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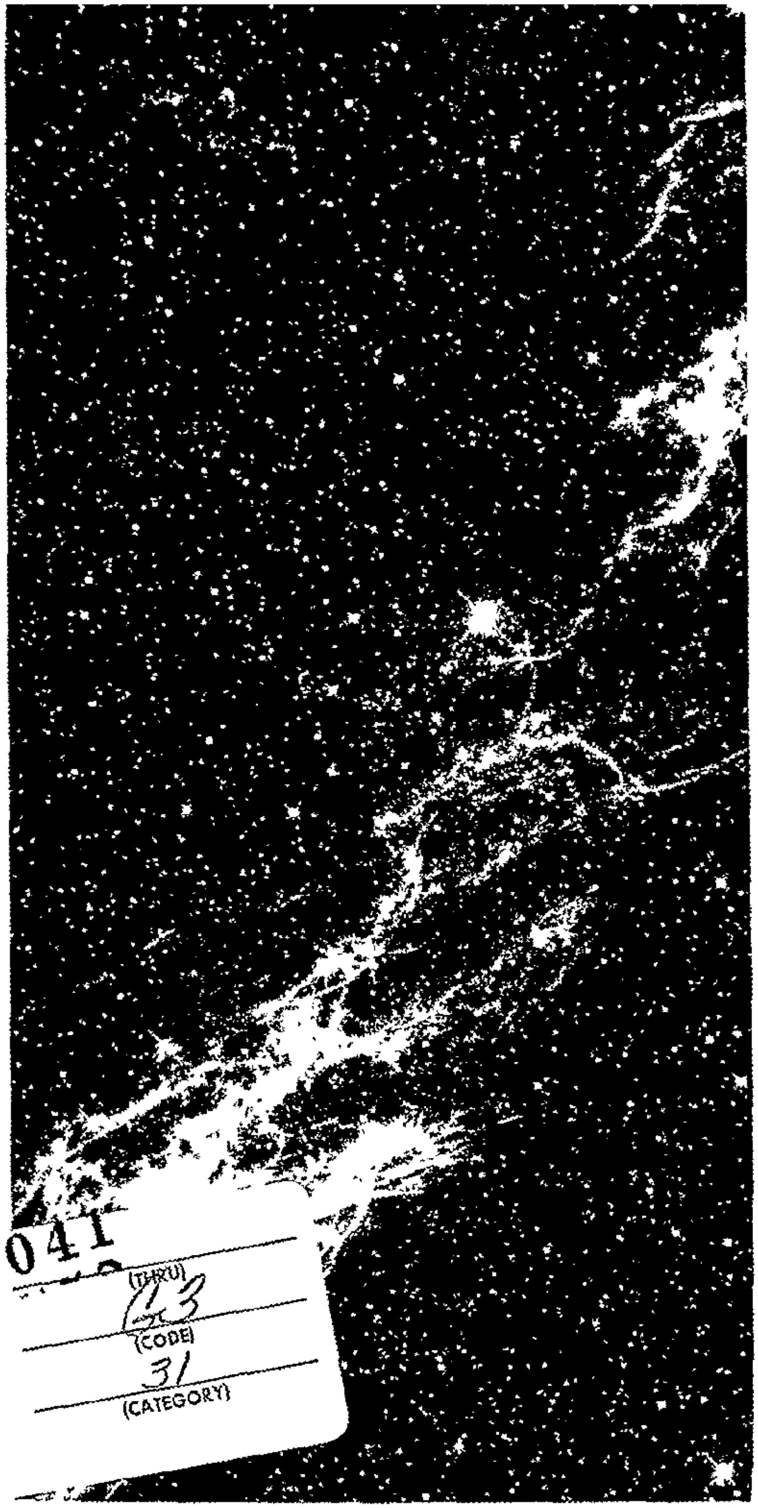

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SOLAR ELECTRIC PROPULSION FOR JUPITER AND SATURN ORBITER MISSIONS

IIT RESEARCH INSTITUTE
10 West 35 Street
Chicago, Illinois 60616

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SOLAR ELECTRIC PROPULSION FOR JUPITER AND SATURN ORBITER MISSIONS

by

A. Friedlander
R. Brandenburg

Astro Sciences
IIT Research Institute
Chicago, Illinois 60616

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APPROVED:

D L Roberts

D. L. Roberts, Manager
Astro Sciences

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FOREWORD

This Technical Report is the final documentation on all data and information required by Task 3: Jupiter Orbiter Mission Study, and Task 4: Saturn Orbiter Mission Study. The work reported herein represents the second 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. Phase 1 (Tasks 1 and 2) study documentation has been reported under separate cover entitled: "Solar Electric Propulsion - A Survey, Technology Status and Mission Applications", March 1970.

SUMMARY AND CONCLUSIONS

This report describes the characteristics and capabilities of solar electric propulsion (SEP) for performing orbiter missions at the planets Jupiter and Saturn. A set of four candidate missions comprising different orbit and payload selections is defined for each planet. These candidate missions provide a suitable basis for achieving a range of desired science objectives. The capabilities of the SEP flight mode are compared against those of the ballistic flight mode.

Several important factors related to the SEP operational concept have evolved from this study:

(1) Use of Titan 3D/Centaur launch vehicle

(2) Selection of direct transfer trajectories (heliocentric transfer angle less than 360°)

(3) SEP system (solar array and thrust subsystem) is jettisoned prior to orbit capture maneuver

(4) Use of common SEP stage (P_o = 15kw, I_{sp} = 3500 sec) for both Jupiter and Saturn missions.

Selection of the SEP jettison option results from the lack of adequate solar array power to satisfy orbit payload requirements at Jupiter and Saturn distances, particularly the latter. This also eliminates the need to retract the large flexible array prior to the high acceleration (0.2-0.5 g's) retro maneuver with the attendant advantage of mechanization simplicity. It becomes inappropriate then to speak of SEP orbiters of Jupiter and Saturn. Rather, the SEP spacecraft is viewed as an additional stage above the Titan 3D/Centaur which delivers the combined mass of the orbiting spacecraft and the chemical retro system needed to achieve the desired orbit. The optimum SEP power rating and specific impulse (yielding maximum net mass in orbit) lie in the range 25-30kw and 2700 - 2900 seconds for each mission application. It is found that an off-optimum but more practical SEP engineering design of 15kw and 3500 seconds could be employed as a common stage for both Jupiter and Saturn missions. This results in a net mass reduction of 15 percent relative to optimum values. Net mass is defined here as consisting of the science payload and spacecraft support subsystems but does not include the retro stage.

Summary Table 1 describes the reference mission selections in terms of the orbit periapse distance, period and inclination, the required net mass in orbit, and the mission science objectives. Suitable orbiter measurements divide into two classes: (1) particles and fields experiments requiring large elliptical orbits, and (2) planetology experiments preferring small short-period orbits. Jupiter Mission No. 1 is representative of the particles and fields orbiter class. The net mass requirement of 226kg per spacecraft is based on Pioneer F/G technology. Note that two spacecraft are specified, one injected into an equatorial orbit and the other into a polar-type orbit. The two-spacecraft concept is thought to be

SUMMARY TABLE I

CANDIDATE MISSION SELECTIONS

PLANET	MISSION NO.	PERIAPSE	ORBIT PERIOD*	INCLINATION	NET MASS IN ORBIT	SCIENCE OBJECTIVE
JUPITER	1	$3R_J$	45^d	0° AND 120°	226KG (PER SPACECRAFT)	PARTICLES AND FIELDS (2 SPACECRAFT)
	2	$3R_J$	15^d	60°	641 KG	PLANETOLOGY
	3	$2.29R_J$	14.222^d	0°	664 KG	PLANETOLOGY/ SATELLITE OBSERVATION
	4	$3R_J$	30^d	60°	705 KG	PLANETOLOGY/ PARTICLES & FIELDS
SATURN	1	$3R_S$	45^d	0° AND 90°	231 KG (PER SPACECRAFT)	PARTICLES & FIELDS/ RING PROBE (2 SPACECRAFT)
	2	$3R_S$	15^d	60°	642 KG	PLANETOLOGY/ RINGS
	3	$1.1R_S$	7.5^d	60°	642 KG	PLANETOLOGY/ RINGS
	4	$3R_S$	15.9^d	0°	660 KG	PLANETOLOGY/ SATELLITE OBSERVATION

*EARTH DAYS

JUPITER AND SATURN MISSION NO. 1: PIONEER F/G TECHNOLOGY
ALL OTHER MISSIONS: TOPS TECHNOLOGY

most suitable to providing the desired coverage of the large region of the Jovian magnetosphere and shock front. Jupiter Mission No. 2, 3, and 4 emphasize planetology experiments, and the net mass requirement in the range 640-700kg is representative of TOPS technology (JPL's proposed Grand Tour spacecraft). Mission No. 3 includes the capability of multiple observations of the Galilean satellites (Io, Europa, Ganymede and Callisto), while Mission No. 4 provides a compromise option for a combined planetology/particles and fields orbiter. A periapse distance of 3 Jupiter radii is selected to avoid the hazard of Jupiter's known radiation belts (the satellite tour mission has a special requirement for a smaller periapse distance).

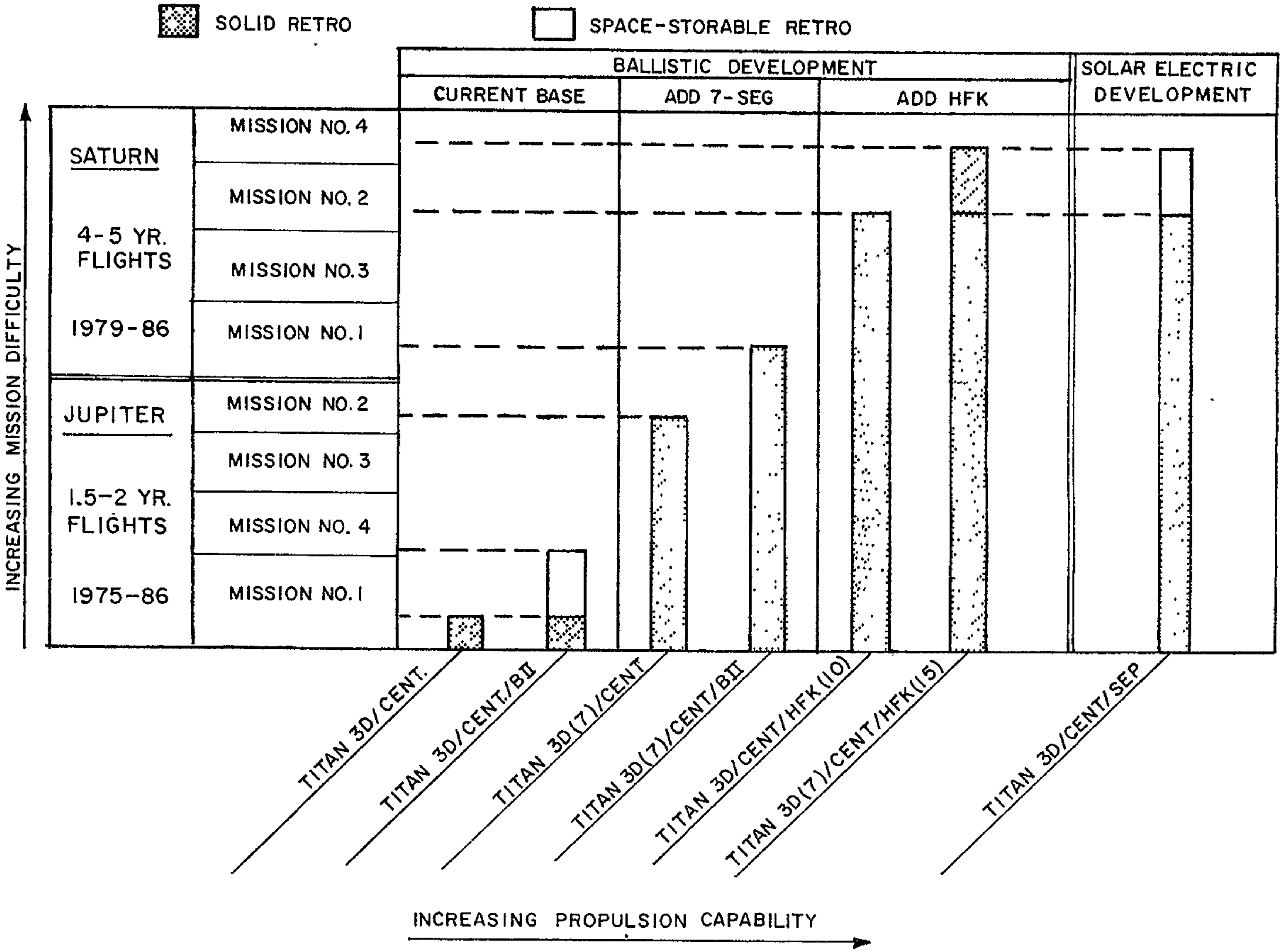
Saturn Mission No. 1 is also a particles and fields class orbiter with an added capability for performing measurements of Saturn's ring system. As tentatively envisioned, the spacecraft in the equatorial orbit would eventually be perturbed via small-step periapse reductions to enter the ring environment. Missions No's. 2 and 3 emphasize planetology and ring photometry experiments, the only difference being that the first selection has a 15-day period and a periapse exterior to the rings whereas the second selection has a 7.5-day period with a periapse between Saturn's surface and inner ring boundary. Mission No. 4 is an equatorial orbiter having a period of 15.9 days which could allow multiple observations of the satellite Titan.

A comparison summary of SEP and ballistic capabilities is shown by the bar chart in Summary Figure 1. These results apply to a launch opportunity period 1975-86 for Jupiter missions and 1979-86 for Saturn missions. Ballistic systems are conveniently subdivided to indicate the two stages of development above the current technology base (Titan 3D/Centaur/BII). The number of missions that can be performed by any specific propulsion combination is indicated by the height of the bar. Mission capability is included only if there are at least three launch opportunities in the time period of interest when that particular mission can be achieved. The range of flight times in the Summary comparison is 1.5-2 years to Jupiter and 4-5 years to Saturn. Ballistic and SEP flight times differ by less than 200 days.

The choice between the solid and space-storable retro stages should also be considered as a development option. It is assumed that once a retro stage is selected its design (and technology experience) will be retained for all orbiter missions. In reading the chart, the solid bar means that the solid retro stage is adequate to perform each of the included missions. The open bar indicates when additional capability is provided by use of the space-storable retro. It is seen that the current base ballistic systems can only perform the minimum-objective particles and fields mission at Jupiter if the solid retro is used. The space-storable retro would allow the combined planetology/particles and fields mission to be performed (i.e.,Jupiter Mission No. 4). Development of the Titan 3D (7)/Centaur/BII offers a capability envelope encompassing all of the Jupiter missions plus the minimum-

SUMMARY FIGURE I.

SUMMARY COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITY FOR JUPITER AND SATURN ORBITER MISSIONS



objective Saturn mission. The desired planetology orbiters at Saturn can be accomplished ballistically only at the cost of developing a high energy hydrogen-flourine kick stage. The Titan 3D/Centaur/HFK(10) performs Saturn Missions No. 1, 2 and 3 with the solid retro stage but does not add Mission No. 4 even with the space-storable retro. Considering the entire set of Jupiter and Saturn orbiter missions, it is seen that the Titan 3D/Centaur/SEP is essentially on a capability par with the Titan 3D/Centaur/ HFK(10). However, the most difficult Saturn orbiter mission could be accomplished with the SEP stage if the space-storable retro is employed.

The principal results of this study are thought to lie in the comparison analysis, the distillation of which is framed as a choice between future development options in solar electric propulsion as against those in chemical launch vehicles (Titan/ Centaur class) and high energy upper stages. If only Titan class vehicles are considered, then it is concluded that the addition of an SEP stage is probably the most useful improvement that can be made in extending the capability of the Titan 3D/Centaur. This conclusion factors in many other unmanned space exploration missions besides Jupiter and Saturn orbiters, e.g., Mercury orbiter, asteroid and comet rendezvous, solar probe. For most of these other missions the SEP power availability at the target yields an added bonus; this is not relevant to outer planet orbiters because SEP jettisoning prior to the capture maneuver appears to be the best operational choice.

Not considered in this study are the all-important questions of program development cost and risk. It is recommended that careful estimates be made in this area and added to the available information on mission performance comparisons. The time is at hand for making a decision between solar electric and ballistic systems for general application to a comprehensive space exploration program in the 1980 decade.

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SOLAR ELECTRIC PROPULSION FOR JUPITER AND SATURN ORBITER MISSIONS

1. INTRODUCTION

1.1 Study Background

The exploration of Jupiter and Saturn from orbiting spacecraft is currently under consideration in the context of advanced mission planning for the time period 1975-90. Orbiters will provide useful scientific data to complement and extend information obtained from early flyby and atmospheric probe mission Particularly suited to the orbit mission mode are measurements made on global atmospheric patterns, particles and fields, planet satellites, and Saturn's ring system which is unique in the Solar System. One of the main advantages of the orbiting mode is the time available for obtaining extended coverage in a dynamic planetary environment.

Two recent studies have investigated the science exploratio objectives of orbiter missions, the instrumentation necessary to perform desired measurements, and the mission concepts and requirements. The first study by IITRI(1) considers Jupiter orbite missions alone and emphasizes the areas of science objectives, measurements and instrumentation, orbit analysis, and requirements of suitable mission class selections. The second study by JPL[2] considers both Jupiter and Saturn orbiters and emphasizes conceptual spacecraft design and its relation to launch vehicle selections. Each of these studies limits the trajectory/payload analysis to the ballistic flight mode and to candidate launch systems which are derivatives of the Titan 3D/Centaur vehicle* and upper stages such as the Burner II (2300) or the conceptual Hydrogen-Flourine Kick stage. If the launch system were restricted to the currently proposed Titan 3D/Centaur/Burner II, then the ballistic spacecraft approach is adequate only for a highly elliptical orbiter of Jupiter which emphasizes particle and fields rather than planetology experiments. To achieve desired orbits at Jupiter would require the development of either Titan 3D(7) vehicle or a high energy upper stage. Both developments may be needed to achieve desired orbiters of Saturn.

The use of a solar powered electric propulsion (SEP) spacecraft has been suggested as an alternative to the ballistic delivery mode.[3] Although this approach would require development of a new technology, the SEP spacecraft offers the potential for improved performance of the Jupiter and Saturn orbiter missions and many other missions as well.[4] The SEP technology is under active development at the present time and is expected to be fully flight-proven by the mid-1970's.

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* Titan 3D is the 5-segment solid version previously designated Titan IIIX (1205).

Titan 3D(7) is the 7-segment solid version previously designated Titan IIIX (1207).

The SEP spacecraft can be considered as an additional stage above the Titan 3D/Centaur launch vehicle. The required mission velocity is attained gradually over a long period of time (about 1/4 to 1/2 the flight time) as a consequence of the low thrust acceleration and high specific impulse operation. With the possible exception of the Jupiter mission, the solar power available at the target planet may be insufficient to meet the orbital power requirements. Hence, for outer planet applications, the SEP spacecraft may be more properly viewed as an interplanetary boost stage to be jettisoned prior to planet approach. The stage delivers the combined mass of the orbiting spacecraft and the chemical retropropulsion system needed to achieve the desired orbit.

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 good orbit missions at Jupiter and Saturn, and to compare the SEP performance with the ballistic flight mode. The quality of the orbit mission is underlined to emphasize a study guideline which attempts to relate payload capability to mission requirements which are consistent with achieving desired science objectives. Such mission requirements are defined by matching science objectives with the selection of candidate orbits, science payloads, and spacecraft support subsystems. In meeting this study guideline, candidate orbiter missions are established for purposes of comparing solar electric and ballistic delivery systems.

The Jupiter orbiter missions are taken directly from the previously mentioned IITRI study.[1] In the case of Saturn, the candidate missions are established as part of the present study and are based on analysis of orbit requirements appropriate to the science objectives at Saturn -- particularly, the Ring System. However, the science payload selections are essentially the same as that of the Jupiter mission. The selected orbiter missions are described in Section 2.

In evaluating the solar electric mission capability, the propulsion system parameters are assumed to have currently demonstrable technology values. Another study groundrule is that the SEP spacecraft is launched by the Titan 3D/Centaur vehicle. Previous analyses have indicated that this currently programmed vehicle should be sufficient for both Jupiter and Saturn missions Two different chemical retrosystems are considered for tradeoff purposes; (1) a solid propellant system having a specific impulse of 300 seconds and an 11% inert fraction, and (2) a space storable liquid propellant system having a specific impulse of 400 seconds and a 25% inert fraction.

Section 3 of this report describes the optimum SEP performance for the candidate orbiter missions at Jupiter and Saturn. Curves of maximum net mass in orbit versus flight time are presented along with supporting data on optimum values of power level, specific impulse, and hyperbolic velocities at launch and arrival conditions. The effect of launch opportunity is shown as well as the performance penalty due to launch window length and off-optimum values of power and specific impulse.

Comparisons of solar electric and ballistic capabilities are presented in Section 4. The differences in planet approach conditions and retro ΔV requirements between the two flight modes are shown for a range of launch opportunities. Assuming a common SEP stage (off-optimum design) for both Jupiter and Saturn missions, the Titan 3D/Centaur/SEP mission performance is compared against a series of ballistic launch vehicle/upper stage combinations of increasing propulsion capability.

2. SELECTED ORBITER MISSIONS

In this section Jupiter and Saturn science is reviewed and the objectives of first generation orbital missions to these planets established. Based on these objectives candidate orbiter missions for Jupiter and Saturn are determined, outlined and evaluated. The science objectives for Jupiter and Saturn are based on several previous AS/IITRI Reports (5, 6), while the discussion of the Jupiter Orbiter missions draws heavily from a concurrent Astro Sciences study; "First Generation Orbiter Missions to Jupiter", AS/IITRI Report No. M-20, which will be published later this year.

2.1 Jupiter Orbiter

2.1.1 Science Objectives and Measurements

The most striking aspects of Jupiter besides its size are the visible features of its atmosphere; the dark belts, light zones and variously colored spots (see Figure 2-1). The belts and zones vary in width, intensity, and complexity as well as in color, ranging from grey to pale blues and reds. The planet's rotation based on observation of the clouds is not constant with latitude. Points within ten degrees of the equator rotate about five minutes faster than those outside this area. Varying cloud velocities have also been observed within these areas. The dynamics of Jupiter's atmosphere are graphically illustrated by the changes in cloud structure, and distribution with time. Local phenomena such

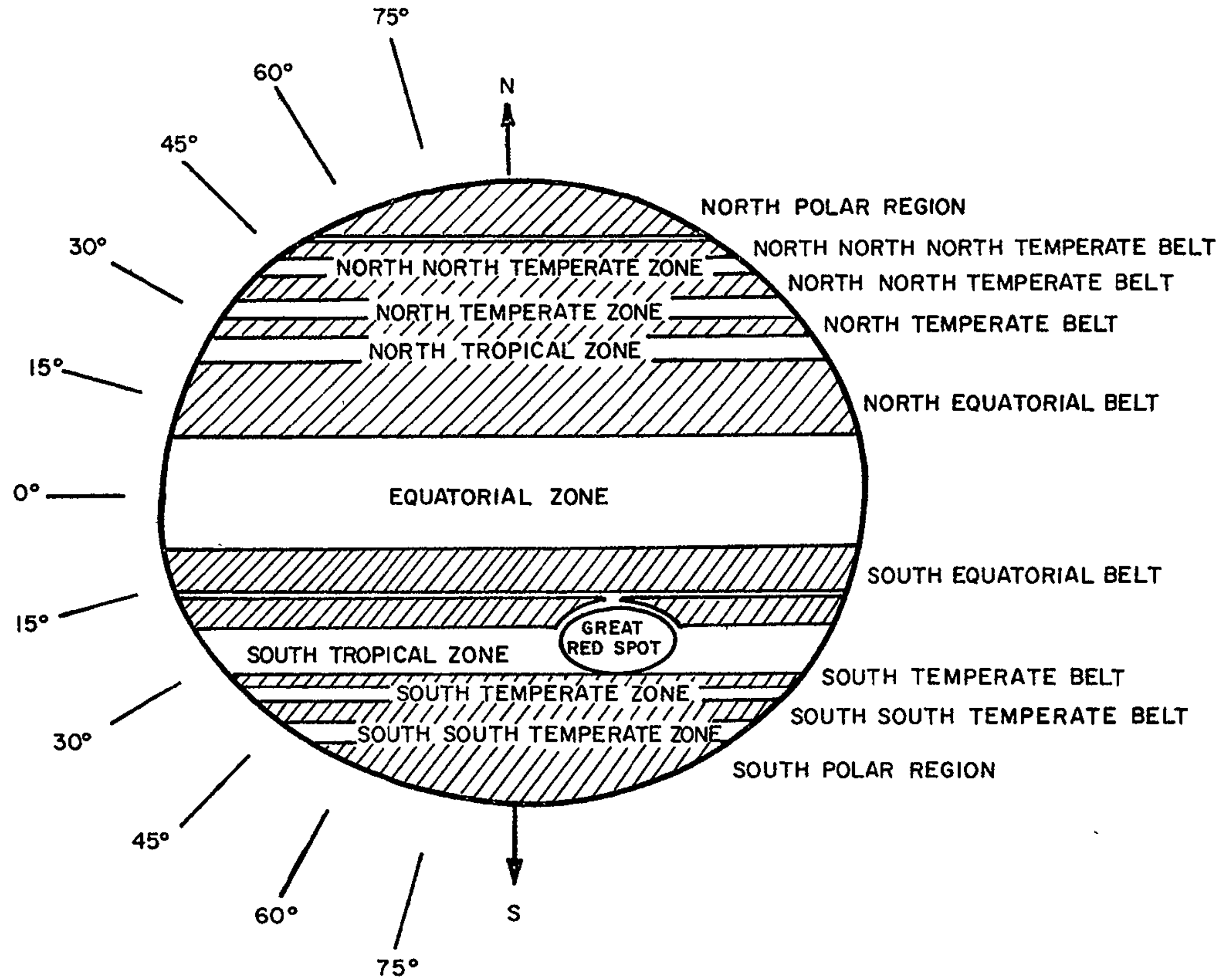


FIGURE 2-1. JOVIAN ATMOSPHERIC BELTS AND ZONES (MICHAUX, 1967)

as spots and cyclones indicate turbulence on a scale of about 1000 km.

The Great Red Spot, measuring 40,000 by 13,000 kilometers, seems to be a permanent feature, though its color does occasionally fade. Other smaller, lighter colored spots appear and disappear from time to time, but the Great Red Spot has existed for at least several hundred years. No satisfactory explanation has been found for it. It is not likely that it is fixed to a surface nor can it be a floating "island" in the atmosphere.[7] Detailed knowledge of the movement, depth and interaction of the Spot with the surrounding cloud structure is necessary to determine its nature.

The atmosphere is mostly hydrogen and helium with some ammonia and methane (perhaps two percent by mass). The clouds are most likely ammonia cirrus but the coloring agent is unknown. This coloration may be attributed directly to simple ammonia or methane ice crystals or perhaps a complex organic polymer, such as pyrene,[7] present as a trace molecule. Particulate matter is almost certainly present in the turbulent atmosphere but its nature is presently unknown.

Accurate atmospheric models for Jupiter, predicting temperature, density, and pressure profiles have not been developed mainly because of the lack of a reliable helium to hydrogen ratio and a good effective temperature. Neither of these are obtainable from earth. However, the composition of Jupiter's atmosphere is probably very similar to that of the solar photosphere, which is assumed to be representative of the primordial solar nebula (excepting D, Li, Be, and B). More importantly it probably resembles the earth's primitive reducing atmosphere where the complex chemical process began which led to terrestrial life. The pastel coloring of Jupiter's clouds may indicate that this type of polymerization is taking place. Energy sources and solvents necessary for these processes may be detectable from orbit, while the temperature, density, pressure, and moisture profiles, of the lower atmosphere may be better obtained by atmospheric probes. Detection of primitive life in the atmosphere, however, would be extremely difficult by either of these means.

Brightness temperatures over a short range of wavelengths are available from earth-based experiments and indicate that the effective temperature is higher than expected from a black body at Jupiter's distance of 5.2 AU from the sun. This indicates that Jupiter has an internal energy source. Hubbard [(9)] maintains that if the excess energy emitted above the absorbed solar radiation exceeds $\sim 10^2$ erg/cm^2, then Jupiter must be wholly convective, fluid throughout, without a solid surface. The interior is probably mostly hydrogen with a small dense core of silicates and metallic elements. A representative model of the Jovian atmosphere and interior is illustrated in Figure 2-2.

Jupiter possesses a large magnetosphere, with high energy radiation belts, shock front, and magnetopause. Observations of

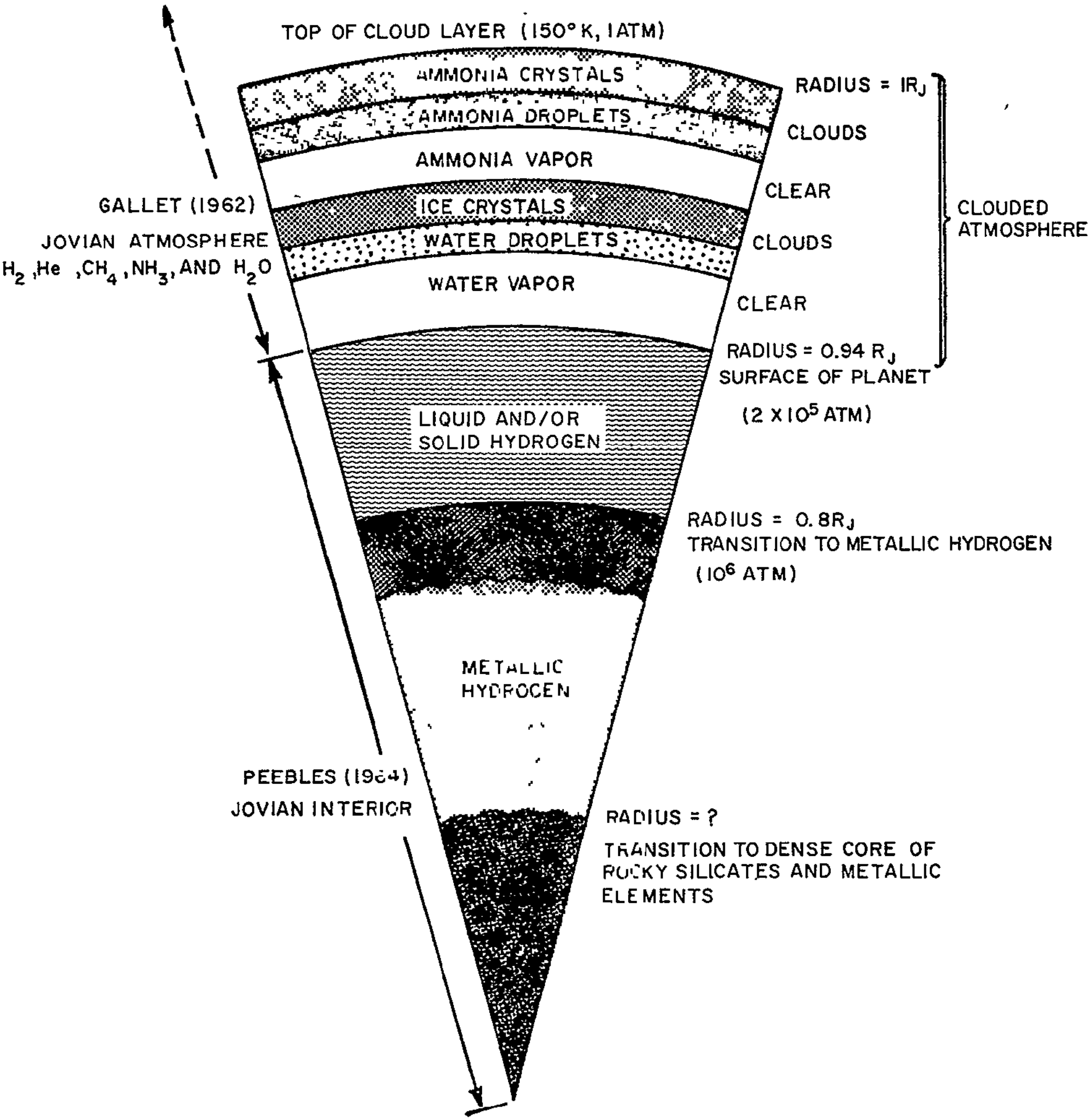


FIGURE 2-2 SCHEMATIC ILLUSTRATION OF JOVIAN ATMOSPHERE AND INTERIOR MODELS MICHAUX, (1967)

non-thermal centimeter and decimeter radiation from Jupiter indicate that it is synchrotron radiation from Van Allen type belts confined to the equatorial plane. These appear to be most intense between one and two Jupiter radii but extend perhaps to ten radii. Jupiter's magnetic field is probably dipolar with the magnetic axis inclined about ten degrees to the rotation axis, with field strengths of ten to twenty gauss at the cloud layer. The shock front is thought to be conically symmetrical about the Sun-Jupiter line with the stagnation point at about 50 to 60 radii. The weak magnetotail extends hundreds or thousands of radii out. Figure 2-3 illustrates the approximate spatial distribution of these phenomena. The area labeled "trapped Charged Particles" refers to that region in which an orbiting spacecraft's particle detectors should be functioning.

The production of the non-thermal decameter radiation (10 - 100m wavelength) emitted by Jupiter is not understood. It appears to originate in or on Jupiter but is coupled to the orbital motion of one of the Galilean satellites, Io. It also appears to be emitted directionally but the geometry is unknown. Complete, accurate mapping of the radiation source cannot be done with earth-based instruments because of the limited coverage available. (The radiation must be directed within several degrees of the ecliptic plane for it to be received by earth).

There are twelve detected natural satellites orbiting Jupiter; five regular ones close to the planet and seven irregular

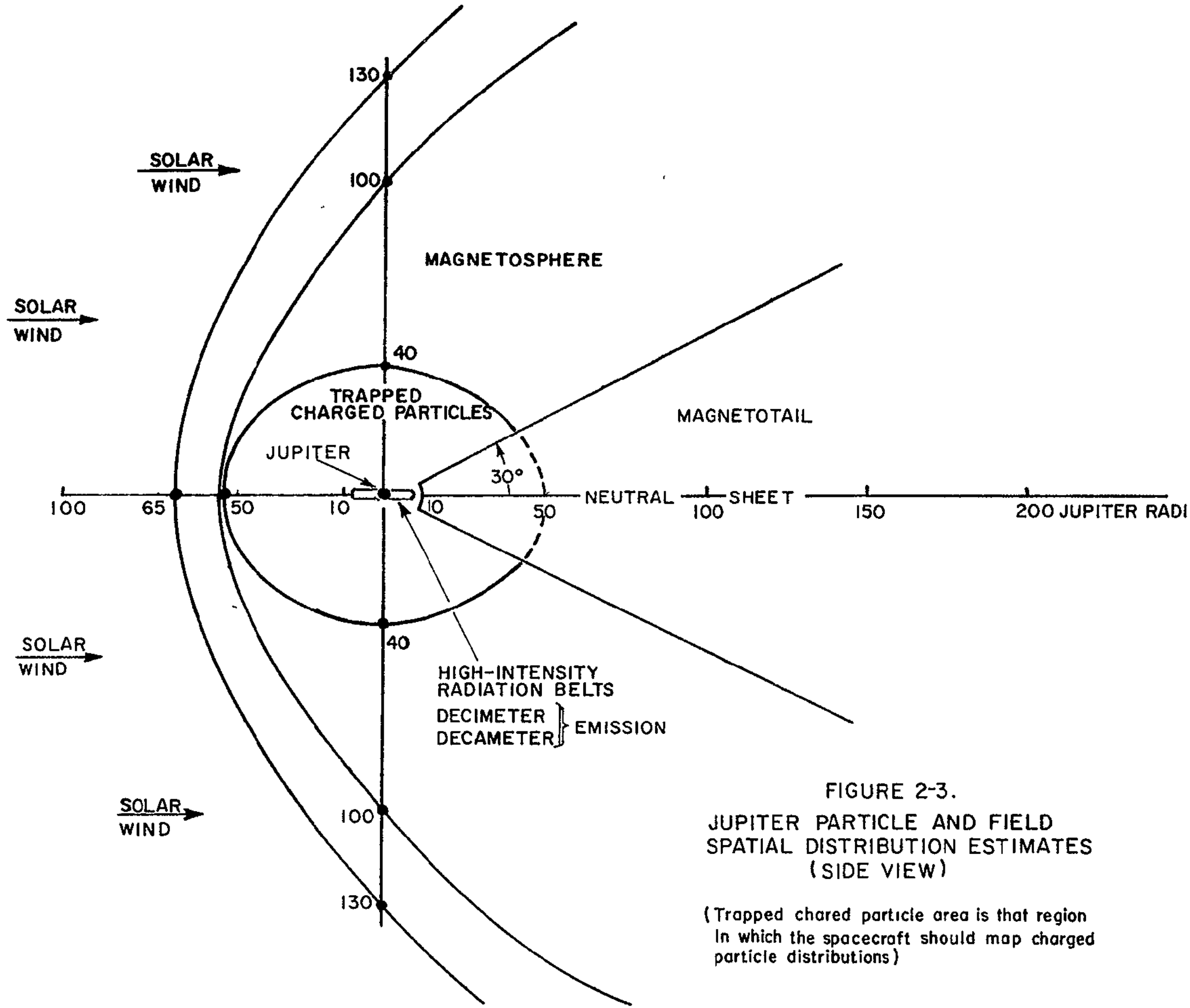


FIGURE 2-3.

JUPITER PARTICLE AND FIELD SPATIAL DISTRIBUTION ESTIMATES (SIDE VIEW)

(Trapped chared particle area is that region In which the spacecraft should map charged particle distributions) ones further out. The regular satellites in direct equatorial orbits are the four Galilean satellites and Amalthea or Jupiter V. Amalthea is small and little is known of it. Io, Europa, Ganymede, and Callisto (Jupiter I, II, III, and IV respectively) are large (0.64 to 2.1 the size of the Moon) and represent important planetary bodies. There are two groups of irregular satellites; four in direct orbits outside the inner regular group and three in retrograde orbits outside these four. They are all small bodies in highly elliptical orbits with inclinations ranging between 26° and 123°.

Figure 2-4 indicates the instruments useful in fulfilling the scientific objectives mentioned in this section. For this figure the goal of total Jupiter exploration was first broken down into regimes closely aligned to the various natural characteristics of the planet, e.g., interior, atmosphere, surrounding magnetic field, and biology. Each regime has specific categories of interest. In the case of the atmosphere these regime categories are composition, dynamics, and structure. Each category has then been refined into objectives. Determining atmospheric composition, for example, really means identifying the objectives; 1) elemental and molecular abundances, 2) isotopic abundances and ratios and 3) particulate matter. Each objective is then broken down into measurables, i.e., the physical evidence of the objective. Figure 2-4 includes only those objectives whose physical measurables are within the capability of first generation Jupiter orbiters. The open boxes indicate a duplication of a measurement.

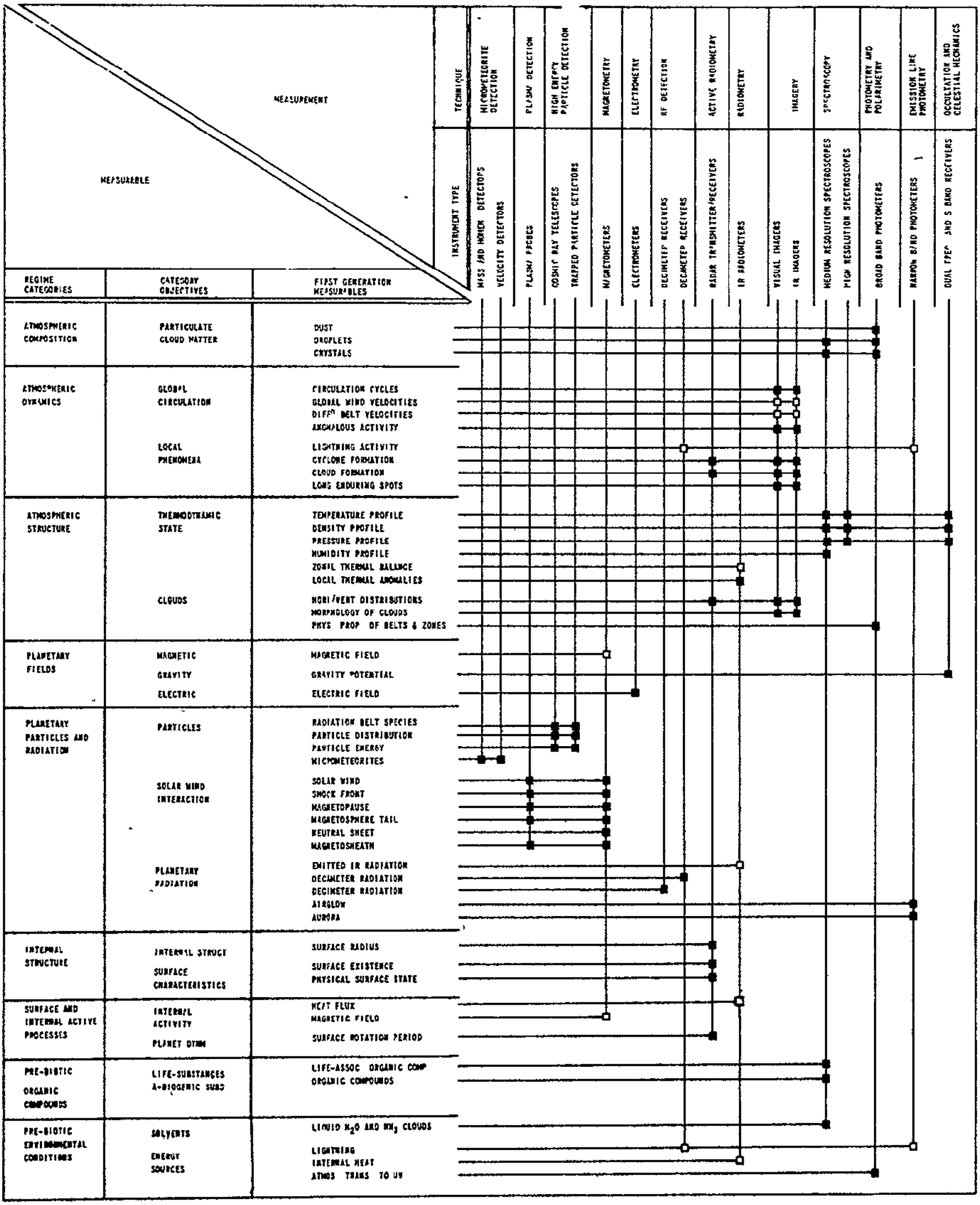


FIGURE 2-4. MEASUREMENT TECHNIQUE AND INSTRUMENT SELECTIONS

2.1.2 Candidate Missions

From the previous section it can be seen that quite an extensive exploration of Jupiter must be done. No single orbiter mission can be designed to sufficiently cover all the necessary science areas, based on what is currently known or assumed to be true for Jupiter. As data from early flybys refine Jupiter science, more efficient orbiter missions may be designed. Breaking Jupiter science areas into planetology, satellite observation, and particle and fields studies, four Jupiter Orbiter Missions were tentatively chosen. They are:

- Mission No. 1 - A particle and fields mission,
- Mission No. 2 - A planetology mission,
- Mission No. 3 - A combined planetology and satellite observation mission,
- Mission No. 4 - A combined planetology and particle and fields mission.

Before describing these four missions more thoroughly it is best to examine some of the reasoning and constraints behind our choice.

Radiation damage of spacecraft components by the energetic particles in the Jovian magnetosphere can greatly reduce spacecraft lifetimes, and provides a major consideration in orbit selection. A previous study has shown that most of the radiation damage is probably done by electrons if a shielding thickness of 1 gm/cm^2 of aluminum is assumed (only a few very high energy protons will be able to penetrate this shielding). Taking the radiation damage threshold as $\sim 10^7$ rads, spacecraft lifetimes in fixed periapse elliptical equatorial orbits are approximately constant for orbital periods greater than 7 days. (1) If 100 revolutions is acceptable as a minimum lifetime, then the orbit periapse must be greater than 2.5 Jupiter radii. Using this criterion the periapse altitude for our reference missions was chosen to be 3 Jupiter radii (except Mission No. 3) which is 2.29 radii to accommodate the specific satellite observation orbit.

Figure 2-5 shows four of the orbit possibilities and lifetimes available to a Jupiter orbiter mission. The position of the solar terminator during the first orbit for the limiting inclinations 0° and 90° is also shown for a representative launch year and arrival date. The terminator's position for the equatorial orbit and this arrival date is preferred over that of the polar orbit because it allows altitude limited instruments an excellent opportunity for covering the sunlit side of Jupiter. The position of the terminator is fortunately about the same for all orbits with inclinations less than 60°, allowing advantageous lighting to be combined with high planet latitude coverage. The terminator's position does vary from orbit to orbit depending on Jupiter's heliocentric position, and the spacecraft's orbital precession.

Figure 2.6 illustrates the relationship between spacecraft altitude and time in Jupiter rotations or days for the four orbits

ORBIT PERIOD, DAYS	RADIATION LIFETIME	
	REVS	DAYS
7.5	425	3188
15	450	6750
30	470	14100
45	475	21375

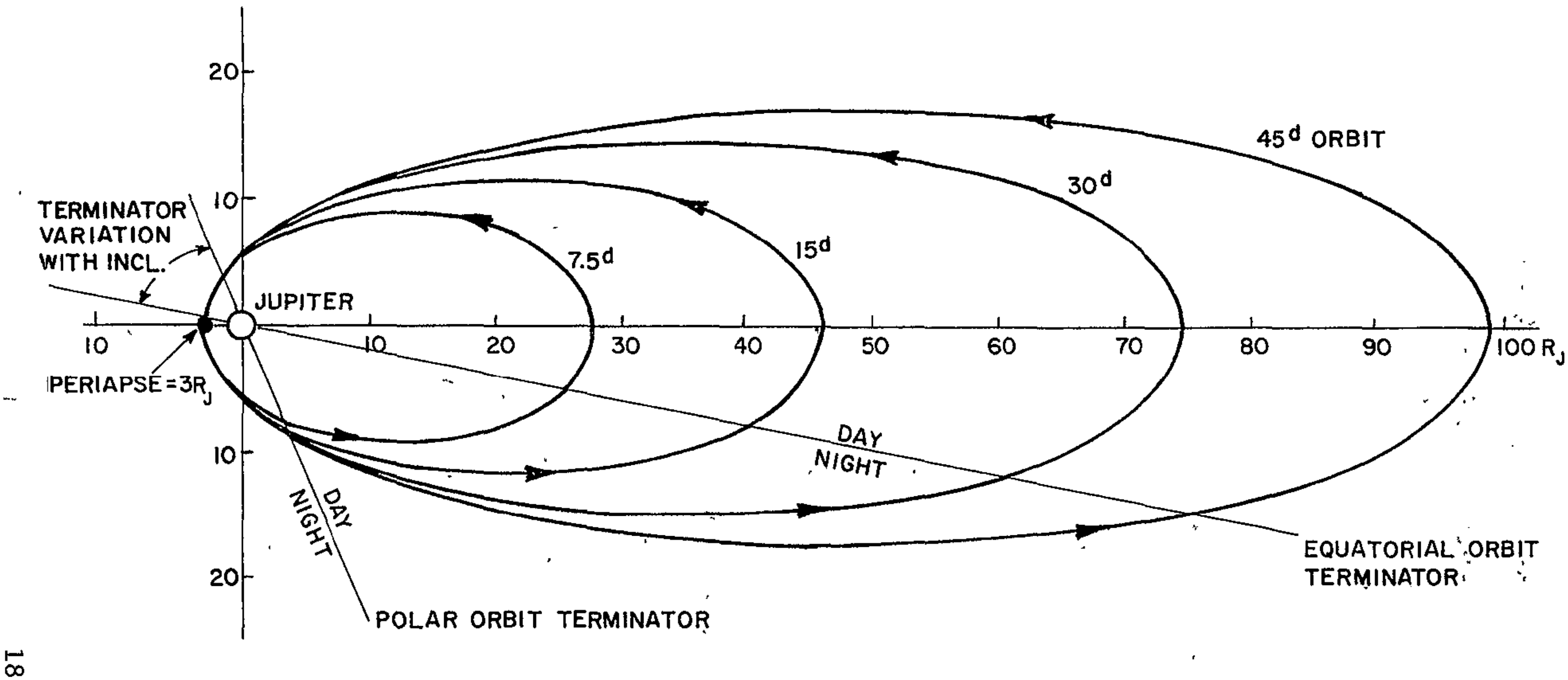


FIGURE 2-5. REPRESENTATIVE JUPITER ORBITS

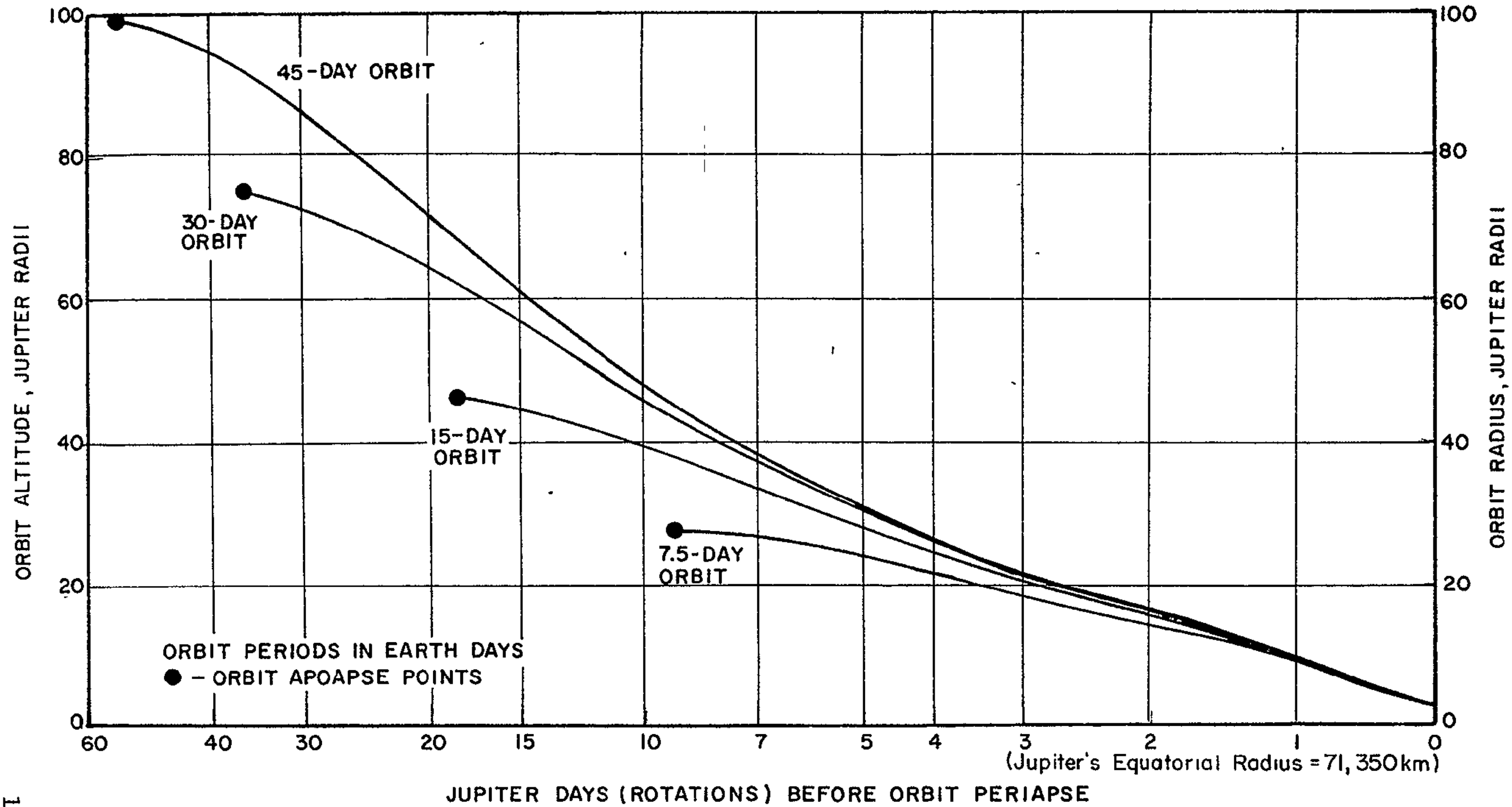


FIGURE 2-6. JUPITER ORBIT ALTITUDE VARIATIONS.

mentioned above. While the 45-day orbit requires 110 Jupiter days for one orbit and the 7.5-day orbit only 18, both spend about the same amount of time between 20 Jupiter radii and periapse. Only two Jupiter days per orbit, regardless of period, are spent within 10 Jupiter radii of the planet.

The subspacecraft ground trace, the relationship between the latitude of the suborbit point and time, is shown in Figure 2-7 for the four orbits at 60° inclination. Once again the four orbits are nearly coincident near periapse. By comparing the highly inclined orbits' ground coverage to that of the equatorial orbits, which is a straight line along 0° latitude (the equator), the great advantage the former offers in coverage for planetology instruments can be seen. The obvious advantage offered by the shorter period orbit is a greater number of low altitude passes during a given time period.

Small, short period orbits, then, offer the best advantages for a planetology oriented orbiter. The constantly varying atmosphere is best viewed and studied from such an orbit. A short period orbiter also spends more of its period within the region near Jupiter where a high concentration of charged particles is expected, and provides more RF occultation measurements of Jupiter's atmosphere than do the longer period orbiters with equal inclinations.

Niehoff [10] has recently shown that a large number of encounters with the Galilean satellites are possible with a specific

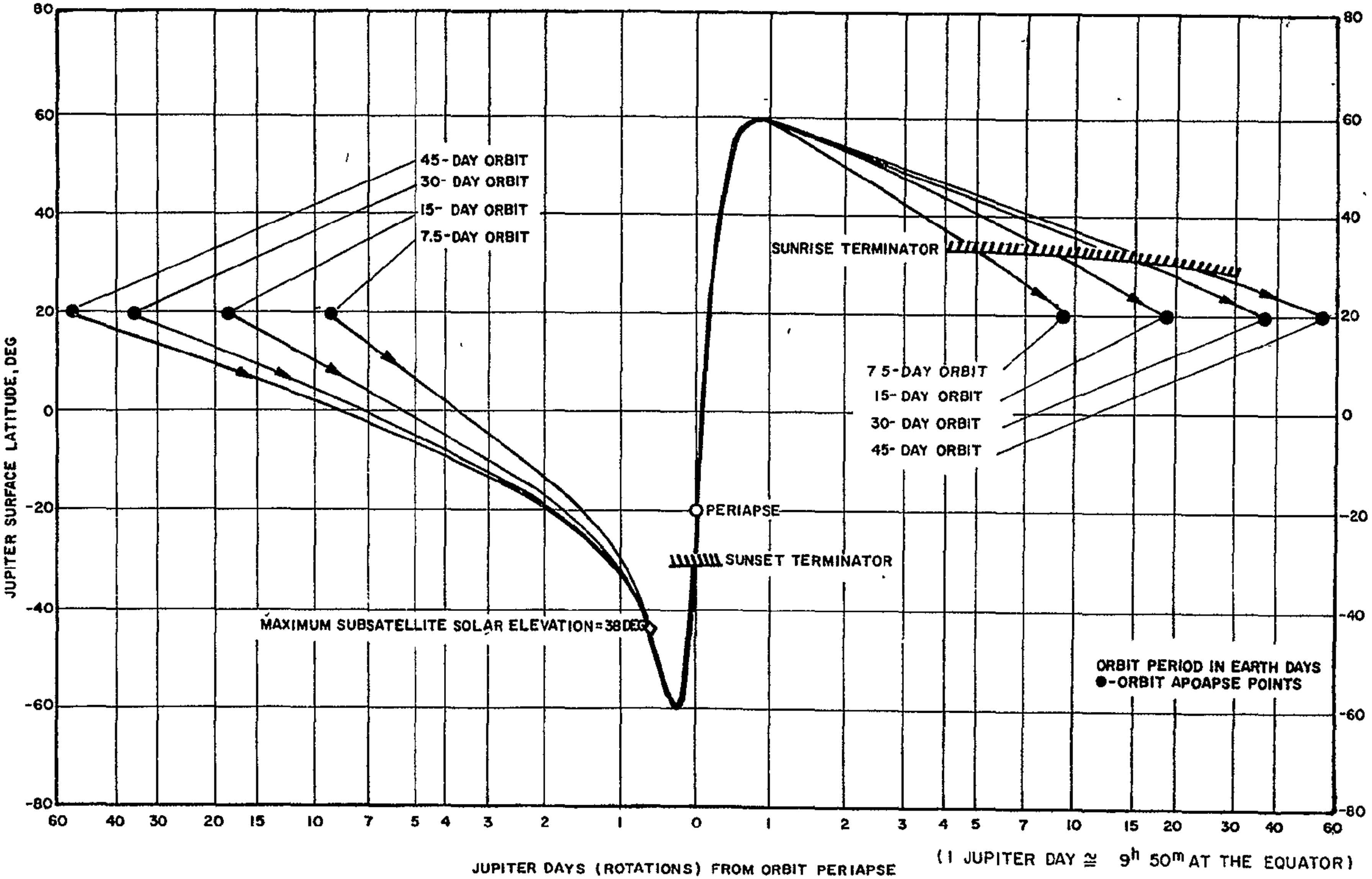


FIGURE 2-7. JUPITER ORBIT LATITUDE VARIATIONS (ORBIT INCLINATION = 60 DEG)

(14.2-day) short period equatorial orbit. A spacecraft correctly placed in this orbit may repeatedly view Jupiter and its four largest satellites.

Long period orbiters, on the other hand, have longer spacecraft radiation lifetimes and are better suited to particle and field experiments. Measurements of the shock front, magnetosheath, magnetopause, and magnetosphere are better done by the long period, large apoapse orbiter.

Based on this reasoning, the four candidate missions were selected. They are listed in Table 2-1.

Mission No. 1 is principally designed to investigate the particle and fields phenomena associated with Jupiter. Two spacecraft, one in a 45-day equatorial orbit, the other in a 45-day, 120° inclined orbit, are used to explore the vast Jovian atmosphere. The science payload, based on the Pioneer F&G payload, is listed in Table 2-2. The LEPEDEA experiment has been added to bridge the gap between detecting the solar wind and the particles in the Jovian magnetosphere. The RF receiver serves both to monitor Jovian decameter radiation and to lend its antenna to the electric field instrument.

The spacecraft subsystem weight breakdown is shown in Figure 2-8. These figures are also based on Pioneer F&G technology. The TRW Pioneer F&G [11] data storage and management, and the X band communication systems are used (9' antenna, 8 watts, X band at 8 kbps)

TABLE 2-1
JUPITER ORBITER MISSION SELECTIONS

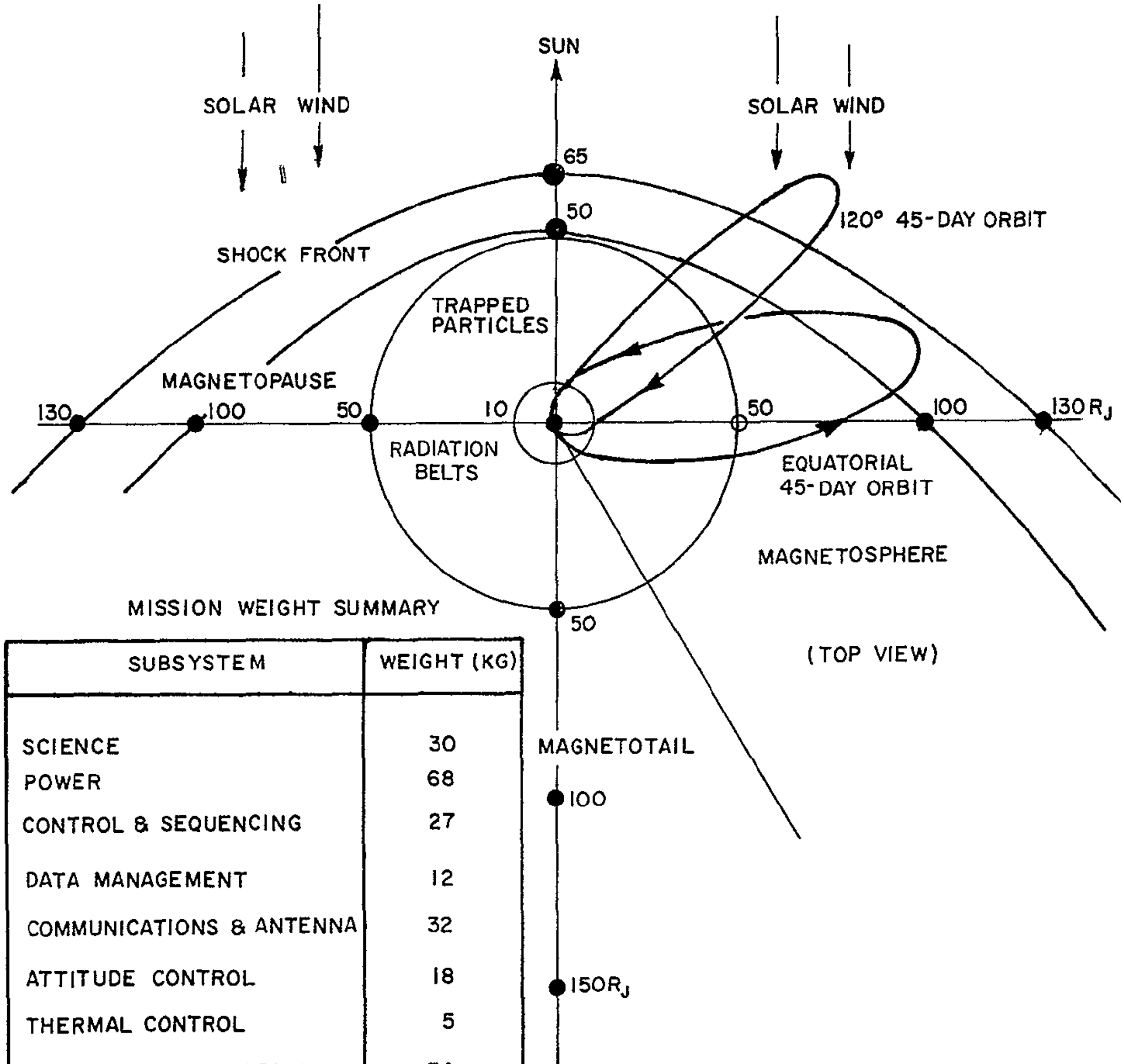
MISSION NUMBER	SCIENCE OBJECTIVE	ORBIT PARAMETERS		
		PERIAPSE (R_J)	PERIOD (DAYS)	INCLINATION (DEG)
1	PARTICLES AND FIELDS[1]	3	45	0 AND 120[1]
2	PLANETOLOGY	3	15	60
3	PLANETOLOGY AND SATELLITE OBSERVATIONS	2.290	14.222	0
4	PARTICLE AND FIELD AND PLANETOLOGY	3	30	60

1. TWO SPACECRAFT ORBITED AT DIFFERENT INCLINATIONS

TABLE 2-2. PAYLOAD DEFINITION, JUPITER MISSION NO. I – PARTICLES AND FIELDS

INSTRUMENT	RANGE	DATA RATE (BPS)	WEIGHT (KG)	POWER (WATTS)
VECTOR HELIUM MAGNETOMETER	0.1γ - 10G	15	4.6	2.5
TWO ELECTROSTATIC PLASMA ANALYZERS	E: 1-500 eV, P: 0.1-8KeV	50	4.3	4
LEPEDEA	E: 100KeV - 80MeV		1.6	2
SOLID STATE PARTICLE DETECTOR AND DOSIMETER	E: 0.5-20MeV, P: >50MeV 10 - 100 MeV/nuc.	50	2.7	2
GEIGER TUBE TELESCOPE	E: 2-5 Mev		1.1	1
TRAPPED RADIATION INSTRUMENTS	E: 0.05 - 50 MeV P: 0.3 - 500 MeV	50	2.5	3
CERENKOV DETECTOR				
ELECTRON SCATTER DETECTOR				
DC SCINTILLATOR				
HIGH ENERGY PROTON DETECTOR				
SWEPT FREQUENCY RF RECEIVER	300 KHz - 300 GHz	15	3.6	3
IMP-1 ELECTRIC FIELD DETECTOR[1]	0 - 300 Hz	15	1.8	1.5
MICROMETEORITE DETECTORS		2	7.8	2
ASTEROID/METEOROID ASTRONOMY	Size: >50µm,			
PRESSURE CELL DETECTORS	Vel.: 0.01 - 70km/sec. $>10^{-9}$ gm			
TOTALS			30	21

1. USES RF RECEIVER ANTENNAE



MISSION WEIGHT SUMMARY

SUBSYSTEM	WEIGHT (KG)
SCIENCE	30
POWER	68
CONTROL & SEQUENCING	27
DATA MANAGEMENT	12
COMMUNICATIONS & ANTENNA	32
ATTITUDE CONTROL	18
THERMAL CONTROL	5
STRUCTURE & SHIELDING	34
SUBTOTAL	226
TWO SPACECRAFT + 5% MOUNTING STRUCTURE (Excluding Propulsion)	475

MISSION NO. 1 SUMMARY WEIGHT BREAKDOWN AND ORBIT CONFIGURATIONS

FIGURE 2-8.

which requires a two hour data transmission period using the 210' Goldstone receiving disk every two days for each spacecraft.

The two orbits selected for Mission No. 1 are also shown in Figure 2-8. These orbits will rotate clockwise as the mission progresses. Unfortunately the equatorial orbit does not pass through the expected shockfront and the 120° inclined orbit intersects it only for a short time. The mission should, however, provide good data on Jupiter magnetic and electric fields, near-planet high energy particle environment and meteorite distribution, and the effects of the solar wind at Jupiter. By tracking its fairly loose orbit some clarification of Jupiter's gravitational field may be obtained. The mission will provide no information on the planet itself apart from monitoring the non-thermal RF radiation and occasional atmospheric discharges.

Mission No. 2

Table 2-3 lists the science payloads for Mission No. 2. Mission No. 2 is specifically a planetology mission having a 15-day, 60° inclined orbit to optimize coverage opportunities for the planetology instruments. The three non-planetology instruments; the micrometeroid detector, the x-ray imager, and ionosonde have been added to study near Jupiter micrometeoroids, the interaction of high energy particles with the atmosphere, and ionospheric electron densities respectively.

Figure 2-9 illustrates the portions of the 15-day orbit

TABLE 2-3. PAYLOAD DEFINITIONS JUPITER ORBITER MISSION NOS. 2, 3 and 4

INSTRUMENTS	WEIGHT (kg)			RANGE	ALTITUDE RANGE JUPITER RADII	MAXIMUM DATA RATE	POWER (WATTS)
	MISSION #2 PLANETOLOGY	MISSION #3 PLANETOLOGY AND SATELLITE OBSERVATION	MISSION #4 PARTICLE AND FIELDS AND PLANETOLOGY				
1.5 INCH RETURN BEAM VIDICON[1]	18	18	18	Resolution: 3.5-100Km	2-50	100 KBPS	26
NEAR-IR LINE SCANNER[1]	13	13	13	Resolution: 34-100Km	2-8	2 KBPS	5
3-CHANNEL IR RADIOMETER[1]	13[2]	13[2]	13[2]	Resolution. 95-200Km	2-5	700 BPS	5
NARROW BAND UV PHOTOMETER/SPECTROMETER[1]	7	7	7	Resolution 40-100Km	2-5	100 BPS	6
X-RAY IMAGER[1]	3.6		3.6	——	2-45	1000 BPS	8
IONOSONDE	11	11	11	——		1 BPS	10
SWEPT FREQUENCY RF RECEIVER	3.6	3.6	3.6	300KHz - 300GHz		15 BPS	3
DC ELECTRIC FIELD DETECTOR (USE RF ANTENNA)			1.8	0 - 300 Hz		15 BPS	1.5
LEPEDEA			1.6	E: 100KeV - 80 MeV		15 BPS	2
LITHIUM-DRIFTED SOLID STATE DETECTOR			2.7	E: 0.5-20 MeV, P>50MeV			2
(AND DOSIMETER)				10 - 100 MeV/Nuc.			
GEIGER TUBE TELESCOPE		1.1	1.1	E: 2-5 MeV		-	1
TRAPPED RADIATION DETECTORS		2.5	2.5	E: 0.05-50 MeV P: 0.3-500 MeV		50 BPS	3
VECTOR HELIUM MAGNETOMETER		4.6	4.6	0.18 - 10G		15 BPS	2.5
MICROMETEOROID DETECTORS	7.8	7.8	7.8	Size: > 50μm, Velocity: 0.01-70Km/sec > 10^{-9} gm		2 BPS	2
RF OCCULTATION	NO	YES	NO				
TOTALS	77.0	81.6	91.3				

1. THESE PLANETOLOGY INSTRUMENTS HAVE LATERAL SCANNING CAPABILITY
2. REQUIRES AN ADDITIONAL 23 Kg. SOLID METHANE COOLING SYSTEM

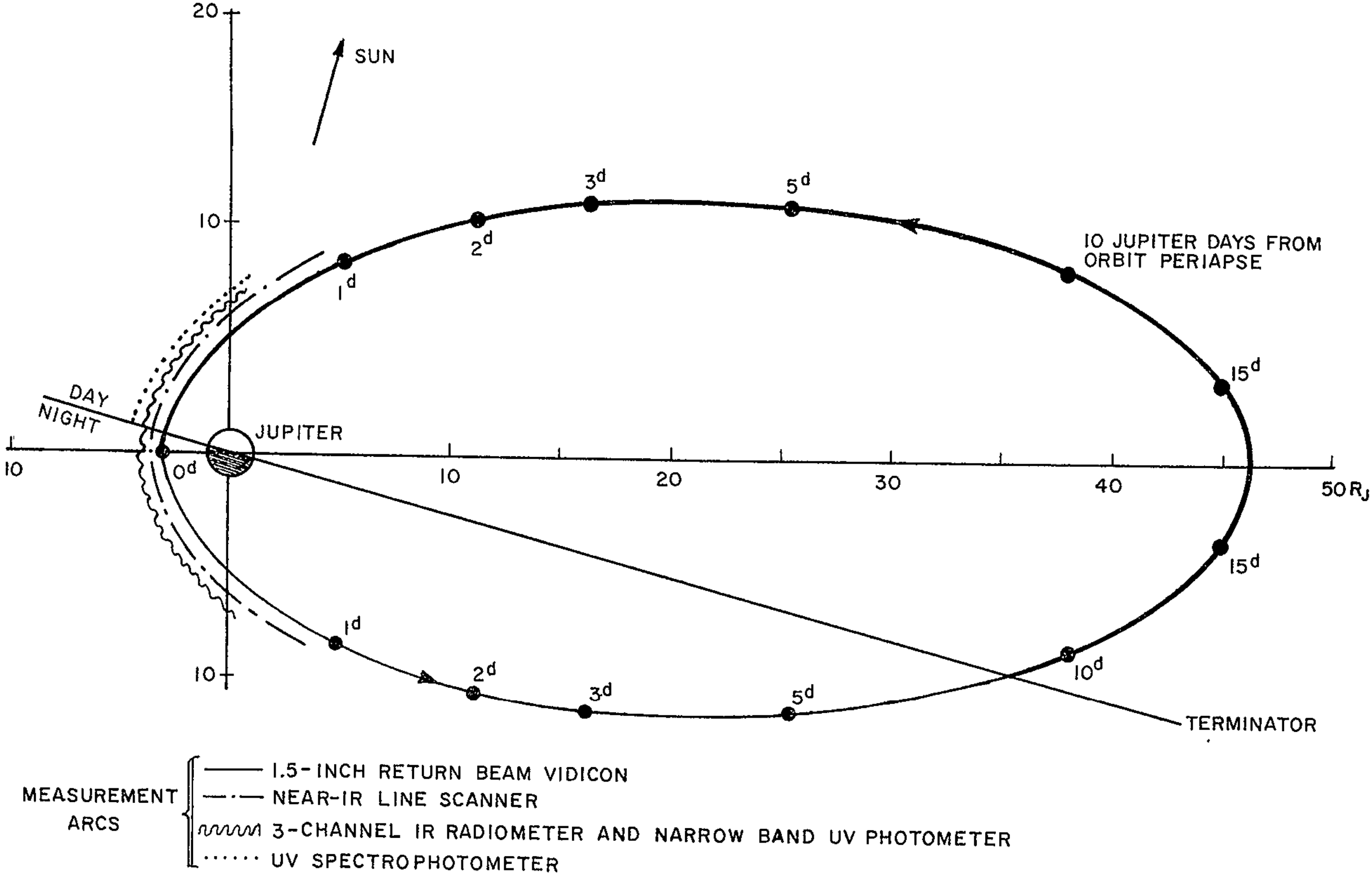


FIGURE 2-9. PLANETOLOGY INSTRUMENT MEASUREMENT ARCS FOR 15-DAY (EARTH DAYS) JUPITER ORBIT. MISSION NO. 2.

over which the planetology instruments function with better than 100 km. resolution. This restricts all the instruments to operation within two Jupiter days of periapse, emphasizing the need for frequent periapse passages (short period orbit). The 15-day orbit was chosen over the 7.5-day orbit because of its smaller ΔV requirement for orbit insertion from an interplanetary trajectory.

Figure 2-10 shows a comparison between the surface coverage capability of the 1.5 inch return beam vidicon in a 15-day equatorial orbit and a 15-day, 60° inclined orbit. The shaded regions show the coverage of a one degree field of view instrument taking one picture on each side of the ground trace (a total field of view of 2°). The dotted curves represent the latitude coverage if lateral scanning up to 200 kilometers resolution is permitted. The 60° inclination allows both poles to be viewed but even the equatorial orbit allows viewing of 85 percent of the planet.

Mission No. 2 is able to provide a good deal of information on Jupiter's appearance in the ultraviolet, visual, and infrared spectral regions. The fairly high resolution spectroscopy needed to determine atmospheric composition is not included in the Mission No. 2 payload. Since it must rely on photometers instead its performance is limited. A serious deficiency of this mission is that it does not allow an RF occultation of Jupiter's atmosphere to be performed which would provide data on atmospheric composition and its thermodynamic state. The 60° inclination and the orbit's orientation relative to the earth are responsible for this. Earth

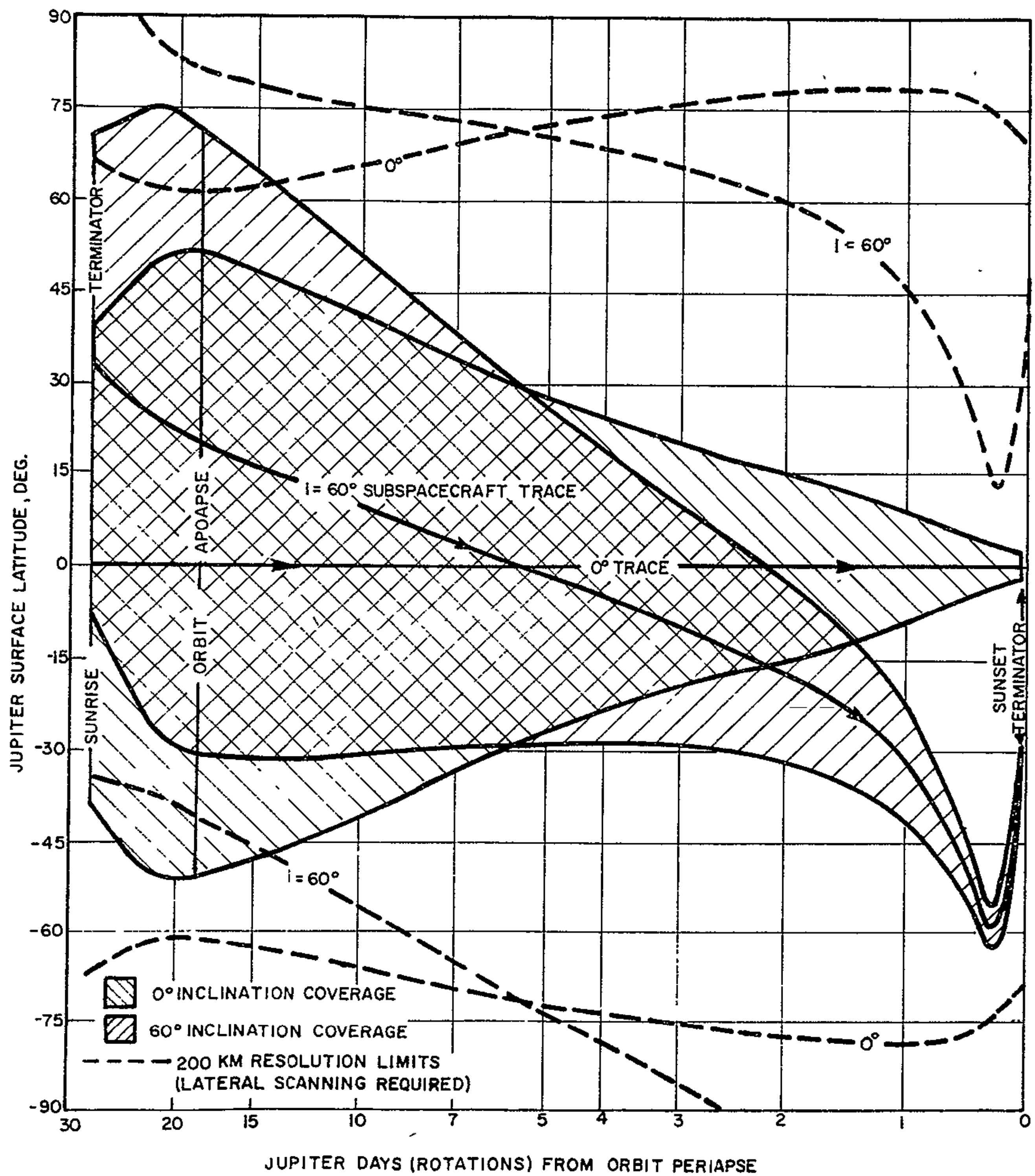


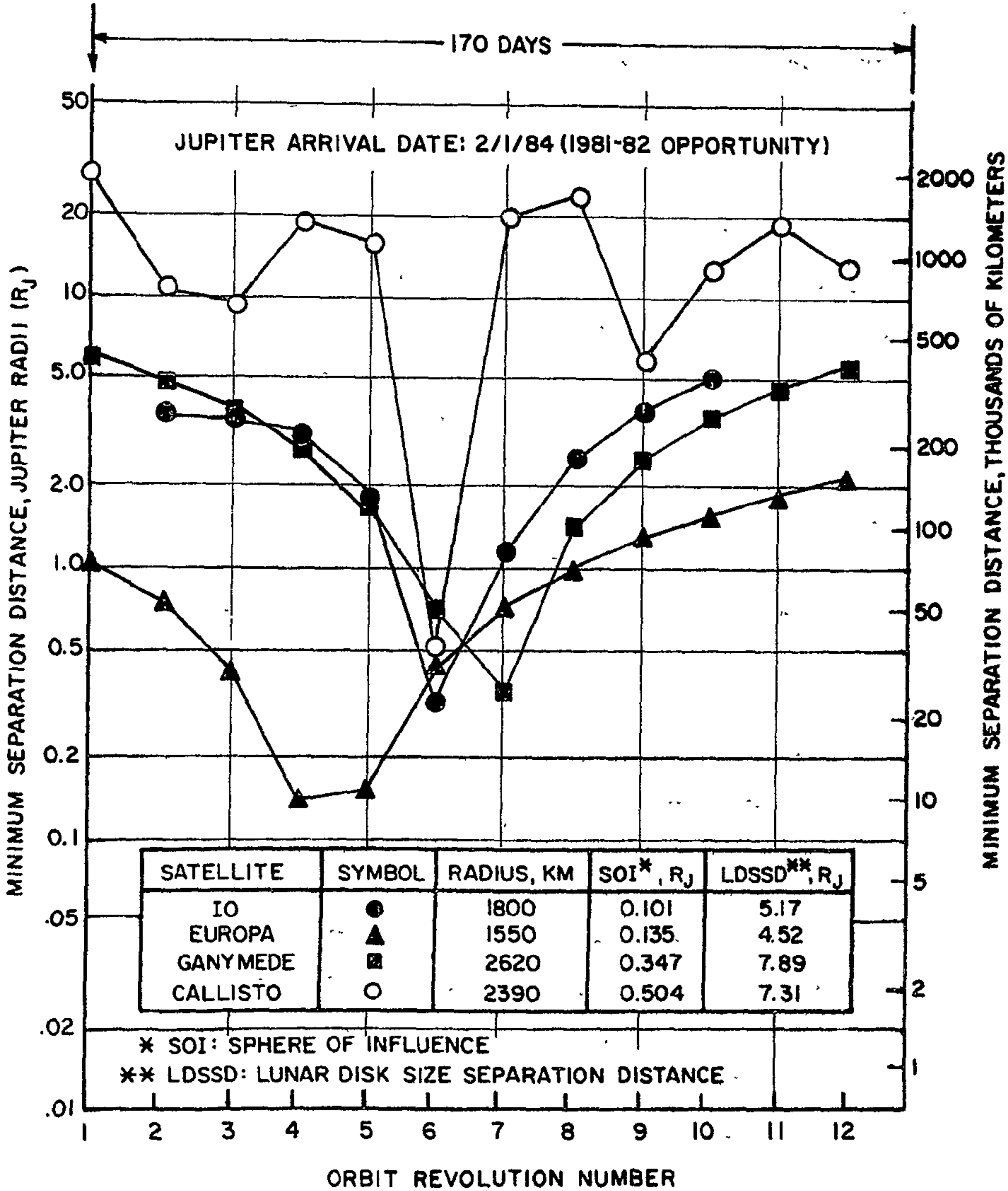
FIGURE 2-10. COMPARISON OF VISUAL COVERAGE (FOV=2°) FOR 15-DAY ORBIT AT 0° AND 60° INCLINATIONS

based tracking of the spacecraft throughout its orbit will provide information on Jupiter's gravitational field, as it does for the other three missions.

Mission No. 3

Mission No. 3 is an unusual planetology orbiter in that it is able to study not only Jupiter but also the four Galilean satellites. The 14.2-day, equatorial orbit with a periapse of 2.29 Jupiter radii provides less latitude coverage of Jupiter than Mission No. 2, but compensates by repeatedly allowing close observations of the four satellites. This is a result of the commensurability of the motions of the first three satellites; Io, Europa, and Ganymede. Their orbital periods are very close to the ratio, 1:2:4. Figure 2-11 illustrates the satellite encounters possible for the 1981 ballistic launch opportunity. Notice that several close encounters with Callisto are possible, even though Callisto does not obey the relationship between orbital periods mentioned above. Figure 2-12 shows the spacecraft's encounter path with Ganymede for four consecutive orbits (listed in Figure 2-11). Since the spacecraft's approach is always from the general direction of the sun, imagery of the satellites is good. The perturbations on the spacecraft's orbit by the satellites have not been taken into account for these figures. However, Niehoff [10] has shown that the perturbations can be corrected by an orbit control scheme (with small impulses) without appreciably effecting the satellite encounter profiles. The science payload for Mission No. 3 is shown in Table 2-3 and is basically the same as that for Mission No. 2. A RF receiver and several particle and

(TIME OF PERIAPSE FOR THE FIRST ORBIT SHOWN IS FEBRUARY 21, 1984 (-244575.918), 20 DAYS AFTER ARRIVAL)



SATELLITE	SYMBOL	RADIUS, KM	SOI*, R_J	LDSSD**, R_J
IO	●	1800	0.101	5.17
EUROPA	▲	1550	0.135	4.52
GANYMEDE	■	2620	0.347	7.89
CALLISTO	○	2390	0.504	7.31

FIGURE 2-11. GALILEAN SATELLITE ENCOUNTERS, MISSION NO.3 (14.2 DAY ORBIT)

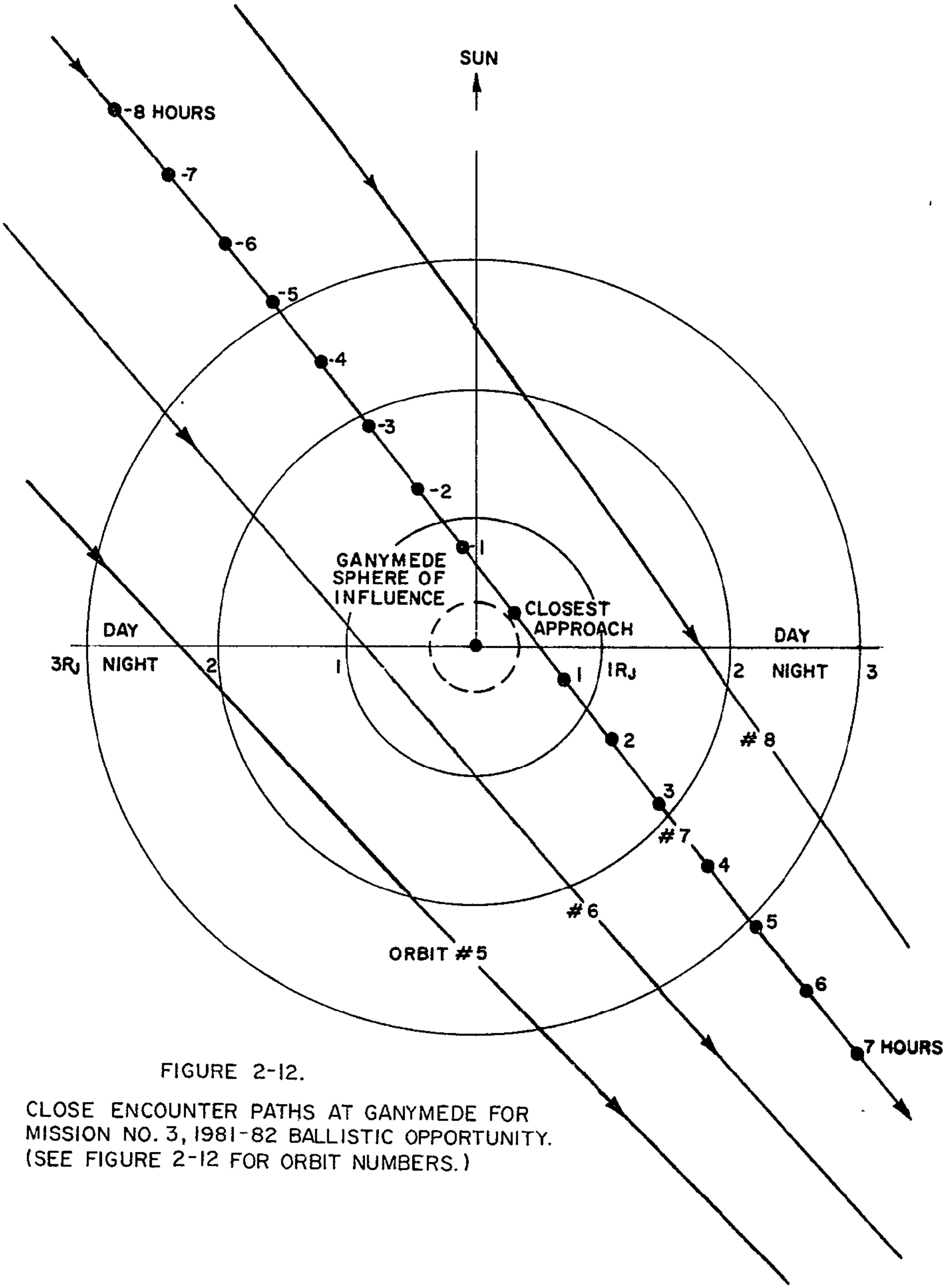


FIGURE 2-12.

CLOSE ENCOUNTER PATHS AT GANYMEDE FOR MISSION NO. 3, 1981-82 BALLISTIC OPPORTUNITY. (SEE FIGURE 2-12 FOR ORBIT NUMBERS.)

field (P/F) instruments have been substituted for the x-ray imager. The x-ray imager, designed to study polar region aurora, has a low relative usefulness in an equatorial orbit. The P/F instruments were added to detect and measure the disturbing influences of the Galilean satellites as they pass through the Jovian magnetosphere. The RF receiver is used to monitor decametric radiation.

This mission is unable to provide imagery of 100 percent of the planet because of its equatorial orbit, but its low (2.29 R_j) periapse distance allows the cloud tops to be imaged at a somewhat finer resolution than available on Mission No. 2 (planetology, periapse - 3 R_j). RF occultations are possible on this mission, allowing a more thorough study of Jupiter's atmosphere than provided by Mission No. 2. The P/F instruments also provide data on the near Jupiter particle and fields environment in addition to information on the Galilean satellites' interaction with the magnetosphere.

Mission No. 4

Mission No. 4 is a composite mission designed for both planetology and particle and field investigations. Its 30-day period and large apoapse allow it to probe the magnetosphere while its 60° inclination provides adequate ground coverage. This orbit does not provide repretitive observations of the Galilean satellites, or a RF occultation of Jupiter's atmosphere, nor does it cross the shock front. The payload is shown in Table 2-3. The solar wind plasma detectors were omitted from the science payload because it does not pass outside the magnetosphere. It does, however, carry all the other instruments that Mission No's. 1, 2 and 3 carry.

The subsystem weight breakdown for Mission No's. 2, 3 and 4 are listed in Table 2-4. These weights are based on a survey of Jupiter orbiter designs. The communications, data storage and management systems are based on the JPL TOPS design, [12] and a data acquisition rate of 10^{10} - 10^{11} bits per orbit. With a storage capability of 2 x 10^9 bits each spacecraft must transmit its collected data (X band, 130 kbps) for about 3 hours every third day (210' disk receiving). The telemetry systems on all three missions are the same because of the common use of the TV imager. The large data acquisition rate of this instrument is the controlli influence on the communication system design. The total power requirement is about 400 watts for each mission.

Table 2-5 gives a subjective indication of the performance of the candidate Jupiter Orbiter missions. Each science objective is fulfilled to a certain degree by each mission. Specific shortcomings of each mission were pointed out in the previous discussion.

2.2 Saturn Orbiter

2.2.1 Science Objectives and Measurements

Saturn is a large, low density, gaseous planet similar in many respects to Jupiter. Many of Saturn's features can be studied in much the same fashion as Jupiter's with similar instruments Much of the previous discussion on the scientific objectives for Jupiter exploratory missions may be applied to Saturn missions, remembering that Saturn is similar but not identical to Jupiter.

TABLE 2-4
SPACECRAFT WEIGHT BREAKDOWN
JUPITER MISSION NOS. 2,3, AND 4-PLANETOLOGY

SUBSYSTEM	MISSION NO.2 PLANETOLOGY	MISSION NO.3 PLANETOLOGY AND SATELLITE OBSERV.	MISSION NO. 4 PARTICLE AND FIELDS AND PLANETOLOGY
SCIENCE	77 KG	82 KG	91 KG
PLANETOLOGY INSTRUMENT PLATFORM	16	16	16
IR RADIOMETER COOLING SYSTEM	23	23	23
PARTICLE/FIELD INSTRUMENT BOOMS	-	11	18
RTG POWER AND CONDITIONING	125	125	136
DATA MANAGEMENT	18	18	18
DATA STORAGE	30	30	30
COMMAND CONTROL AND SEQUENCING	20	23	23
TELEMETRY	41	41	41
ANTENNA	34	34	34
ATTITUDE CONTROL	68	68	68
THERMAL CONTROL	11	11	11
GUIDANCE SENSOR SYSTEM	14	14	14
METEOROID SHIELDING	7	7	7
STRUCTURE, WIRING, ETC.	73	75	82
CONTINGENCIES (15%)	84	86	93
TOTAL ORBITED SPACECRAFT	641	664	705

TABLE 2-5. JUPITER ORBITER MISSION PERFORMANCE

SCIENCE AREA \ MISSION	NO 1 PARTICLE AND FIELDS	NO 2 PLANETOLOGY	NO. 3 PLANETOLOGY AND SATELLITE OBSERVATIONS	NO. 4 PARTICLE AND FIELDS AND PLANETOLOGY
Atmospheric Particulate Matter	NA	GOOD	GOOD	FAIR
Global Circulation	NA	GOOD	FAIR	GOOD
Local Phenomena (Lightning, Spots, etc.)	POOR	GOOD	FAIR	FAIR
Atmospheric Thermodynamic State	NA	GOOD	GOOD	FAIR
Cloud Morphology and Properties	NA	GOOD	FAIR	FAIR
Magnetic Field	GOOD	NA	GOOD	GOOD
Gravitational Field	FAIR	GOOD	FAIR	GOOD
Electric Field	GOOD	NA	NA	GOOD
Particles	GOOD	NA	FAIR	GOOD
Solar Wind Interaction Magnetosphere Shock Front Interaction, etc.)	GOOD	NA	NA	FAIR
Planetary Radiation	POOR	GOOD	FAIR	FAIR
Internal Structure (Radius)	NA	FAIR	FAIR	POOR
Surface Characteristics (Existence)	NA	POOR	POOR	POOR
Internal Activity	POOR	FAIR	POOR	POOR

RATING:

NA Not applicable

POOR The mission provides little if any data

FAIR The mission adds some knowledge

GOOD The mission performs well and adds a significant amount of knowledge

TABLE 2-5 (Continued)

SCIENCE AREA \ MISSION	NO. 1 PARTICLE AND FIELDS	NO. 2 PLANETOLOGY	NO. 3 PLANETOLOGY AND SATELLITE OBSERVATIONS	NO 4 PARTICLE AND FIELDS AND PLANETOLOGY
Active Surface Processes	NA	POOR	POOR	POOR
Planet Dynamics (Rotation)	POOR	FAIR	FAIR	POOR
A-Biogenic Organic Compounds	NA	FAIR	FAIR	POOR
Solvents	NA	FAIR	FAIR	FAIR
Energy Sources	NA	GOOD	GOOD	FAIR

RATING:

NA Not applicable

POOR The mission provides little if any data

FAIR The mission adds some knowledge

GOOD The mission performs well and adds a significant amount of knowledge

Table 2-6 lists a comparison of the mechanical and physical properties of Jupiter and Saturn. The appearance, mass, mean density, and atmospheric composition and structure of the two are similar. There are, however, several significant dissimilarities. Of these Saturn's ring system is the most obvious, but there has also been no detection of a magnetic field, RF synchrotron radiation, or any features similar to Jupiter's Great Red Spot. This may be a result of either the absence of these phenomena or the difficulty in observing Saturn from the earth.

Saturn's most unusual feature is its ring system. Fundamental questions on the ring's structure, composition, and origin cannot be answered satisfactorily at present. Observers, however, have fairly well determined the gross structure of the rings. The rings have been observed to consist of three parts; an inner tenuous crape or C ring, a very bright main or B ring, and a moderately bright outer or A ring. A fourth very dark D ring outside the A ring is thought to have been detected during 1966 when the rings appeared edge-on to earth, but the evidence is still questionable. The dimensions of the ring system are: [13]

- Equatorial radius of Saturn	60,400 km., 1 R_s
- C ring, inner boundary	72,000 km., 1.2 R_s
- C-B ring boundary	90,000 km., 1.5 R_s
- B ring, outer boundary	116,000 km., 1.9 R_s
- Cassini division, outer boundary	120,000 km., 2.0 R_s
- A ring, outer boundary	137,000 km., 2.3 R_s

TABLE 2-6

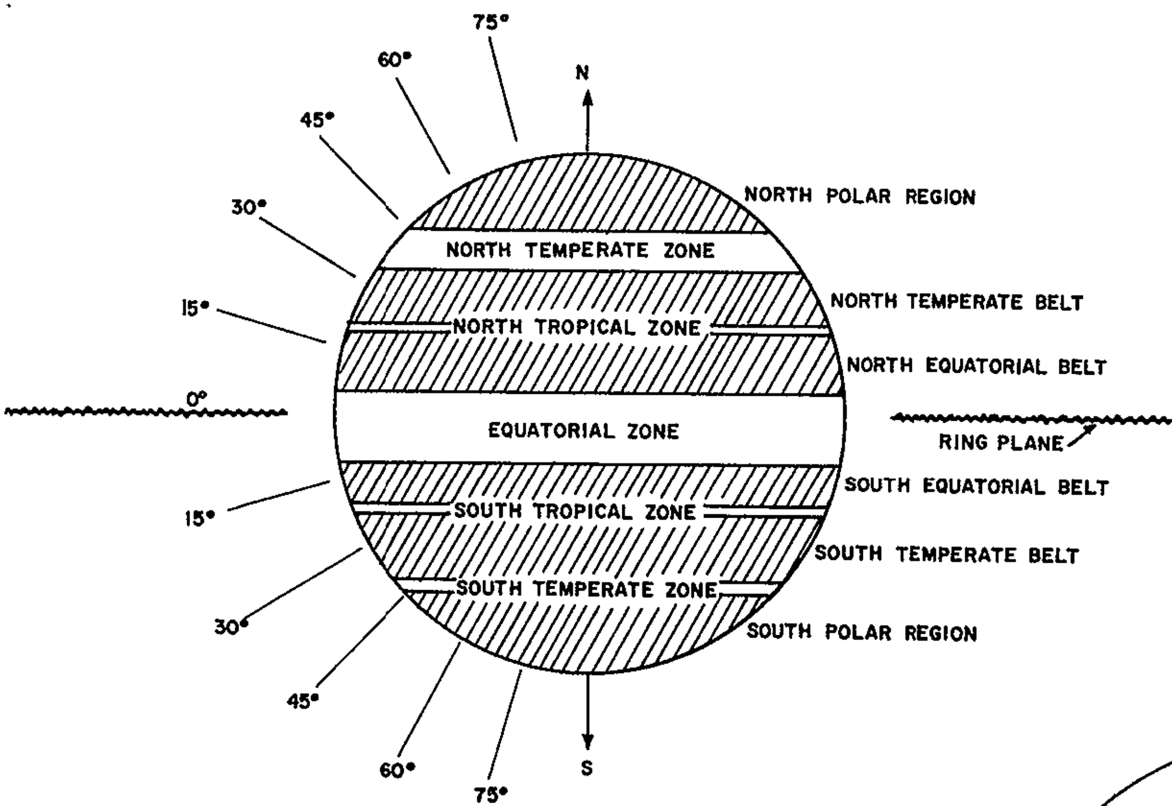
THE PHYSICAL PROPERTIES OF JUPITER AND SATURN

	Jupiter	Saturn
Gravitational Mass, GM_p km^3/sec^2	1.267×10^8	3.793×10^7
Mass, calculated (earth = 1)	317.9	95.1
Equatorial Radius, km (earth = 1)	71,350 11.19	60,400 9.47
Mean Density, g/cm^3	1.334	0.688
Color Index, B-V (sun = 0.63)	0.83	1.04
Average Temperature, °K calculated	105	71
Brightness temperature, °K (measured 8-14μ)	127	93
Mean Surface Gravity (earth = 1)	2.71	1.18
Mean Escape Velocity, km/sec	60.6	36.4
Mean Distance, AU	5.203	9.539
Inclination of Equator to Orbit	3.07°	26.74°
Period of Rotation	9^h50^m-9^h55^m	10^h14^m-10^h38^m

A sketch of the ring system is shown in Figure 2-13. These rings are thought to be continuous and that no other divisions or gaps occur except time varying intensity ripples of 10 or 15 percent. There does seem to be some relationship between Saturn's rings and its second satellite, Mimas. The inner edge of the C ring occurs at a distance corresponding to an orbit with 1/4 the period of Mimas, the boundary between the B and C rings at 1/3, Cassini's division at 1/2, and the outer boundary of the A ring at 2/3. [7] Mimas may not be the sole factor involved here due to several commensurability relationships among the other nine satellites.

A solid or liquid ring rotating about a planet has been shown by Maxwell [18] to be dynamically unstable. Saturn's rings must then consist of a swarm of particles in circilar orbits (if the orbits were elliptical Cassini's division would be filled in). The size distribution and composition of these particles, however, have long been subject to controversy. Particle sizes ranging up to several kilometers have been estimated but recently sophisticated photometric techniques have placed them at between one and ten microns. [14] Infrared scans of the rings indicate that the particles may be composed of ice or snow (or covered with it) but calculated photosputtering rates at Saturn have shown that several centimeters of ice would be eroded in 10^9 years, [15] which is not consistent with particle size. Paraformaldehyde has been offered as a substitute for ice in the ring material in an effort to resolve this problem.

The ring thickness and its degree of flatness cannot be



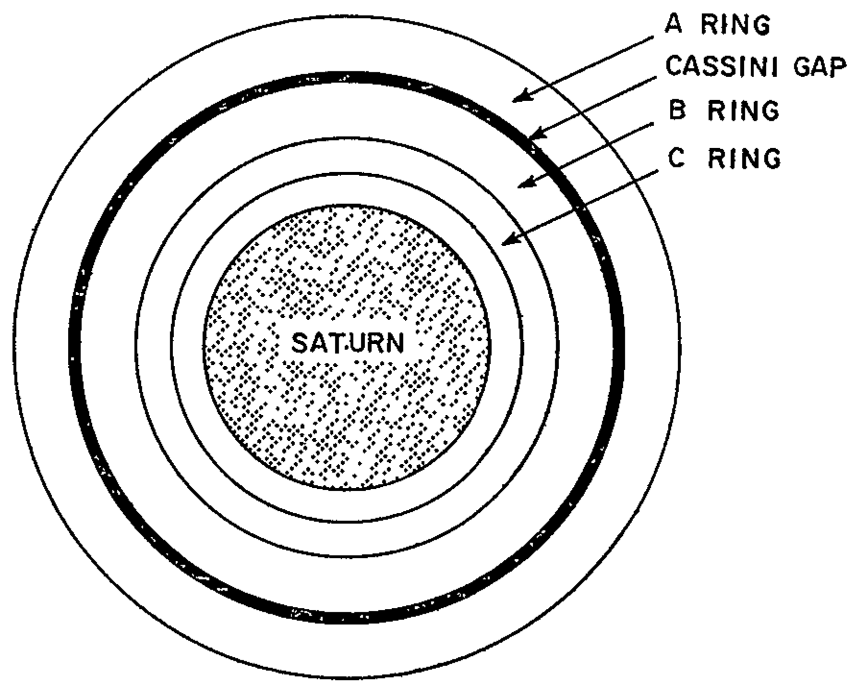


FIGURE 2-13.

SATURN'S ATMOSPHERIC BELTS AND ZONES AND RING STRUCTURE.

determined observationally from earth due to inadequate spatial resolution. The maximum thickness has been placed at several kilometers. The rings may be much thinner, perhaps only several centimeters.

Because the particle size, structure, and composition are poorly known little can be said of the ring's origin. They may be the debris of a disrupted body or non-coalesced material inside the Roche limit,* but this cannot be determined without detailed insitu study of the rings. A spacecraft orbiting Saturn able to perform photmetric, polarization, and spectroscopic studies as well as provide micrometeorite detection and high resolution imagery could advance the state of Saturn ring science considerably.

Saturn's visible disc consists of a series of cloud belts and zones similar to Jupiter's but much less distinctive. The six named belts are shown in Figure 2-13. These belts vary considerably with time and the drawing indicates only their approximate positions. The color shadings of these clouds vary from white to a pale or brownish yellow with an occasional subtle orange or blue tint. Small white spots appear sporatically and fade away within a few weeks, but nothing resembling Jupiter's Great Red Spot has been observed.

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* Within Roche's limit the gravitational attraction between the planet and a particle is greater than the gravitational force between two adjacent particles. In this region a group of particles would not be able to coalesce into a single body. About Saturn this region extends $\sim 1.5 \times 10^5$ km, or to approximately 2.5 radii from the center.

Saturn's cloud top is differentially rotating like Jupiter's but appears to be smoothly changing from a $10^h\ 2^m$ ($\pm$ 4m) rotation period at the equator to a period 11 percent greater at 57 degrees latitude.

Molecular hydrogen, methane, and ammonia have been spectroscopically detected on Saturn. The cloud deck is probably ammonia cirrus as it is at Jupiter with a similar, though modified coloring agent. Helium is also likely to be present to account for the low density, but it has not yet been detected. Aerosols and particulate matter also may be present.

Measured brightness temperatures at Saturn are significantly greater than the planets radiation temperature, as they are for Jupiter. Saturn appears to obtain a greater fraction of the energy it radiates from an internal source than Jupiter does but the energy generation per unit mass is the same for both planets. (7)

Current models of Saturn's atmosphere and interior are based on less accurate data than those for Jupiter. Effective temperatures and molecular abundances are poorly known. Using the known planetary oblateness and gravitational quadrupole moments Peebles (16) has developed a model with a deep adiabatic atmosphere (with a base temperature of 2000° K and a pressure of 2 x 10^5 bars). Assuming a composition of 80% hydrogen, 18% helium and 2% heavy elements an interior model can be developed to fit the known gravitational moments. Probably these numbers give a fair indication of the composition but more accurate data is needed.

Saturn is not emitting detectable non-thermal radio frequency radiation. This suggests that either the magnetic field strength is much lower than Jupiter or that there are no trapped particles in radiation belts. If the magnetic field is proportional to rotation rate, as the "Dynamo" theory predicts, Saturn's magnetic field should be comparable to Jupiter's. This then would mean that either the rings are sweeping up the available charged particles, or that Saturn lies beyond the solar wind and receives no charged particles from it. Little more can be learned about the magnetosphe of Saturn from earth based observations. Investigation by a spacecr will probably be needed to resolve this problem.

Saturn's satellite system is similar to that of Jupiter. Of its ten satellites seven are regular, two semi-regular, and one irregular. The seven regular ones are Janus, Mimas, Enceladus, Tethys, Dione, Rhea, and Titan. These satellites range from between 2.6 Saturn radii (Janus) out to 20.2 R_s (Titan) and their orbital inclinations are all less than 2°. Hyperion's (25 R_s) orbital inclination varies from 17' to 56' while Iapetus' (59 R_s) is 14.7°. Phobe, the irregular satellite, is at over 215 R_s and has an orbital inclination of 150° (retrograde). With the exception of Titan all Saturn's satellites are between 200 and 1400 kilometers in diameter. Titan at 2440 kilometers is larger than the Moon and has the distinction of being the only satellite with a detectable atmosphere in the solar system. (Kuiper [17] spectroscopically detected methan on Titan in 1944). Titan is of the same class as the Galilean satellites, which places a priority on its investigation by Saturn orbiting spacecraft. Iapetus, the only other satellite exhibiting an interesting characteristic, is six times brighter at Western elongation than at Eastern, while retaining the same color. This may mean that it is a long thin chunk of rock with a rotation period about its short axis equal to its 79.3-day orbital period or that one hemisphere is a much more efficient reflector than the other. A good way of resolving this question is by photographing Iapetus from a spacecraft.

2.2.2 Candidate Missions

Four Saturn Orbiter candidate missions have been chosen in a manner analogous to the Jupiter Orbiter selections discussed in Section 2.1.2. Because of the great number of similarities between Jupiter and Saturn orbiters a general discussion of the latter has been deferred to Appendix A. The four Saturn Orbiter mission selections are listed in Table 2-7. These missions are based on the current knowledge of Saturn. As early flybys clarify and enlarge our knowledge of Saturn, the current objectives may have to be modified and updated.

Mission No. 1

Saturn Orbiter Mission No. 1 is a particle and fields orbiter and is very similar to the Jupiter Orbiter Mission No. 1. It consists of two spacecraft in 45-day orbits, one equatorial (0° inclination) and one polar (90° inclination). The science instruments aboard each are identical to the Jupiter Mission with the addition of a spot photometer/polarimeter.[5] (This instrument weighs 4.6 kg and

TABLE 2-7

SATURN ORBITER MISSION SELECTIONS

MISSION NUMBER	SCIENCE OBJECTIVE	ORBIT PARAMETERS		
		PERIAPSE (R_s)	PERIOD (DAYS)	INCLINATION (DEG)
1	PARTICLE AND FIELDS, RING STUDY	3	45	0 AND 90[1]
2	PLANETOLOGY AND RING STUDY	3	15	60
3	PLANETOLOGY AND RING STUDY	1.1	7.5	60
4	PLANETOLOGY, RING STUDY, AND TITAN OBSERVATIONS	3	15.9	0

1. TWO SPACECRAFT ORBITED AT DIFFERENT INCLINATIONS

requires 4 watts of power). The science instruments are listed in Table 2-8 for this mission.

Mission No. 1 is designed to study the near Saturn particle and fields environment as well as to investigate the rings. Because the extent of Saturn's magnetosphere is not known it cannot be determined whether this mission will completely investigate it. This also applies to the trapped or charged particle region which may not exist at Saturn as it does at Jupiter because of the rings. This mission, however, will determine if the solar wind extends out to Saturn.

A polar orbit was chosen for one of the spacecraft because of the wide range of phase angles it provides for viewing the rings. By training the spot photometer/polarimeter on the sunlit portion of the rings for as much of its orbit as possible, information on ring particle size, distribution and composition may be obtained. The micrometeorite detectors aboard will record the particle distribution from 3 to 78 planet radii about Saturn.

The equatorial orbiting spacecraft which views the rings edge on, will train its spot photometer/polarimeter on Saturn to gain data on particulate matter in the atmosphere and cloud composition. Since it is sweeping through the ring plane it will provide data on the radial distribution of matter outside the rings themselves. After several periapse passes (90 - 135 earth days) the spacecraft will employ a short impulse at apoapse to decrease its periapse distance by 0.2 Saturn radii to 2.8 R_s. During each

TABLE 2-8. PAYLOAD DEFINITIONS, SATURN ORBITER MISSION NOS. 1, 2, 3 AND 4.

INSTRUMENTS	WEIGHT (kg)				POWER (WATTS)
	MISSION NO. 1 PARTICLE AND FIELDS AND RING STUDY	MISSION NO. 2 PLANETOLOGY AND RING STUDY	MISSION NO. 3 PLANETOLOGY AND RING STUDY	MISSION NO. 4 PLANETOLOGY, RING STUDY AND TITAN OBSERVATIONS	
1.5 INCH RETURN BEAM VIDICON[1]		18	18	18	26
NEAR - IR LINE SCANNER[1]		13	13	13	5
3 - CHANNEL IR RADIOMETER[1]		13[2]	13[2]	13[2]	5
NARROW BAND UV PHOTOMETER/SPECTROMETER[1]		7	7	7	6
SPOT PHOTOMETER/POLARIMETER PACKAGE	4.6	4.6	4.6		4
X-RAY IMAGER[1]		3.6	3.6		8
IONOSONDE		11	11	11	10
SWEPT FREQUENCY RF RECEIVER	3 6				3
DC ELECTRIC FIELD DETECTOR (USE RF ANTENNA)	1.8				1 5
LEPEDEA	1.6				2
LITHIUM - DRIFTED SOLID STATE DETECTOR (AND DOSIMETER)	2.7				2
GEIGER TUBE TELESCOPE	1.1			1 1	1
TRAPPED RADIATION DETECTORS (SEE TABLE 2-2)	2.5			2.5	3
TWO ELECTROSTATIC PLASMA ANALYZERS	4.3				4
VECTOR HELIUM MAGNETOMETER	4.6			4 6	2 5
MICROMETEOROID DETECTORS	7.8	7.8	7 8	7.8	2
RF OCCULTATION	YES	NO	YES	YES	
TOTALS	34 6	78.0	78.0	78.0	

1. THESE PLANETOLOGY INSTRUMENTS HAVE LATERAL SCANNING CAPABILITY

2. REQUIRES AN ADDITIONAL 23Kg, SOLID METHANE COOLING SYSTEM

succeeding orbit, at apoapse, it will perform a similar maneuver, slowly "lowering" itself into the rings by decreasing its periapse distance in intervals of 0.2 R_s down to 1.1 R_s (3.0, 2.8, 2.6,.... 1.4, 1.2, 1.1). This requires a total propulsion allowance of 125 m/sec and about 450 days for completion, and should provide a complete map of the radial distribution and density of the ring particles. Figure 2-14 illustrates the spacecraft path near periapse and the rings for the equatorial mission. At any time the "drop" into the rings can be terminated should telemetry indicate that the spacecraft is suffering damage from collisions with the ring particles. The spacecraft's velocity near periapse will be very near that of the ring particles (~ 20 km/sec) so that no high velocity collisions should occur.

The spacecraft subsystem weight breakdown for Mission No. 1 is listed in Table 2-9. The weights are nearly identical to those for Jupiter Mission No. 1 (P/F). The data storage and management, and the communications systems are based on the TRW Pioneer F/G system [11] (9' antenna, 8 watt, X band at 2 kbps) and a maximum acquisition bit rate of about 200 bits per second. Each spacecraft requires a six hour data transmission period using the 210' receiving disk every two days.

Mission No. 1 will provide data on the photometric properties of Saturn's rings and cloud tops over a wide range of phase angles (the maximum range of phase angles Saturn can be viewed from the earth is 6°). It will map parts of the magnetosphere and charged

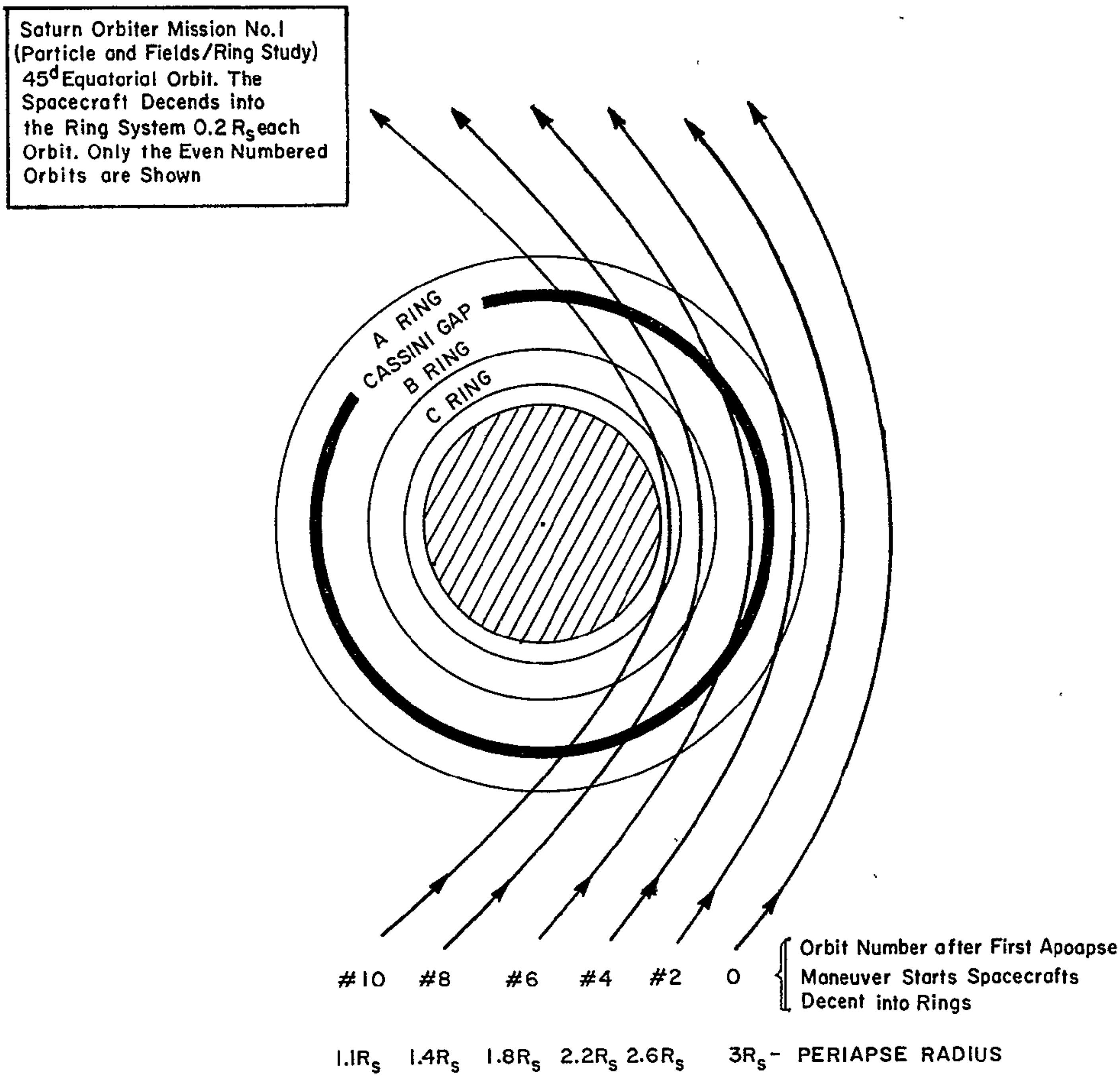


FIGURE 2-14. CONTROLED DECENT OF A SATURN ORBITER INTO THE RING SYSTEM.

TABLE 2-9

SPACECRAFT WEIGHT BREAKDOWN SATURN MISSION NOS. 1, 2, 3 AND 4.

SUBSYSTEM	MISSION NO. 1 PARTICLE AND FIELDS/RING STUDY	MISSION NOS. 2 & 3 PLANETOLOGY AND RING STUDY	MISSION NO. 4 PLANETOLOGY, RING STUDY, TITAN ENCOUNTER
SCIENCE	35 KG	78 KG	78 KG
PLANETOLOGY INSTRUMENT PLATFORM	--	16	16
IR RADIOMETER COOLING SYSTEM	--	23	23
PARTICLE/FIELD INSTRUMENT BOOMS	--	--	11
RTG POWER AND CONDITIONING	68	125	125
DATA MANAGEMENT	12	48	48
DATA STORAGE			
COMMAND CONTROL AND SEQUENCING	27	20	23
TELEMETRY	32	41	41
ANTENNA			
ATTITUDE CONTROL	18	68	68
THERMAL CONTROL	5	11	11
GUIDANCE SENSOR SYSTEM	--	14	14
METEOROID SHIEDLING	34	7	7
STRUCTURE, WIRING, ETC.		73	75
CONTINGENCIES (15%)		84	86
TOTAL ORBITED SPACECRAFT	231 (485) *	642	660

* TWO SPACECRAFT + 5% MOUNTING STRUCTURE

particle environment which will allow later orbiters to more completely and effectively study these areas. Mission No. 3 also allows an RF occultation of Saturn's atmosphere, providing data on atmospheric refraction and the height of the ionosphere.

Missions No. 2 and 3

Missions No. 2 and 3 are both designed to study Saturn and its rings. Both have science payloads similar to Jupiter Mission No. 2 (planetology). Non-thermal RF radiation is not expected to be as intense from Saturn as it is from Jupiter so the RF receiver has been replaced by the photometer/polarimeter package used to study the rings in Mission No. 1. Mission No. 2 payload is listed in Table 2-8.

Mission No. 2 consists of 642 kg spacecraft in a 15-day, 60° inclined orbit which does not pass through the rings. Periapse is just beyond the solar terminator for the first orbit which allows good daylight coverage at resolutions less than 200 km. Figure 2-15 illustrates the portion of the orbit at which the planetology instruments operate at less than 100 km resolution.

Several ring occultations are possible for Mission No. 2 due to the configuration of its orbit. These occultations which allow the spacecraft's photometer and TV imager to view the sun through the rings will provide information on particle size and distribution. Figure 2-16 shows the apparent track of the sun across the rings as viewed by the Mission No. 2 spacecraft for a particular arrival date. This figure also includes a schematic as an aid to understanding the

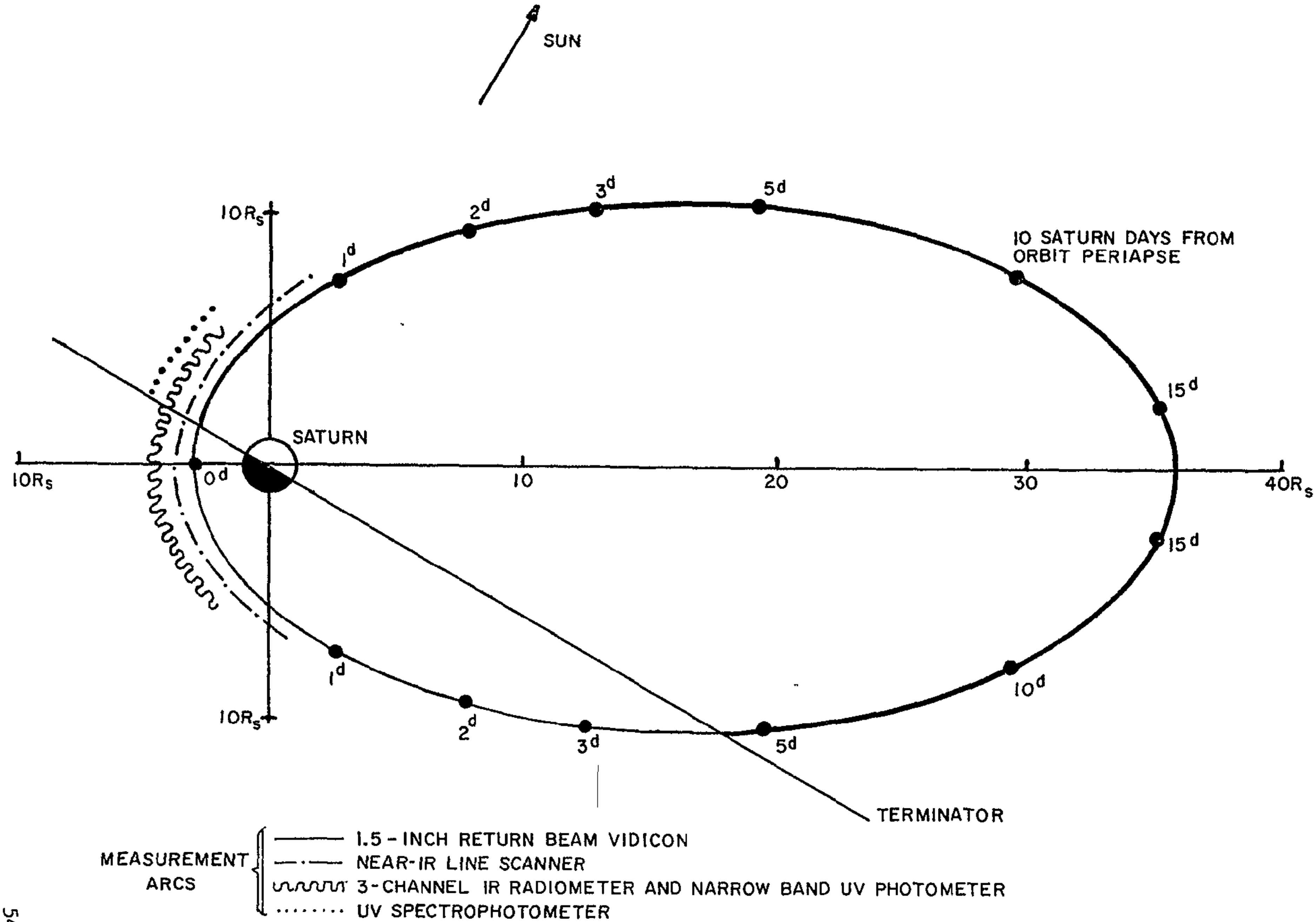
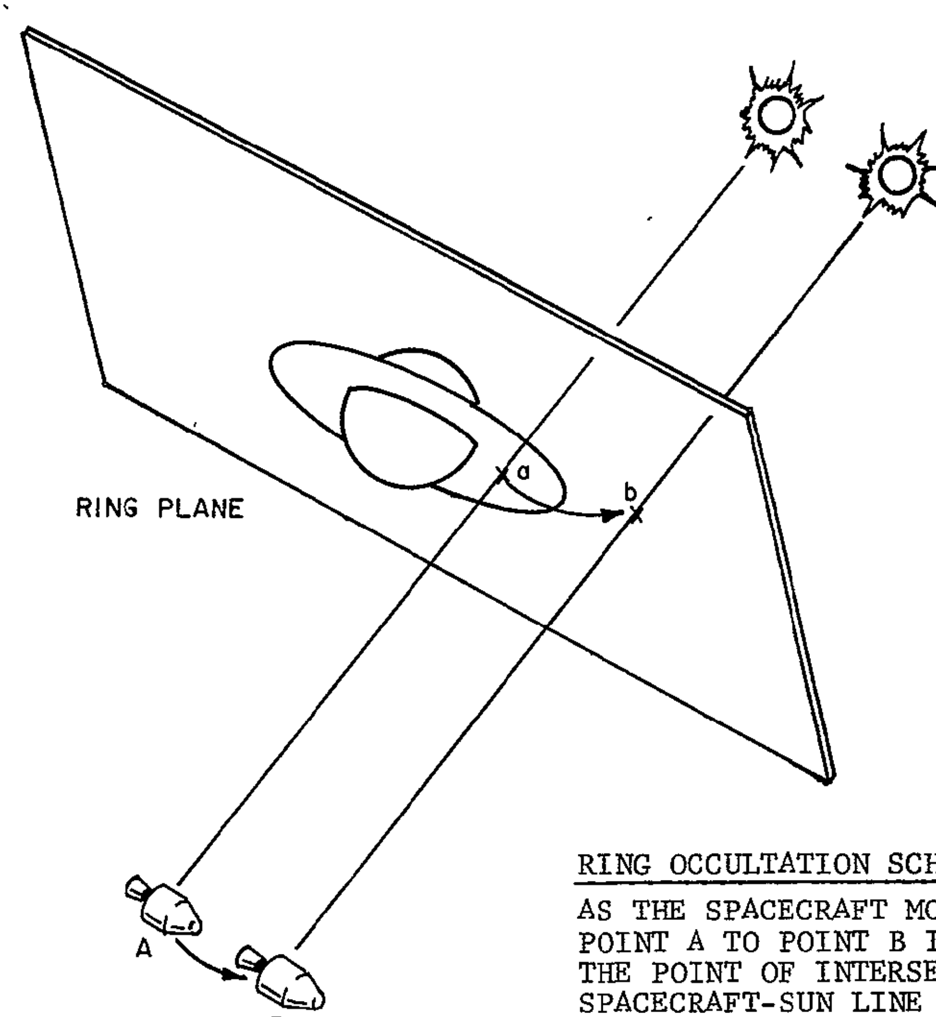


FIGURE 2-15. PLANETOLOGY INSTRUMENT MEASUREMENT ARCS FOR 15-DAY (EARTH DAYS) SATURN ORBIT (MISSION NO. 2).



RING OCCULTATION SCHEMATIC

AS THE SPACECRAFT MOVES FROM POINT A TO POINT B IN ITS ORBIT THE POINT OF INTERSECTION OF THE SPACECRAFT-SUN LINE AND THE RING PLANE MOVES BETWEEN POINTS a AND b, TRACING OUT A CURVE IN THE RING PLANE (THE SUN APPEARS TO MOVE BETWEEN a AND b) WHEN THIS CURVE FALLS WITHIN THE BOUNDARIES OF THE RING SYSTEM (1.2 R_s - 2.3 R_s) A RING OCCULTATION OCCURS

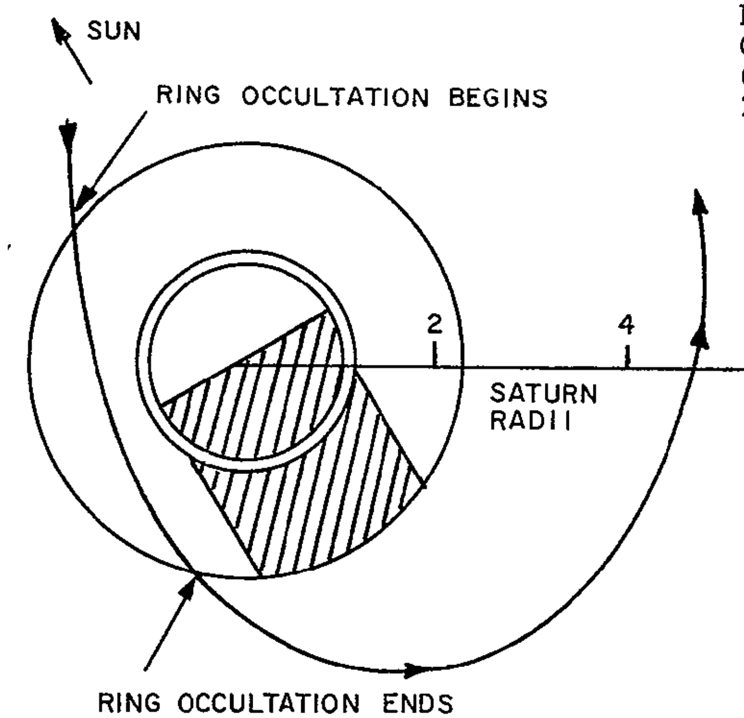


FIGURE 2-16. RING OCCULTATION MISSION NO. 2

geometry involved. (Ring occultations are further discussed in Appendix A).

Mission No. 3 employs the same spacecraft and payload used in No. 2 but in a 7.5-day orbit with a 60° inclination and a periapse radius of 1.1 Saturn radii (a height of ~ 6040 km above cloud top). Placing the periapse inside the ring is hazardous to the spacecraft due to possible collisions with ring particles, but there are several reasons affecting the mission which make the low periapse-short period orbit attractive. First, flight times to Saturn are about 4 to 5 years. The Jupiter and Saturn spacecraft are nearly identical and they will probably have the same reliability and anticipated lifetime. It is therefore important to maintain the shortest orbital period at Saturn commensurate with the objectives. The 7.5-day Saturn orbit allows twice the number of periapse passes as does the 15-day orbit, and will generate more planetology data before spacecraft failure. Secondly, Saturn may not have the high energy radiation belts surrounding it that Jupiter does and the spacecraft can probably remain relatively close to the planet without being damaged by radiation. Lastly, injection into the low periapse (1.1 R_S) orbit from an interplanetary trajectory requires less retro ΔV than for a 3 R_S periapse orbit.

Mission No.3 allows the same high latitude coverage Mission No.2 does with somewhat better resolution of the cloud tops. It also provides twice as many periapse passages as No.2, but risks possible damage by the rings and the unknown Saturn radiation belts. This mission also allows ring occultation opportunities similar to that of Mission No. 2, (See Appendix A). Mission No.3 provides a greater amount of information on Saturn and its rings than Mission No. 2, but at a greater risk to the spacecraft. It also provides opportunities for RF occultations of Saturn's atmosphere which yield data on atmospheric composition and refraction.

Mission No. 4

Saturn Mission No. 4 is similar to the Planetology and Satellite observation mission at Jupiter (Mission No. 3). The spacecraft is placed in a 15.95-day equatorial orbit which allows it to repeatedly observe Titan. The equatorial orbit allows the lowest approach velocity relative to Titan. The spacecraft is able to view Titan from as near as 70,000 km on each orbit, requiring only a small impulsive maneuver to correct for perturbations from the satellites and Saturn's oblateness.

The science payload for Mission No. 4 is listed in Table 2-8. The x-ray imager and spot photometer/polarimeter have been omitted because of their low relative usefulness in an equatorial orbit. Several particle and fields instruments have been added to monitor the effects of Titan on Saturn's magnetosphere.

Mission No. 4's equatorial orbit reduces the latitude coverage available to the planetology instruments, allowing about 85 percent of the planet to be viewed at better than 200 km resolution. It does provide several opportunities for RF occultations of Saturn's atmosphere. The radial distribution of particles in the ring plane outside the rings can also be mapped on this mission. Mission No. 4 doesn't provide as much planetology information or photometric ring data as Missions No. 2 and 3, but is able to study Titan and the distribution of matter in the ring plane outside the rings more thoroughly than either.

The spacecraft subsystem weight breakdowns for Missions No. 1, 2, 3, and 4 are listed in Table 2-9. The communications, data storage and management systems are based on the JPL [12] TOPS Grand Tour spacecraft design. Each spacecraft collects data at an average rate of $\sim 1.5 \times 10^7$ bits/hour over the sunlit side of Saturn. With a storage capability of 2×10^9 bits each spacecraft must transmit its accumulated data (X band, at 50 kbps) for 6 hours every third day (210' disk receiving).

Table 2-10 summarizes the discussion of each of the Saturn Orbiter Reference Missions in this section. As in Section 2.1.2 each mission has been given a subjective performance rating in each of the Saturn science areas or objectives.

TABLE 2-10. SATURN ORBITER MISSION PERFORMANCE

SCIENCE AREA \ MISSION	NO. 1 PARTICLE AND FIELDS/RING STUDY	NO. 2 PLANETOLOGY AND RING STUDY	NO 3 PLANETOLOGY AND RING STUDY	NO 4 PLANETOLOGY, RING STUDY, AND TITAN ENCOUNTER
Atmospheric Particulate Matter	FAIR	GOOD	GOOD	GOOD
Global Circulation	NA	GOOD	GOOD	FAIR
Local Phenomena (Lightning, Spots, etc.)	POOR	GOOD	GOOD	FAIR
Atmospheric Thermodynamic State	NA	GOOD	GOOD	GOOD
Cloud Morphology and Properties	POOR	GOOD	GOOD	FAIR
Mechanical and Physical Properties of the Rings	FAIR	GOOD	GOOD	POOR
Ring Composition (Particle Size, Distribution, Composition)	GOOD	GOOD	GOOD	POOR
Magnetic Field	GOOD	NA	NA	FAIR
Gravitational Field	FAIR	GOOD	GOOD	GOOD
Electric Field	GOOD	NA	NA	NA
Particles	GOOD	GOOD	GOOD	GOOD
Solar Wind Interaction (Magnetosphere Shock Front Interaction, etc.)	FAIR	NA	NA	POOR
Planetary Radiation	POOR	GOOD	GOOD	FAIR
Internal Structure (Radius)	NA	FAIR	FAIR	FAIR

RATING:

NA	Not applicable
POOR	The mission provides little if any data
FAIR	The mission adds some knowledge
GOOD	The mission performs well and adds a significant amount of knowledge

TABLE 2-10. Continued.

SCIENCE AREA \ MISSION	NO. 1 PARTICLE AND FIELDS/RING STUDY	NO. 2 PLANETOLOGY AND RING STUDY	NO. 3 PLANETOLOGY AND RING STUDY	NO. 4 PLANETOLOGY, RING STUDY, AND TITAN ENCOUNTER
Surface Characteristics (Existence)	NA	POOR	POOR	POOR
Internal Activity	POOR	FAIR	FAIR	FAIR
Active Surface Processes	NA	POOR	POOR	POOR
Planet Dynamics (Rotation)	POOR	FAIR	FAIR	FAIR
Life Associated Substances	NA	FAIR	FAIR	FAIR
A-Biogenic Organic Compounds	NA	FAIR	FAIR	FAIR
Solvents	NA	FAIR	FAIR	FAIR
Energy Sources	NA	GOOD	GOOD	GOOD

RATING:

NA Not applicable

POOR The mission provides little if any data

FAIR The mission adds some knowledge

GOOD The mission performs well and adds a significant amount of knowledge

3. SOLAR ELECTRIC CHARACTERISTICS AND CAPABILITIES

The solar electric propulsion (SEP) capability for performing Jupiter and Saturn orbiter missions will now be described. Assuming a nominal set of SEP system parameters, this capability is measured by the net mass delivered into orbit as a function of the interplanetary flight time. Net mass is defined as consisting of the science payload and spacecraft support subsystems; it does not include either the chemical retro stage or the SEP system masses. The data basefor this trajectory analysis was provided by JPL, and corresponds to a complete optimization of launch date, thrust direction, thrust periods, launch and arrival hyperbolic velocities, and the SEP propulsion system parameters (power rating and specific impulse). This data was scaled as needed to apply to the candidate missions according to the scaling relationships developed in a previous study task.[4] Optimum performance is defined here as maximum net mass in orbit.

Two general flight mode options are available for a given heliocentric flight time to the outer planets. The first is termed the direct mode and is characterized by a trajectory profile of steadily increasing radial distance and a total transfer angle less than 360°. The second is termed the indirect mode and involves a total transfer angle greater than 360° where the spacecraft first travels inward toward the sun before heading out to the target planet. The existence of the indirect solution derives from the potential advantage of utilizing the solar power source in a more efficient manner, i.e., higher power for a longer time. Previous studies have shown that the direct mode is optimum in the region of lower net mass capability and shorter flight times, whereas the indirect mode is optimum for longer flights. It can be assumed a priori that the indirect mode is not needed for Jupiter missions. However, the two flight mode options will be evaluated for the Saturn mission application.

In addition to the net mass versus flight time capability, auxiliary data is presented on the optimum values of the various SEP flight parameters such as power, specific impulse and arrival conditions. The question of whether or not to jettison the SEP stage in the case of Jupiter missions is evaluated as a tradeoff between power availability, net mass, and operational simplicity. The tradeoff between different retropropulsion technologies is shown as well as the performance degradation due to finite length launch windows and off-optimum power and specific impulse. A final item of consideration is the possibility of a common SEP stage design to apply to both Jupiter and Saturn missions.

3.1 Nominal System Parameters

Although the missions of interest may not be flown before the 1980 decade, this study conservatively assumes the use of a propulsion technology that is currently available or expected to be available by the mid-1970's. The nominal system parameters are listed in Table 3-1. A rollout solar array having a specific mass

Table 3-1

SOLAR ELECTRIC PROPULSION PARAMETER VALUES *
1970 TECHNOLOGY STATUS

1.	Power subsystem specific mass, α_w (rollout solar array)	15 kg/kw
2.	Thrust subsystem specific mass, α_{ts} (includes power conditioner, thruster array, thrust vector control and redundancy)	12 kg/kw
3.	Effective propulsion system specific mass α_{ps} (20% contingency factor on α_w for solar cell degradation, losses and auxiliary power requirement)	30 kg/kw
4.	Propulsion system efficiency, η(4000 sec)	68%
	Thruster efficiency at I_{sp} = 4000 sec 75%	
	Power conditioning efficiency 91%	
5.	Propellant tankage factor, k_p (percent of propellant mass)	3%

---

* 2-3 kw thruster modules; 10-15 kw system power rating.

of 15 kg/kw is assumed. Thrust subsystem specific mass of 12 kg/kw is thought to be a conservative estimate. A typical specific mass breakdown of the thrust subsystem is 4.5 kg/kw for the power conditioners and 7.5 kg/kw for the thruster array including gimbal-translator and redundancy. It is emphasized, however, that the values stated above are averages and that actual system specific masses are dependent on the operating specific impulse and the particular design configuration. A 20 percent contingency factor is assigned to the power subsystem to account for possible solar cell damage due to radiation and micrometeoroid impacts, array performance uncertainty, and spacecraft subsystem auxiliary power requirements. This brings the total SEP system specific mass to 30 kg/kw.

Propulsion system efficiency is 66 percent at a specific impulse of 3500 seconds. This assumes a 30 cm mercury electron bombardment thruster operating in the power range 2-3 kw and a power conditioning efficiency of 91 percent (see Figure 3-1 for variation of efficiency with specific impulse). Current tankage design for mercury propellant gives a dry weight, including pressurization and expulsion systems, of about 3 percent of the propellant loading.

Another system parameter which influences the SEP performance results is the solar power curve, i.e., the variation of the solar array output power with distance from the sun. Based

- CURVE FIT TO MEASURED EFFICIENCY OF ELECTRON BOMBARDMENT THRUSTERS (30 CM, 2-3 KW)
- INCLUDES 91 PERCENT POWER CONDITIONING EFFICIENCY

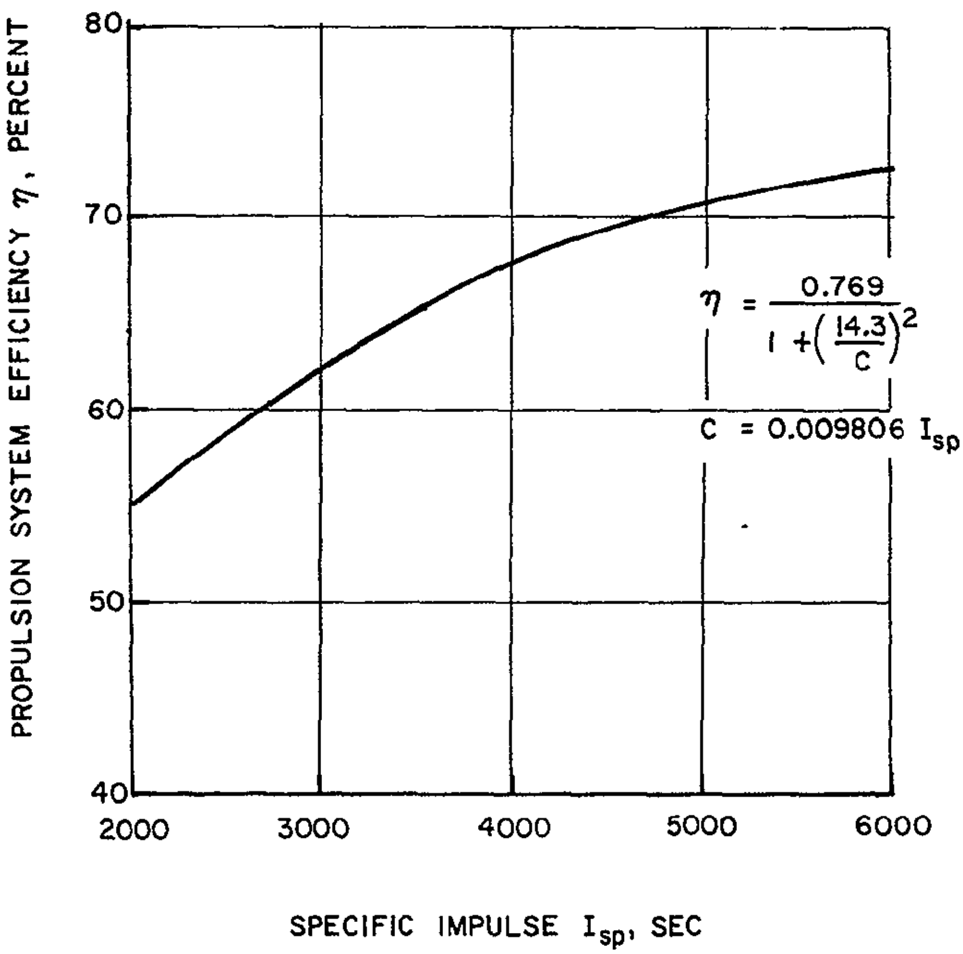


FIGURE 3-1. PROPULSION EFFICIENCY CURVE

on experimental studies, the solar power curve shown in Figure 3-2 was recommended by JPL for use in this study. The power ratio peaks with a value of 1.4 at a solar distance slightly below 0.7 a.u. and then falls off abruptly because of decreased cell efficiency with higher operating temperatures (panel tilting would alleviate this problem). For outer planet missions, the region of major interest is at solar distances greater than 1 AU It will be noted that the power available at Jupiter's mean distance is 5.7 percent of the initial power, while at Saturn it is under 2 percent.

Two different chemical retropropulsion systems needed for orbit capture are compared in the analysis. The first is a beryllliumized solid propellant system with a specific impulse of 300 seconds and a retro inert (hardware) fraction of 11 percent of the propellant loading. The second example is a space-storable liquid propellant system such as flouride-diborane having a specific impulse of 400 seconds. Pressure-fed systems of this type have a relatively large inert fraction of about 25 percent.

3.2 Jupiter Orbiter Missions

3.2.1 Launch Opportunities

Optimum launch dates for Jupiter missions occur approximately 13 months apart, but the trajectory energy requirements vary from year to year because the earth and Jupiter do not move in circular, coplanar orbits. This variation is cyclical with a frequency of

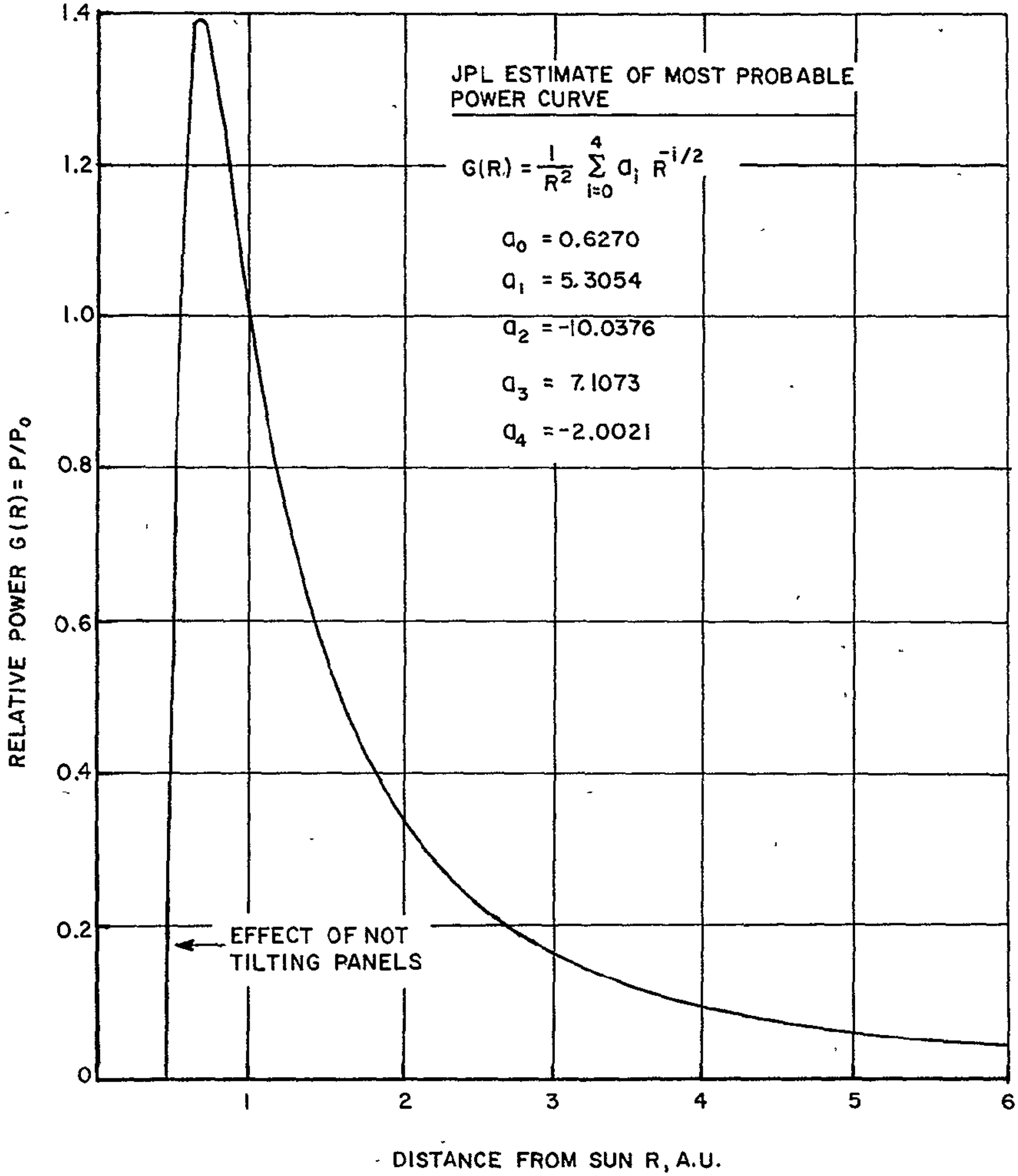


FIGURE 3-2. SOLAR CELL POWER PROFILE

about 11 years commensurate with Jupiter's orbital period. Figure 3-3 illustrates the SEP net mass capability over a launch opportunity cycle (1974-85) for fixed flight times of 600 and 800 days. These are direct mode trajectories having heliocentric travel angles less than 360°. The two options of jettisoning or not jettisoning the SEP propulsion system are shown. It will be recalled that the net mass does not include the SEP propulsion system when it is taken into orbit. The mass difference between the best and worst launch years is 160 - 180 kg for the 600 day flight and 115 - 135 kg for the 800 day flight. The larger difference in each case corresponds to jettisoning the propulsion system. If one takes these differences on a percentage basis relative to the best opportunity, the above characteristic is reversed. For example, the variation is 35% in the 600 day, non-jettison case but only 13% in the 800 day, jettison case.

Figure 3-4 illustrates the 800 day heliocentric trajectory launched in 1984. Thrust cut off occurs at about 3 AU, 290 days after launch. For purposes of describing the general characteristics of the SEP flight mode, it will be convenient to use the condition of circular, coplanar planet orbits. The corresponding results given by the dotted lines in Figure 3-3 are seen to represent average conditions over the launch opportunity cycle. Specific launch years will again be considered when comparing SEP and ballistic performance in Section 4 of this report.

3.2.2 Optimum SEP Flight Parameters

Figures 3-5 and 3-6 show the variation of optimum values of

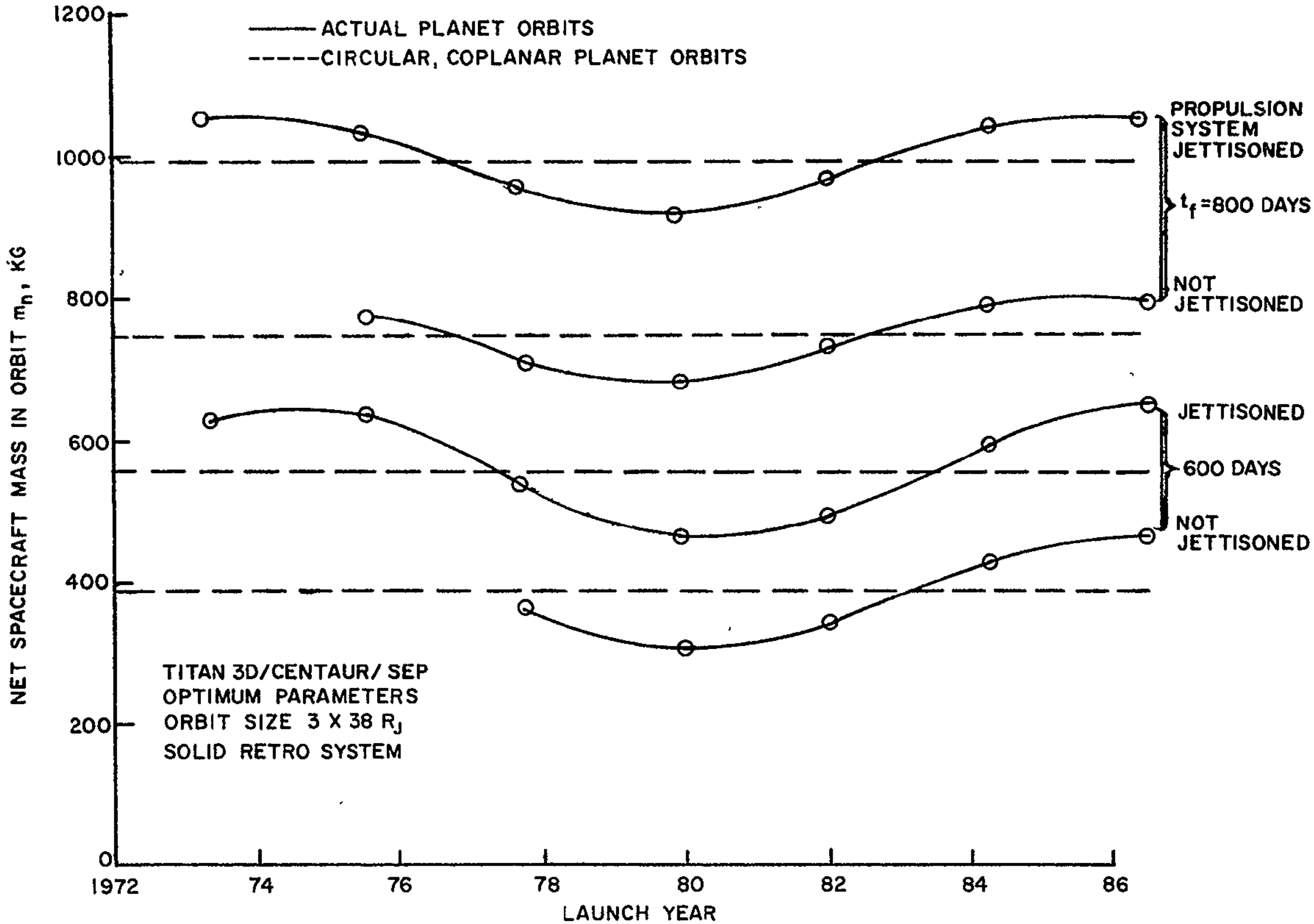


FIGURE 3-3. EFFECT OF LAUNCH OPPORTUNITY ON SOLAR ELECTRIC CAPABILITY FOR JUPITER ORBITER MISSION.

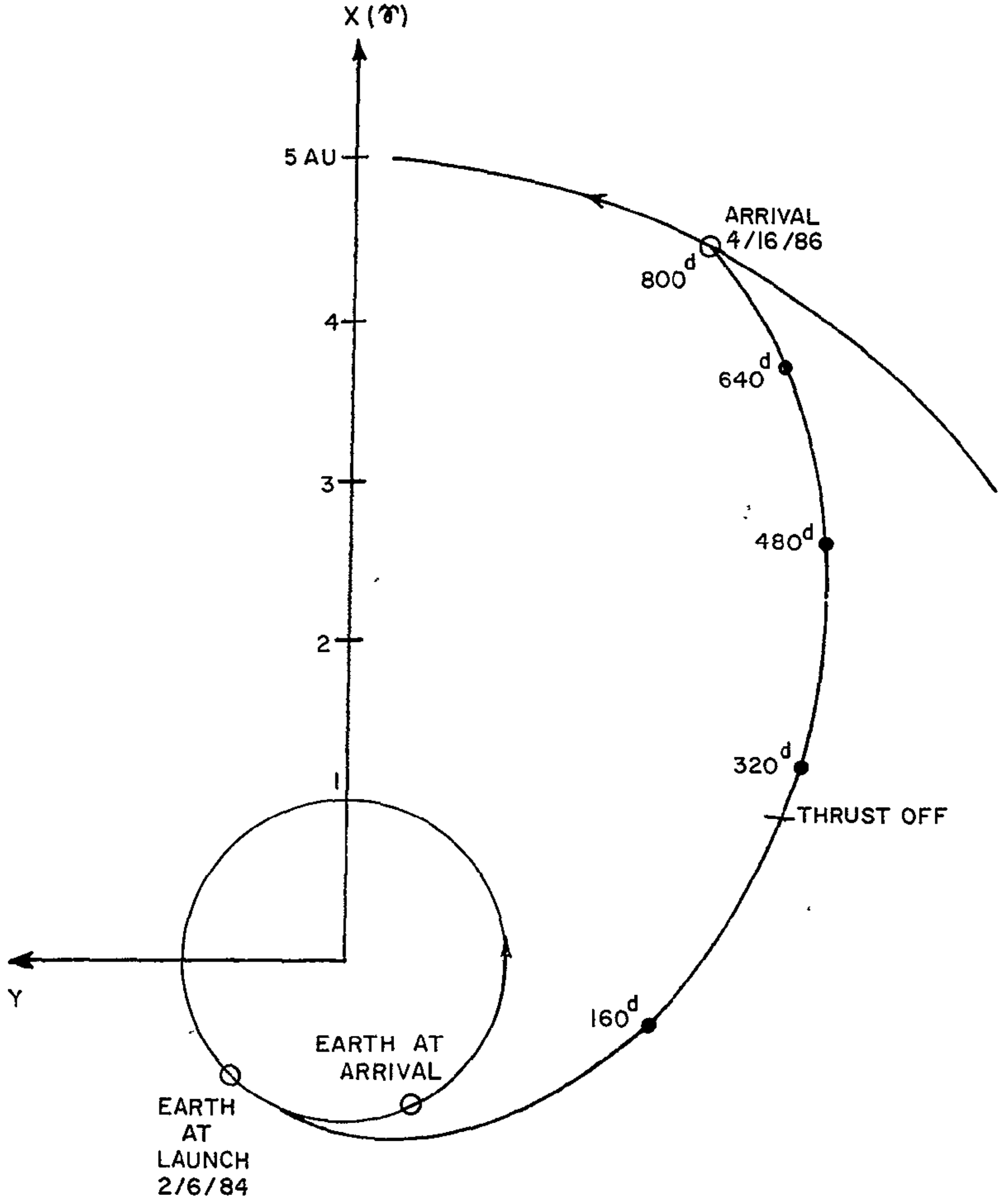


FIGURE 3-4. 800 DAY SOLAR ELECTRIC TRAJECTORY TO JUPITER.

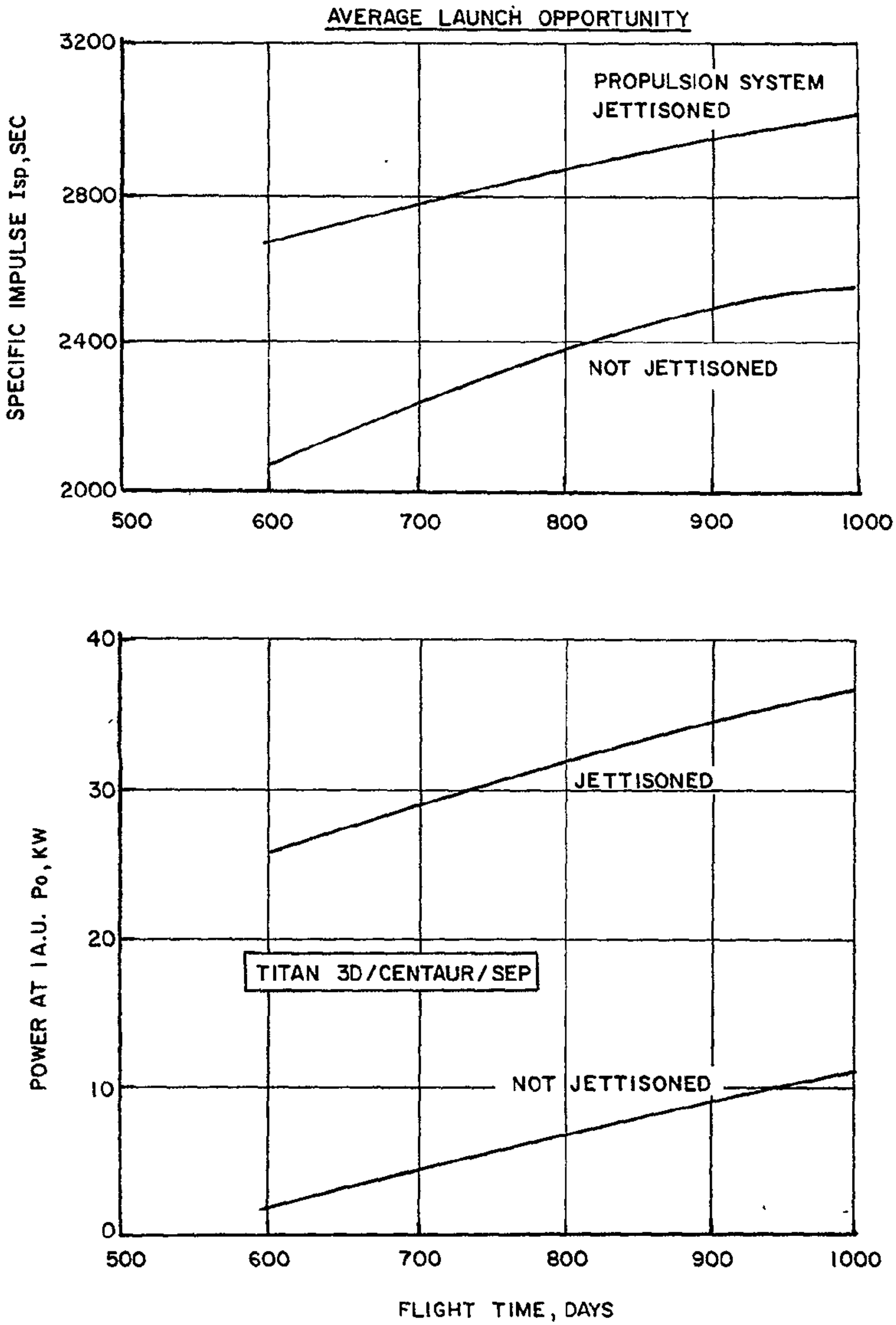


FIGURE 3-5. OPTIMUM POWER RATING AND SPECIFIC IMPULSE FOR JUPITER ORBITER MISSIONS.

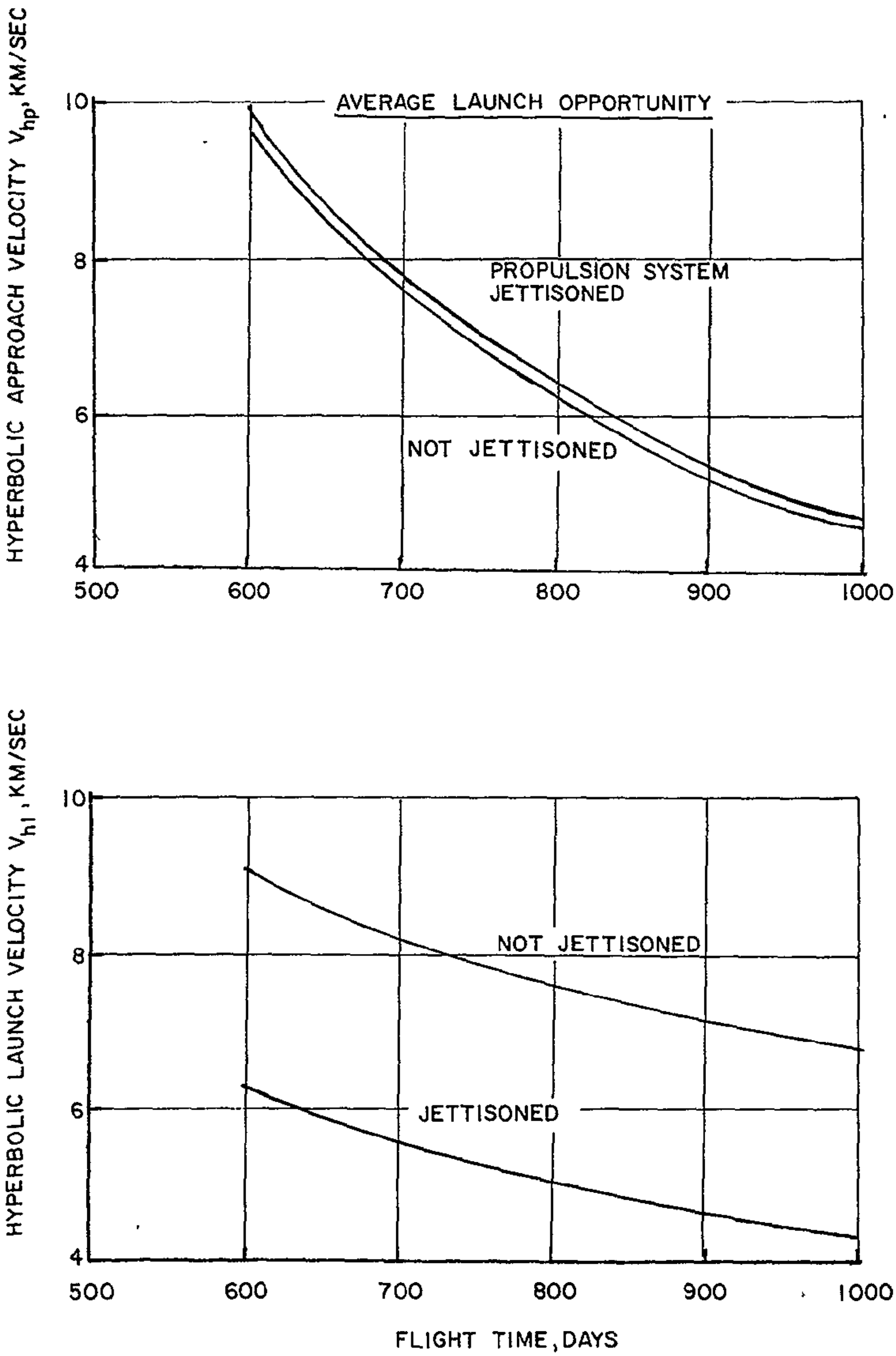


FIGURE 3-6. OPTIMUM LAUNCH AND APPROACH VELOCITIES FOR JUPITER ORBITER MISSIONS

power, specific impulse, and hyperbolic launch and approach velocity with flight times to Jupiter over the range 600-1000 days. A general characteristic of direct mode SEP trajectories is that optimum values of power and specific impulse both increase with longer flight times. Correspondingly, the launch and approach velocities decrease with flight time in a manner similar to ballistic trajectories. Another important characteristic is that the jettison option requires higher values of power and specific impulse. For example, at a flight time of 800 days, the optimum values of P_o and I_{sp} are 32kw and 2860 seconds, respectively, when the propulsion system is jettisoned. In the non-jettison case, the optimum values are 6.7kw and 2380 seconds; the lower power is necessary because taking the SEP powerplant into orbit places an added weight burden on the chemical retro stage.

The low values of optimum specific impulse may present a design problem since the physical spacing of thruster accelerating grids decreases as the specific impulse decreases. Current electron bombardment thruster technology has been concentrated in the I_{sp} region 2700-4000 seconds. The higher operating I_{sp} is preferred from a thruster design standpoint even if it is not optimum for the mission in terms of mass delivered. It will be shown, however, that a higher than optimum I_{sp} (e.g. 3500 sec) incurs only a small mass penalty.

3.2.3 Selection of Jettison Option

Since the solar array area is proportional to the power rating (~100 ft^2/kw), the spacecraft whose propulsion system is not jettisoned would have a substantially smaller size than the spacecraft employing the jettison option (assuming optimum power in each case). Although a smaller SEP spacecraft could lead to a simpler base design configuration and lower cost, these advantages can easily be offset by other factors such as the power mass delivered into orbit, the marginal solar power available for mission operations, and the need to retract the array prior to the orbit capture retromaneuver.

Figure 3-7 compares the net mass/flight time performance of the jettison and non-jettison options for Mission No. 2 -- planetology orbiter, 15 day period, 60° inclination. Note that the power supply mass needed to provide the mission power requirement of 400 watts has been excluded so as to compare the two options on an equal basis. The nominal planetology mission requirement is then 532 kg; this is derived from Table 2-4 by subtracting the RTG power supply mass of 109 kg (400 watts x 0.272 kg/watt). Assuming optimum power (P_o) operation in the non-jettison case and recalling that the solar power available in Jupiter orbit is $0.057P_o$, it is found that an auxiliary power supply is needed to assist the solar panels for flight times under 800 days. This added power is 158 watts for the 700 day flight and 290 watts for the 600 day flight. The lower curve in Figure 3-7 shows that these power deficiencies are made up by adding RTG units of 43 kg and 80 kg, respectively. Additional solar panels could be used instead of the RTG, but the mass tradeoff is about the same since the effective specific mass of the solar panels in Jupiter orbit is 0.015/0.057=0.263 kg/watt.

JUPITER ORBITER - AVERAGE LAUNCH OPPORTUNITY

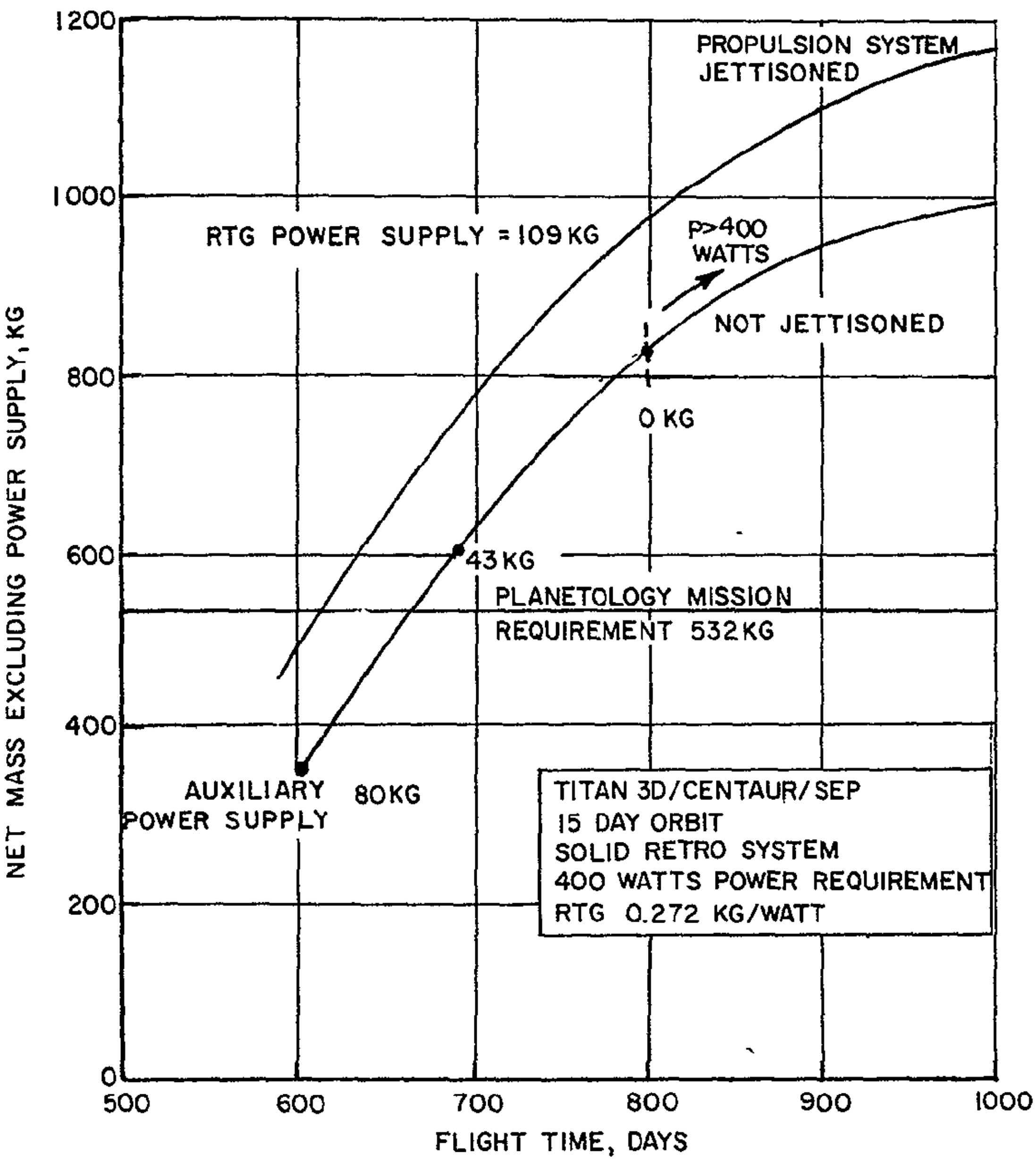


FIGURE 3-7. TRADEOFF SELECTION OF PROPULSION SYSTEM JETTISON OPTION FOR JUPITER MISSIONS.

The comparison shown favors the option of jettisoning the propulsion system even though the flight time advantage is not too significant at lower mass values. A flight time reduction from 660 days to 610 days is obtained for the nominal mission requirement of 532 kg.

The option of jettisoning the SEP system is selected at this point in the analysis as being preferred for Jupiter orbiter missions. Reasons for this selection are:

1. Capability for larger payloads in orbit or shorter flight times.
2. Easier implementation of final midcourse, orbit capture, plane change and orbit trim maneuvers in the absence of a large solar array.
3. Decreased attitude control requirements because of lower spacecraft moments of inertia.
4. Commonality with Saturn orbiter which will employ the jettison option (available power from solar cells is insufficient at Saturn).

The SEP interplanetary stage would be separated following thrust cut-off. For the remainder of the mission the spacecraft configuration is that of a ballistic spacecraft.

3.2.4 Effect of Orbit Selection and Retrosystem

Figure 3-8 shows the net orbiter spacecraft mass as a function of flight time for candidate orbit periods of 7.5, 15, 30

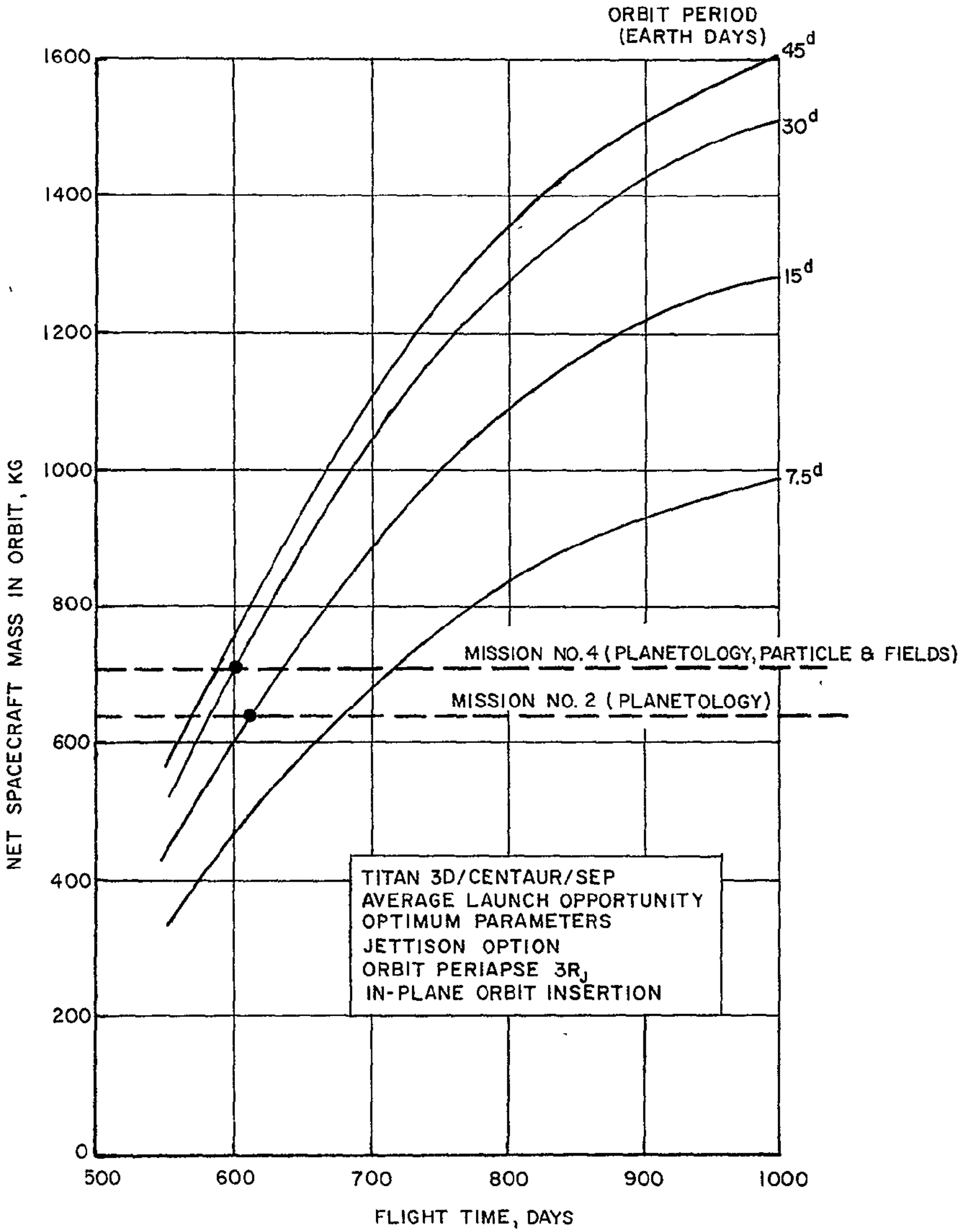


FIGURE 3-8 SOLAR ELECTRIC CAPABILITY FOR JUPITER ORBITER MISSIONS USING SOLID RETRO SYSTEM.

and 45 days. These results assume the SEP jettison option and the solid retrosystem. Reference Missions No.2 and 4 have a net mass requirement of 641 kg and 705 kg, respectively, and each can be performed in a flight time of about 600 days. If desirable, the 641 kg planetology orbiter could be placed into a 7.5 day orbit for a flight time increase of only 70 days. It is seen that a considerable amount of excess capability is available as a tradeoff with flight time. For example, increasing the flight time to 800 days would allow 1080 kg to be placed into the 15 day orbit and 1270 kg into the 30 day orbit. Since an 800 day flight is probably acceptable for Jupiter missions, the excess capability could be used to enhance the mission science value (i.e., larger payload), or to provide a large margin of safety in designing the reference missions. Another important advantage of the excess capability is that it allows the SEP stage to be designed for a significantly lower than optimum power rating. This factor reduces size and cost of the SEP stage and, hence, improves acceptability by mission planners.

Figure 3-9 presents the SEP performance data in a different format where net mass is plotted as a function of orbit period for flight times of 600, 700, and 800 days. A comparison of the two retropropulsion systems shows that the space-storable retro provides a uniformly larger payload capability of 80-100 kg. Although this added mass is not insignificant, a comparison on a flight time basis shows a relatively small difference between the two systems. For example, the use of the space-storable retro for the 15 day planetology orbiter allows a flight time reduction from 610 days to 585 days. If flight time performance is dismissed as a major basis of comparison, then the selection of one

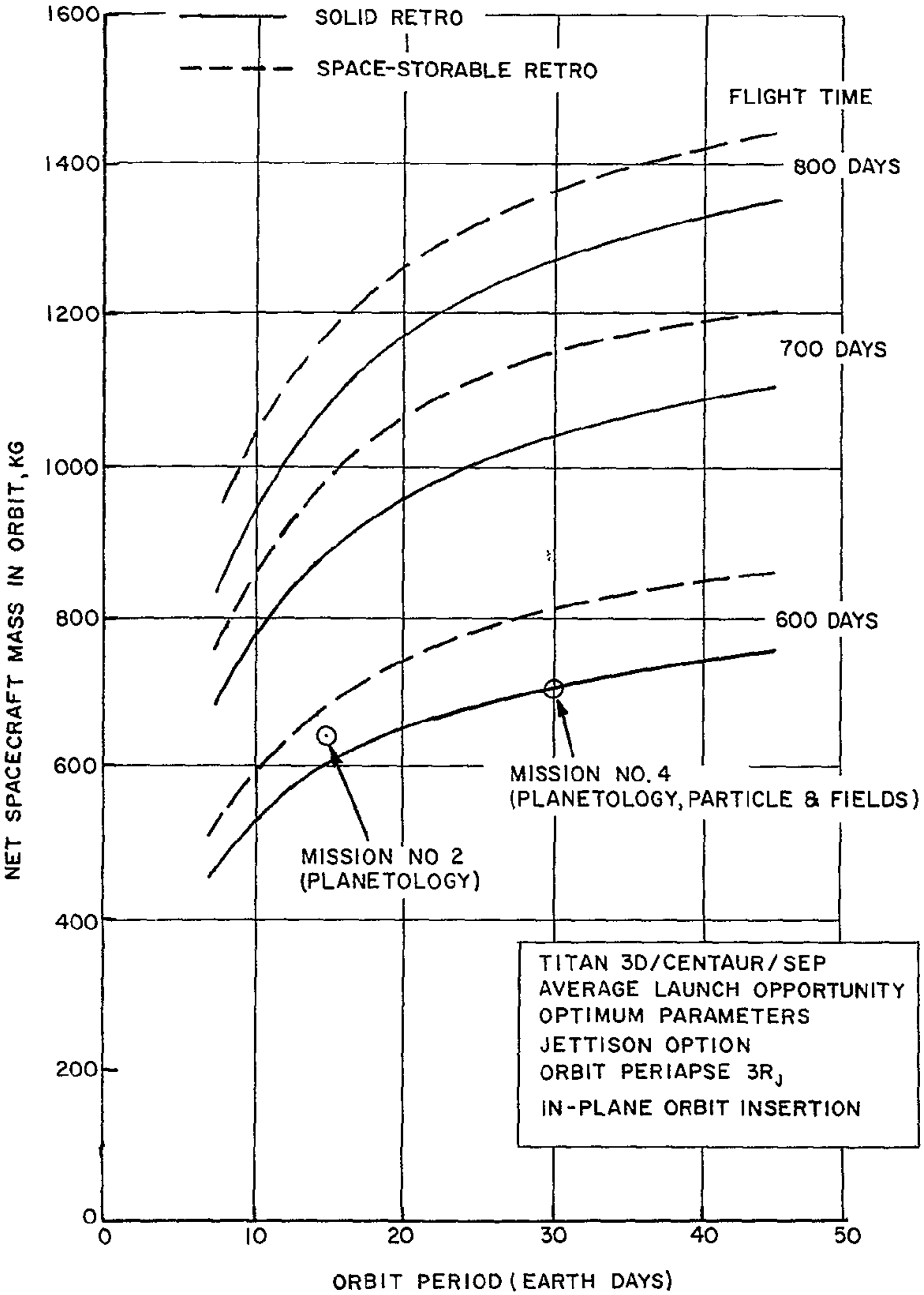


FIGURE 3-9. COMPARISON OF RETRO PROPULSION SYSTEMS FOR JUPITER ORBITER MISSIONS.

system over the other rests on a tradeoff between design simplicity and operational versatility. The simplicity and expected reliability advantages of the solid retro may be offset by its lack of multi-restart capability. Jupiter Reference Mission No's. 1 and 3 require several thrust maneuvers to establish an equatorial orbit (the plane change maneuver sequence will be described in Section 4). In addition to these major maneuvers, there is a requirement for one or two late mid-course guidance corrections and several orbit trim maneuvers. Use of the space-storable retro would obviate the need for an auxiliary propulsion system to perform these multiple maneuvers. Although the above remarks are indicative of the tradeoff considerations, there is little point in selecting one system over the other in this preliminary mission study. Both retrosystems will continue to be compared in the subsequent analysis.

3.2.5 Performance Penalty for Off-Optimum Design

Figure 3-10 illustrates the performance penalty incurred by allowing for a launch window up to 40 days duration. In examining the trajectory variations with launch date, it was found that the penalty is best described by an absolute net mass difference Δm_n rather than a percentage difference. The results of Figure 3-10 represent an average penalty over the launch opportunity cycle and the flight time range of interest. It should be noted, however, that the variation about this average is relatively small. A 10 day window incurs a negligibly small penalty of 4 kg. The penalty is only 50 kg for a 40 day window. Assuming the 40 day window, it is seen that the equivalent flight time penalty in delivering the

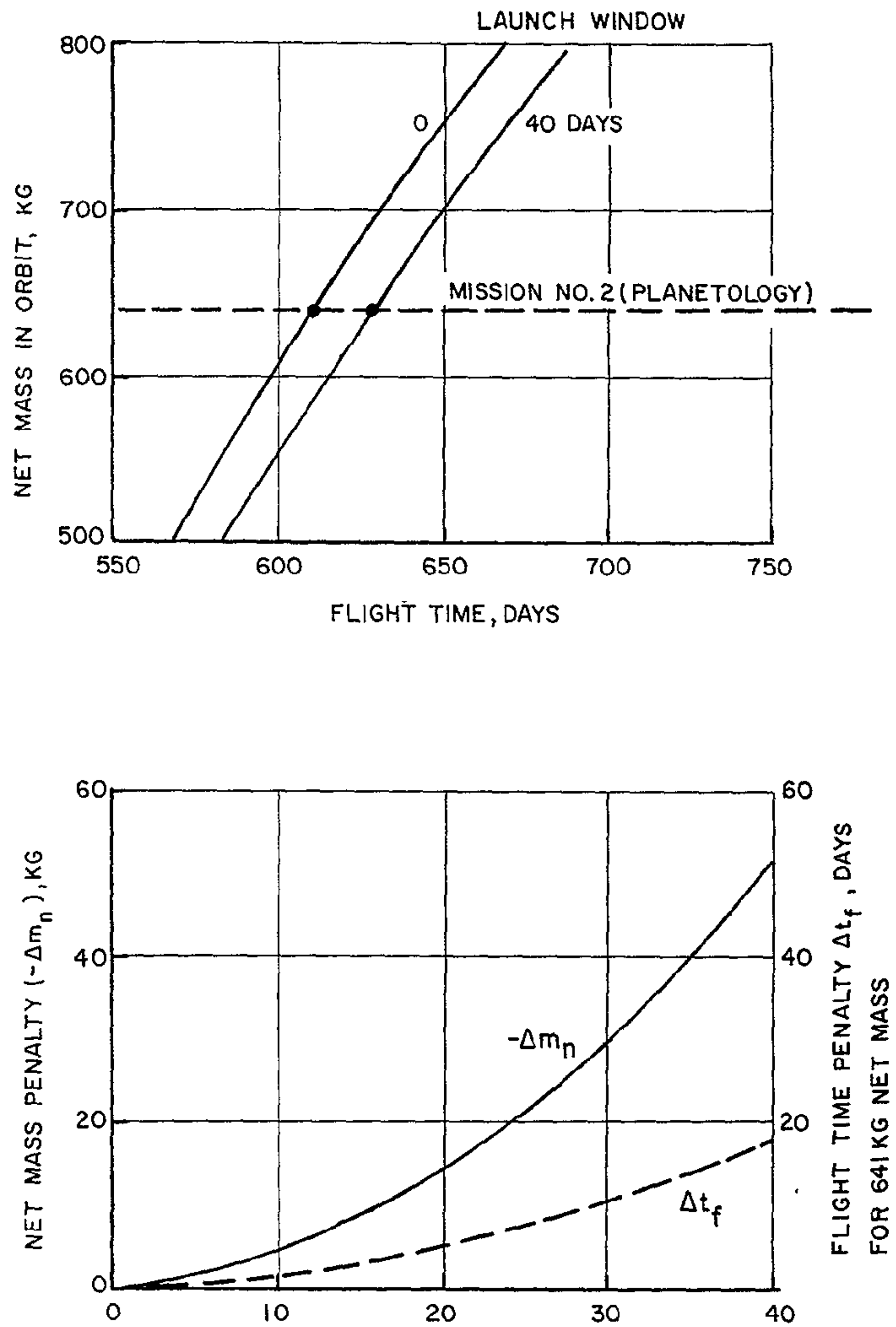


FIGURE 3-10. EFFECT OF FINITE LAUNCH WINDOW
FOR JUPITER ORBITER MISSIONS

641 kg planetology orbiter is only 18 days. These results, characteristic of the SEP flight mode, are in contrast to the ballistic mode where a large payload penalty is often incurred for launch windows greater than 15 days. Given the successful launch-on-time experience of current space missions, it could be argued that a large launch window allowance is not needed. Nevertheless, the SEP characteristic must certainly be viewed as a potential operational advantage.

The effect of designing the SEP spacecraft for off-optimum power and specific impulse is shown in Figure 3-11. Net mass in orbit is plotted as a function of power rating for a fixed specific impulse of 3500 seconds and flight times of 600 and 800 days. Values of optimum launch velocity vary with power rating and are indicated along the curves. Operation at the higher specific impulse incurs a minimum mass penalty of 30 - 40 kg (assuming optimum power). The penalty is fairly insensitive to power down to about one-half of optimum P_o where the mass loss is only 12 percent. Hence, if P_o were selected to be 15 kw, a flight time of about 625 days is required to perform the reference mission. This represents an insignificant flight time penalty to pay for the advantage of designing a smaller SEP stage.

3.3 Saturn Orbiter Missions

3.3.1 Selection of Direct SEP Trajectories

The difference between direct and indirect SEP trajectories was discussed in the introduction to Section 3. The question arises

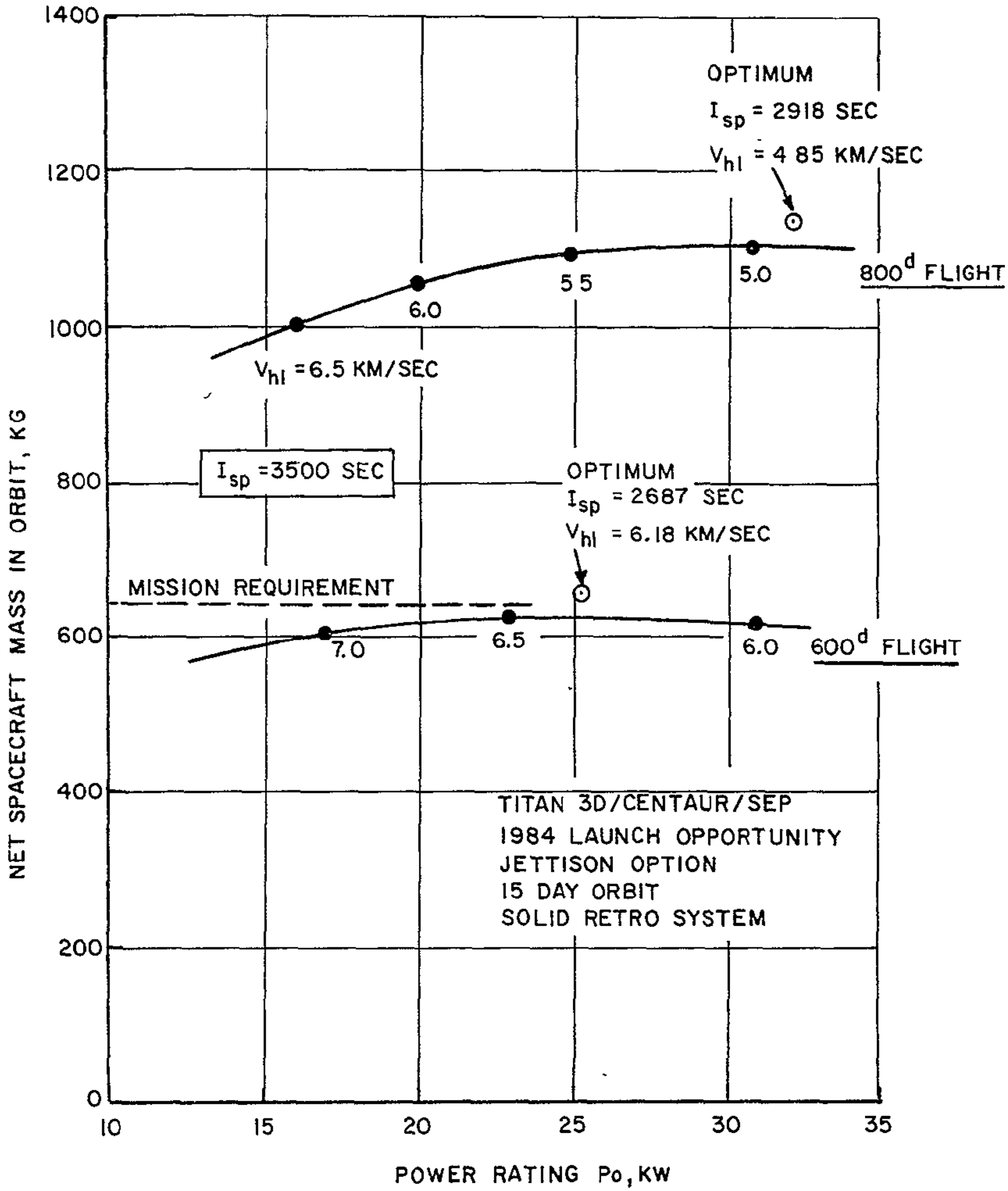


FIGURE 3-11. EFFECT OF OFF-OPTIMUM POWER AND SPECIFIC IMPULSE FOR JUPITER ORBITER MISSIONS.

as to which flight mode is most appropriate for Saturn orbiter missions. Figure 3-12 compares the two flight mode options in terms of net mass capability versus flight time for the 15-day orbit at Saturn. Characteristically, the indirect mode can provide a larger payload beyond a certain flight time. The cross-over point occurs at a flight time of 1960 days an an orbiter mass of 940 kg. Since the nominal net mass requirement is about 640 kg, it is concluded that the indirect mode is not needed for Saturn missions (this conclusion is essentially independent of the orbit size and type of retro stage). The direct mode flight time for this mass requirement is about 1500 days for an average launch opportunity assuming optimum SEP parameters and the solid retrosystem. Another important reason for selecting the direct mode is that it is relatively easier to mechanize the thrust direction program when the transfer trajectory is less than one revolution about the sun, i.e., the thrust direction has a much smaller variation with respect to the sun line.

3.3.2 Launch Opportunities

Optimum launch dates for Saturn mission occur approximately 12 months apart. The launch opportunity cycle is about 30 years commensurate with Saturn's orbital period. Figure 3-13 shows the variation of net mass over one-third of the opportunity cycle, i.e., between 1979 and 1990 which is the time period of interest. Data is given for fixed flight times of 1280, 1480 and 1880 days. Note that launches in the 1980's are associated with less than average payload capability (the average is given by the dotted line curves).

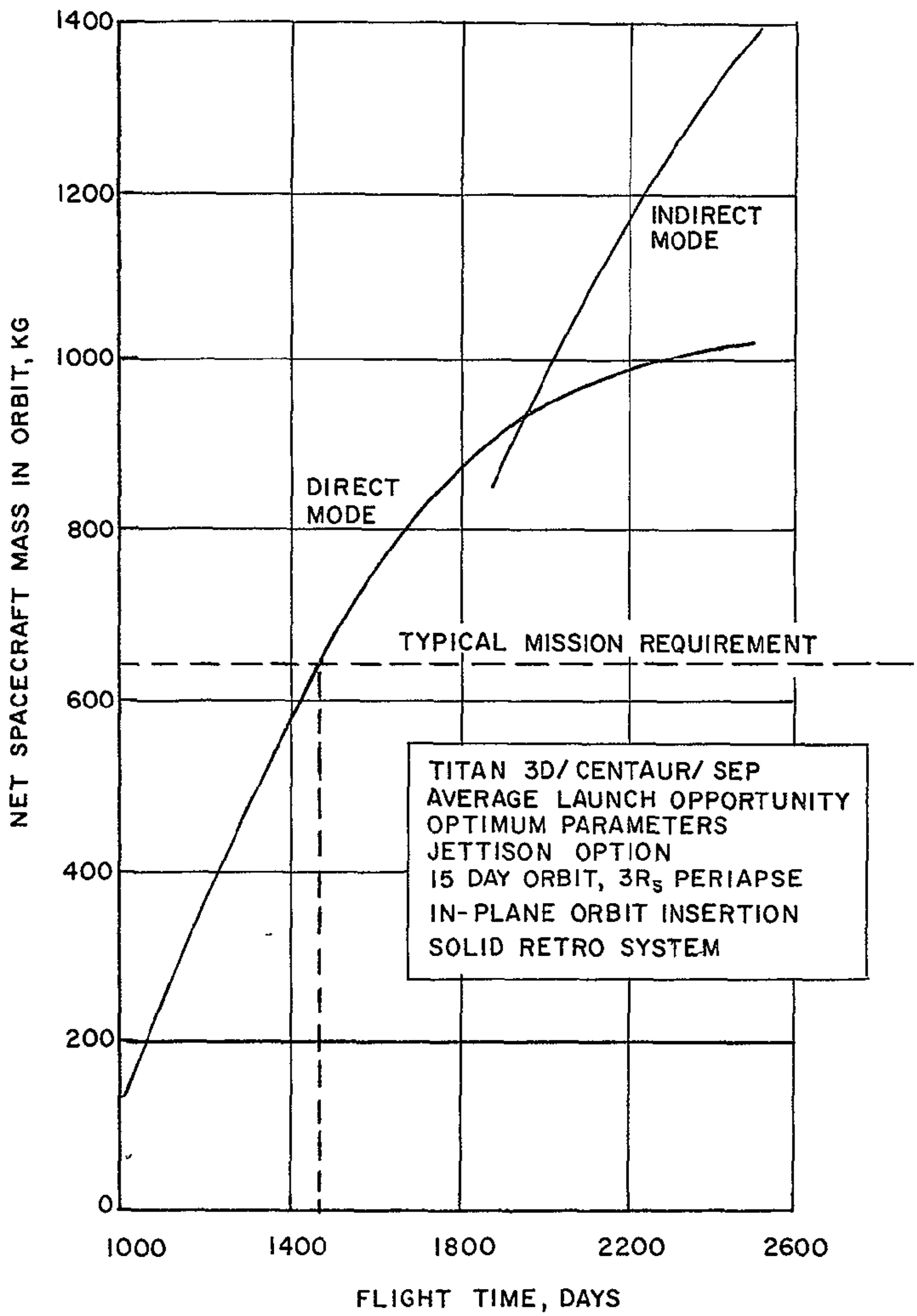


FIGURE 3-12. PERFORMANCE COMPARISON OF DIRECT AND INDIRECT SEP TRAJECTORIES TO SATURN.

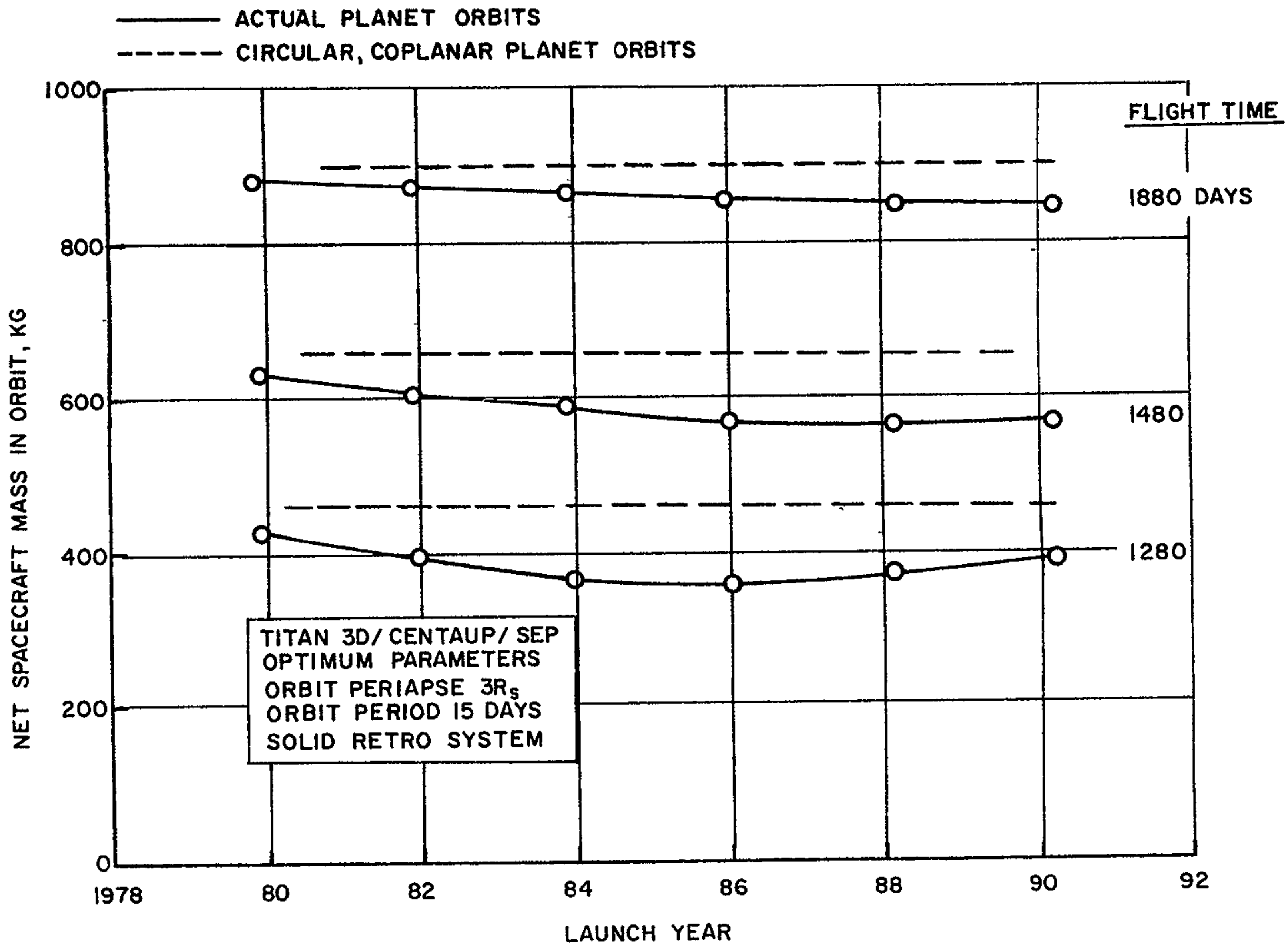


FIGURE 3-13. EFFECT OF LAUNCH OPPORTUNITY ON SOLAR ELECTRIC CAPABILITY FOR SATURN ORBITER MISSION.

The 1985-86 opportunity will be taken as an example to illustrate the general characteristics and tradeoffs for Saturn missions. The flight time range of interest is 1400 - 1800 days for the candidate mission payload requirements. Figure 3-14 illustrates the heliocentric trajectory for a 1680 day flight. Thrust cutoff occurs at about 4 AU, 390 days after launch.

3.3.3 Optimum SEP Flight Parameters

Figure 3-15 shows the variation of optimum power, specific impulse, and hyperbolic launch and approach velocity as a function of flight time to Saturn. For flight times between 1400 and 1800 days, power and specific impulse vary over the narrow range 2775 - 2875 sec and 21.8 - 26.3 kw, respectively. Similarly, the optimum values of launch and approach velocities vary between 6.3 - 7.0 km/sec and 6.0 - 8.5 km/sec, respectively. The orbit capture ΔV is, of course, much smaller than the hyperbolic approach velocity. Taking the 15-day orbit (Mission No. 2) as an example, the ΔV requirement decreases from 2.45 km/sec to 1.65 km/sec over the flight time range 1400 - 1800 days.

3.3.4 Effect of Orbit Selection and Retrosystem

Net spacecraft mass is plotted as a function of flight time in Figure 3-16 for Saturn orbit periods of 7.5, 15 and 45 days. These results assume the solid retrosystem and an in-plane orbit insertion maneuver at a periapse distance of 3 Saturn radii (15^d, 45^d orbits) or 1.1 Saturn radii (7.5^d orbit). The fact that the retro ΔV requirement decreases with periapse distance accounts

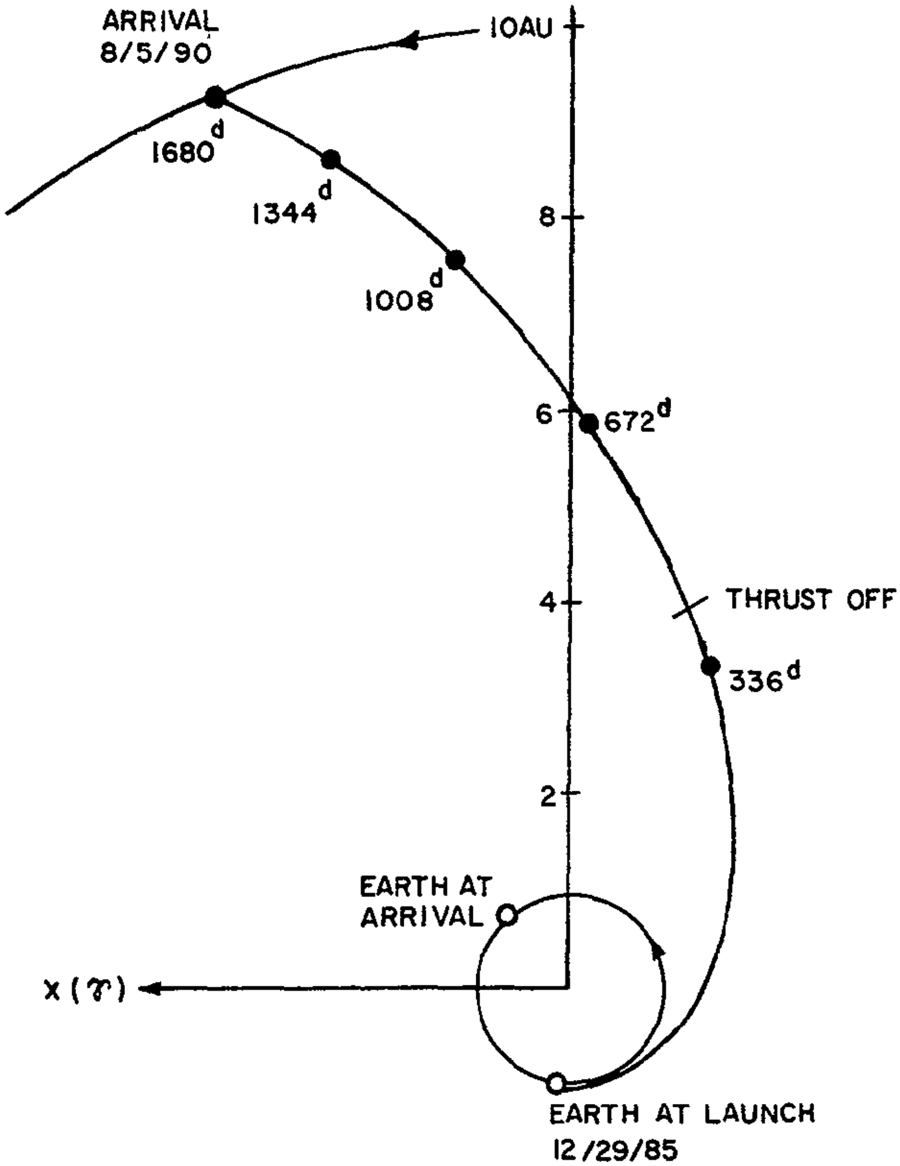


FIGURE 3-14. 1680 DAY SOLAR ELECTRIC TRAJECTORY TO SATURN

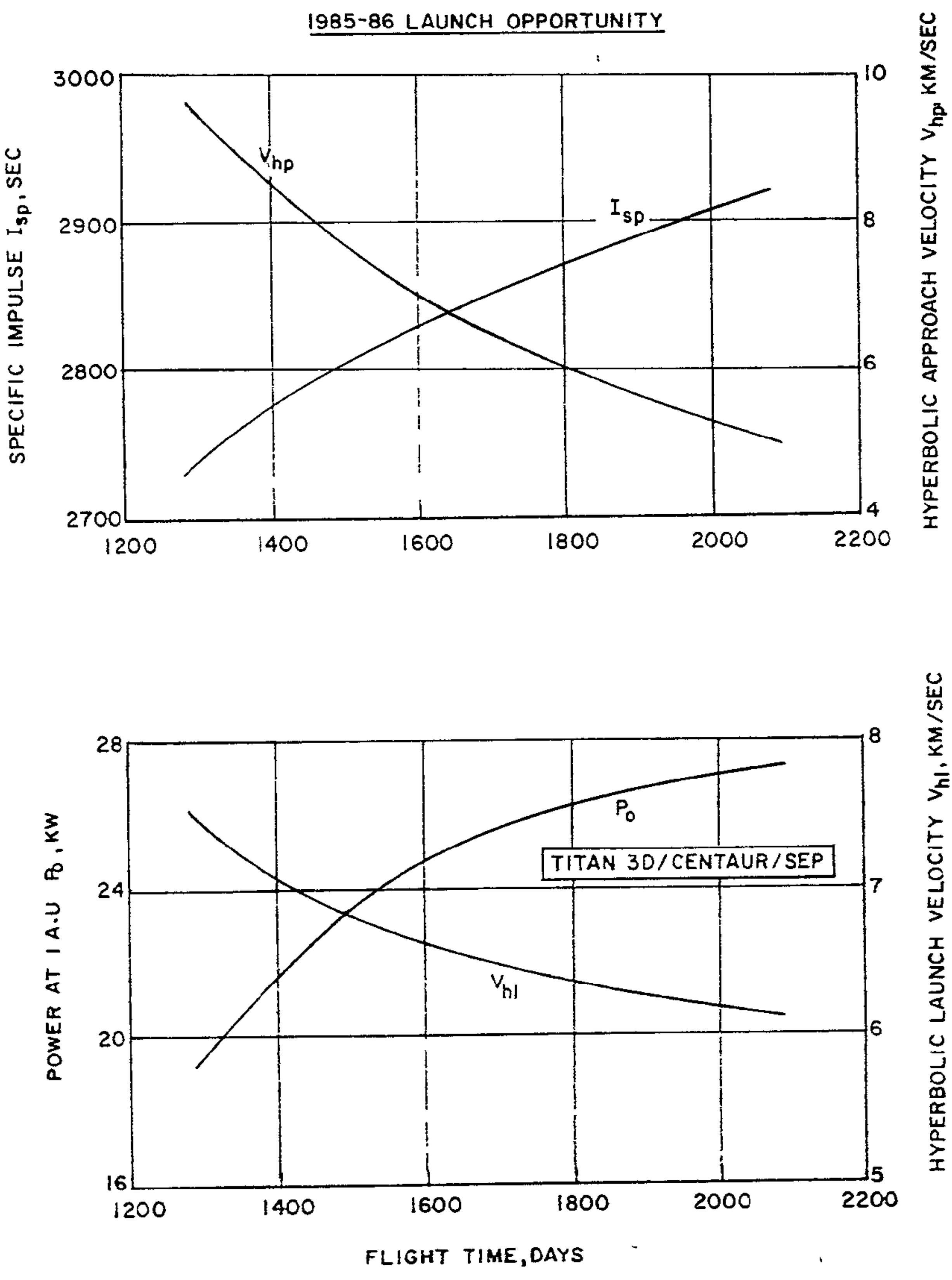


FIGURE 3-15. OPTIMUM FLIGHT PARAMETERS (P_0, I_{sp}, V_{hl}, V_{hp}) FOR SATURN ORBITER MISSIONS.

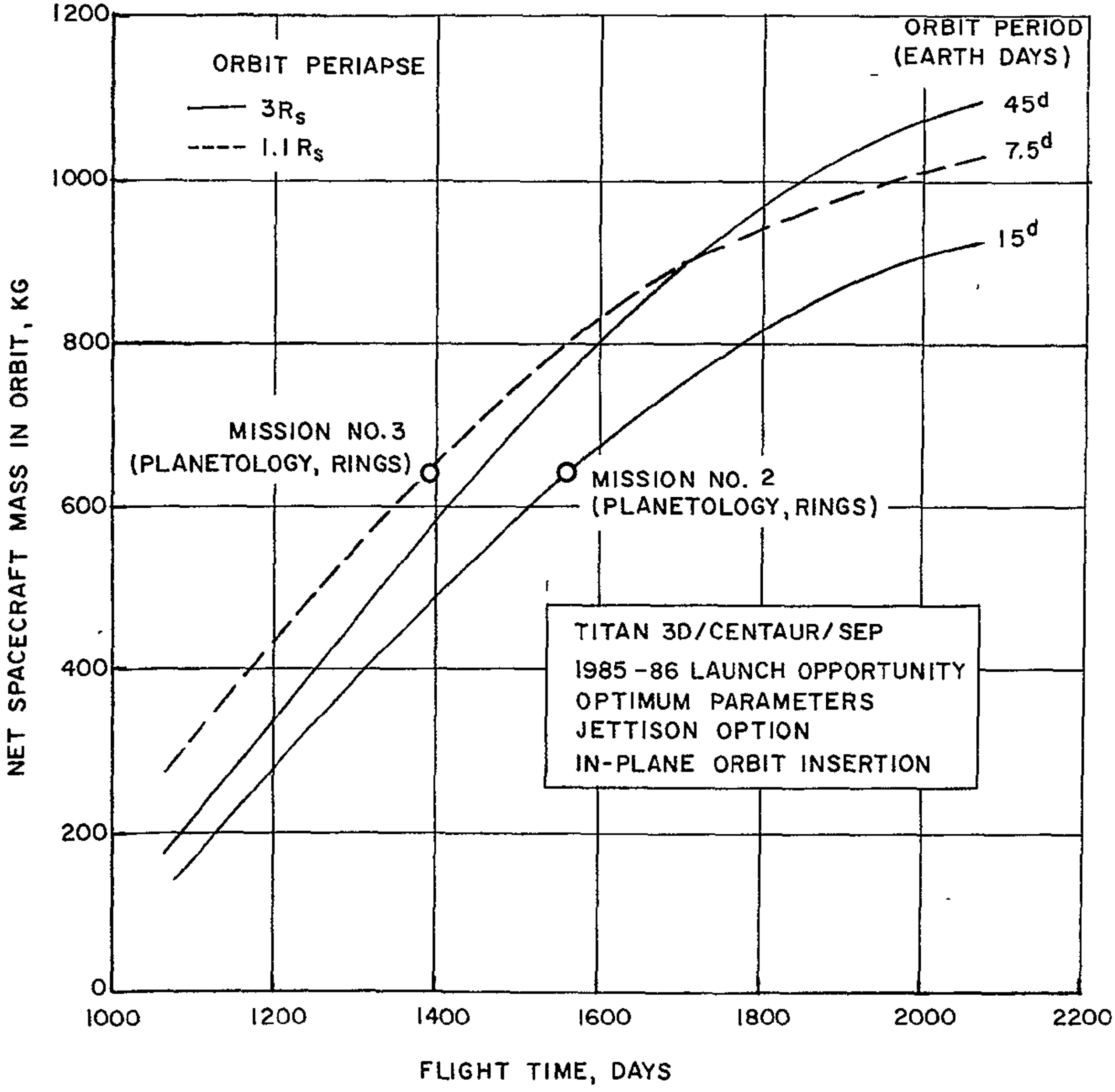


FIGURE 3-16. SOLAR ELECTRIC CAPABILITY FOR SATURN ORBITER MISSIONS USING SOLID RETRO SYSTEM

for the improved capability of performing Mission No. 3 (7.5^d orbit) relative to Mission No. 2 (15^d orbit); the flight time difference for the same mass requirement is about 170 days.

A comparison of the two retropropulsion systems in Figure 3-17 shows that the space-storable retro provides an increased payload capability of about 70 kg for a given flight time. However, as in the case of Jupiter missions, this mass difference can be made up by a small increase in flight time. For Mission No. 2 this increase is only 70 days.

3.3.5 Performance Penalty for Off-Optimum Design

Figure 3-18, shows the mass penalty incurred by allowing for a finite launch window and its affect on the net mass versus flight time characteristic of Candidate Mission No. 2. A 40-day window results in an average mass reduction of 55 kg -- about the same as in the Jupiter mission examples. The equivalent flight time penalty in delivering the 642 kg planetology orbiter is about 70 days.

The performance penalty due to off-optimum power and specifi impulse operation is illustrated in Figure 3-19 for Saturn Candidate Mission No. 2. Optimum SEP parameters for the 1680-day flight are P_o = 25.6 kw, I_{sp} = 2851 sec and ${}^{V}h_1$ = 6.46 km/sec; these yield a maximum net mass in orbit of 737 kg. If I_{sp} is increased to 3500 sec, a minimum penalty of only 7 kg is incurred by operating at a higher power rating (31 kw) and lower launch velocity (6 km/sec). However, at a lower than optimum power the

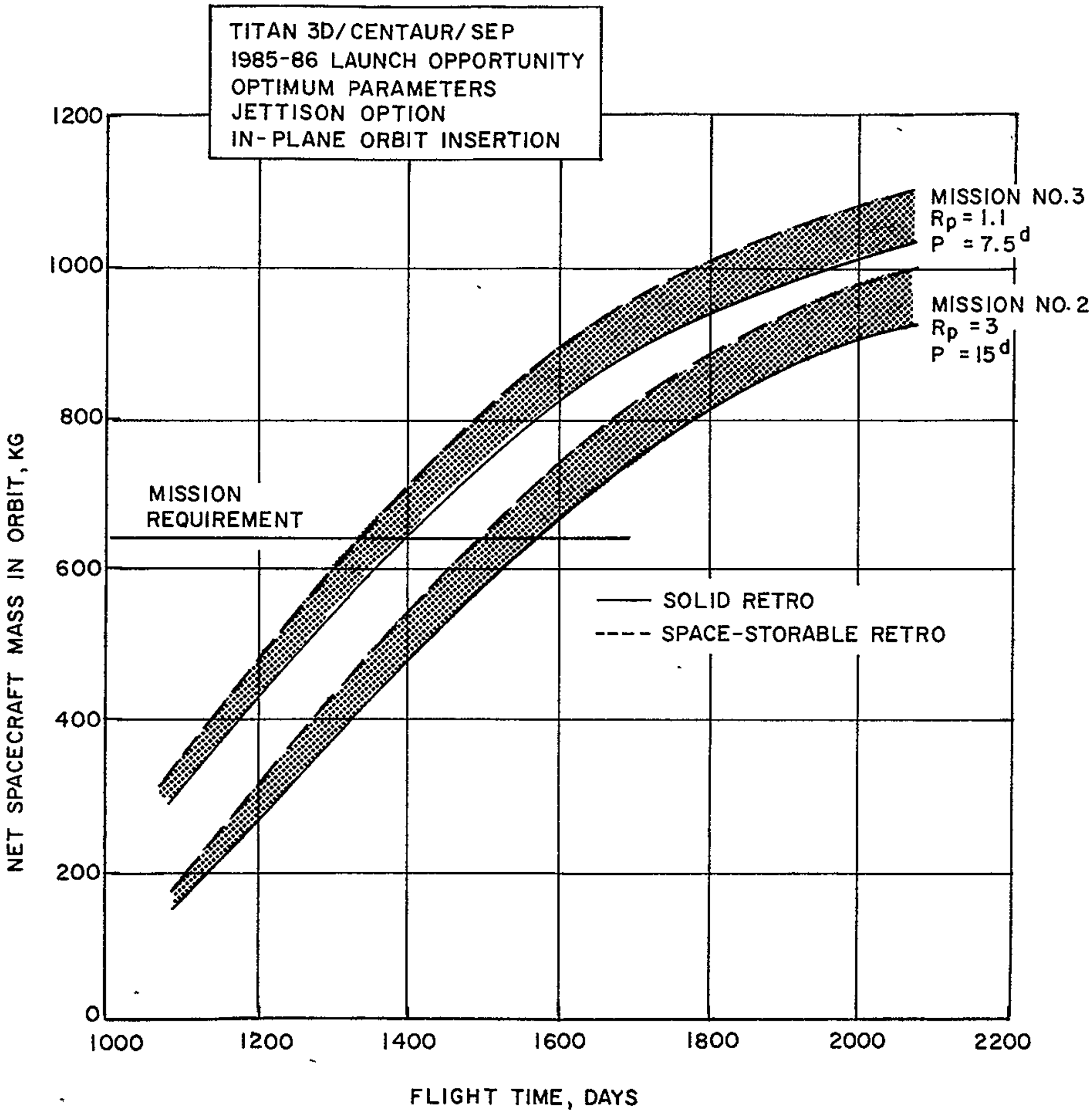


FIGURE 3-17. COMPARISON OF RETRO PROPULSION SYSTEMS FOR SATURN ORBITER MISSIONS.

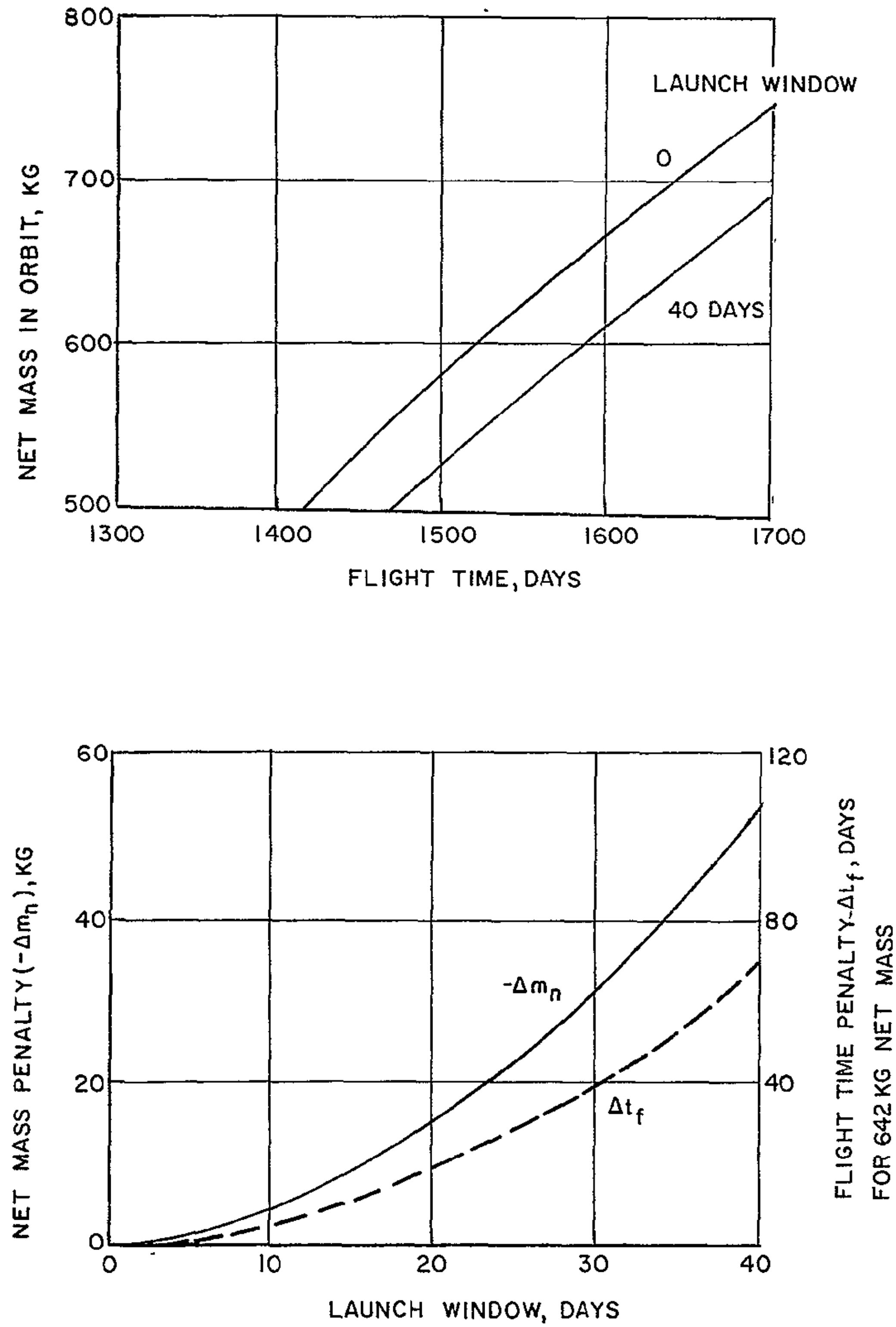


FIGURE 3-18. EFFECT OF FINITE LAUNCH WINDOW FOR SATURN ORBITER MISSIONS.

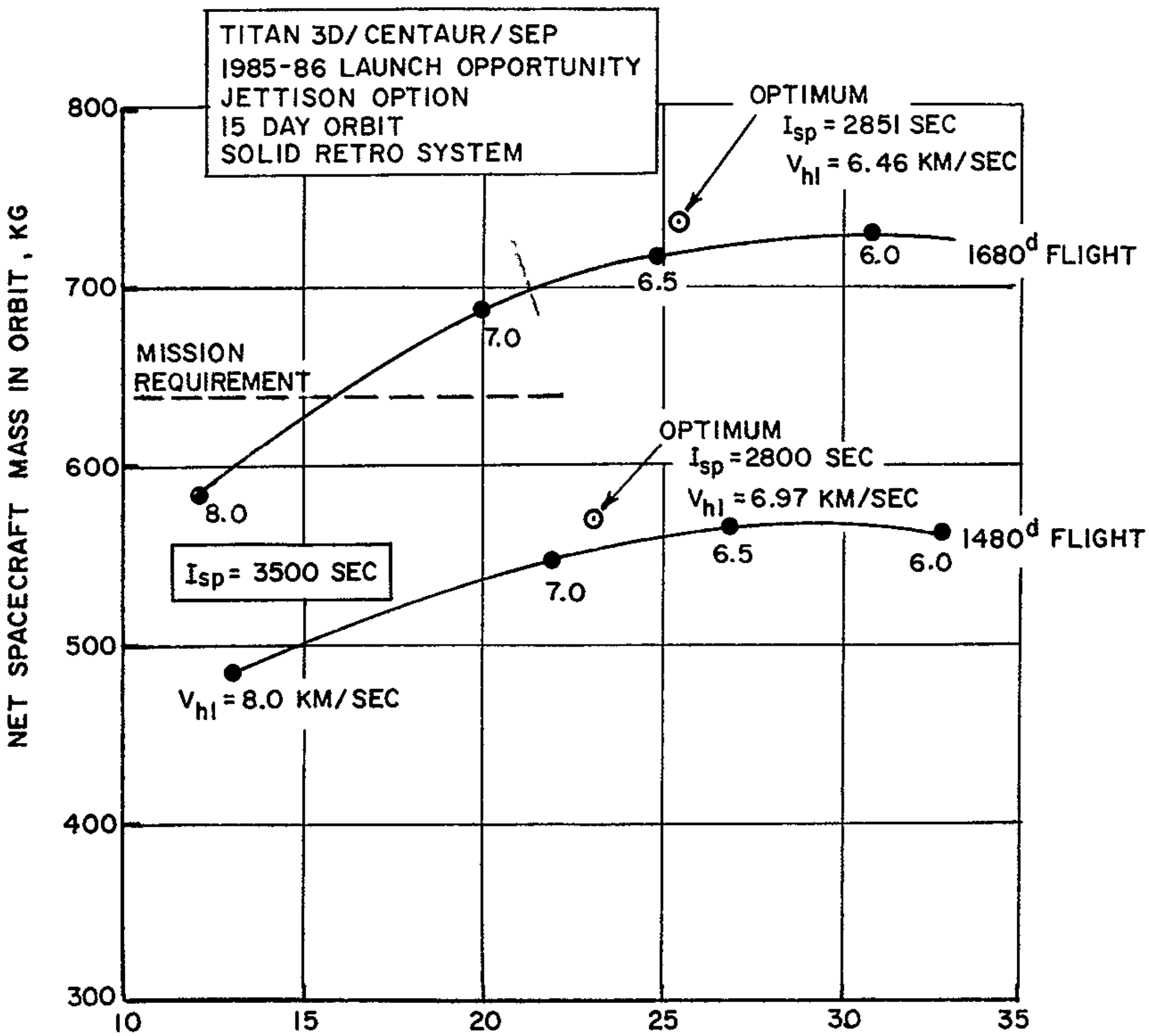


FIGURE 3-19. EFFECT OF OFF-OPTIMUM POWER AND SPECIFIC IMPULSE FOR SATURN ORBITER MISSIONS.

launch velocity must increase to partially compensate for the loss in electric propulsion capability. The mass penalties for operation at 20 kw and 15 kw are, respectively, 47 kg (6.5 percent) and 107 kg (14.5 percent). It is seen that the orbiter mission requirement of 642 kg can be delivered on a 1700-day flight using a 15 kw - 3500 sec SEP stage. Hence, the flight time penalty relative to the optimum SEP stage is about 140 days, or a 9 percent increase in flight time.

3.4 Common SEP Stage for Jupiter and Saturn Missions

Jupiter and Saturn orbiters may be considered together as a class of outer planet exploration missions with potential application in the 1975-90 time period. The desirability of designing a fixed SEP stage which may be utilized for both missions is assumed. A common design is enhanced by the fact that the jettison option appears to be the best choice for each mission.

Off-optimum design performance for Jupiter and Saturn orbiters has been illustrated previously by Figures 3-11 and 3-19. Assuming thruster operation at 3500 sec specific impulse, these results are summarized below in terms of the percent net mass penalty as a function of design power rating.

P_o	Jupiter Orbiter		Saturn Orbiter	
(I_{sp} = 3500 sec)	600^d	800^d	1480^d	1680^d
25 kw	-1.4%	-3.8%	-1.4%	-2.4%
20 kw	-5.2%	-7.6%	-5.8%	-6.5%
15 kw	-9.1%	-13.1%	-12.0%	-14.5%

The 15 kw power plant represents an off-optimum power design of 50 - 60 percent, yet incurs a net mass penalty less than 15 percent. For purposes of the present analysis, we will select the 15 kw SEP stage for common application to both Jupiter and Saturn missions. In the comparison of SEP and ballistic capabilities presented in Section 4, the optimum SEP net mass data will be reduced uniformly by 15 percent. This reduction should also take into account the penalty due to allowing a finite launch window of about 20 days. One important factor in the power selection is that current SEP studies have tended to focus on a relatively small power rating of about 15 kw for multi-mission applications. Such a system matched to the Titan 3D/Centaur launch vehicle could have a wide capability envelope for performing such missions as outer planet flybys and orbiters, Mercury orbiter, comet and asteroid rendezvous, 0.1 AU solar probe, and out-of-ecliptic flights.

4. COMPARISON OF SOLAR ELECTRIC AND BALLISTIC FLIGHT MODES

The preceeding discussion has illustrated the basic characteristics, design tradeoff options, and capabilities of SEP spacecraft for accomplishing orbiter missions at Jupiter and Saturn. Performance potential was shown to be high even for significantly off-optimum SEP stage design. With the mission planning task in mind, it is recognized that the SEP flight mode must be viewed as a competitor or an alternative to the usual ballistic delivery systems. Assuming the desirability of outer planet orbiters as part of an overall space exploration program, the program planner may have to decide between two alternative development routes: (1) develop a SEP upper stage with multi-mission capability, or (2) develop a high energy chemical upper stage and/or modify current Titan class launch vehicle designs. Such a decision will rest on the dual criteria of mission capability and overall cost effectiveness. Our purpose in this section is to compare the two alternatives on the basis of capability alone. It will be convenient to discuss both Jupiter and Saturn missions together rather than separately as in the previous section.

4.1 Ballistic Launch Requirements and Injected Mass

Analysis of ballistic mission applications assumes the following set of launch vehicle/upper stage combinations listed in order of increasing propulsion capability:

Titan 3D/Centaur — Jupiter Applications
Titan 3D/Centaur/BII
Titan 3D(7)/Centaur
Titan 3D(7)/Centaur/BII — Saturn Applications
Titan 3D/Centaur/HFK(10)
Titan 3D(7)/Centaur/HFK(15)

Of the above vehicle combinations, only the Titan 3D/Centaur is presently being developed for a specific flight program (Viking). The Burner II is under development as an upper stage for the Atlas/Centaur and its matching to the Titan 3D/Centaur has been proposed for the Grand Tour missions. The Titan 3D(7) is a proposed modification of the Titan 3D which employs two 7-segment, 120-in.-diameter, solid motor strapons replacing the basic 5-segment units. Finally, two proposed versions of a hydrogen-flourine kick stage (HFK) having propellant loadings of 10,000 lbs. and 15,000 lbs., respectively, are considered as high energy upper stage examples.

Table 4-1 shows the launch velocity requirements for 760-day transfers to Jupiter in each of the eleven launch years from 1975 to 1986. The characteristic velocities (V_c) are representative of a 10-day launch window and a 36° constraint on the departure asymptote declination, the latter being a common assumption for purposes of range safety. Also listed in the table are the injected mass capabilities of the four launch vehicles considered for Jupiter applications. Launch vehicle performance data are taken from the 1970 OSSA Estimating Factors Handbook. It is noted that 1978 and 1985 are particularly poor ballistic launch years

TABLE 4-1

BALLISTIC LAUNCH REQUIREMENTS AND INJECTED MASS FOR JUPITER MISSIONS

760^d FLIGHT, 10^d LAUNCH WINDOW

LAUNCH DATE (FIRST DAY)	LAUNCH CHARACTERISTIC VELOCITY Vc (FT/SEC)	INJECTED MASS CAPABILITY (KG)			
		TITAN 3D/ CENT	TITAN 3D/ CENT/BII	TITAN 3D(7)/ CENT	TITAN 3D(7)/ CENT/BII
6/20/75	47020	1110	1340	1750	1810
7/24/76	47215	1090	1290	1720	1770
8/31/77	47840	910	1180	1520	1610
10/11/78	49940	500	860	1000	1200
11/4/79	48160	860	1130	1450	1540
11/30/80	47530	1000	1250	1610	1700
12/28/81	47820	910	1180	1520	1610
1/28/83	46600	1200	1400	1880	1900
3/13/84	47940	890	1160	1500	1590
4/24/85	48860	680	1000	1250	1380
6/1/86	47325	1040	1270	1650	1720

due mainly to the declination constraint. SEP mission performance is less sensitive to this constraint since velocity direction changes are easily accommodated by continuous thrusting. The best ballistic opportunity in the cycle occurs in 1983 and provides an interplanetary payload 1.5 to 2.5 times the 1978 payload capability. It will be recalled that SEP missions have a characteristically smaller variation over the launch opportunity cycle.

Ballistic velocity requirements and launch vehicle performance for the Saturn mission are shown in Table 4-2. Data is given for seven launch years from 1979 to 1986 assuming a 1680-day (4.6 years) trajectory to Saturn. Payload capability improves steadily during this period with the 1986 opportunity providing about 1.5 times the 1979 injected mass. This is a result of the decreasing inclination change required to intercept Saturn over the corresponding range of arrival dates even though Saturn's orbit distance is increasing during this period. In the solar electric case the inclination change is easily accomplished via low thrust maneuvers and the increasing intercept distance causes a relatively small decrease in capability from 1979 to 1986 (see Figure 3-13).

4.2 Planet Approach Conditions and Equatorial Orbits

Table 4-3 compares SEP and ballistic approach conditions at Jupiter for nearly equivalent flight time trajectories (~ 2.1 years). The approach asymptote is described here by the hyperbolic speed (V_{hp}), the approach path angle relative to the solar direction (ξ), and the declination of the asymptote relative to Jupiter's

TABLE 4-2

BALLISTIC LAUNCH REQUIREMENTS AND INJECTED MASS FOR SATURN MISSIONS

1680^d FLIGHT, 10^d LAUNCH WINDOW

LAUNCH DATE (FIRST DAY)	LAUNCH CHARACTERISTIC VELOCITY Vc (FT/SEC)	INJECTED MASS CAPABILITY (KG)		
		TITAN 3D(7)/ CENT/BII	TITAN 3D/ CENT/HFK(10)	TITAN 3D(7)/ CENT/HFK(15)
11/28/79	52974	770	1070	1360
12/1/80	52547	820	1110	1450
12/16/81	52123	860	1180	1540
12/21/82	51426	970	1290	1670
1/5/84	50940	1040	1360	1790
1/9/85	50462	1110	1450	1900
1/25/86	50055	1180	1540	1990

TABLE 4-3

APPROACH CONDITIONS FOR JUPITER ORBITER MISSIONS

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (800^d FLIGHT)				BALLISTIC (760^d FLIGHT)			
	V_{hp} Km/sec	ξ Deg	δ_{eq} Deg	A (R_P=3R_J) Deg	V_{hp} Km/sec	ξ Deg	δ_{eq} Deg	A (R_P=3R_J) Deg
1975	5.48	126.4	-5.23	17.27	6.40	116.9	-1.88	5.32
1976					6.48	121.8	-5.79	16.37
1977	6.28	133.3	-3.39	9.79	6.67	125.9	-6.51	17.98
1978					6.92		-4.69	12.44
1979	6.81	137.5	2.20	5.89	7.20	131.5	-2.17	5.53
1980					7.28	131.7	0.48	1.21
1981	6.80	135.4	5.40	14.58	7.26	129.8	3.12	7.90
1982-83					7.12	125.4	5.99	15.55
1984	6.26	129.1	2.67	7.72	6.88	120.2	6.78	18.22
1985					6.66	115.7	5.18	14.25
1986	5.83	125.2	-3.67	11.36	6.42	114.8	2.45	6.92

V_{hp} = Hyperbolic approach velocity

ξ = Angle between sun vector and V_{hp} vector

δ_{eq} = Declination of V_{hp} vector to Jupiter's equator plane

A = Plane change required for equatorial orbits equator plane (δ_{eq}). Several of these approach parameters are illustrated in Figure 4-1. Smaller values of V_{hp} for SEP missions will result in somewhat lower ΔV requirements for in-plane capture. Differences in ξ result in the solar terminator position (on the first orbit) being displaced about 10° closer to periapse in the ballistic case. This should have little effect on long-term mission operations since the terminator position will change by about 40° after one year in orbit due to Jupiter's orbital motion about the sun. The most significant difference between SEP and ballistic missions lies in the declination angle δ_{eq} which has an important effect on the ΔV requirements for equatorial orbits. Any desired orbit having an inclination less than δ_{eq} requires a plane change maneuver. Equatorial orbits can be established by making a plane change in the amount A listed in Table 4-3. The value of A depends mainly on δ_{eq}, but it is also a weak function of the orbit periapse distance R_p. For example, Jupiter Mission No. 4 has a periapse of 2.29 R_j and would require a plane change of 1-2 degrees larger than the values listed in the table. In comparing SEP and ballistic plane change requirements over the launch opportunity cycle it is seen that the overall variation in magnitude is similar but the two cyclic characteristics are not in phase. Hence, in any given launch year, there is likely to be a large difference in A. For example, in 1975 the SEP plane change is 12° larger than the ballistic case, but in 1984 it is about 10° smaller.

Unless δ_{eq} is equal to zero, equatorial orbits cannot be established with a single coplanar periapse capture maneuver. A

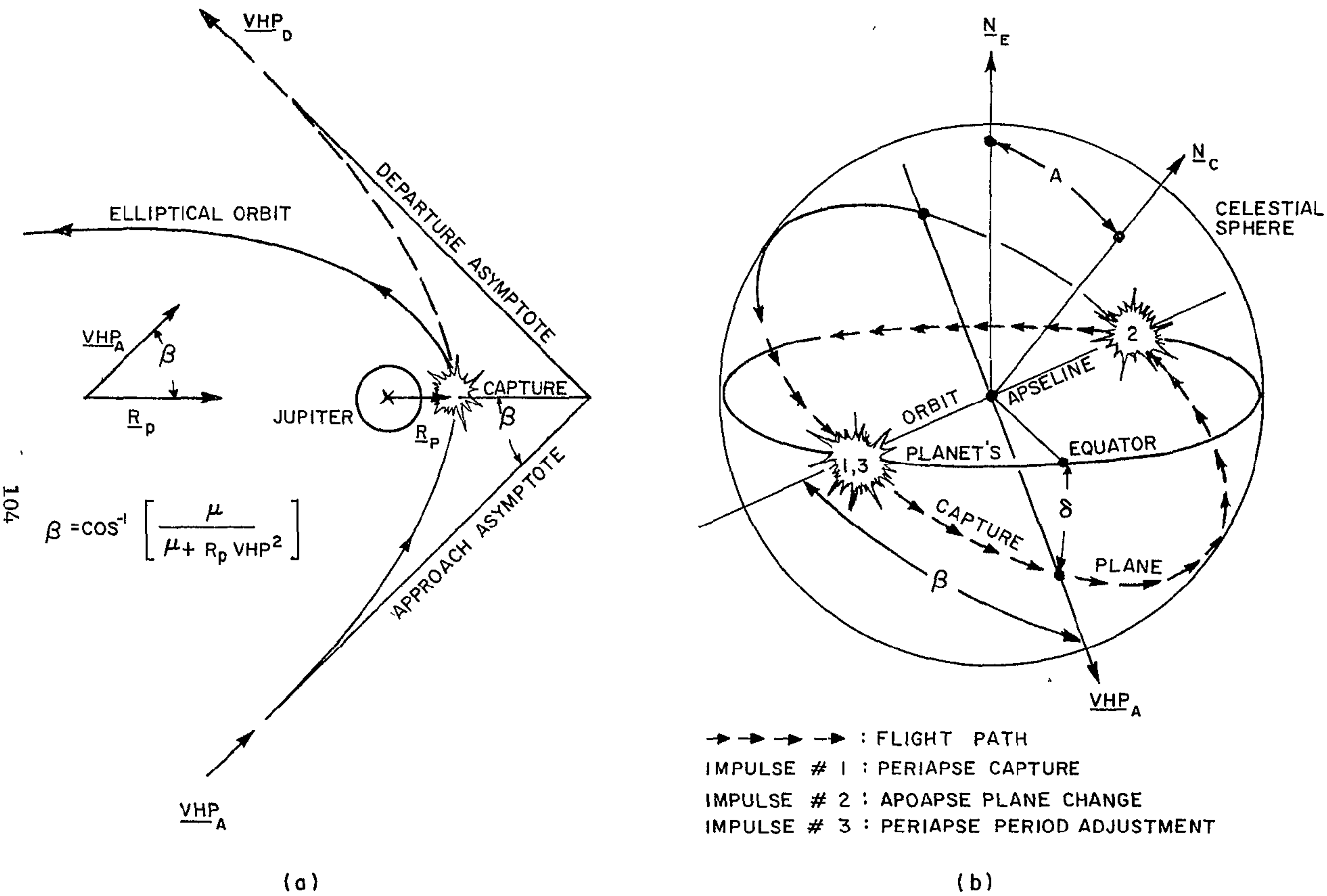


FIGURE 4-1. ORBITAL GEOMETRY FOR CAPTURE AND PLANE-CHANGE MANEUVERS.

sequence of three impulses illustrated in Figure 4-1 has been used in the analysis to determine the added retro velocity requirement of equatorial orbits. The first impulse is a coplanar periapse capture into a loose elliptical orbit, the impulse being constrained to occur at the intersection of the approach plane and equatorial plane. A second impulse, occurring at the apoapse distance of the initial orbit, accomplishes the plane change maneuver. A third impulse is then added at the next periapse pass to reduce the orbit period to the desired value. Clearly, for the same periapse distance and final orbit period, the sum of ΔV_1 and ΔV_3 is equal to the single coplanar capture ΔV required for inclined orbits.

Table 4-4 gives the added velocity increment (ΔV_2) needed to establish an equatorial orbit for both SEP and ballistic missions. A 600-day SEP flight is also shown for purposes of comparison. The 1975 SEP opportunity yields the largest values of ΔV_2; 177m/sec and 308 m/sec, respectively, for the 600 and 800-day flights. The largest ΔV_2 for the 760-day ballistic mission is 325 m/sec and is associated with the 1984 opportunity.

Tables 4-5 and 4-6 present information on approach conditions and equatorial orbit requirements for Saturn missions. The main result of an overview of this data is that the plane change requirement for equatorial orbits increases monotonically over the period of launch opportunities considered. Also, for the same flight time, the SEP missions require a larger plane change maneuve than ballistic missions after 1980 -- particularly, the 1985-86 opportunity. In the case of ballistic missions, the increasing ΔV_2

TABLE 4-4

ADDITIONAL VELOCITY INCREMENT FOR JUPITER EQUATORIAL ORBITS

$\Delta V_2 = \Delta V(EQ) - \Delta V(INC)$, KM/SEC

LAUNCH	SOLAR ELECTRIC		BALLISTIC
OPPORTUNITY	600^d FLIGHT	800^d FLIGHT	760^d FLIGHT
1975	0.177	0.308	0.095
1976			0.292
1977	0.149	0.175	0.320
1978			0.222
1979	0.039	0.106	0.099
1980			0.022
1981	0.170	0.261	0.141
1982-83			0.277
1984	0.129	0.138	0.325
1985			0.254
1986	0.099	0.203	0.124

$\Delta V(INC)$ is a single periapse capture impulse for inclined orbits

ΔV (EQ) is a three impulse capture sequence for equatorial orbits

1) ΔV_1 - periapse capture into 45^d orbit

2) ΔV_2 - plane change (A) at apoapse ($99R_J$)

3) ΔV_3 - periapse period change to final orbit

TABLE 4-5

APPROACH CONDITIONS FOR SATURN ORBITER MISSIONS

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (1680^d FLIGHT)				BALLISTIC (1680^d FLIGHT)			
	V_{hp} Km/sec	ξ Deg	δ_{eq} Deg	A ($R_p=3R_S$) Deg	V_{hp} Km/sec	ξ Deg	δ_{eq} Deg	A ($R_p=3R_S$) Deg
1979	6.06	130.5	-0.27	0.51	6.17	120.9	1.20	2.25
1980					6.23	122.6	-5.30	9.90
1981	6.27	133.3	-11.00	20.71	6.31	124.6	-9.70	18.10
1982-83					6.37	125.7	-15.20	28.68
1983-84	6.44	135.2	-19.17	36.59	6.40	126.5	-18.20	34.73
1984-85					6.67	129.6	-21.70	40.83
1985-86	6.54	135.6	-28.10	57.69	6.70	129.7	-23.10	43.74

V_{hp} = Hyperbolic approach velocity

ξ = Angle between sun vector and V_{hp} vector

δ_{eq} = Declination of V_{hp} vector to Saturn's equator plane

A = Plane change required for equatorial orbits

TABLE 4-6

ADDITIONAL VELOCITY INCREMENT FOR SATURN EQUATORIAL ORBITS

$\Delta V_2 = \Delta V(EQ) - \Delta V(INC)$, KM/SEC

LAUNCH OPPORTUNITY	SOLAR ELECTRIC		BALLISTIC
	1480^d FLIGHT	1680^d FLIGHT	1680^d FLIGHT
1979	0.044	0.007	0.030
1980			0.133
1981	0.269	0.278	0.243
1982-83			0.383
1983-84	0.436	0.485	0.461
1984-85			0.539
1985-86	0.524	0.745	0.575

ΔV (INC) is a single periapse capture impulse for inclined orbits

ΔV (EQ) is a three impulse capture sequence for equatorial orbits

1) ΔV_1 - periapse capture into 45^d orbit
2) ΔV_2 - plane change (A) at apoapse ($78R_s$)
3) ΔV_3 - periapse period change to final orbit

requirement with launch year is more than compensated for by the increase in injected mass capability (see Table 4-2).

4.3 Candidate Mission Capabilities

Solar electric and ballistic flight mode comparisons for each of the four candidate orbiter missions at Jupiter and Saturn are presented in Tables 4-7 through 4-14. The reader is referred to Section 2 for a detailed description of the candidate missions. The net orbiter mass required for each mission is taken from the results of Section 2 of the report and is given at the top of each table. Listed in the vertical columns are the net mass capabilities as a function of launch opportunity, delivery system, and retro propulsion system. The SEP and ballistic flight times chosen for the examples are not always identical, but, generally, they are comparable within 200 days. The 760-day ballistic flight to Jupiter represents a near-minimum energy transfer. Saturn tranfers have been restricted to a duration of about 5 years; this would be commensurate with a total mission lifetime of 8 years assuming a maximum orbit lifetime of 3 years.

Two additional conditions are to be noted in regard to the capability data. First, the solar electric stage is assumed to have an off-optimum power rating of 15kw and a specific impulse of 3500 seconds. The maximum net mass capability has been reduced by 15 percent to account for this off-optimum design and, also, a 20-day launch window. Second, 200m/sec has been added to the retro ΔV requirement for both SEP and ballistic missions to account for midcourse and orbit trim maneuvers. These two factors account for

TABLE 4-7

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

JUPITER ORBITER MISSION NO. 1

NET ORBITER MASS REQUIRED 226kg/SPACECRAFT (2 SPACECRAFT)

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (600^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (760^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D/CENT		TITAN 3D/CENT/BII	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1975	315	350	310	335	375	405
1976	295	330	300	330	345	375
1977	270	300	240	260	310	335
1978	250	280	130	145	225	245
1979	240	270	230	245	300	325
1980	245	275	265	285	330	360
1981-82	250	280	235	255	305	330

Mission No. 1 Definition: Particle and Fields, 0° and 120° inclination, 45^d period, $R_p = 3R_J$

Solar Electric Stage: P_o = 15kw, I_{sp} = 3500 sec

Guidance Allowance: 200 m/sec for midcourse and orbit trim maneuvers

Two-Spacecraft Mounting Structure: 5% of total approach mass

TABLE 4-8

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

JUPITER ORBITER MISSION NO. 2

NET ORBITER MASS REQUIRED 641 KG

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (800^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (760^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D/CENT/ BII		TITAN 3D(7)/CENT	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1979	785	870	510	565	655	725
1980	795	880	560	615	720	795
1981-82	820	910	525	585	675	750
1983	860	945	635	705	850	945
1984	890	975	535	590	690	765
1985	900	990	470	515	585	645
1986	895	985	610	670	790	870

Mission No. 2 Definition: Planetology, 60° inclination, 15^d period, $R_p = 3R_J$

Solar Electric Stage: P_o = 15kw, I_{sp} = 3500 sec

Guidance Allowance: 200m/sec for midcourse and orbit trim maneuvers

TABLE 4-9

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

JUPITER ORBITER MISSION NO. 3

NET ORBITER MASS REQUIRED 664 KG

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (800^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (760^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D/CENT/BII		TITAN 3D(7)/CENT	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1979	815	900	530	590	680	750
1980	810	880	600	655	775	855
1981-82	800	890	540	600	700	770
1983	840	950	620	685	825	920
1984	900	995	505	560	650	725
1985	900	995	455	505	575	630
1986	885	975	625	685	810	890

Mission No. 3 Definition: Planetology/Satellite Observation, 0° inclination $14.^d222$ period, $R_p = 2.29\ R_J$

Solar Electric Stage: $P_o = 15$ kw, $I_{sp} = 3500$ sec

Guidance Allowance: 200 m/sec for midcourse and orbit trim maneuvers

TABLE 4-10

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

JUPITER ORBITER MISSION NO. 4

NET ORBITER MASS REQUIRED 705 KG

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (800^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (760^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D/CENT/BII		TITAN 3D(7)/CENT	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1979	925	1010	600	655	770	840
1980	940	1025	655	715	845	920
1981-82	970	1055	620	680	800	875
1983	1020	1105	750	835	1000	1100
1984	1050	1135	630	680	815	885
1985	1060	1145	550	600	690	750
1986	1060	1140	720	835	930	1080

Mission No. 4 Definition: Planetology/Particle and Fields, 60° inclination, 30^d period, $R_p = 3R_J$

Solar Electric Stage: P_o = 15kw, I_{sp} = 3500 sec

Guidance Allowance: 200 m/sec for midcourse and orbit trim maneuvers

the reduced values of SEP capability relative to the results presented in Section 3. Effectively, a conservative estimate of SEP mission performance is being presented here.

The 2-spacecraft, particles and fields orbiter at Jupiter is, relatively, the easiest mission to accomplish. Table 4-7 shows that the 600-day SEP flight provides an ample margin of capability above the 226 kg requirement in each of the launch years Also, use of the solid retro system would be quite adequate. Since the equatorial orbiter requires two capture ΔV impulses (e.g., ΔV_1 = 1.876 km/sec, ΔV_2 = 0.149 km/sec in 1977), the solid retro would require a restart capability. Alternatively, an auxiliary propulsion device would be needed to provide the second capture impulse plus additional small orbit trim impulses Table 4-7 also shows that the ballistic flight utilizing the Titan 3D/Centaur alone satisfies the mission payload requirement. Addition of the Burner II stage provides a payload margin which could be used to reduce the ballistic flight time; e.g., to 500 days in 1975 and to 700 days in 1979.

Solar electric capabilities are comparatively more favorable when considering the Jupiter planetology orbiter missions. Payload margins of several hundred kilograms are available on the 800-day flight. Hence, the excess capability offers significant flight time reductions of 50-200 days depending on the launch opportunity and which retro system is employed. For example, with reference to Mission No. 2 shown in Table 4-8, the 641 kg orbiter launched in

1986 can be delivered in 640 days (solid retro) or 600 days (space-storable retro). Corresponding flight times for the 1979 opportunity are 710 or 670 days. Ballistic flights utilizing the Titan 3D/Centaur/BII do not generally provide sufficient mass to accomplish the planetology orbiter missions. At best, this launch vehicle combined with a space-storable retro stage could perform Mission No. 4 in only two launch years, 1980 and 1983. It is seen, however, that the Titan 3D(7)/Centaur has more than adequate capability over several launch opportunities. Some degree of ballistic flight time reduction and/or launch window extension would be possible. Even though the Titan 3D/Centaur/SEP has a better performance envelope than the Titan 3D(7)/Centaur, either delivery system is capable of accomplishing each of the candidate missions. For Jupiter then, one may view the development alternatives as being between the SEP upper stage or uprating the Titan 3D core vehicle to the proposed 7-segment version.

Saturn orbiter mission comparisons are shown in Tables 4-11 to 4-14 for the 1979-86 launch opportunity period. Note that the SEP capability decreases during this period whereas the ballistic capability increases. As in the Jupiter case, the least difficult of the four candidate missions is the 2-spacecraft particles and fields/ring probe orbiter. The equatorial orbiter has an initial periapse distance of 3 Saturn radii which is then reduced in small steps to 1.1 radii. The total maneuver requirement of about 125 m/sec has been accounted for by the ΔV reserve of 200 m/sec. This is the only Saturn mission that can be performed ballistically

TABLE 4-11

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

SATURN ORBITER MISSION NO. 1

NET ORBITER MASS REQUIRED 231kg/SPACECRAFT (2 SPACECRAFT)

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (1480^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (1680^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D(7)/CENT/BII		TITAN 3D/CENT/HFK(10)	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1979	270	300	200	220	280	305
1980	265	290	210	230	285	310
1981	255	280	215	235	290	320
1982-83	250	275	235	255	310	340
1983-84	240	270	240	265	315	350
1984-85	230	260	245	270	320	355
1985-86	225	250	255	285	335	370

Mission No. 1 Definition: Particle and Fields/Ring Probe, 0° and 90° inclination, 45^d period, $R_p = 3R_s$

Solar Electric Stage: $P_o = 15kw$, $I_{sp} = 3500$ sec

Guidance Allowance: 200 m/sec for midcourse and orbit trim maneuvers

Two-Spacecraft Mounting Structure: 5% of total approach mass

TABLE 4-12

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

SATURN ORBITER MISSION NO. 2

NET ORBITER MASS REQUIRED 642 KG

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (1680^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (1680^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D/CENT/HFK(10)		TITAN 3D(7)/CENT/HFK(15)	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1979	610	675	500	550	635	700
1980	600	665	515	570	675	740
1981	595	660	540	600	710	775
1982-83	590	655	590	650	770	850
1983-84	585	650	620	685	815	900
1984-85	580	645	635	710	845	935
1985-86	575	640	675	750	875	970

Mission No. 2 Definition: Planetology/Rings, 60° inclination, 15^d period, $R_p = 3R_s$

Solar Electric Stage: $P_o = 15kw$, $I_{sp} = 3500$ sec

Guidance Allowance: 200 m/sec for midcourse and orbit trim maneuvers

TABLE 4-13

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

SATURN ORBITER MISSION NO. 3

NET ORBITER MASS REQUIRED 642 KG

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (1680^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (1680^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D(7)/CENT/BII		TITAN 3D/CENT/HFK(10)	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1979	715	775	425	460	585	635
1980	710	770	445	485	610	660
1981	705	765	470	510	640	700
1982-83	700	760	530	575	700	760
1983-84	695	755	565	615	735	805
1984-85	695	755	590	640	770	840
1985-86	690	750	625	680	815	885

Mission No. 3 Definition: Planetology/Rings, 60° inclination, 7.5^d period, $R_p = 1.1\ R_s$

Solar Electric Stage: P_o = 15kw, I_{sp} = 3500 sec

Guidance Allowance: 200 m/sec for midcourse and orbit trim maneuvers

TABLE 4-14

COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITIES

SATURN ORBITER MISSION NO. 4

NET ORBITER MASS REQUIRED 660 KG

LAUNCH OPPORTUNITY	SOLAR ELECTRIC (KG) (1880^d FLIGHT, 20^d WINDOW)		BALLISTIC (KG) (1680^d FLIGHT, 10^d WINDOW)			
	TITAN 3D/CENT/SEP		TITAN 3D/CENT/HFK (10)		TITAN 3D(7)/CENT/HFK(15)	
	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO	SOLID RETRO	S/S RETRO
1979	675	745	490	545	625	695
1980	640	710	485	540	635	705
1981	610	675	490	545	640	715
1982-83	580	640	500	565	650	730
1983-84	545	605	510	575	670	755
1984-85	520	580	510	575	660	750
1985-86	495	560	525	600	680	775

Mission No. 4 Definition: Planetology/Satellite Observation, 0° inclination, 15.9^d period, $R_p = 3R_s$

Solar Electric Stage: P_o = 15 kw, I_{sp} = 3500 sec

Guidance Allowance: 200 m/sec for midcourse and orbit trim maneuvers

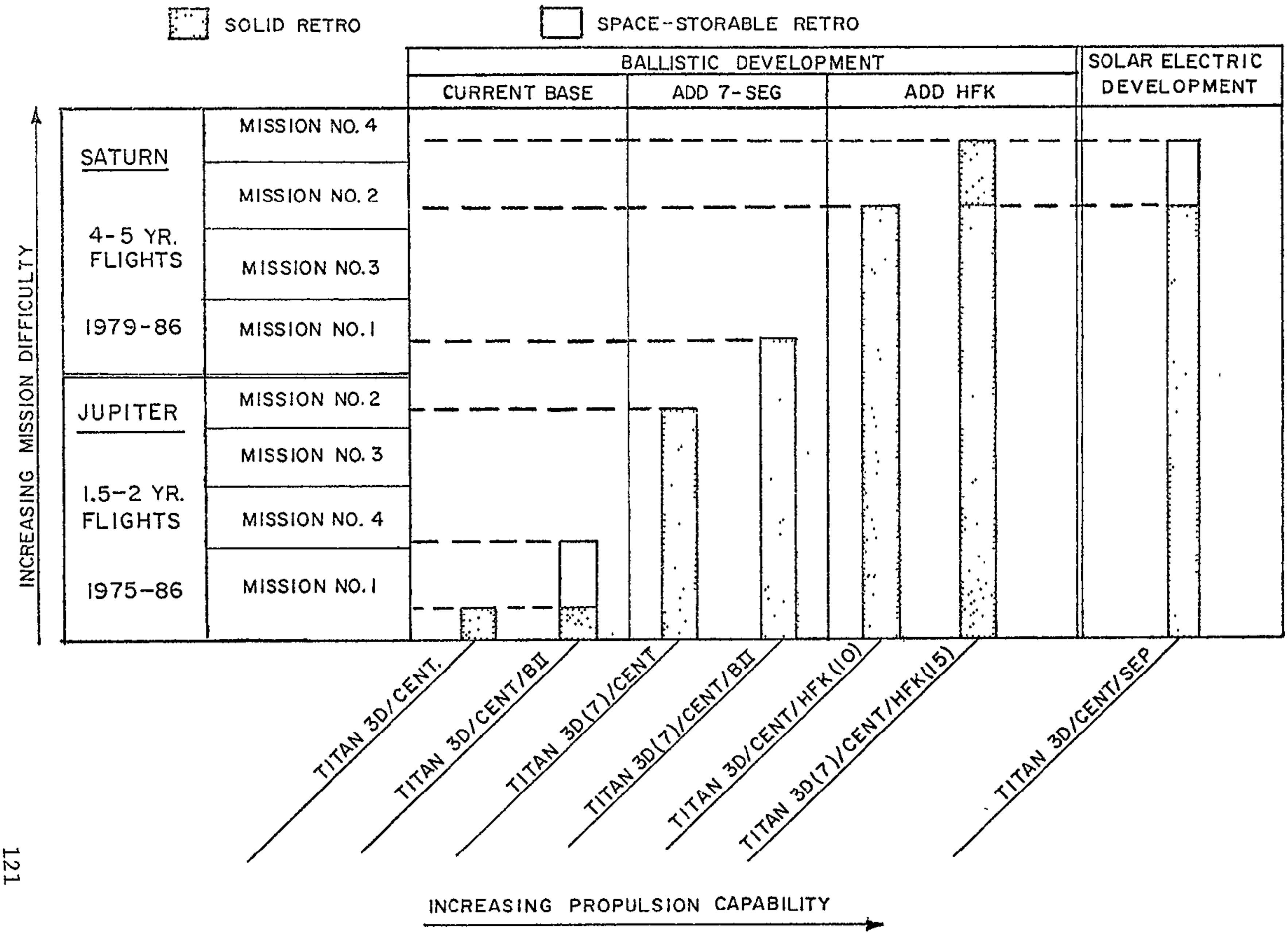
by the Titan 3D(7)/Centaur/BII. The comparative advantage of the solar electric system is a 200-day shorter flight time. Table 4-12 shows that Mission No. 2 cannot be performed in 1680 days with the SEP-solid retro combination. The necessary mass capability is provided, however, by increasing the flight time to 1880 days; e.g., in 1981 the orbiter mass of 595 kg would increase to 680 kg. The same increase in flight time for the Titan 3D/Centaur/HFK(10) - Solid Retro system increases the 1981 ballistic capability from 540 kg to only 580 kg, but would open up the following two launch opportunities. Focusing on the Titan 3D/Centaur vehicle, it would be fair to say that the required orbiter mass of 642 kg for Missions No's. 2 and 3 could be delivered by either the SEP or HFK (10) upper stages. Mission No. 4 is more difficult because of the equatorial orbit requirement. In this case the space-storable retro is needed for solar electric applications and the mission capability tradeoff is between the Titan 3D/Centaur/SEP and the Titan 3D(7)/Centaur/HFK(15), the latter ballistic vehicle providing somewhat better performance in flight time and number of launch opportunities. It may be noted that the space-storable retro is needed to accomplish Mission No. 4 even if the SEP power rating is increased to the optimum value.

4.4 Comparison Summary

An overview of the preceeding results are presented as a bar chart in Figure 4-2. The various Jupiter and Saturn orbiter missions are listed on the left side of the chart in terms of increasing mission difficulty. The seven launch vehicle/upper stage

FIGURE 4-2.

SUMMARY COMPARISON OF SOLAR ELECTRIC AND BALLISTIC CAPABILITY FOR JUPITER AND SATURN ORBITER MISSIONS



delivery systems considered in the analysis are given at the bottom of the chart increasing, from left to right, in propulsion capability. Ballistic systems are conveniently subdivided to indicate the two stages of development above the current technology base (Titan 3D/Centaur/BII). The number of missions which can be performed by any specific propulsion combination is indicated by the height of the bar. Mission capability is included in the summary chart only if there are at least three launch opportunities in the time period of interest when that particular mission can be achieved. The range of flight times in the summary comparison is 1.5-2 years to Jupiter and 4-5 years to Saturn. Ballistic and SEP flight times differ by less than 200 days.

The choice between the solid and space-storable retro stages should also be considered as a development option. It is assumed that once a retro stage is selected its design (and technology experience) will be retained for all orbiter missions. In reading the chart, the solid bar means that the solid retro stage is adequate to perform each of the included missions. The open bar indicates when additional capability is provided by use of the space-storable retro. It is seen that the current base ballistic systems can perform only the minimum-objective particles and fields mission at Jupiter if the solid retro is used. The space-storable retro would allow the combined planetology/particles and fields mission to be performed (but only for 1980 and 1983 launches). Development of the Titan 3D(7)/Centaur/BII offers a capability envelope encompassing all of the Jupiter missions plus the minimum-objective Saturn mission. The desired planetology orbiters at Saturn can be accomplished ballistically only at the cost of developing a high energy hydrogen-flourine kick stage. The Titan 3D/Centaur/HFK(10) performs Saturn Missions No's. 1, 2 and 3 with the solid retro stage but does not add Mission No. 4 even with the space-storable retro. Considering the entire set of Jupiter and Saturn reference missions, it is seen that the Titan 3D/Centaur/SEP is essentially on a capability par with the Titan 3D/Centaur/HFK(10).

In conclusion, this study has attempted to provide information to the program planner in three areas relevant to Jupiter and Saturn orbiter missions:

(1) Mission definition options and corresponding sample payloads which are suited to accomplishing meaningful science objectives.

(2) Basic characteristics, capabilities and design tradeoff options which describe the application of solar electric propulsion to these missions.

(3) Comparison of solar electric and ballistic capabilities in performing the set of candidate missions.

The principle results of this study are thought to lie in the comparison analysis, the distillation of which is framed as a choice between future development options in the solar electric propulsion as against those in chemical launch vehicles (Titan/Centaur class) and high energy upper stages. In addition to the ballistic combinations considered here, the Intermediate-20 (Saturn class) must be recognized as a competing capability. The Intermediate-20 introduces an element of over-kill for Jupiter and Saturn missions, but this might be countered by broader applications (Uranus and Neptune missions) and the possibility of multi-mission (dual spacecraft) launches. If this factor is disregarded and only Titan class vehicles are considered, then it is concluded that the addition of an SEP stage is probably the most useful improvement that can be made in extending the capability of the Titan 3D/Centaur. This conclusion factors in many other unmanned space exploration missions besides Jupiter and Saturn orbiters; e.g., comet and asteroid rendezvous, Mercury orbiter, and solar probe missions. For most of these other missions the SEP power availability at the target yields an added bonus.

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APPENDIX A - SATURN ORBITS

The presence of the ring system surrounding Saturn places a restraint on the orbit selection for Saturn missions. The orbits must be chosen so that the spacecraft will not pass through and collide with the observed ring material. There are two ways of doing this. One is to select those orbits whose distance of closest approach to the planet, or periapse, falls well outside the rings' outer boundary. The other is to chose those orbits which allow the spacecraft to slip through the gap between the inner edge of the C ring and the cloud-top.

The outer edge of the A ring appears to be at approximately 2.3 R_s from the planet's center but material, possibly in the D ring, may extend further out. Placing the periapse distance at 3 R_s provides a relatively safe orbit which keeps the spacecraft at least 40,000 km from the observed ring material at all times. This periapse distance was chosen for three of the four candidate Saturn Orbiter Missions (Nos. 1, 2, and 4) discussed in this report

Selecting an orbit which avoids the rings by taking the spacecraft between the inner edge of the C ring (1.2 R_s) and Saturn's cloud top (1.0 R_s) is a considerably more exacting task than simply avoiding the rings by remaining outside them. If an orbital periapse of 1.1 R_s (from Saturn's center) is chosen in order to remain well outside the atmosphere, then the orbital inclination must exceed 30°. Now with this fixed periapse distance

(1.1 R_s), a chosen orbital inclination Ø (where Ø $\geq$ 30°), and a fixed incoming asympote due to the spacecraft's interplanetary trajectory, there are still two posigrade orbits into which the spacecraft may be injected. These two orbits differ in the position of their ascending nodes and arguments of periapse. Only one will allow the spacecraft to pass through the gap. Throughout this report the orbit option which passed through the gap was assumed for Mission No. 3 (which has an orbit periapse distance of 1.1 R_s, inclination of 60°, and 7.5 day period.)

Figure A-1 shows six representative orbits available to a Saturn orbiter mission (arrival date: JD 2448109, 8/5/90). Four orbits with a periapse distance of 3 R_s are illustrated together with the sunrise/sunset terminators position during the first orbit for a polar (90°) and a 30° inclined orbit. The terminator's position in the 30° orbit allows the altitude limited planetology instruments a much better opportunity for covering the sunlit side of Saturn than it does in the polar orbit. The terminator's position does not change much as the inclination is increased to 60°, providing for increased latitude coverage with good illumination. Two orbits with periapse distance of 1.1 R_s have been included for comparison. These must have orbital inclinations of 30° or greater to avoid collisions with the ring material.

Figure A-2 illustrates the relationship between the spacecraft's orbital radius and time. The 45-day orbit requires 106 Saturn days for one revolution and the 7.5 day orbit, 18

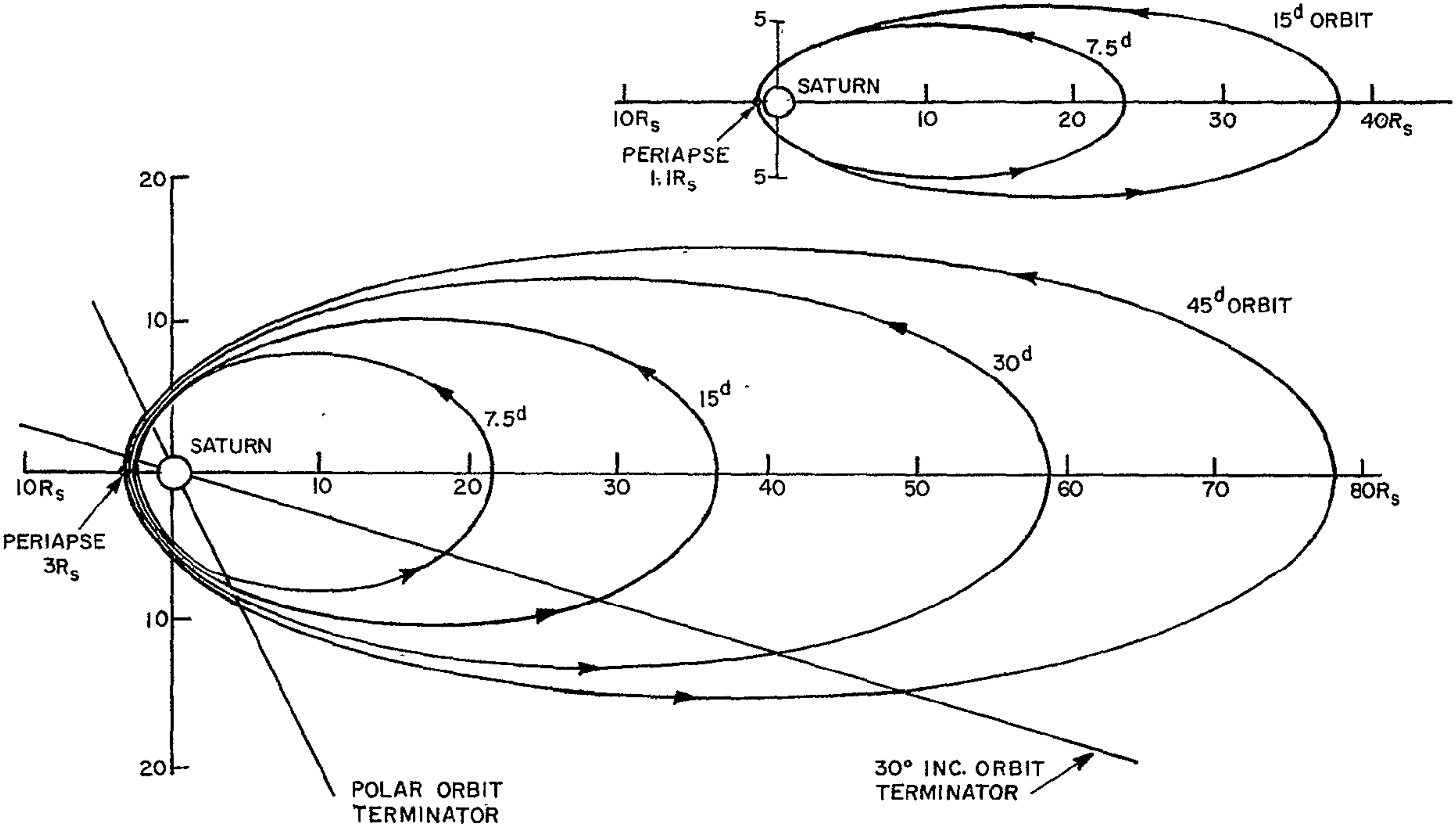


FIGURE A-1. REPRESENTATIVE SATURN ORBITS

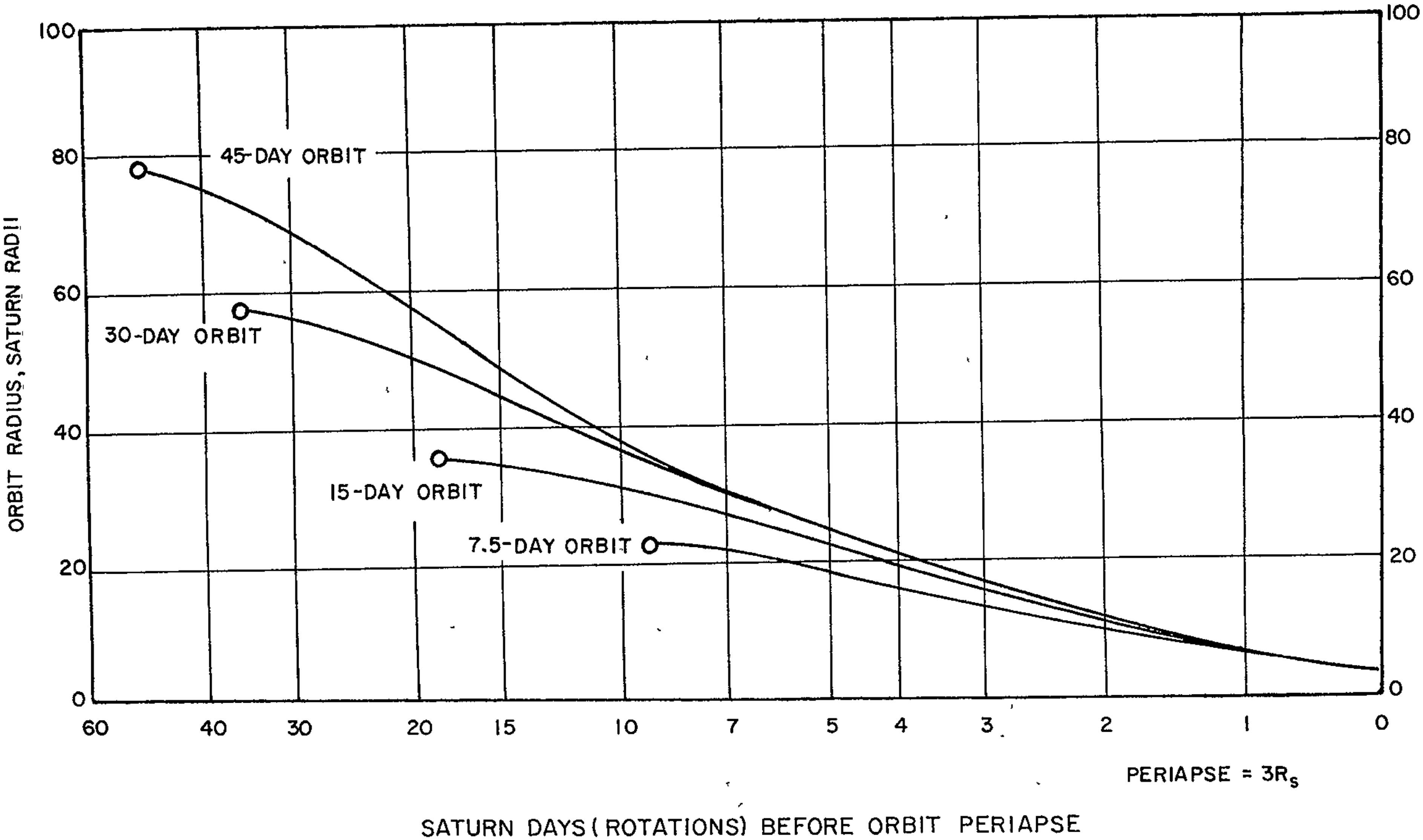


FIGURE A-2. SATURN ORBIT ALTITUDE VARIATIONS.

Saturn days. Each spends about the same length of time near periapse. As with Jupiter, only two Saturn days per orbit are spent within 10 R_s of the planet, regardless of orbital period.

Figure A-3 shows the subspacecraft ground trace for the four orbits with a periapse of 3 R_s shown in Figure A-1, and an orbital inclination of 60°. The four orbits are again nearly coincident near periapse, pointing out the advantage for planetology instruments offered by the short period orbit with its greater number of low altitude passes available.

Ring occultations occur when the line of sight between the spacecraft and the sun (virtually at infinity) intersects the ring structure. Figure A-4 is a schematic representation of the geometry involved. As a spacecraft moves between points A and B in its orbit the intersection point of the spacecraft-sun line and the ring plane traces a curve between points a and b. A ring occultation occurs during the time the curve lies within the boundary of the rings (1.2 - 2.3 R_s).

Figure A-5 shows part of this curve of the sun's apparent path in the ring plane for a particular case for Mission No. 2 (arrival date: 8/5/90). Also shown are plots of the "line of sight" distance from the rings during this occultation. Negative time arguments indicate time before periapse passage.

Occasionally, during parts of its orbit, a spacecraft is able to observe its "shadow" or backscatter point on the rings. Figure A-6 schematically illustrates this. The spacecraft-sun

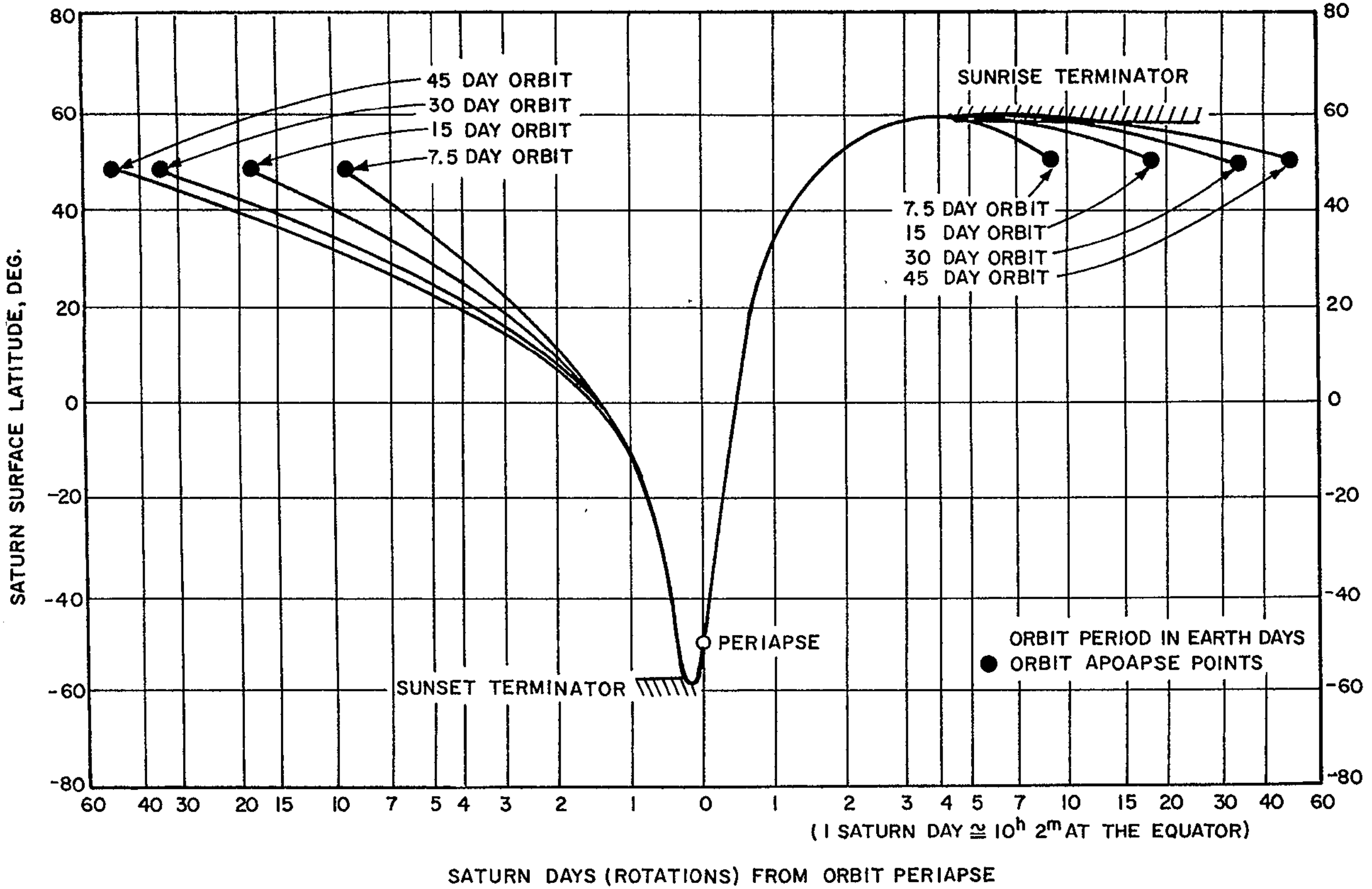
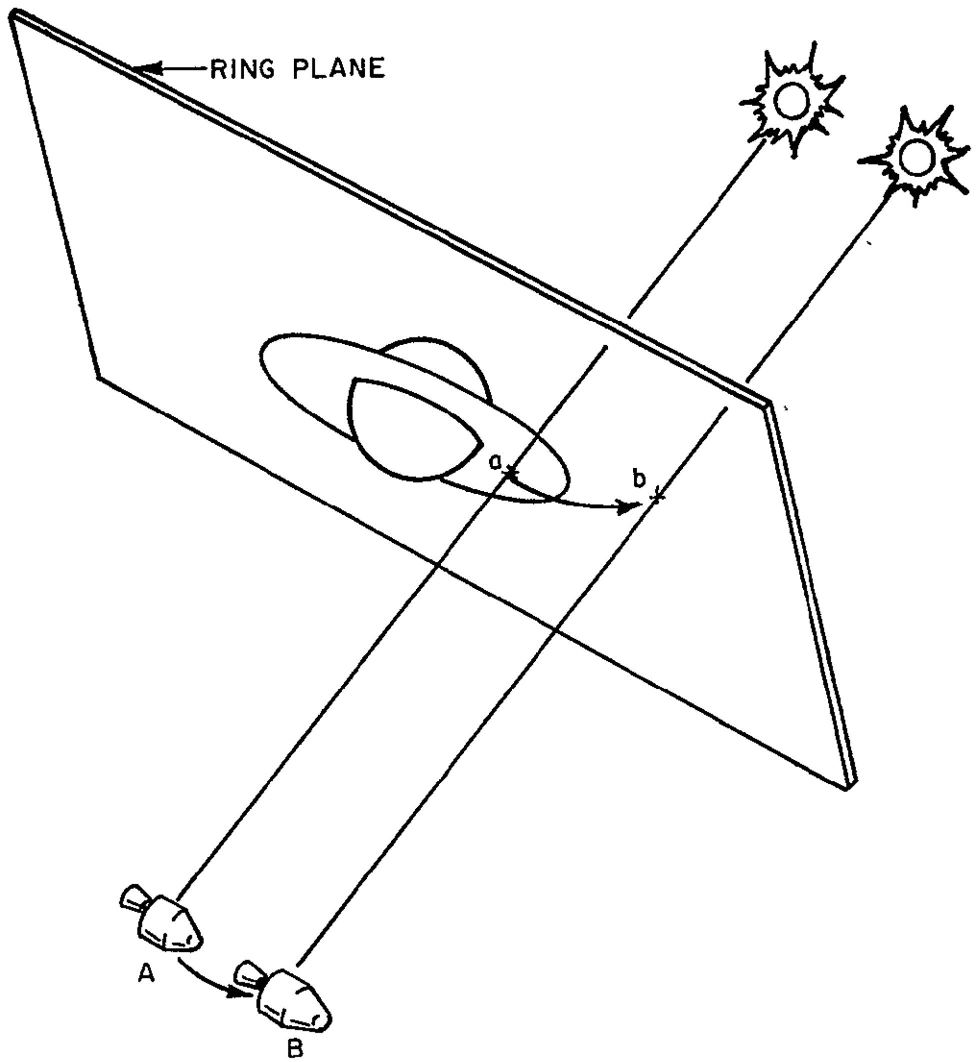
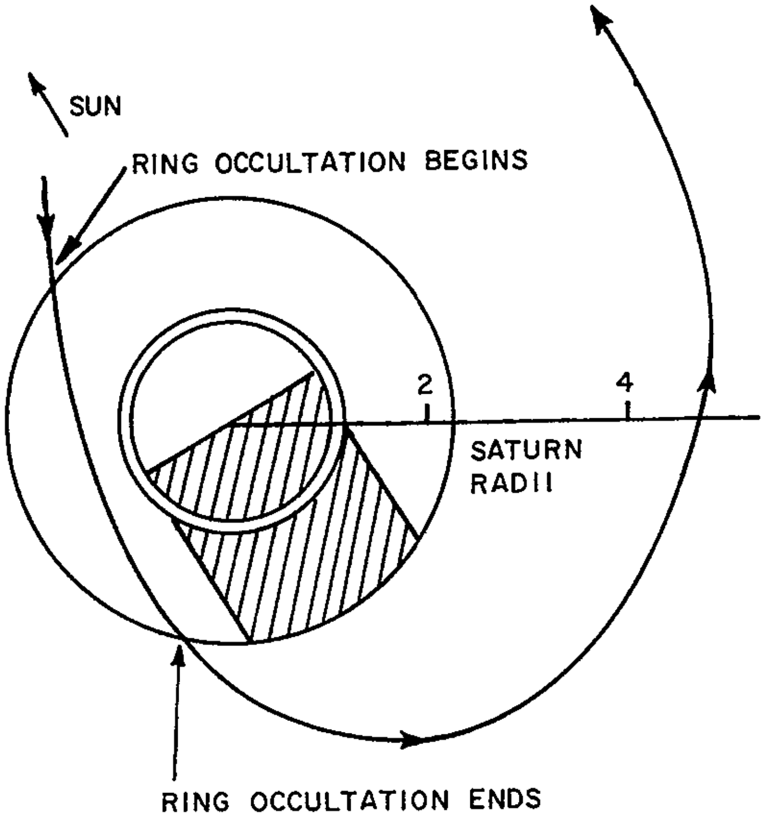


FIGURE A-3. SATURN ORBIT LATITUDE VARIATIONS (ORBIT INCLINATION = 60 DEG., PERIAPSE ≑ $3R_S$)

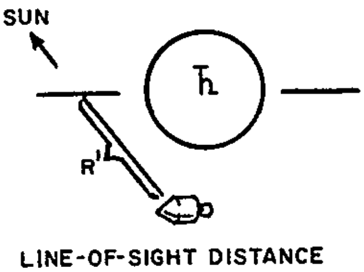


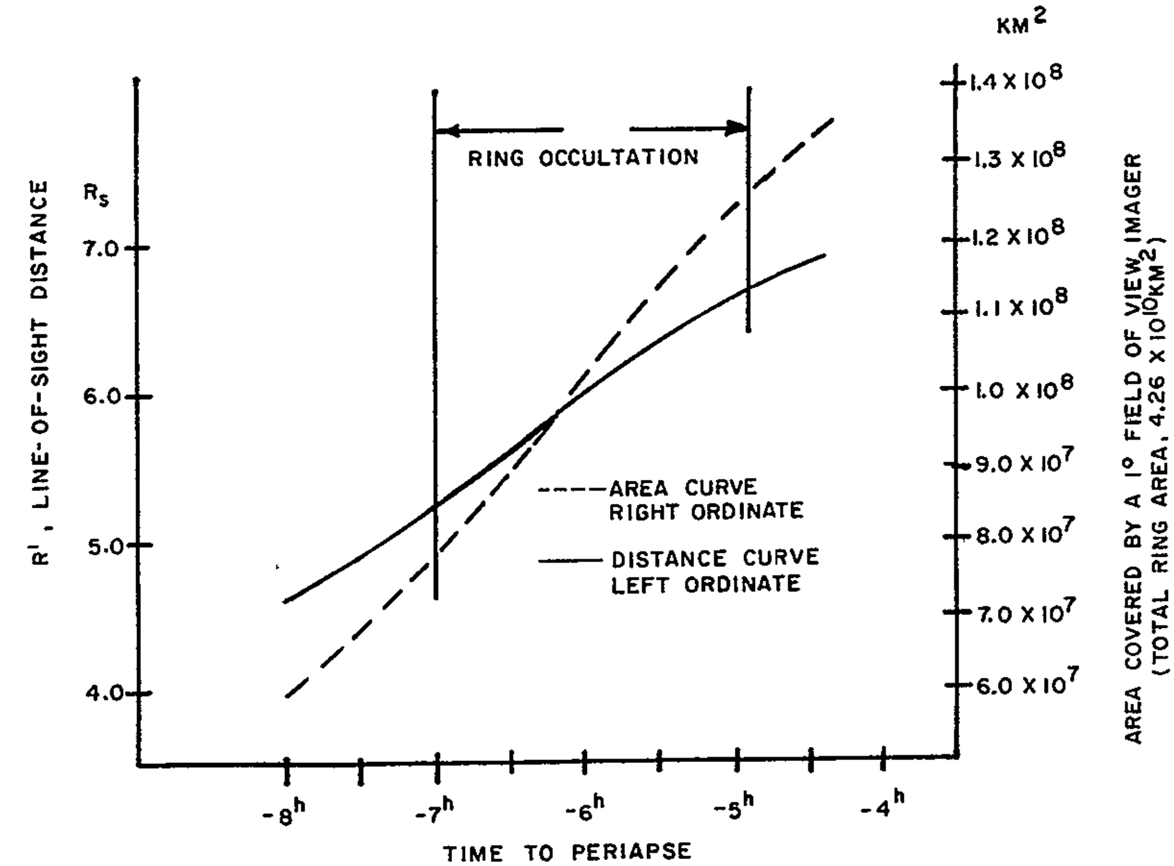
AS THE SPACECRAFT MOVES FROM POINT A TO POINT B IN ITS ORBIT THE POINT OF INTERSECTION OF THE SPACECRAFT-SUN LINE AND THE RING PLANE MOVES BETWEEN POINTS a AND b IN THE RING PLANE, TRACING OUT A CURVE. WHEN THIS CURVE FALLS WITHIN THE BOUNDARIES OF THE RING SYSTEM (1.2 R_s - 2.3 R_s) A RING OCCULTATION OCCURS.

FIGURE A-4. RING OCCULTATION



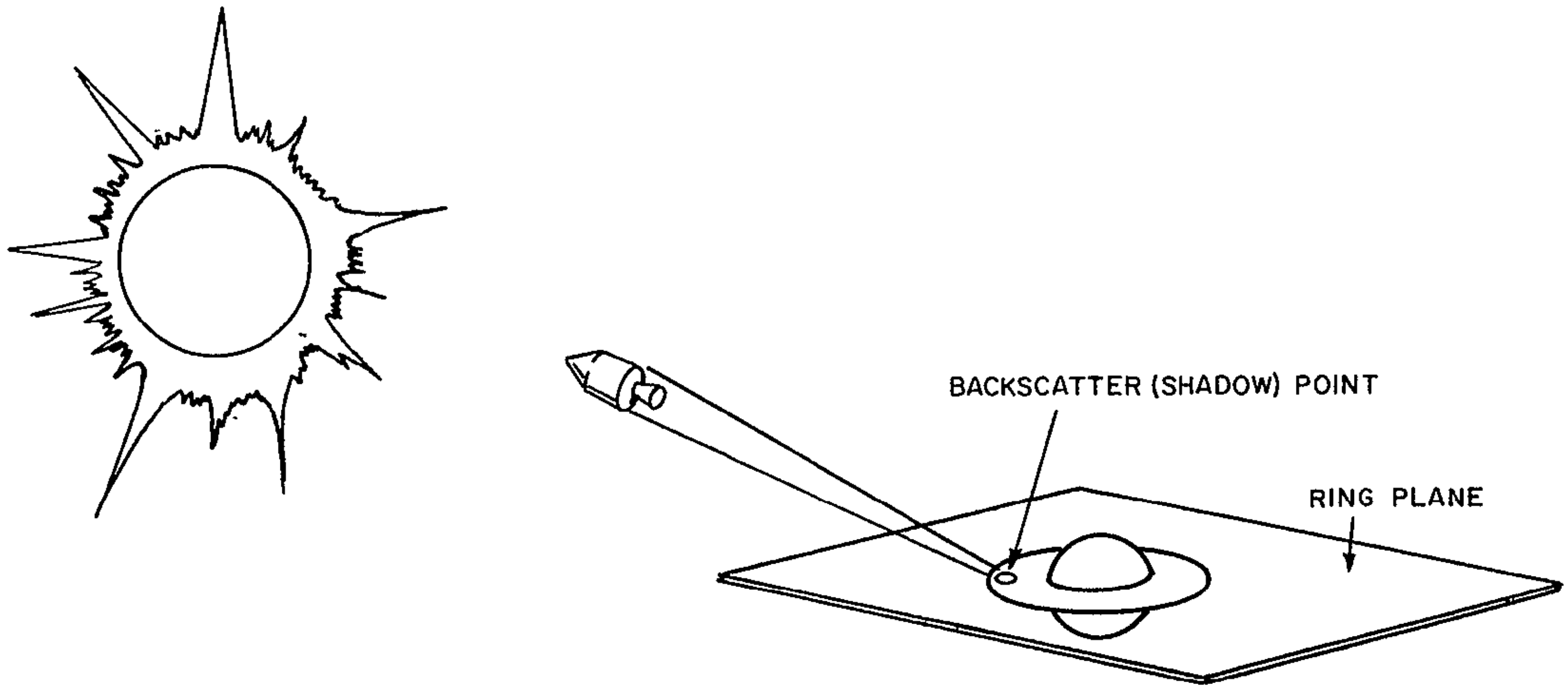
THE SUN'S APPARENT PATH IN THE RING PLANE AS VIEWED BY THE MISSION NO.2 SPACECRAFT (15^d, 60° INCLINATION, 3 R_s PERIAPSE, ARRIVAL DATE 2448109, 8/5/90). THE RING OCCULTATION SHOWN LASTS 2.1 HOURS.
(THE DECLINATION OF THE SUN IS ~23°)





DISTANCE TO THE RING AND AREA IMAGED ON THE RING DURING OCCULTATION.

FIGURE A-5. RING OCCULTATION-MISSION NO. 2.

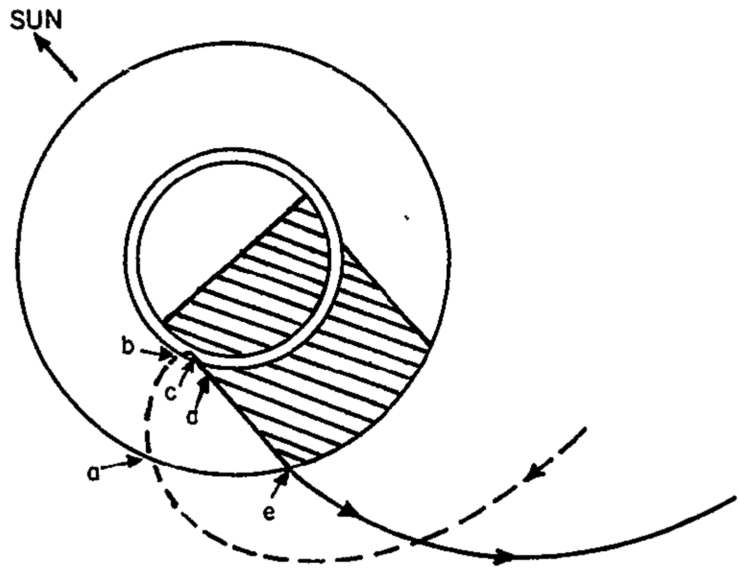


WHEN BOTH THE SPACECRAFT AND THE SUN ARE POSITIONED ON THE SAME "SIDE" OF THE RING PLANE THE SPACECRAFT-SUN LINE TRACES OUT A CURVE ON THE RING PLANE WHICH WHEN WITHIN THE RINGS BOUNDARIES IS THE BACKSCATTER POINT OF THE SPACECRAFT.

FIGURE A-6. BACKSCATTER POINT OBSERVATION.

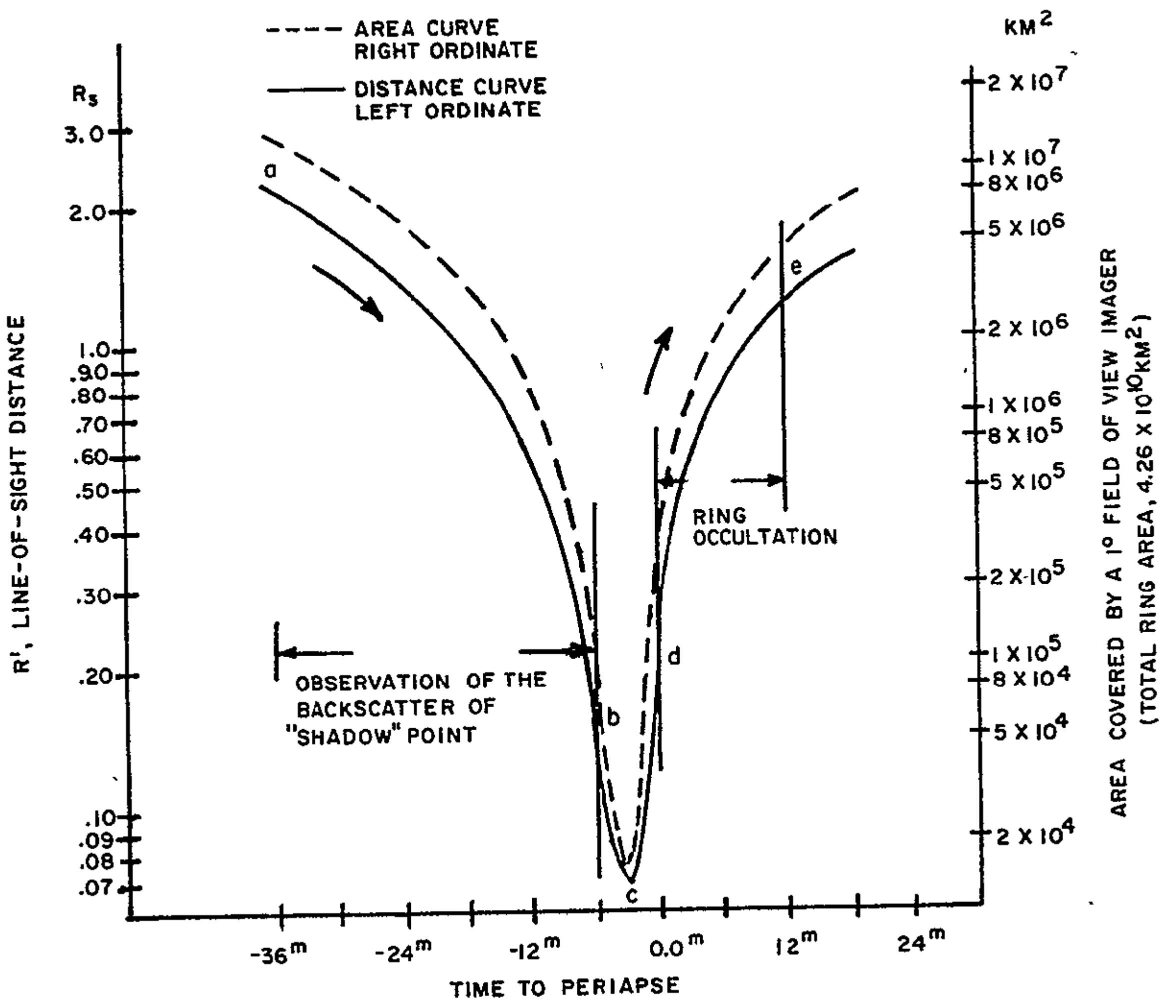
line is extended until it intersects the ring plane. The intersection point traces out a curve on the ring plane as the spacecraft moves in its orbit. When this curve crosses the ring structure a backscatter point on the rings is observable. (Backscatter point observations are possible when both the spacecraft and the sun are on the same "side" of the ring plane. Ring occultations occur when the spacecraft and sun are on opposite "sides" of the ring plane).

Figure A-7 illustrates the backscatter and ring occultation arcs which occur during the first orbit of a particular case for Mission No. 3 (arrival date: 8/5/90). At point a on the figure observation of the backscatter point begins. This ends at point b 30 minutes later, when the "shadow" moves into the gap between the C ring and cloud top. At c the spacecraft is crossing the ring plane, passing from the Northern to Southern hemisphere. Point d marks the beginning of the ring occultation which ends at point e 12 minutes later. Also included in Figure A-7 are plots of the "line of sight" distance and area image by a 1 degree field of view instrument during both the backscatter point observation and the ring occultation. This particular opportunity allows a much higher resolution study to be made of the rings than does the case illustrated for Mission No. 2 in A-5, because R', the line of sight distance, is much smaller. However, the total observation time, 42 minutes, is much shorter than the 126 minutes shown for Mission No. 2.



A representative ring occultation (———) and backscatter point observation (----) for mission No. 3 (7.5^d, 60° inclination, 1.1 R_s periapse, arrival date 2448109, 8/5/90)

a-Backscatter Pt. observation begins
b-Backscatter Pt. observation ends
c-Spacecraft crosses ring plane
d-Ring occultation begins
e-Ring occultation ends.



DISTANCE TO THE RING AND AREA IMAGED ON THE RING VS. TIME.

FIGURE A-7. RING OCCULTATION AND BACKSCATTER POINT OBSERVATION-MISSION NO. 3.

The equatorial orbiting spacecraft of Missions Nos. 1 and 4 can only view the rings edge-on and can observe neither of the phenomena mentioned above. Also, ring occultations are not available to the polar orbiter of Mission No. 1.

APPENDIX B

SOLAR ELECTRIC TRAJECTORY/PAYLOAD DATA

The data base used in the analysis of solar electric mission performance was provided by the Jet Propulsion Laboratory. A portion of this data is presented in Figures B-1 through B-3. Curves of net spacecraft mass in orbit versus launch date (in the vicinity of optimal launch date) are shown for a range of flight times and launch opportunities. The net mass data was generated for an arbitrarily fixed set of mission parameters or input conditions (i.e., launch vehicle, propulsion system specific mass, orbit size, etc.). It was therefore necessary to scale this data to the parameter set employed in the present mission analysis. The scaling relationships developed in a previous study task [4] were used for this purpose. Results presented in Sections 3 and 4 of this report reflect this scaling of the original data base.

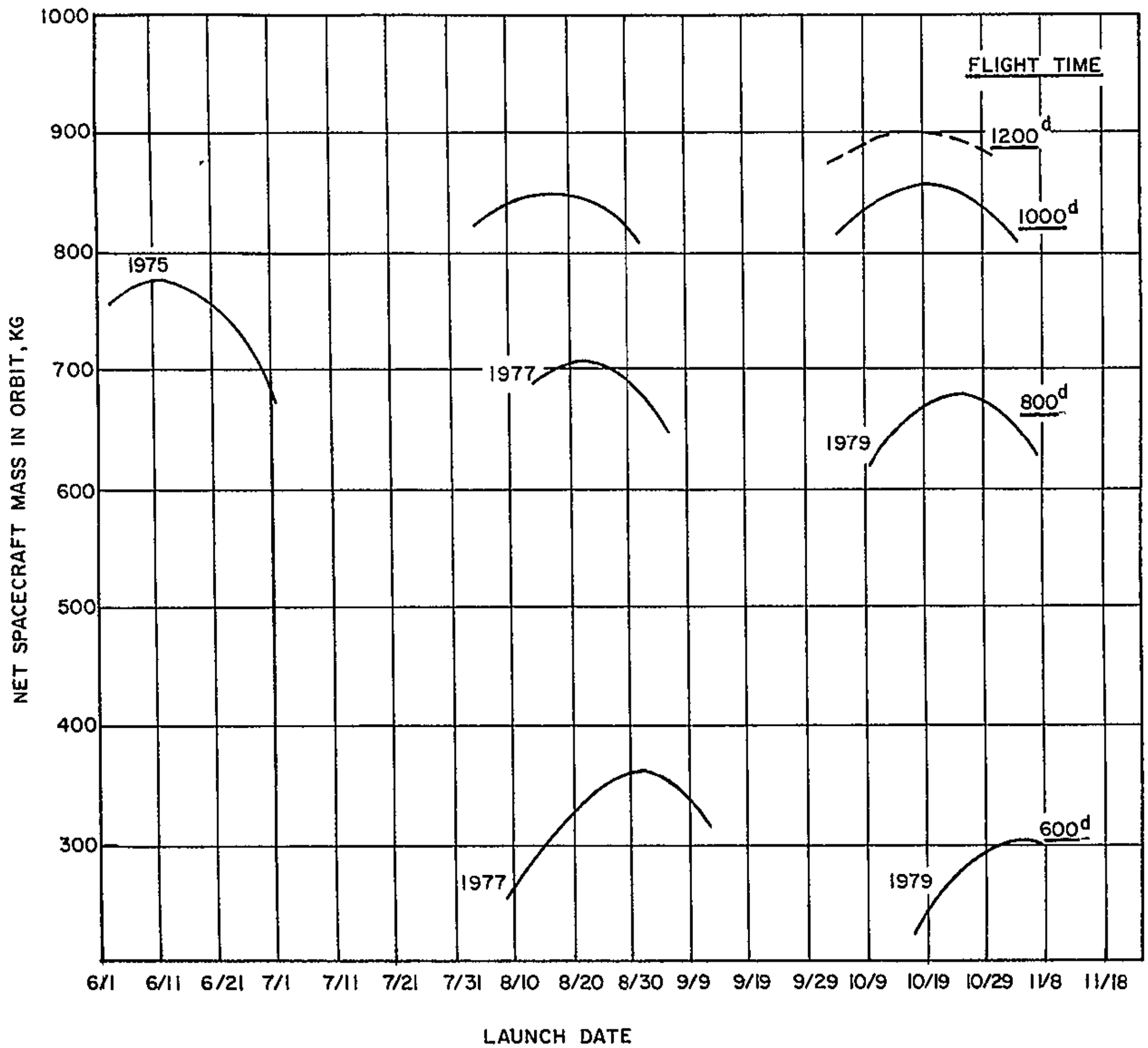


FIGURE B-1 OPTIMAL SOLAR ELECTRIC JUPITER ORBITER PAYLOAD PERFORMANCE FOR LAUNCH OPPORTUNITIES FROM 1975 TO 1986, PROPULSION SYSTEM NOT JETTISONED

MISSION PARAMETERS: L V = TITAN 3D/CENTAUR, α = 30 KG/KW, K_{pt} = 0.03, K_{re} = 0.111, (b = 0.769, d = 14300 thruster coefficients), Isp (retro) = 300 SEC, ORBIT = 3 X 38, OPTIMIZED PARAMETERS: POWER, EXAUST VELOCITY, DEPARTURE AND ARRIVAL C_3

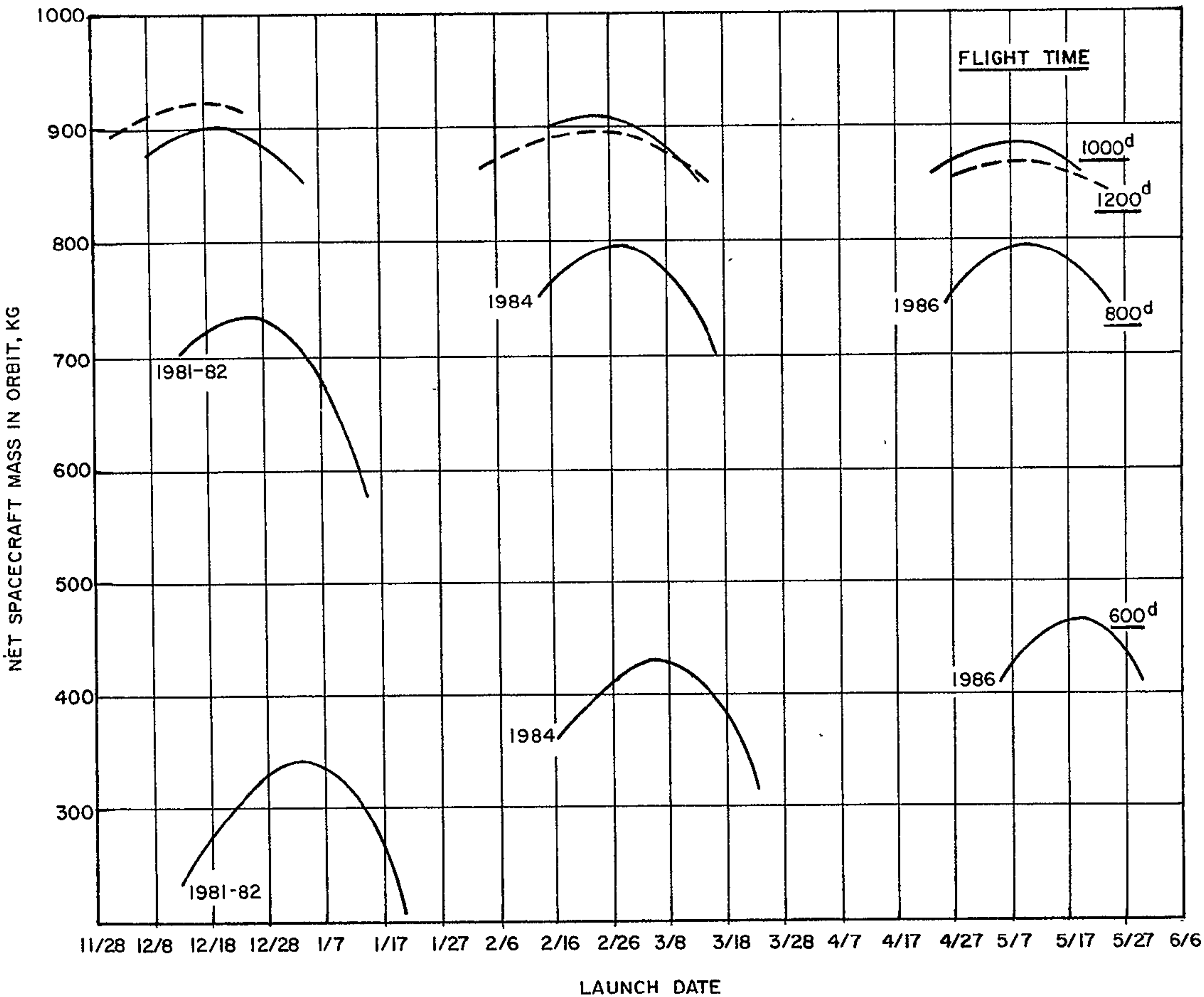


FIGURE B-1 (CONTINUED)

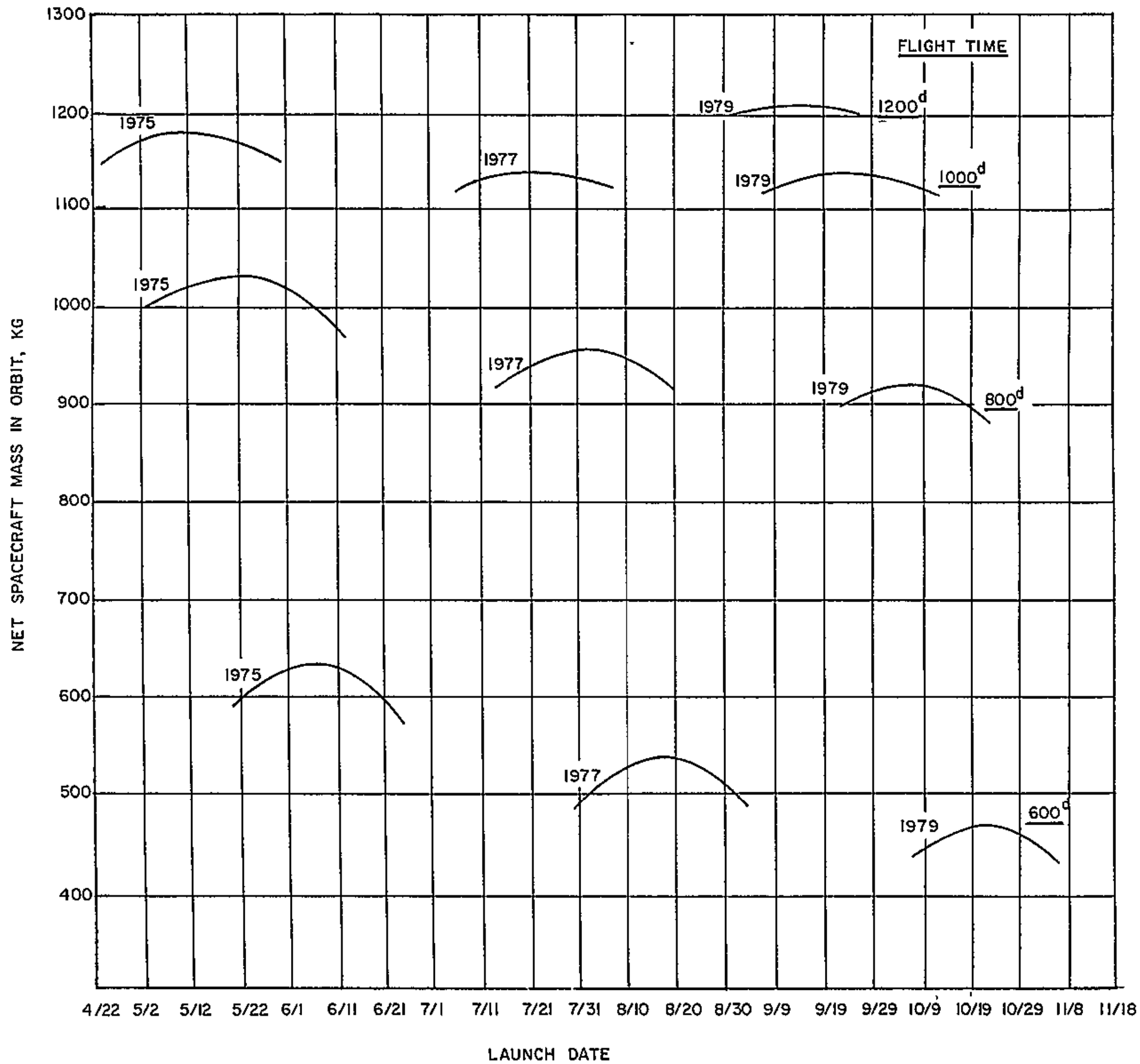


FIGURE B-2 OPTIMAL SOLAR ELECTRIC JUPITER ORBITER PAYLOAD PERFORMANCE FOR LAUNCH OPPORTUNITIES FROM 1975 TO 1986, PROPULSION SYSTEM JETTISONED

MISSION PARAMETERS L V = TITAN 3D/CENTAUR, α = 30 KG/KW, K_{pt} = 0 03, K_{re} = 0 111, (b = 0 769, d = 14300 thruster coefficients), Isp (retro) = 300 SEC, ORBIT = 3 X 38, OPTIMIZED PARAMETERS POWER, EXAUST VELOCITY, DEPARTURE AND ARRIVAL C_3

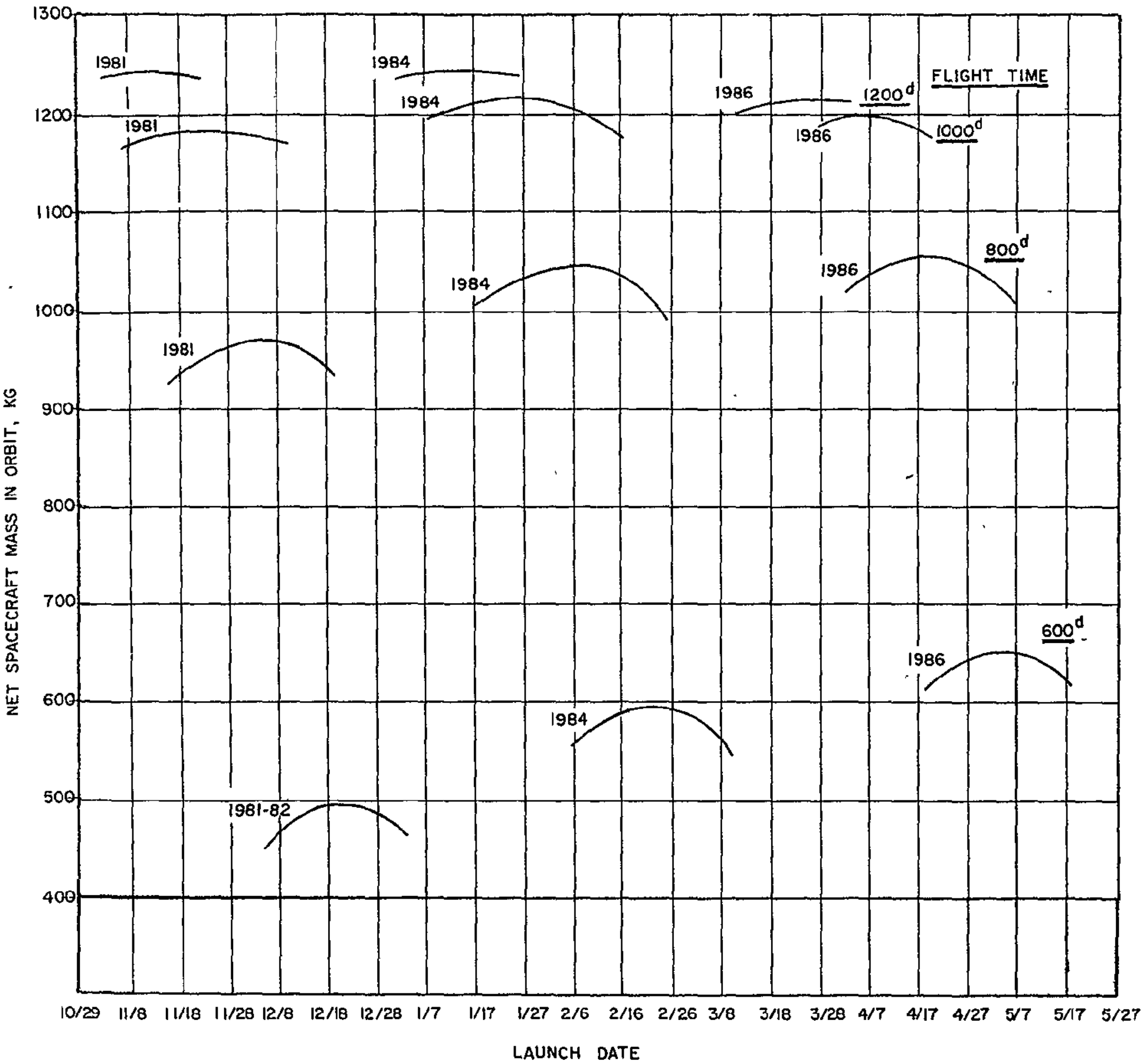


FIGURE B-2. (CONTINUED)

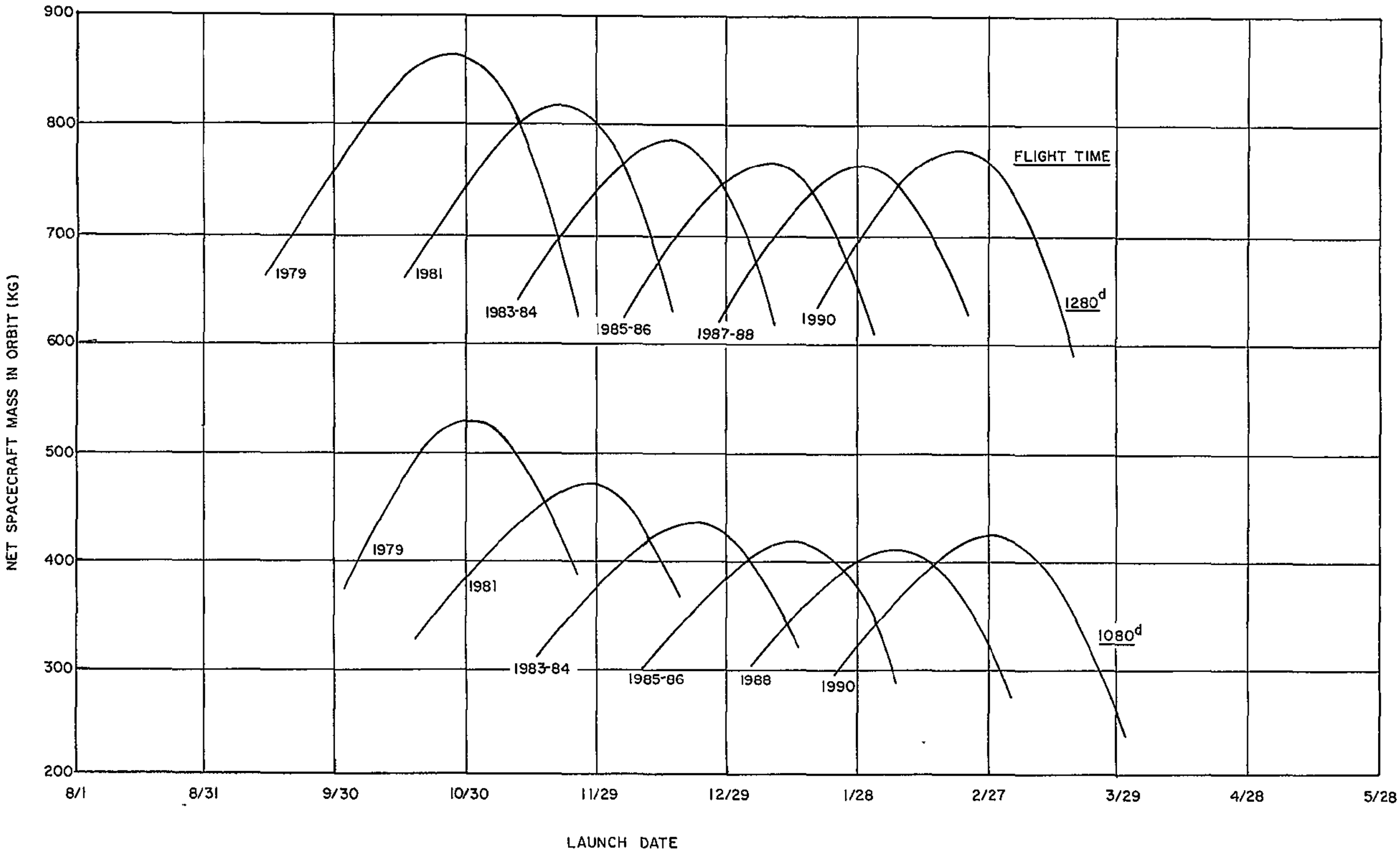


FIGURE B-3. OPTIMAL SOLAR ELECTRIC SATURN ORBITER PAYLOAD PERFORMANCE FOR LAUNCH OPPORTUNITIES FROM 1979 TO 1990

MISSION PARAMETERS: L V = TITAN 3D (7)/CENTAUR α = 34 KG/KW, K_{pt} = 0.03, K_{re} = 0.111, (b = 0.763, d = 16600 thruster coefficients), Isp (retro) = 385 SEC, ORBIT = 2.3 X 59.3 (30 DAY PERIOD), OPTIMIZED PARAMETERS. POWER, EXHAUST VELOCITY, DEPARTURE AND ARRIVAL C3

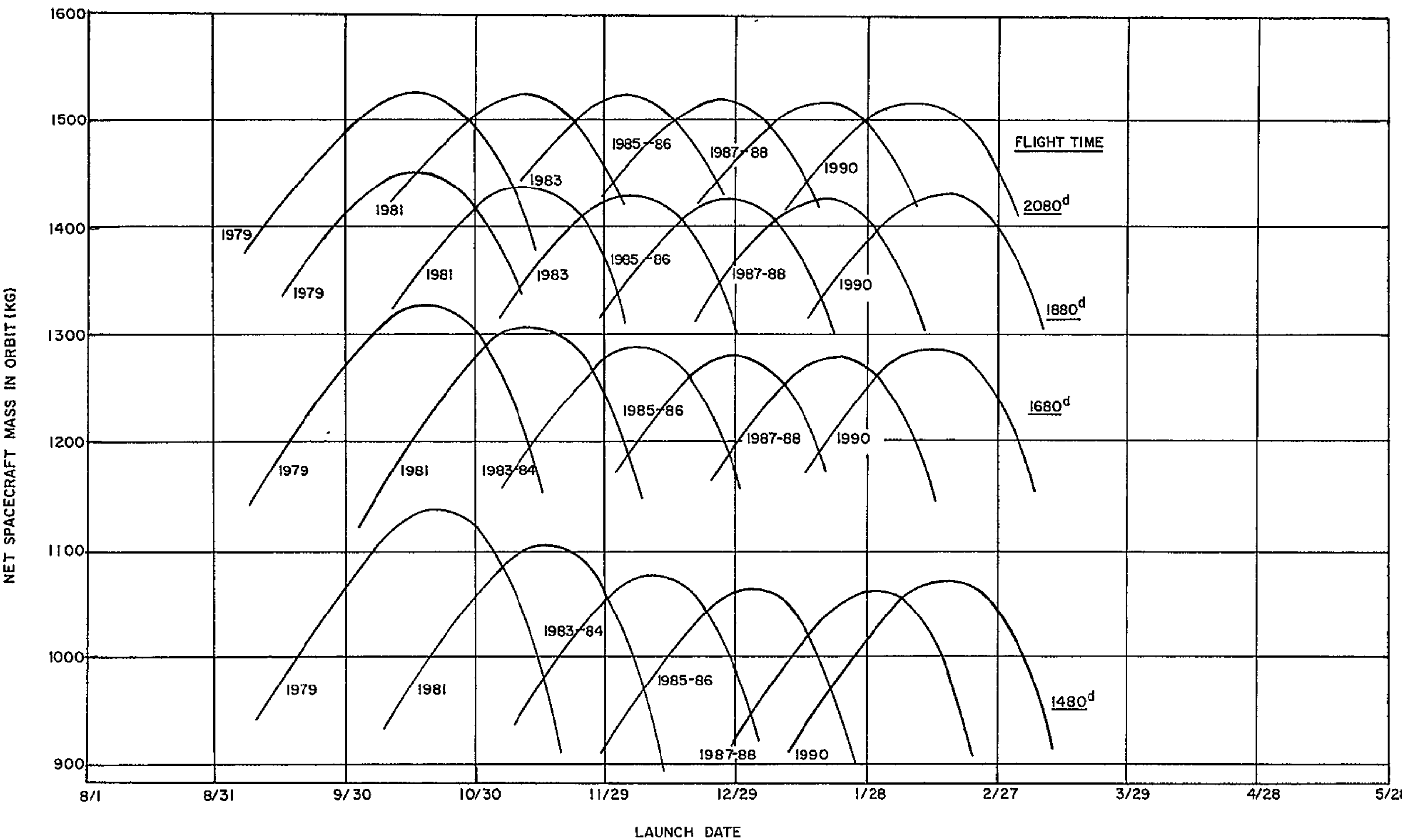


FIGURE B-3 (CONTINUED)