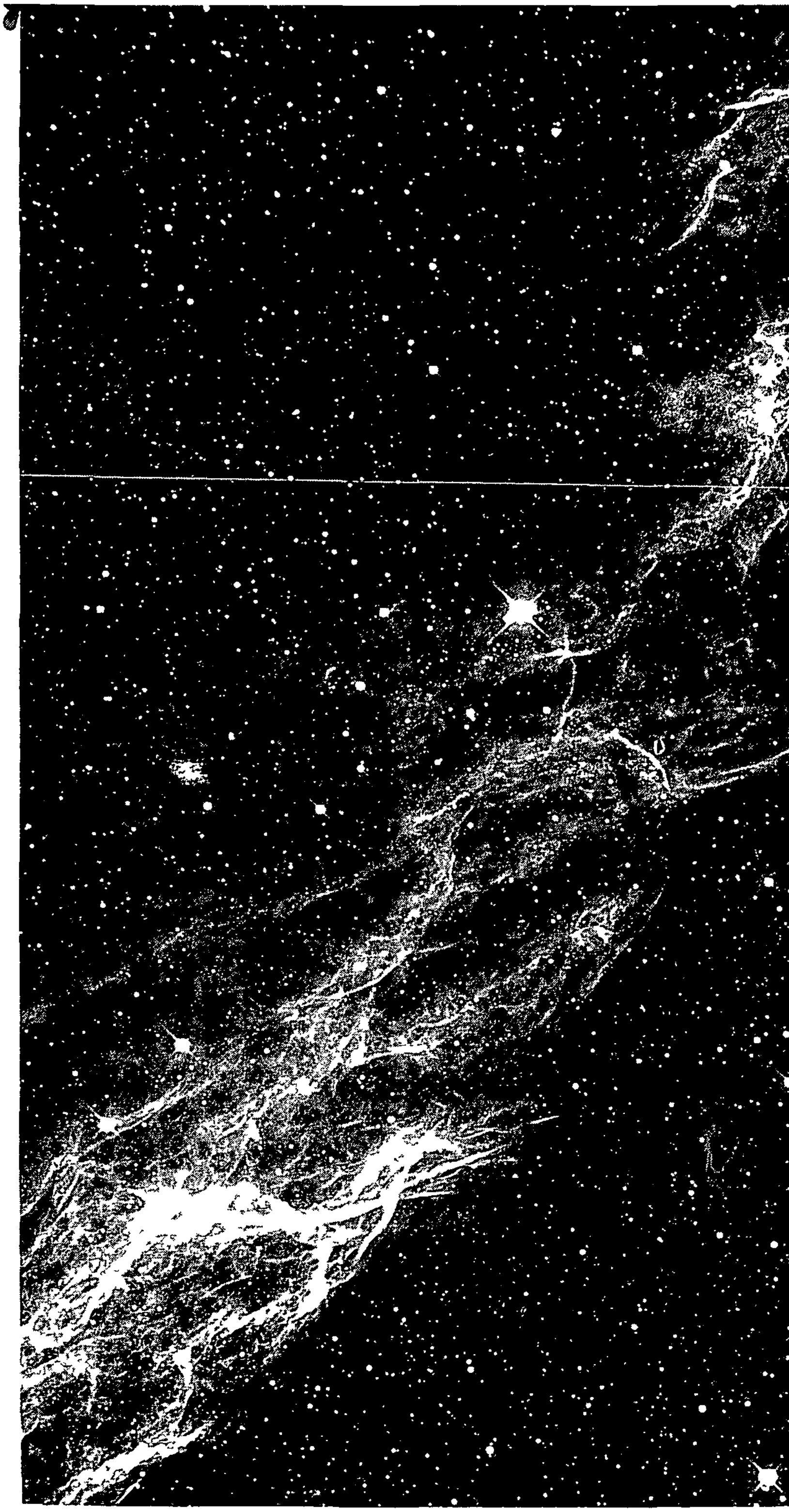
N72-18540

ASTRO SCIENCES



Report No. T-30

ASTEROID (FLORA AND EROS) SAMPLE-RETURN MISSIONS VIA SOLAR ELECTRIC PROPULSION

IIT RESEARCH INSTITUTE
10 West 35 Street
Chicago, Illinois 60616

Report No. T-30

ASTEROID (FLORA AND EROS) SAMPLE-RETURN MISSIONS VIA SOLAR ELECTRIC PROPULSION

by

A. L. Friedlander

Astro Sciences
IIT Research Institute
Chicago, Illinois 60616

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, as sponsored by the National Aeronautics and Space Administration under Contract NAS7-100, Task Order No. RD-26.

APPROVED BY:

D L Roberts

D. L. Roberts, Manager
Astro Sciences

October 1971

FOREWORD

This Technical Report is the final documentation on all data and information required by Task 8: Asteroid Sample Return. 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. Task 9 of this study will be reported separately.

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration under Contract NAS7-100.

ACKNOWLEDGEMENT

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

SUMMARY

This report describes the characteristics and capabilities of solar electric propulsion (SEP) for performing sample-return missions to the asteroids Flora and Eros. The scope of the study emphasizes trajectory/payload analysis and mission design trade-off options. Science objectives, instrumentation, operations, and SEP spacecraft design are treated only in a limited manner. The reader is referred to previous studies for a more comprehensive discussion of these important topics. Realistic subsystem weights and scaling relationships used herein are based on such independent studies.

Launch opportunities in the 1980-90 time period seem reasonable in light of current NASA program planning and SEP spacecraft development. The mission to Flora is investigated for the first three opportunities after 1980; the interval between launch periods is about 1.3 years. In the case of Eros only the 1984 opportunity is examined. Previous studies have looked at SEP missions to Eros over the period 1975-82; launch periods recur at approximately 2 year intervals. Performance variations between opportunities do not appear to be very significant for either asteroid mission when SEP is employed. In regard to mission duration, the round-trip time is rather narrowly restricted as a result of the optimum Earth-asteroid orbit geometries. Thus the Flora mission trip time is about 3.7 years and the Eros trip time is about 3 years. The Flora mission is more demanding from a propulsive standpoint since (1) it is farther out in the asteroid belt (affecting SEP requirements), and (2) it is about 1000 times more massive than Eros (affecting auxiliary chemical propulsion requirements for landing/separation maneuvers).

Major results of the study are presented as performance curves of Earth departure mass versus surface sample size. Hyperbolic launch velocity and SEP power are given parametrically along these curves. Presented in this way, the data may easily be matched with the injected payload capability of particular launch vehicles of interest. Sample return capability is shown for each of two Earth recovery options: (1) orbit capture into a 555 x 9000 km altitude orbit which would be compatible with retrieval by a manned vehicle such as the Apollo CSM or a Shuttle Orbiter System, and (2) direct Earth re-entry with air or water recovery. The direct re-entry mode yields slightly better performance (e.g., larger samples), but the question of possible back-contamination of Earth is still subject to controversy.

Characteristics of two baseline mission examples of Flora and Eros sample-return are summarized in Table S-1. Each returns a 25 kg surface sample to Earth orbit. A solid chemical retro stage (147 kg) executes the capture ΔV maneuver of 2.5 km/sec. The loaded sample capsule placed into orbit weighs 80 kg and includes necessary attitude stabilization and instrumentation to aid recovery. Spacecraft subsystems include a 450 kg bus or equipment module for such functions as communications, data handling, attitude control, etc., and a 250 kg rendezvous, docking, science and sampling module (RDSS). The RDSS module includes a basic science payload (55 kg), sample acquisition and handling devices (30 kg), navigation and guidance (31 kg), and other subsystems to accomplish asteroid docking and attachment. The mission concept assumes that the entire spacecraft including the quiescent SEP system is landed on the asteroid. Except for the sample return capsule, the RDSS module is left on the surface at separation from the asteroid.

The Flora mission is nominally launched on Dec. 15, 1982 by a Titan IIID/Centaur which injects the 2423 kg gross mass

TABLE S-1

ASTEROID SAMPLE-RETURN MISSION SUMMARY

PARAMETER	FLORA MISSION	EROS MISSION
LAUNCH DATE	DECEMBER 15, 1982	JANUARY 20, 1984
LAUNCH VEHICLE	TITAN III D/CENTAUR	TITAN III D/BII
GROSS INJECTED MASS	2423 KG	1680 KG
SEP POWER	15 KW	10 KW
SEP SPECIFIC IMPULSE	3500 SEC	3500 SEC
MISSION DURATION	1360 DAYS	1080 DAYS
ASTEROID STAY TIME	90 DAYS	120 DAYS
SEP PROPULSION TIME	923 DAYS	443 DAYS
EARTH APPROACH MASS	1240 KG	1035 KG
SAMPLE RETURNED TO EARTH ORBIT	25 KG	25 KG

to a hyperbolic escape velocity of 5 km/sec. The SEP system employs a 15 kw powerplant (power input to thrust subsystem) with ion thrusters operating at 3500 sec specific impulse. A possible design configuration would be five 3.5 kw thrusters, each having at least a 2 to 1 throttling capability, and having from 1 to 3 thrusters in standby during nominal propulsion periods. Total trip time is 1360 days which is broken down into 640 days for the outbound transfer, 90 days staytime at Flora, and 630 days for the return transfer. The nominal propulsion on-time is 923 days; 562 days on the outbound transfer and 361 days for return. The SEP system including propellant and tankage comprise 1061 kg, or about 46% of the Earth departure mass. A propellant pad sufficient for a 30-40 day launch window is allowed. The auxiliary chemical propulsion system employs liquid Earth-storable propellants and weighs 345 kg. This is utilized principally for all post-rendezvous maneuvers including landing and separation, and is sized for a total ΔV of 500 m/sec.

The Eros mission is launched on Jan. 20, 1984 by a Titan IIID/BII vehicle. A gross mass of 1680 kg is injected to a hyperbolic escape velocity of 4 km/sec. This mission being less demanding requires only a 10 kw SEP system. Four 3.5 kw thrusters would be sufficient with either one or two thrusters always in standby. Total trip time is 1080 days of which 480 days each are specified for the outbound and return transfers and 120 days for Eros encounter operations. The nominal propulsion on-time is 443 days; 276 days on the outbound transfer and 167 days for the return. The SEP system weighs 618 kg, or about 39% of the Earth departure vehicle. Again a propellant pad is included for an extended launch window. The auxiliary chemical propulsion system is sized for only 150 m/sec and weighs 80 kg.

In conclusion, this study has shown that solar electric propulsion can be used quite effectively to accomplish asteroid sample-return missions, at least to the class of bodies represented by Flora and Eros. There is considerable flexibility in designing the appropriate SEP system, selecting the sample return size, and even in launch vehicle selection. The baseline missions described are only to be taken as reasonable examples. The proposed Shuttle orbiter systems may be considered in place of Titan class vehicles for mission opportunities in the 1980's. Finally, based on previous analyses of ballistic flight mode requirements, it may be concluded that SEP offers significant advantages over chemical propulsion systems for this type of mission. Such advantages acrue generally, not in large flight time reductions, but rather in smaller launch vehicles and perhaps greater flexibility in launch opportunities.

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF FIGURES (continued)

LIST OF TABLES

ASTEROID (FLORA AND EROS) SAMPLE-RETURN MISSIONS VIA SOLAR ELECTRIC PROPULSION

1. INTRODUCTION

1.1 Study Background

Acquisition and laboratory analyses of surface samples from various bodies in the solar system are of great scientific value. The results obtained will contribute not only to a better understanding of the individual bodies but also to verifying or formulating theories of solar system origin and evolution. The recent success of the Soviet Luna 16 mission has demonstrated that automated (unmanned) sample return is a technically feasible concept. A comprehensive analysis of a future automated lunar program has been made (Blahnik (1971). Mars sample-return is a logical follow-up to the Viking project, and numerous studies of this mission concept have been performed (Odom 1970, Spadoni and Friedlander 1971).

Asteroids and comets may represent a more suitable class of target than Mars for sample-return. These small bodies are likely to have recorded and preserved more information of early solar system history than the planets. Alfven and Arrhenius (1971), and others, have advanced strong arguments for supporting asteroid exploration, including sample return. The small gravitational field of most asteroids aids in reducing propulsive requirements for landing and separation. However, their heliocentric orbits are larger and more inclined than Mars so that we may expect a significantly larger propulsive requirement to achieve rendezvous and return to Earth.

The asteroid Eros is one of our closest neighbors approaching to within 0.2 AU at periodic intervals. It has been well observed in the past and its orbit, shape and rotation rate are fairly well known (Gehrels 1970). Although Eros may not be representative of some of the larger Main Belt asteroids, it was natural to have received initial attention as a target for sample-return. The ballistic flight mode employing chemical propulsion was studied by Mascy and Niehoff (1971) for a number of launch opportunities between 1975 and 1984, and by Adams, et al (1971) for the 1977 launch period. A major conclusion is that the round-trip mission takes about 3 years and requires the Titan IIID(7)/Centaur launch vehicle with a high-energy upper stage for post-injection maneuvers.

A comparative analysis of solar electric propulsion (SEP) capability is also described by Mascy and Niehoff, and by the CARD study for Marshall Space Flight Center (Northrop 1971). The result as expected is that the greater propulsive efficiency of SEP yields significant advantages such as a smaller launch vehicle requirement and a larger sample size.

1.2 Study Objectives and Approach

The objective of the present study is to determine the performance capability and characteristics of solar electric propulsion for accomplishing asteroid sample-return missions to two diverse targets, Flora and Eros. The intent is to extend the data base and understanding of such missions beyond the earlier studies. Towards this end the major new results are for the Flora mission. Flora was selected by JPL for the following reasons: (1) it is somewhat representative of Main Belt asteroids of moderate size, (2) good astrometric and photometric observations are available, (3) it is the main body of a related group of about 156 asteroids known as the Flora family; these bodies have similar orbits but are distributed in longitude relative to Flora, and (4) no previous trajectory data exists even for rendezvous missions. New results for the

Eros mission are included in the report with attention focused on the 1984 launch opportunity. It is noted that only post-1980 mission opportunities are considered. This seems reasonable in light of current NASA program planning and SEP spacecraft development.

The scope of this study emphasizes trajectory and payload analysis. Science objectives, instrumentation, operations, and spacecraft design are treated only in a limited descriptive sense as they relate to trajectory requirements and vehicle mass fractions. The reader is referred to the literature for a more comprehensive discussion of these topics. Realistic subsystem weights and scaling relationships have, in fact, been taken from such previous studies.

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. Another study guideline is the use of launch vehicles in the Titan family, e.g., the Titan IIID/Centaur. Returned sample sizes in the range 10-50 kg are of interest, with 25 kg being considered a nominal value. The possibility of back-contamination of Earth by sample micro-organisms seems remote. Yet the question remains open to controversy. Two options for sample recovery are therefore considered in the analysis. The first option returns the sample to Earth orbit to be retrieved later by a manned vehicle, and the second is direct atmospheric re-entry with an air or water recovery.

The report is organized as follows: Section 2 discusses the analysis structure and presents numerical definitions of spacecraft subsystems used for the trajectory/payload calculations.

Section 3 describes the trajectory characteristics of the Flora and Eros missions, and presents graphical data of sample size as a function of initial vehicle mass, launch velocity and SEP power rating. Performance sensitivity to specific impulse, propulsion time, and other mission design parameters is discussed. Section 4 gives weight summaries and profile data for several baseline mission examples.

2. ANALYSIS MODELS AND DEFINITION

2.1 Mission-Phases

The sample-return mission may be separated into the following distinct phases: (1) Earth launch and departure, (2) Earth-Asteroid transfer, (3) Asteroid rendezvous, station keeping/circum-navigation, site selection and docking, (4) Surface operations including sample acquisition, (5) Asteroid separation, (6) Asteroid-Earth transfer, and (7) Earth recovery. These mission phases are illustrated schematically in Figure 2-1.

The SEP interplanetary spacecraft may be considered as an upper stage of a high-thrust chemical launch vehicle such as the Titan IIID/Centaur. It must be boosted, at least, to Earth orbital energy before thruster startup. The standard launch mode is to inject the spacecraft to some hyperbolic escape velocity (V_{HL}) via an intermediate Earth parking orbit of about 100 n. mile altitude. Solar electric propulsion is used primarily for the heliocentric transfer between the orbits of Earth and the target asteroid, and for the corresponding return transfer. Because of the relatively small mass of most asteroids, their gravitational field offers little assistance in the capture maneuver. Hence, rendezvous at nearly zero relative velocity is the required terminal condition.

A number of circum-navigation and station-keeping maneuvers are required during the interval between rendezvous and docking. These maneuvers would allow investigation of the asteroid and its environment by remote sensing measurements, and most importantly would achieve the necessary reconnaissance for landing site selection. This phase of the mission may also include a gravitational orbit about the asteroid at an altitude of several radii. Once a preliminary landing site or region is

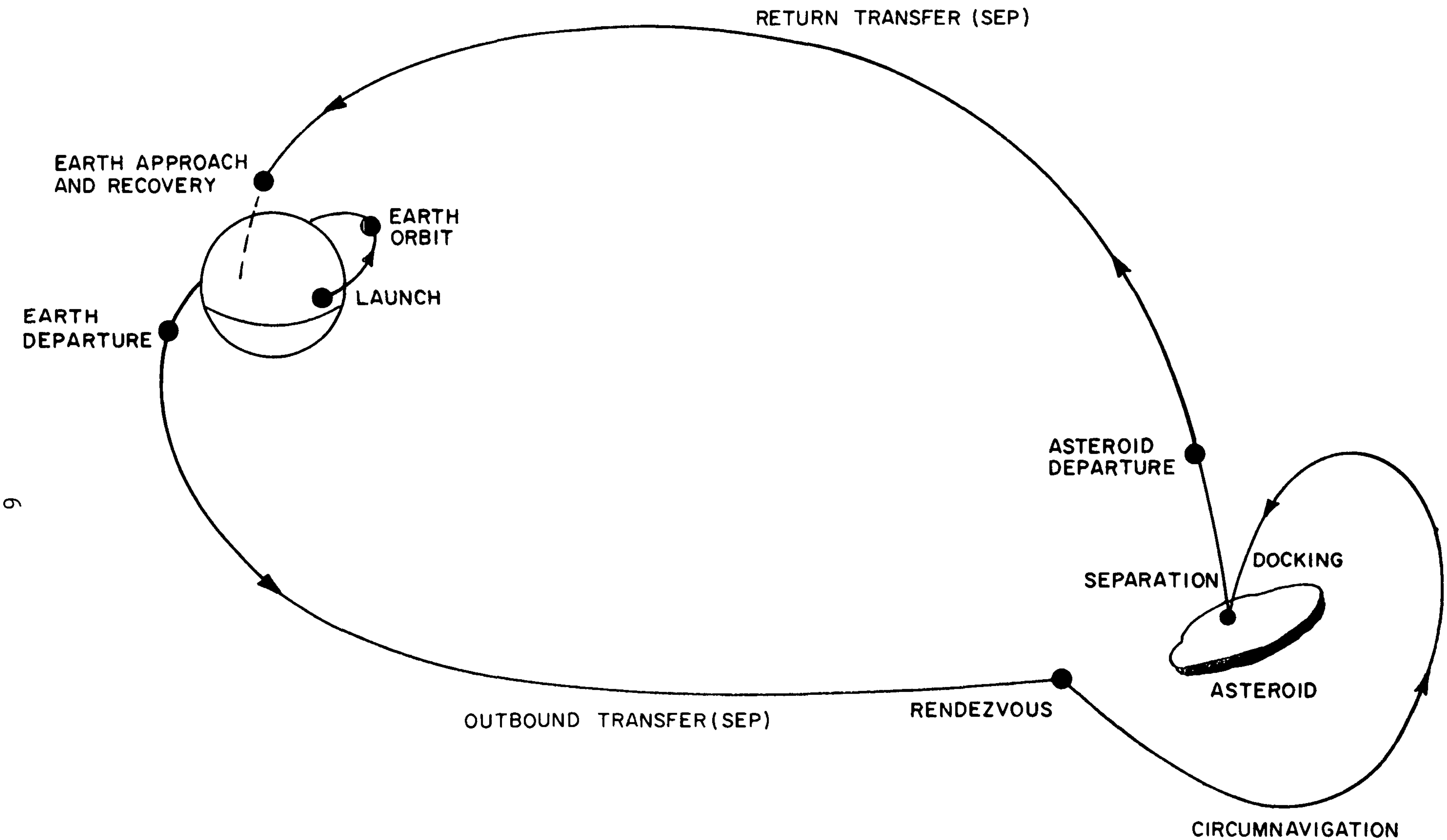


FIGURE 2-1 SCHEMATIC OF SAMPLE-RETURN MISSION PHASES

selected (Earth-based decision), the vehicle is commanded to perform a terminal descent maneuver and achieve a docking/attachment on the surface. Prior to actual docking, a hover mode may be included (at several kilometers altitude) to allow time for an Earth tele-operator link to select the final touch-down location, or to abort the landing if necessary. For purposes of this analysis, it is assumed that all post-rendezvous maneuvers including asteroid separation are performed by an auxiliary chemical propulsion system having multiple restart capability. It is also assumed that the entire spacecraft lands on the asteroid, but that certain subsystems not needed for the return phase are staged off at separation.

The return transfer to Earth begins with asteroid rendezvous conditions and terminates at some hyperbolic excess velocity at Earth approach. The question of whether or not the asteroid sample should be quarantined has not yet been answered with certainty. Therefore, two options for sample recovery will be considered. The first, direct re-entry, assumes that the sample container enters the Earth's atmosphere directly from the hyperbolic approach trajectory, and is retrieved either in the air or on the surface. The second option, orbit capture, assumes that the container and necessary recovery aids are placed by a chemical (solid) retro stage into an elliptical orbit (555 x 9000, km altitude), and later retrieved by a manned vehicle and placed into quarantine.

2.2 Orbital and Physical Characteristics (Flora and Eros)

Trajectory energy requirements and launch opportunities for sample-return missions depend upon the orbit characteristics of the target asteroid and its relative phasing with the motion of Earth. Table 2-1 lists the basic orbital elements and other pertinent parameters of Flora and Eros. Situated in the Main Asteroid Belt, Flora has a mean orbital distance of 2.2 AU

TABLE 2-1

ORBITAL ELEMENTS OF FLORA AND EROS

ELEMENT	FLORA	EROS
SEMI-MAJOR AXIS, a	2.2016 a.u.	1.4581 a.u.
ECCENTRICITY, e	0.1569	0.223
INCLINATION, i	5°.894	10°.829
LONGITUDE OF NODE, Ω	110°.85	304°.004
LONGITUDE OF PERIHELION, $\bar{\omega}$	35°.15	122°.069
LONGITUDE OF EPOCH, L	7°.74	112°.954
EPOCH JULIAN DATE, T	2434000.5	2430200.5
PROJECTED PERIHELION, T_p	8 MAY 1978	1 AUG. 1978
ORBITAL PERIOD, P	3.28 YRS.	1.76 YRS.
EARTH SYNODIC PERIOD	1.44 YRS.	2.32 YRS.

and an orbital period of 3.28 years (1193 days). The perihelion and aphelion distances are, respectively, 1.86 AU and 2.55 AU. The orbit is inclined about 6° with respect to the ecliptic plane. Optimum launch opportunities would recur at intervals approximately equal to the Earth synodic period of 1.44 years (525 days).

Eros is a Mars-crossing asteroid having a mean distance of about 1.5 AU and an orbital period of 1.76 years (643 days). Aside from the Moon, it is one of Earth's closest neighbors approaching to within 0.2 AU at periodic intervals. The perihelion and aphelion distances are, respectively, 1.13 AU and 1.78 AU. The orbit has a moderate inclination of about 11°. Launch opportunities recur at nearly two year intervals.

Figure 2-2 compares the ecliptic projections of the orbits of Flora and Eros. Note that the apsidal (major axis) line and the nodal line of Eros are nearly coincident. In such cases the optimum Earth launch position usually occurs near the asteroid's perihelion longitude. This will not generally be true for missions to Flora since the apsidal and nodal lines are displaced by 76°. A gross comparison of the trajectory energy requirements for Flora and Eros rendezvous can be made by considering a ballistic Hohmann transfer between Earth and the asteroid aphelion distance. The Hohmann transfer time to Flora is 430 days, and the sum of the transfer and rendezvous impulses is 9.05 km/sec. For Eros, the Hohmann time is 300 days, and the total impulse is 7.13 km/sec. Relatively speaking then, the Flora mission is about 27 percent more difficult. It must be realized, of course, that these numbers do not translate directly for SEP trajectories. In particular, the outbound transfer times for the round trip missions will be shown to be significantly longer than the Hohmann times.

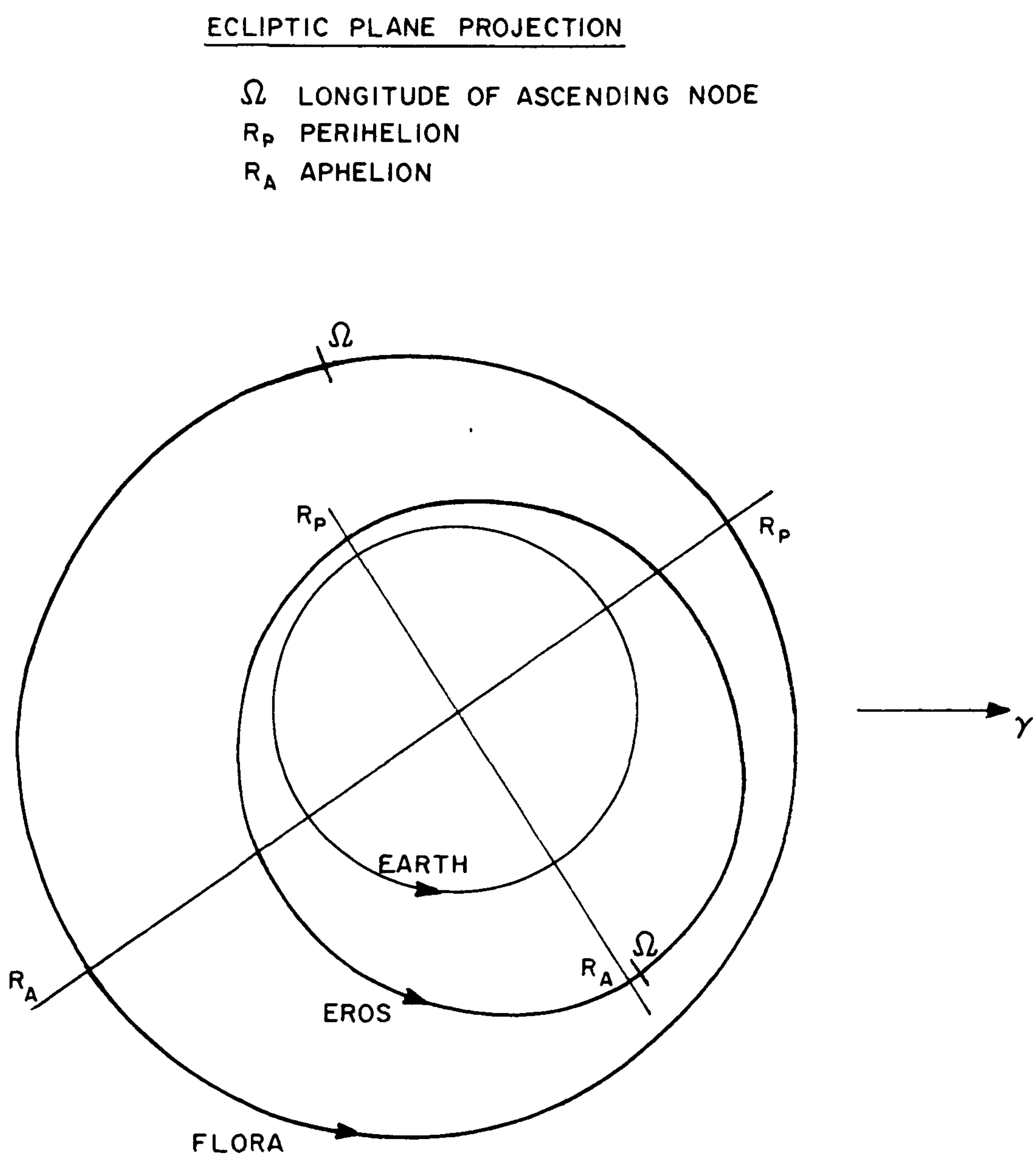


FIGURE 2-2. ORBITS OF THE ASTEROIDS FLORA AND EROS.

Those physical parameters which are pertinent to the post-rendezvous and docking maneuvers are listed in Table 2-2. Asteroid size is the most basic of these parameters. While direct size measurements can be made for larger asteroids such as Ceres, the size of the smaller bodies are usually inferred from their measured brightness characteristics and certain assumptions regarding albedo (Dolfus 1970). By assuming Flora to have the same albedo as Ceres (p = 0.11), the radius of Flora is estimated to be about 82 km. Since only small variations in Flora's light curve have been observed, a spherical shape is assumed. In contrast, the brightness variation of Eros implies an elongated or "cigar-shaped" body having semi-axes measurements of 17.5 x 8 x 3.5 km (Gehrels 1970). Fairly reliable estimates of the rotation periods of Flora and Eros are 13.6 hours and 5.3 hours, respectively. The probable composition of the asteroids gives a density estimate of 3-4 gm/cc; a value of 3.5 gm/cc is assumed in this study.

The remaining parameters listed in Table 2-2 are derived from the size and density estimates. Thus the mass and gravitational constant of Flora are about three orders of magnitude higher than those of Eros. The surface gravity of Flora is about 1/100th of Earth's gravity, and the gravity of Eros is smaller yet by a factor of 2 to 50 depending upon the surface location. Velocity requirements for landing and separation are equivalent to the surface escape velocity which is 115 m/sec for Flora and 16.5 m/sec (maximum) for Eros. The gravitational sphere-of-influence, relative to the Sun, is about 10,000 km for Flora but only several hundred kilometers for Eros. Note that all post-rendezvous maneuvers at Flora will take place within its sphere-of-influence.

TABLE 2-2

ESTIMATED PHYSICAL CHARACTERISTICS OF FLORA AND EROS

CHARACTERISTIC	FLORA	EROS	
SIZE	82 KM RADIUS ASSUMED SPHERICAL	17.5 x 8 x 3.5 KM ELLIPSOID SEMI-AXES	
ROTATIONAL PERIOD	13.6 HRS.	5.3 HRS.	
DENSITY	3.5 GM/CC	3.5 GM/CC	
MASS	8.1×10^{21} GM	7.2×10^{18} GM	
GRAVITATIONAL CONSTANT (μ)	0.54 KM^3/SEC^2	4.8×10^{-4} KM^3/SEC^2	
SURFACE GRAVITY	0.08 M/SEC^2	0.039 M/SEC^2	MAXIMUM AT 3.5 KM
ESCAPE VELOCITY	115 M/SEC	16.5 M/SEC	MAXIMUM AT 3.5 KM
SPHERE-OF-INFLUENCE (AT MEAN DISTANCE)	9150 KM	365 KM	

2.3 Post-Rendezvous/Docking Maneuvers

A representative circum-navigation/station-keeping profile for the Eros mission is illustrated in Figure 2-3 (Northrop 1971). Starting at point 1, 1000 km from the asteroid in the solar direction, the sequence of translation maneuvers places the spacecraft at various points on a sphere of 200 km about the asteroid. The entire sequence would take 30 days and require a ΔV expenditure of about 7 m/sec. Viewed from various aspect angles, a large amount of information would be obtained as to surface features, rotation axis and rate, and the possibility of a dust cloud or atmospheric halo around Eros. A similar maneuver sequence could be defined for the Flora mission; the ΔV requirement would be somewhat higher due to gravity losses within the sphere-of-influence.

It is reasonable also to consider establishing a gravitational orbit about the asteroid, particularly in the case of Flora. Table 2-3 lists the circular velocity, insertion ΔV and period for orbits between 1 and 5 asteroid radii distance. The ΔV calculation assumes a parabolic free-fall to the orbit distance. For Flora, 19.4 m/sec is required to establish a 3 radii orbit (164 km altitude) having a period of about 9 hours. For Eros, the ΔV is only 1.66 m/sec for a 3 radii orbit ($\sim$ 20 km altitude) having a period of 13 hours.

Additional maneuvers to be considered are descent from orbit, hover, terminal landing, and separation. Hovering at low altitude (5 km) can be rather expensive in terms of ΔV when the two-way communication time to Earth is taken into account. Earth-Flora distance is typically 1 AU at encounter, giving a two-way time delay of about 17 minutes. Assuming a 30-minute hover requirement at $g = 0.08$ m/sec^2, the ΔV expenditure is 144 m/sec. For Eros missions the encounter occurs near conjunction with a communications distance and two-way time

REF SOURCE (NORTHROP 1971)

STATION POINT	TRANSFER TIME (DAYS)	STAY TIME (DAYS)	TRANSFER ΔV (M/SEC)	STATION-KEEPING ΔV (M/SEC)
1	—	1	—	0.18
2	10	1	0.96	0.91
3	5	1	0.66	0.65
4	5	1	0.65	0.66
5	5	1	0.93	0.97

TOTAL ΔV = 7 M/SEC

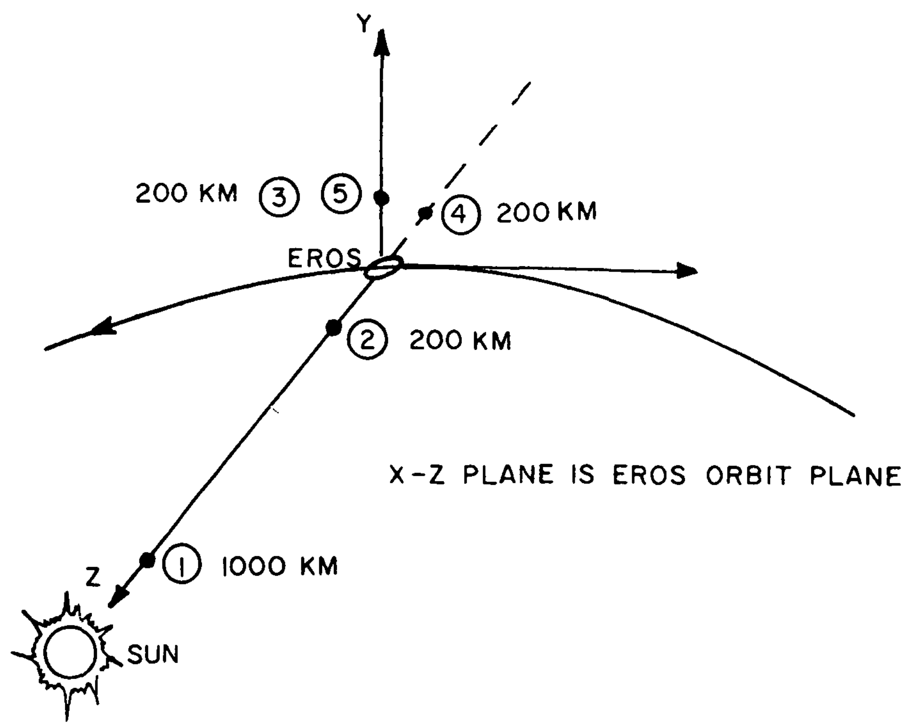


FIGURE 2-3. EROS CIRCUMNAVIGATION/STATIONKEEPING PROFILE

TABLE 2-3

ASTEROID ORBIT REQUIREMENTS

ORBIT RADIUS*	FLORA			EROS		
	CIRCULAR VELOCITY (M/SEC)	INSERTION ΔV (M/SEC)	ORBIT PERIOD (HRS.)	CIRCULAR VELOCITY (M/SEC)	INSERTION ΔV (M/SEC)	ORBIT PERIOD (HRS.)
R_o	81.5	33.6	1.76	6.92	2.86	2.52
$2R_o$	57.4	23.7	5.00	4.89	2.02	7.14
$3R_o$	46.9	19.4	9.16	4.00	1.66	13.2
$4R_o$	40.7	16.8	14.1	3.46	1.43	20.2
$5R_o$	36.3	15.0	19.7	3.09	1.28	28.2

* FLORA R_o = 82 KM

EROS R_o = 10 KM (ASSUMED AVERAGE VALUE) delay of 2.3 AU and 38 minutes, respectively. Assuming a 60-minute hover at $g = 0.0066\ m/sec^2$, the ΔV expenditure is 24 m/sec.

Table 2-4 summarizes the ΔV budget for all post-rendezvous maneuvers including asteroid separation. Also included is an allowance of 70 m/sec for guidance maneuvers on the outbound and return transfers. The total ΔV's for Flora and Eros missions are 500 m/sec and 150 m/sec, respectively. It is assumed that the velocity requirement is imparted by an auxiliary chemical propulsion system having a 310 second specific impulse and a 20 percent inert fraction. For purposes of calculating the system propellant and inert weight, the total ΔV is split into two components as shown, i.e., a pre-docking and post-docking ΔV.

2.4 Science Payload

The overall scientific goal of missions to the asteroids is to resolve the question of their formation and how this contributes to solar system origin and evolution. One hypothesis suggests that the asteroid belt represents an intermediate stage in the formation of planets (Alfven 1970). Opposing this thesis of accretion is the possibility that asteroids resulted from the fragmentation of an old planet. Yet another suggestion is that some asteroids are the "burnt out" nuclei of comets.

The major scientific goal of asteroid investigation requires that a number of different types of bodies be visited. However, much can be learned from a comprehensive study of even one asteroid, particularly if a surface sample can be analyzed. Among the science objectives for a given mission are to determine the properties of the asteroid body, to measure its ambient environment, and to determine its interaction with the solar wind. Asteroid properties of interest are size, shape, mass,

TABLE 2-4

VELOCITY REQUIREMENTS FOR AUXILIARY CHEMICAL PROPULSION

MANUEVER	FLORA MISSION		EROS MISSION	
EARTH-ASTEROID TRANSFER	35 M/SEC		35 M/SEC	
CIRCUMNAVIGATION/STATIONKEEPING	9		8	
ORBIT CAPTURE	20		2	
DESCENT FROM ORBIT	109		11	
HOVER	144		24	
TERMINAL LANDING	28	(345)	15	(95)
ASTEROID SEPARATION	120		20	
ASTEROID-EARTH TRANSFER	35	(155)	35	(55)
TOTALS		500 M/SEC		150 M/SEC

rotation, surface and interior structure and composition, and age. Environmental characteristics include gravitational field, particles, and remnant atmosphere or dust clouds.

Candidate instruments are listed in Table 2-5 along with related science measurables, the applicable mission phases, and estimates of weight and power. It should be noted that many of these instrument selections are based on current program technology such as Mariner, Pioneer and Viking. The instruments are divided into two categories, (1) those remote sensing and in-situ instruments associated with the usual type of science payload, and (2) those devices associated with sample acquisition and handling. The weights in each category are 55 kg and 30 kg, respectively, for a total experimental payload of 85 kg.

2.5 Spacecraft System Mass Allocation

The purpose of this section is to summarize the various system mass components and scaling assumptions employed in the subsequent calculation of SEP sample-return capability. These consist of: (1) launch vehicle performance, (2) SEP system, (3) spacecraft bus and structure, (4) auxiliary chemical propulsion, (5) rendezvous, docking, science and sampling module (RDSS), and (6) sample return module (direct re-entry or orbit capture).

Figure 2-4 presents curves of maximum injected mass as a function of hyperbolic launch velocity for three launch vehicles considered in the study; Titan IIID, Titan IIID/Burner II, and Titan IIID/Centaur. Data for the Titan III class vehicles was provided by the Jet Propulsion Laboratory. It is understood that a particular mission design may not necessarily utilize the maximum launch vehicle performance at a given value of V_{HL}. This can occur when the SEP power rating is significantly less than optimum (maximum net mass). If full launch vehicle

TABLE 2-5

CANDIDATE EXPERIMENTS FOR ASTEROID SAMPLE-RETURN MISSIONS

GENERIC INSTRUMENT	SCIENCE MEASURABLE	MISSION PHASES						WEIGHT (KG)	POWER (W)
		OUTBOUND TRANSFER	RENDEZ-VOUS	CIRCUM-NAVIGATION	LANDING	SURFACE ACTIVITY	RETURN TRANSFER		
SCIENCE								55	72
TELEVISION	SHAPE, FEATURES			X	X	X		18	33
FACSIMILE CAMERA	SAMPLE OBSERVATION					X		3	4
IR RADIOMETER	SURFACE TEMPERATURE			X				2	5
METEORITE DETECTOR	PARTICLE VELOCITY/MOM.	X	X	X			X	5	2
MAGNETOMETER	MAGNETIC FIELD	X	X	X		X	X	2	3
PLASMA GAUGE	SOLAR WIND	X	X	X			X	5	4
GRAVITY GRADIOMETER	GRAVITY, MASS			X	X	X		4	5
MASS SPECTROMETER	OUTGASSING			X		X		4	5
X-RAY SPECTROMETER	SURFACE COMPOSITION					X		3	6
PENETROMETER	STRUCTURAL PROPERTIES					X		1	2
TEMPERATURE SENSOR	HEAT FLOW					X		1	1
ACTIVE SEISMOMETER	INTERNAL STRUCTURE					X		7	2
SAMPLE ACQUISITION								30	
SCOOP, TONGS, ETC	SURFACE SAMPLE					X		5	
ROTARY CORE DRILL	SUBSURFACE SAMPLE					X		15	BAT.
SAMPLE CANNER	STORAGE, LABELING					X		10	
TOTAL								85 KG	

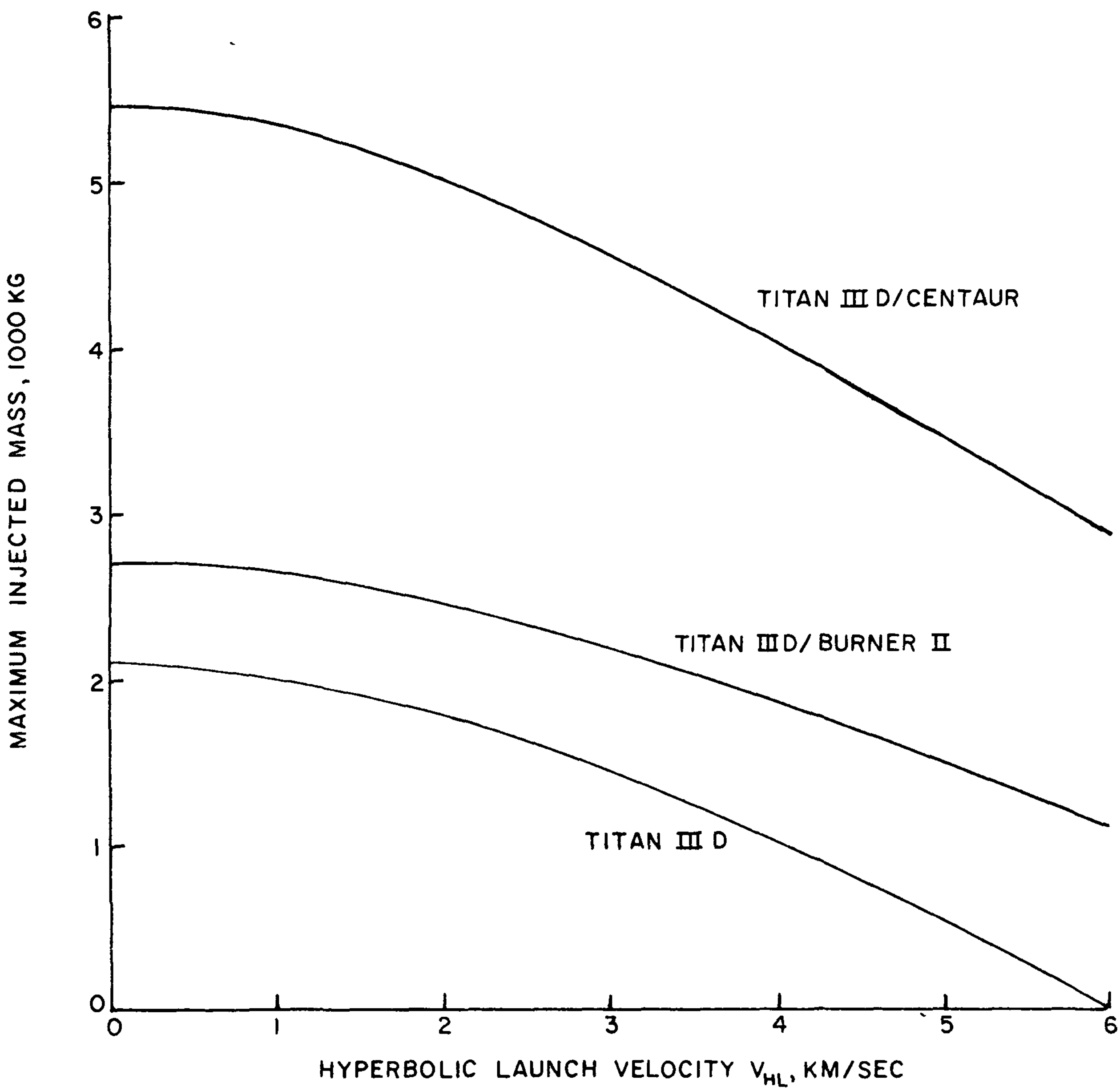


FIGURE 2-4 LAUNCH VEHICLE PERFORMANCE CURVES

capability is not needed for payload delivery then it may be used to obtain other engineering advantages such as a reduction in propulsion on-time, if desired.

Characteristics of the solar electric propulsion system are listed below:

Specific mass, α_{ps}	30 kg/kw
Tankage factor, k_p	3% of propellant loading
Specific impulse, I_{sp}	3500 sec
Overall efficiency, η	66%

The SEP mass exclusive of propellant and tankage is α_{ps} P_o where P_o is the input power to the thrust subsystem at 1 AU. The specific mass includes the solar array, power conditioners, thrusters, thrust vector control actuators, and a contingency to account for solar array power losses due to possible radiation damage and housekeeping power during cruise. The 3500 second I_{sp} is a baseline value for the analysis and is representative of current ion thruster design. Performance sensitivity to I_{sp} will be examined; other values and corresponding efficiencies are 2500 sec (57%), 3000 sec (62%), and 4000 sec (68%).

A mass of 450 kg has been assigned to the interplanetary bus which comprises the engineering support subsystems such as communications and data handling, computer and sequencer, attitude control, thermal control, meteoroid shielding, and structure. Table 2-6 shows a typical mass breakdown which is based on the CARD study (Northrop 1971) and recent IITRI studies. Note that the navigation and guidance subsystem is not included here. Rather it is accounted for as part of the RDSS module. The auxiliary chemical propulsion system used for midcourse, approach and post-rendezvous maneuvers is assumed to be of the

TABLE 2-6

SPACECRAFT BUS SUBSYSTEMS

COMMUNICATIONS AND ANTENNA	85 KG
DATA HANDLING AND STORAGE	24
CENTRAL COMPUTER/SEQUENCER	11
ATTITUDE CONTROL	56
ELECTRIC POWER/BATTERY	45
THERMAL CONTROL	34
METEOROID SHIELDING	45
STRUCTURE AND CONTINGENCY	150
	450 KG

Earth-storable propellant type (310 sec I_{sp} and 20% inert fraction). A typical mass range for chemical propulsion is found to be 300-400 kg (Flora mission) and 60-100 kg (Eros mission).

Table 2-7 shows the subsystems comprising the rendezvous docking, science and sampling module (Northrop 1971). The Navigation and Guidance subsystem includes an approach guidance vidicon, a medium range radar for circum-navigation and descent, a multi-beam radar for docking, platform gyros and an horizon scanner. The Docking and Attachment subsystem requirements are to provide a firm connection for a range of possible surface hardnesses and to provide a normal holding force through the landed vehicle. Major functional devices are an explosively driven piton (hard surfaces) and a rocket fired harpoon. Thermal control is a scaled down version of that required for the spacecraft bus and consists of a combination of louvers, insulation and heaters. Also, as in the spacecraft bus, the meteroid shielding consists of a multiple aluminum sheet with low density filler material. The total mass of the RDSS module is 250 kg. This system is jettisoned prior to asteroid escape.

Scaling relationships for the sample-return module are given in Figure 2-5. The basic sample container, including the sample and environment control equipment, is assumed to be a linear function of sample size M_s:

$$M_{sc} = 2.1\ M_s + 5\ \text{(kg)}$$

In the case of direct re-entry (re-entry speed < 40,000 ft/sec), the sample container is placed into an aerobraking system comprised of an aeroshell, parachute and recovery beacon; the total mass including M_{sc} is shown by the broken-line curve in Figure 2-5. The orbit capture mode of recovery assumes an

TABLE 2-7

RDSS MODULE SUBSYSTEMS

NAVIGATION AND GUIDANCE	31 KG
DOCKING AND ATTACHMENT	30
SCIENCE PAYLOAD	55
SAMPLE ACQUISITION AND HANDLING	30
THERMAL CONTROL	7
METEROID SHIELDING	14
STRUCTURE AND CONTINGENCY	83
	250 KG

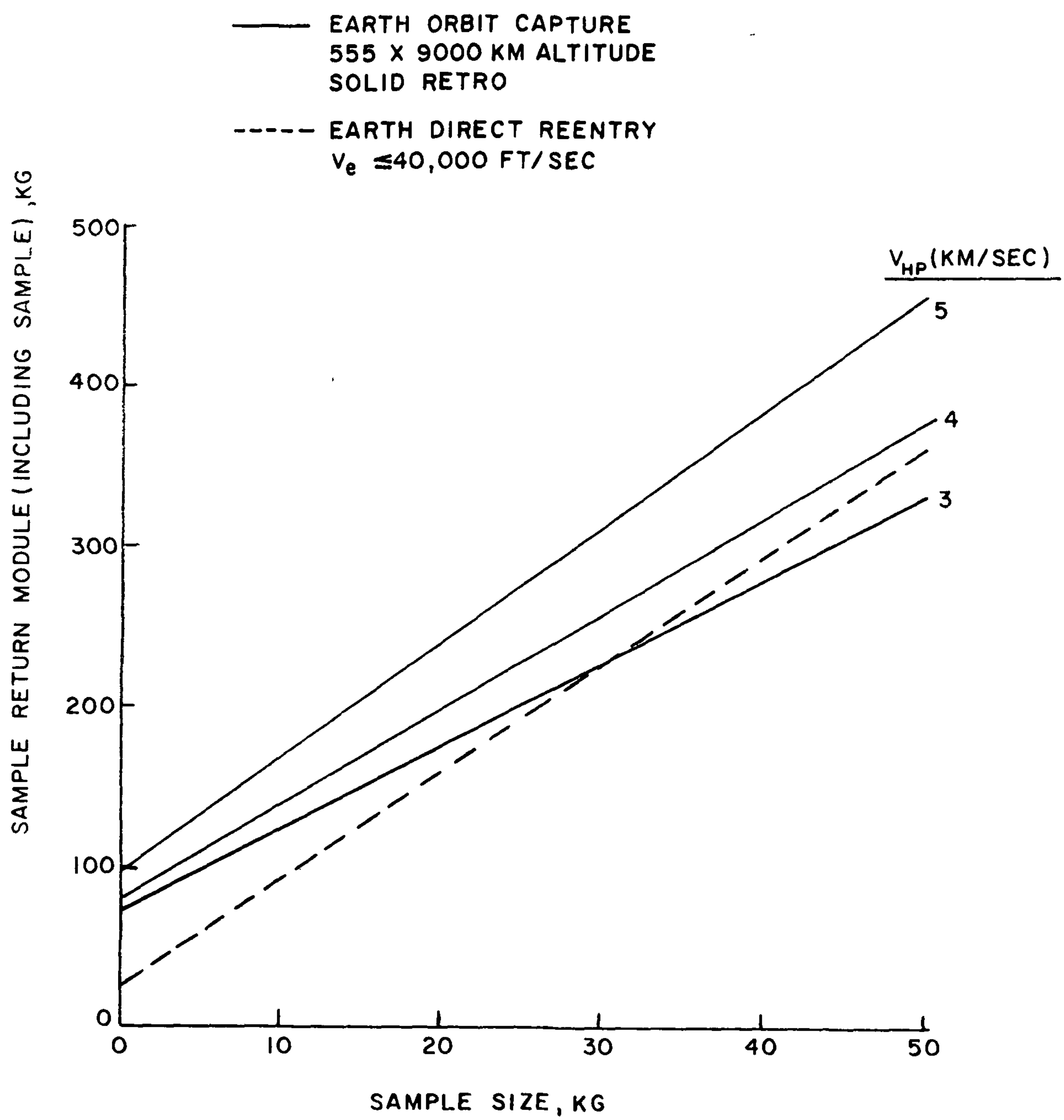


FIGURE 2-5. SAMPLE RETURN MODULE SCALING

elliptical orbit of 555 x 9000 km altitude. This selection is based on the use of an orbit-launched, fully loaded Apollo CSM, but is not necessarily incompatible with a refueled Earth Orbital Shuttle (for further discussion of orbit size selection see Odom 1970). The orbited "payload" or sample capsule consists of the sample container plus a 23 kg allowance for attitude stabilization and recovery aids. This capsule is placed into orbit by a solid propellant retro stage having an I_{sp} of 300 seconds and an inert fraction of 0.11. Clearly the retro mass depends upon the approach hyperbolic velocity V_{HP}. The solid-line curves in Figure 2-5 give the combined mass of the retro stage and sample capsule for typical values of V_{HP}.

To summarize, the interplanetary vehicle mass at Earth departure consists of the following components.

$$M_o = 30\ P_o + 1.03\ M_p + M_{ACP} + (M_{SRM} - M_S)$$

$$+ M_{BUS}\ (450\ \text{kg}) + M_{RDSS}\ (250\ \text{kg})$$

where M_{ACP} is the auxiliary chemical propulsion system and M_{SRM} is the sample return module. The next section of the report describes the characteristics and requirements of the outbound and return interplanetary trajectories, and how these affect the sample return capability.

3. TRAJECTORY AND PAYLOAD ANALYSIS

The analysis of round-trip missions requires a survey of compatible outbound and return trajectories. SEP trajectories were generated using the CHEBYTOP computer program (Hahn, et al 1969). The propellant mass fraction data obtained was combined with the system mass allocation discussed in the previous section in order to determine the sample-return capability. Results presented in this section describe the SEP trajectory/payload characteristics, and show the effect of such mission design parameters as launch opportunity, flight time, launch and return hyperbolic velocities, launch vehicle selection, SEP power rating and propulsion on-time. The mission to the asteroid Flora has not been studied previously. Results for this mission will therefore be presented in a fairly comprehensive manner. Launch opportunities in 1980, 81 and 82 have been investigated. Missions to Eros have received prior attention by others, particularly launch opportunities in the 1970's and 1980's. Our treatment of Eros sample-return will be less detailed and will consider only the 1984 launch opportunity.

3.1 Flora Mission

3.1.1 Launch Opportunities and Energy Maps

A convenient way of presenting the trajectory energy requirements is shown in Figure 3-1 for the 1980 launch opportunity to Flora. The energy measure used is "J" which is given by the time-integral of $a^2/G(R)$, where $a(t)$ is the thrust acceleration magnitude and $G(R)$ is the normalized solar power (relative to $R = 1$ AU) available to the thrust subsystem. The parameter J is related to the propellant expenditure; i.e., the lower the J value the lower the propellant expenditure. It can be shown that the total value of J should not exceed about 6 m^2/sec^3 if viable payloads and practical size powerplants are to be achieved with Titan class vehicles.

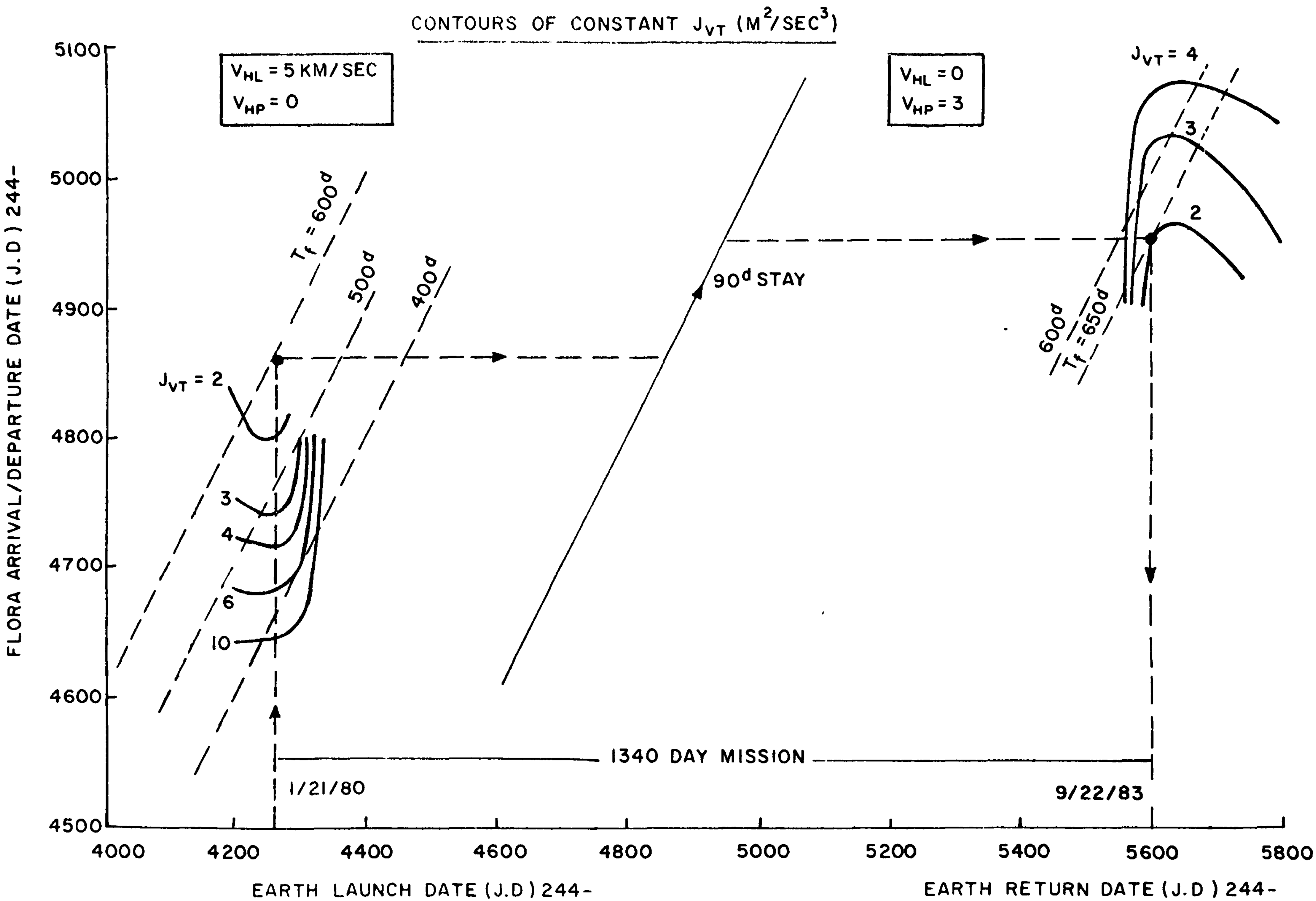
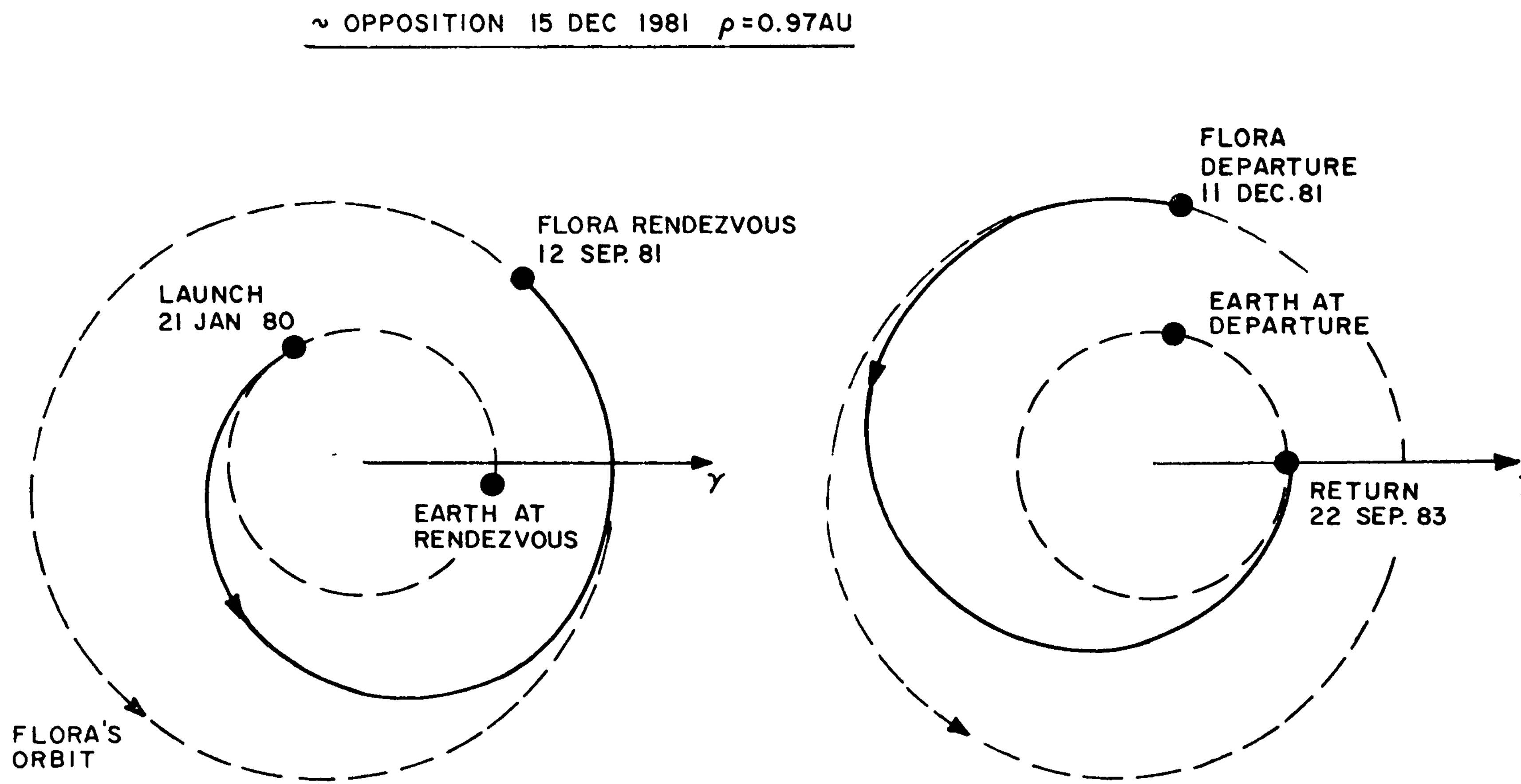


FIGURE 3-1 SOLAR ELECTRIC ENERGY CONTOUR MAP FOR 1980 FLORA SAMPLE RETURN MISSION

The figure shows constant J contours plotted in a grid of Earth launch and return dates (abscissa) and Flora arrival and departure dates (ordinate). Both outbound and return transfers are of the direct type, i.e., the heliocentric travel angle is typically less than 360°. This type of data map is convenient for determining suitable launch and return dates and the effect of varying trip time and stay time at the asteroid. If the mission objective were only a rendezvous with Flora then a one-way flight time of about 400 days would be an acceptable design. However, in the case of round-trip missions, the outbound J must be lowered and this requires a longer transfer time to Flora. Furthermore, the steep-ridge characteristic of both outbound and return transfers indicates a strong limiting effect on the minimum round-trip time. This is simply due to the optimum geometric phasing of Earth and Flora. The limiting characteristic is true, more or less, for round-trip missions to any solar system target. Figure 3-1 illustrates a 1340-day mission example, departing Earth on Julian date 2444260 (1/21/80), arriving Flora 2444860 (9/12/81), staying 90 days, departing Flora 2444950 (12/11/81), and returning to Earth on 2445600 (9/22/83). It will be noted that both the outbound and return legs are near-minimum energy transfers. Any attempt to reduce trip time much below 1340 days will meet with a rapidly increasing energy requirement.

Figure 3-2 shows the transfer profiles of the 1340-day mission projected into the ecliptic plane. Earth-Flora opposition occurs on about Dec. 15, 1981 with a minimum geocentric distance of just under 1 AU. Fortunately, the rendezvous/docking/separation maneuvers also occur near opposition. This results in a very favorable communications profile for encounter operations.

Energy maps and transfer profiles for the 1981 and 1982 launch opportunities are shown in Figures 3-3 to 3-6.



EARTH TO FLORA, 600 DAYS

FLORA TO EARTH, 650 DAYS

STAY TIME = 90 DAYS

FIGURE 3-2. SOLAR ELECTRIC TRANSFER PROFILES FOR A 1340-DAY FLORA SAMPLE RETURN MISSION, 1980 LAUNCH.

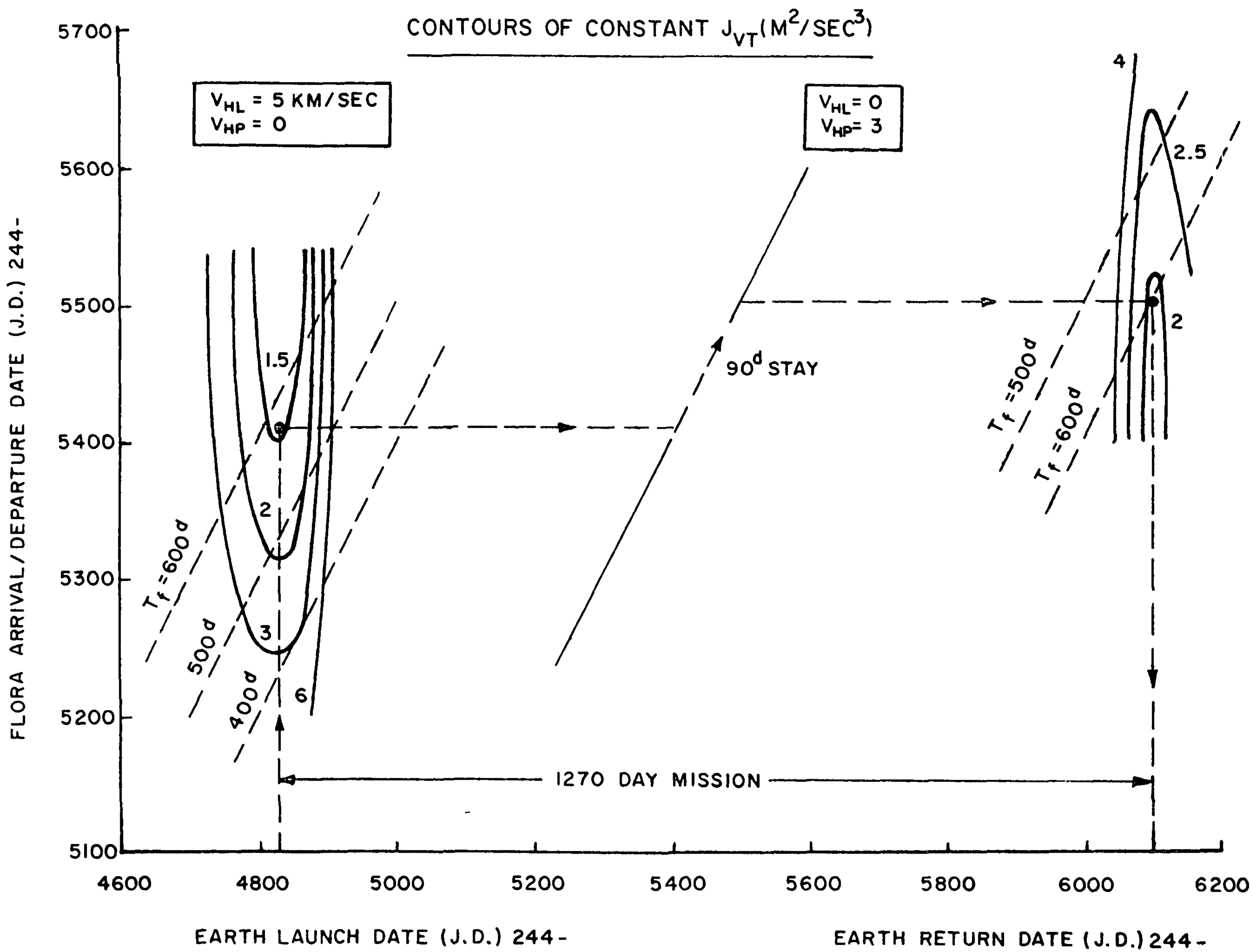


FIGURE 3-3. SOLAR ELECTRIC ENERGY CONTOUR MAP FOR 1981 FLORA SAMPLE RETURN MISSION

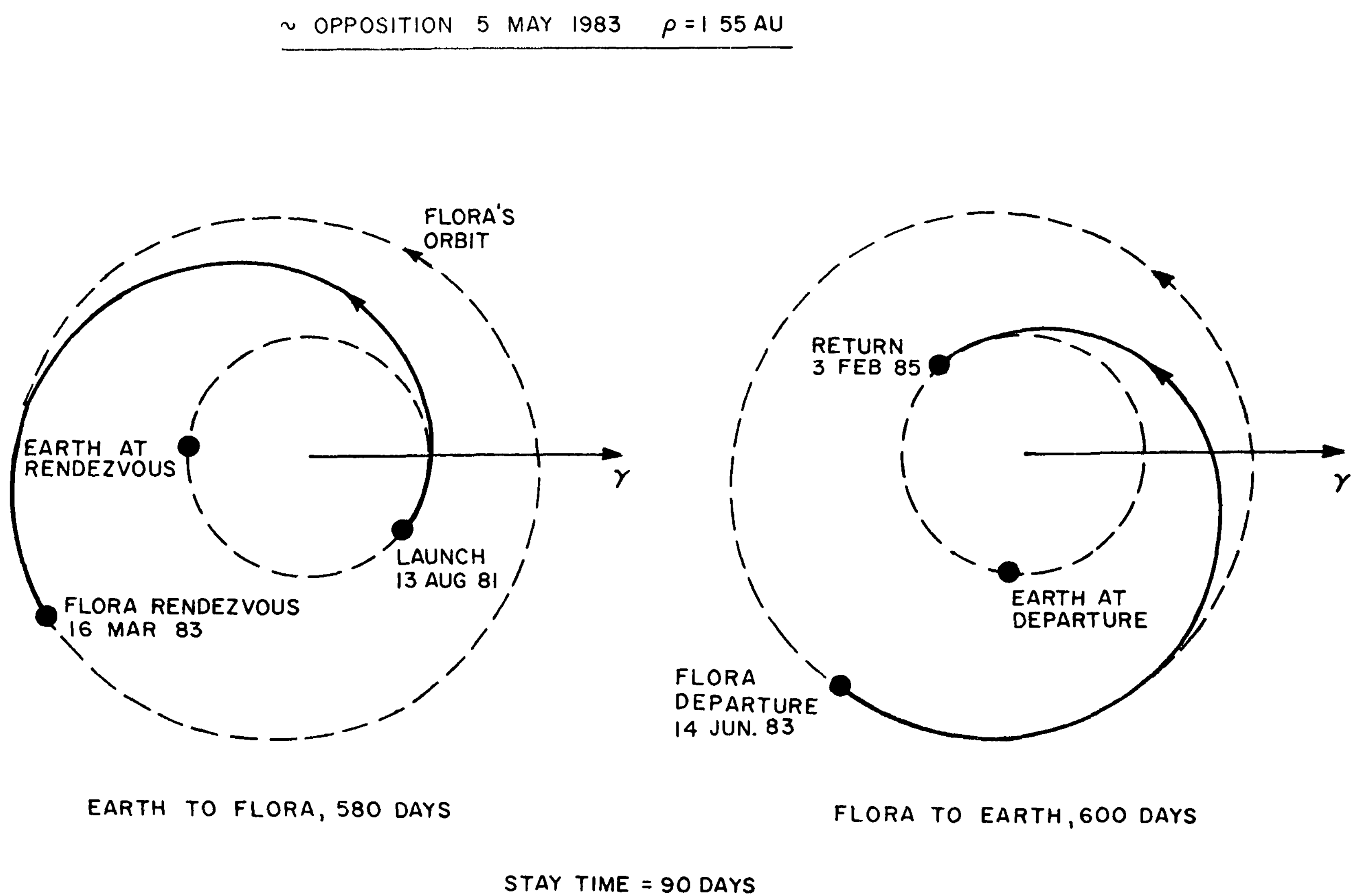


FIGURE 3-4. SOLAR ELECTRIC TRANSFER PROFILES FOR A 1270-DAY FLORA SAMPLE RETURN MISSION, 1981 LAUNCH

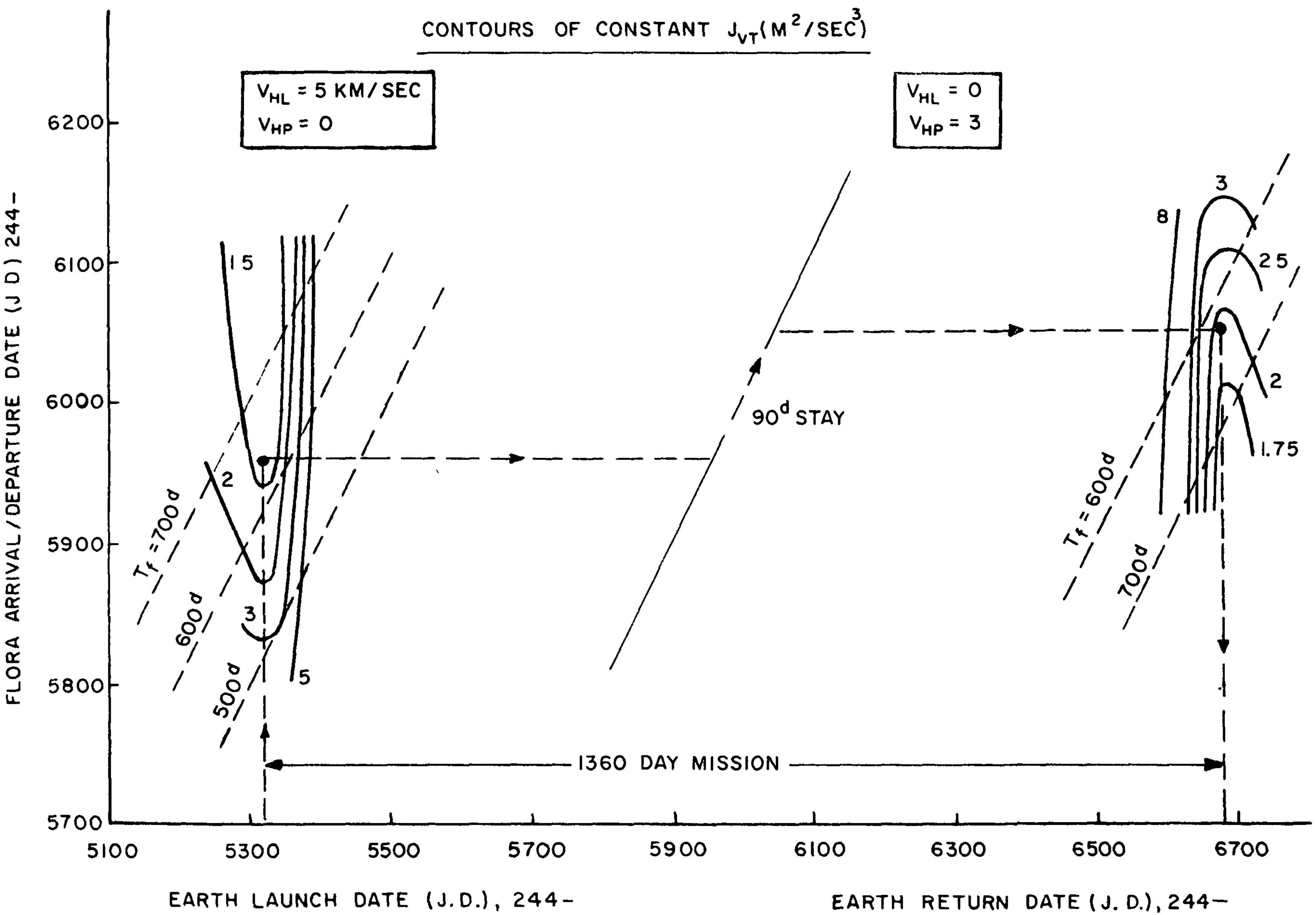


FIGURE 3-5 SOLAR ELECTRIC ENERGY CONTOUR MAP FOR 1982 FLORA SAMPLE RETURN MISSION

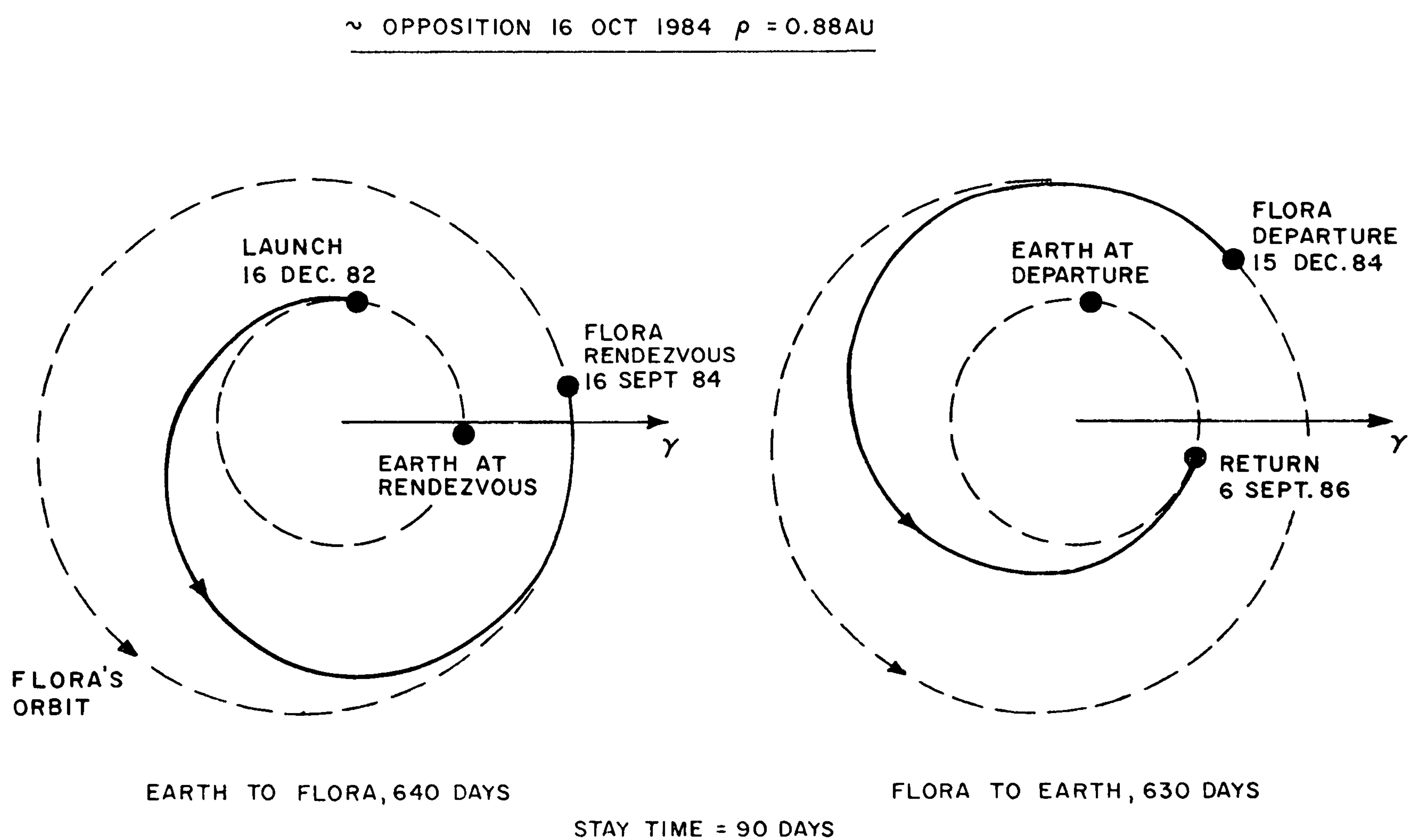


FIGURE 3-6 SOLAR ELECTRIC TRANSFER PROFILES FOR A 1360-DAY FLORA SAMPLE RETURN MISSION, 1982 LAUNCH.

The Earth-Flora synodic period and transfer characteristics are such that encounter generally occurs near opposition, although the opposition distance varies from year to year. A 1270-day trip is illustrated for the 1981 launch period occurring about 470 days after the previous opportunity (synodic period is 525 days). In this case encounter takes place near Flora's aphelion, and the communications distance is about 1.6 AU. The 1982 opportunity closely repeats the 1980 geometry with encounter occurring near Flora's perihelion at minimum distance opposition. Communications distance at encounter for the example 1360-day mission is about 0.9 AU.

3.1.2 Sample-Return Performance

The 1982 launch opportunity has been chosen for the purpose of describing sample-return payload characteristics. Results for the 1980 and 1981 opportunities will be summarized at a later point in Section 3.

Consider first the ability to deliver a given net spacecraft mass to rendezvous conditions. In this context net mass is equal to the initial vehicle mass at Earth departure less the sum of the SEP system mass ($\alpha_{ps}P_o$) and the outbound propellant and tankage; it includes the propellant and tankage required for the return transfer. Figure 3-7 shows the maximum* net mass capability of two Titan class launch vehicles as a function of SEP power for the 640-day outbound transfer. Values of hyperbolic launch velocity are shown parametrically along each curve. The broken line starting at the origin and tangent to each net mass curve represents the off-loaded or scaled launch vehicle performance. At low SEP power this could result in a larger

---

* A general characteristic of "optimum" rendezvous is that the propulsion on-time is equal or nearly equal to the flight time. This yields the maximum value of net mass for a given power rating.

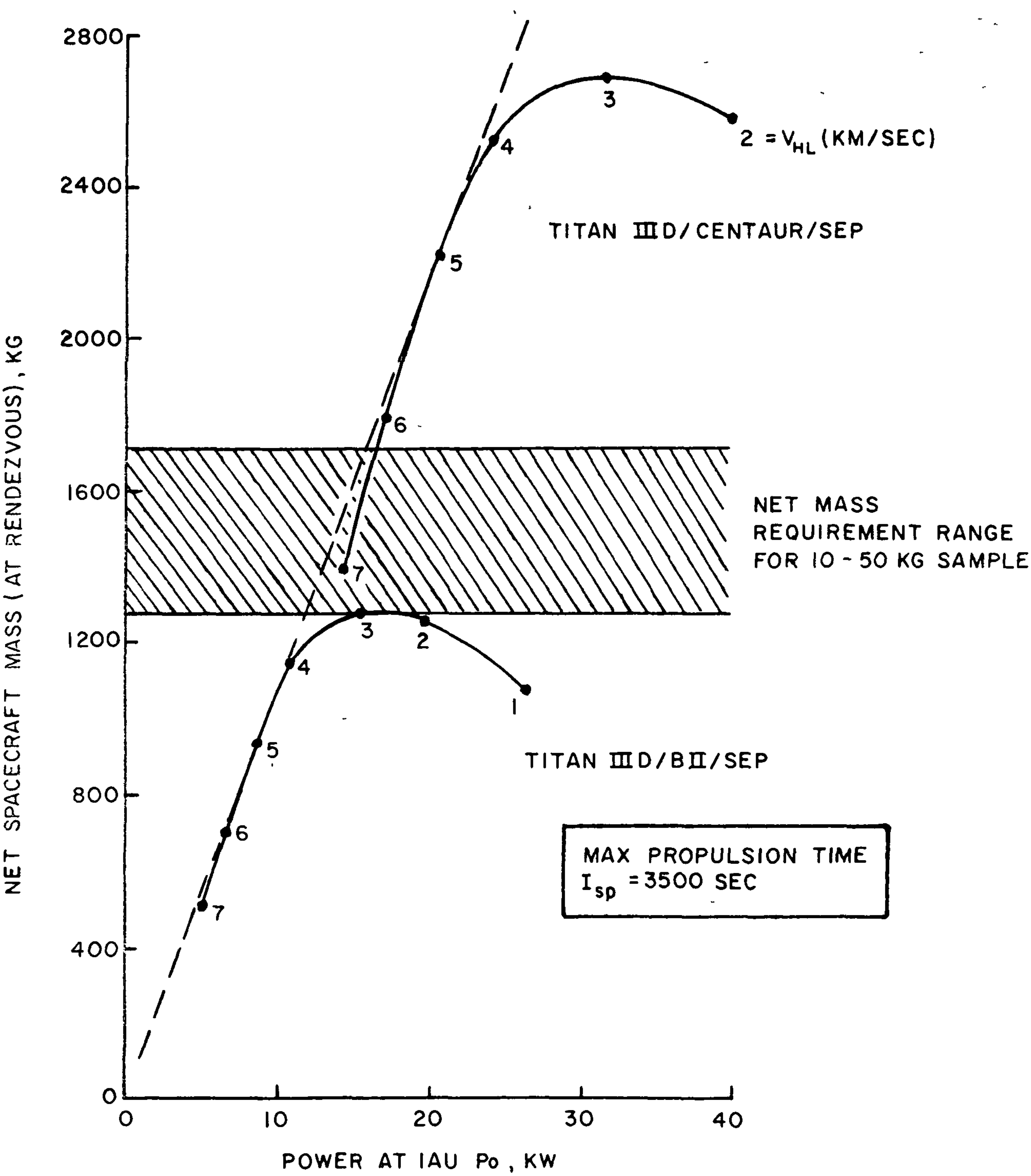


FIGURE 3-7. SOLAR ELECTRIC CAPABILITY FOR FLORA RENDEZVOUS, LAUNCH 12/16/82, FLIGHT TIME 640 DAYS.

net mass capability than the fully loaded (maximum injected mass) launch vehicle. In this case the tangent point occurs at about V_{HL} = 5 km/sec. For the Titan IIID/Centaur, this value of V_{HL} would give the largest net mass for all power design points less than 20 kw. Analysis of the round-trip mission has shown that net mass at rendezvous must lie in the range 1275-1710 kg in order to return a 10-50 kg sample. The Titan IIID/BII has a very marginal capability even at the optimum power design of 15 kw. On the other hand the Titan IIID/Centaur has more than adequate capability at optimum power (30 kw), but is well suited to this mission at power designs in the range 12-16 kw.

Another convenient presentation of performance data is shown in Figure 3-8 for the orbit capture mode of recovery. Earth departure mass is given as a function (essentially linear) of sample size for several values of launch velocity. Note that the lowest curve is for V_{HL} = 5 km/sec; this reflects the result that the net mass tangency point occurs at the same value of V_{HL}. Although the basic data in Figure 3-8 is independent of launch vehicle selection, the limiting performance of a particular launch vehicle may be superimposed on this data (see Fig. 2-4; M_o vs V_{HL}). Thus the Titan IIID/BII is seen to have the marginal capability of a 3 kg sample return to Earth orbit. The upper performance limit of the Titan IIID/Centaur does not even appear on the scale of the figure. However, at the off-optimum power of 15 kw, this launch vehicle provides a 40 kg sample return for a departure mass of 2500 kg.

Figure 3-9 presents equivalent performance data for the direct reentry mode of recovery. The increase in sample size is only several kilograms above the orbit capture mode. A trade off between the two recovery modes on the basis of sample size alone exists if the Titan IIID/BII were the selected launch vehicle. In this case direct reentry allows a maximum sample

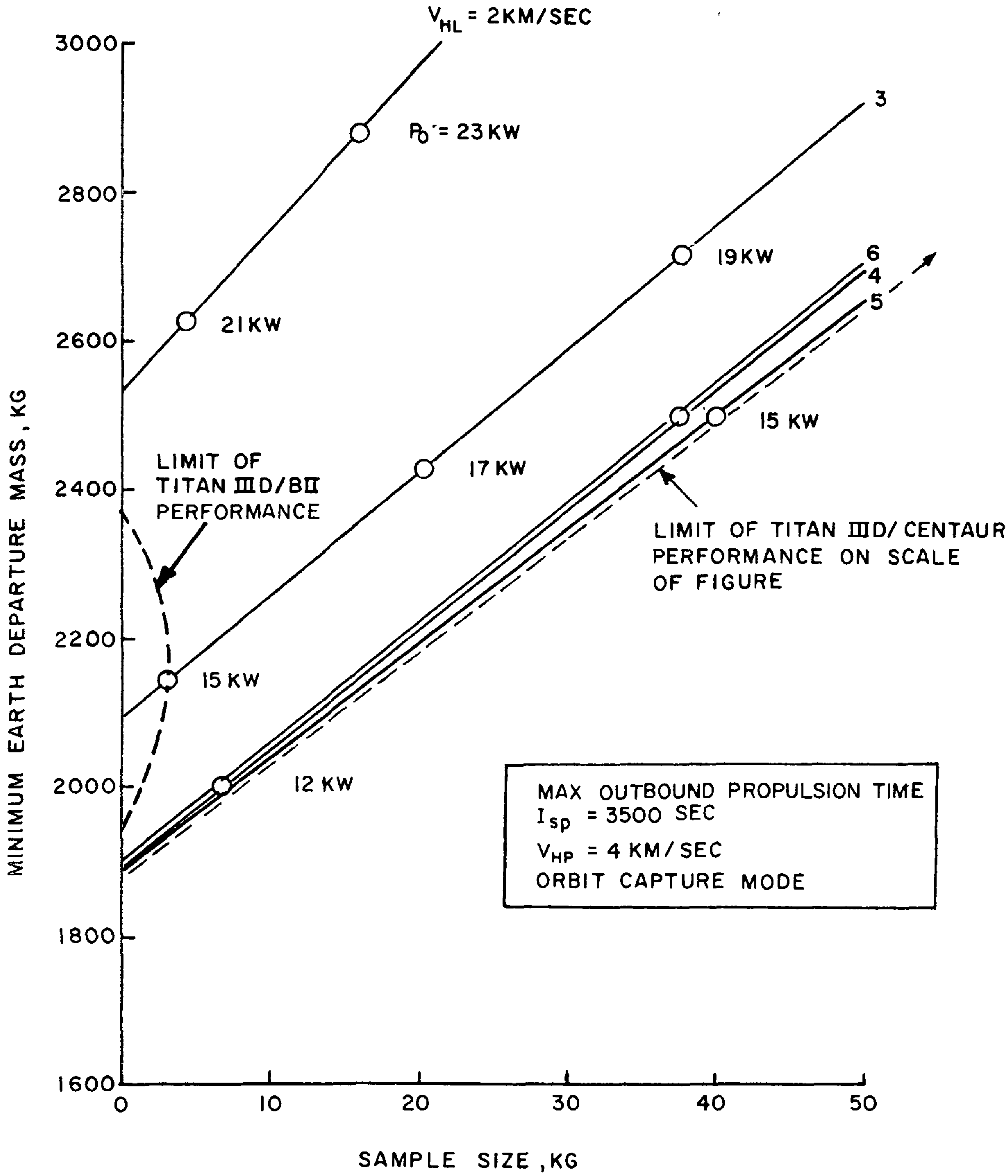


FIGURE 3-8. SOLAR ELECTRIC PERFORMANCE FOR FLORA SAMPLE-RETURN TO EARTH ORBIT, LAUNCH 12/16/82, TRIP TIME 1360 DAYS

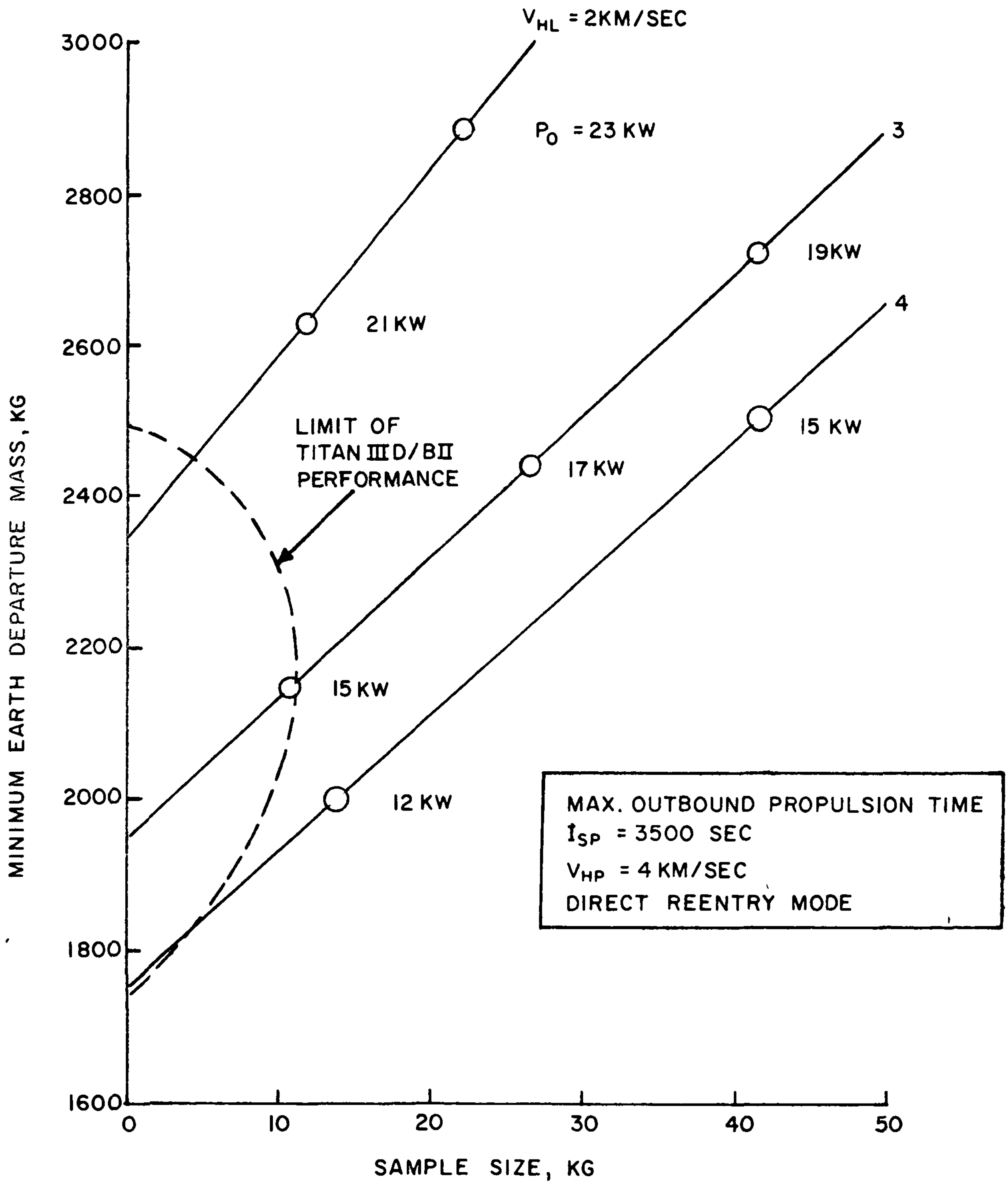


FIGURE 3-9. SOLAR ELECTRIC PERFORMANCE FOR FLORA SAMPLE-RETURN TO EARTH DIRECT REENTRY, LAUNCH 12/16/82, TRIP TIME 1360 DAYS.

return of 11 kg. Yet if one allows for a reasonable margin of safety it must be concluded that the Titan IIID/Centaur is a more viable choice for the Flora mission.

The above results have tacitly assumed that a hyperbolic velocity at Earth return of 4 km/sec is optimum or nearly so for the 1340-day mission. In the analysis V_{HP} was varied between 3 and 6 km/sec. A value of about 4 km/sec was found to be optimum for the orbit capture mode. In the case of direct reentry a value of 5 km/sec increased the sample return capability by 2-5 kg.

3.1.3 Parametric Analysis

It is of interest to examine the sensitivity of sample size to various baseline parameter assumptions. Among these are the mass values assigned to the spacecraft bus and RDSS module, the ΔV requirement and specific impulse of the auxiliary chemical propulsion, the total trip time, and the stay time at Flora. In addition, the effect of low-thrust specific impulse, propulsion on-time and launch window will be described. The nominal conditions for the parametric analysis are the Titan IIID/Centaur launch vehicle, a 15 kw SEP system, and the orbit capture mode of recovery. Hence the "reference" sample return capability is 39.7 kg as shown in Figure 3-8.

The sensitivity to four system parameters is listed in Table 3-1. Adding 50 kg to the spacecraft bus or RDSS module decreases the sample return by 8.4 kg or 6.4 kg, respectively. An additional 100 m/sec ΔV requirement results in a 10 kg sample reduction. If the chemical propulsion subsystem operated at 240 sec specific impulse (monopropellant) the sample size would decrease by 13.4 kg. Taken individually, the performance degradations do not seem to be too serious on the basis of a 40 kg nominal sample size. The combined effect would however nullify any sample-return capability of the 15 kw SEP system.

TABLE 3-1

SENSITIVITY OF SAMPLE SIZE TO SEVERAL PARAMETER ASSUMPTIONS*

PARAMETER	PARAMETER VALUE A	B (Nominal)	C
SPACECRAFT BUS (KG)	500	450	400
RDSS MODULE (KG)	300	250	200
CHEMICAL ΔV (M/SEC)	600	500	400
CHEMICAL Isp (SEC)	240	310	380

PARAMETER	SAMPLE SIZE (KG) A	B	C	APPROXIMATE SENSITIVITY
SPACECRAFT BUS	31.3	39.7	48.1	−0.17 KG/KG
RDSS MODULE	33.3	39.7	46.1	−0.13 KG/KG
CHEMICAL ΔV	29.7	39.7	49.0	−0.10 KG/M/SEC
CHEMICAL Isp	26.4	39.7	48.0	+0.15 KG/SEC

* 1360-DAY MISSION TO FLORA (1982 LAUNCH)
TITAN III D/CENTAUR/SEP
P_o = 15 KW, Isp = 3500 SEC
MAX OUTBOUND PROPULSION TIME
ORBIT CAPTURE RECOVERY

Figure 3-10 shows the effect of varying the total trip time and the stay time at Flora. The maximum returned sample is 41 kg at a trip time of about 1340 days. Sample size decreases quite rapidly for trip times less than 1320 days or greater than 1360 days. There does not appear to be any significant change in sample size for stay times in the range 0-100 days. This allows considerable freedom to perform the desired operations at Flora. A delayed departure from Flora much beyond the nominal 90-day stay time is seen to be prohibitive. These results correlate well with the information displayed by the energy contour map (Figure 3-5).

The discussion so far has been restricted to the maximum sample returned at a specified power level such as 15 kw. It has been mentioned that this optimum condition is characterized by a propulsion time on the outbound trajectory very nearly equal to the flight time. If the maximum sample size is greater than needed to satisfy the science objectives, then the excess capability may be used to enhance other engineering design objectives such as reducing the propulsion time. Presumably the mission reliability can be increased by a shorter system operating time. A reduction in propulsion time requires an increase in the thrust acceleration. For fixed P_o and I_{sp} this can be achieved by reducing the initial mass at Earth departure and executing an early Centaur thrust termination or using ballast. Figure 3-11 illustrates the off-optimum design procedure. It should be noted that the propulsion time on the return transfer is also reduced since the vehicle mass at Flora departure is lower.

The combined effect of propulsion time and specific impulse is shown in Figure 3-12. At the baseline value of I_{sp} = 3500 sec, the maximum 40 kg sample requires a propulsion time of 1010 days. If a 25 kg sample were acceptable the propulsion time decreases to 900 days, or about 11 percent.

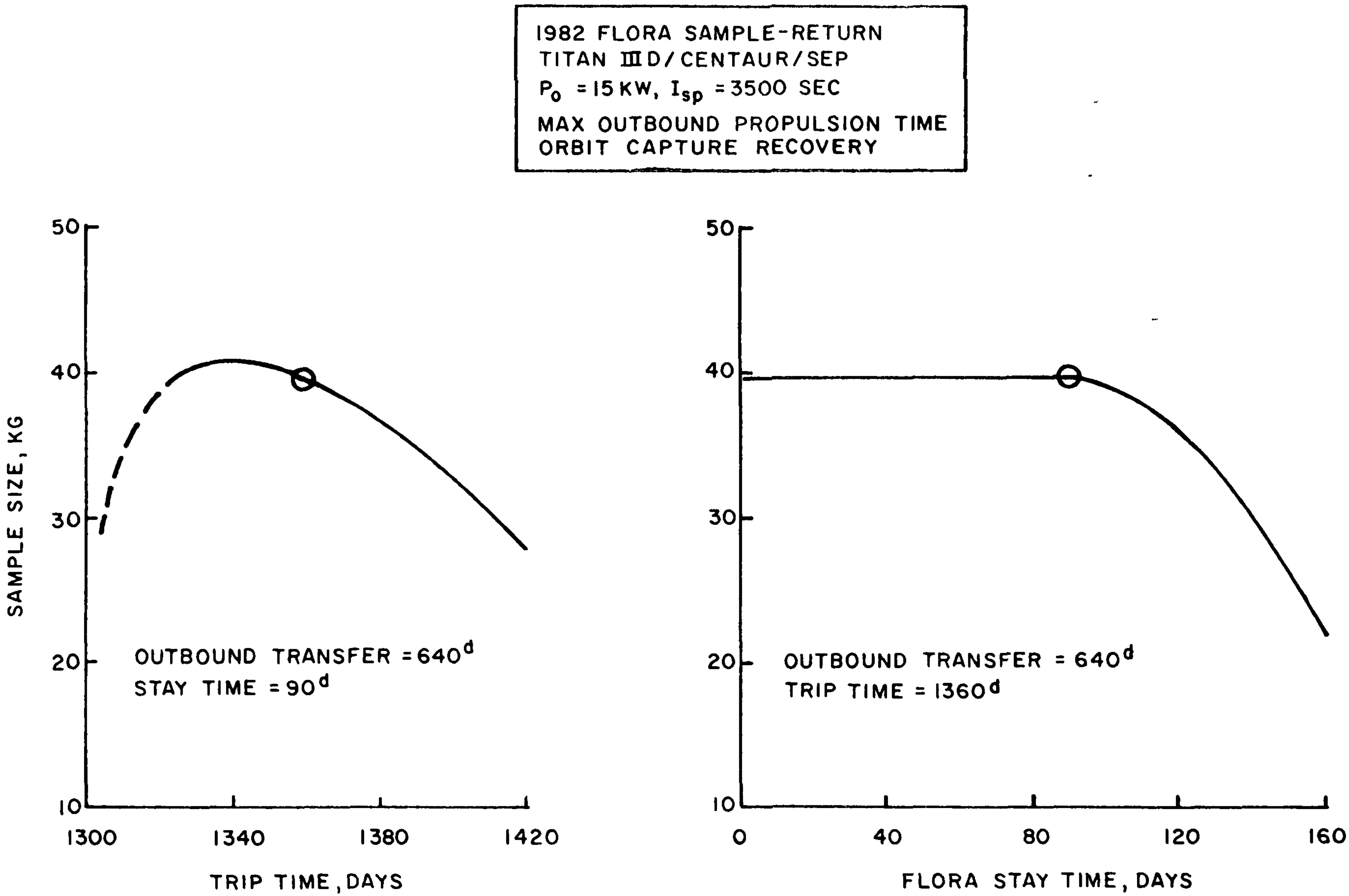


FIGURE 3-10. TRIP TIME AND STAY TIME EFFECTS ON SAMPLE RETURN CAPABILITY.

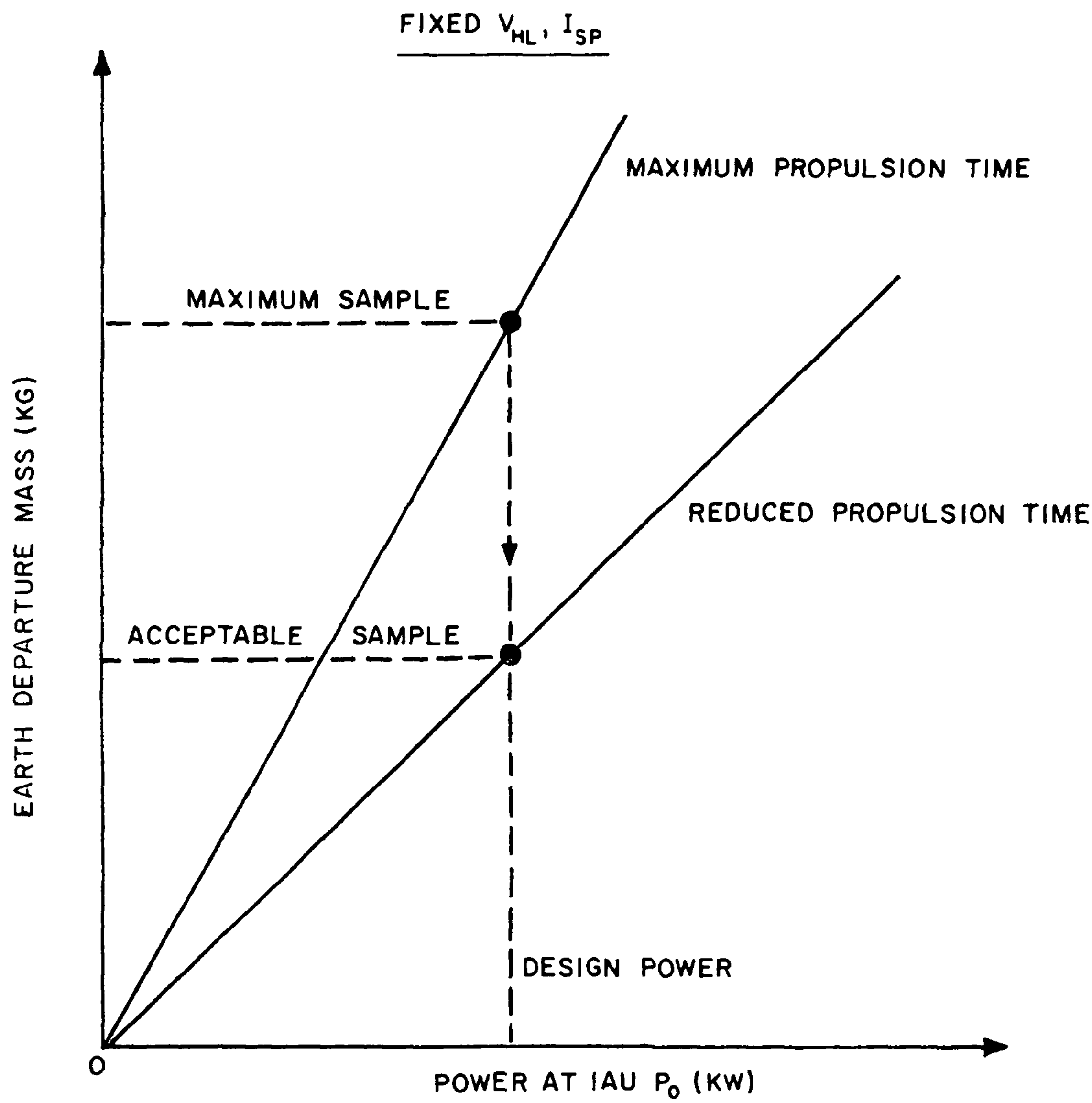


FIGURE 3-11. ILLUSTRATION OF OFF-OPTIMUM DESIGN SELECTION TO REDUCE PROPULSION TIME

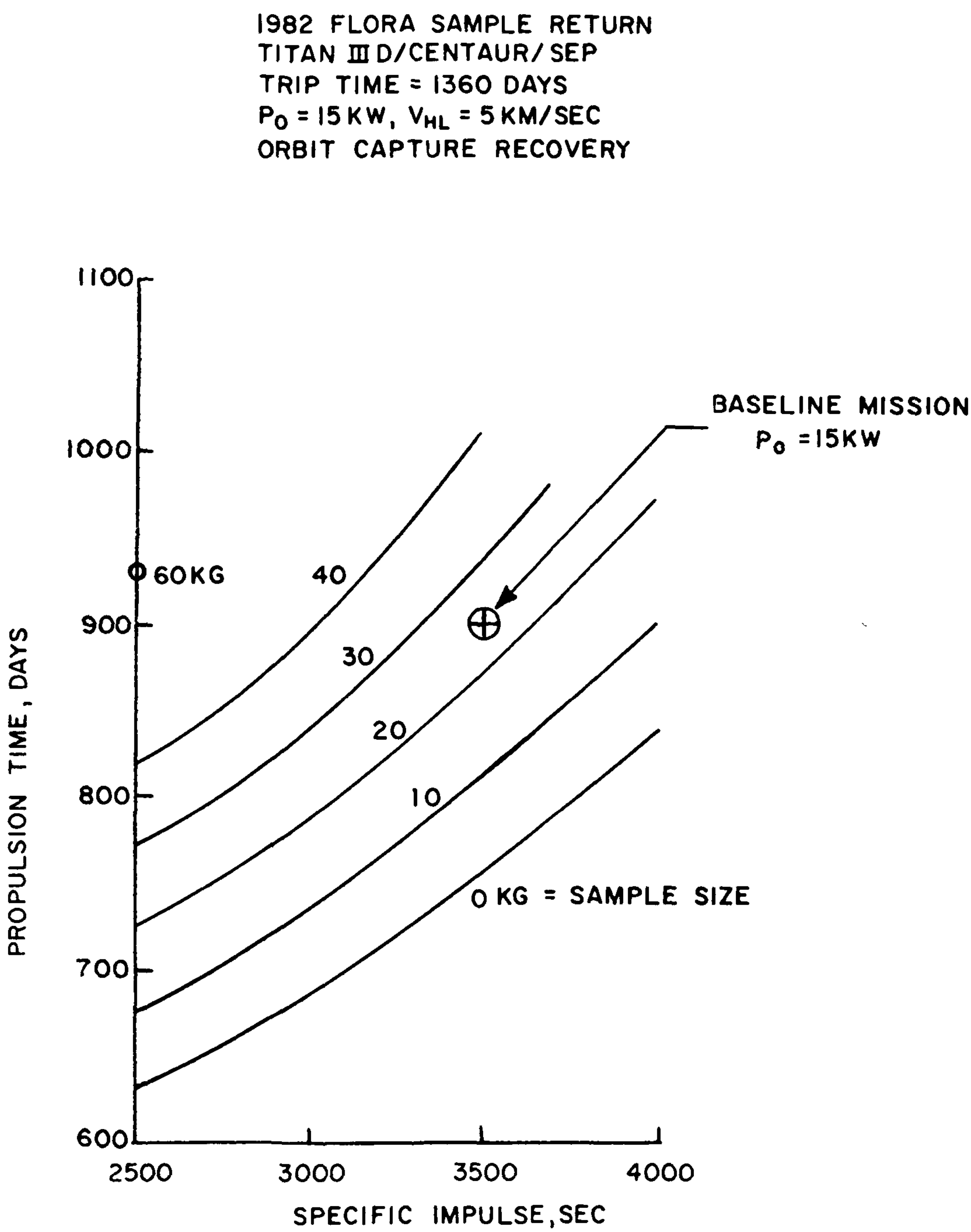


FIGURE 3-12. EFFECT OF SPECIFIC IMPULSE ON SAMPLE SIZE AND PROPULSION TIME.

Specific impulse is seen to have a strong effect on sample return capability and propulsion time. If a 2500 sec thruster design could be achieved, the maximum sample size is 60 kg and the propulsion time 930 days. Alternatively, a 25 kg sample would require a propulsion time of only 750 days.

The performance variation over the launch window is shown in Figure 3-13. Mass at Earth departure is held constant at 2308 kg and the nominal arrival date at Flora is also fixed (9/16/84). A 37-day launch window provides a sample-return capability of 25 kg. This is achieved at the expense of adding 27 kg to the nominal propellant loading.

3.1.4 Effect of Launch Year Opportunity

It will be recalled that launch opportunities for missions to Flora occur approximately every 1.3 to 1.5 years. Solar electric performance is not expected to vary significantly from one opportunity to the next. The table below gives the variation for the first three opportunities in the 1980 decade assuming the baseline conditions of the Titan IIID/Centaur, P_o = 15 kw, I_{sp} = 3500 sec, and M_o = 2308 kg.

Launch Date	Flight Time	Propulsion Time	Sample Size (Orbit Capture)
1/21/80	1340 days	964 days	18.6 kg
8/13/81	1270	944	26.0
12/16/82	1360	918	27.7

Ballistic flight mode results have not been obtained for the Flora mission. However, it has been shown that the "easier" Eros mission requires the Titan IIID(7)/Centaur and a space-storable (I_{sp} = 385 sec) retro system to perform the multi-impulse outbound and return transfers (Mascy and Niehoff, 1971).

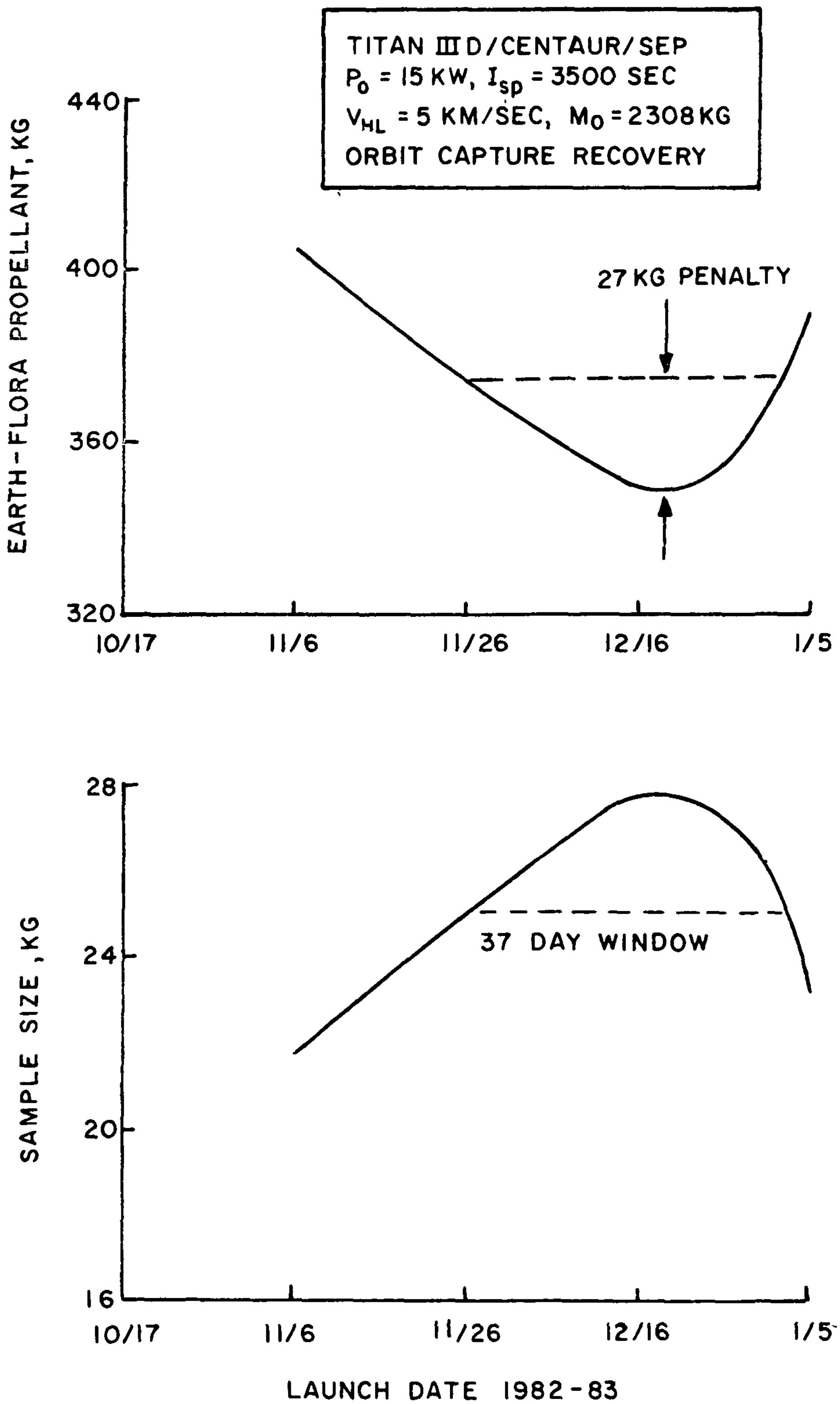


FIGURE 3-13. LAUNCH WINDOW PENALTY FOR NOMINAL 1360-DAY FLORA SAMPLE RETURN MISSION

It may be inferred therefore that SEP offers performance advantages over ballistic systems for the Flora sample-return mission.

3.2 Eros Mission

Mascy and Niehoff investigated SEP sample return from Eros for four launch opportunities in the time period 1975-82. They found that round-trip times are essentially invariant at approximately 3 years and that staytimes up to 4 months duration have little effect on sample sizes. They also found that the lowest energy opportunity (1977) would be closely repeated in the 1984 launch period. We have chosen to examine this latter opportunity since it is more favorable than 1977 from a programmatic viewpoint. A scan of launch and return dates was made for outbound and return transfer times of 480 days. Figure 3-14 shows the transfer profiles of a 1080-day mission launched on Jan. 20, 1984. The stay time at Eros is 120 days and the Earth return date is Jan. 4, 1987. It will be noted that Earth and Eros are in near-conjunction during encounter operations. Unfortunately, this poor communications geometry is characteristic of all Eros sample-return launch opportunities. Current DSN communications capability requires that the Earth-spacecraft line-of-sight be displaced at least 2° from the Earth-Sun line. It appears that this condition will be satisfied over the staytime interval (recall that Eros is inclined 11° to the ecliptic plane).

Figure 3-15 shows the maximum net mass (at rendezvous) capability of three Titan class launch vehicles as a function of SEP power. Net mass requirements are 950-1300 kg for a sample return of 10-50 kg. The Titan IIID performance falls within this range for SEP power between 6 and 12 kw; at optimum power the maximum sample return is about 30 kg. The Titan IIID/BII performance encompasses the full range of sample sizes and

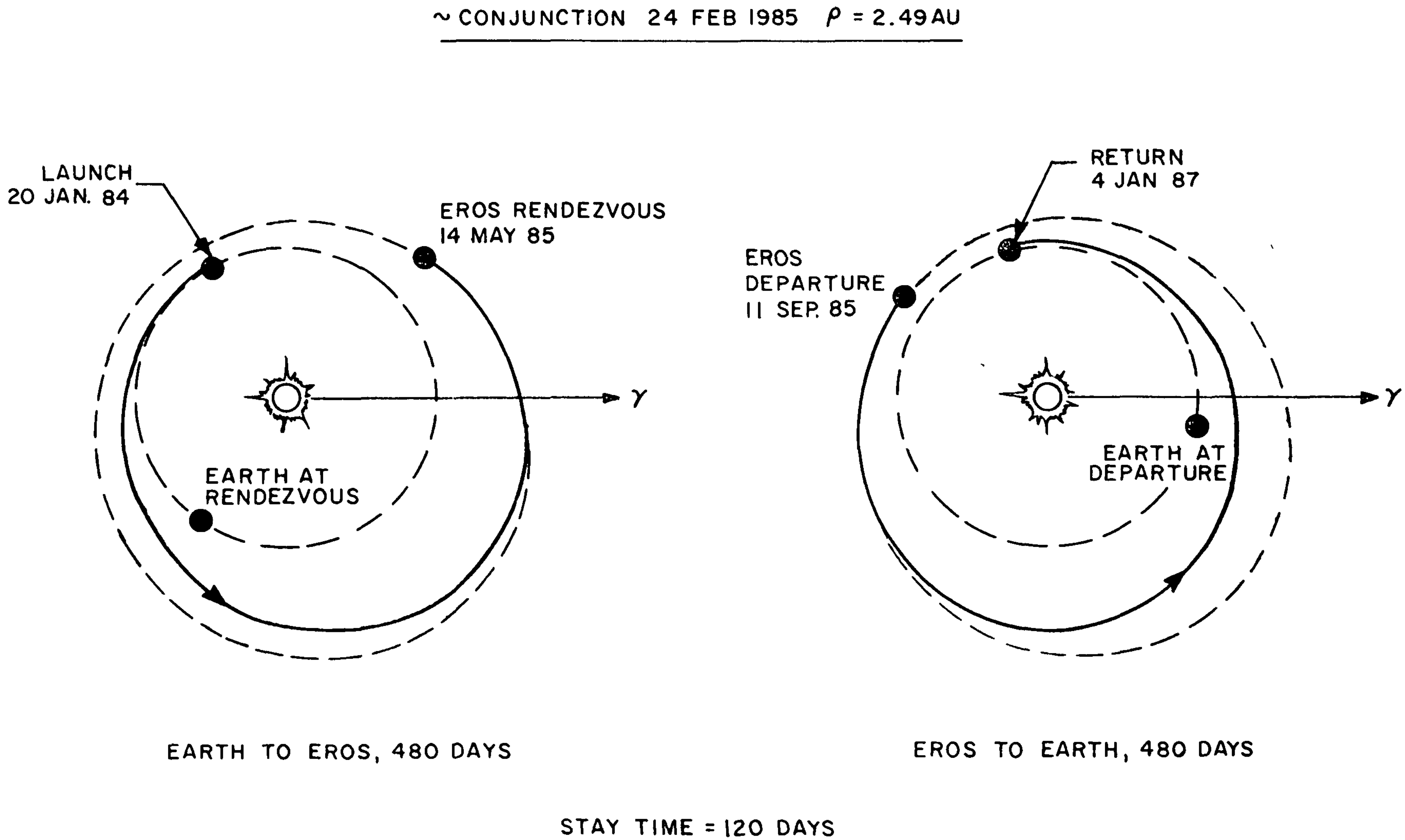


FIGURE 3-14. SOLAR ELECTRIC TRANSFER PROFILES FOR A 1080-DAY EROS SAMPLE RETURN MISSION, 1984 LAUNCH.

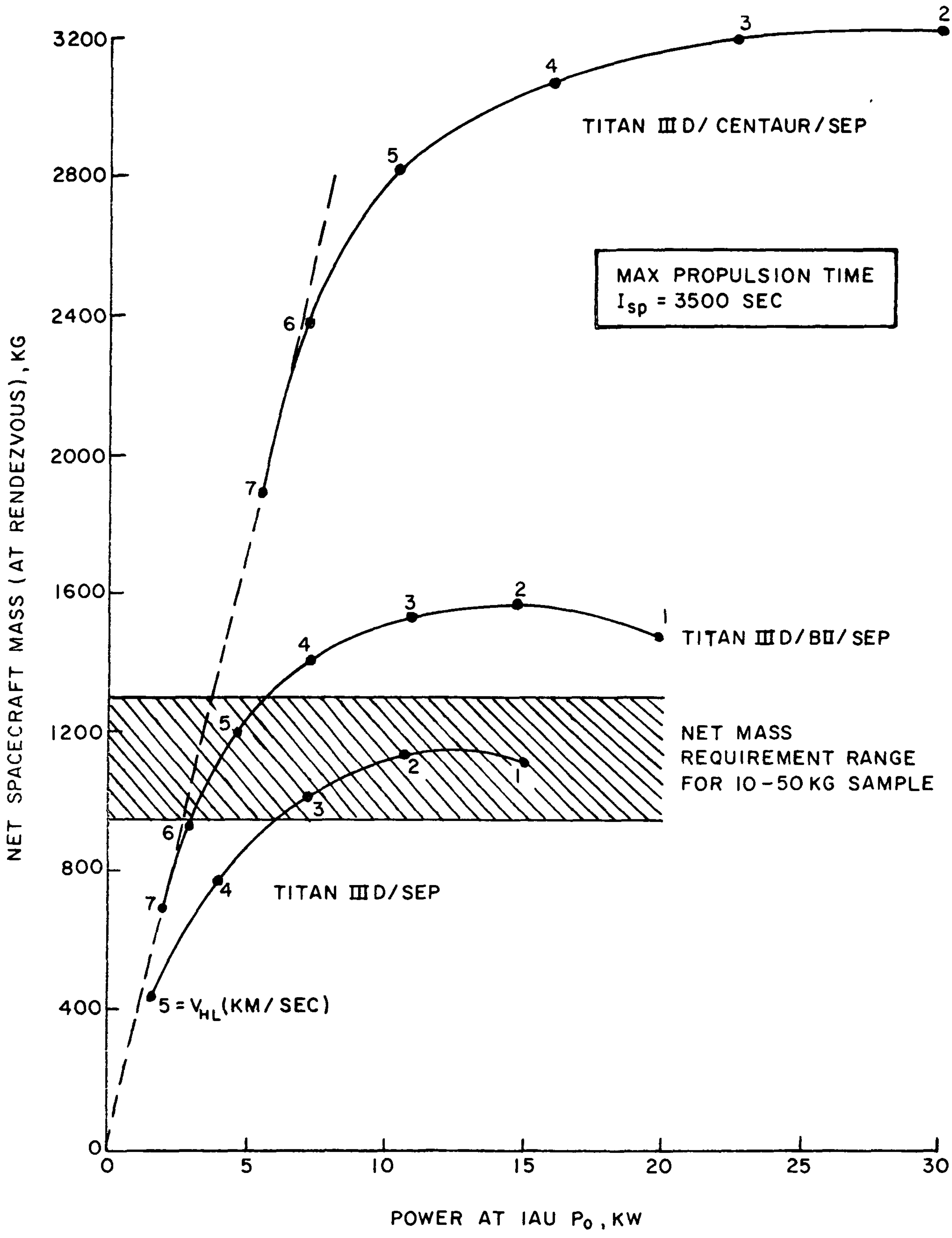


FIGURE 3-15. SOLAR ELECTRIC CAPABILITY FOR EROS RENDEZVOUS, LAUNCH 1/19/84, FLIGHT TIME 480 DAYS.

would require only 3-6 kw powerplants. One notes that the Titan IIID/Centaur capability is excessively high. Use of this launch vehicle should allow a reduction in trip time, but probably no less than 900 days. Alternatively, the Titan IIID/ Centaur could be used for the 1080-day mission, say at P_o = 10 kw, and result in an electric propulsion on-time of only 250-300 days. This type of mission design may be quite sensible. The SEP system would operate essentially as a low acceleration multi-impulse device, although a very efficient one compared to chemical propulsion.

Figures 3-16 and 3-17 show the basic SEP performance data as curves of Earth departure mass versus sample size for the orbit capture and direct reentry modes of recovery. As in the case of the Flora mission direct reentry provides slightly better performance. The capability limit of the Titan IIID shows a maximum sample return of about 27 kg (orbit capture) and 32 kg (direct reentry).

Figure 3-18 describes the effect of propulsion time on sample return capability for the three Titan class launch vehicles. These results assume a 10 kw powerplant operating at 3500 sec specific impulse. The Titan IIID mission has a maximum propulsion time of about 600 days and can return an 18 kg sample to Earth orbit. The Titan IIID/BII mission returns a 55 kg sample for a maximum propulsion time of 515 days. Alternatively, an 18 kg sample allows a propulsion time reduction to 415 days. Use of the Titan IIID/Centaur further reduces the propulsion time to only 250 days for an 18 kg sample.

A great deal of flexibility has been shown in utilizing SEP for the Eros sample-return mission. To reiterate a previously stated result, the 3-year ballistic mission would require the Titan IIID(7)/Centaur plus a high energy retro-propulsion system.

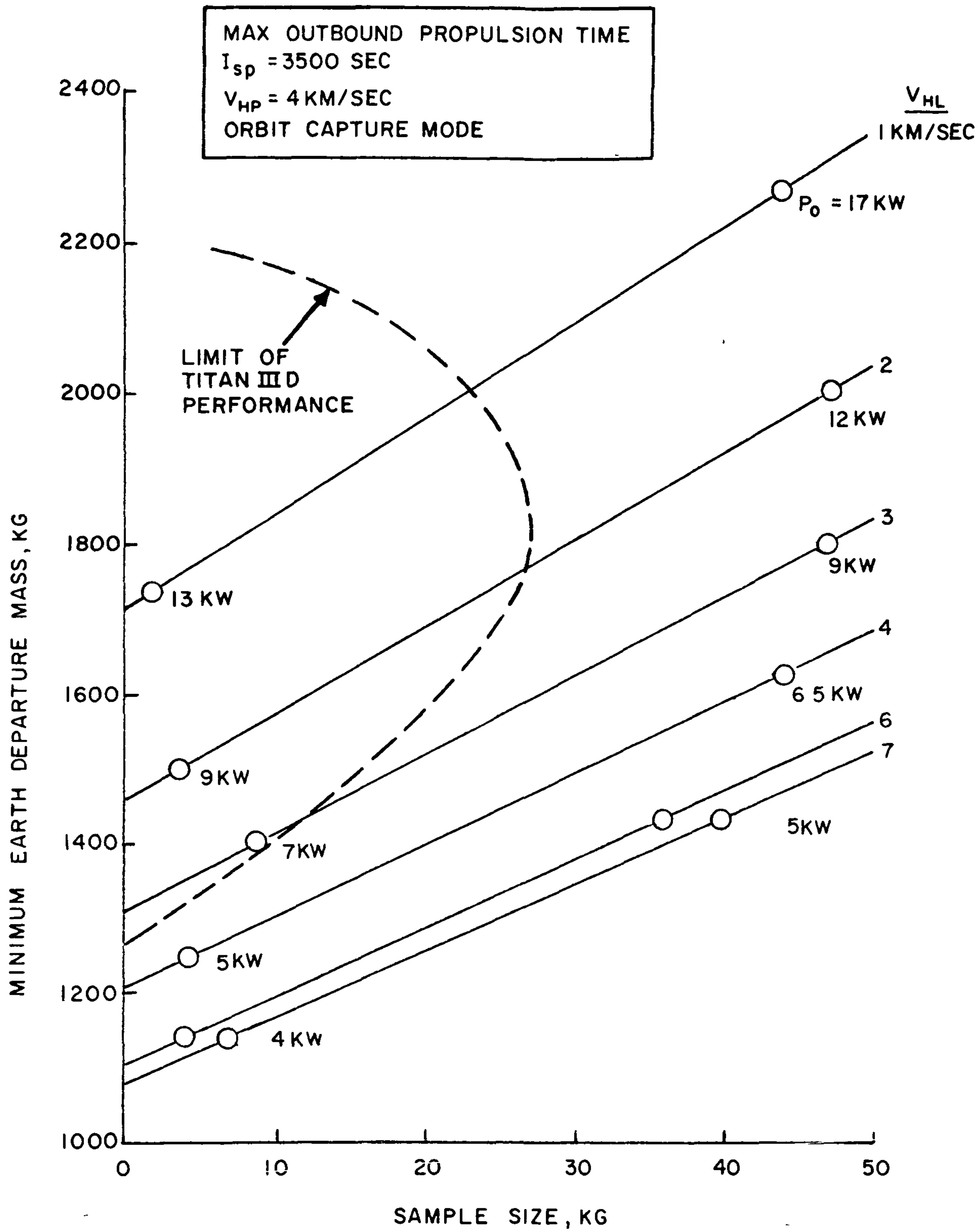


FIGURE 3-16. SOLAR ELECTRIC PERFORMANCE FOR EROS SAMPLE-RETURN TO EARTH ORBIT, LAUNCH 1/19/84, TRIP TIME 1080 DAYS.

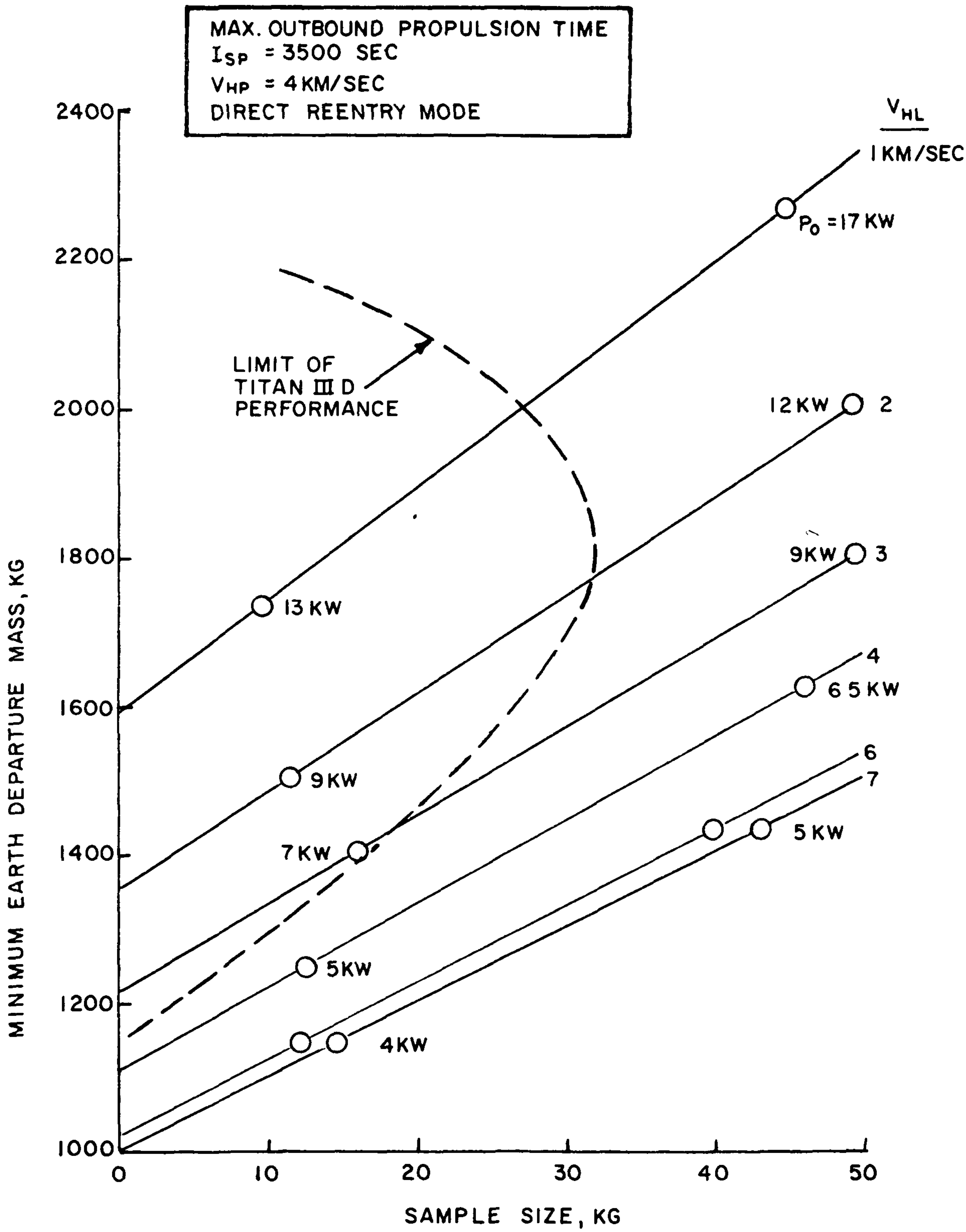


FIGURE 3-17. SOLAR ELECTRIC PERFORMANCE FOR EROS SAMPLE-RETURN TO EARTH DIRECT REENTRY, LAUNCH 1/19/84, TRIP TIME 1080 DAYS

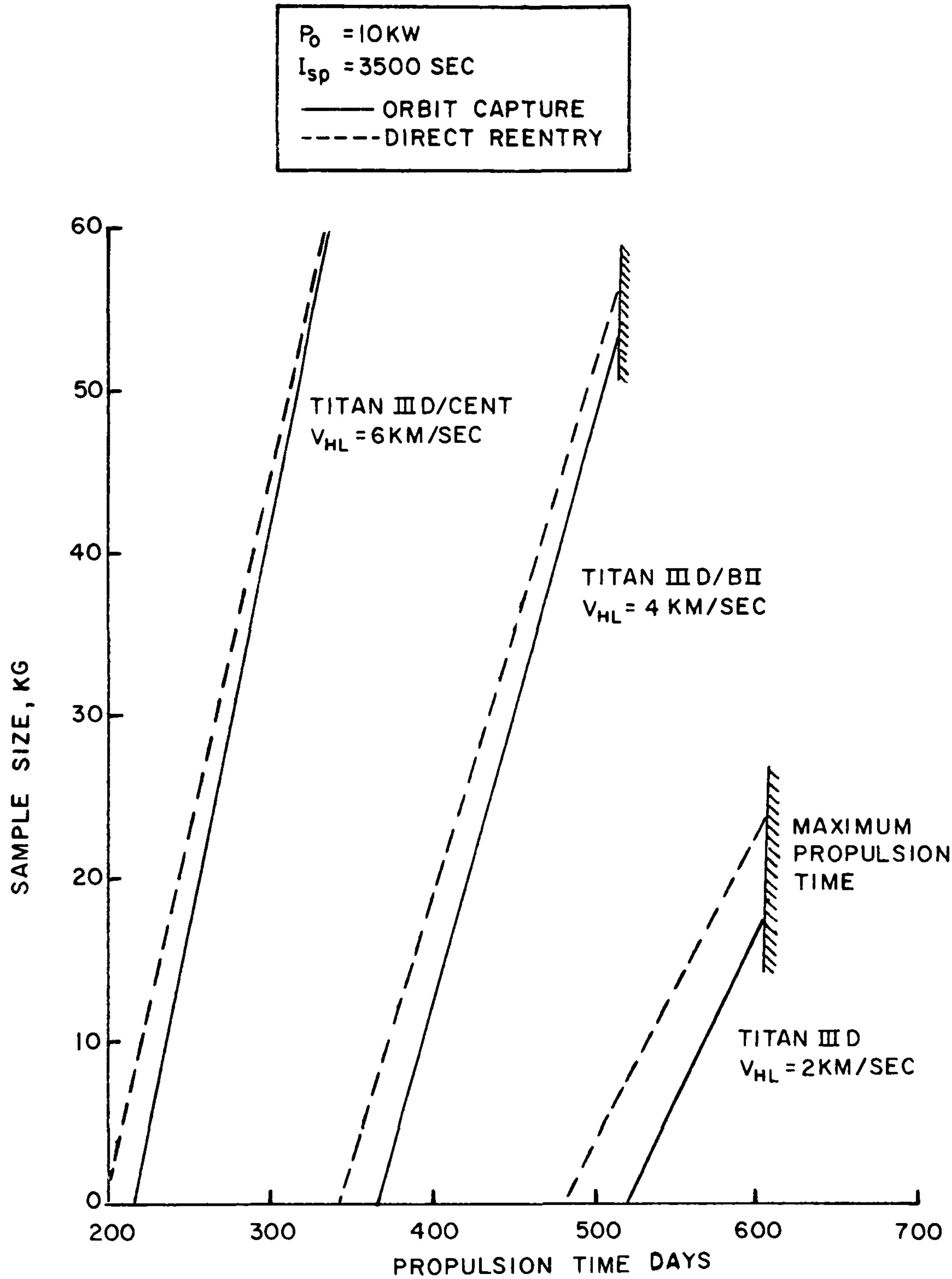


FIGURE 3-18. EFFECT OF PROPULSION TIME ON SAMPLE RETURN CAPABILITY FOR 1984 EROS MISSION

4. BASELINE MISSION PERFORMANCE SUMMARY

As a means of bringing together the study results it will be useful to select representative design-point examples for the Flora and Eros missions. A sample size is chosen for each baseline mission which allows a nominal margin between Earth departure mass and launch vehicle capability. Sample recovery in Earth orbit is assumed for each example. The mercury propellant loading includes an increment to provide a launch window of 30-40 days.

The Flora mission returns a 25 kg sample and utilizes the Titan IIID/Centaur and a 15 kw SEP system. A spacecraft weight summary and performance sequence of events are given in Tables 4-1 and 4-2. Of approximately 2300 kg initial mass, 46% is allocated to solar electric propulsion and 21% to chemical propulsion maneuvers. The total trip time is 1360 days (3.73 years) of which 923 days are nominally required for low-thrust propulsion. Note that the declination of the hyperbolic asymptote at launch (DLA) is 34° which should satisfy the ETR range safety requirements. Titan IIID/Centaur payload capability at V_{HL} = 5 km/sec is 3450 kg; this is 1000 kg more than the required gross mass of the interplanetary vehicle.

Figure 4-1 shows the time profiles of the SEP power input and the optimum thrust cone angle (sun-spacecraft-thrust vector). There are two main thrust periods, the first beginning 78 days after launch and ending at Flora rendezvous, and the second beginning at Flora departure and lasting 361 days. The input power varies between 5.4 and 12 kw throughout the mission. Note that the thrust cone angle variation is relatively small (72° - 88°). This indicates that a fixed orientation of the thrusters with respect to the solar array (e.g., 80°) would incur a small performance penalty but achieve a much simpler engineering design.

TABLE 4-1

WEIGHT SUMMARY FOR 1982 FLORA SAMPLE-RETURN MISSION

(25 KG SAMPLE, TITAN III D/CENTAUR)

RENDEZVOUS, DOCKING, SCIENCE, AND SAMPLING (RDSS) MODULE		250 (KG)
SAMPLE CAPSULE (EMPTY)		55
EARTH CAPTURE STAGE		147
SPACE BUS SUBSYSTEMS AND STRUCTURE		450
AUXILIARY CHEMICAL PROPULSION		345
SOLAR ELECTRIC SYSTEM		1061
PROPULSION SYSTEM (15 KW)	450	
PROPELLANT + TANKAGE	611	
LAUNCH VEHICLE ADAPTER		115
GROSS EARTH DEPARTURE MASS		2423

TABLE 4-2

PERFORMANCE SEQUENCE FOR 1982 FLORA SAMPLE-RETURN MISSION

SEP: P_0 = 15KW, I_{sp} = 3500 SEC

EVENT	PARAMETER	DATES	MASS (KG)	TIME (DAYS)	PARAMETER
LAUNCH	DATE	12/15/82			
	GROSS MASS		2423		
	LAUNCH VEHICLE				TITAN IIID/CENT.
	VHL(KM/SEC)/DLA(DEG)				5/34
	LV CAPABILITY(KG)				3450
EARTH DEPARTURE	DATE	12/16/82			
	INJECTED MASS		2308		
	OUTBOUND TRANSFER TIME			640	
	PROPULSION TIME(DAYS)				562
FLORA RENDEZVOUS	DATE	9/16/84			
	MASS		1959		
DOCKING	DATE	11/15/84			
	MASS		1749		
SEPARATION	DATE	12/13/84			
	MASS		1524		
	STAYTIME			90	
FLORA DEPARTURE	DATE	12/15/84			
	MASS		1448		
	RETURN TRANSFER TIME			630	
	PROPULSION TIME				361
EARTH RETURN/ CAPTURE	DATE	9/6/86			
	APPROACH MASS		1240		
	SEPARATED ORBITER MASS		227		
SYNOPSIS	RETURN SAMPLE		25		
	TOTAL TRIP TIME			1360	

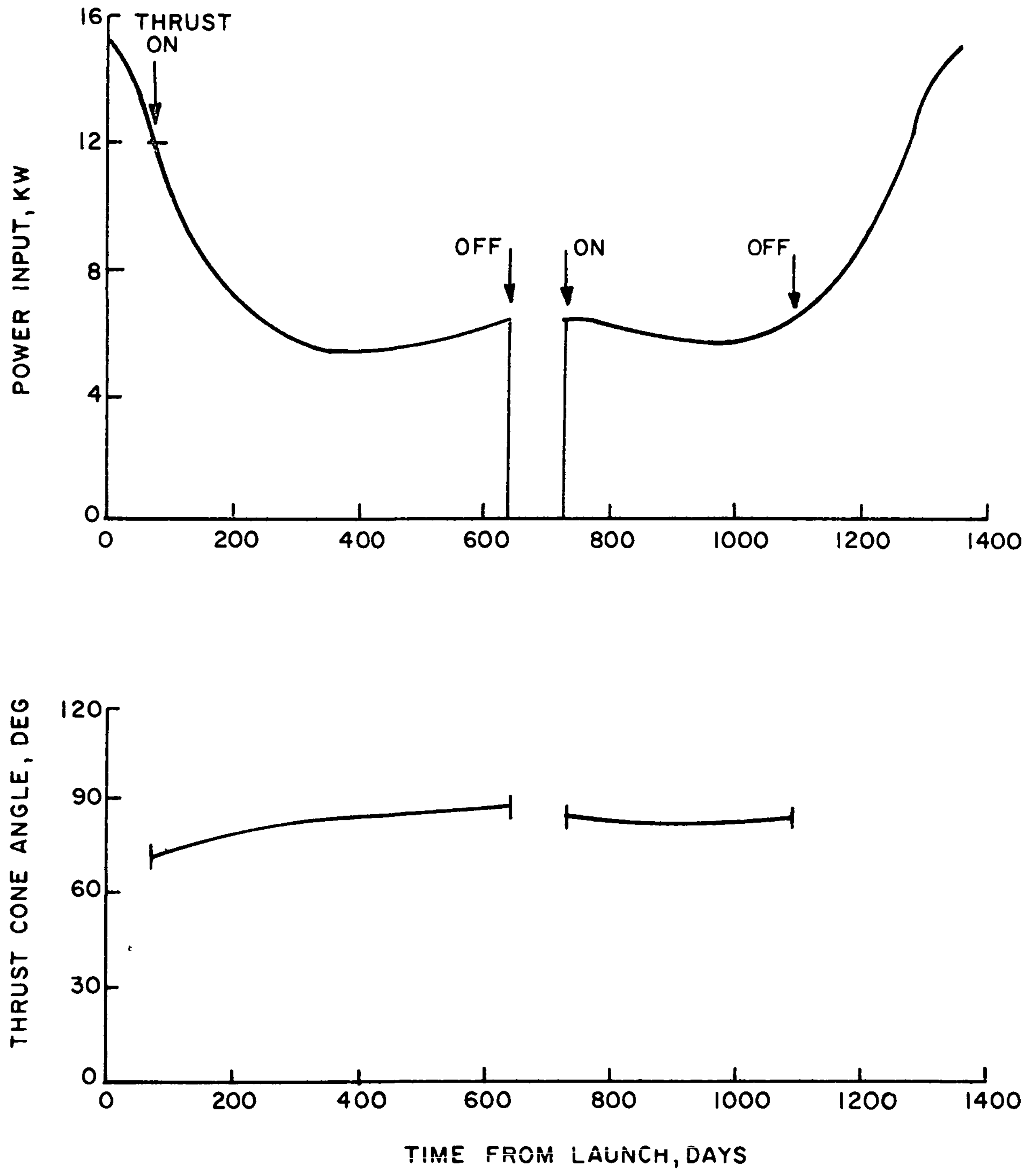


FIGURE 4-1. POWER PROFILE AND THRUST CONE ANGLE FOR FLORA SAMPLE-RETURN MISSION.

The Eros mission launched in 1984 has a total trip time of 1080 days (2.96 years) and utilizes a 10 kw SEP system. Two mission/design examples are described to illustrate a possible trade-off in sample return capability, launch vehicle and propulsion time. Tables 4-3 and 4-4 show the spacecraft weight summary and performance timeline.

Mission No. 1 can be launched by the Titan IIID, requires 577 days propulsion time, and returns a 10 kg sample. Mass at Earth departure is 1600 kg of which 44% is allocated to solar electric propulsion and only 10% to chemical propulsion. Launch vehicle capability at V_{HL} = 2 km/sec is 90 kg above the required gross mass. The DLA of 48° is somewhat higher than ETR safety requirements. From a practical standpoint the Titan IIID may not be a good choice since it does not have restart capability in Earth orbit. Hence the mission would necessitate a direct launch to hyperbolic escape injection with a corresponding narrow daily launch window. Injection from Earth orbit is much more flexible and is preferred. An alternate launch vehicle is the Titan IIIC which can be restarted in orbit and has significantly more payload capability than is necessary.

Eros Mission No. 2 is launched by the Titan IIID/BII, requires only 443 days propulsion time, and returns a 25 kg sample. Earth departure mass is also 1600 kg of which the SEP fraction is 39% and the chemical propulsion fraction is 14%. Launch vehicle capability at V_{HL} = 4 km/sec is 140 kg above the required gross mass. The excess performance may have to be used to achieve the 64° DLA requirement.

Figure 4-2 shows the input power and thrust angle profiles for the second Eros mission example. The computer results gave three thrust periods for the outbound transfer

TABLE 4-3

WEIGHT SUMMARY FOR 1984 EROS SAMPLE-RETURN MISSION

	MISSION EXAMPLE 1 10 KG SAMPLE, TITAN IIID		MISSION EXSAMPLE 2 25 KG SAMPLE, TITAN IIID/BII	
RENDEZVOUS, DOCKING, SCIENCE, AND SAMPLING (RDSS) MODULE		250 (KG)		250 (KG)
SAMPLE CAPSULE (EMPTY)		39		55
EARTH CAPTURE STAGE		89		147
SPACECRAFT BUS SUBSYSTEMS AND STRUCTURE		450		450
AUXILIARY CHEMICAL PROPULSION		73		80
SOLAR ELECTRIC SYSTEM PROPULSION SYSTEM (10 KW)	300	699	300	618
PROPELLANT + TANKAGE	399		318	
LAUNCH VEHICLE ADAPTER		80		80
GROSS EARTH DEPARTURE MASS		1680		1680

TABLE 4-4

PERFORMANCE SEQUENCE FOR 1984 EROS SAMPLE-RETURN MISSIONS

SEP. P_0 = 10 KW, I_{sp} = 3500 SEC

EVENT	PARAMETER	DATES	MISSION EXAMPLE 1			MISSION EXAMPLE 2		
			MASS (KG)	TIME (DAYS)	PARAMETER	MASS (KG)	TIME (DAYS)	PARAMETER
LAUNCH	DATE	1/19/84						
	GROSS MASS		1680			1680		
	LAUNCH VEHICLE				TITAN III D			TITAN IIID/BII
	VHL(KM/SEC)/DLA(DEG)				2/-48			4/-64
	LV CAPABILITY (KG)				1770			1820
EARTH DEPARTURE	DATE	1/20/84						
	INJECTED MASS		1600			1600		
	OUTBOUND TRANSFER TIME			480			480	
	PROPULSION TIME (DAYS)				429			276
EROS RENDEZVOUS	DATE	5/14/85						
	MASS		1330			1441		
DOCKING	DATE	8/12/85						
	MASS		1289			1396		
SEPARATION	DATE	9/9/85						
	MASS		1049			1171		
	STAY-TIME			120			120	
EROS DEPARTURE	DATE	9/11/85						
	MASS		1030			1150		
	RETURN TRANSFER TIME			480			480	
	PROPULSION TIME				148			167
EARTH RETURN/ CAPTURE	DATE	1/4/87						
	APPROACH MASS		929			1035		
	SEPARATED ORBITER MASS		138			227		
SYNOPSIS	RETURN SAMPLE		10			25		
	TOTAL TRIP TIME			1080			1080	

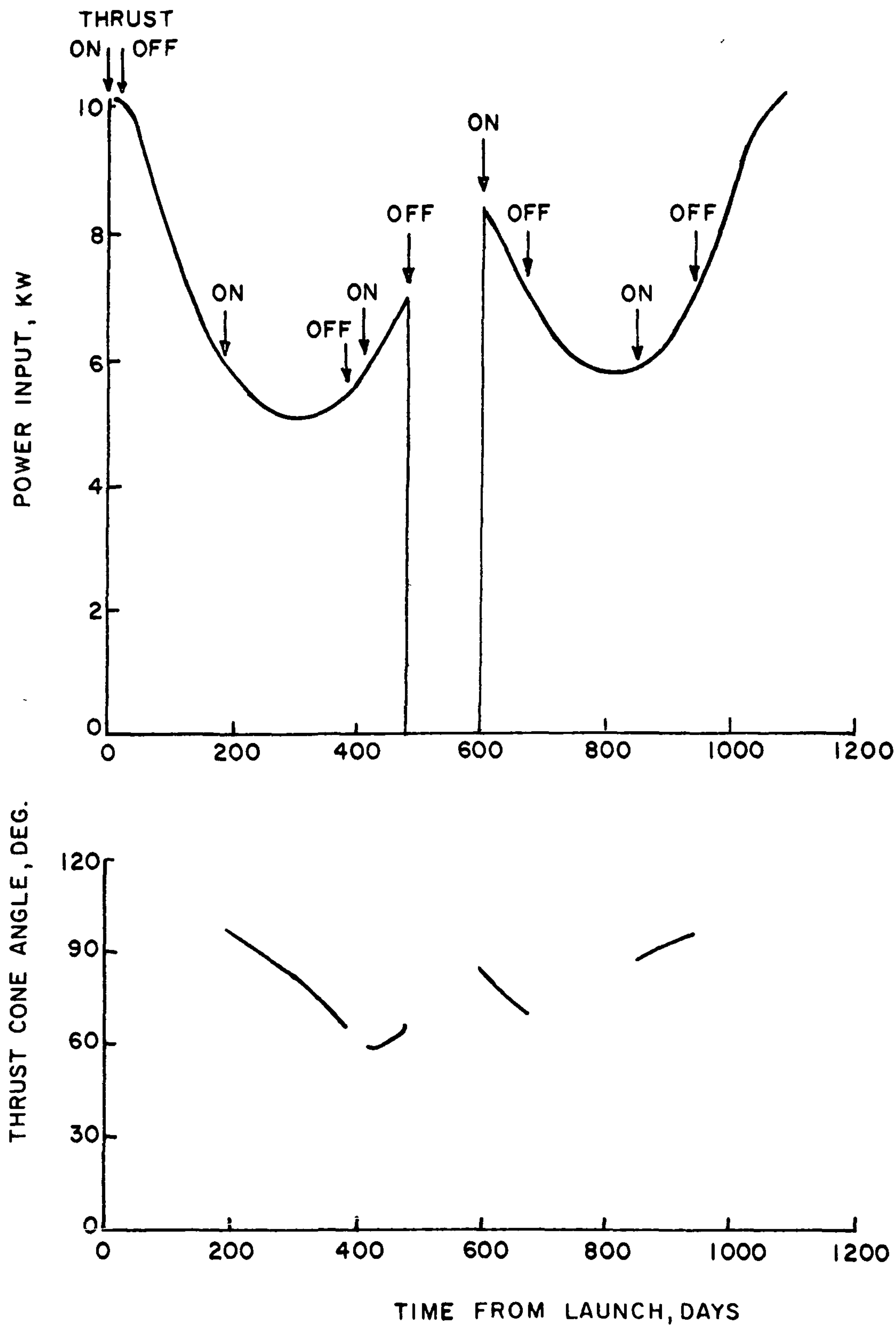


FIGURE 4-2. POWER PROFILE AND THRUST CONE ANGLE FOR EROS SAMPLE-RETURN MISSION.

and two for the return transfer. There is reason to believe that the 15-day initial period is a spurious result. In any event the outbound trajectory could be constrained, with negligible performance loss, to eliminate the first propulsion interval. In such case the input power variation is reduced to the range 5 - 8.5 kw; this should result in a more favorable thruster switching sequence. The optimum thrust cone angle varies between 60° and 98°, and is generally less uniform than in the Flora mission. However, a single or perhaps two fixed orientations should be adequate to closely approximate the optimum thrust vector performance.

5. CONCLUSIONS

The study results have shown that solar electric propulsion can be used quite effectively to accomplish sample-return missions to the asteroids Flora and Eros. Flora, being representative of the orbit class and size of other Main Belt asteroids, is a more difficult mission in terms of propulsive requirements and trip time. The minimum SEP powerplant size is about 15 kw to return a nominal 25 kg sample. Within the Titan family of launch vehicles the Titan IIID/Centaur is required. Although sample-return size does vary between different launch opportunities it appears that any opportunity in the 1980 decade would yield acceptable performance. The Eros mission characteristics may be generally representative of other small Mars-crossing asteroids. A 10 kw SEP system launched by the Titan IIID/BII is a suitable design choice. If desirable, a common 15 kw powerplant and the Titan IIID/Centaur could be employed for both Flora and Eros missions. Even though the Eros mission would be "overpowered", certain engineering advantages such as a very short thruster on-time is obtained. However, the total spacecraft would not be common to both missions since the auxiliary chemical propulsion system is considerably larger for the Flora mission.

Sample-return capability data has also been presented in a form which is independent of the launch vehicle employed. Therefore launch vehicle selections other than the Titan family may be examined. In particular the Shuttle vehicle systems may be a more timely choice for missions in the 1980 decade.

Sample-return missions to the large Main Belt asteroids such as Ceres and Vesta have yet to be studied. Such missions have higher propulsive requirements which means that the performance advantage of solar electric propulsion compared to ballistic systems will be even more significant. It is expected however that powerplants in the 20-30 kw range will be necessary to accomplish these missions.

REFERENCES

Adams, J. D., Drowns, R. E. and Hazelrigg, G. A., "Analysis of a Round-Trip Mission to Eros Using Chemical Propulsion", AAS Paper No. 71-369, AAS/AIAA Astrodynamics Specialists Conference, Ft. Lauderdale, Fla., Aug. 17-19, 1971.

Alfven, H. and Arrhenius, G., "Missions to an Asteroid", Science, Vol. 167, Jan. 9, 1970, pp. 139-141.

Blahnik, J. E. (ed.), "Automated Lunar Exploration Study", Astro Sciences Report No. M-29, IIT Research Institute, Chicago, Illinois, October 1971.

Gehrels, T., Photometry of Asteroids, Chapter 6, in <u>Surface and Interiors of Planets and Satellites</u>, A. Dollfus, ed., Academic Press, London and New York, 1970, Chapter 6.

Hahn, D. W., et al., "Chebychev Trajectory Optimization Program (CHEBYTOP)", Report No. D2-121308-1, Boeing Company, July 1969.

Mascy, A. C., and Niehoff, J. C., "Sample Return Missions to the Asteroid Eros", presented at Twelfth Colloquium of the International Astronomical Union, Physical Studies of Minor Planets, University of Arizona, Tucson, Ariz. Mar. 10, 1971.

Northrop Services, Inc., "A Feasibility Study of Unmanned Comet and Asteroid Rendezvous and Docking Concepts Using Solar Electric Propulsion", Final Briefing Document for MSFC under Contract NAS8-27206, September 1971.

Odom, P. R., "Automated Mars Surface Sample Return Mission/System Study, Vol. II A - Mission Analysis", Report No. TR-793-717, Northrop Corporation, February 1970.

Spadoni, D. J. and Friedlander, A. L., "Mars Surface Sample Return Missions Via Solar Electric Propulsion", Astro Sciences Report No. T-29, IIT Research Institute, Chicago, Illinois,October, 1971.