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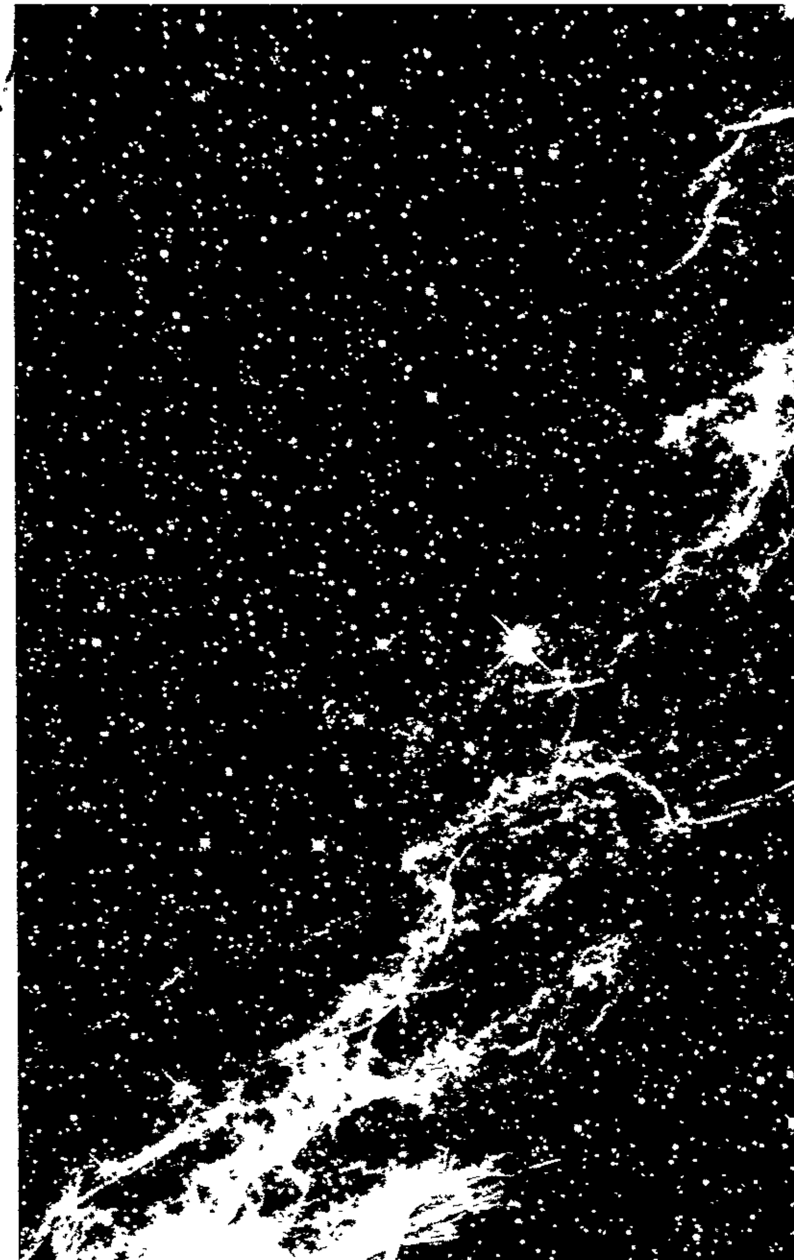

Report No. T-28

HALLEY'S COMET FLYTHROUGH AND RENDEZVOUS MISSIONS VIA SOLAR ELECTRIC PROPULSION

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HALLEY'S COMET FLYTHROUGH AND RENDEZVOUS MISSIONS VIA SOLAR ELECTRIC PROPULSION

Contract 952701

by

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FOREWORD

This Technical Report is the final documentation on all data and information required by Task 6: Halley's Comet Flythrough. 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 7, 8 and 9 of this study will be reported separately.

ACKNOWLEDGEMENT

The author expresses his appreciation to Mr. Frank Rinaldo of Astro Sciences, IITRI for his valuable assistance in performing the trajectory computational work required in this study.

SUMMARY AND CONCLUSIONS

This report describes the characteristics and capabilities of solar electric propulsion (SEP) for missions to Halley's Comet which is due to return in 1985-86. The 76 year period of this comet coupled with its large scientific and public interest makes this mission a rare and exciting opportunity. This study emphasizes trajectory/payload analysis and related requirements on flight time, launch vehicles, SEP powerplant size, propulsion on-time, and other SEP design characteristics. Questions concerning comet science objectives, instrumentation, mission operations and spacecraft design are being treated in other concurrent studies.

The Halley mission opportunity is investigated over a very wide range of encounter conditions. These conditions are described primarily in terms of the spacecraft approach velocity relative to the comet (0-55 km/sec), and the arrival date (0-200 days before perihleion). Depending upon the approach speed, the type of encounter may be classified as fast flythrough, slow flythrough or rendezvous. In a qualitative sense, rendezvous may be the preferred mission mode but also the most difficult to achieve. This is due to the high eccentricity and retrograde motion of Halley's Comet.

SEP system parameters assumed in the study are representative of current technology and design goals. Baseline values are 30 kg/kw specific mass, 3500 sec specific impulse, 3 percent tankage fraction, and solar array power under 20 kw. The use of launch vehicles in the Titan family is emphasized. It is noted that the baseline specific impulse and power rating are generally not optimum in the sense of maximum delivered payload (net spacecraft mass). This is particularly true of the optimum power requirement which could exceed 40 kw for certain mission applications. The effect of these parameters is discussed in the text.

A distillation of study results is presented in Figure S-1. The bar chart shows flight time and launch vehicle requirements over a range of approach velocity. The six mission examples cover the spectrum of Halley exploration capability. Payload delivered to the comet is approximately 450 kg. The "easiest" mission is clearly the 6 month, fast flythrough arriving 60 days before perihelion. In an operational sense this mission may not be easy in that the 55 km/sec approach velocity could cause instrument operational difficulties. Halley's coma would be traversed in less than two hours. Imaging on the nucleus would prove extremely difficult due to such limiting factors as attitude control errors, TV exposure time and uncertainty in where the nucleus is situated. Nevertheless, a 3 kw SEP stage atop the Titan IIIB/Centaur could deliver a maximum payload of 740 kg. Actually, SEP is not needed for this mission since the same launch vehicle could deliver a ballistic spacecraft of 600 kg.

Intermediate speed flythroughs in the range 30-20 km/sec are possible with flight times between 1.3 and 2.5 years. The arrival date is about 150 days before perihelion. SEP application could utilize the Titan IIID/Burner II launch vehicle and a 10 kw powerplant. This class of mission may also be assigned to ballistic spacecraft but the approach velocity would increase by about 5 to 10 km/sec. The advantage of SEP begins to be significant when the desired approach speed is under 20 km/sec. Employing the Titan IIID/Centaur and a 15 kw powerplant, the mission example shown has a flight duration of

3.5 years and arrives 130 days before perihelion with a fly-through speed of 18 km/sec.

The two examples of slow flythrough and rendezvous utilize the Jupiter gravity-assist flight mode which allows a substantial reduction in trajectory energy requirements. A 15 kw SEP system is adequate for these missions, but the Titan IIID(7)/Centaur launch vehicle is required. With a launch date in 1978, the slow flythrough encounter occurs 150 days before perihelion at a relative speed of 6.4 km/sec. It is important to point out that while the total mission duration is 7 years, the SEP system is used <u>only</u> for the Earth-Jupiter leg of the mission; propulsion on-time is 228 days.

The best rendezvous mission profile requires a launch in 1977 and a trip time of almost 8.5 years. Halley encounter occurs 50 days before perihelion at which time the spacecraft and comet velocities are exactly matched. Total propulsion on-time is 1326 days; 324 days on the Earth-Jupiter leg and 1002 days on the Jupiter-Halley leg. Figure S-2 illustrates the rendezvous trajectory profile.

In conclusion, this study has attempted to deliniate the trajectory possibilities and requirements from which mission planners may assess the preliminary feasibility of Halley missions and the potential role of solar electric propulsion. It must be admitted that an extremely attractive mission profile has not been found. That is to say, the easy high velocity flythrough missions may as well be performed ballistically, and the difficult rendezvous mission places rather severe requirements on SEP spacecraft design. To place the SEP rendezvous mission in perspective, it is noted that Halley rendezvous may be accomplished ballistically (with Jupiter swingby) but the requirements are a Saturn V/Centaur launch, a two-stage retro-propulsion system, and a flight time of 8 years. The most promising delivery mode is nuclear electric propulsion which can utilize the Titan IIID(7)/Centaur launch vehicle and has a flight time under 3 years. The decision as to which type of mission, if any, should be programmed lies with NASA mission planners. Suffice it to say that of all periodic comets accessible to space exploration, Halley's Comet is undeniably the most unique and interesting example. It is recommended that the results presented herein be used to initiate a further and more comprehensive investigation of mission feasibility from a practical design and cost standpoint.

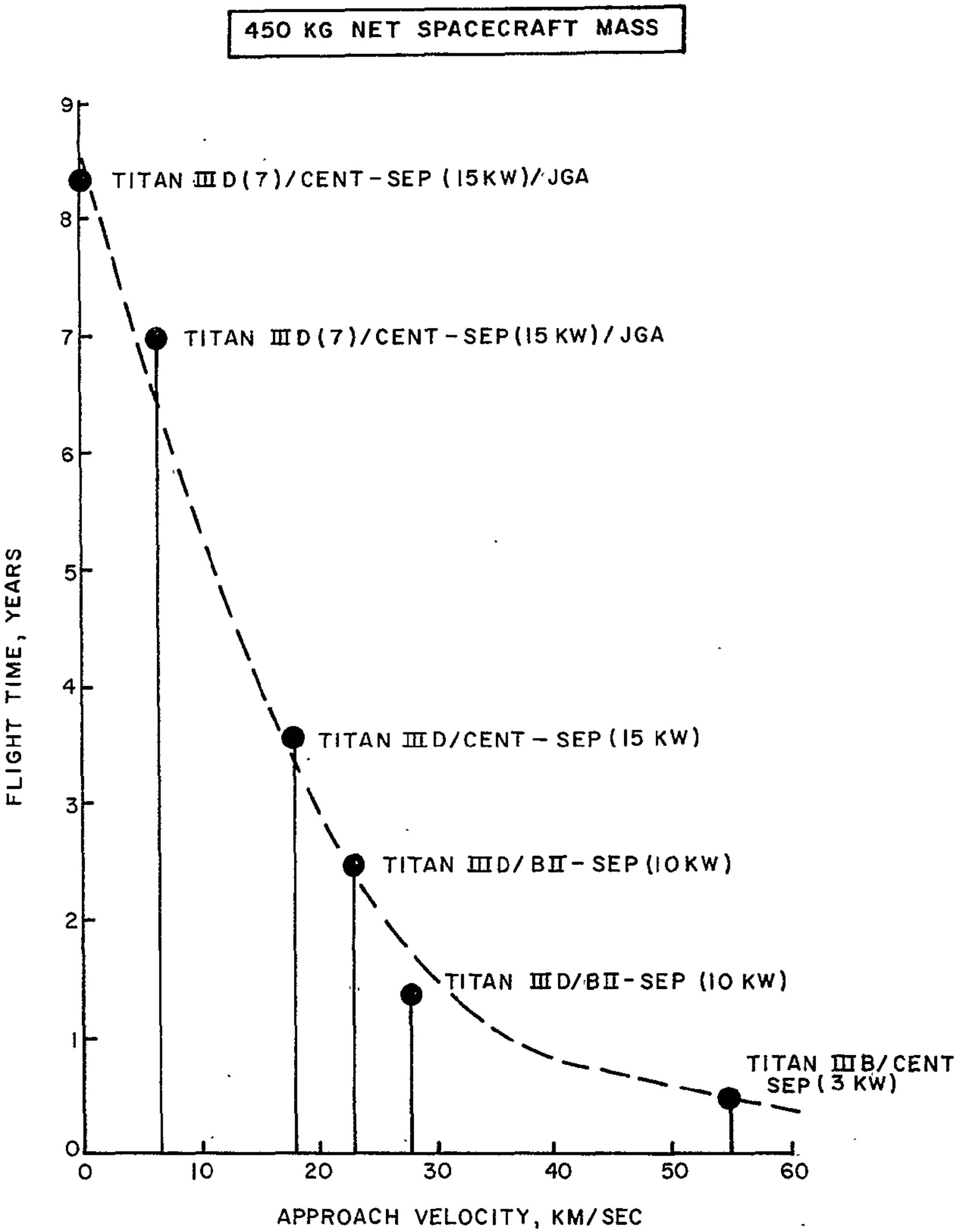

FIGURE. S-1. SUMMARY OF SOLAR ELECTRIC CAPABILITY FOR MISSIONS TO HALLEY'S COMET

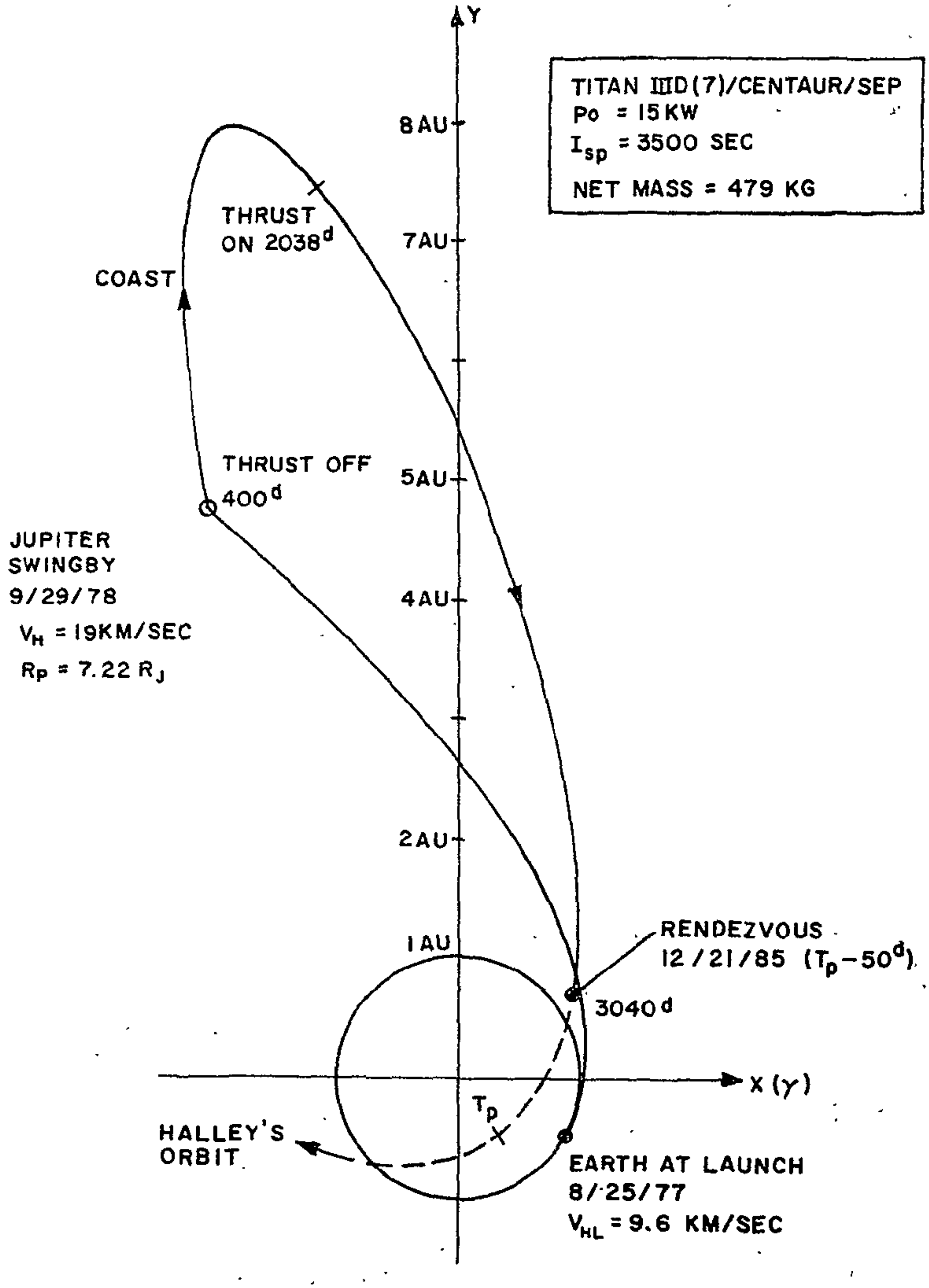

FIGURE S-2. SOLAR ELECTRIC RENDEZVOUS TRAJECTORY WITH JUPITER GRAVITY-ASSIST.

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HALLEY'S COMET FLYTHROUGH AND RENDEZVOUS MISSIONS VIA SOLAR ELECTRIC PROPULSION

1. INTRODUCTION

1.1 Study Background

The most outstanding comet mission from the standpoint of scientific and public interest is that of Halley's Comet which is due to return in 1985-86. Recognition of this rare and exciting opportunity has motivated several recent studies of the characteristics and requirements of the Halley mission (Michielsen 1968, Kruse and Fox 1968, and Friedlander et al. 1970). Primary emphasis has naturally been placed on the rendezvous mission which would allow the spacecraft to remain in the near vicinity of the comet for many months. A rendezvous with Halley is especially difficult because of the unique retrograde feature of its orbital motion.

In previous studies, two flight modes having sufficient energy capability to deliver a rendezvous spacecraft of at least 450 kg have been identified. The ballistic mode utilizes gravity-assist via a Jupiter swingby. A Saturn V/Centaur launch plus a two-stage retro-propulsion system is required, and the flight time is almost 8 years. The low-thrust mode utilizes a nuclear-electric spacecraft launched by the less costly Titan IIID(7)/Centaur, and requires a flight time of only 2.6 years. While the latter mode is clearly preferable, it is uncertain whether nuclear electric propulsion can be developed and made operational by 1983.

Solar electric propulsion (SEP) has been shown to be very attractive for rendezvous missions to a number of short-period comets. However, the SEP Halley rendezvous mission has received only cursory study (Friedlander 1970). The preliminary results, while not too encouraging, have been inconclusive.

Assuming the distinct possibility that Halley rendezvous will not be feasible or attractive but that the Halley mission itself is imperative, one may wish to consider the reduced goal of a flythrough mission. Flythrough is defined here to mean that the spacecraft's approach velocity relative to the comet is some finite, nonzero value. The time available for scientific experimentation in the cometary environment is an inverse function of the approach velocity. The spectrum of flythrough missions extends from the typical fast flyby mission (55 km/sec), to the "slow" flythrough (5-15 km/sec), and, in the limit, to the rendezvous mission (0 km/sec). In general, fast flythroughs are associated with short flight times and slow flythroughs with long flight times.

1.2 Study Objectives and Approach

This report presents the results of a study undertaken to determine the capability and characteristics of the SEP flight mode for performing missions to Halley's Comet. The objective is to provide mission planners with the necessary trajectory/payload data and system parameter tradeoffs in order to evaluate both the flythrough and rendezvous missions using solar electric propulsion. Both direct SEP flights and gravity-assisted SEP flights via a planet swingby (principally Jupiter) are investigated. Specific results will describe the SEP payload capability as a function of the following parameters: (1) flight time, (2) approach velocity, (3) arrival date, (4) launch vehicle, (5) solar array power rating, and (6) propulsion on-time.

In evaluating the SEP mission capability, the propulsion system parameters are assumed to have current technology values. Baseline values of 3500 seconds specific impulse, 30 kg/kw specific mass and 3 percent tankage factor are employed. The effect of variations in the baseline parameters can be obtained by using appropriate scaling relationships (Bartz and Friedlander 1971). Another study guideline is the use of launch vehicles in the Titan III family, e.g., the Titan IIID/Centaur. Arrival dates at Halley in the range 0-200 days before perihelion are emphasized in this study. The term payload is used synonomously with net spacecraft mass which is defined as the spacecraft science and engineering support subsystems; it does not include the SEP propulsion system. A particular mission will be considered potentially attractive if the payload delivered is 450 kg or more. A 450 kg spacecraft would have a science payload of about 65 kg, and is a useful guideline for assessing comet exploration capability (Friedlander and Wells, 1971).

Section 2 of this report describes the orbital characteristics of Halley's Comet and the Earth-based sighting conditions during the 1985-86 apparition. The method and conditions of the trajectory/payload analysis are discussed in Section 3. Section 4 presents results of the direct SEP flight mode for both flythrough and rendezvous missions. Similar results are given in Section 5 for the gravity-assisted SEP flight mode. Section 6 contains detailed trajectory profile data for several baseline mission selections representing a spectrum of Halley exploration capability.

2. ORBITAL AND SIGHTING CHARACTERISTICS OF HALLEY'S COMET

A better understanding of the Halley mission will be gained by first reviewing the comet's orbital motion and its predicted position-time history in the vicinity of the 1986 perihelion. In addition, the characteristics of comet observability with Earth-based telescopes are described. Earth-based sighting can play an important rôle in the mission operations in that: (1) an early recovery (first sighting) provides a redetermination of the comet's position-in-orbit and thus facilitates terminal guidance maneuvers, and (2) post-recovery observations may provide calibration data which are useful in correlating spacecraft science instruments.

2.1 Orbital Motion

The orbital period of Halley's Comet is approximately 76 years. The comet has been observed at numerous prior apparitions, but the 1910 apparition was particularly well observed using modern photographic techniques. A definitive orbit of Halley connecting the 1835 and 1910 apparitions has been published (Brady and Carpenter 1967). This orbit is based on a least-squares fit of about 2000 observations and accounts for the gravitational perturbations of all the planets. Table 2-1 lists the predicted orbital elements for the 1985-86 apparition. The osculation date* is November 21, 1985, which is 76 days before the gravitationally predicted perihelion date of

*Orbital elements vary in time due to the planetary perturbations. The osculation date is some arbitrary time (near perihelion in this case) at which specific values of the elements are stated.

TABLE 2-1

ORBITAL ELEMENTS OF HALLEY'S COMET

(Brady and Carpenter, 1967)

Osculation Date	t	1985 Nov. 21.0	(2446390.5)
Perihelion Date	T_p	1986 Feb. 5.3683	(2446466.8683)
		* 1986 Feb. 9.5	(2446471.0)
Semi-Major Axis	a	17.934279 a.u.	
Perihelion Distance	q	0.587108 a.u.	
Eccentricity	e	0.967263	
Inclination	i	162°.23827	mean ecliptic and equinox of 1950.0
Longitude of Node	Ω	58°.15363	
Argument of Perihelion	ω	111°.85711	

* This perihelion date accounts for estimated nongravitational effects and is used in the present study

February 5, 1986. The comet has a large eccentricity of 0.967 and a fairly close perihelion distance of 0.587 a.u. The aphelion distance is 34.7 a.u. which is beyond the orbit of Neptune. Halley is in a retrograde orbit having an inclination of 162 degrees.

Halley's Comet has long been suspected of being affected by nongravitational forces, probably due to mass ejection under the influence of solar radiation heating. Michielsen has undertaken a recent study of the nongravitational effect on perihelion date (Michielsen 1968). Starting with Brady's 1910 orbit, he integrated the equations of motion backward through seven prior apparitions to the year 1378. The osculating semi-major axis (or period) was adjusted as necessary at each successive apparition in order to accurately fit the recorded observations at these apparitions. The surprising result obtained by this process was that the amount of period adjustment (about 4 days) was essentially the same for all 2-apparition pairs. This is sufficiently strong evidence to expect that Halley has a secular (nongravitational) deceleration of 4 days per $(\text{period})^2$. Extrapolating this result with reasonable assurance yields February 9 as a better prediction of the 1986 perihelion date. This date is employed in the present trajectory study.

Figure 2-1 illustrates the ecliptic and out-of-plane projections of motion in the vicinity of perihelion. Also shown are the relative positions of the Earth and Halley at several time points. Figure 2-2 presents a detailed time history of geocentric distance and elongation (sun-earth-comet) angle. The two oppositions occur about 3 months before and 2 months after perihelion passage. The closest approaches are 0.6 and 0.45 a.u., respectively. At perihelion passage Halley will be near superior conjunction and, hence, not readily observable from the Earth (i.e., for visual observations).

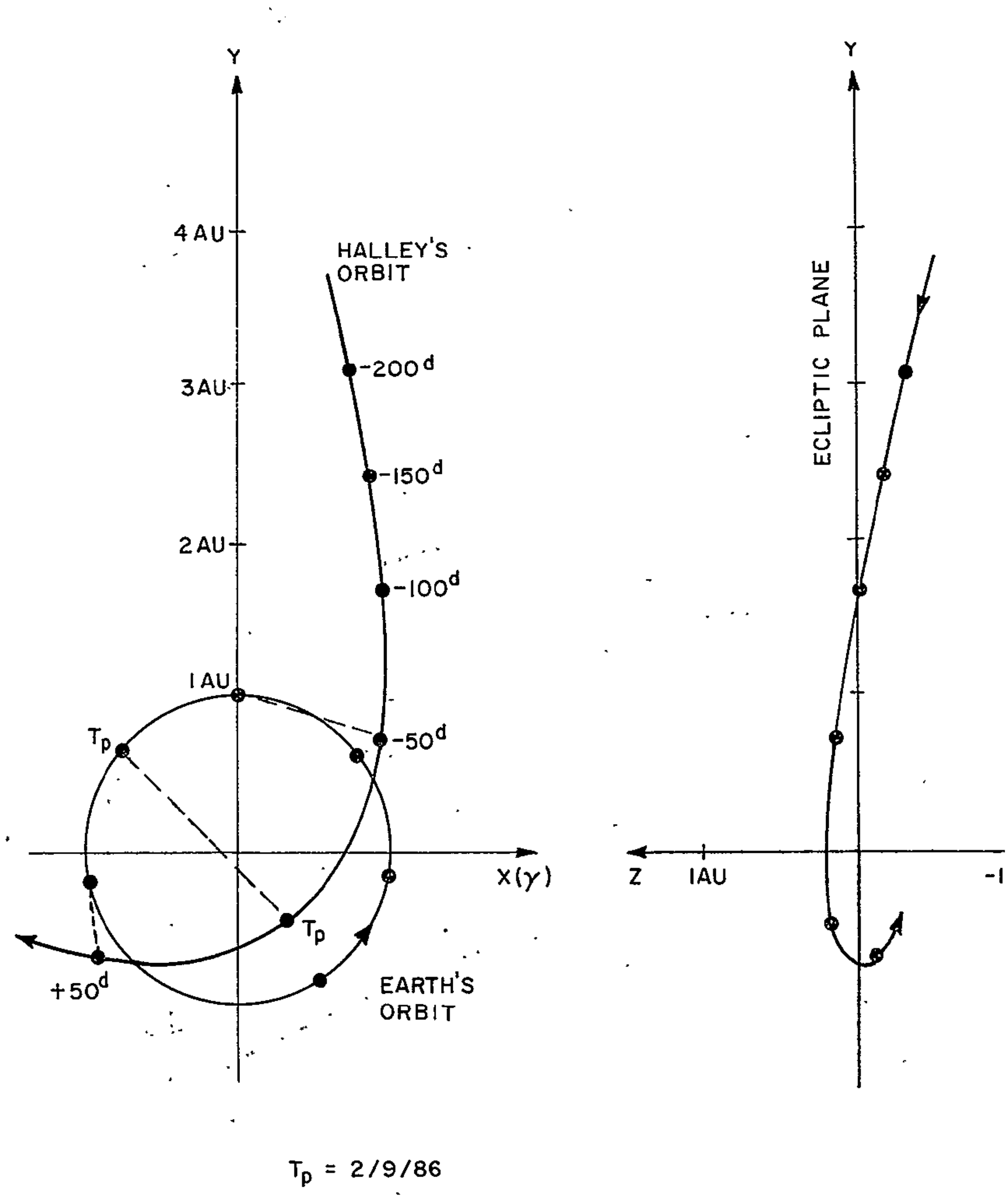

FIGURE 2-1. ORBIT OF HALLEY'S COMET NEAR PERIHELION

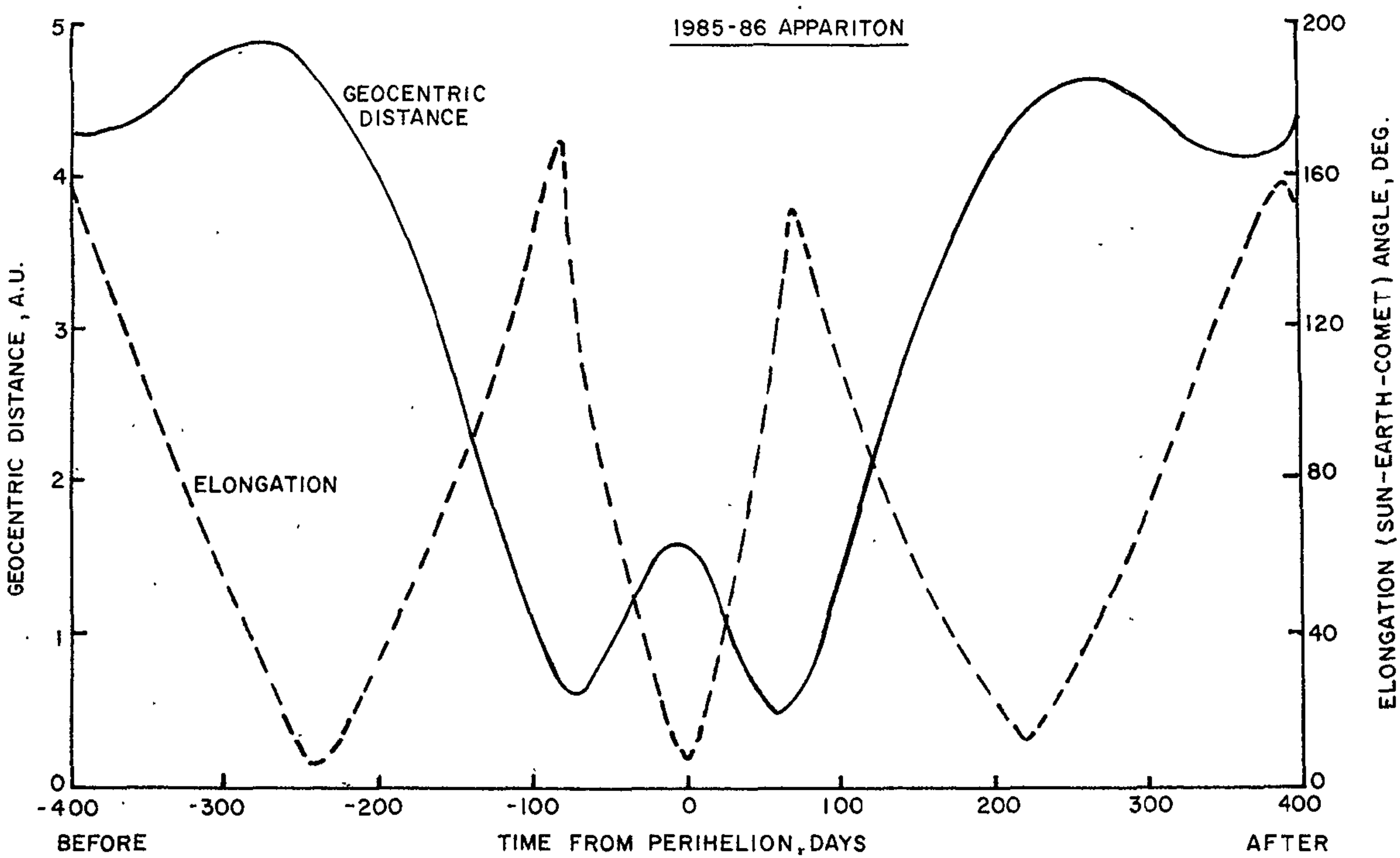

FIGURE 2-2. GEOCENTRIC DISTANCE AND ELONGATION ANGLE OF HALLEY'S COMET

2.2 Earth-Based Sighting Characteristics

The two measures of comet brightness are referred to as nuclear magnitude and total magnitude. Nuclear magnitude (or central condensation) is used to predict the stellar brightness of a comet for recovery and tracking purposes. It applies to the brightness of the comet nucleus only but is a fair approximation of total brightness when the comet is greater than 1 a.u. from the Sun and Earth. The nuclear magnitude of Halley's Comet is given by the following expression (Marsden 1970).

$$M_n = 8.5 + 5 \log_{10} \Delta + 10 \log_{10} r \qquad (1)$$

where Δ and r are the geocentric and heliocentic distances (a.u.), respectively. Total magnitude is a measure of the total brightness exhibited by the comet nucleus and coma and applies when the comet is active in the region of perihelion. Total magnitude is a somewhat less reliable prediction and generally is not valid above values of 12. The total magnitude of Halley's Comet is given by the expression (Marsden 1970)

$$M_t = 4 + 5 \log_{10} \Delta + 15 \log_{10} r \quad . \qquad (2)$$

Figure 2-3 shows the predicted time history of nuclear and total brightness for the period $\pm$ 400 days about perihelion. Using appropriate Earth-based telescopes, comet recovery usually occurs when the nuclear brightness is about 20th magnitude. Therefore, Halley should be recovered well before perihelion (perhaps in 1983). Spectroscopic and photometric measurements from Earth are usually obtained when the total brightness is greater than 12th magnitude. For Halley then, such measurements would be possible whenever the comet is observable during the period $\pm$ 140 days about perihelion.

The basic condition of observability is that the comet is visible in a dark sky. Specifically, the comet must be above the local horizon of the observing site when the Sun is at least 18° below the horizon. Figure 2-4 shows the daily period of observation for two representative observatory latitudes in the Northern and Southern Hemispheres. Photographic plate exposures of at least one hour are needed for recovery and tracking work. Halley's comet should be observable from northern latitudes during the two near-perihelion periods: $T_p - 180^d$ to $T_p - 30^d$ and $T_p + 60^d$ to $T_p + 120^d$. From southern latitudes the observable periods are: $T_p - 170^d$ to $T_p - 50^d$ and $T_p + 30^d$ to $T_p + 170^d$.

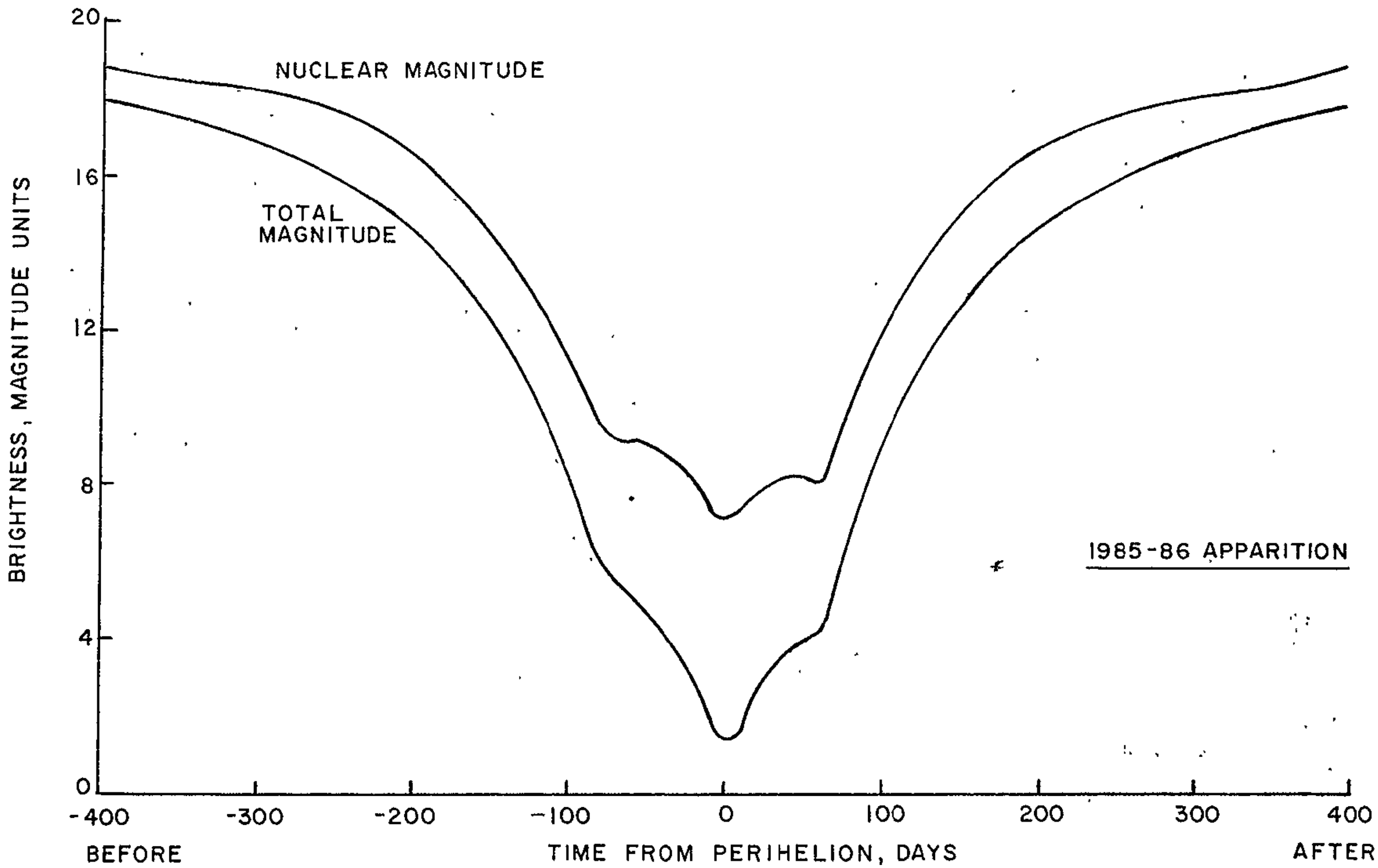

FIGURE 2-3. BRIGHTNESS OF HALLEY'S COMET

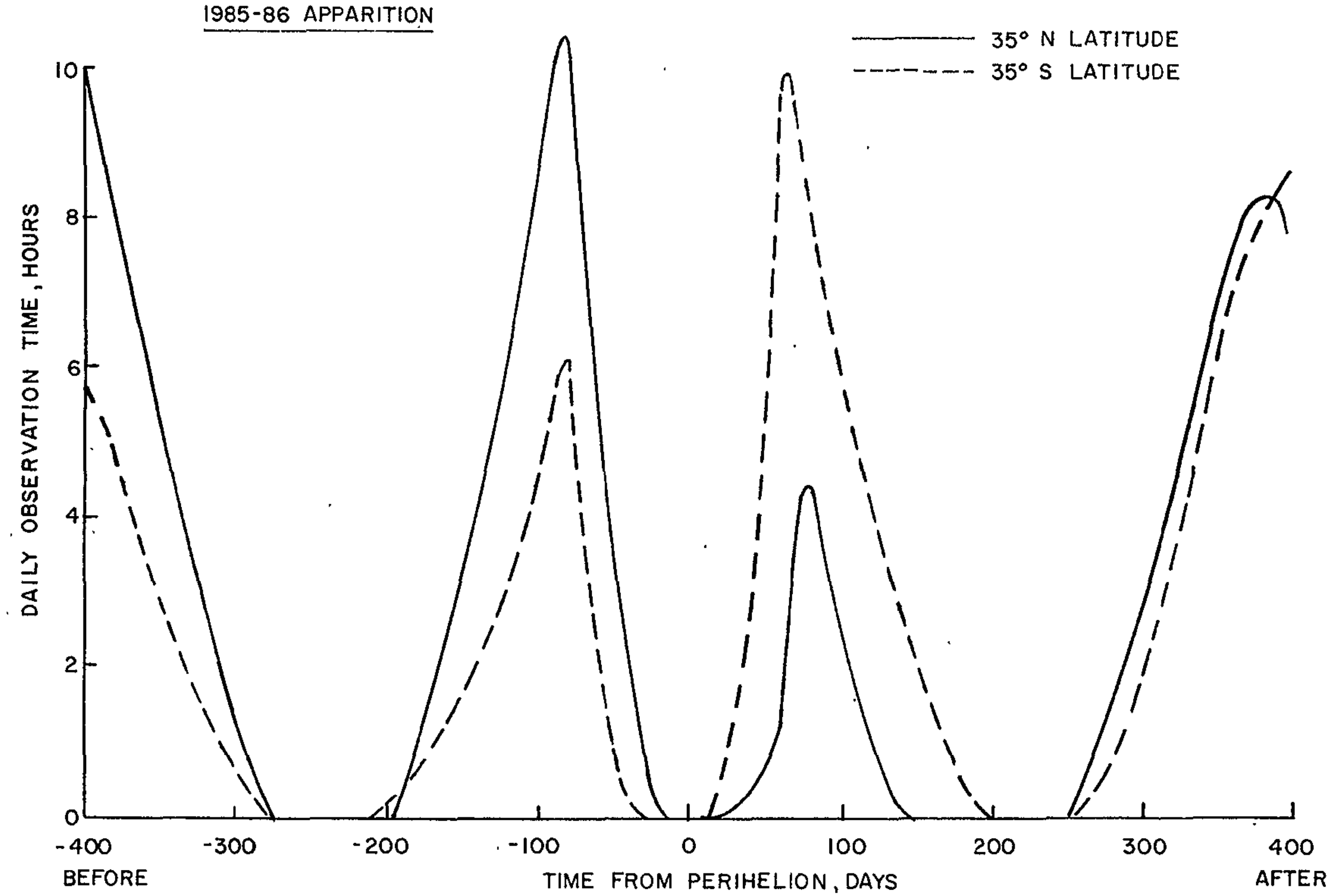

FIGURE 2-4. EARTH-BASED OBSERVATION PERIODS FOR HALLEY'S COMET

3. METHOD OF ANALYSIS

The trajectory model is three-dimensional and assumes an inverse-square solar gravitational field. The trajectory is shaped by the spacecraft velocity at Earth departure and the subsequent low-thrust propulsion phase. Optimization of low-thrust trajectories takes place on two, basically separate, levels. First, "kinematic" optimization determines the best launch and arrival positions (dates) and thrust vector program for specified values of flight time and terminal hyperbolic velocities. These results are fairly insensitive to the propulsion system parameters. Second, "payload" optimization determines the best values of solar array power, specific impulse, and launch velocity for a specified launch vehicle. More generally, the net spacecraft mass may be maximized for constrained values of power and specific impulse. This two-level optimization procedure has been followed in the present analysis.

SEP trajectory requirements have been generated using the computer program CHEBYTOP (Hahn 1969). The basic trajectory performance index is the "energy parameter" J_{VT} which is defined by the integral expression

$$J_{VT} = \int_{o}^{t_f} \frac{a^2(t)}{G(R)} \, dt \; ; \quad G(R) = \frac{P\,[R(t)]}{P_o} \qquad (3)$$

where a(t) is the thrust acceleration magnitude, G(R) is the normalized solar power (relative to 1 a.u.) available to the thrust subsystem, and t_f is the flight time. Figure 3-1 illustrates the solar power profile assumed in the present analysis*. The computational algorithm used in CHEBYTOP minimizes J_{VT} subject to the trajectory boundary conditions. This basic result, termed the Variable Thrust Solution, assumes a freely variable thrust acceleration magnitude and direction. In practice, acceleration magnitude is constrained by the constant specific impulse operation of current ion thrusters. This constraint is stated by the following equations:

$$a(t) = \frac{a_o\, G(R)\ \sigma(t)}{1 - \frac{a_o}{c} \int_o^t G(R)\sigma(t)\,dt} \tag{4}$$

$$a_o = \frac{2\eta}{c} \quad (P_o/m_o) \tag{5}$$

$$c = 9.806\ I_{sp} \tag{6}$$

$$\sigma(t) = \begin{cases} 1 \text{ during propulsion periods} \\ o \text{ during coast periods} \end{cases} \tag{7}$$

where a_o is the initial thrust acceleration at 1 a.u., c is the constant exhaust velocity of the thrusters, η is the propulsion system efficiency, and m_o is the initial spacecraft mass. The denominator in Equation (4) is recognized as the instantaneous mass fraction $m(t)/m_o$.

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* This power profile is built in the CHEBYTOP program, and derives from an analysis by Strack (Strack 1967). Recent experimental studies at JPL indicate that Strack's power curve may be overly optimistic by about 10 percent at large R.

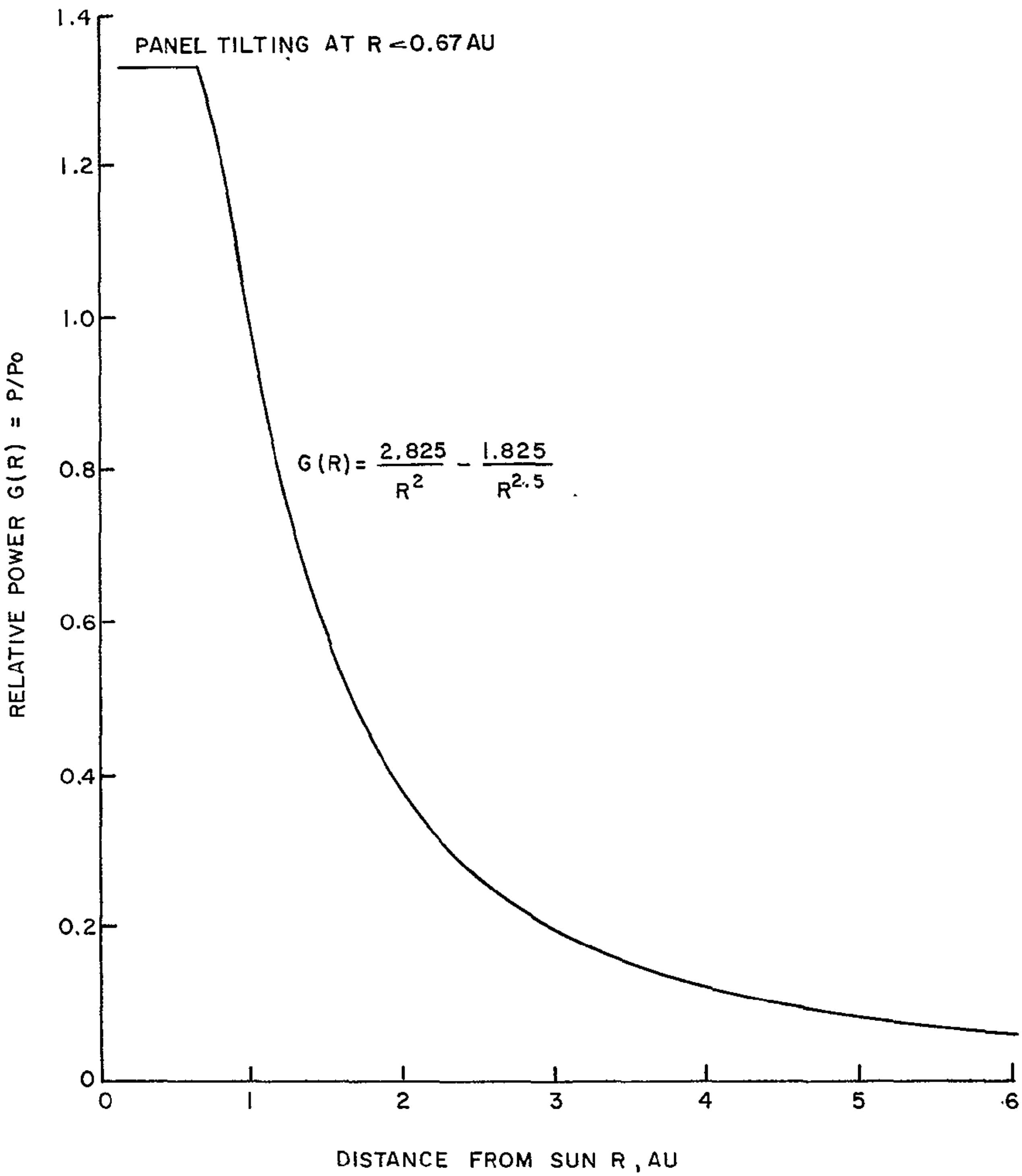

FIGURE 3-1. SOLAR CELL POWER PROFILE

CHEBYTOP also generates the Constant Specific Impulse Solution for specified values of the propulsion system parameters (a_o, I_{sp}). The corresponding energy parameter J is typically 5 to 15 percent higher than the ideal solution J_{VT}. All payload data presented in this report are based on the Constant Specific Impulse Solution.

Initial spacecraft mass is equivalent to the injected mass of the launch vehicle. Figure 3-2 shows the performance of three Titan-class vehicles assumed in the present analysis; Titan IIID/Burner II, Titan IIID/Centaur and Titan IIID(7)/Centaur. Only the Titan IIID/Centaur is an actual programmed vehicle and, hence, will be taken as our baseline choice provided sufficient net mass capability is available. However, it will be shown subsequently that the smaller Titan IIID/Burner II is adequate for certain low-energy Halley flythrough missions, and the larger Titan IIID(7)/Centaur is required for the high-energy Halley rendezvous missions. The hyperbolic launch velocity (V_{HL}) range of interest for missions to Halley is 4-7 km/sec for the Titan IIID/Burner II and 7-10 km/sec for the two Titan/Centaur vehicles. Typically, then, m_o will lie in the range 750-3000 kg.

The "link" between the trajectory kinematics, launch vehicle and SEP propulsion system is given by the following expression

$$\frac{m_n}{m_o} = 1 - \frac{1 + k_p}{1 + ca_o/J} - \alpha \left(\frac{ca_o}{2\eta} \right) \qquad (8)$$

where m_n is the net spacecraft mass, k_p is the propellant tankage fraction, and α is the propulsion system specific mass. This expression is graphed in Figure 3-3 for the following

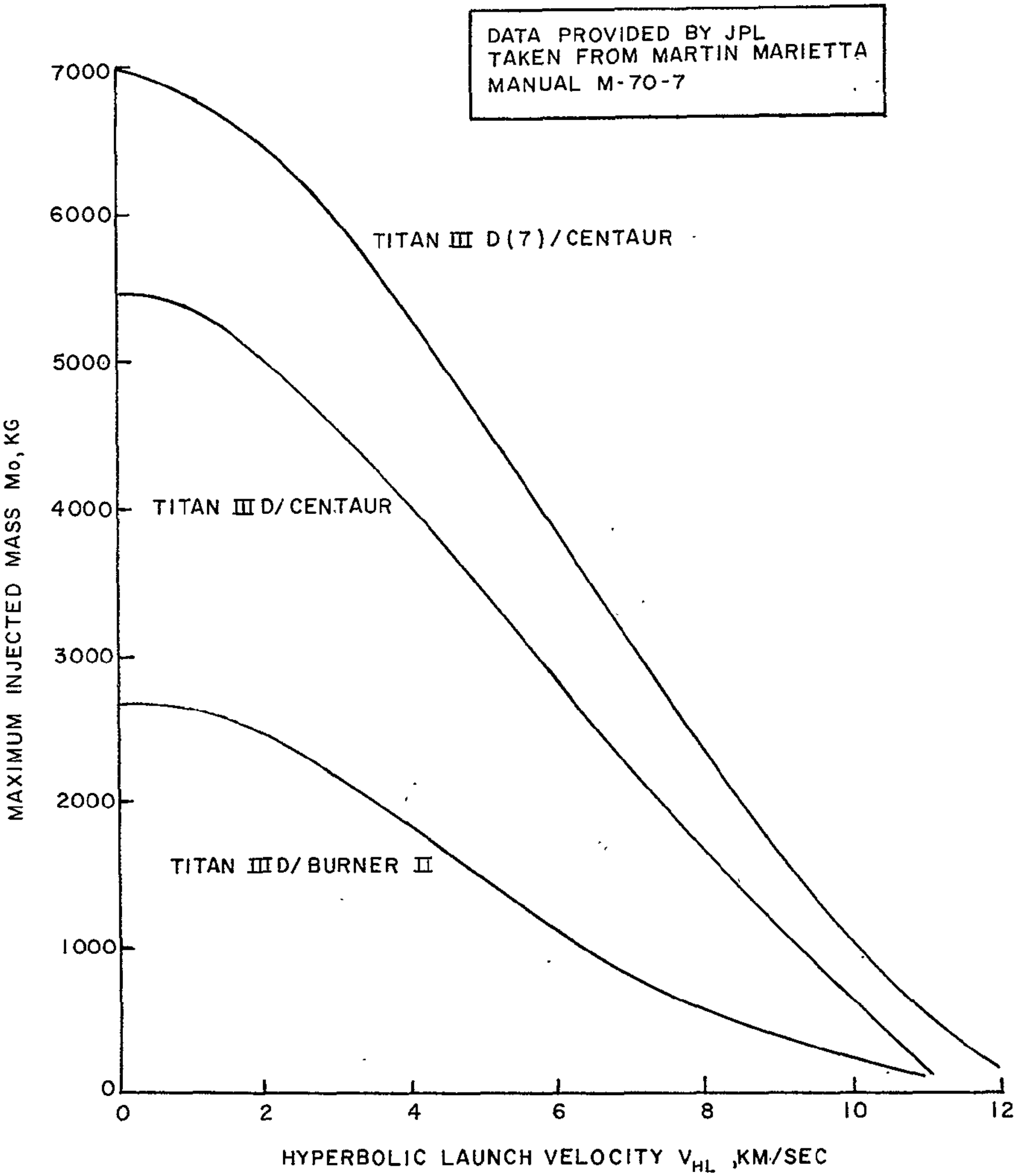

FIGURE 3-2. LAUNCH VEHICLE PERFORMANCE CURVES

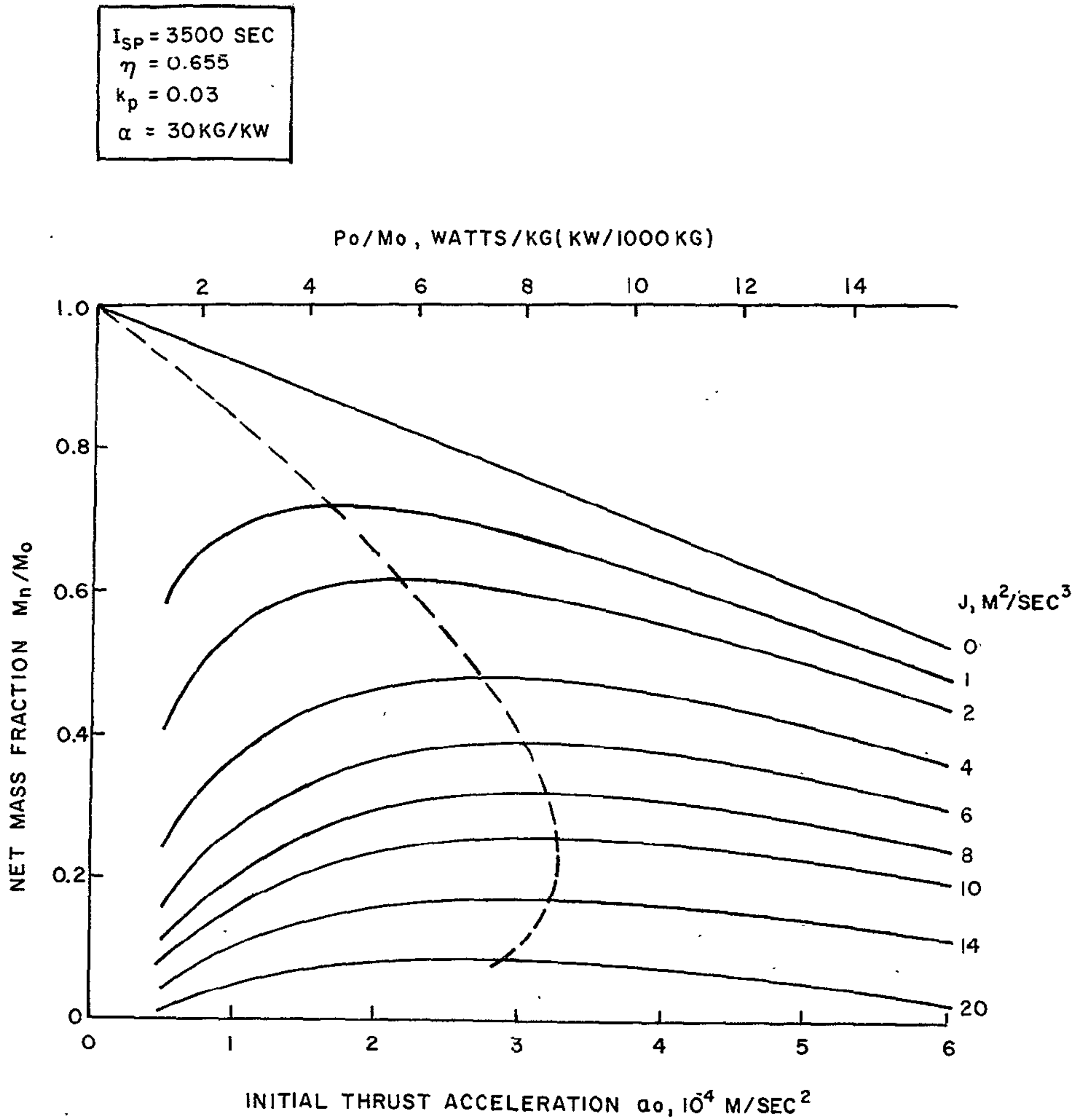

FIGURE 3-3. RELATION BETWEEN SEP TRAJECTORY KINEMATIC REQUIREMENTS AND NET MASS CAPABILITY

baseline parameter values employed throughout this study:

$$I_{sp} = 3500 \text{ sec}$$

$$\eta = 0.655$$

$$\alpha = 30 \text{ kg/kw}$$

$$k_p = 0.03 \quad .$$

The figure indicates that there is an optimum a_o for a given value of J (broken line locus). Actually, though, J is a function of a_o (Equations (3) and (4)), and both parameters are mission-dependent in the Constant Specific Impulse Solution. In general, the mission kinematic conditions require that a_o be greater than a certain minimum value in order to accomplish the mission in a specified flight time. This minimum a_o is associated with an all-propulsion flight path, i.e., no coast periods. The missions to Halley's Comet that will be described have a typical a_o requirement in the range 4×10^{-4} to 6×10^{-4} m/sec^2. Values of J should be less than about 10 m^2/sec^3 if viable payloads and practical size powerplants are to be achieved with Titan/Centaur vehicles. For example, taking P_o/m_o = 14 kw per 1000 kg (a_o = 5.34×10^{-4} m/sec^2) as a representative value, a net mass of 436 kg is obtained for values of J = 10 m^2/sec^3, P_o = 28 kw and m_o = 2000 kg. If the powerplant is constrained to 14 kw (m_o = 1000 kg), then the J requirement must be less than 3 m^2/sec^3 for the same 436 kg net mass. These examples are given to simply indicate that attractive SEP missions to Halley's Comet will typically have a J requirement in the range 3-10 m^2/sec^3.

In summary, the two steps in the trajectory/payload analysis are as follows:

1. For specified approach velocity conditions and time of flight, Variable Thrust Solutions are generated to identify the optimum launch and arrival dates corresponding to minimum J_{VT}. Alternatively, the arrival date may be fixed and the optimum launch date (flight time) is found. These solutions are obtained for a single, but representative value of hyperbolic launch velocity. Experience has shown that the optimum geometry does not change significantly over the V_{HL} range of interest.

2. Given the optimum or near-optimum launch/arrival geometry, Constant Specific Impulse Solutions are generated for a range of V_{HL} and P_o/m_o values. The resulting payload or net mass data is thus normalized to an initial spacecraft mass of 1000 kg. The power input and net mass capability for any given launch vehicle is then simply scaled using the launch vehicle performance data shown in Figure 3-2. This method avoids unnecessary trajectory recomputation and is accurate provided that a sufficiently close grid of P_o/m_o is employed.

The above procedure is also followed for the gravity-assisted flight mode. In this case the two trajectory legs are computed separately with appropriate matching of the hyperbolic velocity magnitude at the swingby planet. Swingby distance (r_p) is found from the scalar product of the inbound and outbound velocity vectors which result from the separate solutions:

$$\underline{V}_{Hi} \cdot \underline{V}_{Ho} = \cos \psi \tag{9}$$

$$1 + \frac{r_p V_H^2}{\mu} = \frac{1}{\sin (\psi/2)} \tag{10}$$

where ψ is the deflection angle of the hyperbolic asymptotes, V_H is the hyperbolic velocity magnitude, and μ is the planet's gravitational constant. Overall net mass capability is found by appropriate matching of the individual-leg performance functions m_n/m_o versus P_o/m_o.

4. DIRECT FLIGHT MODE RESULTS

The direct flight mode refers to the usual single-target mission; in this case, a direct trajectory between Earth and Halley's Comet. Flythrough and rendezvous missions will be discussed separately because they are basically different mission concepts - at least for planning purposes.* Of course, in a kinematic sense, rendezvous is simply the limiting case of a slow flythrough. The results presented in this section describe the SEP trajectory characteristics and payload capability, and show the effect of such mission design parameters as approach velocity, arrival date, flight time, launch vehicle selection and SEP power rating.

4.1 Flythrough Missions

In this mission concept the spacecraft approaches and passes through the cometary environment at some finite (nonzero) relative velocity. The velocity direction is essentially constant during the time interval of interest. Therefore, the flythrough region is necessarily limited by the trajectory kinematics and the choice of the nominal aim point (miss distance) vector. For example, the spacecraft may be targeted either to pass in front of Halley on the sunward side, or through the tail region, but not both. This situation is illustrated in Figure 4-1.

In terms of velocity magnitude alone, the spectrum of possible flythrough conditions is quite broad in extent. At

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* An assessment of the comparative science value of flythrough versus rendezvous or the detailed mission analysis of each mode is beyond the scope of the present study.

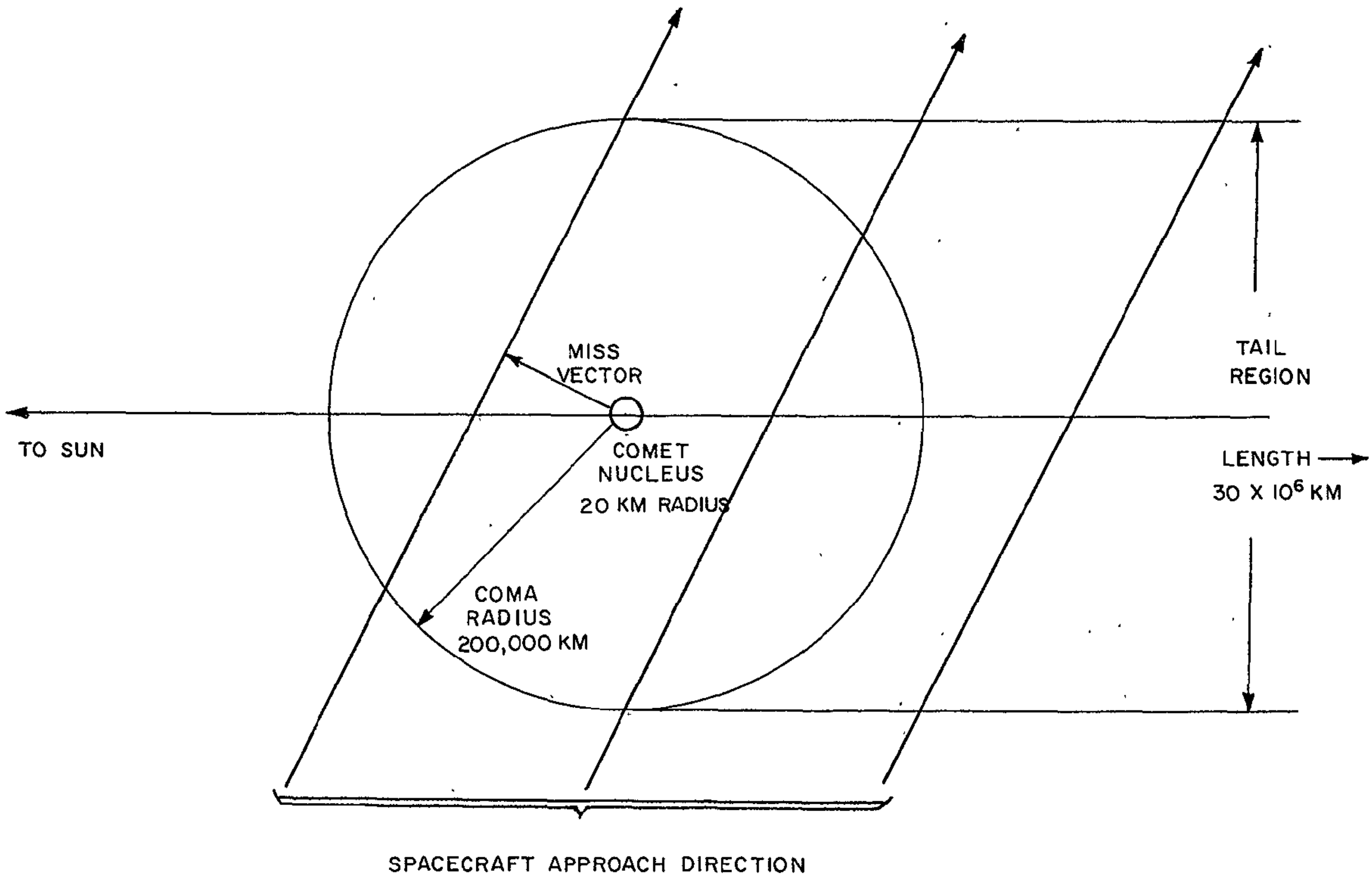

FIGURE 4-1. ILLUSTRATION OF POSSIBLE APPROACH PATHS ON HALLEY FLYTHROUGH

one extreme, there is the very fast flythrough having an approach velocity in excess of 70 km/sec. At the other extreme, there is the slow flythrough at velocities in the range 5-15 km/sec. It is understood, a priori, from trajectory considerations that the slow flythrough missions will require much longer flight times.

Unlike ballistic spacecraft, the low-thrust SEP spacecraft has considerable flexibility in controlling the approach velocity magnitude V_{HP}. To exemplify this point suppose that both the launch date and arrival date are specified. Two types of flythrough missions may be defined as follows:

1. <u>Unconstrained flythrough</u> -- V_{HP} is not specified. This is an "optimum" flythrough in that the energy parameter J is minimized (maximum net mass). The value of V_{HP} is a result of the solution.

2. <u>Constrained flythrough</u> -- V_{HP} is specified, typically at some value less than the unconstrained flythrough velocity. The energy parameter J will be larger in this case since an additional velocity change is needed to slow down the spacecraft.

The usual procedure followed in the trajectory analysis was to first examine the short flight time - high velocity missions and then gradually work towards the long flight time - low velocity missions. At each step, the unconstrained flythrough conditions were determined first. Presentation of results will generally follow this sequence.

Figure 4-2 shows the "variable thrust" J requirements for fast trips as a function of Earth launch date for six arrival dates between 50 and 100 days before perihelion. The flight time range is 120 to 250 days with launches between February and August 1985. These missions are unconstrained fly-throughs having approach velocities in the range 46 to 66 km/sec. The optimum (minimum J) flight geometry is immediately apparent from the figure -- it is the 170 day trip launched on J. D. 2446241 (6/24/85) and arriving 60 days before perihelion. The corresponding approach velocity is 55.6 km/sec. Actually, the value of J is so small that this mission is nearly ballistic. In fact, the optimum ballistic flight has the same launch/arrival dates, a hyperbolic launch velocity of 3.997 km/sec, and an approach velocity of 55.4 km/sec.

The payload capability for this mission is shown in Figure 4-3 assuming smaller launch vehicles than were discussed previously. For the ballistic missions ($P_o = 0$), the Titan IIIB/Centaur and TAT/Delta vehicles can deliver payloads of 600 kg and 331 kg, respectively. A 3 kw SEP spacecraft atop the Titan IIIB/Centaur can increase the payload to 740 kg, but such a large payload is probably not needed. Hence, one could fairly conclude that the 170 day fast flythrough mission is easier to accomplish ballistically.

The effect of constraining the approach velocity to lower values on the 170 day mission is shown by the table below:

V_{HL}	V_{HP}	J_{VT}
4 km/sec	55.6 km/sec	0.002 m^2/sec^3
	50	4.977
	45	18.901
	40	44.290

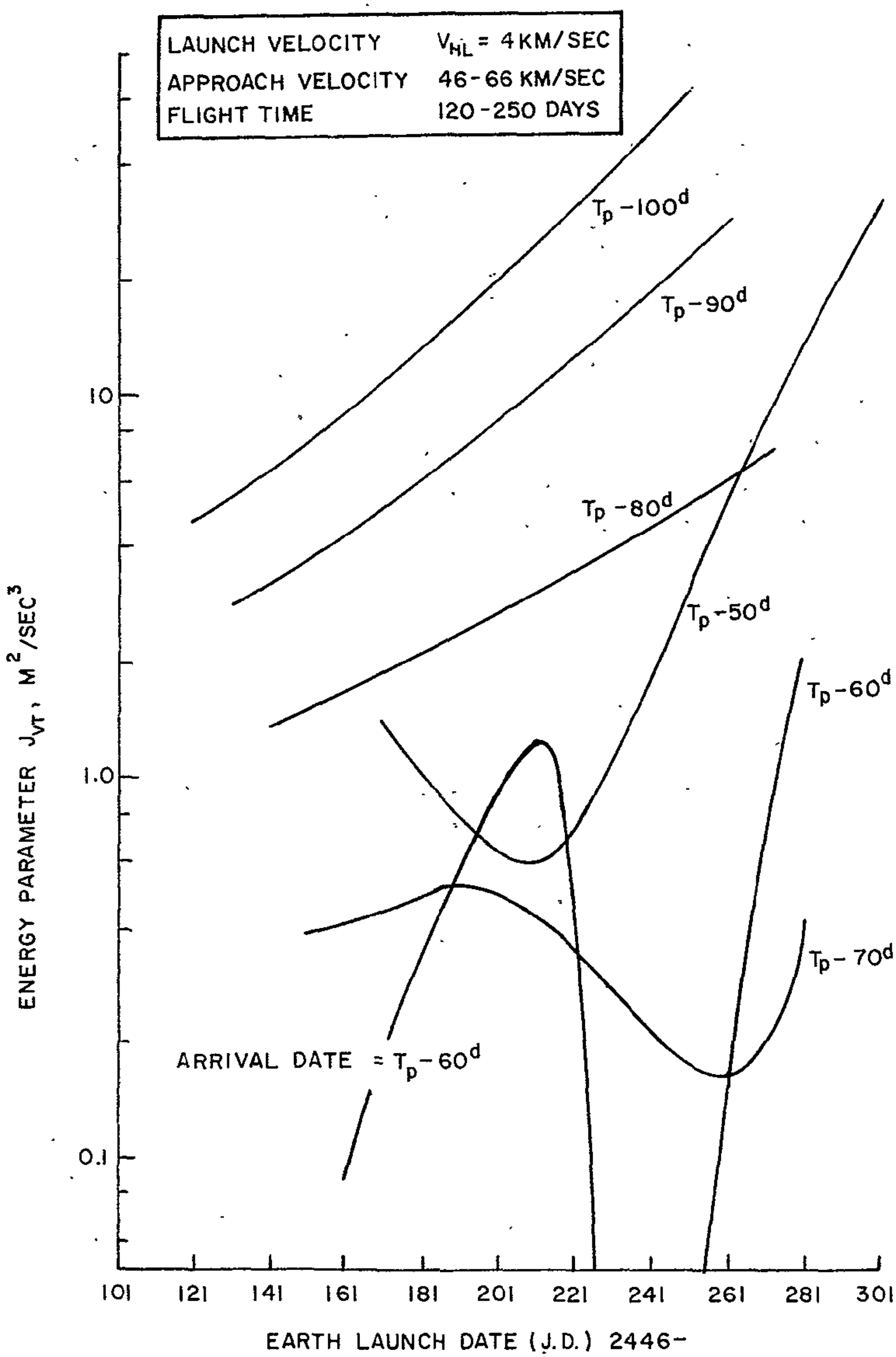

FIGURE 4-2. TRAJECTORY ENERGY REQUIREMENTS FOR SHORT FLIGHT TIME MISSIONS TO HALLEY'S COMET, UNCONSTRAINED FLYTHROUGH MODE.

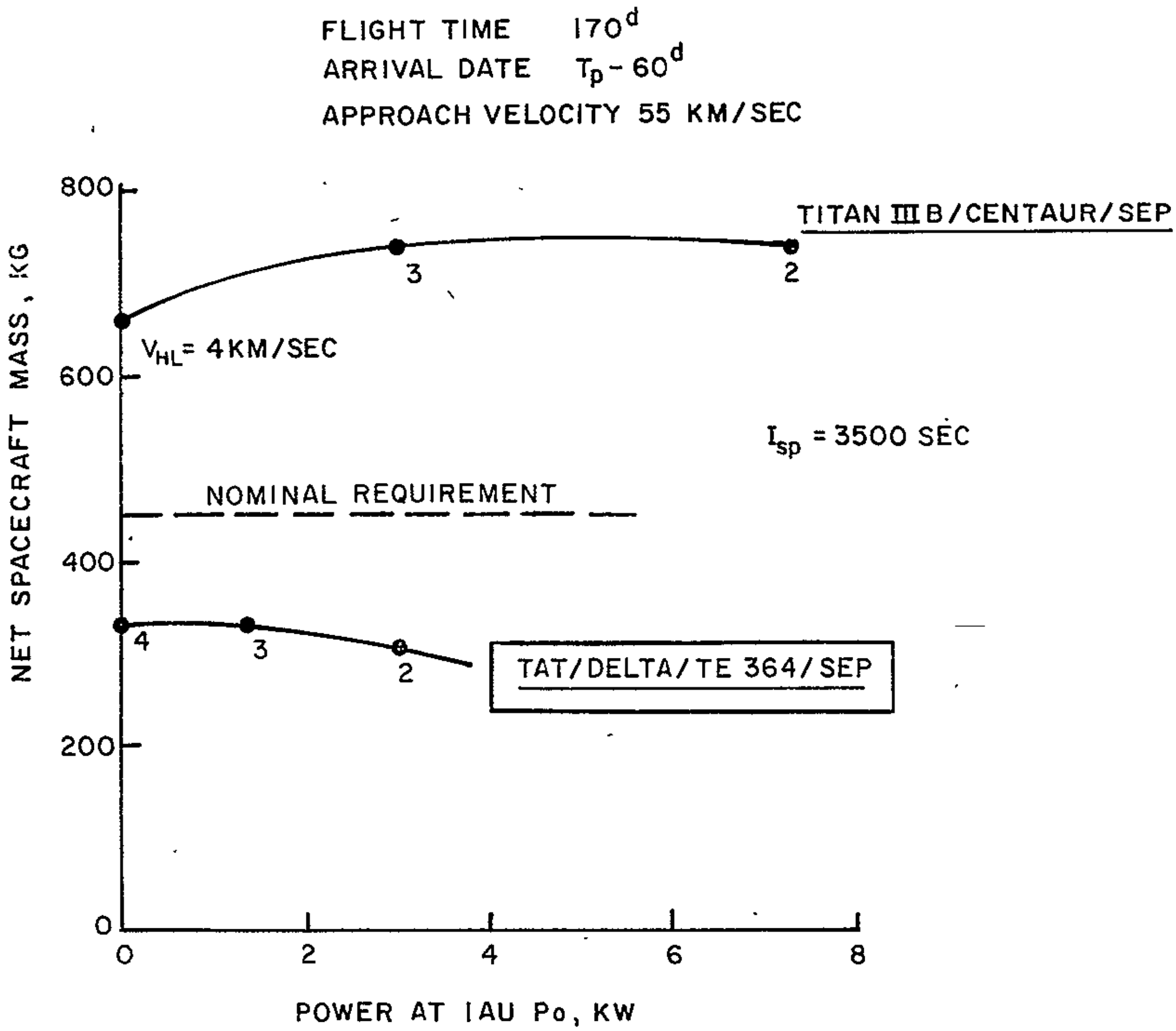

FIGURE 4-3. SOLAR ELECTRIC CAPABILITY FOR HIGH VELOCITY HALLEY FLYTHROUGH MISSION.

The rapidly increasing J requirement would indicate a vanishing payload capability at V_{HP} between 50 and 45 km/sec. A simple approximate* payload calculation can be made as follows: take the case V_{HP} = 50 km/sec and increase the J requirement by 12 percent to account for the constant specific impulse solution. Referring to Figure 3-3, the net mass fraction is 0.37 at J = 6 m^2/sec^3 and a_o = 4 x 10^{-4} m/sec^2. The Titan IIIB/Centaur injected mass at V_{HL} = 4 km/sec is 660 kg. Therefore, the SEP net mass capability is approximately 245 kg at a power input of about 7 kw. Since the difference between 55 and 50 km/sec approach velocity is not very significant, it is still concluded that the fast trip to Halley is in the ballistic domain.

Trajectory characteristics of the unconstrained fly-through mode are listed in Table 4-1 for a flight time range from 170 to 2600 days. The launch dates (or arrival dates) are near-optimum for each flight time. The general trend shown as flight time increases is an earlier arrival date, a lower approach velocity and a higher J requirement. However, even for the very long 2600 day trip, the unconstrained approach velocity is still fairly high at 18 km/sec. Figure 4-4 illustrates the trajectory profiles of the 170, 500 and 900 day missions.

A further explanation of the trajectory characteristics may be given by considering a particular flight time subclass such as 900 days. Figure 4-5 shows the energy parameter and approach velocity as a function of the arrival date

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* Payload data presented in figures and tables in this report are not approximated in this fashion. Rather, they result from actual computation of the Constant Specific Impulse Solution.

TABLE 4-1

SOLAR ELECTRIC TRAJECTORY CHARACTERISTICS

OF UNCONSTRAINED FLYTHROUGH MISSION TO HALLEY'S COMET

FLIGHT TIME (DAYS)	LAUNCH DATE	ARRIVAL DATE	V_{HP} (KM/SEC)	J_{VT}* (M^2/SEC^3)
170	6/24/85	$T_p - 60^d$	55	0.002
500	5/10/84	$T_p - 140^d$	34	0.67
900	3/17/83	$T_p - 160^d$	31	1.18
1200	3/22/82	$T_p - 220^d$	25	2.00
2300	3/18/79	$T_p - 220^d$	22	3.29
2600	4/12/78	$T_p - 260^d$	18	4.17

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* $V_{HL} = 4$ km/sec

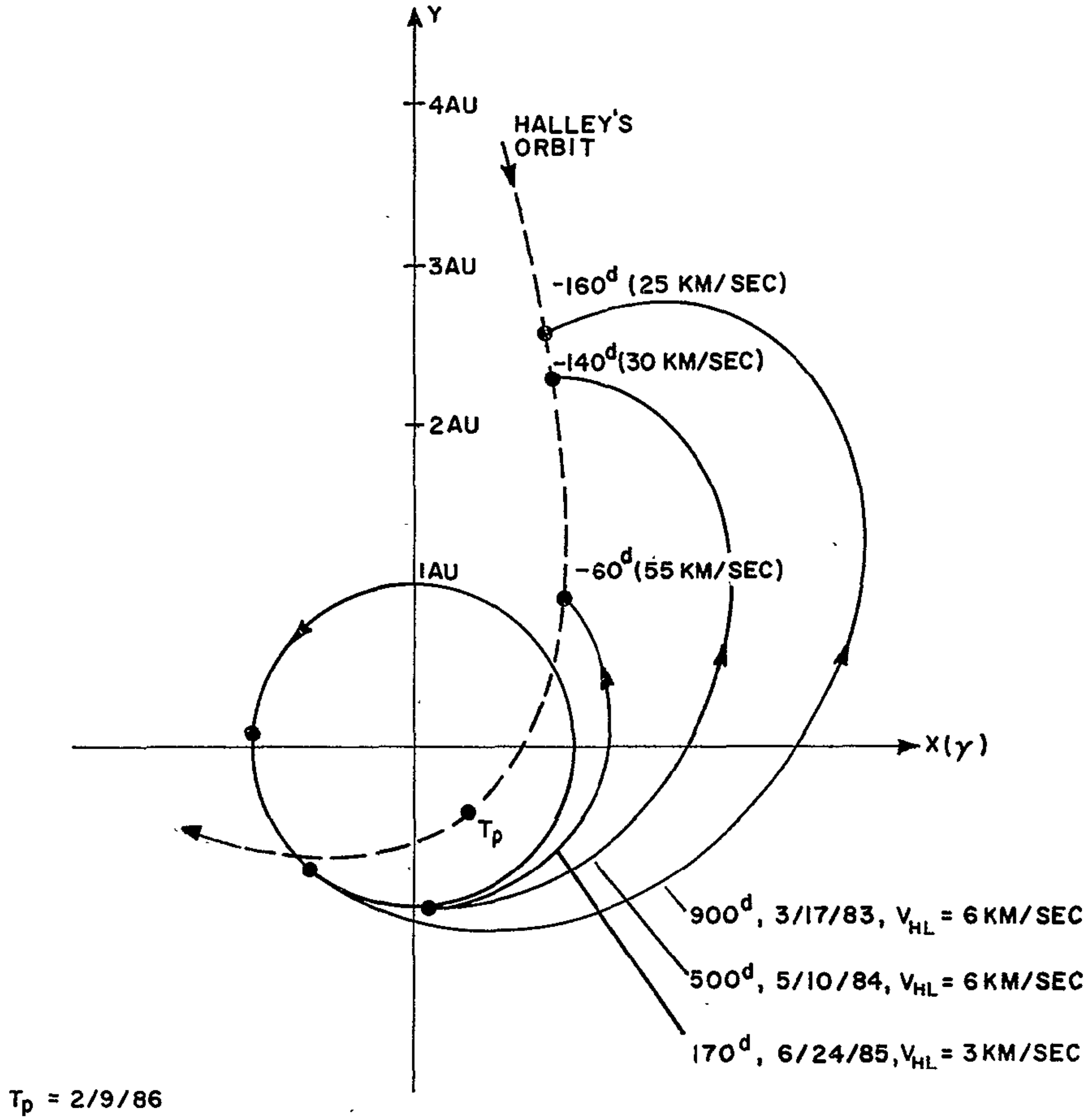

FIGURE 4-4. SOLAR ELECTRIC TRAJECTORIES TO HALLEY'S COMET, 170, 500, AND 900 DAY FLYTHROUGH MISSIONS.

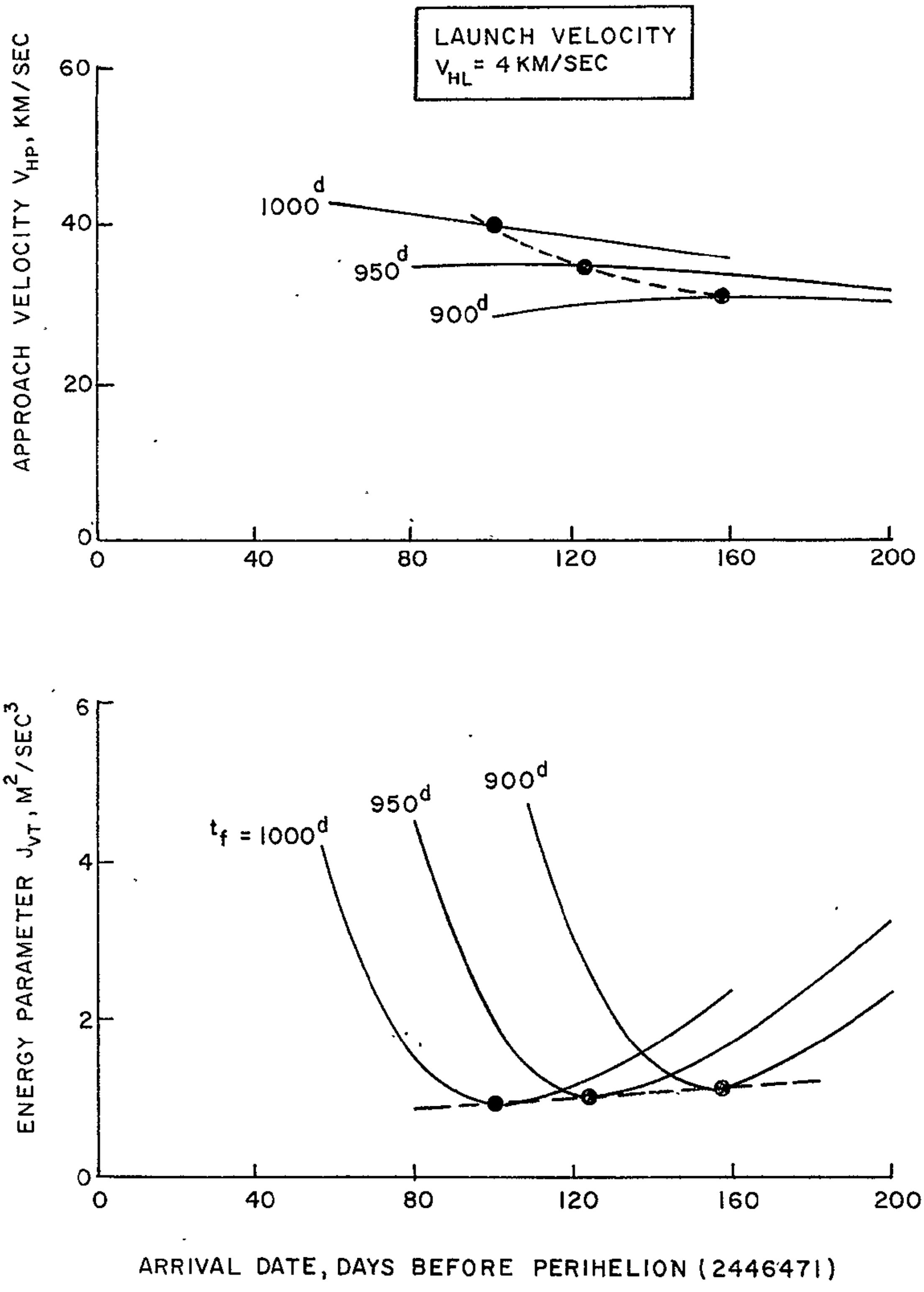

FIGURE 4-5. ILLUSTRATION OF FLIGHT TIME/ARRIVAL DATE CHARACTERISTICS FOR HALLEY'S COMET FLYTHROUGH (UNCONSTRAINED)

for flight times of 900, 950, and 1000 days. This subclass has preferred arrival points from 160 to 100 days before perihelion. One notes the corresponding shift of optimum arrival date with flight time. As perihelion is approached the J requirement decreases while the approach velocity increases. Similar characteristics (but different J and V_{HP} levels) occur for other trajectory subclasses launched approximately one year earlier or later, e.g., 500 - 600 day flights.

Figure 4-6 shows the sensitivity to hyperbolic velocities for the 900 day flight. For the unconstrained flythrough, J decreases with increasing launch velocity but the approach velocity remains fairly constant at about 30 km/sec. It is then seen that the J requirement increases rapidly as the approach velocity is constrained below 30 km/sec. Taking V_{HL} = 9 km/sec as representative of off-optimum (low power) SEP design on the Titan IIID/Centaur, the lowest practical approach velocity for the 900 day flight would be about 20 km/sec.

Solar electric payload capability for the 900 day unconstrained flythrough is shown in Figure 4-7. The Titan IIIB/Centaur launch vehicle would require an SEP powerplant of 4 kw to deliver the nominal 450 kg net mass. The Titan IIID/Burner II/SEP is rather ill-matched to this mission in that much greater payloads than needed could be delivered with powerplants in the 2-5 kw range. As a point of comparison, a 670 kg ballistic spacecraft could be launched by the Titan IIID/Burner II.

Figures 4-8 through 4-12 present payload results of the constrained flythrough mission analysis. The range of approach velocity is 30 to 5 km/sec for flight times between 500 and 2600 days. Optimum SEP power for the Titan IIID/Burner II is in the 20-30 kw region. This launch vehicle selection is appropriate for the 500 and 900 day missions; the minimum approach

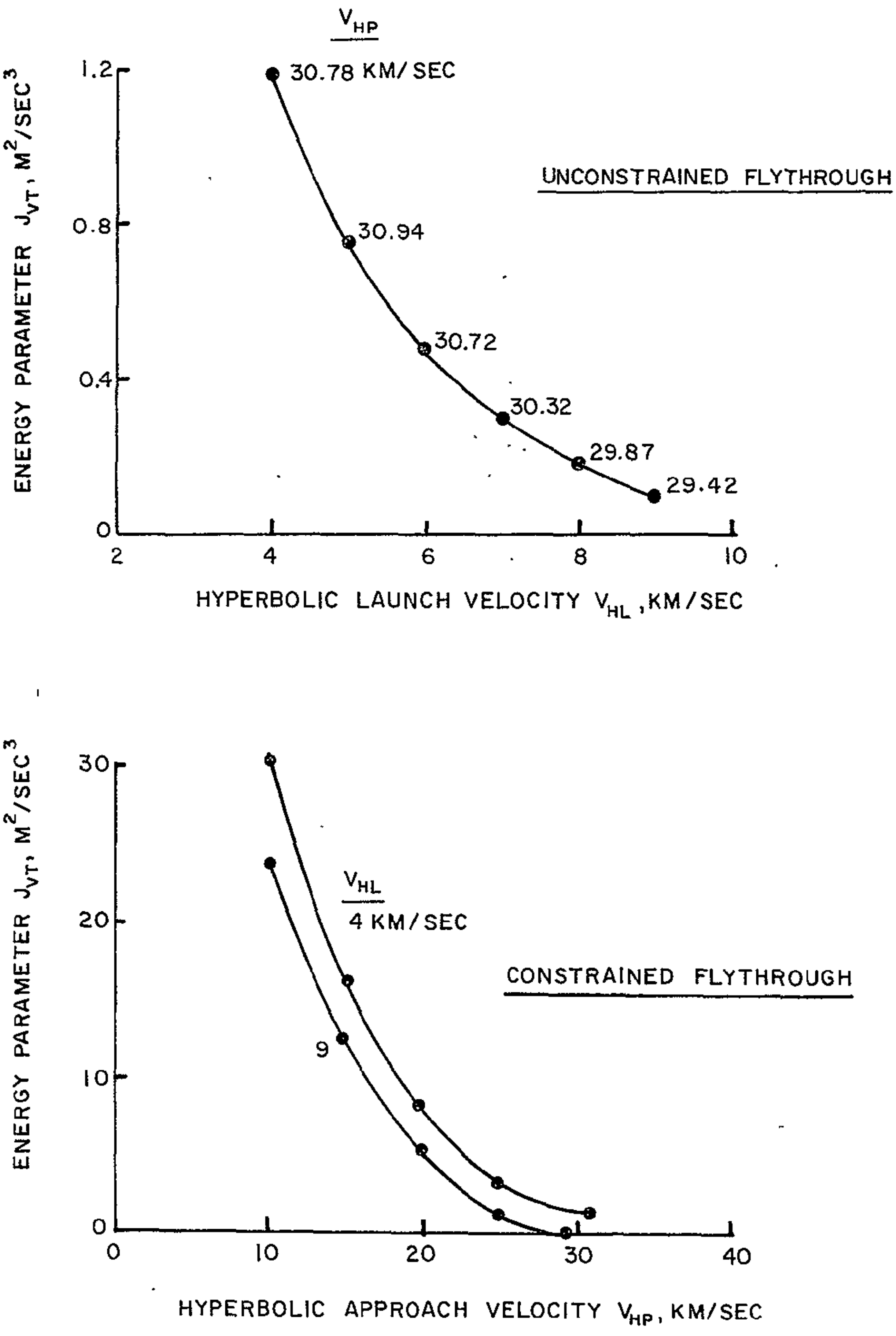

FIGURE 4-6. SENSITIVITY TO HYPERBOLIC LAUNCH AND APPROACH VELOCITIES FOR HALLEY'S COMET FLYTHROUGH, FLIGHT TIME = 900^d, ARRIVAL DATE = $T_p - 160^d$

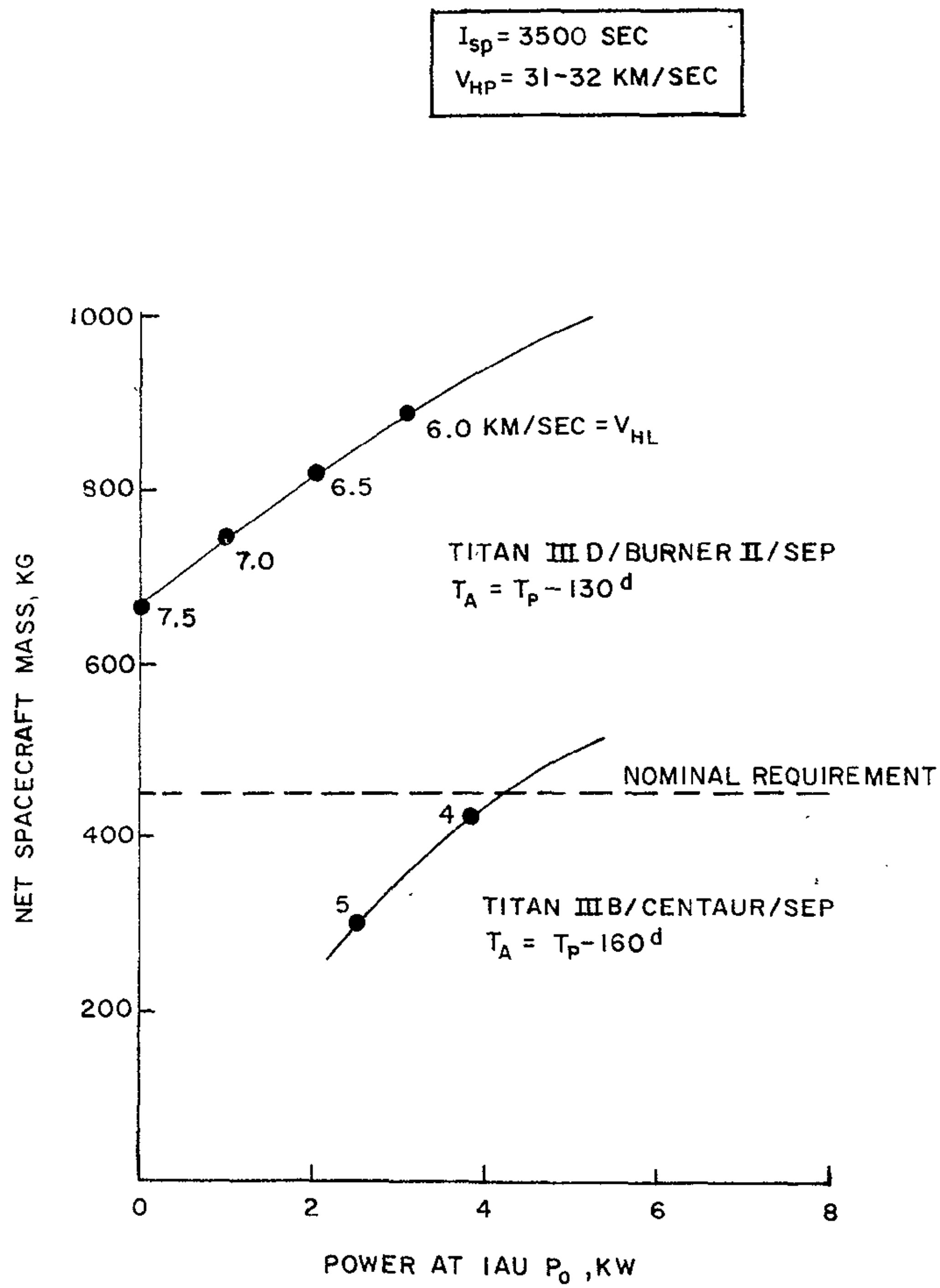

FIGURE 4-7. SOLAR ELECTRIC CAPABILITY FOR 900 DAY HALLEY'S COMET FLYTHROUGH MISSION (UNCONSTRAINED MODE)

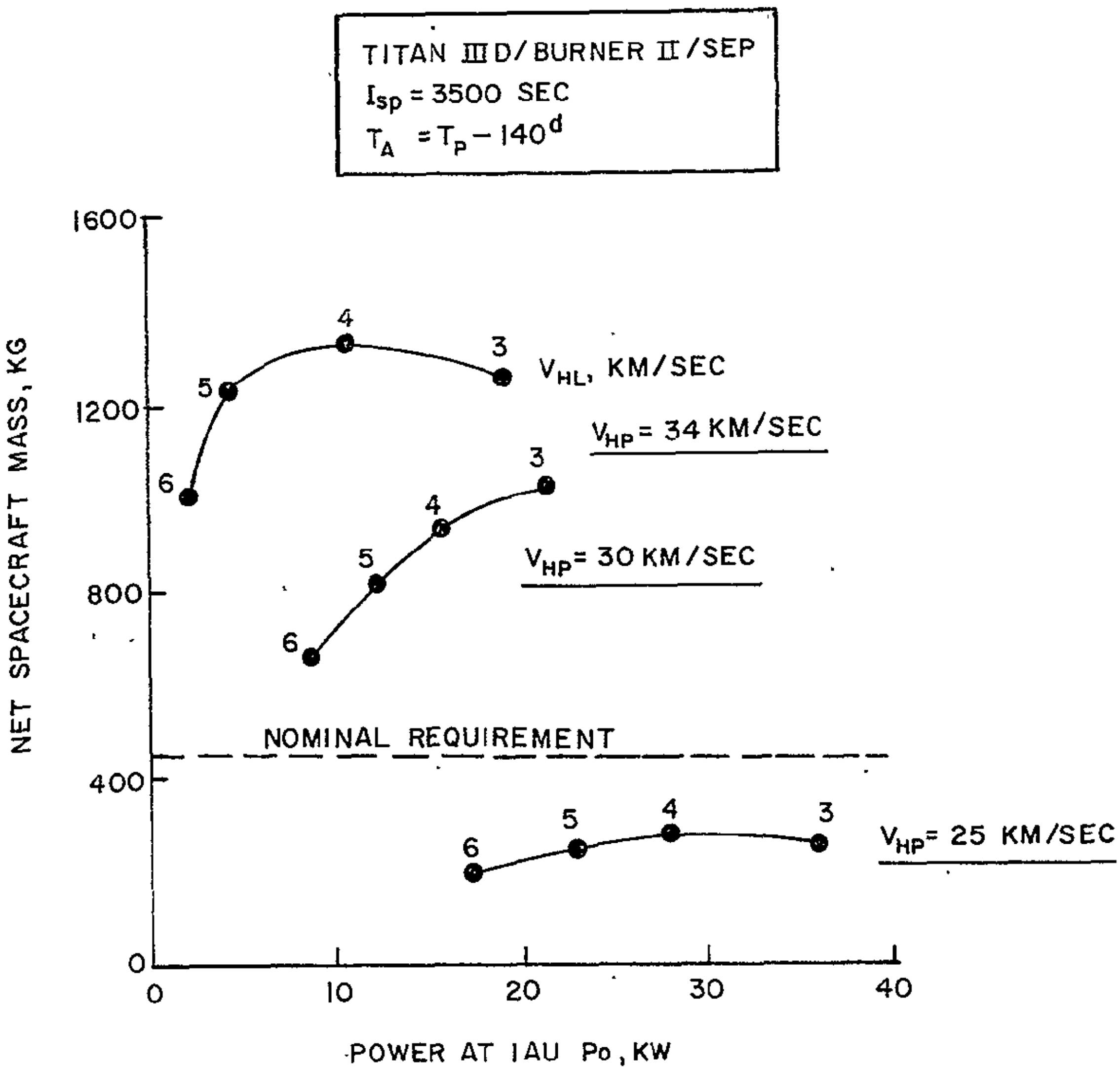

FIGURE 4-8. SOLAR ELECTRIC CAPABILITY FOR 500 DAY FLYTHROUGH MISSIONS TO HALLEY'S COMET

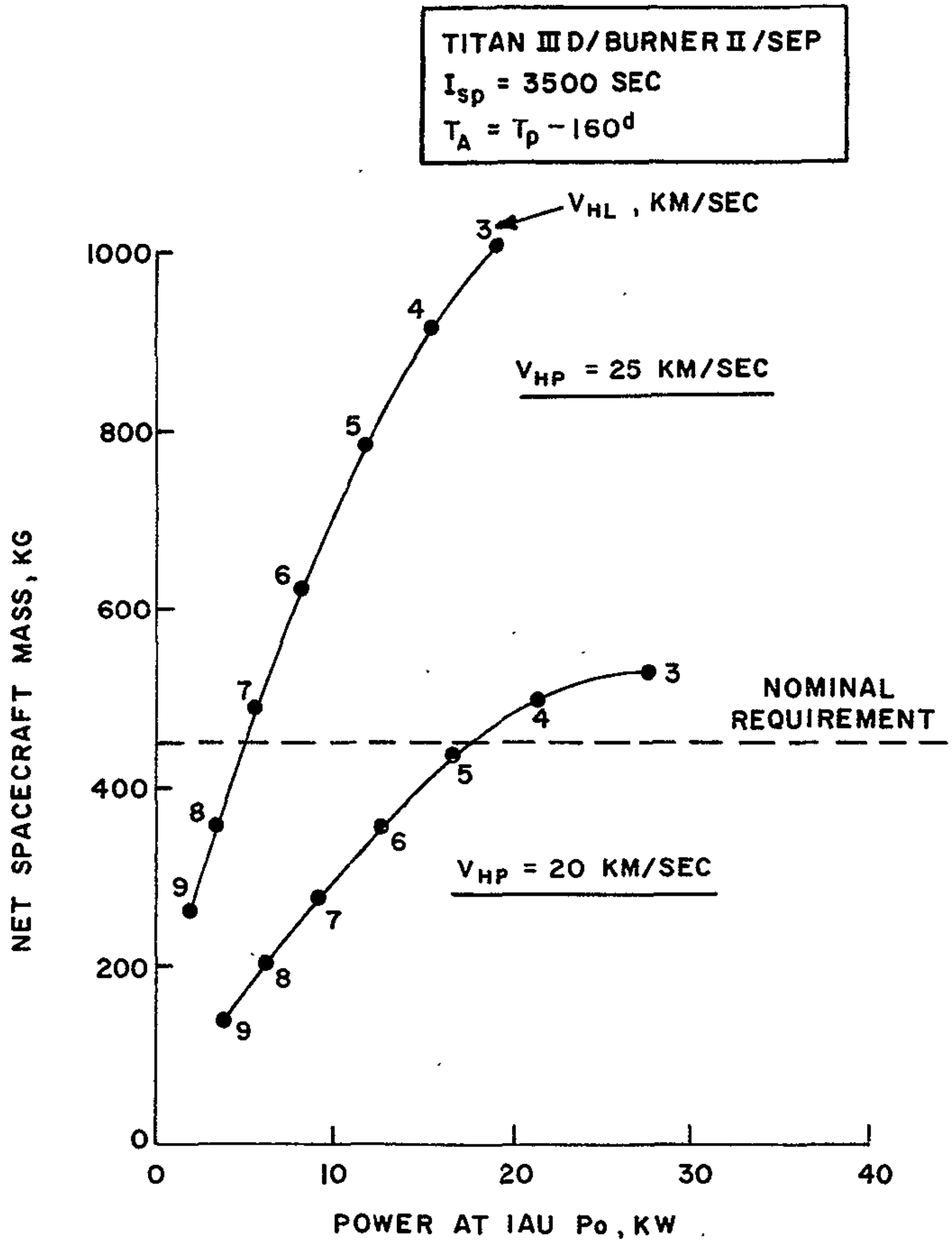

FIGURE 4-9. SOLAR ELECTRIC CAPABILITY FOR 900 DAY FLYTHROUGH MISSIONS TO HALLEY'S COMET.

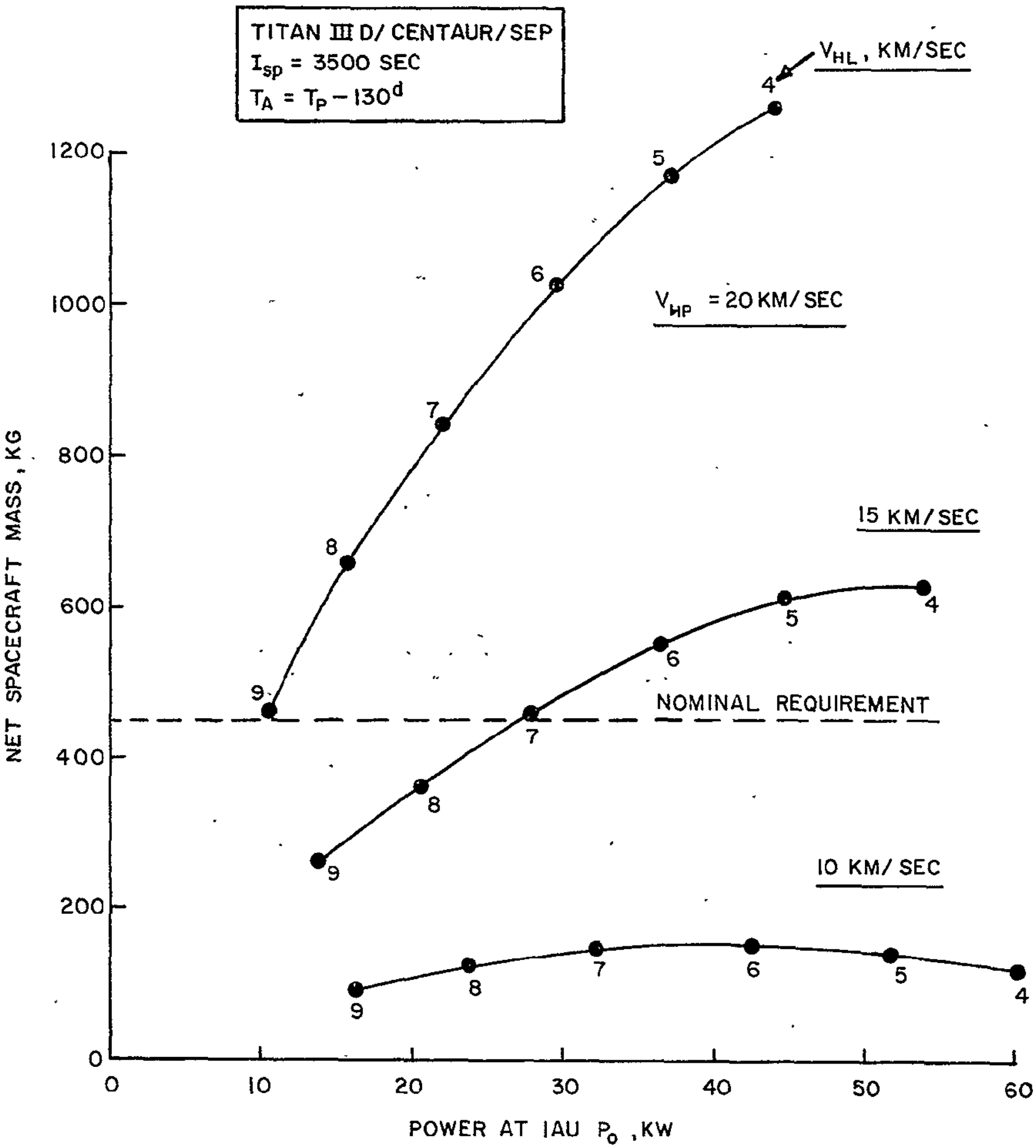

FIGURE 4-10. SOLAR ELECTRIC CAPABILITY FOR 1300 DAY FLYTHROUGH MISSIONS TO HALLEY'S COMET

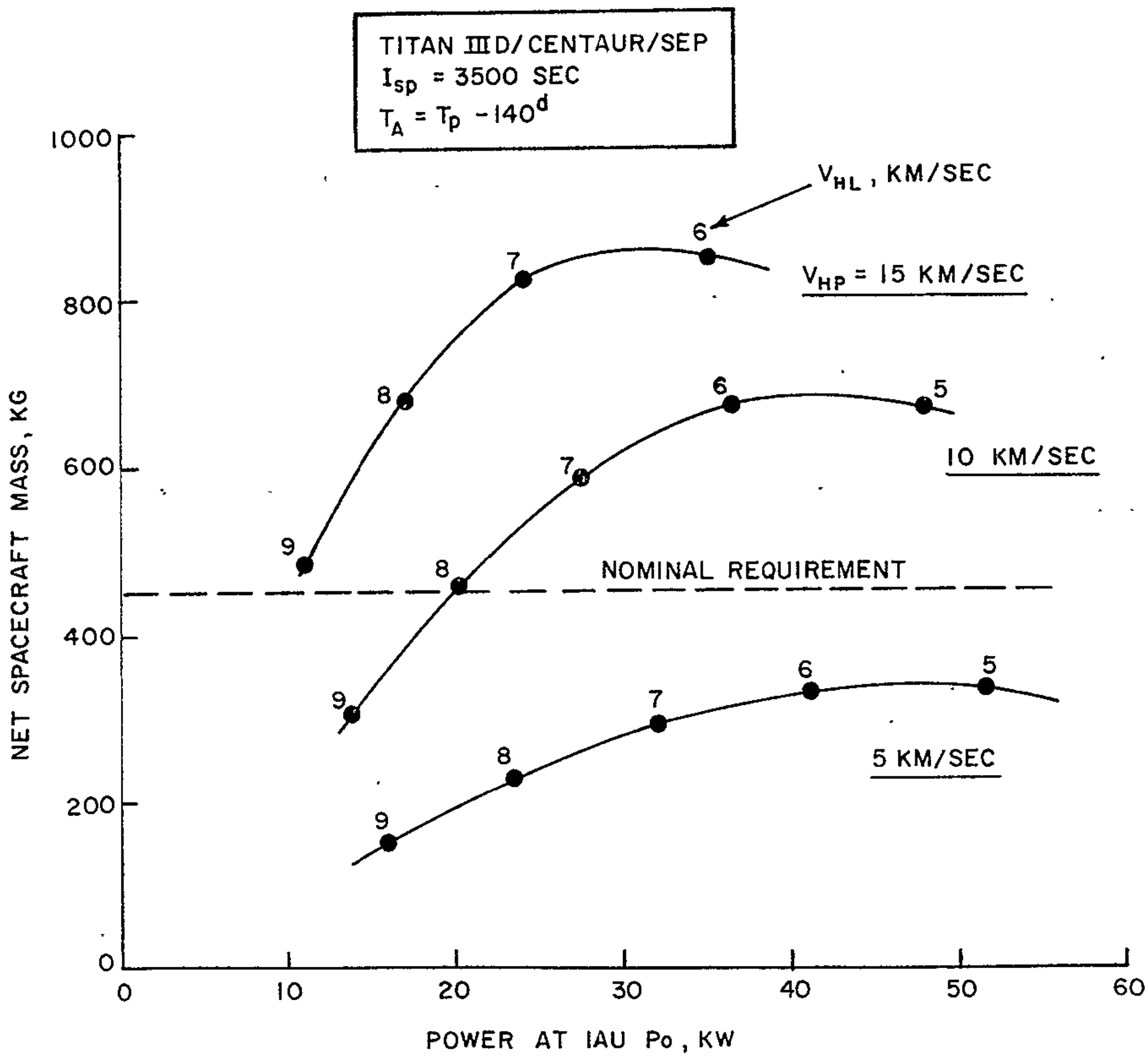

FIGURE 4-11. SOLAR ELECTRIC CAPABILITY FOR 2300 DAY FLYTHROUGH MISSIONS TO HALLEY'S COMET

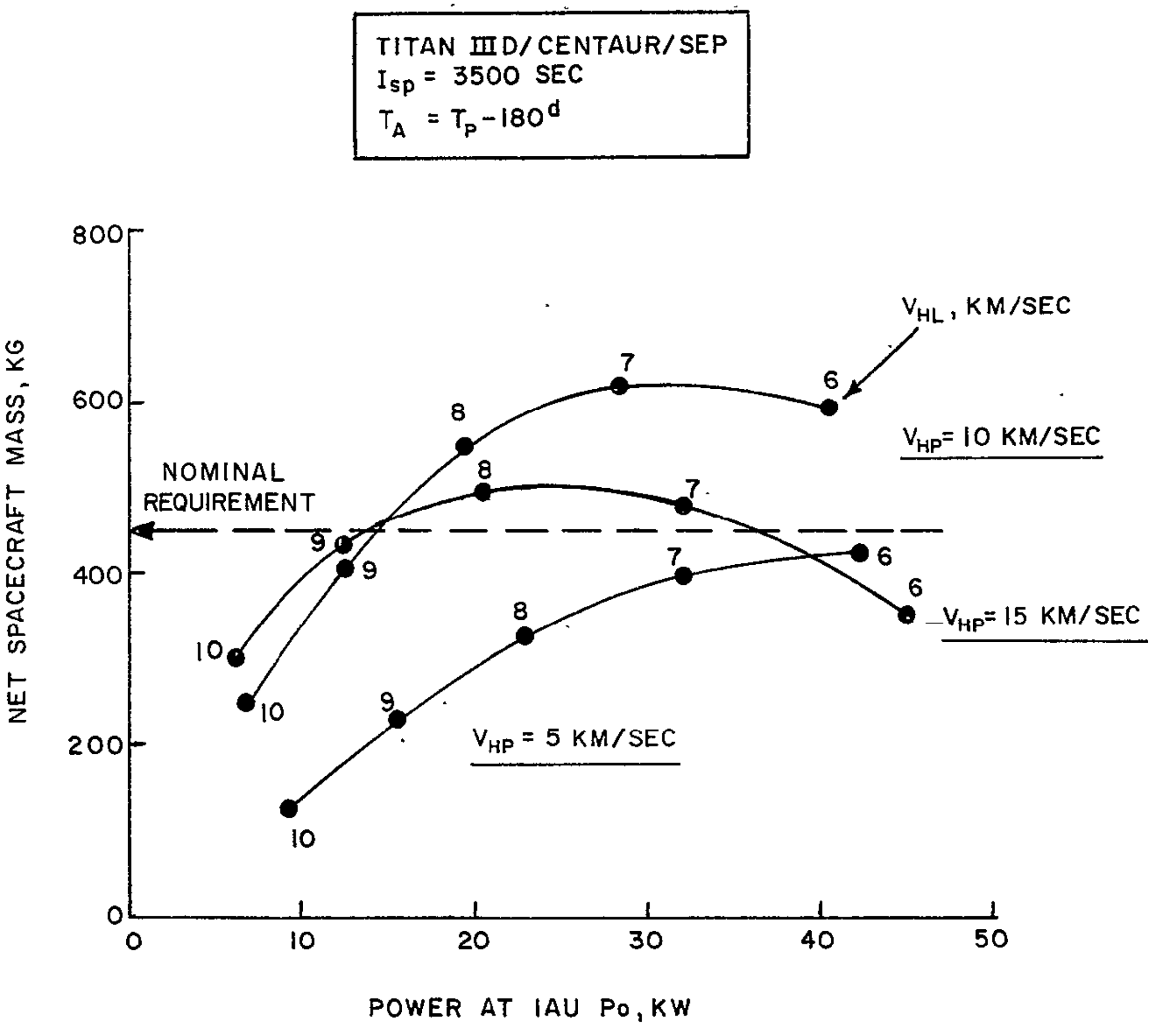

FIGURE 4-12. SOLAR ELECTRIC CAPABILITY FOR 2600 DAY FLYTHROUGH MISSIONS TO HALLEY'S COMET

velocities for a 450 kg net mass are 27 km/sec and 20 km/sec, respectively.

The optimum power levels are rather high in light of current design trends. Figure 4-13 presents an overview of flythrough mission capability for a more representative power input of 15 kw. The limiting approach velocity for the nominal payload requirement varies from 10 km/sec (2600* day flight) to 27 km/sec (500 day flight).

Figure 4-14 illustrates the trajectory profiles of the 1300, 2300 and 2600 day missions. Aphelion distance/time points are (4.33 a.u., 780 days), (6.69 a.u., 1288 day), and (7.42 a.u., 1456 days), respectively. The spacecraft is 0.2-0.4 a.u. below the ecliptic plane at the aphelion points with a positive $\dot{Z}$ component. The approach direction is made from outside the orbit of Halley.

The main results of the flythrough mission analysis may be summarized as follows:

1. Solar electric propulsion offers no performance advantage for the fast intercept class of missions. These missions, characterized by a 6 month flight duration and a flythrough velocity of 55 km/sec, can easily be performed ballistically using small launch vehicles.

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* The 2600 day curve shown in Figure 4-13 peaks at a V_{HP} of about 12.5 km/sec which represents the unconstrained flythrough solution for this flight time/arrival date combination. The reader is also referred to Figure 4-12 for explanation of this seemingly anomalous characteristic.

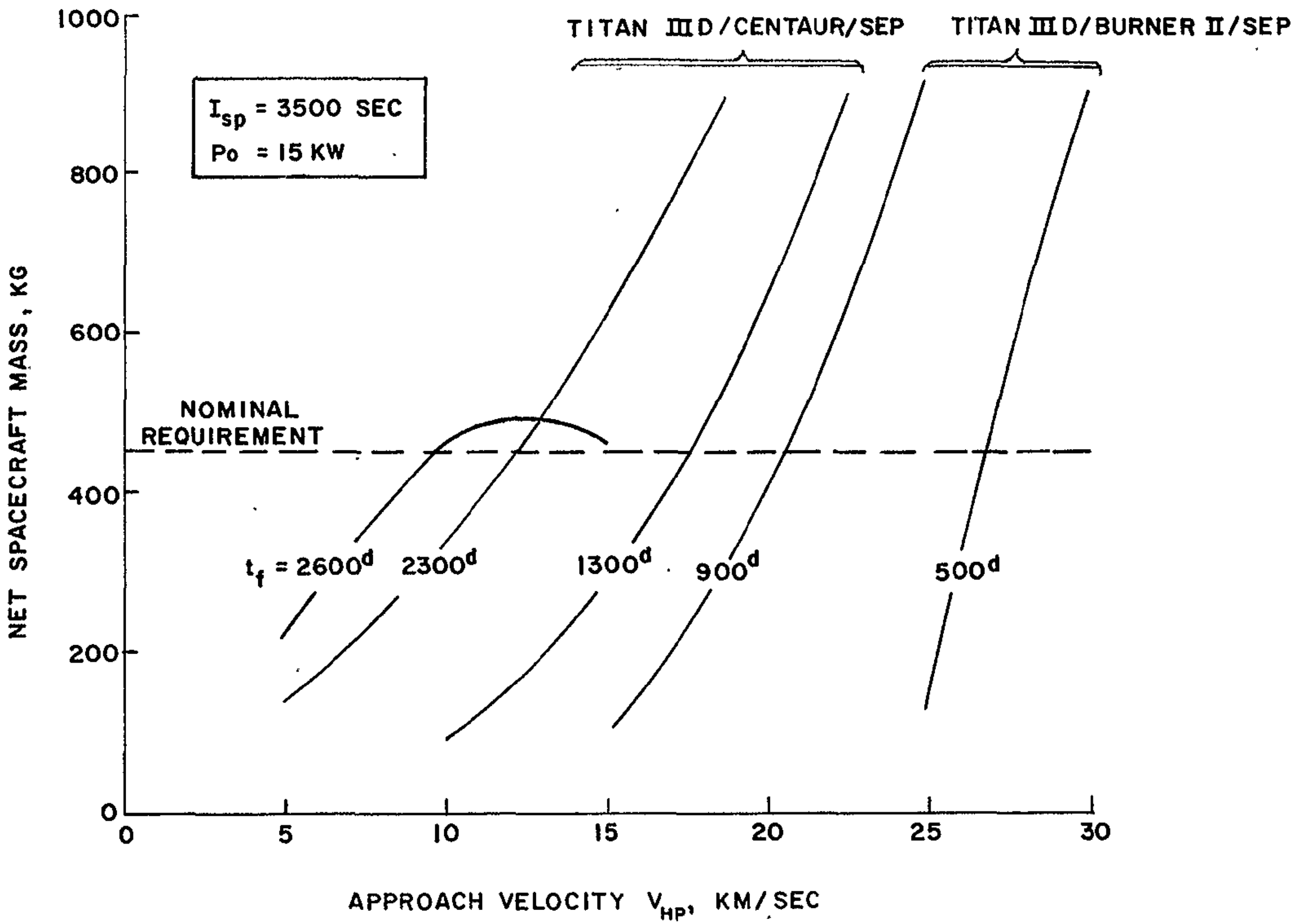

FIGURE 4-13. SOLAR ELECTRIC CAPABILITY SUMMARY FOR FLYTHROUGH MISSIONS TO HALLEY'S COMET.

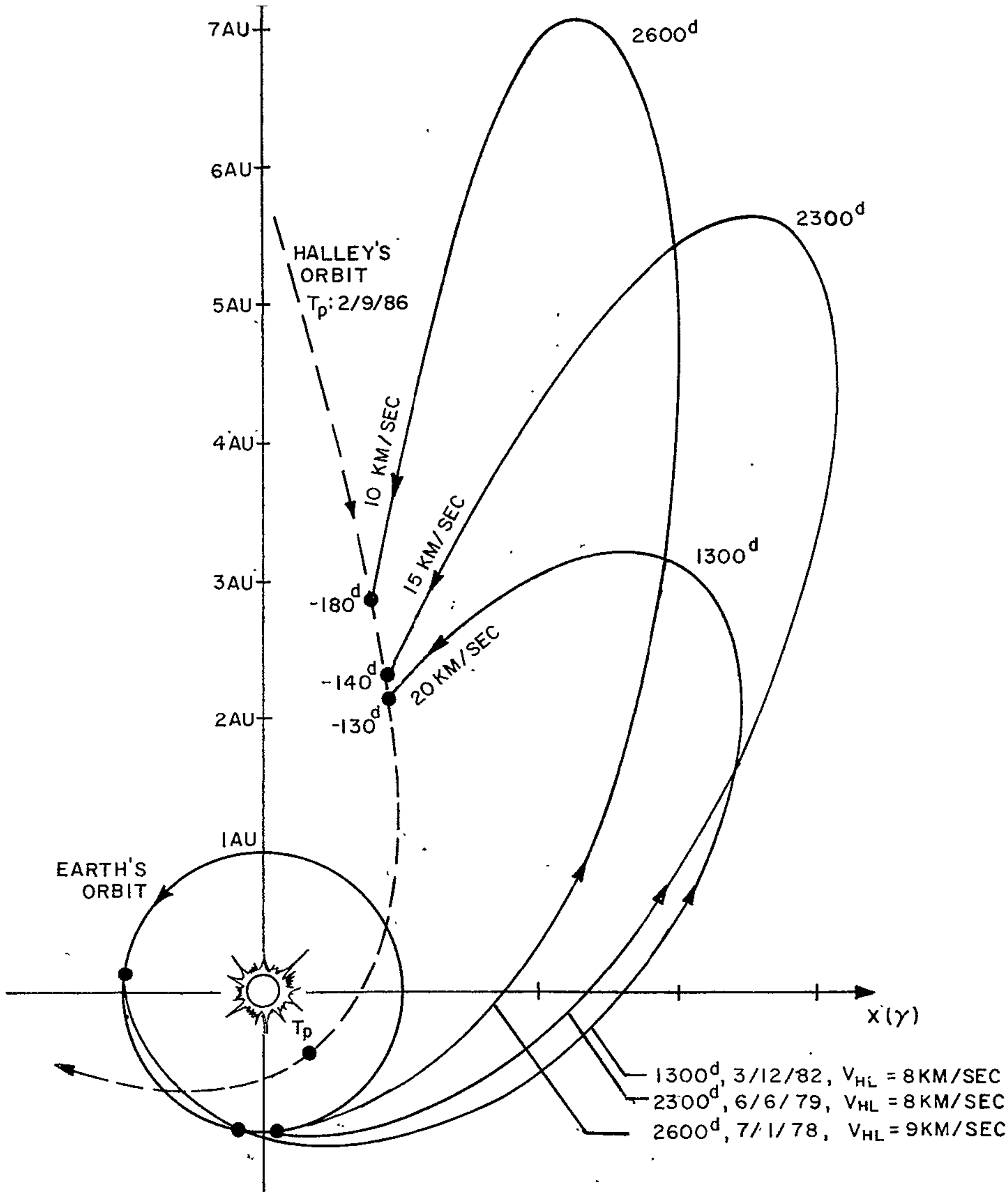

FIGURE 4-14. SOLAR ELECTRIC TRAJECTORIES TO HALLEY'S COMET, 1300, 2300 AND 2600 DAY FLYTHROUGH MISSIONS

2. The 40-30 km/sec velocity region is accessible to ballistic spacecraft launched by the Titan IIID/Burner and flight durations of 1.5-2.5 years. The addition of a 10-15 kw SEP stage allows a 10 km/sec reduction in fly-through velocity for this mission class, i.e., to the region 30-20 km/sec.

3. The 20-5 km/sec velocity region is accessible to SEP spacecraft launched by the Titan IIID/Centaur. Flight time and solar array power requirements are in the range 3.5-7 years and 15-40 kw. If the SEP power is reasonably constrained to 15 kw, the 3.5 and 7 year missions have limiting flythrough velocities of 18 km/sec and 10 km/sec, respectively, for a 450 kg net mass capability. The longer mission is probably not justified in view of the relative reduction in velocity - 10 km/sec is still far removed from rendezvous conditions. As a point of perspective, the Titan IIID/Centaur/SEP can readily accomplish a rendezvous mission to short-period comets other than Halley.

4.2 Rendezvous Missions

The preceeding results have underlined the great difficulty in achieving a low relative velocity at Halley using solar electric propulsion. They also point the direction one must take to arrive at a reasonable rendezvous mission profile -- if indeed one exists. Unfortunately, this direction is toward even longer flight times than 2600 days, larger launch vehicles and higher SEP power. It may be expected also that the best arrival dates will be closer to perihelion in order to gain the necessary increase in propulsive power as solar distance decreases.

Figure 4-15 shows the rendezvous J requirements as a function of arrival date for several flight times with launches in 1977-78. The 2650-2750 day flight time subclass has preferred arrival points from 110 to 20 days before perihelion. Values of J are quite high at 13.5 to 16 m^2/sec^3. The characteristic shift of optimum arrival date with flight time is again noted. The 2700 and 3050 day flights have similar terminal positions, the longer flight being launched approximately one year earlier. However, the relative reduction in energy requirements is only 8 percent. An even longer flight time might improve the situation somewhat, but launches earlier than 1977 were not considered in the study. The 2700 day trajectory profile is illustrated in Figure 4-16.

Net spacecraft mass is plotted as a function of input power in Figure 4-17 for the 2700 and 3050 day missions. The Titan IIID(7)/Centaur launch vehicle is assumed. The maximum payload is only 280 kg with a 46 kw powerplant for the 3050 day flight. In a further attempt to "stretch" the SEP capability, specific impulse was varied about the baseline value of 3500 sec. The results given by the upper graph in Figure 4-17 show that the optimum I_{sp} is somewhat less than 2500 sec. Still, the maximum payload is only 325 kg.

It must be concluded that the direct mode rendezvous utilizing solar electric propulsion is not a viable mission concept. The following section of this report will describe how a gravity-assist via a planet swingby can improve this situation.

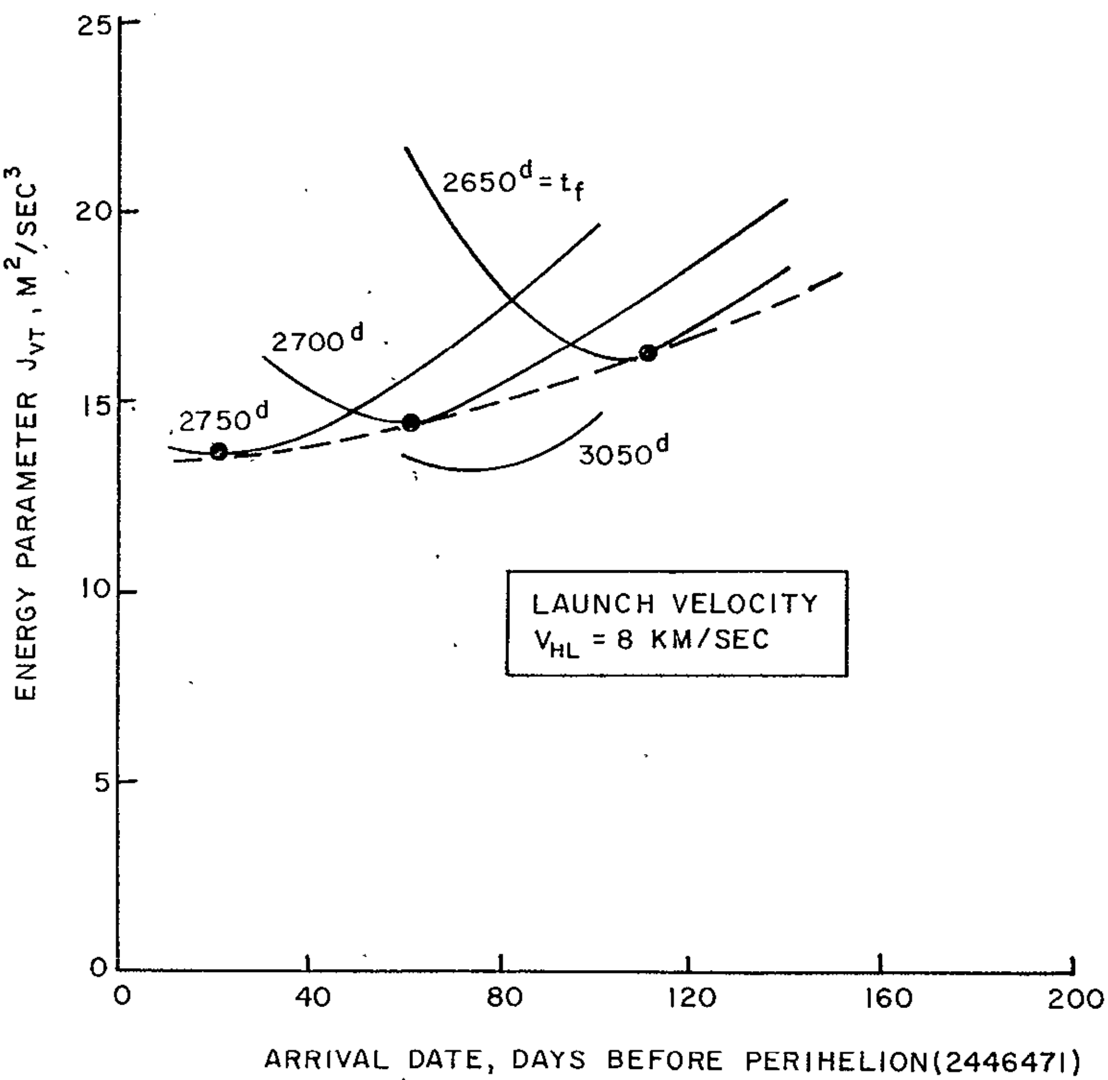

FIGURE 4-15. ILLUSTRATION OF FLIGHT TIME/ARRIVAL DATE CHARACTERISTICS FOR HALLEY'S COMET RENDEZVOUS.

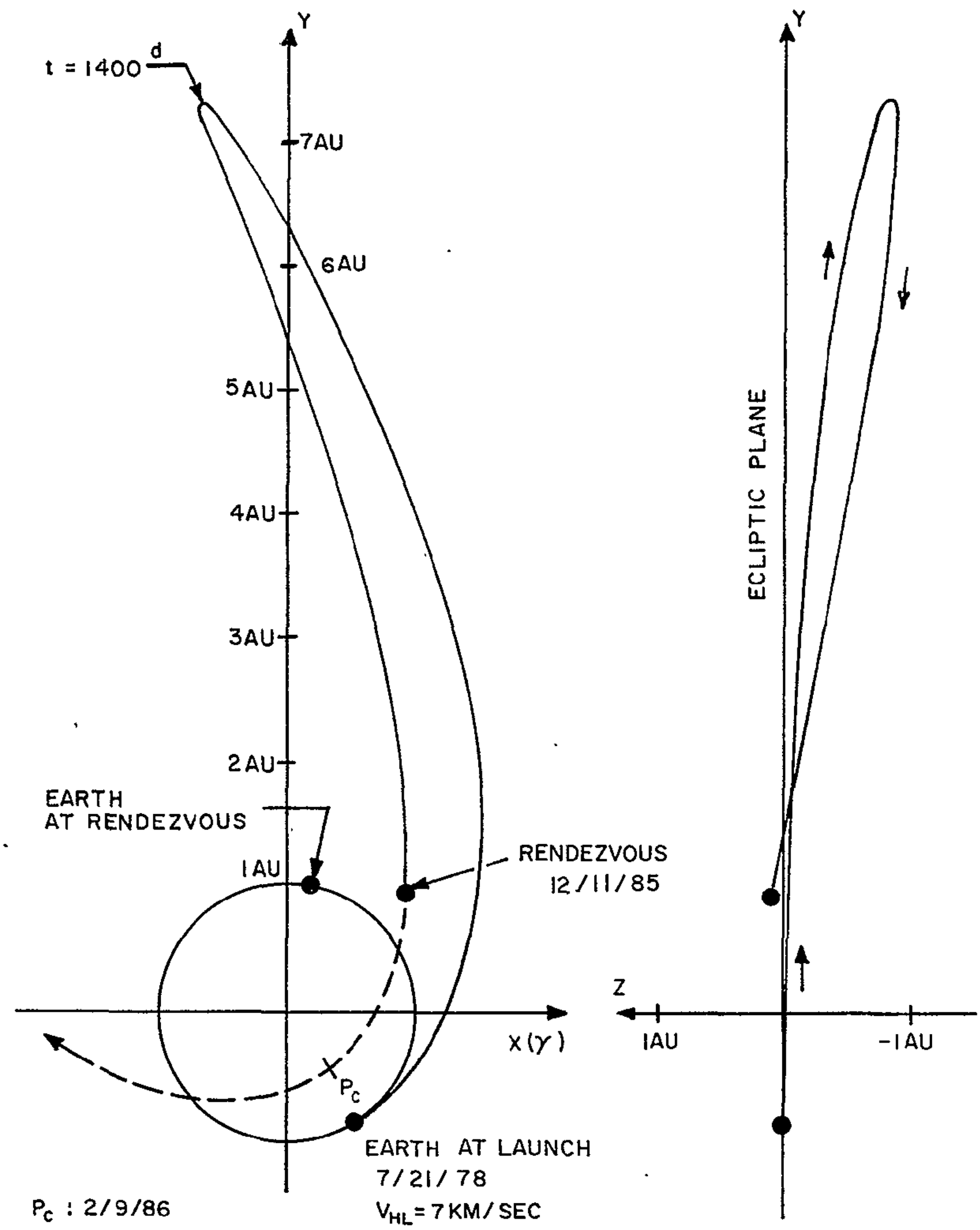

FIGURE 4-16. SOLAR ELECTRIC RENDEZVOUS WITH HALLEY'S COMET

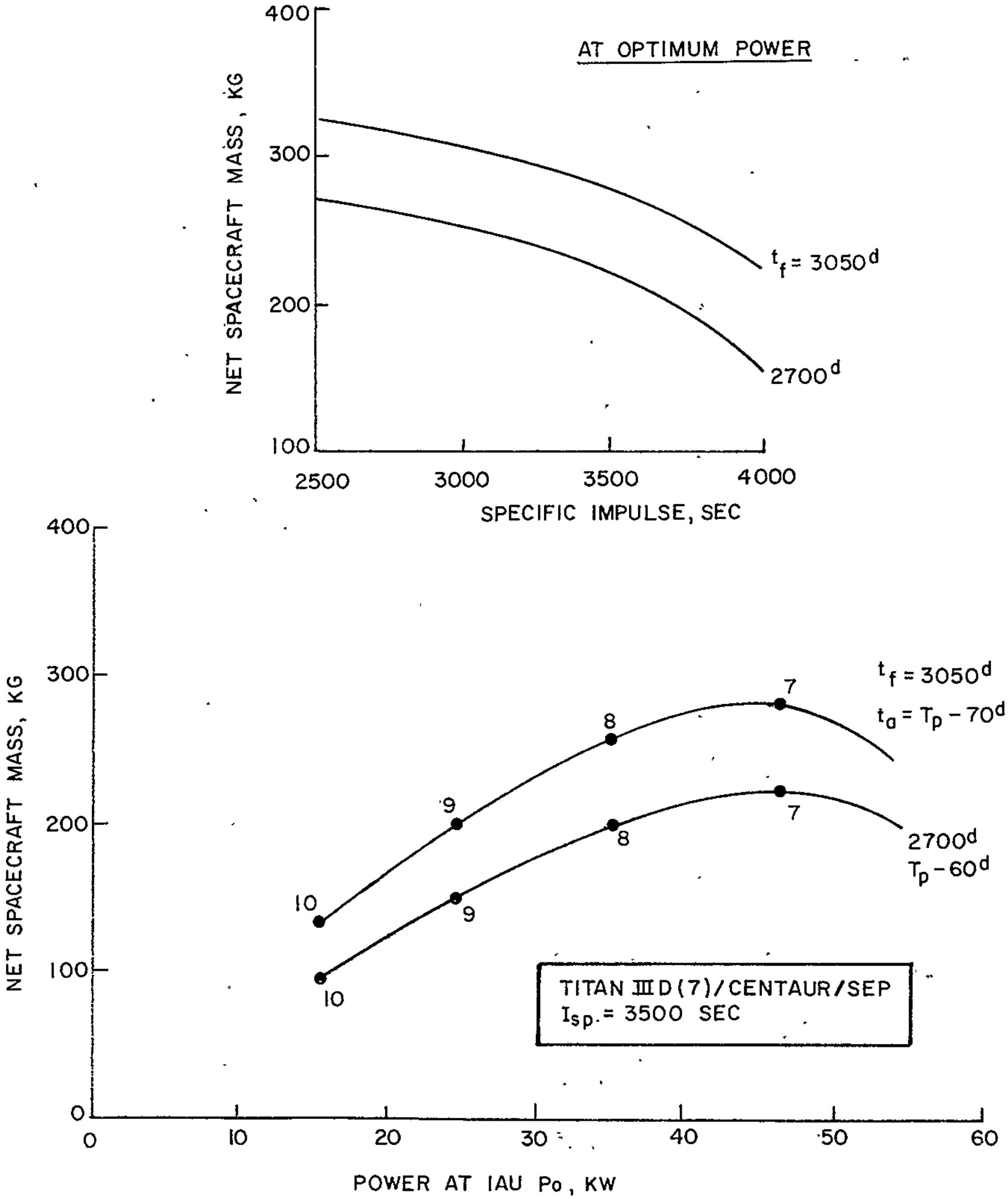

FIGURE 4-17. SOLAR ELECTRIC CAPABILITY FOR HALLEY'S COMET RENDEZVOUS.

5. GRAVITY-ASSISTED FLIGHT MODE RESULTS

5.1 Planet Swingby Potential

The physical and mathematical treatment of planetary swingby has been described in earlier reports and will not be repeated here (Flandro 1966). However, for reference purposes, Figure 5-1 illustrates a typical planetocentric and heliocentric velocity diagram and defines other relevant swingby parameters.

One measure of swingby potential is the maximum deflection angle in planetocentric coordinates; i.e., the rotation of the hyperbolic asymptotes. Figure 5-2 shows ψ_{max} as a function of hyperbolic excess velocity for the planets Venus, Mars, Earth and Jupiter. The hard limit of a grazing pericenter passage is assumed. The lower velocity region is applicable to Venus and Mars transfers. For example, at V_H = 6 km/sec, these two planets provide maximum deflection angles of 73° and 30°, respectively. Because of its great mass Jupiter is capable of the largest path deflection for any given hyperbolic speed. The higher velocity region will be shown to be a typical requirement for the Halley mission. Thus, even at V_H = 20 km/sec, Jupiter still provides a maximum deflection of 110°.

The real potential of a planet swingby must be related to the mission objectives. That is, how does the gravity-assist affect the subsequent shaping of the heliocentric trajectory? Two important measures in this regard are the amount of heliocentric velocity change (ΔV) and heliocentric velocity deflection (γ). These quantities are graphed in Figures 5-3 to 5-5 for the planets Venus, Mars, and Jupiter.

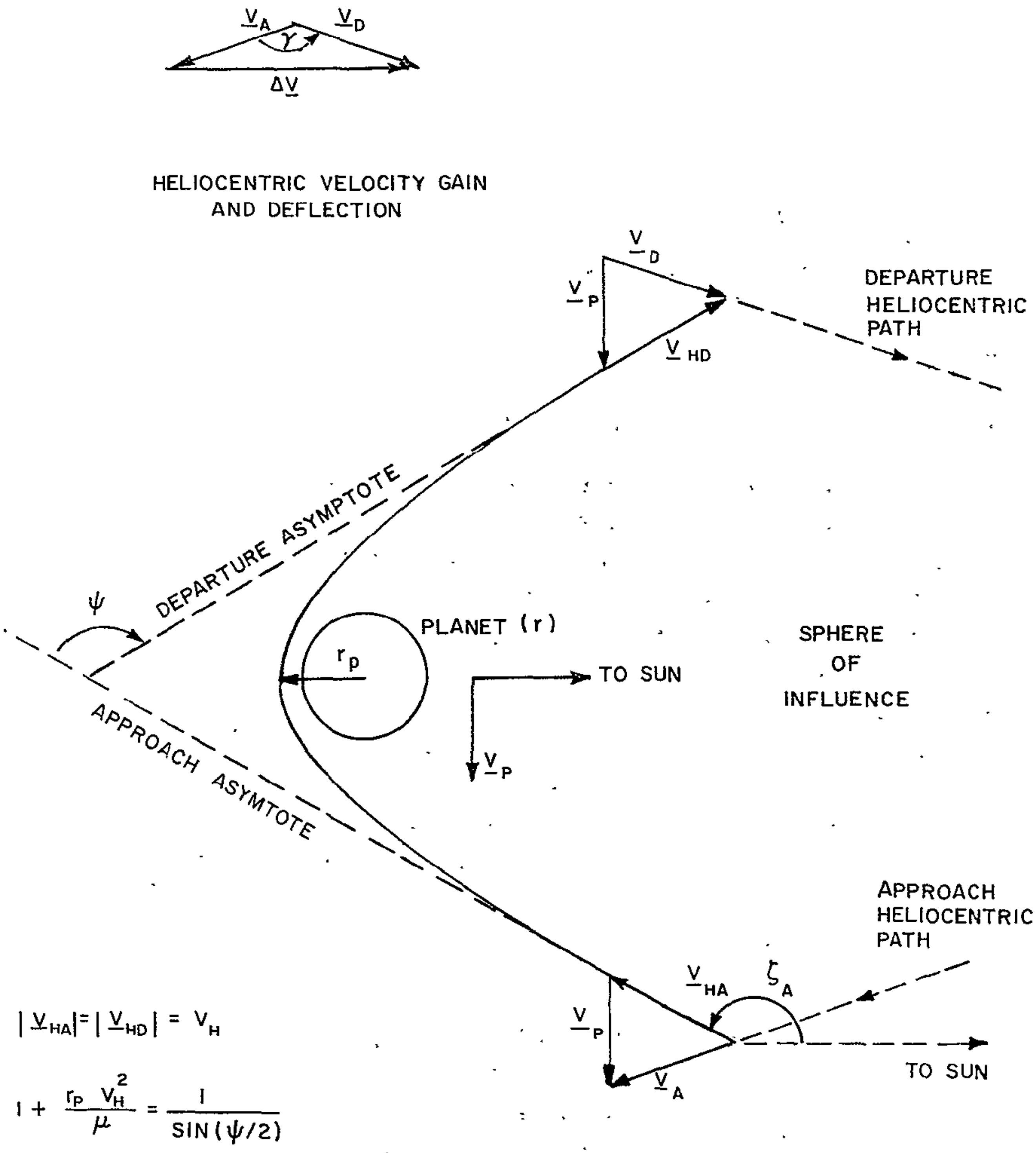

FIGURE 5-1. GRAVITY-ASSIST GEOMETRY

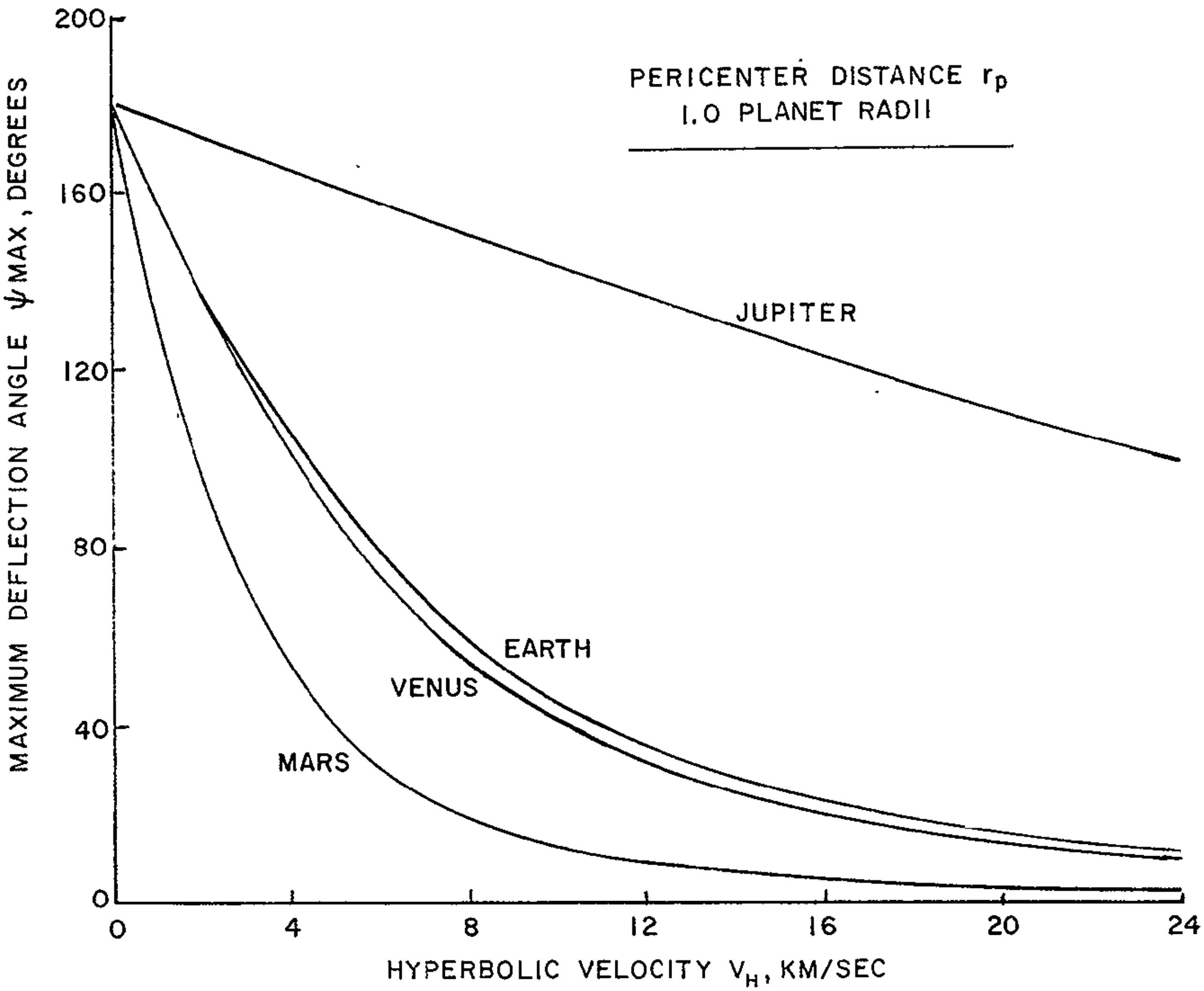

FIGURE 5-2. MAXIMUM DEFLECTION OF ASYMPTOTE FOR PLANET SWINGBYS

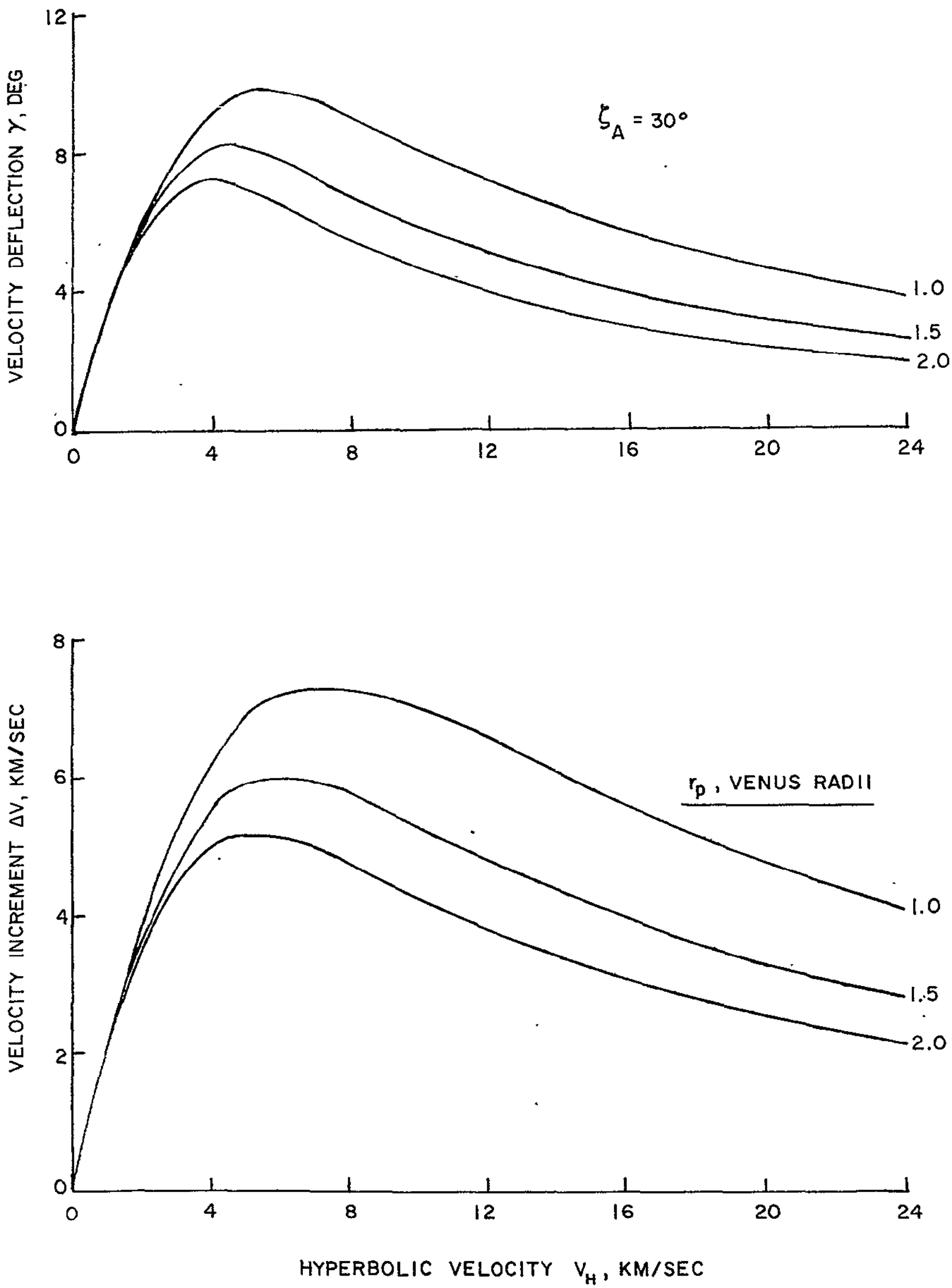

FIGURE 5-3. HELIOCENTRIC VELOCITY CHANGE FOR VENUS GRAVITY-ASSIST (ENERGY ADDITION)

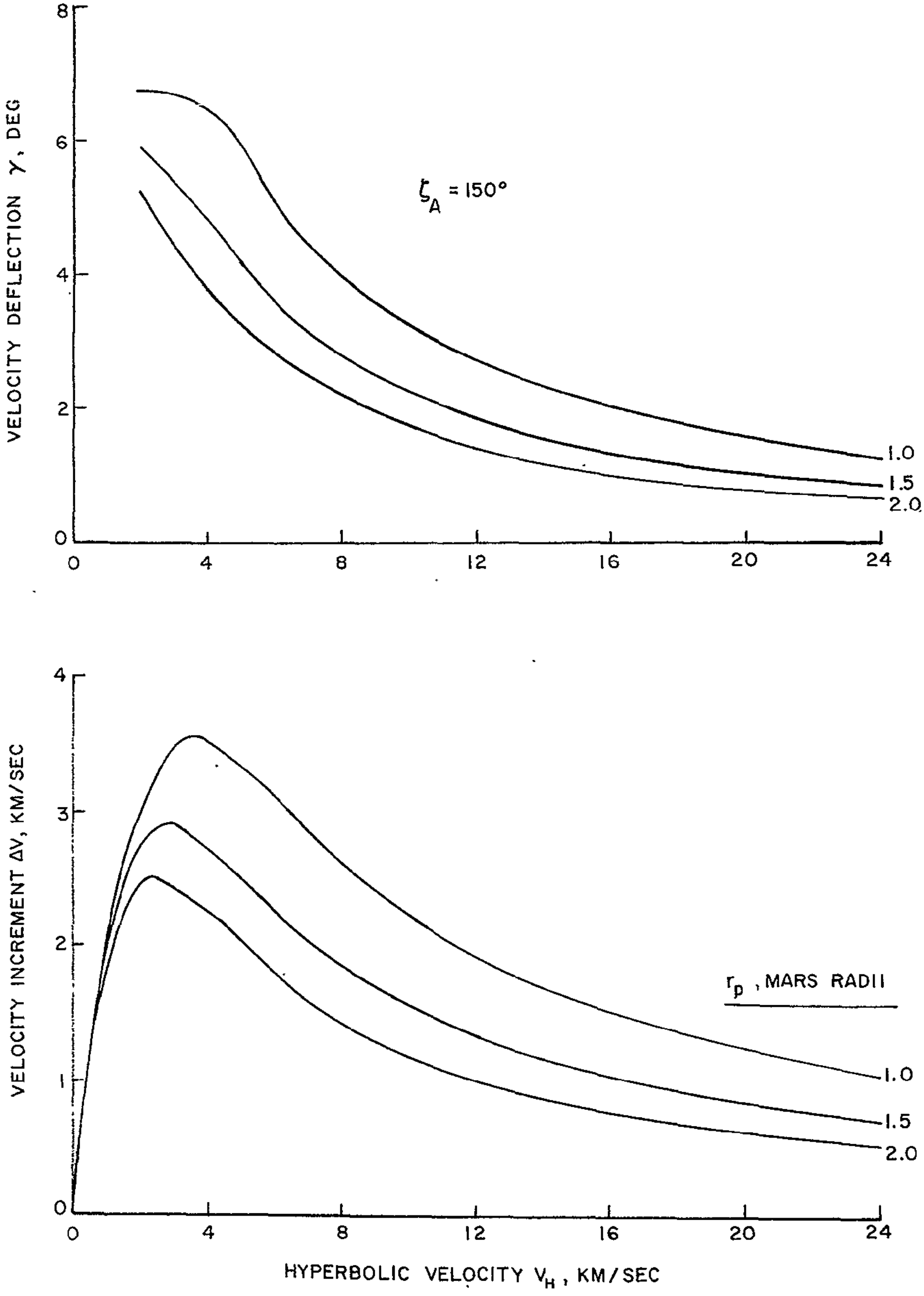

FIGURE 5-4. HELIOCENTRIC VELOCITY CHANGE FOR MARS GRAVITY-ASSIST (ENERGY ADDITION)

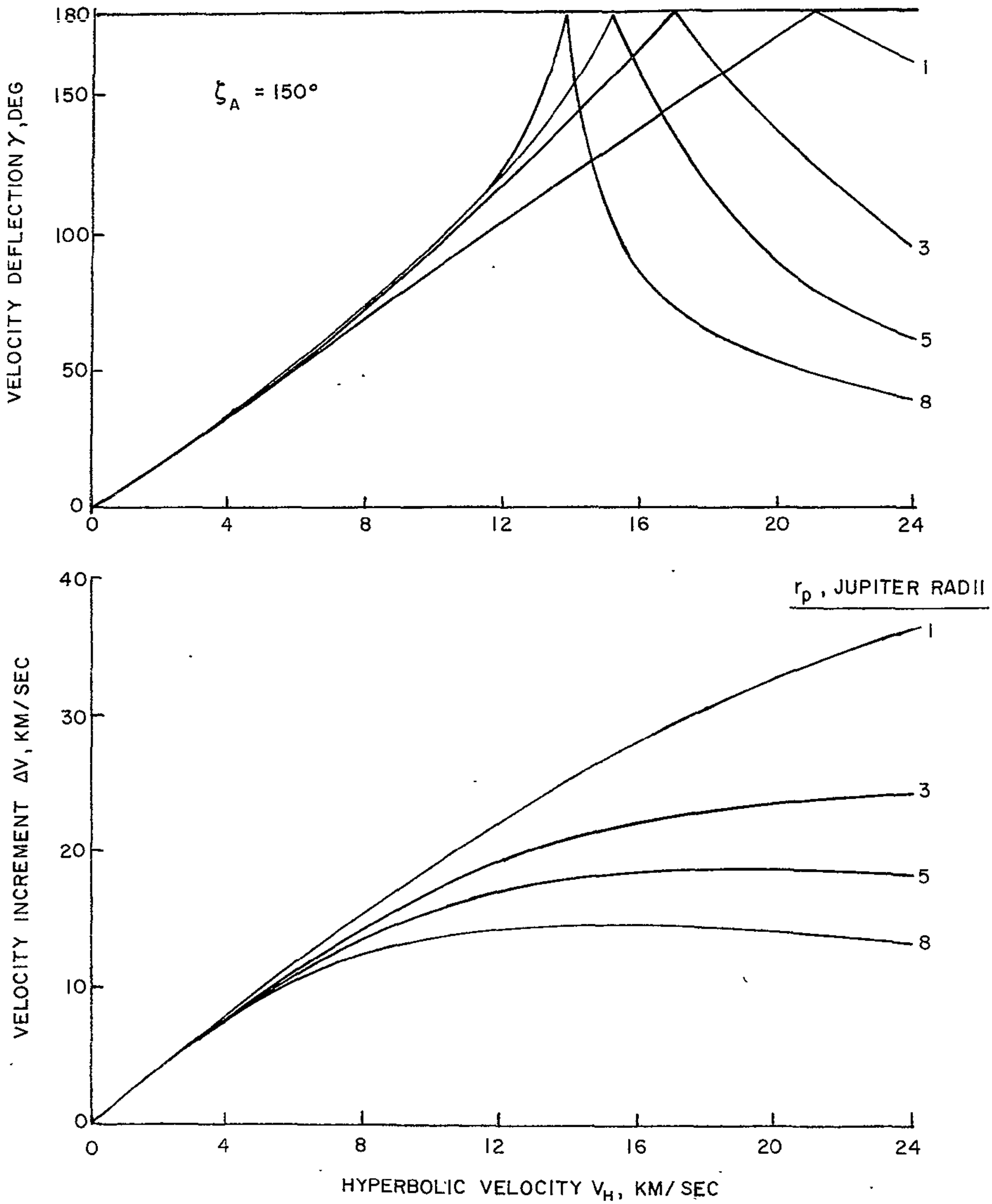

FIGURE 5-5. HELIOCENTRIC VELOCITY CHANGE FOR JUPITER GRAVITY-ASSIST (ENERGY REDUCTION)

The values of the planet approach direction (ζ_A) chosen for the γ calculation are representative of Earth-planet transfers. For the case of a grazing passage at Venus, the maximum ΔV is about 7 km/sec at V_H = 7 km/sec. While this appears to be a substantial velocity gain, it is noted that the maximum path deflection is only 10°. The smaller planet Mars provides maximum ΔV gain and path deflection of only 3.5 km/sec and 6.5°, respectively. In contrast, a Jupiter swingby offers a maximum ΔV increment of about 35 km/sec and a 180° path deflection. Jupiter would be utilized to reduce the energy of the heliocentric trajectory. For example, at V_H = 20 km/sec and a pericenter distance of 5 radii, the heliocentric speed is decreased from 17.6 km/sec to 7 km/sec while the velocity direction is rotated by 89° into retrograde motion.

The Halley mission has rather unique gravity-assist requirements. Ideally, the post-assist trajectory should be retrograde and closely matched to Halley's orbit. Of the planets considered in this study, only Jupiter can provide this desired characteristic. There may be some benefit obtained from a multiple swingby mission concept, e.g., Earth-Mars-Jupiter-Halley. Of course the intermediate swingby planet must have the proper position-time relationship linked to the Jupiter-Halley trajectory requirements. Multiple swingbys were not considered in this study with the one exception of the Earth swingby concept proposed by Meissinger (1970). This concept will be discussed in Section 5.4 after the Jupiter-assisted missions are described.

5.2 Jupiter-Assisted Flythrough Missions

The gravitational field of Jupiter is utilized to deflect the spacecraft into a retrograde trajectory aligned as closely as possible with Halley's orbit. Michielsen has shown that the best launch opportunities for the all-ballistic mission occur in the fall of 1977 and 1978. Typical trajectory characteristics are summarized in the following table:

	1977	1978
Time to Jupiter	1.0 year	1.0 year
Hyperbolic launch velocity	13.6 km/sec	14.1 km/sec
Hyperbolic swingby velocity	19.28 km/sec	20.46 km/sec
Swingby distance	7.87 R_J	4.96 R_J
Halley arrival	$T_p - 258^d$	$T_p - 152^d$
Halley approach velocity	5.83 km/sec	6.42 km/sec

The 1977 opportunity has a slight advantage of a lower launch and approach velocity, but the early arrival date is perhaps less favorable from a science standpoint. Ballistic spacecraft mass injected by the Titan IIID(7)/Centaur/Burner II is 265 kg and 200 kg, respectively, in 1977 and 1978. This payload capability is clearly inadequate.

One objective in the present study is to determine the performance improvement offered by utilizing a SEP stage for the Earth-Jupiter leg of the flythrough mission. For this mission concept the Jupiter-Halley leg is still ballistic, i.e., SEP is not used at all after Jupiter swingby, except possibly for guidance corrections. The ballistic data given by Michielsen are used as a reference point for the analysis. Additional SEP results are obtained for the 1979 launch opportunity.

Trajectory characteristics of the Jupiter-assisted flythrough missions are listed in Table 5-1. Flight time to Jupiter is about 13 months in each of the launch years. The range of swingby deflection angle and pericenter distance is 44°-74° and 7.9-2.8 planet radii, respectively.

TABLE 5-1

TRAJECTORY CHARACTERISTICS OF JUPITER-ASSISTED HALLEY FLYTHROUGH MISSIONS

	1977 LAUNCH	1978 LAUNCH	1979 LAUNC
LAUNCH DATE	8/22/77	9/29/78	11/3/79
TIME TO JUPITER	390^d	380^d	400^d
ENERGY PARAMETER* J_{VT} at V_{HL} = 9 km/sec	3.62 m^3/sec^3	4.55 m^2/sec^3	3.51 m^2/sec
SWINGBY DATE	9/16/78	10/14/79	12/7/80
HYPERBOLIC VELOCITY	19.28 km/sec	20.5 km/sec	20.4 km/se
ASYMPTOTE DEFLECTION	$44.\overset{\circ}{2}$	$54.\overset{\circ}{7}$	$74.\overset{\circ}{1}$
PERICENTER DISTANCE	7.87 R_J	4.96 R_J	2.80 R_J
HALLEY ARRIVAL	$T_p - 258^d$	$T_p - 150^d$	$T_p - 60^d$
APPROACH VELOCITY	5.83 km/sec	6.41 km/sec	15.88 km/s
TOTAL FLIGHT TIME	7.75^{yr}	6.95^{yr}	6.10^{yr}

* SEP used only on Earth-Jupiter leg

Because Jupiter is not well placed for the 1979 launch opportunity, the arrival date is closer to perihelion and the approach velocity has a relatively high value of almost 16 km/sec. The data presented here is a result of a rather loose scan of Jupiter swingby dates (20 day increment), and, therefore, should be viewed as "near-optimum" solutions. A small modification of the swingby date for the 1978 mission could possibly reduce the energy requirement which is the largest of the three launch opportunities. Figure 5-6 compares the trajectory profiles of the three Jupiter-assisted missions.

Net spacecraft mass is shown as a function of power input in Figure 5-7. The Titan IIID(7)/Centaur capability is at least 450 kg in each launch year for 15-20 kw SEP powerplants. The Titan IIID/Centaur capability is adequate in 1979, marginal in 1977, and inadequate in 1978. In summary, the SEP stage provides about twice the payload of the Burner II stage matched to the 7-segment Titan/Centaur. Furthermore, use of the existing Titan IIID/Centaur is marginally possible for this type of mission with solar electric propulsion.

An intermediate step toward rendezvous is taken by examining the slow flythrough mission which includes low-thrust maneuvers on the Jupiter-Halley leg. The additional propulsion period allows more freedom in the Jupiter swingby conditions which may be used to improve the performance on the first leg. Specifically, the swingby velocity can be decreased and the swingby date adjusted accordingly. The 1978 launch opportunity was chosen for this investigation. Payload results are presented in Figure 5-8 for a flythrough velocity range of 2-10 km/sec. The arrival date in this case is 50 days before perihelion. Optimum SEP power is between 20 and 25 kilowatts, and the Titan IIID(7)/Centaur launch vehicle can provide a payload of

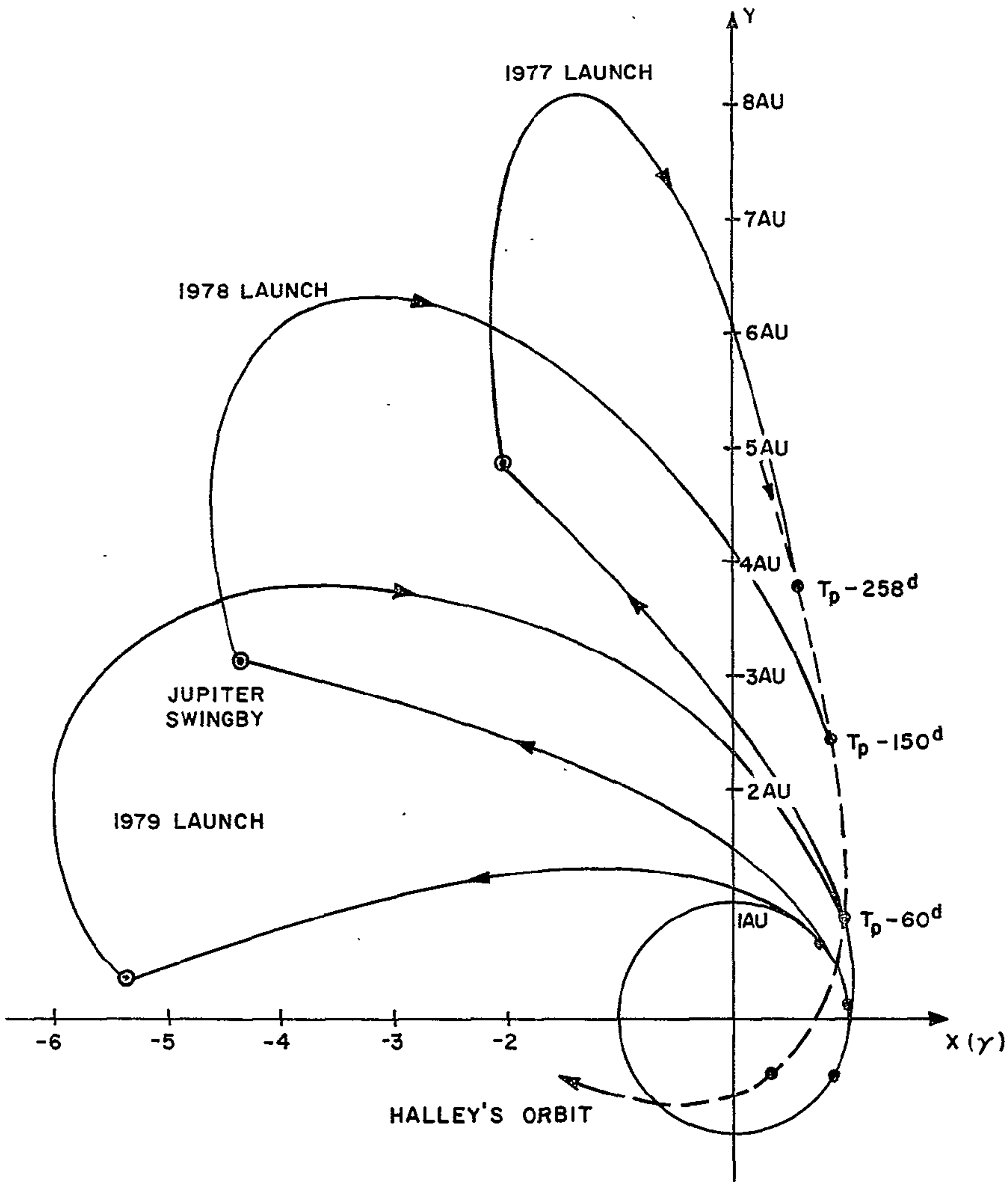

FIGURE 5-6. SOLAR ELECTRIC FLYTHROUGH TRAJECTORIES WITH JUPITER GRAVITY-ASSIST.

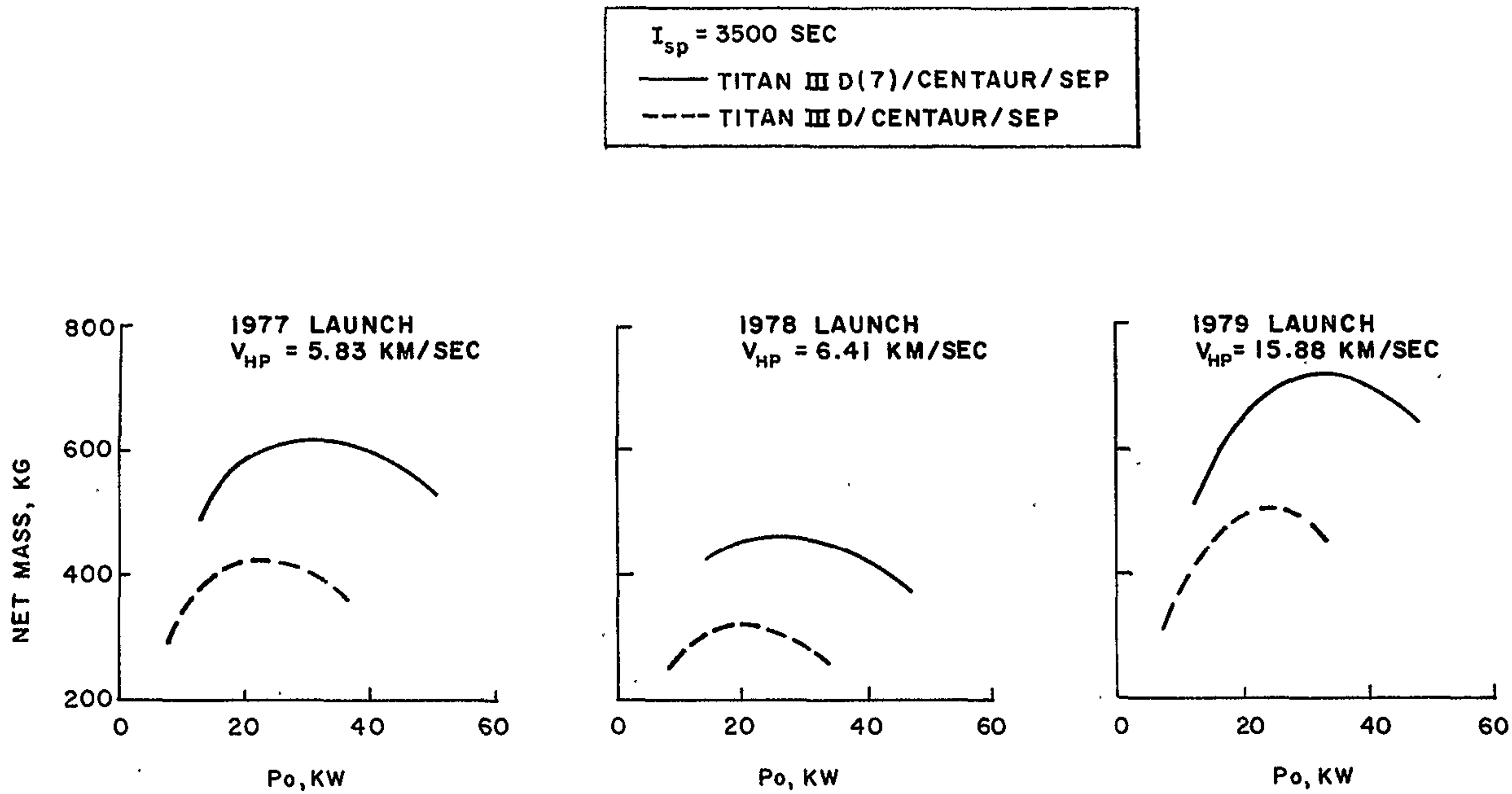

FIGURE 5-7. SOLAR ELECTRIC CAPABILITY FOR JUPITER-ASSISTED FLYTHROUGH MISSIONS TO HALLEY'S COMET.

TITAN IIID(7)/CENTAUR/SEP
I_{sp} = 3500 SEC
JUPITER SWINGBY
11/3/79, 19 KM/SEC
HALLEY ARRIVAL
T_p -50^d

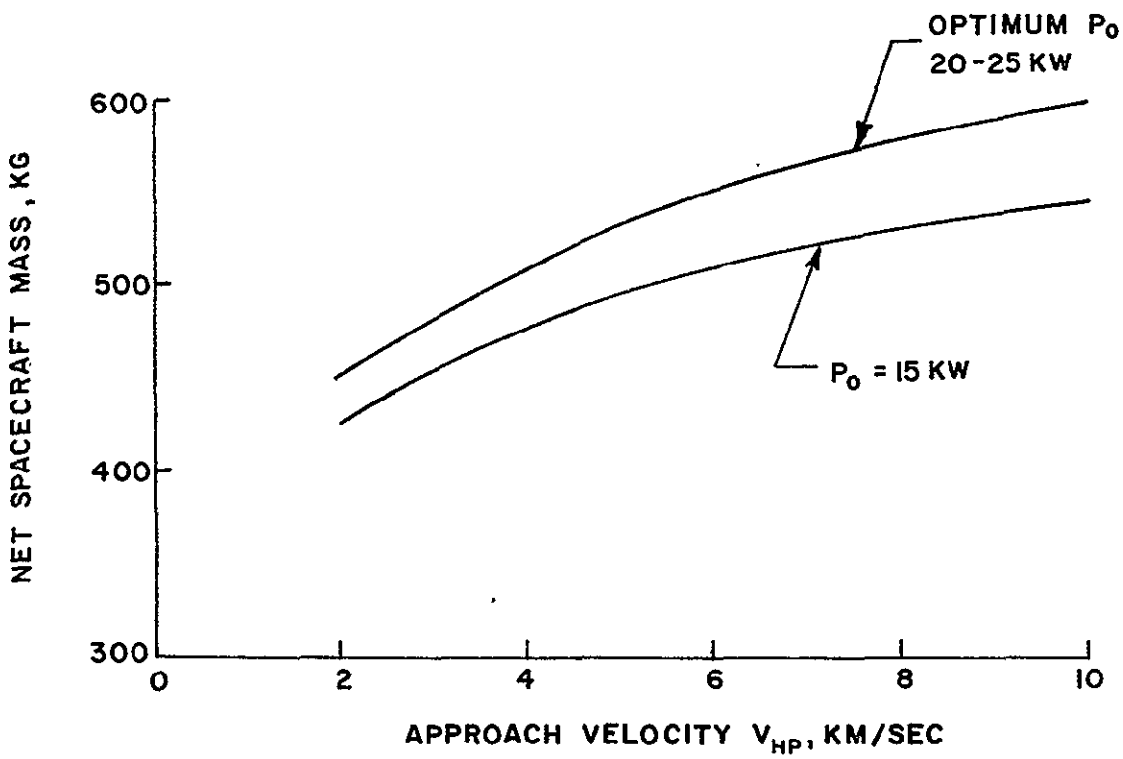

FIGURE 5-8. SOLAR ELECTRIC CAPABILITY FOR HALLEY'S COMET FLYTHROUGH, 1978 LAUNCH OPPORTUNITY WITH SEP USED AFTER JUPITER SWINGBY.

at least 450 kg for flythrough velocities as low as 2 km/sec. If power is constrained to 15 kw then the limiting velocity is about 3 km/sec. Propulsion on-time for the Jupiter-Halley leg is about 1300 days.

The above discussion has described two types of Jupiter-assisted flythrough missions which offer a choice of sorts. If SEP is not utilized beyond Jupiter then the flythrough velocity is about 6 km/sec and the Halley encounter is rather early. If SEP is used, the optimum arrival date tends to advance toward perihelion but a long propulsion period is required to obtain further velocity reduction. Since the total flight time is quite long in either case, it may be that neither of these missions is particularly attractive. Rather, one would prefer to go "all the way" and achieve rendezvous conditions, if possible.

5.3 Jupiter-Assisted Rendezvous Missions

Trajectory energy requirements for the 1977 launch opportunity are shown in Figure 5-9. The optimum swingby date is approximately 9/29/78 for a 400 day transfer to Jupiter arriving with a hyperbolic velocity of 19 km/sec. The J_{VT} requirement on this Earth-Jupiter leg is 2.9 m^2/sec^3. For the Jupiter-Halley leg we see the familiar characteristic of decreasing J as the arrival date approaches perihelion. At 50 days before perihelion the total energy requirement for both legs is 5.7 m^2/sec^3; this value should allow adequate payload capability.

Payload data is shown in Figure 5-10 assuming the Titan IIID(7)/Centaur launch vehicle. Optimum SEP power is above 25 kw and tends to be larger for early arrival dates.

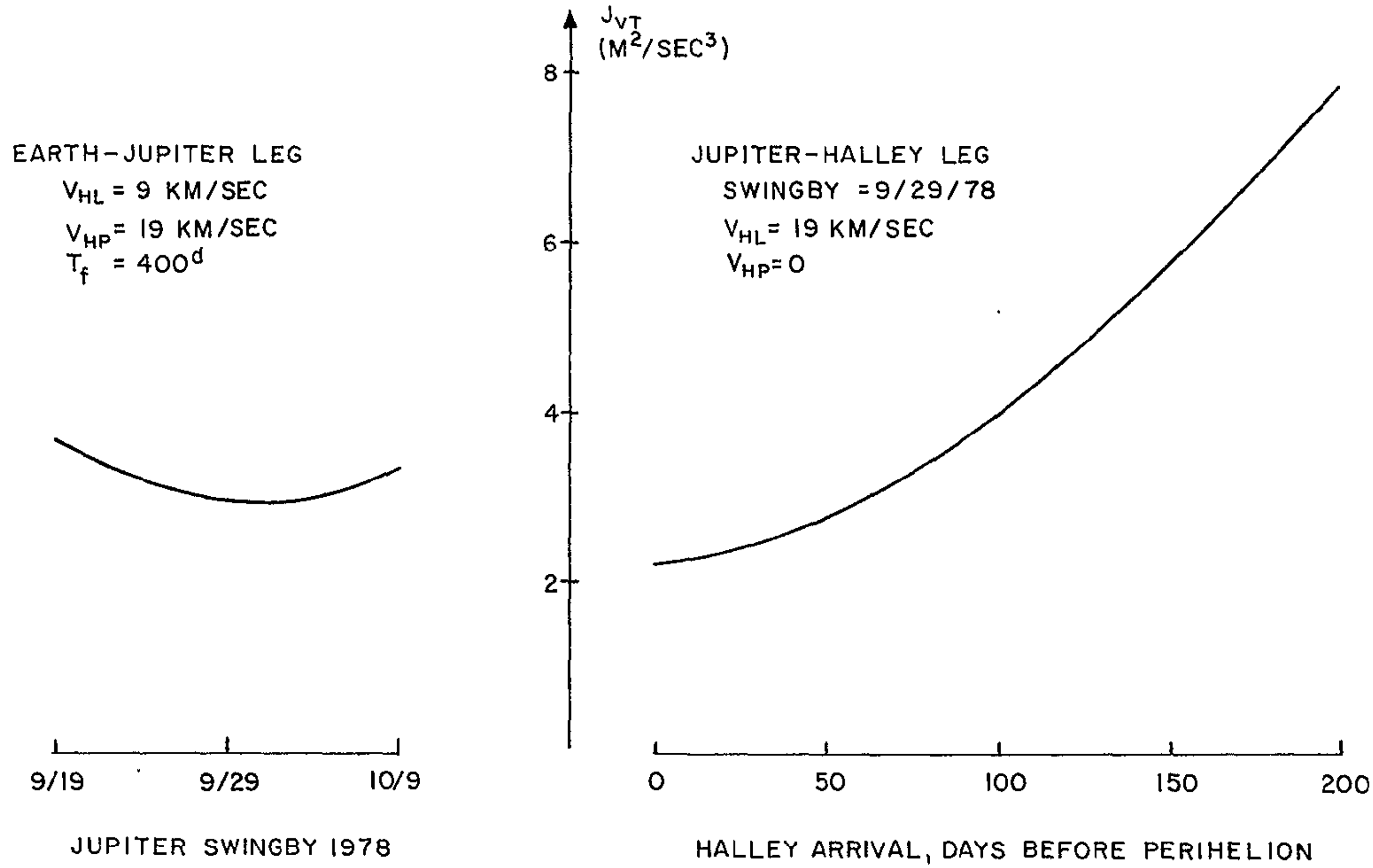

FIGURE 5-9. TRAJECTORY ENERGY REQUIREMENTS FOR JUPITER-ASSISTED RENDEZVOUS MISSION, 1977 LAUNCH OPPORTUNITY

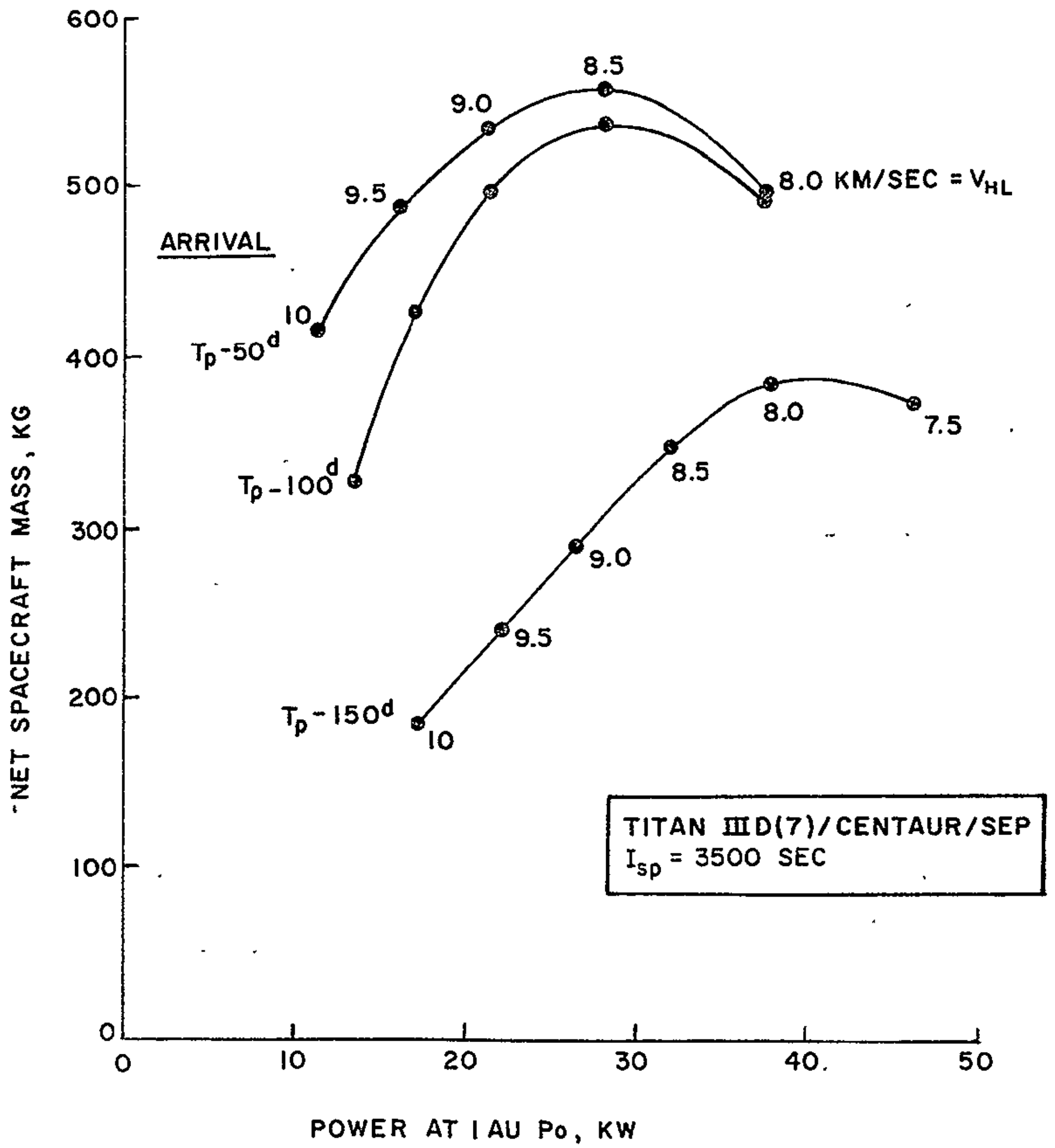

FIGURE 5-10. SOLAR ELECTRIC CAPABILITY FOR JUPITER-ASSISTED RENDEZVOUS MISSIONS TO HALLEY'S COMET, 1977 LAUNCH OPPORTUNITY.

The mission arriving 150 days before perihelion would have inadequate payload even at optimum power. At 15 kw, the earliest arrival date for a 450 kg payload is about 50 days before perihelion.

Figure 5-11 illustrated the Jupiter-assisted trajectory profile for the 1977 launch and the T_p - 50^d arrival date. Pericenter distance at swingby is 7.22 Jupiter radii. The particular example is for a hyperbolic launch velocity of 9.6 km/sec and a 15 kw powerplant. Total propulsion on-time is 1326 days; 324 days on the Earth-Jupiter leg and 1002 days on the Jupiter-Halley leg. There is a 1638 day coast period after Jupiter departure. Further data on this mission profile will be given in Section 6 of this report.

Launches in 1978 and 1979 were also investigated in the hope of obtaining adequate payload for shorter flight times. The 1978 launch may be the best programmatic opportunity inasmuch as it falls between the two Grand Tour years and would not have to compete for launch pad operations. Trajectory data for these opportunities as well as 1977 are given in Table 5-2.

Figure 5-12 summarizes the effect of launch year, arrival date and SEP power on payload capability. The best arrival date for the 1978 launch opportunity is near perihelion where the maximum payload is about 425 kg at a power input of 20 kw. For the 1979 launch, the best arrival date is about two months after perihelion where the maximum payload is just under 400 kg at a power input of 17 kw. It should be noted that a further attempt to optimize the 1978 and 1979 missions could possibly yield some performance improvements. For example, by changing the specific impulse to 3000 seconds, the 1978 launch maximum payload was found to increase by about 15 kg for the

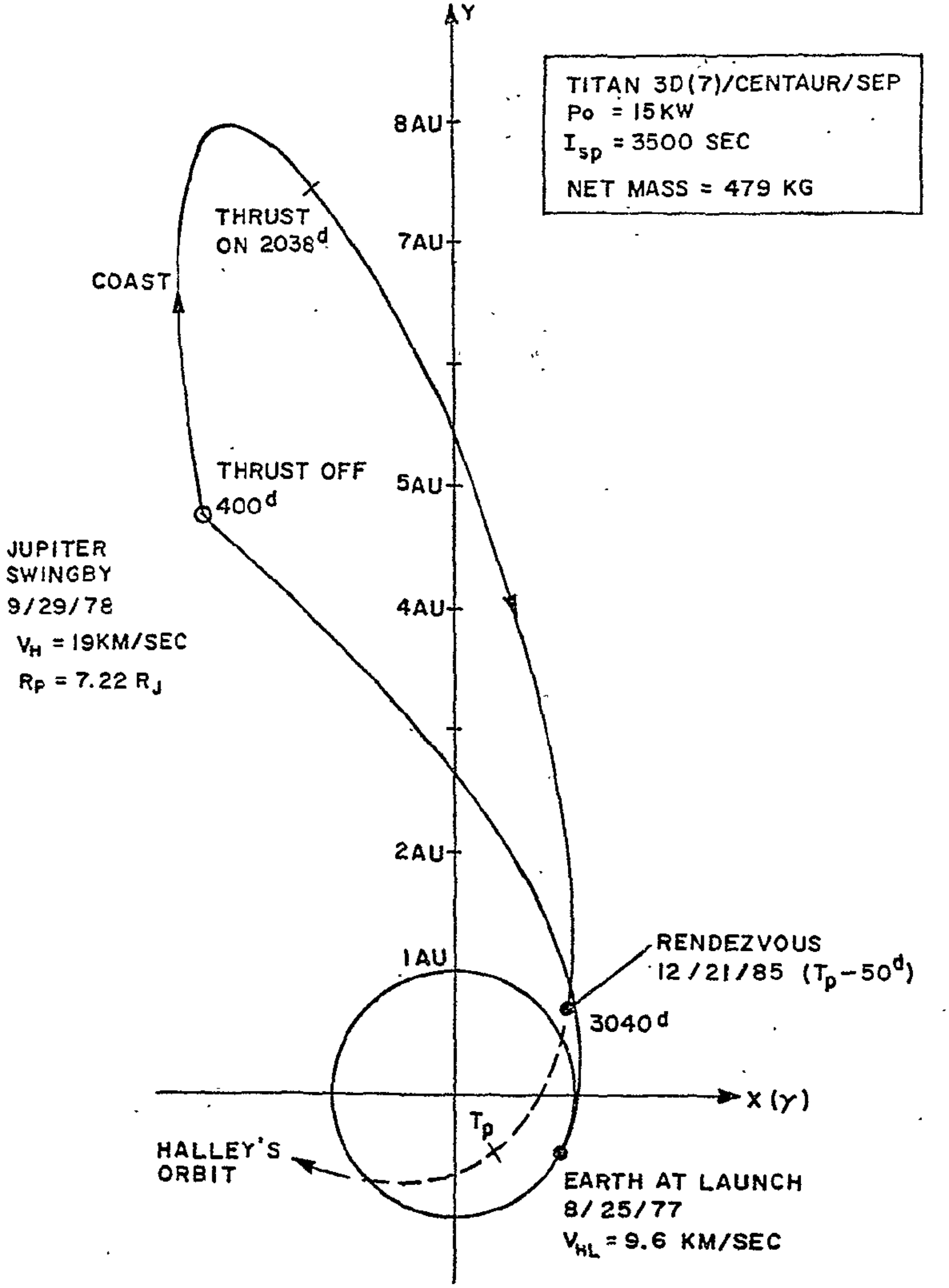

FIGURE 5-11. SOLAR ELECTRIC RENDEZVOUS TRAJECTORY WITH JUPITER GRAVITY-ASSIST.

TABLE 5-2

TRAJECTORY CHARACTERISTICS OF JUPITER-ASSISTED HALLEY RENDEZVOUS MISSIONS

	1977 LAUNCH	1978 LAUNCH	1979 LAUNCH
LAUNCH DATE	8/25/77	9/29/78	11/3/79
TIME TO JUPITER (days)	400	400	410
SWINGBY DATE	9/29/78	11/3/79	12/17/80
HYPERBOLIC VELOCITY (km/sec)	19	19	19
ASYMPTOTE DEFLECTION (deg)	48.4	54.0	61.6
PERICENTER DISTANCE (R_J)	7.09	5.90	4.67
HALLEY ARRIVAL (from T_p)	-50^d	0^d	$+50^d$
TOTAL FLIGHT TIME (years)	8.33	7.35	6.40
TOTAL ENERGY PARAMETER (m^2/sec^3) J_{VT} at V_{HL} = 9 km/sec	5.69	5.57	7.32

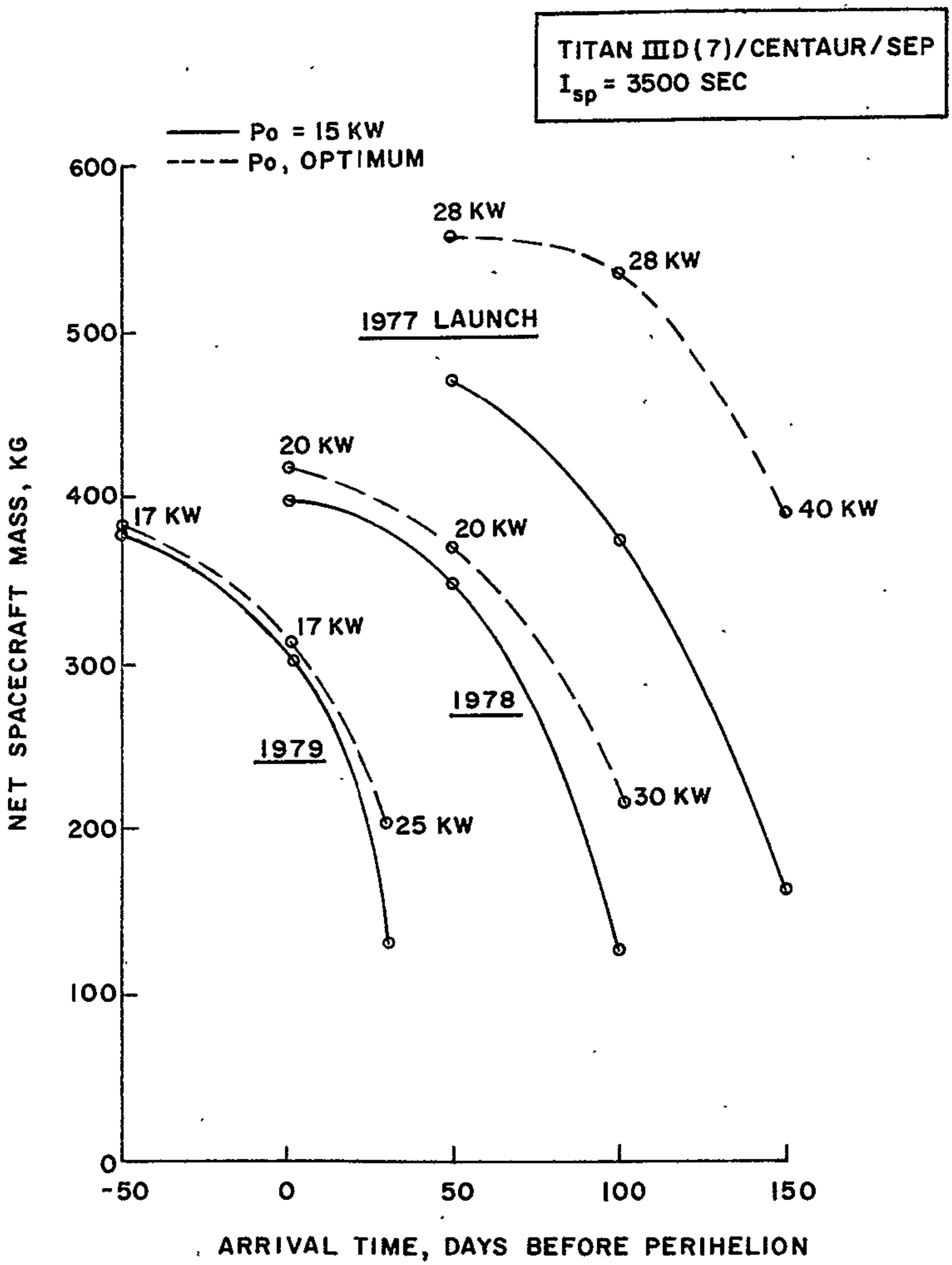

FIGURE 5-12. EFFECT OF LAUNCH YEAR AND ARRIVAL TIME FOR JUPITER-ASSISTED SOLAR ELECTRIC RENDEZVOUS WITH HALLEY'S COMET.

perihelion arrival date. Admittedly, this improvement is not very significant. Another possibility would be to adjust the Jupiter swingby velocity--probably to a lower value. This was not investigated in the present study. These suggestions have the flavor of trying to squeeze out the last kilogram of payload. Perhaps the margin of safety is simply not large enough to justify the effort to make the 1978 or 1979 opportunities more attractive. In the final analysis, the 1977 launch opportunity is clearly the best case.

5.4 Earth Swingby Mode (Earth - Earth Transfer)

A novel and unconventional mission mode has been proposed as one possible means of improving SEP payload capability for interplanetary missions (Meissinger 1970). The spacecraft is launched from Earth at some hyperbolic velocity V_{HL} (0) and arrives back at Earth 6 months later at an increased hyperbolic velocity V_{HL} (1). Thrust is applied in a direction normal to the orbital plane (north or south) and then reversed for the return to Earth. A short coast period may be included between the ascending and descending nodes where thrust has little effect on inclination. The resulting Earth - Earth transfer is approximately 1 a.u. from the Sun and inclined relative to the ecliptic plane. At Earth swingby the velocity gained by out-of-plane thrusting is converted into a velocity increment that adds a positive or negative energy change to the subsequent mission phase depending on the direction of swingby. For outer planet missions the spacecraft would pass behind the Earth in order to add heliocentric energy.

It should be made clear that this Earth swingby concept differs from the usual "free" gravity-assist obtained at other planets. The gain in hyperbolic velocity is obtained by expending propulsive energy which is then converted into a suitably oriented velocity increment via the Earth swingby. In effect, this mission mode may be compared to the indirect type of heliocentric transfers which initially swing inside the Earth's orbit before traveling to the outer planets. It may also be likened to the Earth spiral escape mode. In each of these cases the initial mission phase makes effective use of the SEP system at a more favorable solar distance. The penalty paid for increased payload capability is twofold: (1) the flight time is increased and (2) the required power level is higher. The latter penalty is much more significant in that the power requirement may be higher than the current goal of 10-20 kw.

There is another serious penalty incurred that makes the application of this concept questionable or ineffective for the Halley mission. This penalty is again twofold. First, the out-of-plane Earth - Earth transfer requires that the initial launch velocity V_{HL} (0) be directed normal to the ecliptic; i.e., at high departure declination. Assuming that the Eastern Test Range launch safety requirements constrain the maximum northern azimuth to 45°, then an azimuth penalty of about 120 m/sec must be applied to the launch vehicle performance curve. This penalty is somewhat minor compared to the second effect which relates to incomplete conversion of the swingby potential due to the hard constraint of minimum swingby distance (1 Earth radii). The approach asymptote of the Earth swingby is essentially normal to the ecliptic plane. However, for missions to the outer planets and to Halley, the Earth departure asymptote should lie in or near the ecliptic plane. Therefore, the required planetocentric deflection angle is essentially 90°, -- certainly no less than 80°. With reference to Figure 5-2 it is seen that the maximum deflection angle is less than 90° for hyperbolic velocities greater than 5.1 km/sec. In other words, at the higher range of velocity the swingby maneuver becomes rapidly less effective due to the inability of obtaining the full 90° deflection angle. This penalty is charged against the departure hyperbolic velocity as a cosine loss.

Following Meissinger, the net advantage of the swingby mission mode is shown in Figure 5-13 in terms of the effective gain in launch vehicle performance. The three effects of launch azimuth penalty, pre-encounter low thrust propellant expenditure, and incomplete velocity deflection are taken into account. This particular example assumes a 3 km/sec gain in hyperbolic velocity magnitude due to pre-encounter thrust which translates into a propellant expenditure of about 8.7 percent at 3500 sec specific impulse. The performance gain could be quite significant but it is limited to the hyperbolic velocity range from 3 to 7 km/sec. Previous results have shown that the high-energy Halley missions require a launch velocity between 9 and 10 km/sec if practical sized SEP powerplants are to be utilized with Titan/Centaur vehicles. Hence, the Earth swingby mode does not appear to be apropos for Halley missions.

As an example calculation, consider the 2700 day direct mode rendezvous mission. Figure 4-17 has shown that the maximum payload capability of the Titan IIID(7)/Centaur/SEP is 225 kg at P_o = 46.5 kw and V_{HL} = 7 km/sec. With Earth swingby at V_{HL} (1) = 7 km/sec (V_{HL} (0) = 4 km/sec), the net spacecraft mass is increased to 292 kg and P_o increases to 60 kw. Alternatively, the former Titan IIID(7) capability could be achieved with the Titan IIID if Earth swingby is employed. However, the mission remains unattractive in either case.

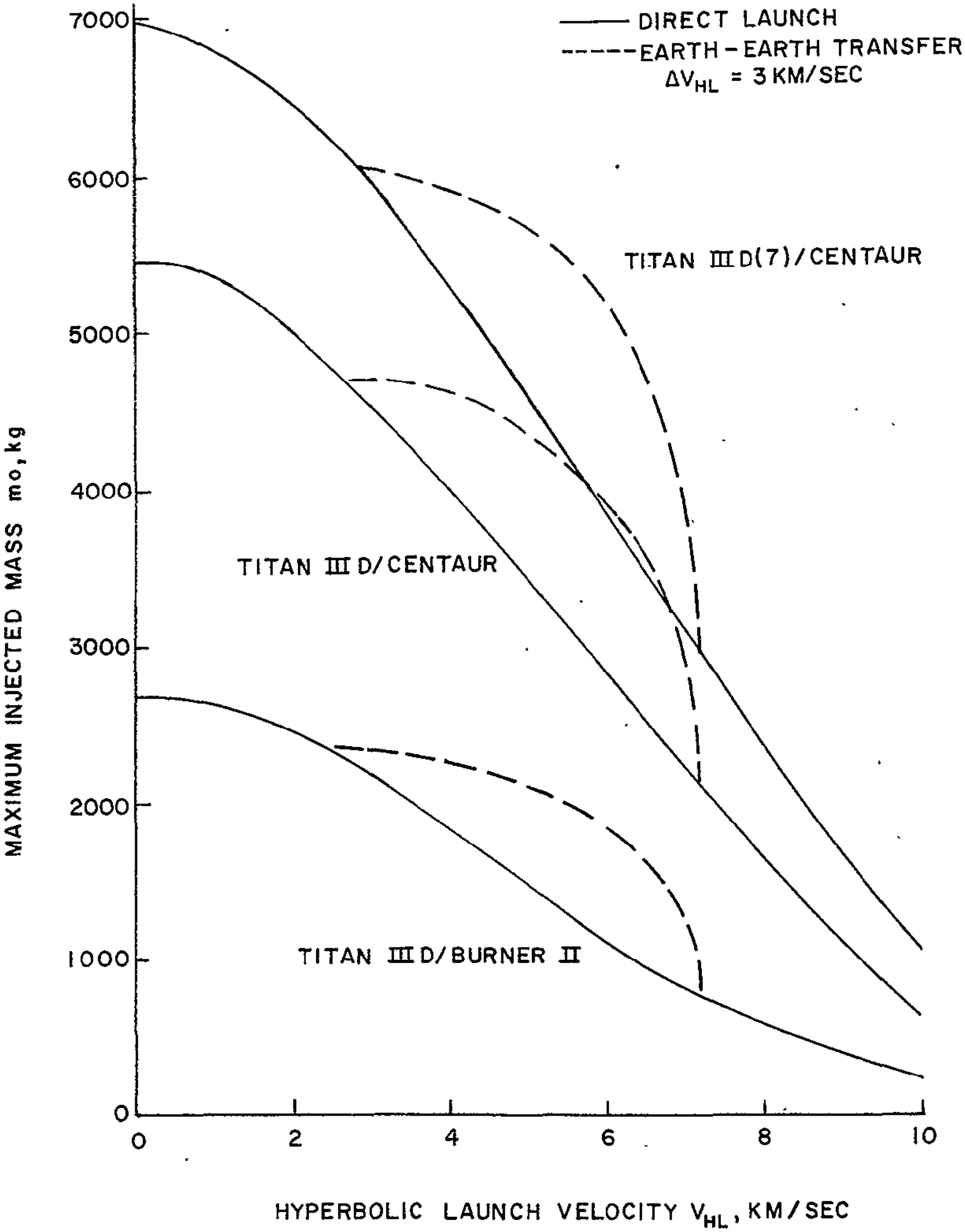

FIGURE 5-13. EFFECTIVE GAIN IN LAUNCH VEHICLE PERFORMANCE USING EARTH SWINGBY MODE

6. BASELINE MISSION SELECTIONS -- PROFILE DATA

The purpose of this final section of the report is to sort out some of the study results described in the preceeding sections. In addition, supporting data on power profiles, propulsion time, thrust angles and the effect of changing the specific impulse will be presented. Three mission examples are selected for this purpose: (1) 900 day direct mode flythrough, (2) 2540 day Jupiter-assisted flythrough, and (3) 3040 day Jupiter-assisted rendezvous. Tables 6-1 and 6-2 show the relevant trajectory data and vehicle mass breakdown and are presented without discussion. Figure 6-1 has been prepared to show, in a general way, the expected communication and tracking coverage at encounter of the Deep Space Network for any Halley mission. The $\pm$ 35° latitude selection is representative of the three DSN sites in the Northern and Southern hemispheres. It is seen that essentially 24 hours of overlapping coverage per day is possible except in the region 50-80 days after perihelion when the comet is at high southern declination (20°-50°) and not readily viewed from either Goldstone or Madrid.

6.1 900 Day Direct Flythrough

The trajectory profile for this mission is essentially the same as that shown in Figure 4-4 (page 30). For the specific baseline mission parameters the total propulsion on-time is 668 days. There is an initial coast period of 21 days and a 211 day coast period beginning 134 days after launch. Initial thrust acceleration is 3.94×10^{-4} m/sec^2. The solar power profile is shown in Figure 6-2. A possible thruster array configuration would be five 2.5 kw rated modules with one in spare. Assuming a 2:1 throttling capability, the first propulsion period would have 4 operating thrusters. The second

TABLE 6-1

TRAJECTORY CHARACTERISTICS OF BASELINE MISSION SELECTIONS

	MISSION NO. 1	MISSION NO. 2	MISSION NO. 3
MISSION MODE	Direct Flythrough	Jupiter-Assisted Flythrough	Jupiter-Assisted Rendezvous
LAUNCH VEHICLE	Titan IIID/BII	Titan IIID(7)/Cent	Titan IIID(7)/Cent
LAUNCH DATE	3/17/83	9/29/78	8/25/77
FLIGHT TIME (years)	2.46	6.95	8.33
SEP POWER (kw)	10	15	15
SPECIFIC IMPULSE (sec)	3500	3500	3500
LAUNCH VELOCITY (V_{HL}, km/sec)	6.4	10.0	9.6
JUPITER PERICENTER (R_J)	--	4.96	7.22
HALLEY ARRIVAL (from T_p)	-160^d	-150^d	-50^d
APPROACH VELOCITY (V_{HP}, km/sec)	24.0	6.41	0
ENERGY PARAMETER J_{CISP} (m^2/sec^3)	3.36	3.36	5.69
TOTAL PROPULSION TIME (days)	668	228	1326

TABLE 6-2

VEHICLE MASS CHARACTERISTICS OF BASELINE MISSION SELECTIONS

	MISSION NO. 1	MISSION NO. 2	MISSION NO. 3
MISSION MODE	Direct Flythrough	Jupiter-Assisted Flythrough	Jupiter-Assisted Rendezvous
LAUNCH VEHICLE	Titan IIID/BII	Titan IIID(7)/Cent	Titan IIID(7)/Cent
INITIAL MASS (kg)	970	1030	1280
SEP POWERPLANT (kg)	300	450	450
PROPELLANT (kg)	193	154	341
TANKAGE (kg)	6	5	10
NET SPACECRAFT (kg)	471	421	479

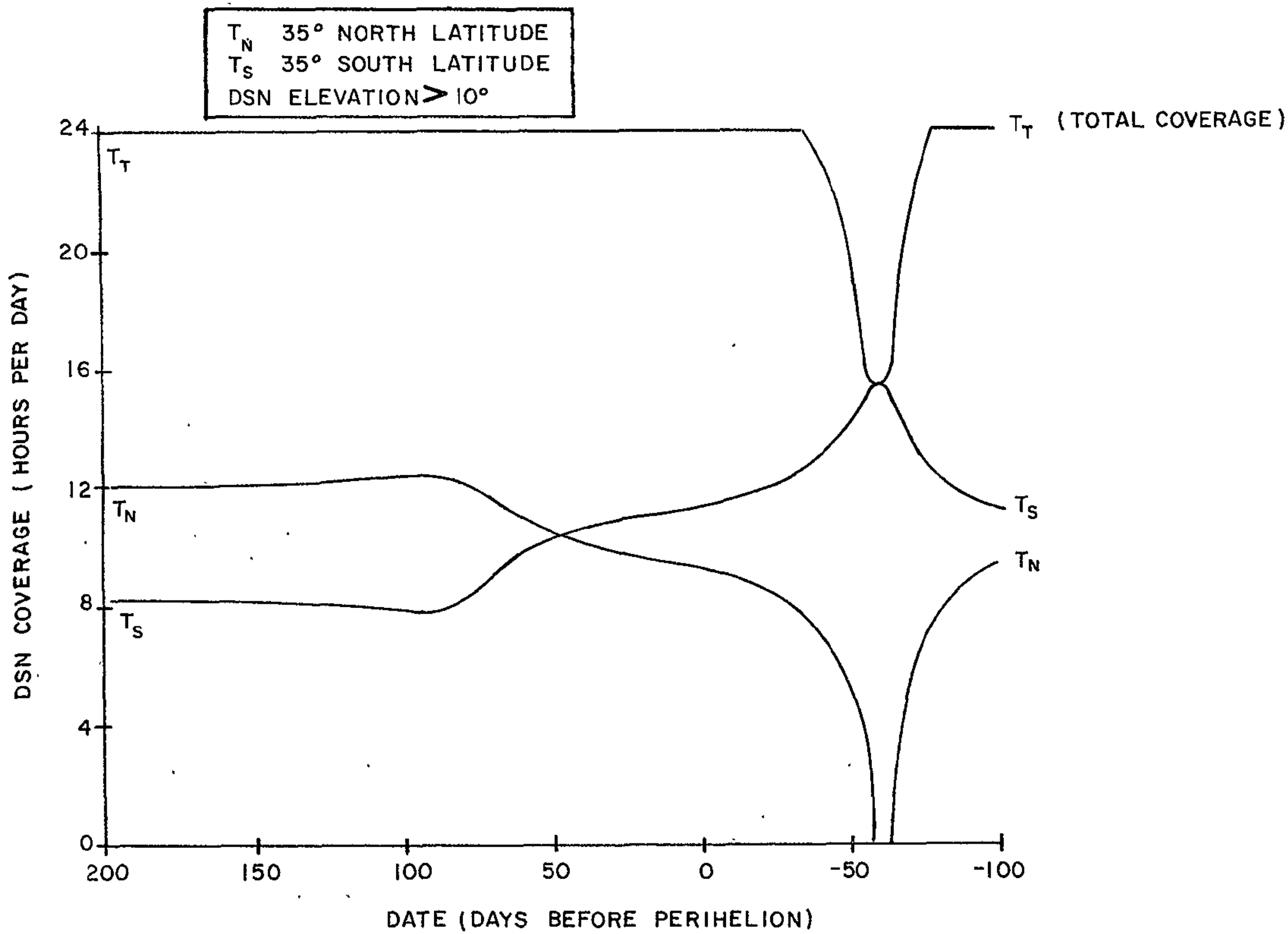

FIGURE 6-1. DEEP SPACE NETWORK COVERAGE FOR HALLEY ENCOUNTERS

propulsion period would have a single thruster operating at any time (not necessarily the same thrusters) and have 4 units in standby.

Figure 6-3 shows the cone and clock angles for the optimum thrust vector program. The cone angle is the displacement of the thrust vector from the solar direction and, hence, bears directly on the problem of mechanizing the steering program in the face of a solar array pointing constraint. The simplest mechanization is to fix the thruster array at 90° from the solar array pointing axis. This would incur a non-optimum performance penalty which may not be too large in this case since the optimum cone angle differs by no more than 35° from the normal direction.

Figure 6-4 shows the effect of specific impulse on propulsion time and payload. The 3500 sec baseline value is clearly not optimum. For example, if I_{sp} were 2500 sec* the maximum payload capability would be more than 700 kg. Alternatively, at the nominal 450 kg requirement the propulsion on-time can be reduced to 590 days.

6.2 2540 Day Jupiter-Assisted Flythrough

This mission is launched in Sept.-Oct. 1978 and arrives at Halley 150 days before perihelion with an approach velocity of 6.4 km/sec. The trajectory profile has been shown in Figure 5-6 (page 59). Although the SEP is not needed for thrust maneuvers after Jupiter departure the solar array can provide spacecraft power requirements. At maximum solar distance of 7.3 a.u. the array power output is about 600 watts.

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* Electron bombardment thrusters operating at an I_{sp} of 2500 sec or less are not current state-of-the-art.

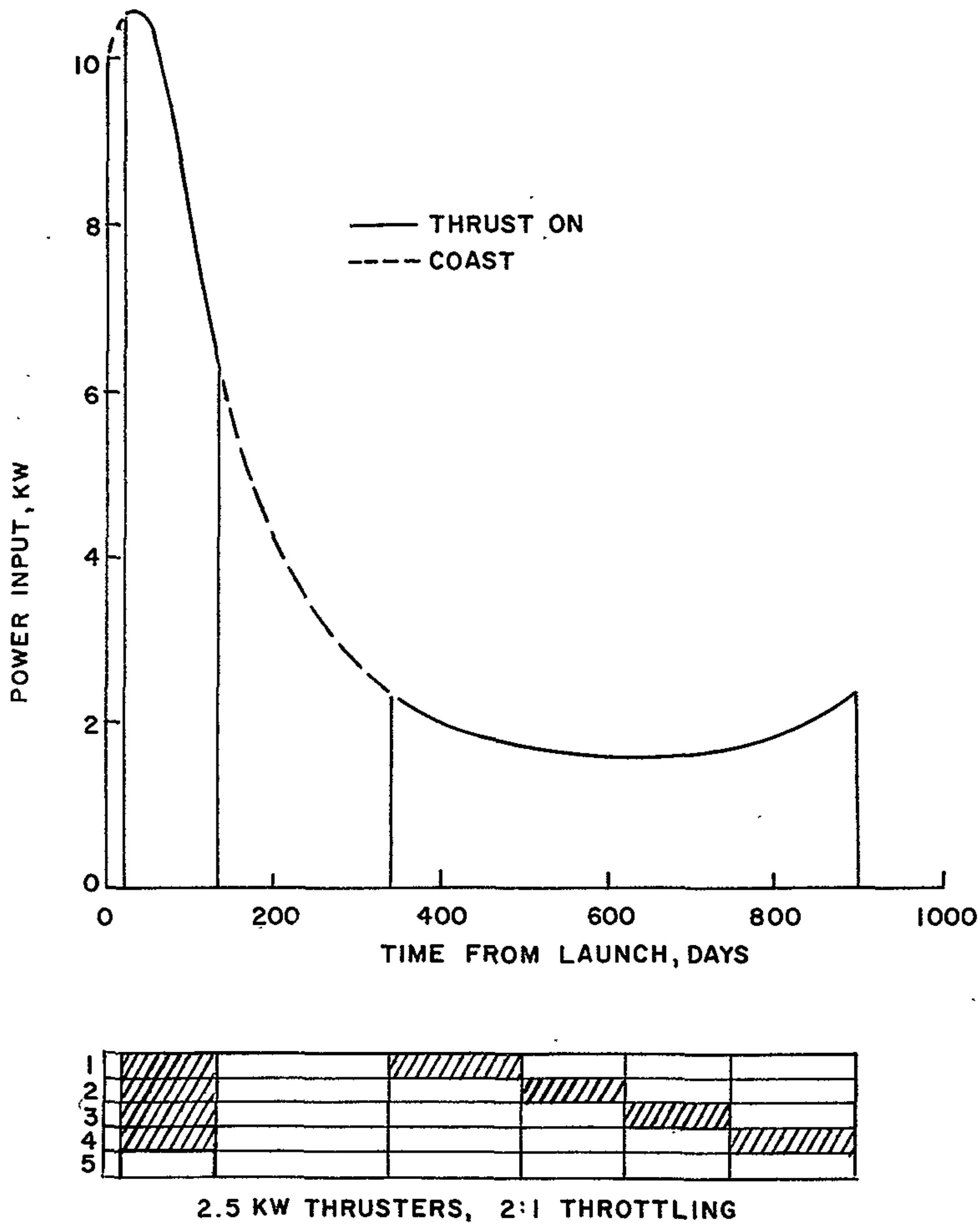

FIGURE 6-2. POWER PROFILE AND THRUSTER SWITCHING FOR 900 DAY FLYTHROUGH MISSION TO HALLEY'S COMET.

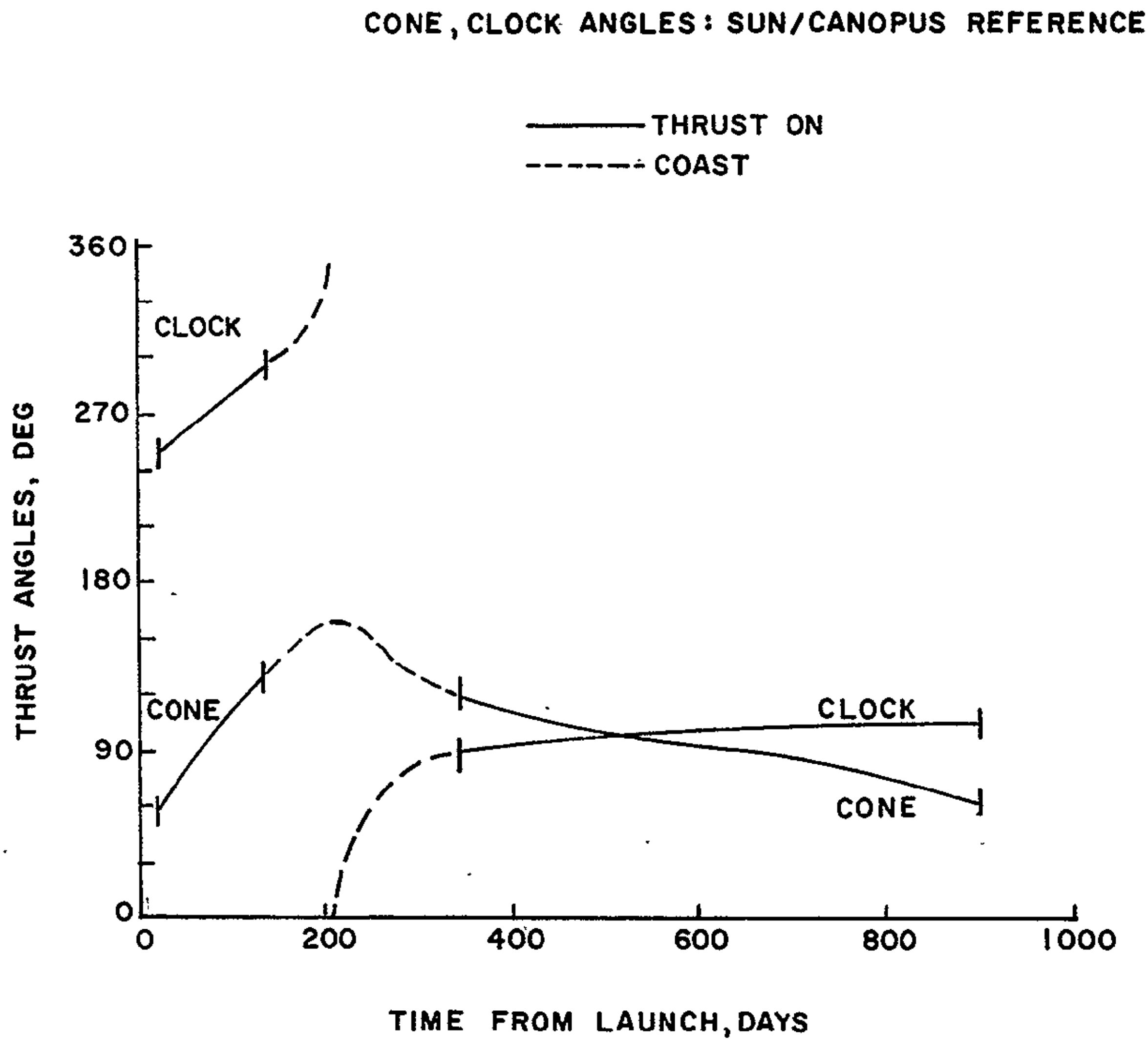

FIGURE 6-3. THRUST VECTOR PROFILE FOR 900 DAY FLYTHROUGH MISSION TO HALLEY'S COMET.

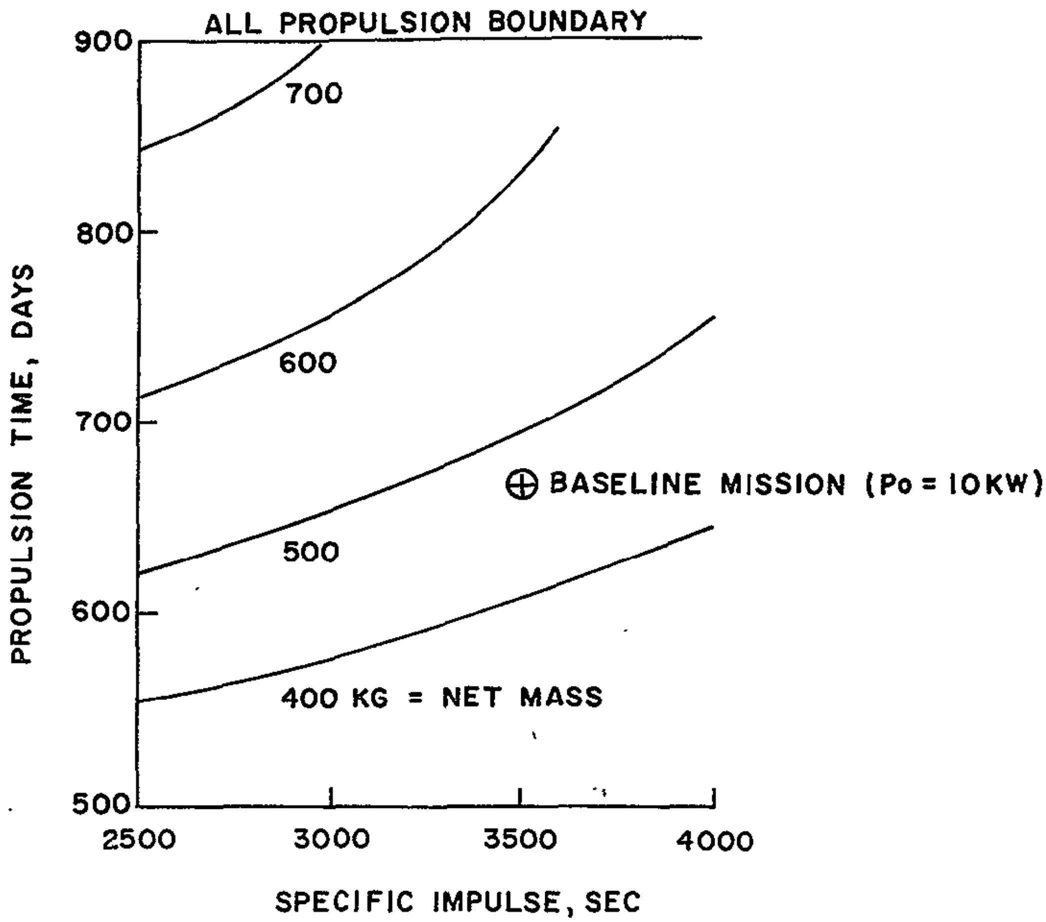

FIGURE 6-4. PERFORMANCE SENSITIVITY TO SPECIFIC IMPULSE FOR 900 DAY FLYTHROUGH MISSION TO HALLEY'S COMET.

Total propulsion on-time for the 380 day transfer to Jupiter is 228 days; there is a 152 day coast period beginning 178 days after launch. Initial thrust acceleration is 5.56×10^{-4} m/sec^2. The solar power profile is given in Figure 6-5. An example thruster switching sequence is shown assuming 2.5 kw rated thrusters with a 2:1 throttling ratio.

Clock and cone angles of the optimum thrust vector program are shown in Figure 6-6. A small positive out-of-plane component is needed as indicated by the near-270° clock angle. Cone angle differs from the normal direction by as much as 60° during the first thrust period. However, the second thrust period requires a cone angle between 10° and 20° which could be difficult to mechanize. It may be possible to redistribute the propulsion effort (by constraint) thereby eliminating the need for the second thrust period. Further investigation of this mission should look into this question and determine the payload penalty -- the penalty is likely to be small.

Figure 6-7 shows the effect on propulsion time and payload when specific impulse is varied. The baseline payload is seen to be somewhat shy of 450 kg when I_{sp} is 3500 sec. If instead an I_{sp} of 3000 sec were employed the payload would increase to between 450 and 500 kg. Although propulsion time reduction also could be obtained at lower I_{sp} operation, it is the payload gain which is most important for this mission.

6.3 3040 Day Jupiter-Assisted Rendezvous

The ability to achieve rendezvous conditions with SEP makes this mission the most interesting. True, it is not a particularly attractive mission from a design standpoint because of the very long 8.3 year flight duration (1977 launch requirement). The trajectory profile has been shown in

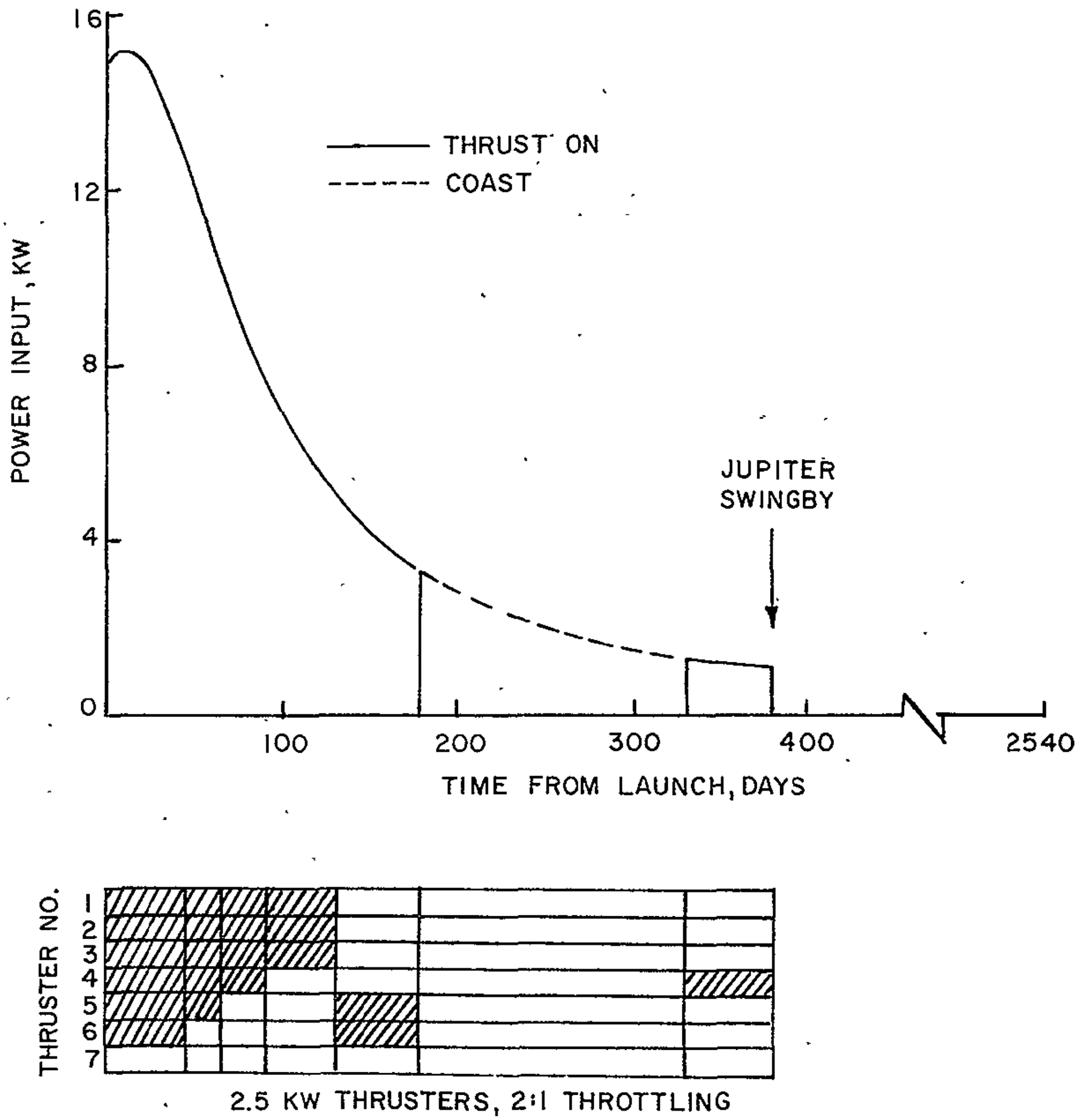

FIGURE 6-5. POWER PROFILE AND THRUSTER SWITCHING FOR JUPITER-ASSISTED FLYTHROUGH MISSION TO HALLEY'S COMET

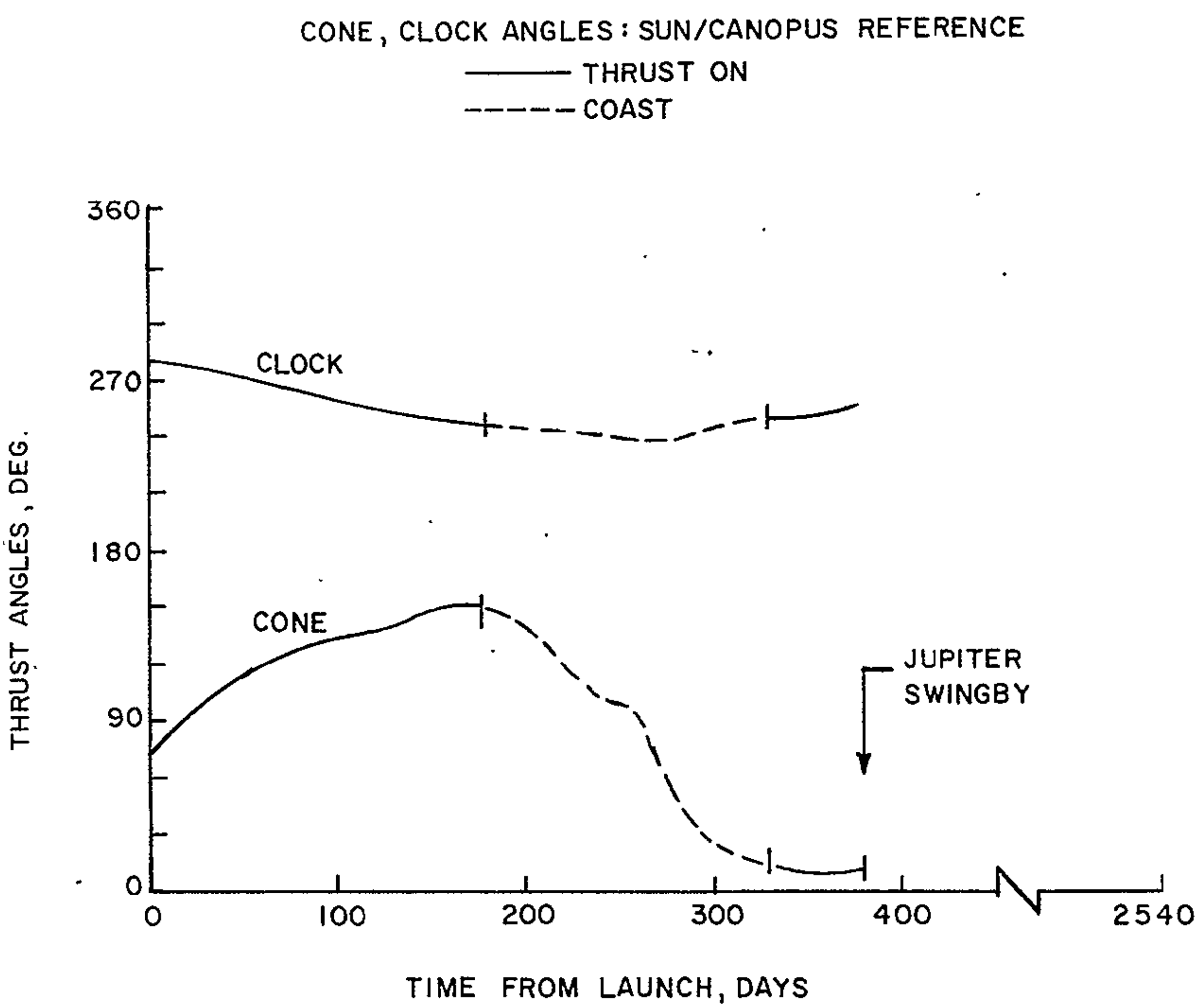

FIGURE 6-6. THRUST VECTOR PROFILE FOR JUPITER-ASSISTED FLYTHROUGH MISSION TO HALLEY'S COMET

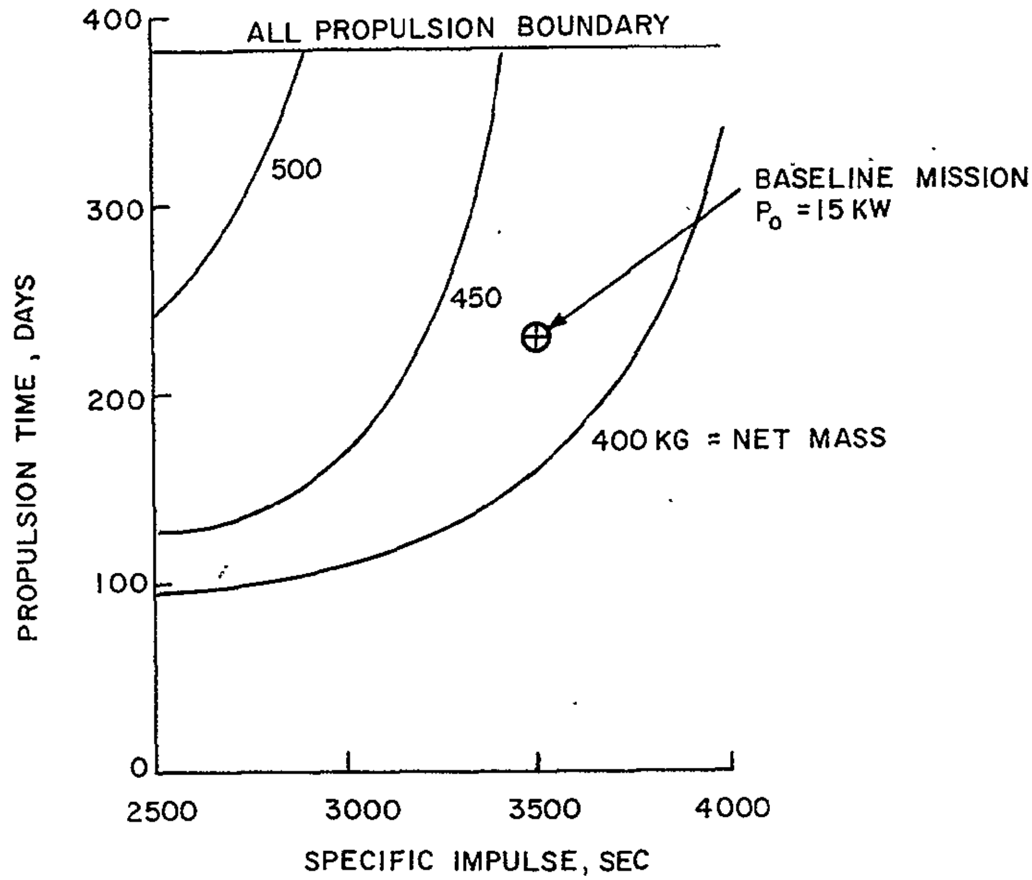

FIGURE 6-7. PERFORMANCE SENSITIVITY TO SPECIFIC IMPULSE FOR JUPITER-ASSISTED FLYTHROUGH MISSION TO HALLEY'S COMET

Figure 5-11 (page 66). Initial thrust acceleration is 4.47×10^{-4} m/sec^2. Total propulsion time is 1326 days; 324 days on the Earth-Jupiter leg and 1002 days on Jupiter-Halley leg. The spacecraft coasts for 1638 days after leaving Jupiter. The power profile shown in Figure 6-8 points out the potential difficulty in matching the power variation with a suitable thruster switching sequence. The minimum power input is about 550 watts*; this occurs when the final propulsion period is initiated at a solar distance of 7.65 a.u. Several small thruster units rated at about 0.4 km may be required during the interval 2038 - 2700 days after launch.

Figure 6-9 shows the optimum thrust angle program. The cone angle is substantially above 90° during the transfer to Jupiter which indicates mechanization difficulty. However, during the approach to Halley the cone angle is nearly constant at about 70°. The out-of-plane thrust component is small during the Jupiter transfer except during the short thrust reversal interval near 300 days (indicated by spike). During most of the Halley approach the out-of-plane and in-plane thrust components are nearly equal.

The effect of specific impulse on propulsion time and payload is shown in Figure 6-10. Here again lower I_{sp} would effect a small payload increase relative to the baseline value.

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* This is a potential problem area inasmuch as the solar array should also provide in-transit spacecraft subsystem power requirements (perhaps 200-400 watts). Any further study of this mission should account for an auxiliary power requirement by deleting this amount from the available propulsion power, or alternatively, by reducing the payload mass by an amount equal to the auxiliary system (e.g., RTG's).

More importantly, it would allow a substantial reduction in propulsion on-time. For example, if a 2500 sec I_{sp} design could be achieved, the same payload of 480 kg could be delivered with a propulsion time of about 950 days; this is a reduction of almost 30 percent compared with the baseline time of 1326 days.

Finally, the sensitivity of payload to launch date variations is shown in Figure 6-11 for the baseline rendezvous mission. This calculation is made for a fixed spacecraft design and injected mass. A 30 day window incurs a penalty of only 25 kg propellant. The design payload would then be fixed at 456 kg.

6.4 Conclusions

This study has attempted to deliniate the trajectory possibilities and requirements from which mission planners may assess the preliminary feasibility of Halley missions and the potential role of solar electric propulsion. It must be admitted that an extremely attractive mission profile has not been found. That is to say, the easy high velocity flythrough missions may as well be performed ballistically, and the difficult rendezvous mission places rather severe requirements on SEP spacecraft design. The decision as to which type of mission, if any, should be programmed lies with NASA mission planners. Suffice it to say that of all periodic comets accessible to space exploration Halley's Comet is undeniably the most unique and interesting example. It is recommended that the results presented herein be used to initiate a further and more comprehensive investigation of mission feasibility from a practical design and cost standpoint.

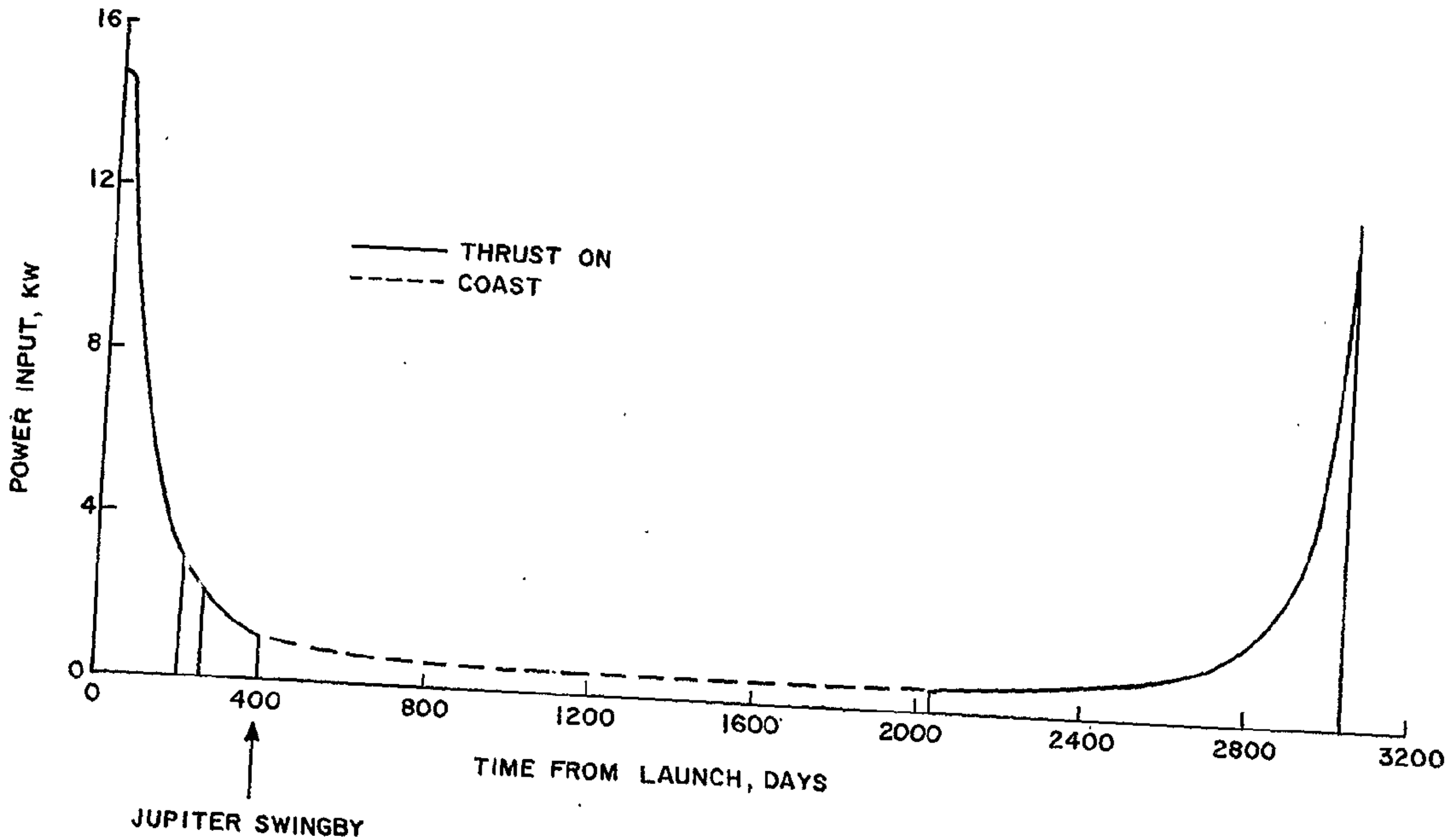

FIGURE 6-8. POWER PROFILE JUPITER-ASSISTED RENDEZVOUS MISSION TO HALLEY'S COMET.

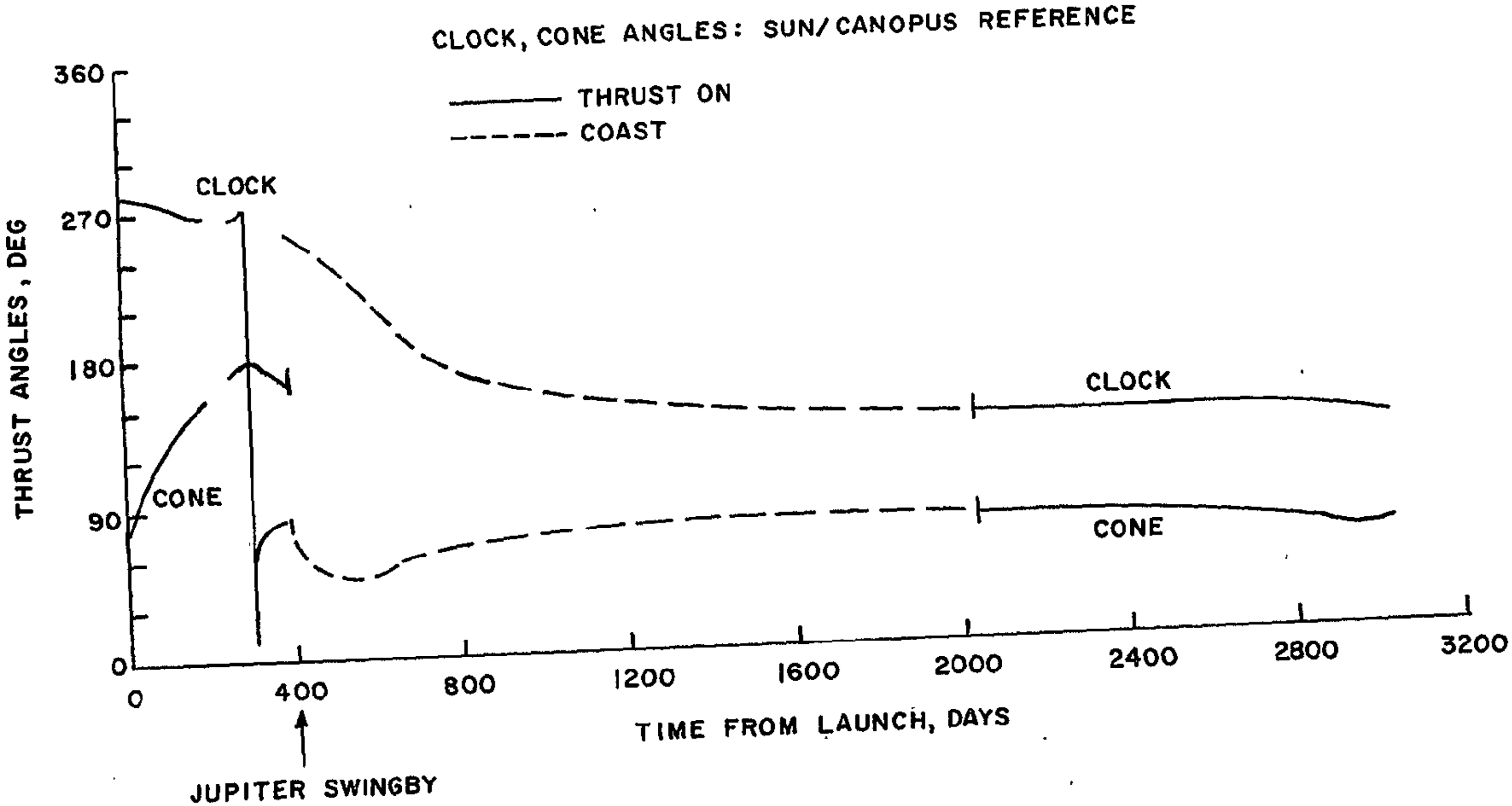

FIGURE 6-9. THRUST VECTOR PROFILE FOR JUPITER-ASSISTED RENDEZVOUS MISSION TO HALLEY'S COMET.

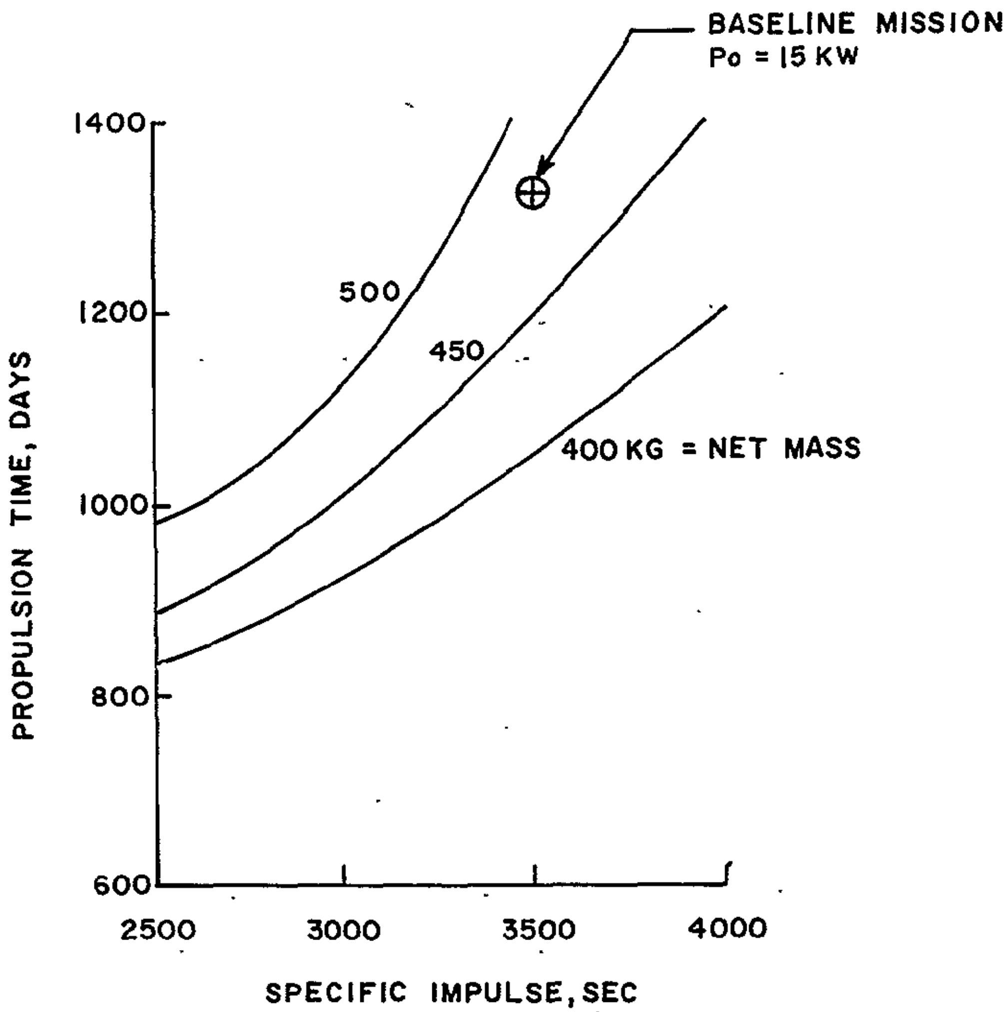

FIGURE 6-10. PERFORMANCE SENSITIVITY TO SPECIFIC IMPULSE FOR JUPITER-ASSISTED RENDEZVOUS MISSION TO HALLEY'S COMET.

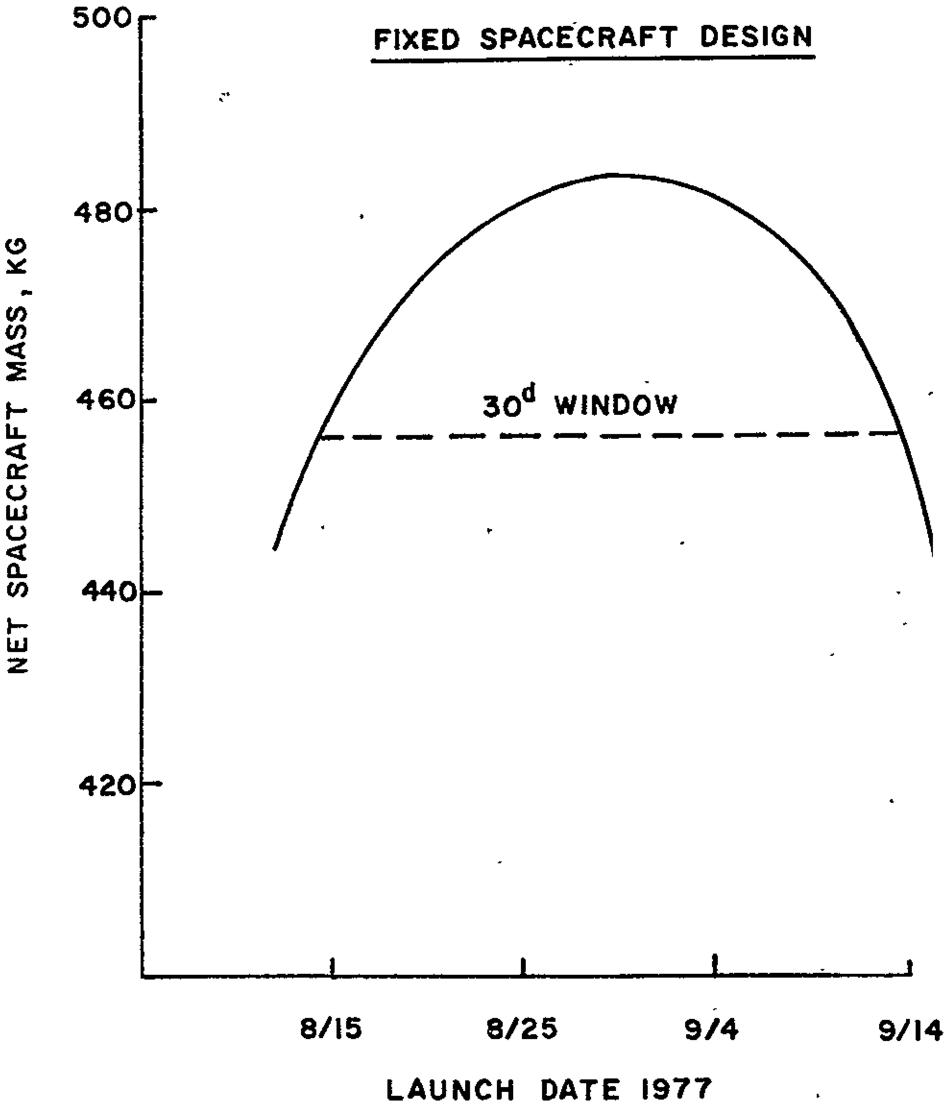

FIGURE 6-11. LAUNCH WINDOW PENALTY FOR JUPITER-ASSISTED RENDEZVOUS MISSION TO HALLEY'S COMET.

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