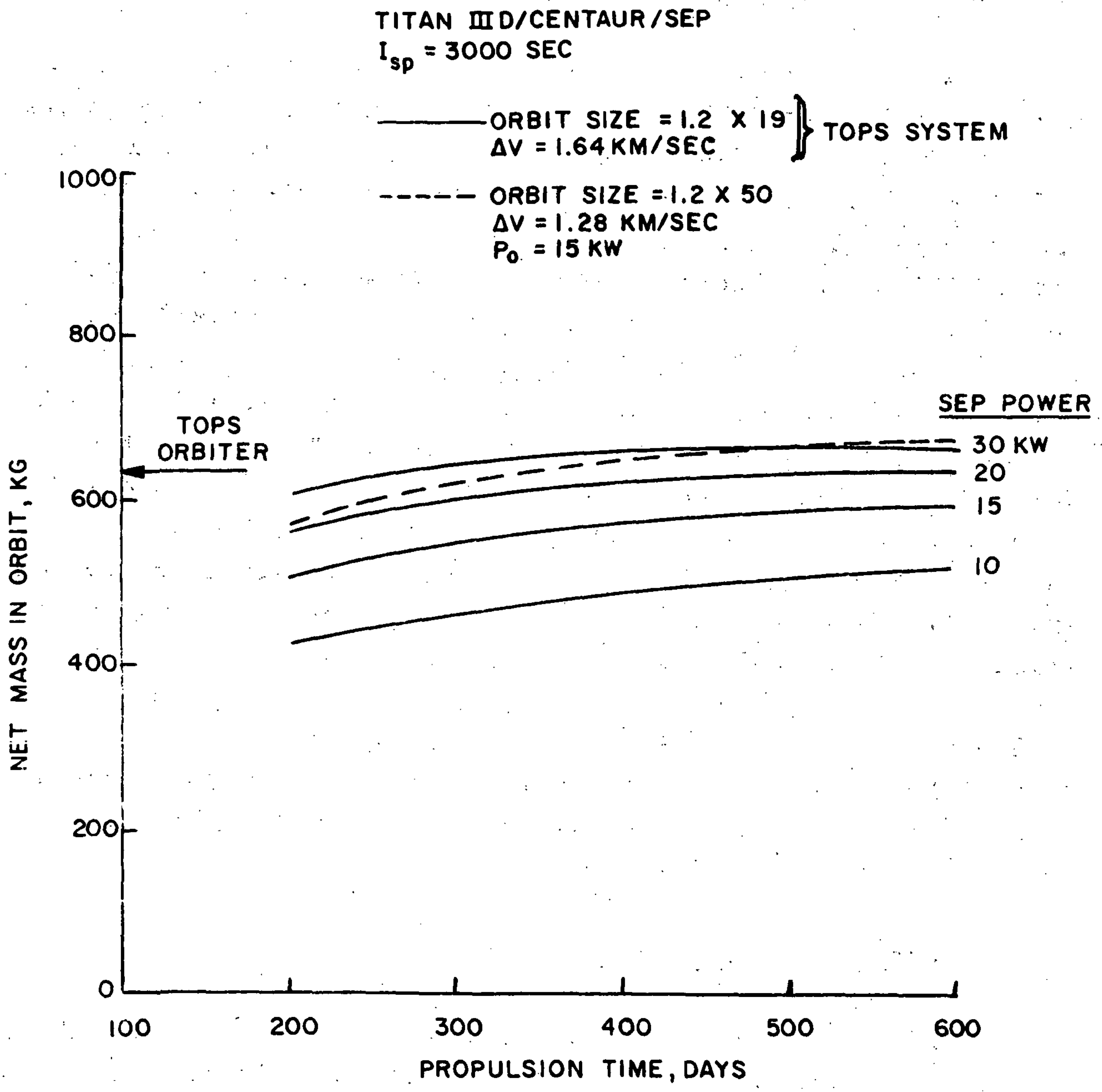

FIGURE 3-9. EFFECT OF REDUCING PROPULSION TIME FOR 3600-DAY URANUS ORBITER MISSION. (a.) TITAN III D CENTAUR APPLICATION.

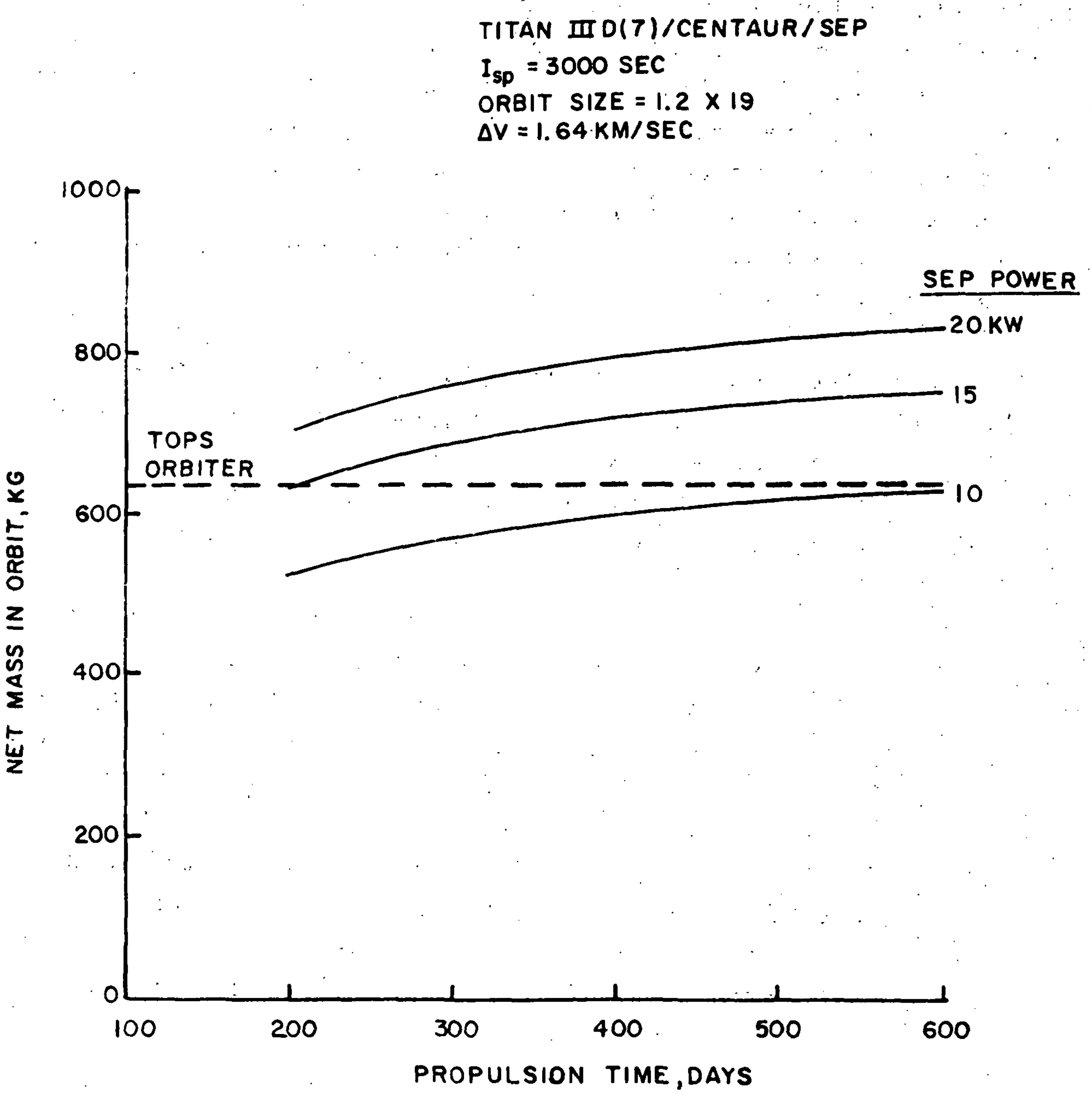

FIGURE 3-9. EFFECT OF REDUCING PROPULSION TIME FOR 3600-DAY URANUS ORBITER MISSION (b.) TITAN III D(7)/CENTAUR APPLICATION.

Figure 3-10 compares the two trajectory profiles for a 5100 day mission to Neptune. Upon closer examination from a practical design standpoint, it is found that the indirect mode is, on balance, inferior to the direct mode. Among the important disadvantages are: (1) higher values of optimum power, (2) a more rapid falloff of net mass with reduction in power rating, (3) very long propulsion on-time, (4) a higher V_{HP} for the same flight time which means a larger retro stage requirement. Indirect mode data will be included in this section to illustrate several of the above disadvantages.

Since the Neptune trajectory/payload data is presented in the same format as that for Uranus, brevity will be served by not going through as detailed a description of the figures. The data is presented in Figures 3-11 through 3-19. The following points are made in summary of the results:

1. The maximum value of hyperbolic approach velocity is 8.3 km/sec for spacecraft insertion into a 1.2 x 100 orbit, assuming that the TOPS retro system (ΔV = 1.64 km/sec) is employed. While V_{HP} is similar to that at Uranus, the flight times are much longer. Again, for the TOPS system, the minimum flight time is 4900 days (13.4 years) for the direct mode and 5600 days (15.4 years) for the indirect mode. If the retro capability were increased to 2.5 km/sec, the corresponding flight times are 11.2 years and 13.6 years, respectively.

2. Over the flight time range 3400-5800 days, the optimum power (Titan IIID/Centaur) varies between 30 kw and 35 kw for the direct mode, and between 48 kw and 37 kw for the indirect mode. Optimum I_{sp} values are about 2750 sec (direct mode) and 3800-4600 sec. (indirect mode).

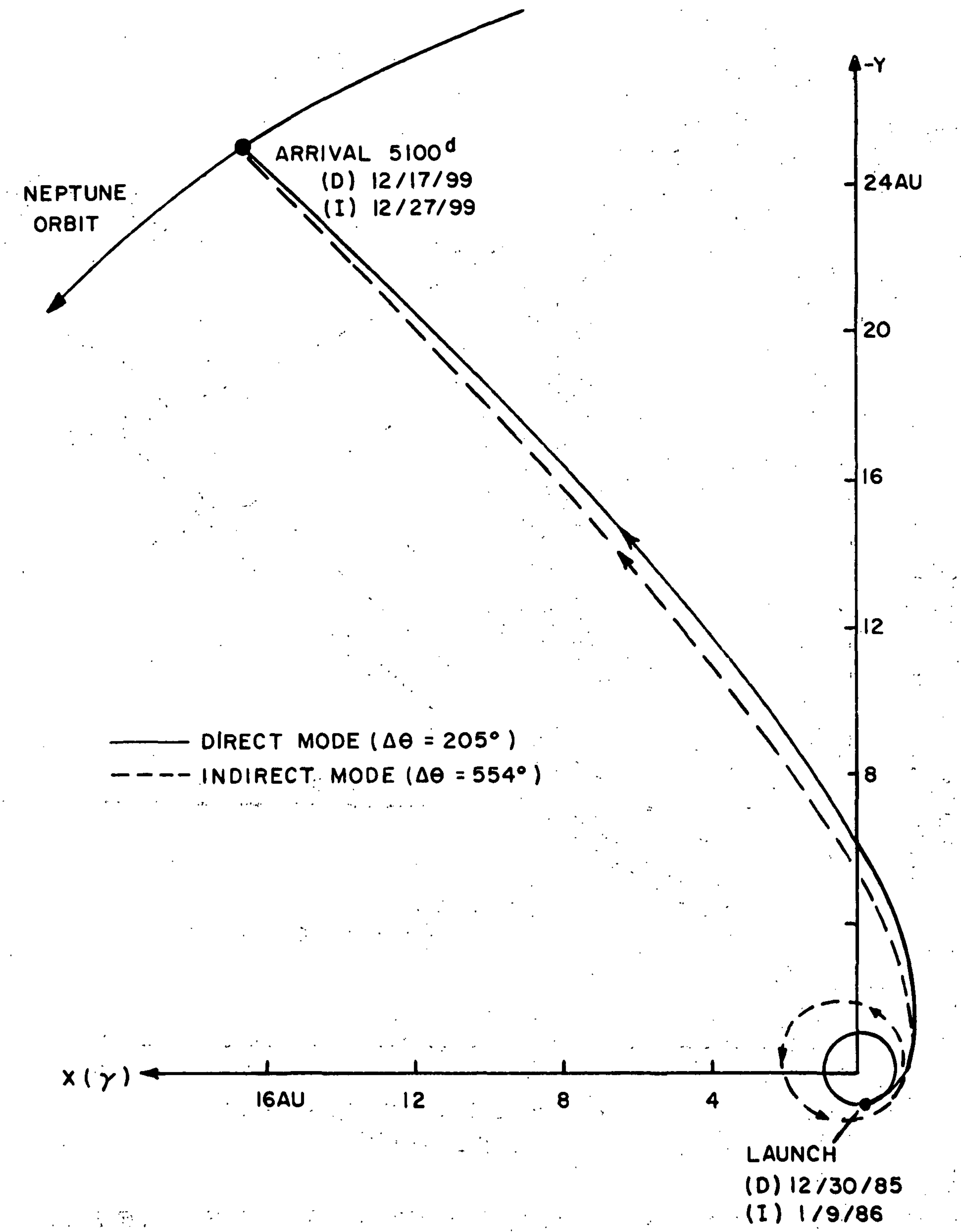

FIGURE 3-10. SOLAR ELECTRIC TRAJECTORIES TO NEPTUNE.

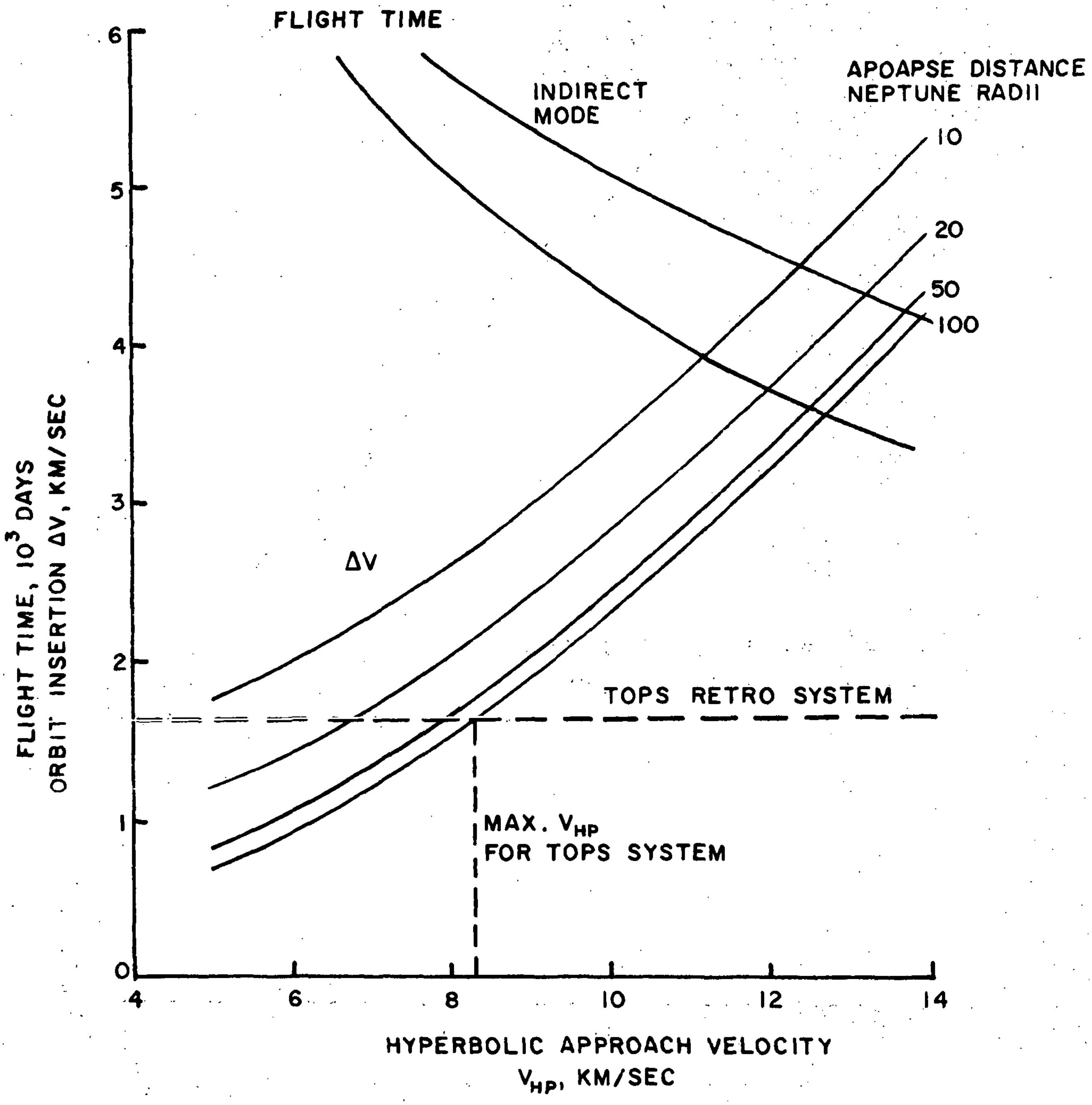

FIGURE 3-11. NEPTUNE ORBIT INSERTION CHARACTERISTICS
(a.) PERIAPSE DISTANCE = 1.2 NEPTUNE RADII.

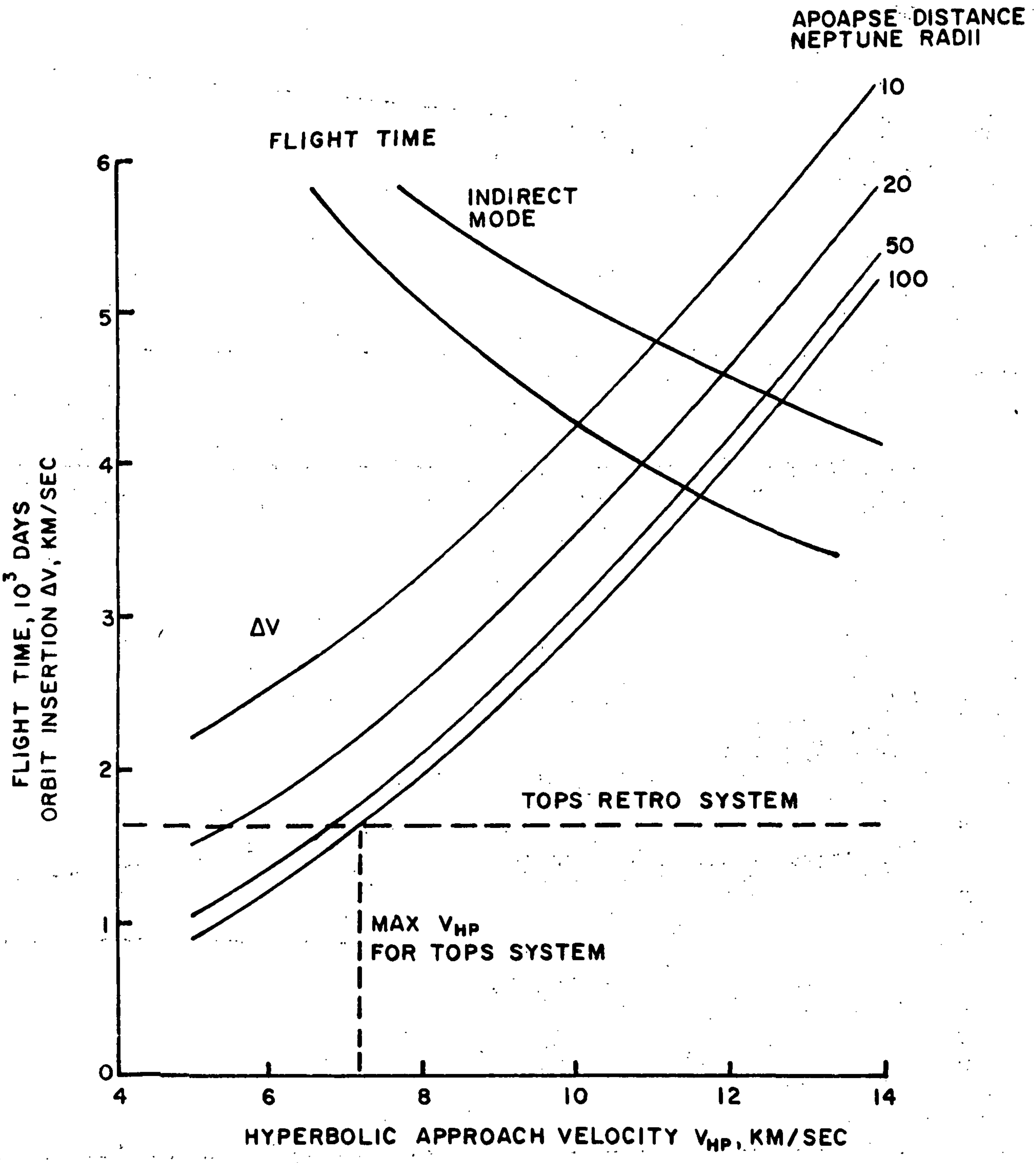

FIGURE 3-11. NEPTUNE ORBIT INSERTION CHARACTERISTICS
(b.) PERIAPSE DISTANCE = 2 NEPTUNE RADII.

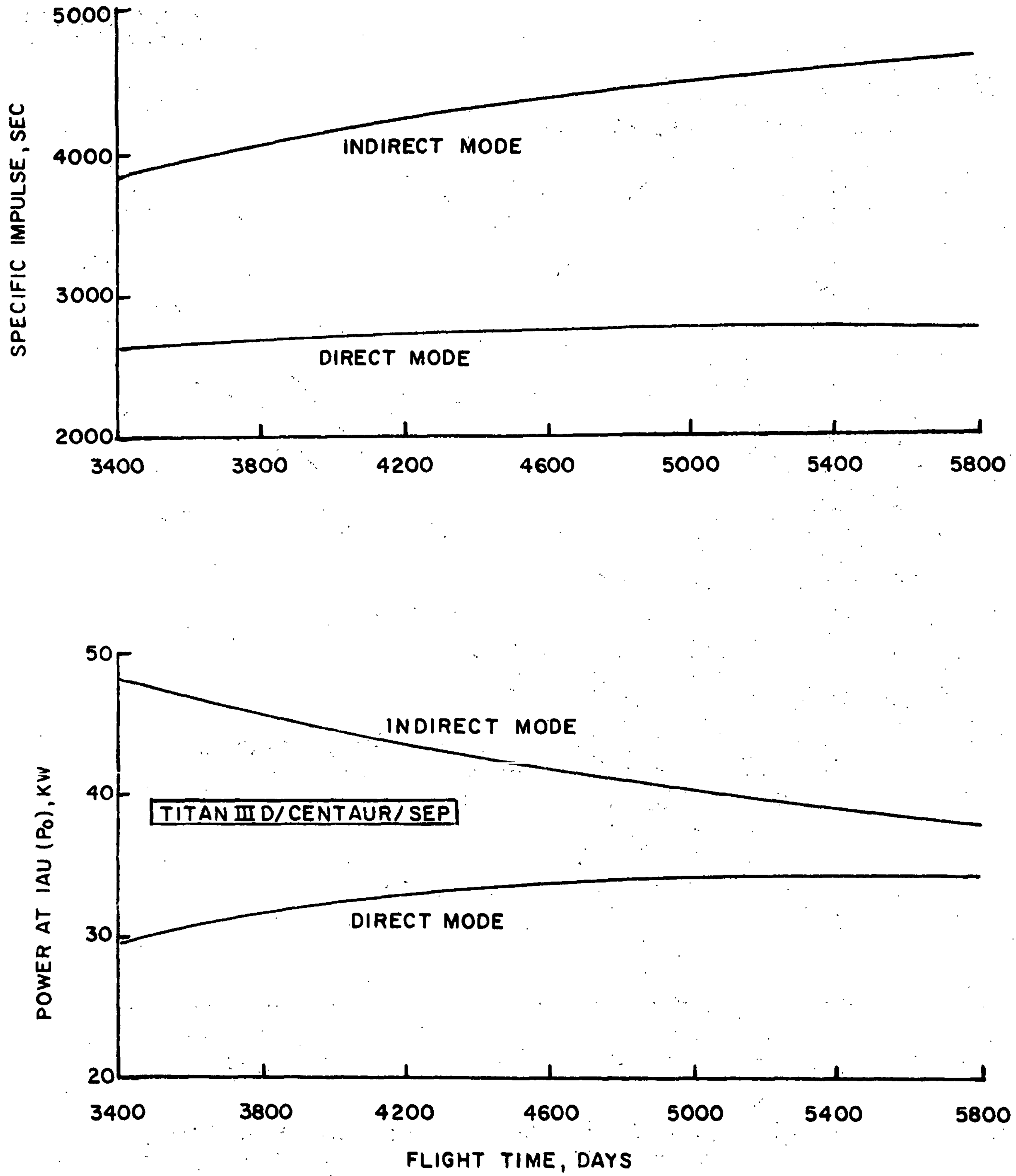

FIGURE 3-12. OPTIMUM POWER RATING AND SPECIFIC IMPULSE FOR SOLAR ELECTRIC NEPTUNE MISSIONS.

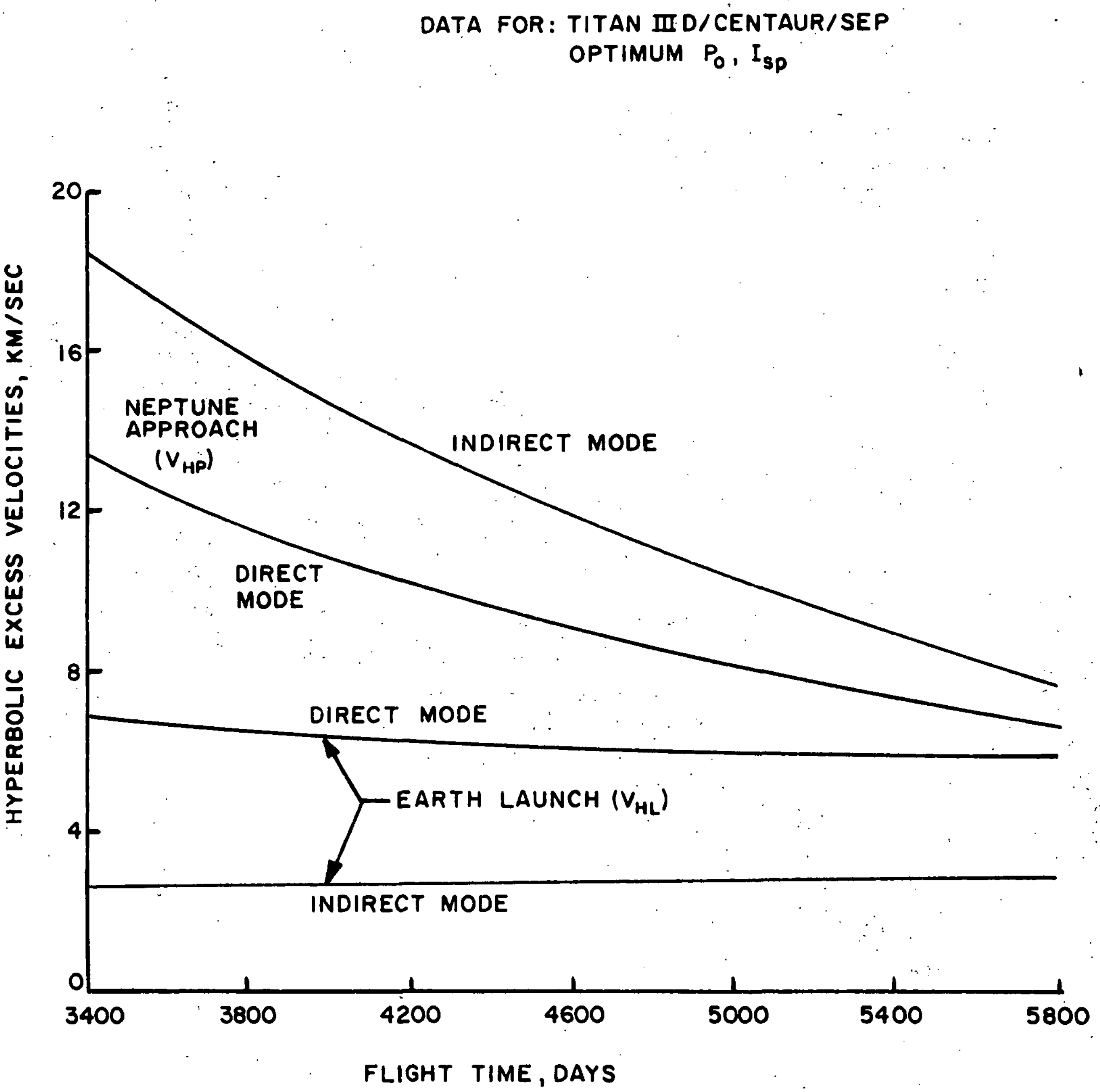

FIGURE 3-13. OPTIMUM LAUNCH AND APPROACH VELOCITIES FOR SOLAR ELECTRIC NEPTUNE MISSIONS.

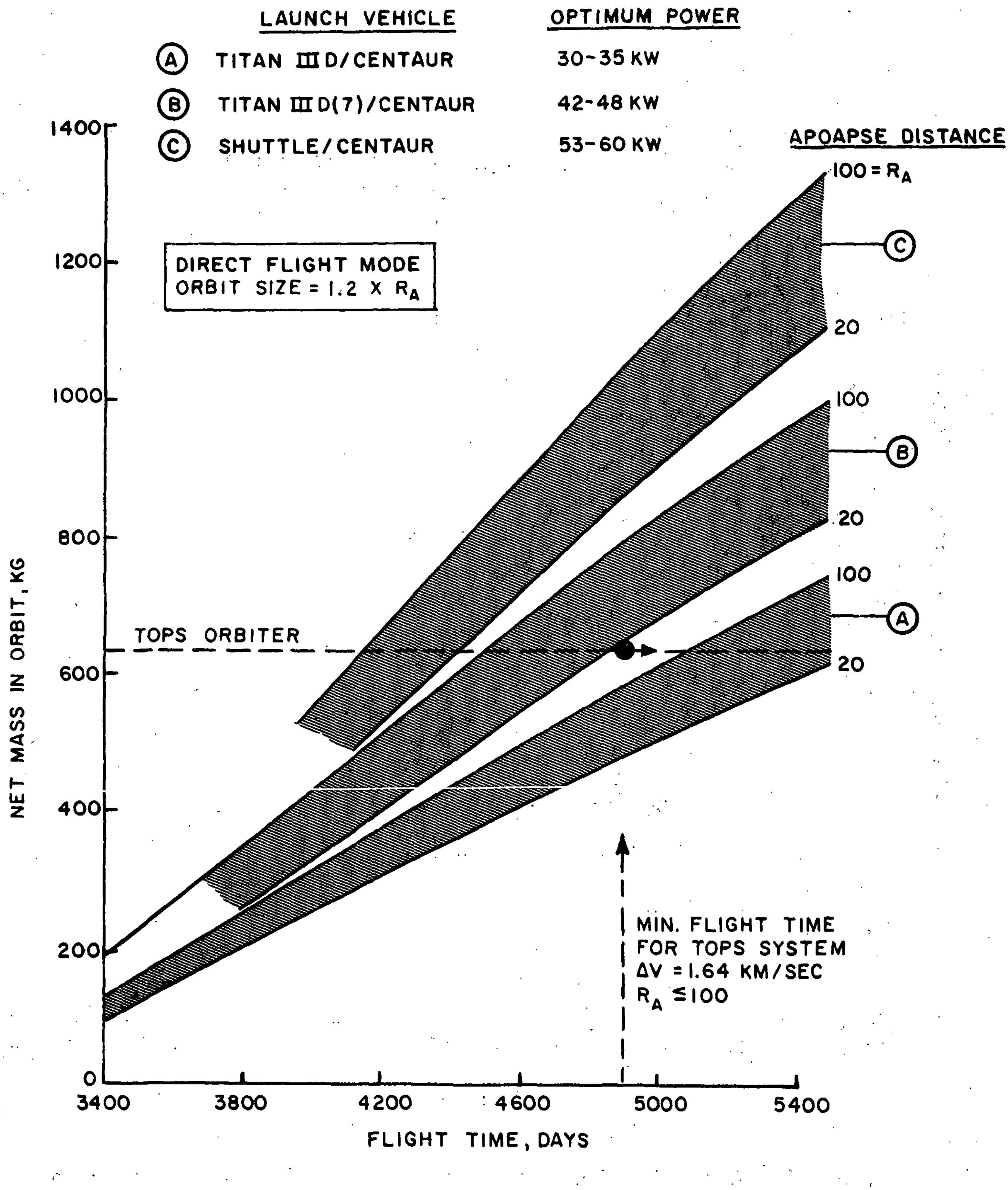

FIGURE 3-14. SOLAR ELECTRIC PROPULSION CAPABILITY FOR NEPTUNE ORBITER MISSIONS, DIRECT FLIGHT MODE, AVERAGE LAUNCH OPPORTUNITY.

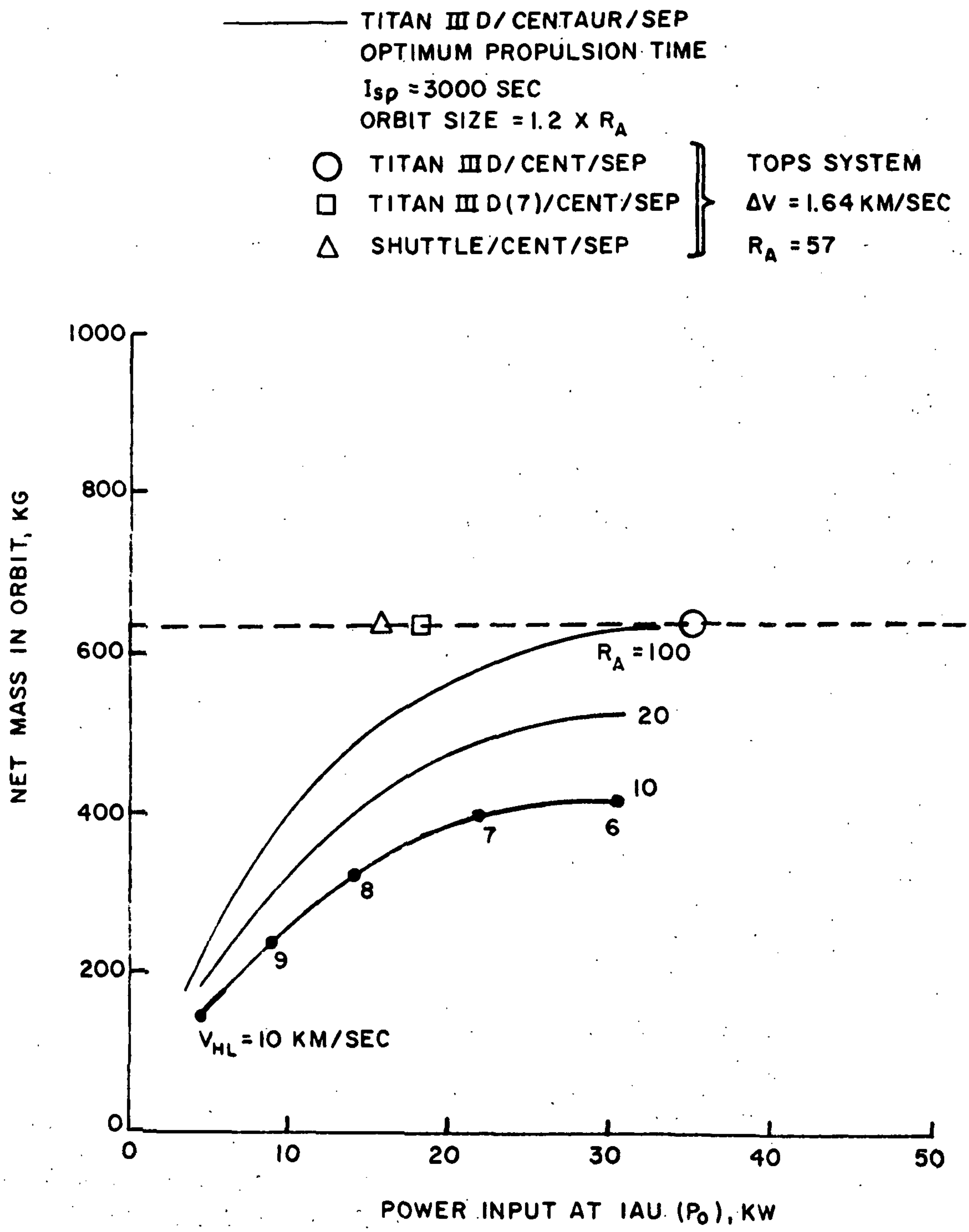

FIGURE 3-15. NET MASS IN ORBIT VERSUS SEP POWER FOR 5100-DAY NEPTUNE ORBITER MISSION, DIRECT FLIGHT MODE, LAUNCH 12/30/85.

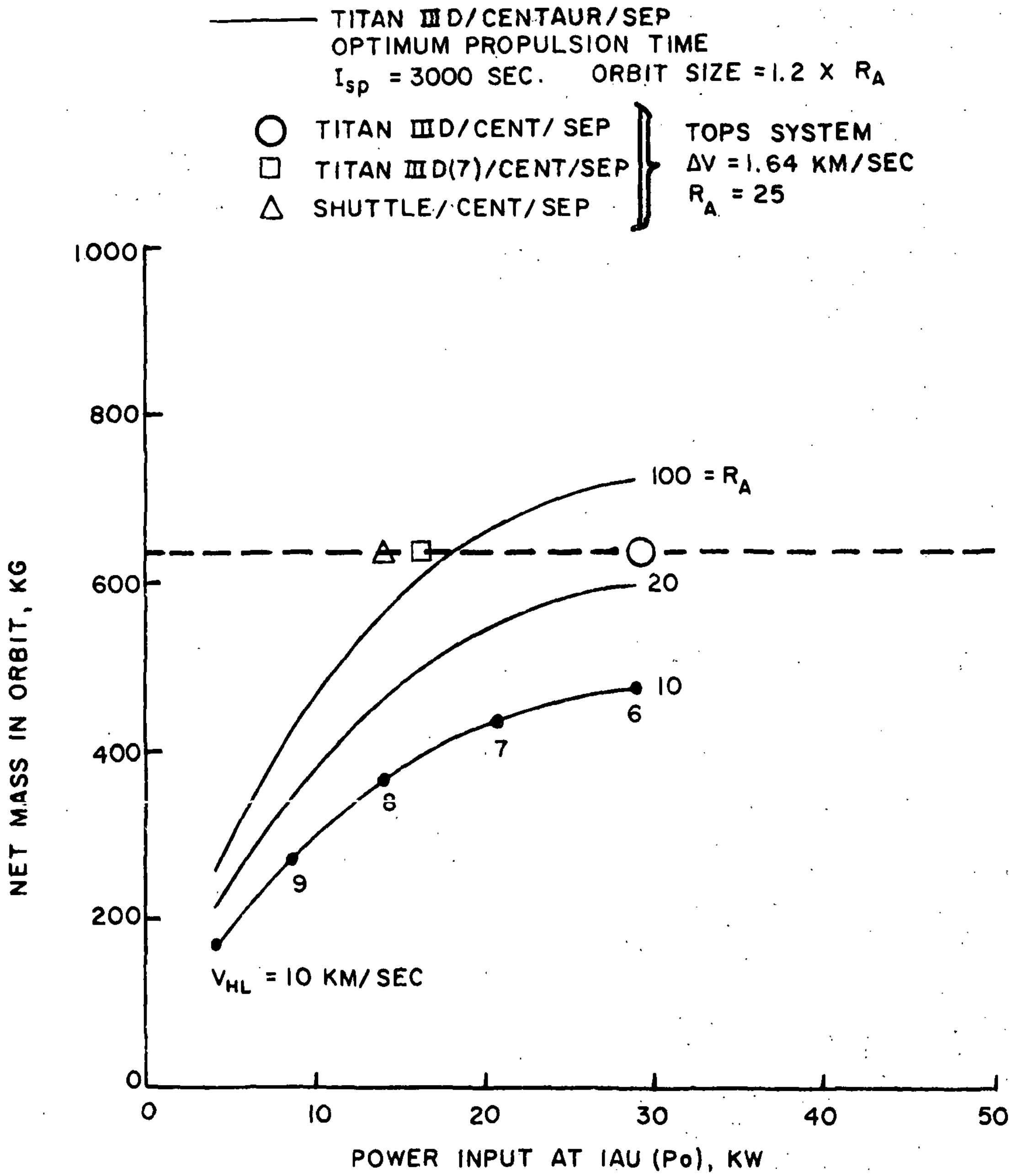

FIGURE 3-16. NET MASS IN ORBIT VERSUS SEP POWER FOR 5500-DAY NEPTUNE ORBITER MISSION, DIRECT FLIGHT MODE, LAUNCH 12/30/85.

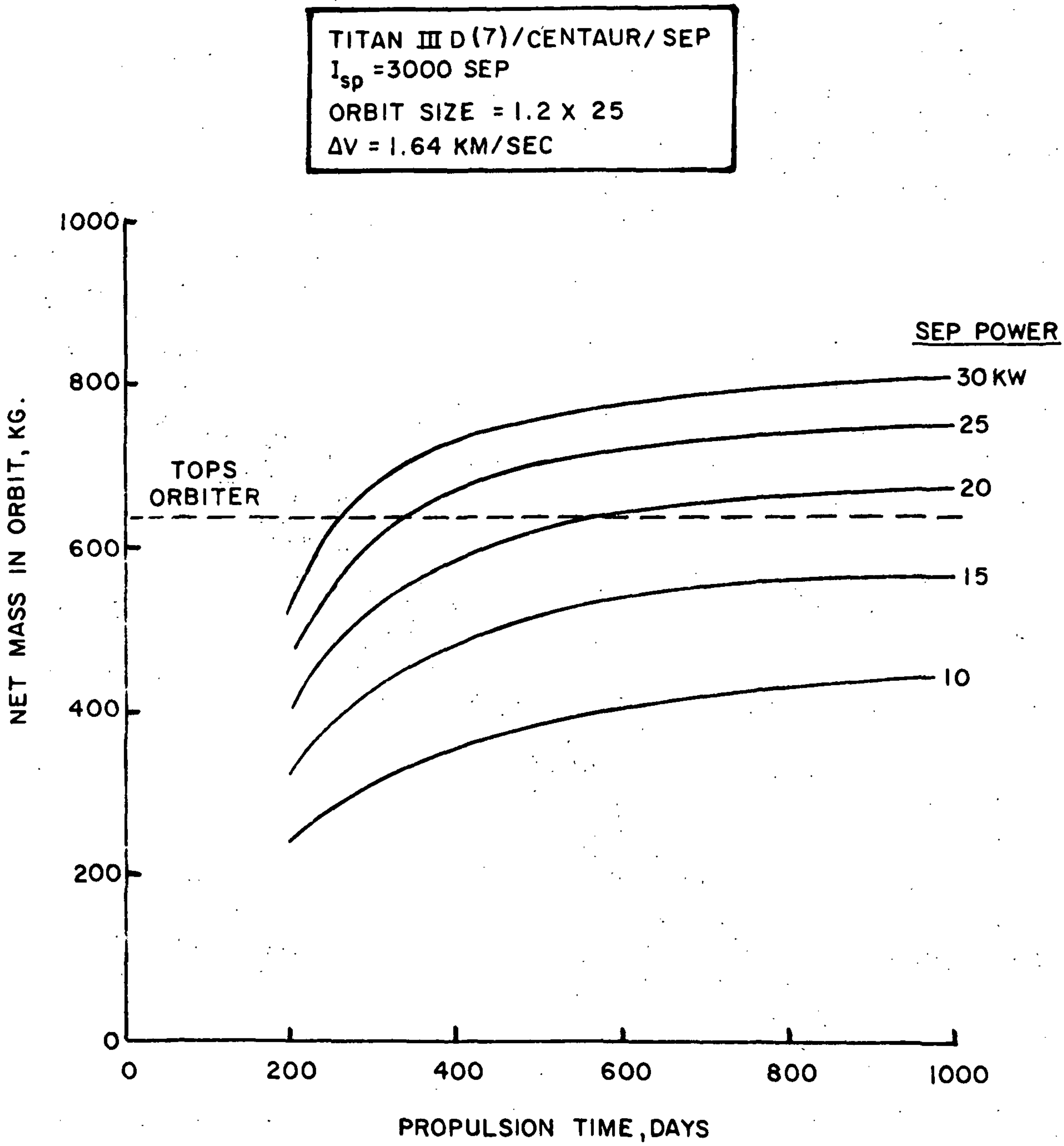

FIGURE 3-17. EFFECT OF REDUCING PROPULSION TIME FOR 5500-DAY NEPTUNE ORBITER MISSION.

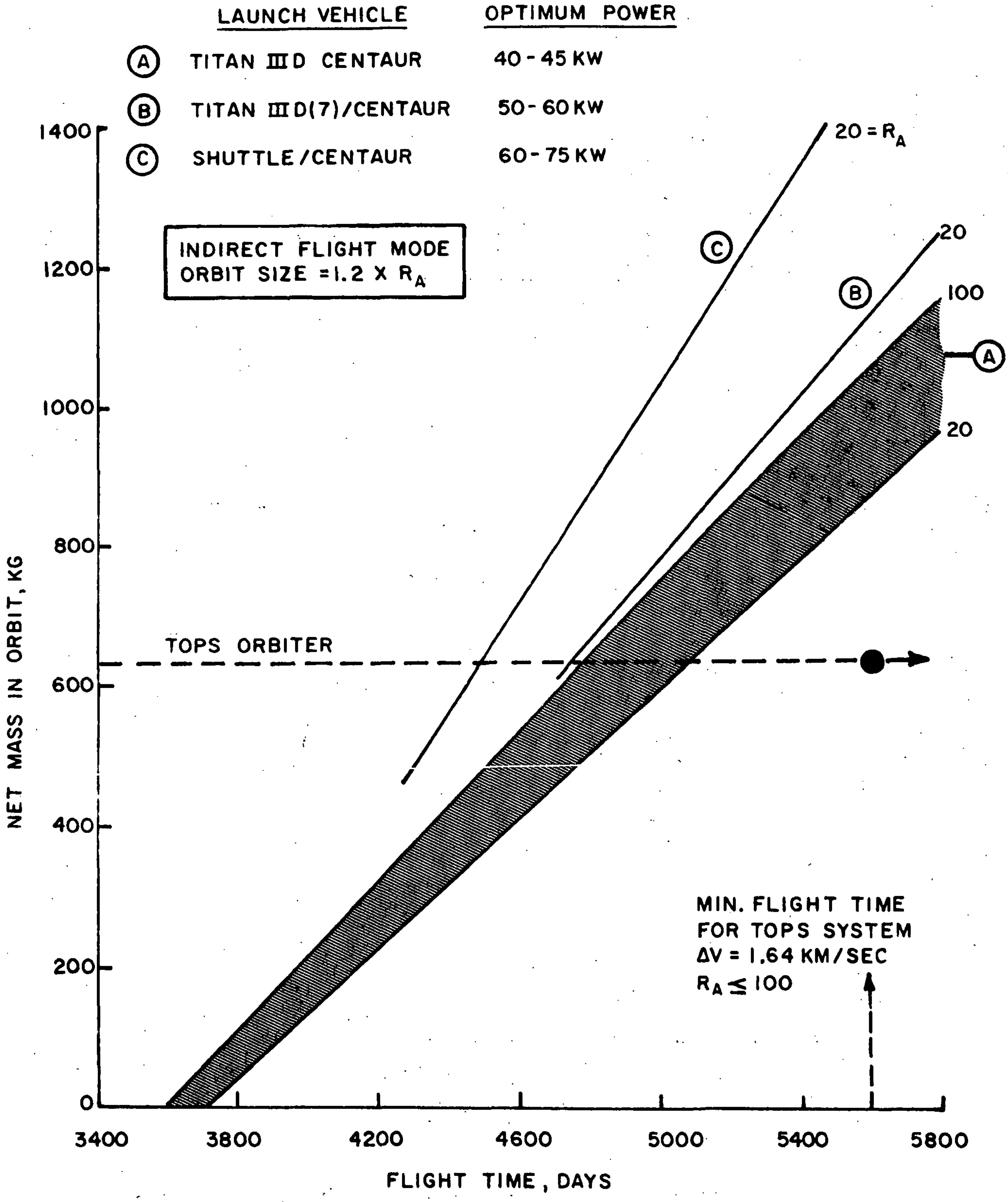

FIGURE 3-18. SOLAR ELECTRIC PROPULSION CAPABILITY FOR NEPTUNE ORBITER MISSIONS, INDIRECT FLIGHT MODE, AVERAGE LAUNCH OPPORTUNITY.

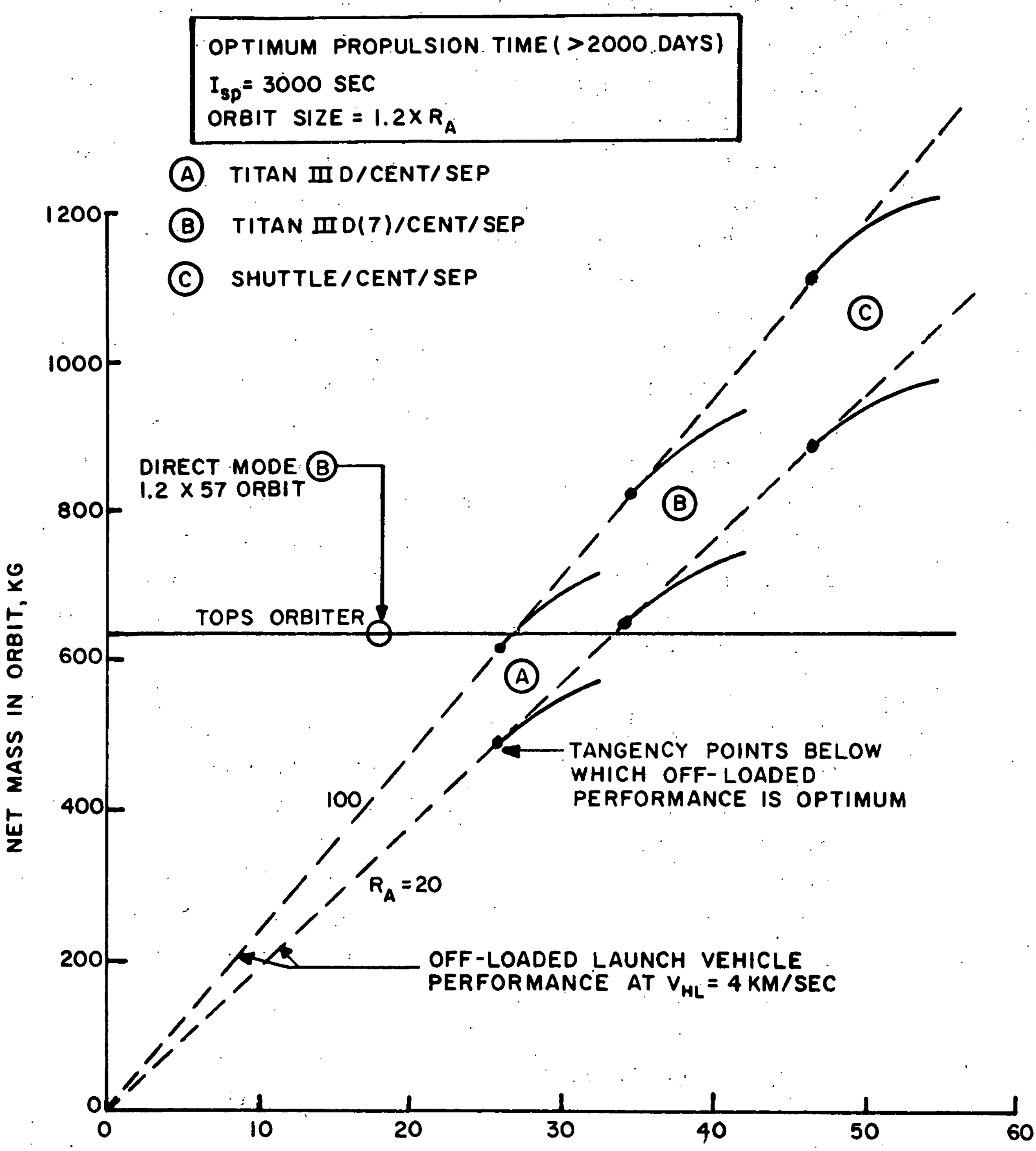

FIGURE 3-19. NET MASS IN ORBIT VERSUS SEP POWER FOR 5100-DAY NEPTUNE ORBITER MISSION, INDIRECT FLIGHT MODE, LAUNCH 1/9/86.

3. Assuming the Titan IIID/Centaur and optimum SEP power, a small Pioneer-type orbiter could be placed into a 1.2 x 50 orbit for a flight time of 4030 days (11 years). The insertion ΔV is 2.75 km/sec and retro stage weight is 550 kg. Under the same conditions, the TOPS orbiter (632 kg) would require a flight time of 5200 days (14.2 years), an insertion ΔV of 1.6 km/sec, and a retro weight of 480 kg.

4. When off-optimum power and propulsion time are considered, it is found that the Titan IIID/Centaur application is marginal even for a 5500 day (15 year) mission. A 20 kw powerplant is required to place the TOPS orbiter into a 1.2 x 50 orbit, and the propulsion time exceeds 1000 days. For the same flight time and powerplant size, the Titan IIID(7)/Centaur would effect a propulsion time reduction to 600 days and place TOPS into a 1.2 x 25 orbit.

5. Figure 3-19 compares direct and indirect mode performance as a function of SEP power. Although it is true that the indirect mode can deliver larger payloads at very high power inputs, the direct mode performance is clearly better in the power range 10-20 kw. Furthermore, the indirect mode propulsion time is several thousand days, and this cannot be decreased without severe loss in payload.

3.4 Flight Mode Comparisons

Trajectories to the outer planets via a Jupiter swingby maneuver are well known for both ballistic and SEP flight modes. In general, they have been considered more applicable to flyby rather than orbiter missions because of the high relative velocities at the target planet. However, if longer flight times are allowed, swingby trajectories can be useful for orbiter missions as well.

Figure 3-20 compares direct and swingby modes for Uranus orbiter missions. The direct ballistic flights require very high launch velocities (V_{HL} > 11.5 km/sec) which is reflected in the relatively poor performance shown for this flight mode. Even when the proposed high-energy Versatile Upper Stage (VUS) is matched to the Titan IIID/Centaur, the net mass in orbit is less than 600 kg for flight times under 4000 days. A 15 kw SEP upper stage yields significantly better performance than the VUS. It should be mentioned, however, that the Titan IIID(7)/Centaur/VUS is capable of placing 632 kg into orbit for a flight time of 3200 days. The upper performance curve in Figure 3-20 is for the Jupiter swingby mode and applies equally to a 15 kw SEP stage or the VUS stage, again assuming the Titan IIID/Centaur. The flight time required to deliver the TOPS orbiter is about 3000 days, or 1 year less than the direct SEP mode. At 3000 days, the Uranus approach velocity via Jupiter swingby is 9.6 km/sec. From Figures 3-2 and 3-3(a) we find that the orbit insertion ΔV for a 1.2 x 50 orbit is 2.45 km/sec, and the retro stage weight for the TOPS orbiter is 940 kg. In comparison, the direct SEP flight of 3400 days has V_{HP} = 7 km/sec, ΔV = 1.45 km/sec, and M_{retro} = 420 kg. Another way of comparing the two flight modes is on the basis of the same flight time, say 3400 days. The swingby mode has a payload advantage of about 225 kg (in orbit) which translates to an increase of 415 kg at planet approach. This would allow a mission which combined atmospheric probe(s) as well as the TOPS orbiter. Swingby opportunities are restricted by the Jupiter-Uranus synodic period of 13.8 years. Hence the 1979 launch opportunity, which is probably too early for orbiter missions, would not be repeated again until 1993. One

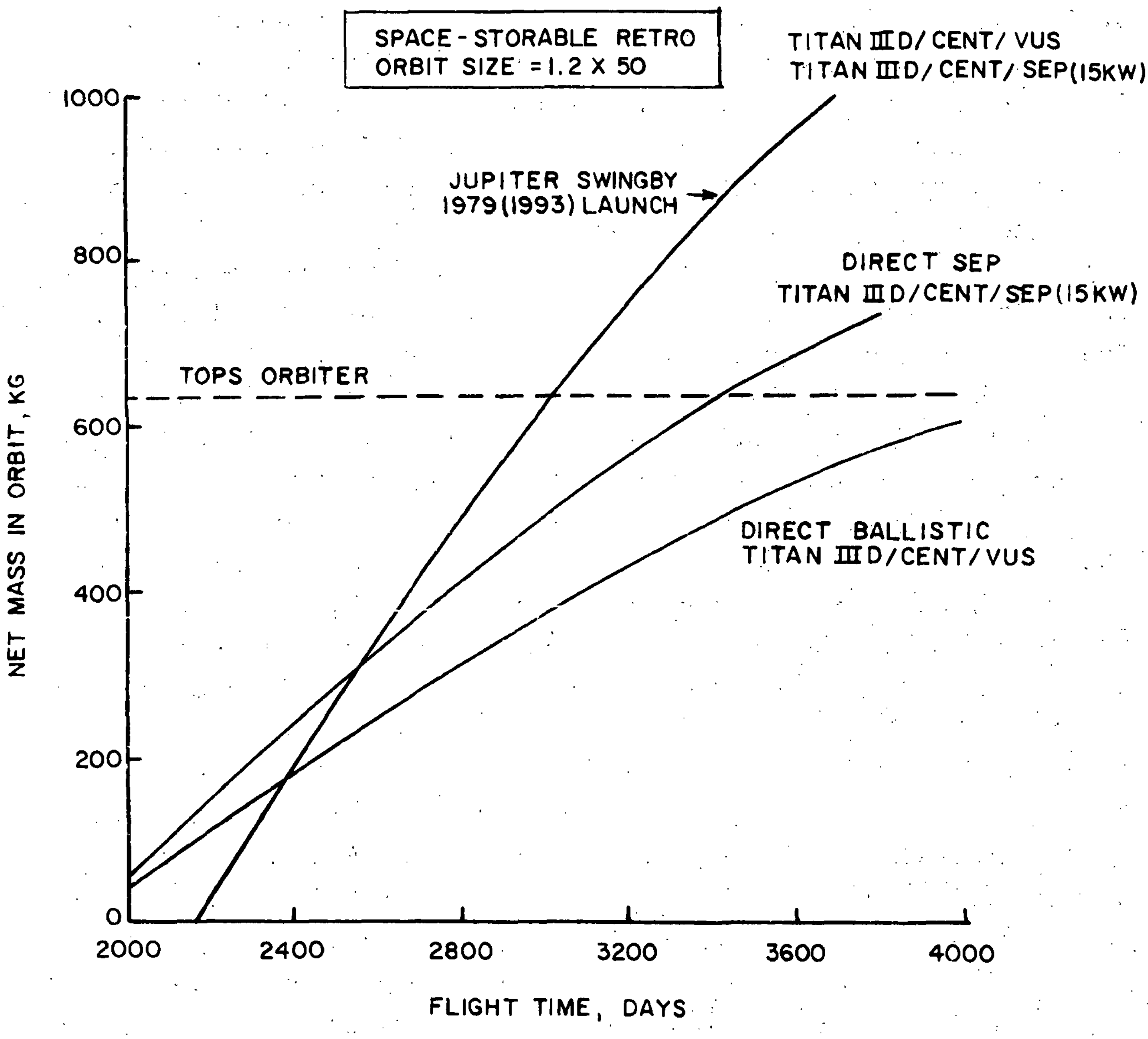

FIGURE 3-20. FLIGHT MODE COMPARISON FOR URANUS ORBITER MISSIONS.

advantage, then, of direct SEP missions is that launch opportunities are available every year.

Figure 3-21 shows flight mode comparisons for the Neptune orbiter missions. Similar conclusions may be inferred from the results. For orbiters of the TOPS class, the direct ballistic mode is even less favorable than for the Uranus mission (although Titan IIID(7)/Centaur/VUS does provide sufficient performance). The minimum flight time to insert TOPS into a 1.2 x 50 orbit is 4800 days (13 years) using a Jupiter swingby. Again, a 15 kw SEP stage has essentially the same performance as the VUS chemical stage matched to the Titan IIID/Centaur launch vehicle. At 4800 days, the orbit insertion parameters are V_{HP} = 10.2 km/sec, ΔV = 2.5 km/sec, and M_{retro} = 790 kg. With a Jupiter-Neptune synodic period of 12.8 years, the next two launch opportunities are 1979 and 1992.

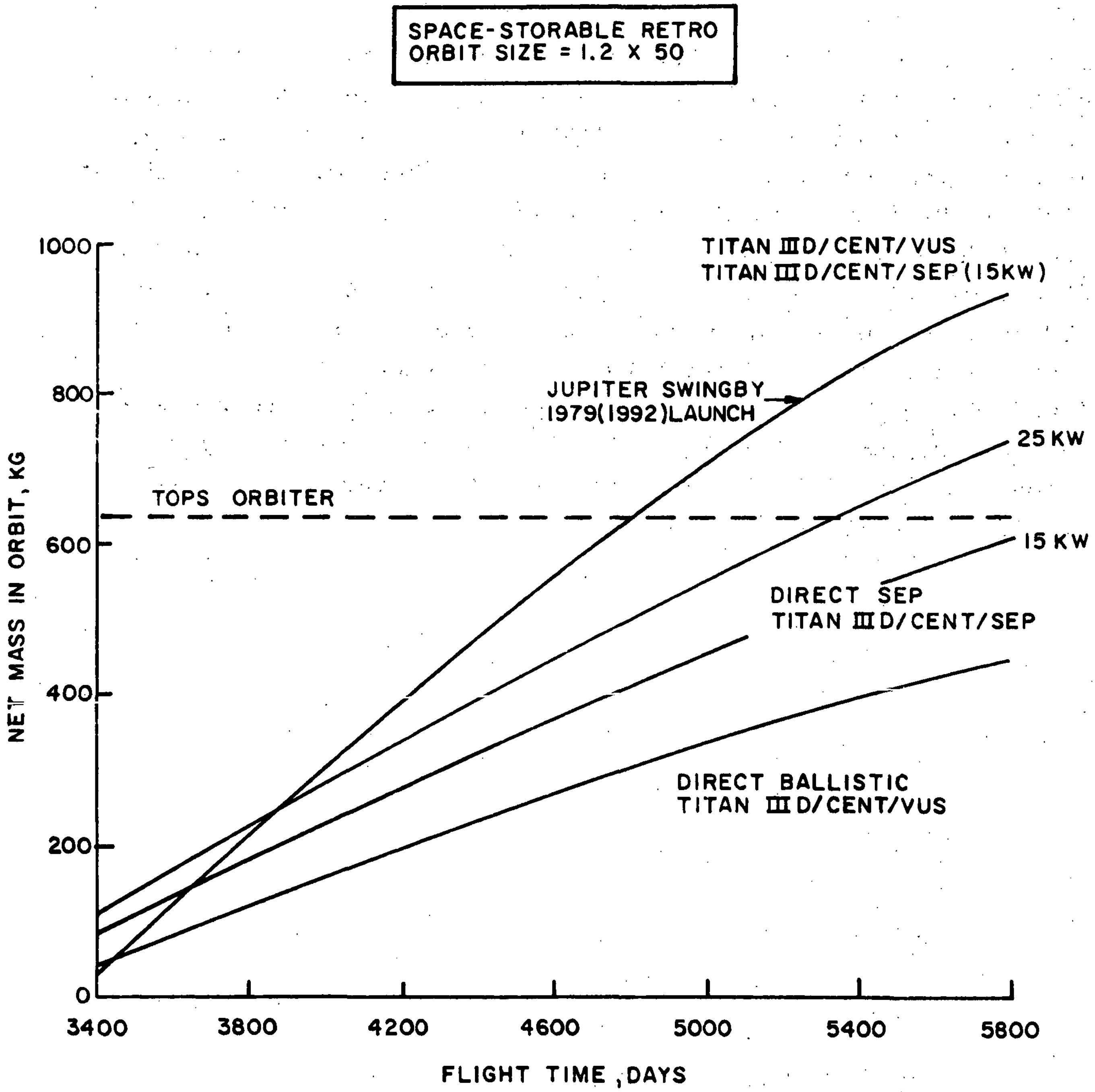

FIGURE 3-21. FLIGHT MODE COMPARISON FOR NEPTUNE ORBITER MISSIONS.

4. BASELINE MISSION PERFORMANCE SUMMARY

The purpose of this final section of the report is to bring together some of the study results described in the preceding sections. It will be helpful to select representative design-point examples for the Uranus and Neptune missions. The criterion, of course, is that the orbiter payload and orbit size selection reasonably satisfy the science objectives, and that the SEP phase of the mission is practical. These examples are just that - clearly, the parametric data presented in Section 3 would allow one to arrive at other choices.

Table 4-1 lists the system weights and key parameters of the baseline missions. A common SEP/orbiter system design is chosen for both Uranus and Neptune. The SEP "stage" has a total weight of 1008 kg and is jettisoned shortly after its propulsion function is accomplished. The propulsion system is rated at 20 kw (power input at 1 AU) with the ion thrusters operating at 3000 sec. specific impulse. Mercury propellant and tankage comprise 408 kg of the total SEP stage weight; this includes added propellant for an extended launch window of 20 days or more. The chemical retro stage, using a space-storable propellant such as Fluorine/Hydrazine, weighs 420 kg and provides an orbit insertion* ΔV of 1.5 km/sec. The proposed TOPS orbiter (632 kg) is assumed for the science-dependent mission function. The total weight, at Earth departure, of the SEP/orbiter system is 2060 kg.

The direct mission to Uranus is about 10 years duration and can be launched by the Titan IIID/Centaur. Maximum operating

* The re-startable retro stage could also provide for midcourse and orbit correction maneuvers. These were not specifically accounted for in the study. A typical maneuver budget of 100-200 m/sec would require a slightly larger retro, or, alternatively, a larger orbit size.

TABLE 4-1

EXAMPLE BASELINE MISSION SUMMARY
FOR URANUS AND NEPTUNE ORBITERS

1. Common SEP/Orbiter System Weights

Solar Electric System		1008 (KG)
Propulsion System (20 kw/3000 sec)	600	
Propellant + Tankage	408	
Chemical Retro Stage (383 sec/0.25)		420
ΔV = 1.5 km/sec		
Orbiter Systems (TOPS)		632
Earth Departure Weight		2060 (KG)

2. Mission Parameters

	URANUS	NEPTUNE
Launch Vehicle	Titan IIID/Centaur	Titan IIID(7)/Centaur
Launch V_{HL} (KM/sec)	7.2	8.2
Max. Injected Weight (KG)	2110	2200
Flight Time (days)	3600	5500
Max. Propulsion time (days)	440	455
Approach Velocity V_{HP} (KM/sec)	6.5	7.2
Orbit Size (Radii)	1.2 x 29	1.2 x 43
Orbit Period (Earth Days)	7.2	10.4

time for the SEP system is only 440 days. The orbit size established is 1.2 x 29 Uranus radii and has a period of 7.2 Earth days. Fifty orbit passes are accumulated after one year.

The direct mission to Neptune has an extremely long flight time of 15 years. In this case, the Titan IIID(7)/Centaur has been taken as the baseline launch vehicle since the standard 5-segment Titan is inadequate. It may be noted that the Titan IIID/Centaur(F), if developed, could be substituted for the baseline launch vehicle. Maximum propulsion time is again relatively short - 455 days. The nominal orbit size is 1.2 x 43 Neptune radii. With an orbital period of 10.4 Earth days, about 35 orbit passes are made in one year's time.

Supporting data are presented in Figures 4-1 to 4-4. Launch window characteristics are shown as a plot of planet approach mass (retro stage plus orbiter) versus launch date. The launch opportunity 1985-86 is used here only as an example (the early 1990's would be a more promising date).

The optimum launch date for the Uranus mission is approximately Jan. 9, 1986. For an initial mass of 2060 kg, the maximum approach mass is 1085 kg. Hence, the maximum excess over the nominal approach mass of 1052 kg represents a 33 kg propellant penalty for an extended launch window. The Neptune data in Figure 4-3 shows a propellant addition of 93 kg giving a launch window of at least 35 days. The solar power profiles of the Uranus and Neptune missions are similar. A possible thruster array configuration would be nine 2.5 kw rated modules with one in spare. Each thruster should have a 2:1 throttling capability. Individual thrusters would be successively switched off as power input decreases. Also shown in Figures 4-2 and 4-4 are the optimum thrust cone angles (thrust vector displacement from the solar direction). The cone angle varies between 84° and 150° for the Uranus mission, and 63° to 131° for the Neptune mission.

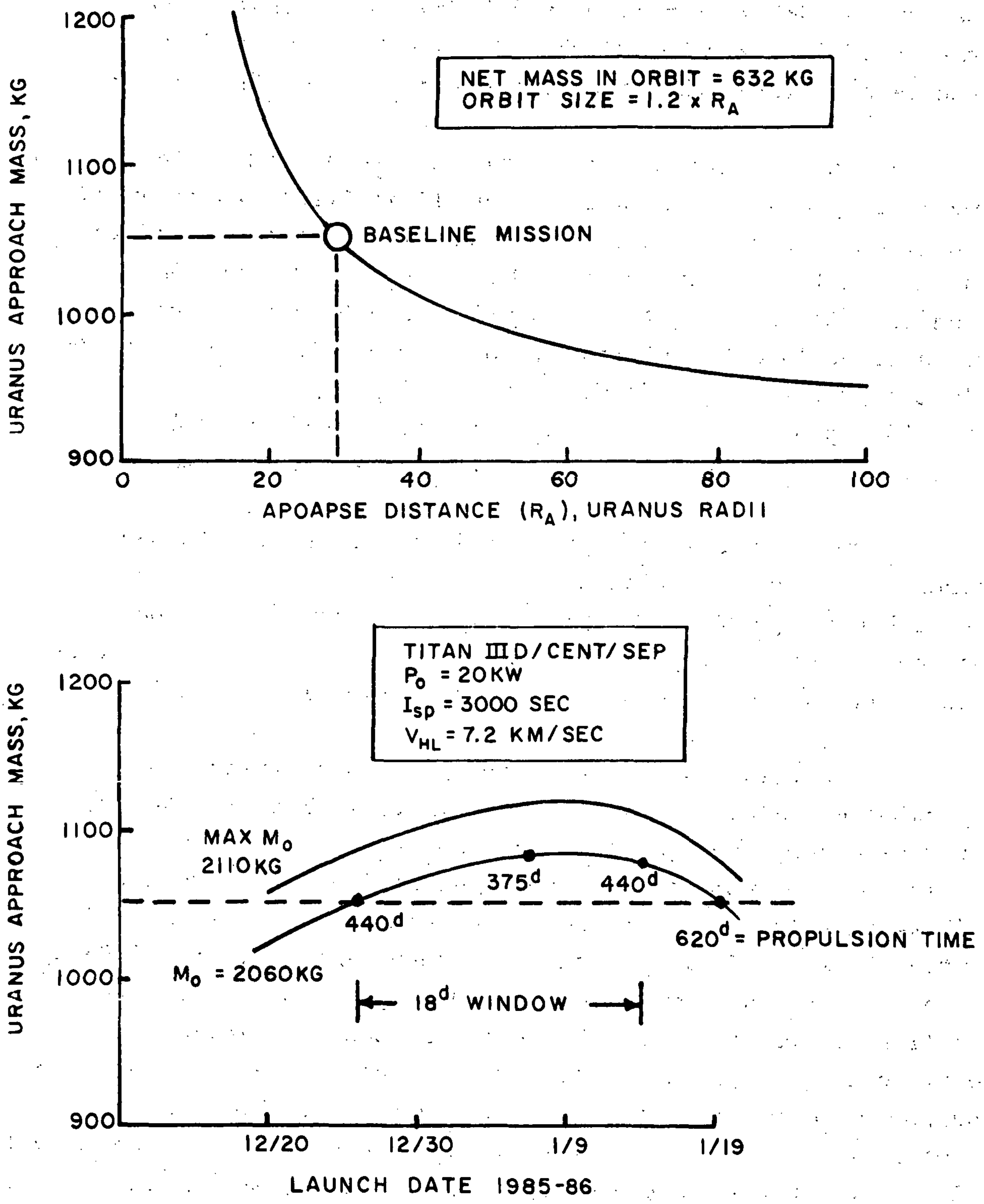

FIGURE 4-1. LAUNCH WINDOW PENALTY AND ORBIT SIZE CAPABILITY FOR 3600-DAY URANUS ORBITER MISSION.

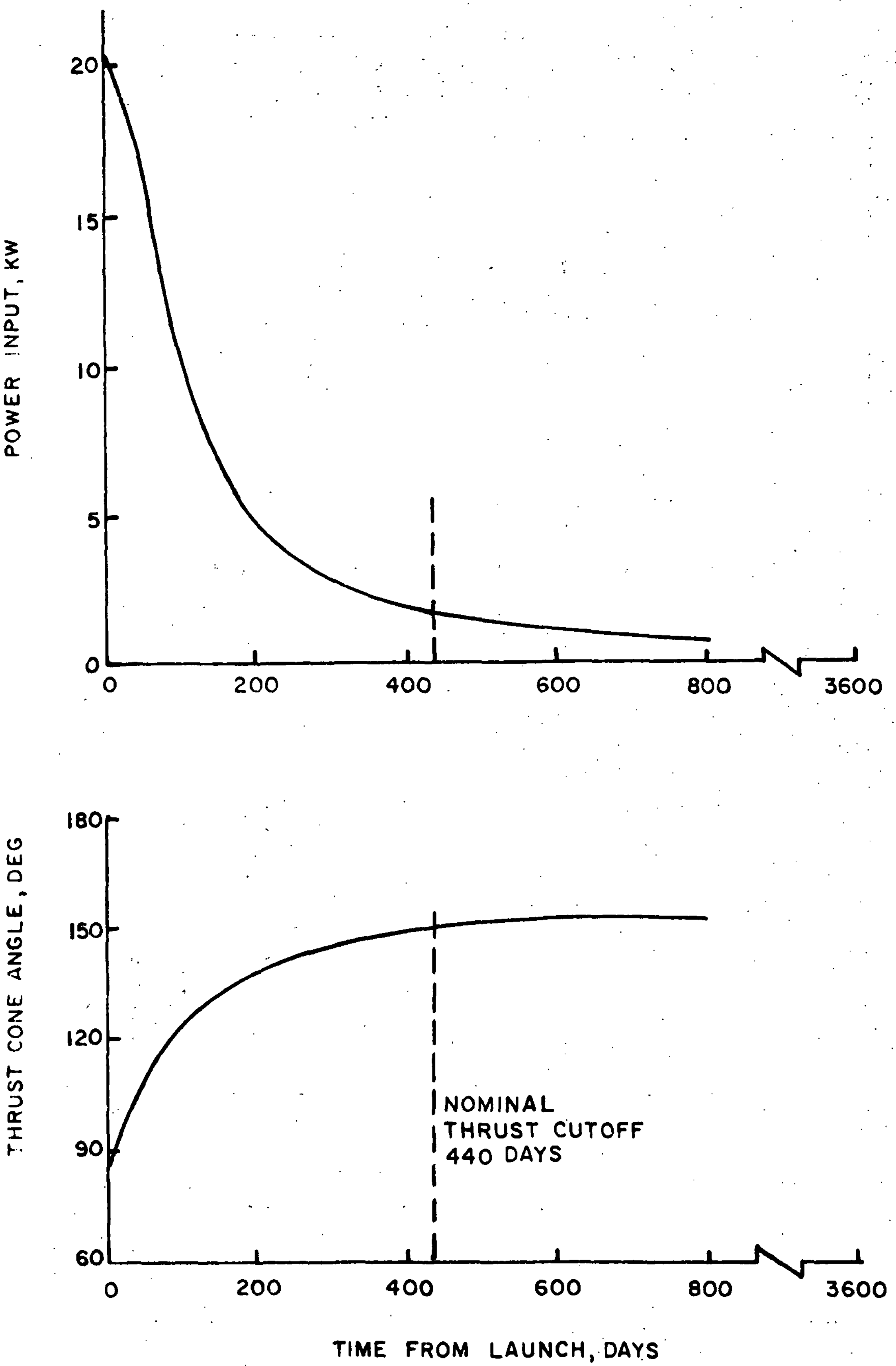

FIGURE 4-2. POWER PROFILE AND THRUST CONE ANGLE FOR URANUS ORBITER MISSION.

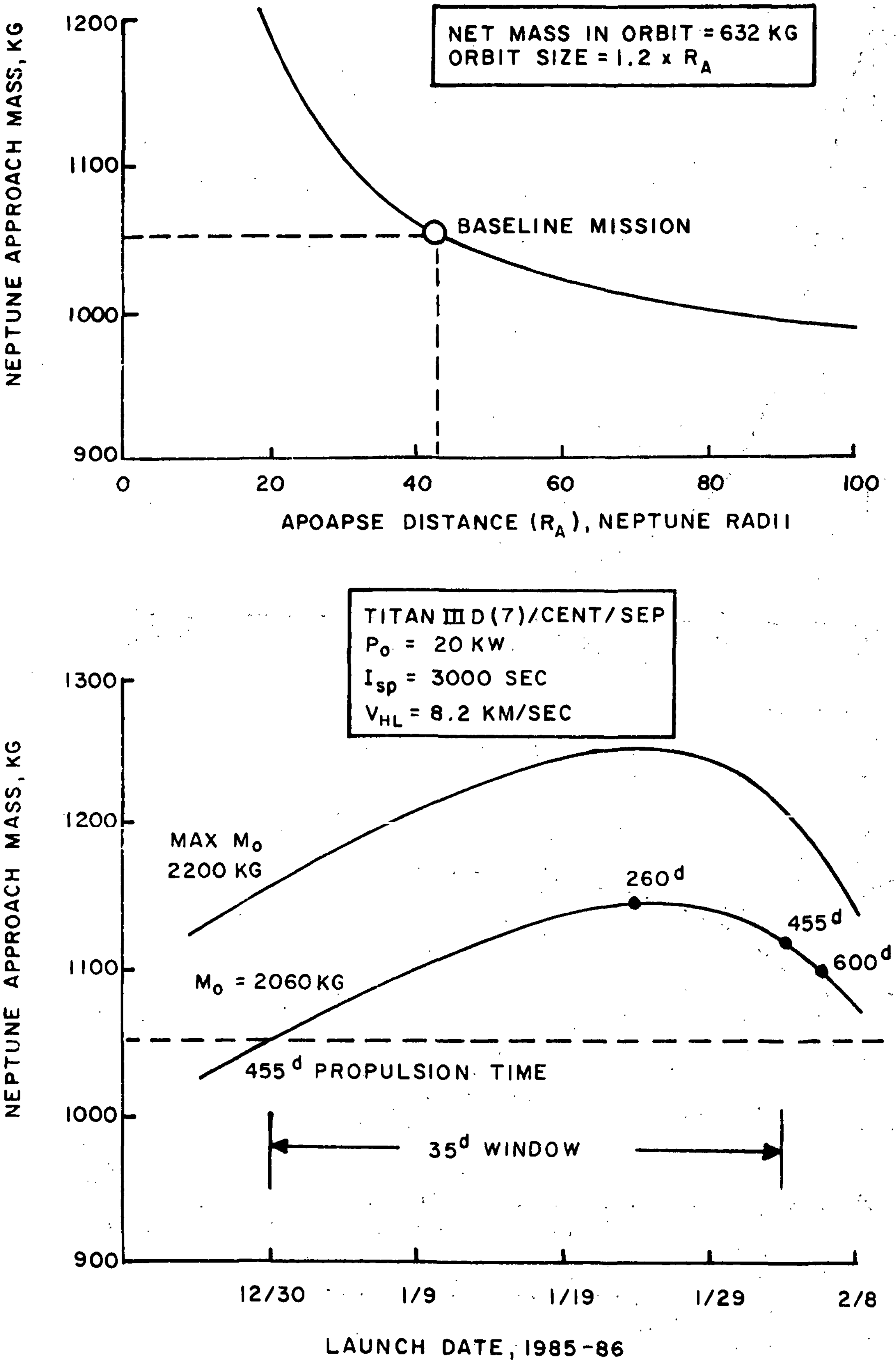

FIGURE 4-3. LAUNCH WINDOW PENALTY AND ORBIT SIZE CAPABILITY FOR 5500-DAY NEPTUNE ORBITER MISSION

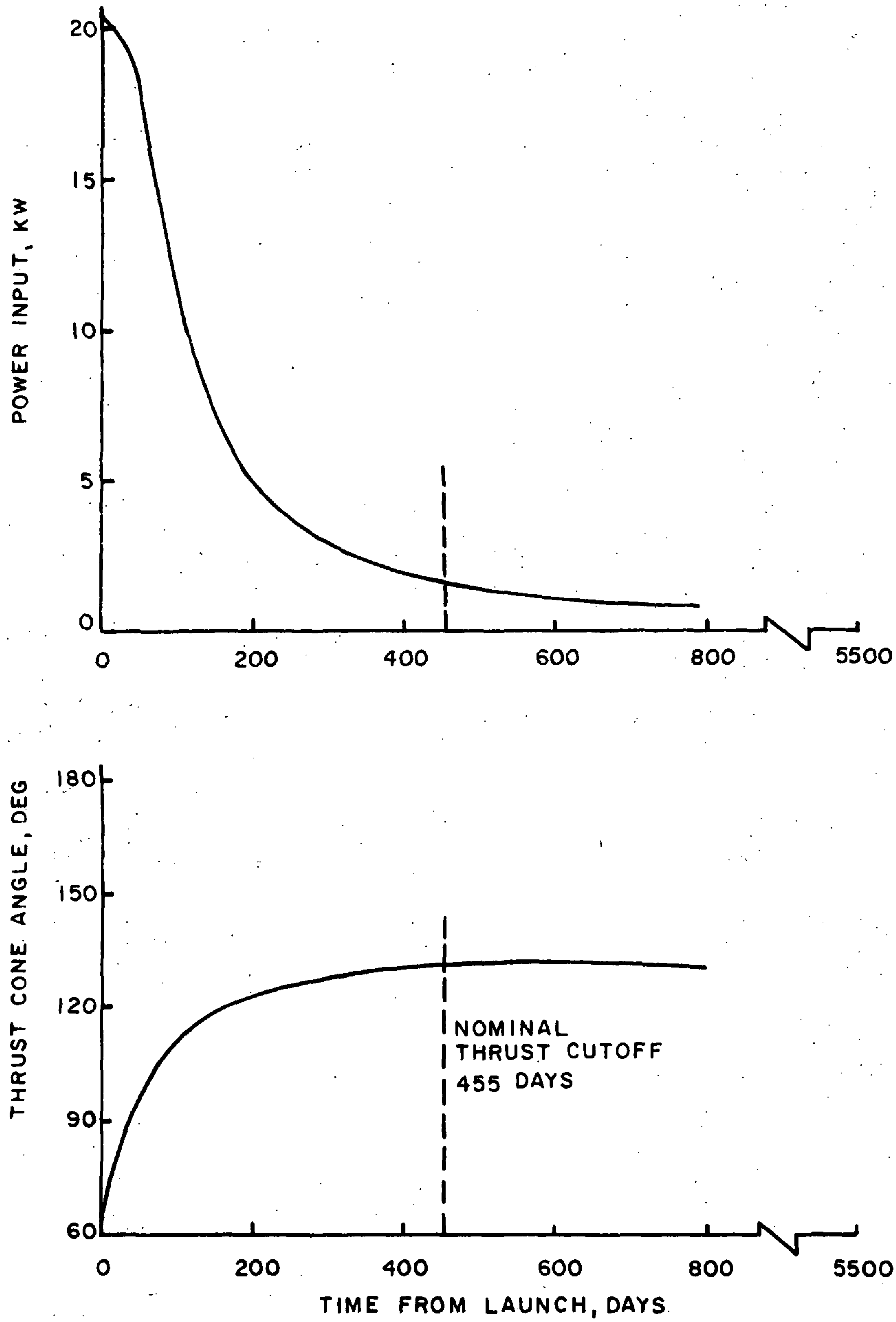

FIGURE 4-4. POWER PROFILE AND THRUST CONE ANGLE FOR NEPTUNE ORBITER MISSION.

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