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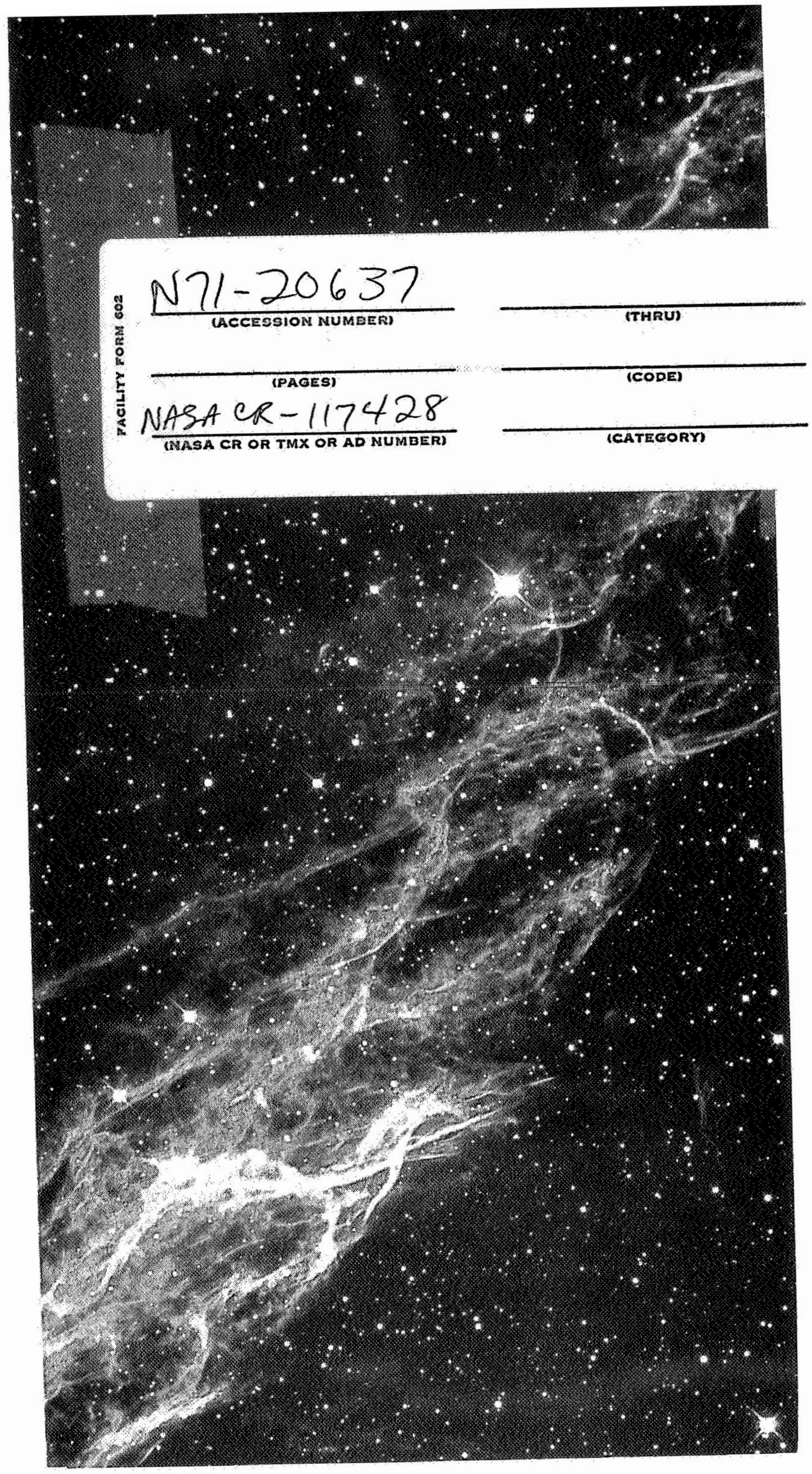

Report No. M-25

A PRELIMINARY STUDY OF COMPOSITE ORBITER/LANDER MISSIONS TO SATELLITES OF THE OUTER PLANETS

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A PRELIMINARY STUDY OF COMPOSITE ORBITER/LANDER MISSIONS TO SATELLITES OF THE OUTER PLANETS

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SUMMARY

A preliminary study has been made of the scientific objectives and payload requirements for landing unmanned spacecraft on satellites of the four giant outer planets. Scientific and operational rationale are developed for selecting six major satellites for composite orbiter/lander missions. Specific missions to Io (Jupiter I), Europa (Jupiter II), Ganymede (Jupiter III), Callisto (Jupiter IV), Titan (Saturn VI) and Triton (Neptune I) are considered. Two classes of lander missions (of equal mass in satellite orbit) are discussed, 1) a single soft-lander, and 2) multiple (10) rough-landers.

The major objective of such missions would be the collection of scientific data pertinent to:

1) a better understanding of the mode(s) of formation of the satellites and smaller planets of the solar system,
2) the study of the origin of planetary/satellite systems,
3) comparing theories for the evolution of planet/satellite systems with those for the evolution of the solar system itself.

Lander experiments should emphasize identification of fundamental chemical and physical properties of the satellite. The orbiting bus, regarded as an essential communication link between the lander and earth, could enhance these measurements by generating global surface feature and thermal maps through orbital imagery. With these data the satellites could be compared with the smaller terrestrial planets and the moon, hopefully providing new insight into the vast differences of the inner and outer planets. Surface experiments and instrumentation appropriate for initial landings are briefly discussed.

A second objective for satellite lander missions is the use of these bodies as bases for the remote observation of their parent planets. A satellite base has the inherent advantage of platform stability compared with an orbiting spacecraft. Also, if the satellites' rotation periods are locked to their orbital periods (e.g. the moon and earth), as is predicted, then the parent planet is continuously observable from any landing site on the "front-face" of the satellite. Since the six satellites selected all apparently revolve well outside the intense regions of planetary radiation belts, radiation hazards should not be a major concern. The constant altitude of these regular satellites above their parent planets would simplify imagery requirements for planetary observations.

The satellites were also considered as bases for monitoring the magnetospheres surrounding the parent planet. However, it was concluded that, at least until the existence and characteristics of outer planet magnetospheres have been better established, the disruptive effects of the satellites presence would make such measurements difficult to interpret. Hence a payload consisting solely of particle and field instruments was considered inappropriate for any lander mission.

Trajectory and payload analyses were performed for the composite orbiter/lander mission to each of the six regular satellites identified above. The class of soft-lander missions were sized to a useful landed payload of 1000 lbs., exclusive of terminal guidance (descent radar), variable-thrust propulsion and landing gear. Experiment instrumentation was limited to 100 lbs. For the class of rough-lander missions 60 lbs. useful payload (exclusive of the impact limiter) was allowed at impacts $\leq$ 200 ft/sec. The associated science was limited to 10-15 lbs. In either case the supporting orbiter was assumed to weigh 1500 lbs.

Payload requirements were determined by separating the mission into four distinct phases and applying various propulsion systems* to each phase. The phase breakdown, in reverse order of occurrence, is as follows:

1) Terminal landing maneuver; variable-thrust chemical propulsion considered for soft-lander, free-fall assumed for rough-lander.
2) Deorbit and braking maneuvers; chemical propulsion considered for both deorbit impulse and constant-thrust braking maneuver just prior to terminal descent.
3) Polar satellite orbit insertion; a) chemical three-impulse maneuver sequence from planet approach of either ballistic or solar-electric low-thrust interplanetary transfers, or b) spiral low-thrust approach from nuclear-electric transfer followed by single-impulse chemical propulsion capture maneuver.
4) Interplanetary transfer; a) ballistic, b) solar-electric low-thrust, and c) nuclear-electric low-thrust flight modes considered.

Payload results indicated that a nominal total useful weight of 4000 lbs was required in a 100-km polar circular satellite orbit to perform the defined soft-lander missions (this includes the 1500 lbs communications relay and mapping orbiter). In order to apply the interplanetary trajectory and payload analyses equally to each mission class, the rough-lander missions were also permitted a total useful in-orbit weight of 4000 lbs. Analysis of the rough-lander propulsion requirements showed that ten landers, their carrier structure and the orbiter were within this weight allowance.

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* Candidate chemical propellants include earth-storable, solid, space-storable and cryogenic.

Assuming that combinations of earth-storable, space-storable, cryogenic and solid propulsion systems can be made available for satellite capture, deorbit, braking and landing maneuvers, the payload feasibility of either lander-class mission can be summarized in terms of the interplanetary flight mode employed. This is done in Table S-1.

Missions using ballistic interplanetary trajectories are conceptually possible to all six selected satellites using Saturn-class launch vehicles. A mission to Callisto is feasible with the Intermediate-20/Centaur if cryogenic propulsion is used for the capture and braking maneuvers. The Saturn V provides mission capability to Europa, Ganymede and Callisto without regard to the type of propulsion used at the satellite. Adding a Centaur to the Saturn V makes possible missions to all four Galilean satellites of Jupiter and the more distant satellites, Titan (Saturn) and Triton (Neptune), with flight times ranging from about 2 years to the Galilean satellites to 11 years to Triton.

Solar-electric low-thrust missions are possible to Ganymede, Callisto and Titan with the Intermediate-20/Centaur as a launch vehicle. The flight times are comparable to the ballistic flight mode for Callisto and somewhat longer for the other three satellites. The Titan 3F/Centaur launch vehicle may perform a marginal solar-electric mission to Callisto, consisting of an orbiter and two or three rough-landers, but this mission has not been studied in detail.

The nuclear-electric low-thrust flight mode makes possible missions to all six satellites with a Titan-class launch vehicle. Missions to the satellites of Jupiter and Saturn require the Titan 3F vehicle (seven-segment solids). A mission to Triton requires the Titan 3F/Centaur.* Flight time requirements to the

* Subject to confirmation that integration of Titan 3F/Centaur nuclear-electric stage is feasible from a flight launch dynamics standpoint.

TABLE S-1

MISSION SUMMARY

Interplanetary Transfer Mode Launch Vehicle Combination	Io	Europa	Ganymede	Callisto	Titan	Triton
BALLISTIC:						
Intermediate-20/Centaur	No	No	No	$600\text{-}700^d$	No	No
Saturn V	No	$600\text{-}700^d$	$500\text{-}700^d$	$500\text{-}600^d$	No	No
Saturn V/Centaur	$600\text{-}700^d$	$500\text{-}700^d$	$500\text{-}700^d$	$500\text{-}700^d$	$3\frac{1}{2}\text{-}4^y$	$11\text{-}12^y$
SOLAR-ELECTRIC:						
Intermediate-20/Centaur	No	$900\text{-}1100^d$	$900\text{-}1000^d$	$600\text{-}700^d$	$4\text{-}4\frac{1}{2}^y$	No
NUCLEAR-ELECTRIC:						
Titan 3F	$1100\text{-}1200^d$	$900\text{-}1000^d$	$900\text{-}1000^d$	$800\text{-}900^d$	$3\frac{1}{2}\text{-}4^y$	No
Titan 3F/Centaur	NA	NA	NA	NA	NA	$8\text{-}9^y$

Galilean satellites are somewhat longer than the ballistic and solar-electric counterparts, this being attributed to the use of spiral earth-departure and Jupiter-approach maneuvers employed with the nuclear-electric mode. For a nuclear-electric flight to Titan (Saturn) this time deficit is made up on the interplanetary transfer. For a Triton (Neptune) mission the flight time is from 2 to 4 years shorter with nuclear-electric propulsion.

Based on the results of this study, it is concluded that composite orbiter/lander missions to the outer planet satellites are deserving of further study. Specifically, we recommend a prephase-A mission study for missions to Ganymede (Jupiter) and Titan (Saturn). Primary emphasis should be given to definition of scientific objectives, instruments, subsystem requirements, operations, propulsion system tradeoffs, and comparisons of the exploration potential of a soft-lander versus multiple rough landers.

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1. INTRODUCTION

The purposes of this study are: 1) to summarize the characteristics of the outer planet satellites and assess their role in the exploration of the outer planets, 2) to define mission configurations for landing unmanned science experiments on the satellites, and 3) to establish approximate payload requirements necessary to perform these missions to a selected set of the most interesting satellites. This study is an extension to earlier work by Price and Spadoni (1970) which was restricted to the feasibility of direct soft-landings on the Galilean satellites of Jupiter. It is to be determined from the results of these analyses whether outer planet satellite lander missions merit further study, and if so which satellites and mission configurations are appropriate for more detailed mission analysis.

Characteristics and exploration uses of the outer planet satellites are reviewed in Section 2. The properties of the satellites are discussed and compared with the smaller planets Mercury and Mars, and with the Moon. The possibilities of using the satellites as bases for remote observation of their parent planets, and for monitoring magnetospheres surrounding the outer planets are also presented.

Section 3 deals with mission synthesis. Here mission configurations are considered and a rationale developed for selecting a composite orbiter/lander profile as the baseline for subsequent payload analysis. Two classes of landers are identified: 1) a soft-lander of a design similar to Surveyor and Viking, and 2) multiple rough-landers (total weight in satellite orbit equal to the soft-lander) similar to Ranger and proposed designs for Apollo missions. A number of science instruments appropriate for an initial satellite lander mission are briefly discussed. A science payload of about 100 lbs. is envisioned for soft-lander missions. One or two instruments totaling 10-15 lbs. are suggested for the rough-landers. A total weight of about 4000 lbs. in orbit is determined necessary to conduct either the soft-lander or multiple rough-lander missions.

Analysis of trajectory and payload requirements is the subject of Section 4. Results are presented for missions to the six largest regular satellites, which include Io (Jupiter I), Europa (Jupiter II), Ganymede (Jupiter III), Callisto (Jupiter IV), Titan (Saturn VI), and Triton (Neptune I). Three interplanetary flight modes are considered, ballistic, solar-electric low-thrust and nuclear-electric low-thrust. Results are summarized in terms of launch vehicle and flight time requirements for each flight mode. The effects of using several different propulsion systems to perform satellite capture, deorbit, braking and terminal descent maneuvers are discussed. Earth-storable, solid, space-storable and cryogenic systems are considered in various combinations.

Study conclusions and recommendations are presented in Section 5.

2. SATELLITE CHARACTERISTICS AND USES

2.1 The Satellites as Individual Planetary Bodies

2.1.1 Orbital Parameters

The orbital parameters of the satellites of Jupiter, Saturn, Uranus and Neptune are listed in Table 1. For each planet, the satellites are listed in order of increasing radial distance. The data for the Galilean satellites (Jupiter I through IV) are taken from Melbourne et al (1968). Data for all other satellites are taken from Allen (1963). Unless otherwise noted, the inclination of the plane of the orbit is measured from the equatorial plane of the parent planet orbits are direct, unless the inclination is proceded by the symbol R in which case they are retrograde.

2.1.2 Basic Physical Parameters

The basic physical parameters of the satellites of Jupiter, Saturn, Uranus and Neptune are listed in Table 2. Again, for each planet, the satellites are listed in order of increasing radial distance. The masses are taken from a review by Brouwer and Clemence (1961). The radii for the Galilean satellites (Jupiter I through IV) are taken from a review by Price (1970) of all measured values. Radii for all other satellites are taken from Allen (1963).

The radius of Titan (Saturn VI) is derived from micrometric measurements made by experienced observers; its value is accurate to 10-20 per cent. All other satellites are either too small or too faint for micrometric measurements to be made. Their radii must be estimated from their apparent brightnesses, with assumptions made concerning their surface albedos at photographic wavelengths. Such estimates are uncertain to a factor of 2 or 3.

TABLE 1

SATELLITES OF THE OUTER PLANETS: ORBITAL PARAMETERS

PLANET	SATELLITE NUMBER	SATELLITE NAME	SEMI-MAJOR AXIS OF ORBIT IN 10^6 KM	SEMI-MAJOR AXIS OF ORBIT IN PLANET RADII	ECCENTRICITY OF ORBIT	ORBIT INCLINATION (IN DEGREES)	SIDEREAL PERIOD OF REVOLUTION (IN DAYS)
JUPITER	V	AMALTHEA	0.181	2.537	0.003	0.4	0.498179
	I	IO	0.422	5.915	~0	~0	1.769138
	II	EUROPA	0.671	9.404	0.0003	~0	3.551181
	III	GANYMEDE	1.070	14.996	0.0015	~0	7.154553
	IV	CALLISTO	1.883	26.391	0.0075	~0	16.689018
	VI		11.470	160.757	0.158	28	250.59
	VII		11.740	164.541	0.206	26	259.7
	X		11.850	166.083	0.135	28.5	255
	XII		21.200	297.127	0.16	R33*	631
	XI		22.560	316.188	0.207	R16.5*	692
	VIII		23.500	329.362	0.40	R33*	737
	IX		23.700	332.165	0.27	R25*	758
SATURN	X	JANUS	0.160	2.65	~0	~0	0.748958
	I	MIMAS	0.186	3.079	0.0201	1.5	0.942422
	II	ENCELADUS	0.238	3.940	0.0044	0.0	1.370218
	III	TETHYS	0.295	4.884	0.0	1.1	1.887802
	IV	DIONE	0.377	6.242	0.0022	0.0	2.736915
	V	RHEA	0.527	8.725	0.0010	0.3	4.517500
	VI	TITAN	1.222	20.232	0.0290	0.3	15.945452
	VII	HYPERION	1.481	24.520	0.104	0.5	21.27666
	VIII	IAPETUS	3.560	58.940	0.0283	14.7	79.33082
	IX	PHOEBE	12.950	214.404	0.1633	R30	550.41
URANUS	V	MIRANDA	0.128	5.378	<0.01	~0	1.414
	I	ARIEL	0.192	8.067	0.003	~0	2.52038
	II	UMBRIEL	0.267	11.218	0.004	~0	4.14418
	III	TITANIA	0.438	18.403	0.0024	~0	8.70588
	IV	OBERON	0.586	24.622	0.0007	~0	13.46326
NEPTUNE	I	TRITON	0.353	15.901	0.0	R20.1	5.87683
	II	NEREID	5.600	252.252	0.76	27.5	360

NOTATION:

* ANGLE OF INCLINATION MEASURED FROM ORBITAL PLANE OF PARENT PLANET

R RETROGRADE ORBIT

TABLE 2

SATELLITES OF THE OUTER PLANETS: BASIC PHYSICAL PARAMETERS

PLANET/SATELLITE NUMBER		SATELLITE NAME	MASS (IN 10^{25} g)	RADIUS (IN KM)
	V	AMALTHEA		70
	I	IO	7.22 ± 0.57	1800 ± 163
	II	EUROPA	4.70 ± 0.09	1549 ± 98
J	III	GANYMEDE	15.45 ± 0.19	2621 ± 367
U	IV	CALLISTO	9.64 ± 0.76	2389 ± 389
P	VI			50
I	VII			10
T	X			7
E	XII			6
R	XI			8
	VIII			10
	IX			8
	X	JANUS		150:
	I	MIMAS	0.00380 ± 0.00007	300
S	II	ENCELADUS	0.00721 ± 0.00204	300
A	III	TETHYS	0.06475 ± 0.00114	500
T	IV	DIONE	0.10338 ± 0.00227	500
U	V	RHEA	0.23:	700
R	VI	TITAN	13.69 ± 0.07	2440
N	VII	HYPERION		200
	VIII	IAPETUS		500
	IX	PHOEBE		100
U	V	MIRANDA		100
R	I	ARIEL		300
A	II	UMBRIEL		200
N	III	TITANIA		500
U	IV	OBERON		400
S				
N	I	TRITON	13.56 ± 2.36	2000
E	II	NEREID		100
P				
T				
U				
N				
E				

Clearly, substantial uncertainties and gaps exist in our knowledge of the basic physical parameters of the satellites. Much more accurate data on the masses and radii of these satellites will be required of earlier outer planet missions before lander missions can be attempted.

2.1.3 Regular and Irregular Satellite Groups

On the basis of their orbital and physical parameters, reviewed in Sections 2.1.1 and 2.1.2, the satellites of the outer planets divide naturally into two distinct groups, Regular and Irregular.

The Regular group has the following properties:

1. The orbits are direct with respect to the direction of rotation of the parent planet.
2. The orbits are very nearly circular.
3. The orbits are very close to the equatorial plane of the parent planet.
4. The periods of rotation are probably equal to the periods of revolution (see Section 2.2).
5. The radii range from $\sim$ 50 km to $\sim$ 2600 km.

The Irregular group has the following properties:

1. The orbits are either direct or retrograde with respect to the direction of rotation of the parent planet.
2. The orbits are appreciably non-circular.
3. The orbits are inclined, in some cases substantially, to the equatorial plane of the parent planet.
4. The satellites are always the outermost in any system.
5. The satellites are always small i.e. less than $\sim$ 500 km in radius.

The natural division of the satellites into distinct groups does not merely provide a convenient method for labelling them. It also provides a means for discussing their respective origins. On the basis of current ideas concerning the formation of satellite systems (Kuiper 1956) the two groups originated through different processes. The Regular satellites are believed to have been formed at nearly the same epoch as their parent planets, as the final stage in the systematic process of formation of the solar system from the solar nebula. The Irregular satellites, on the other hand, are believed to be captured asteroids, the gravitational captures having occurred fairly recently on an astronomical time scale.

The satellites are grouped as Regular and Irregular in Table 3. For each planet, the satellites are listed in order of radial distance. That the two satellites of Neptune have been put in the Regular class may be questioned. By comparing their properties with those of the two groups one might suppose that these two moons should more properly be put in the Irregular class. However, the satellite system of Neptune appears to be a special case. The hypothesis has been advanced (Kuiper 1956) that this system has been drastically modified since its formation. It is speculated that originally Neptune was the parent of three satellites, rather than the two it now possess. Soon after the formation of the system the satellites suffered severe dynamical interaction. The result was the drastic modification of the system. One satellite was ejected - now the planet Pluto - while the orbital parameters of the remaining two were completely changed. The theory explains the physical and orbital parameters of Pluto. In addition, it accounts for the current peculiar properties of the satellite system of Neptune. If, indeed, the disruption hypothesis is correct the two satellites of Neptune should be classified as Regular rather than Irregular, on the basis of their probable origin.

Lander missions may be unnecessary to test the hypothesis that the Regular and Irregular satellites had basically different

TABLE 3

SATELLITES OF THE OUTER PLANETS: DIVISION INTO REGULAR AND IRREGULAR GROUPS

Satellite Group	PARENT PLANET			
	Jupiter	Saturn	Uranus	Neptune
REGULAR	V (Amalthea)	X (Janus)	V (Miranda)	I (Triton)
	I (Io)	I (Mimas)	I (Ariel)	II (Nereid)
	II (Europa	II (Enceladus)	II (Umbriel)	
	III (Ganymede)	III (Tethys)	III (Titania)	
	IV (Callisto)	IV (Dione)	IV (Oberon)	
		V (Rhea)		
		VI (Titan)		
		VII (Hyperion)		
IRREGULAR	VI	VII (Iapetus)		
	VII	IX (Phoebe)		
	X			
	XII			
	XI			
	VII			
	IX			

origins. Visual imaging of the satellites during fly-by reconnaissance missions could provide the necessary scientific data. The large Regular satellites, for example the Galilean satellites of Jupiter and Titan (Saturn), appear to be spherical bodies closely resembling small planets (Section 2.1.4). If all the Regular satellites, even the smallest, were formed by the same physical process as the Inner planets, they will appear as spherical objects i.e. miniature planets. The Irregular satellites, however, will look very different, if they are captured asteroids. They will appear to be irregularly shaped objects, probably parts of a larger solid body which broke up. Evidence that the asteroids have irregular shapes comes from photometric studies of their rotational periods. Some photometric evidence also exists to suggest that the Irregular satellites are non-spherical. Iapetus (Saturn VIII) shows a very large variation in its apparent brightness with position in its orbit (Harris 1961). The range in brightness is over 2 magnitudes - a factor of over 6. Since the brightness variation repeats from one orbit to the next the probable explanation is that the periods of rotation and revolution of the satellite are equal, so that it keeps the same face permanently turned towards Saturn (see Section 2.2). For such a large range in brightness, the satellite is almost certainly of irregular shape.

2.1.4 Comparison with the Smaller Planetary Bodies and Asteroids

The Regular satellites span a very broad range in size, a factor of nearly 40 separating the radii of the largest, Ganymede, from the smallest, Amalthea. Table 4 compares the masses, radii and mean densities of the six largest satellites with those of the Moon, Mercury and Mars. The planets and satellites are listed according to decreasing mass. Basic physical data for the planets are taken from Allen (1963).

TABLE 4

MOON, MERCURY AND MARS, AND THE SIX LARGEST

SATELLITES: LISTED IN ORDER OF DECREASING MASS

Celestial Body	Mass (in 10^{25} g)	Radius (in Km)	Mean Density (in g/cc)
Mars	63.95	3400	3.95
Mercury	31.68	2420	5.3
Ganymede (Jupiter III)	15.45	2621	2.1
Titan (Saturn VI)	13.69	2440	2.3
Triton (Neptune I)	13.56	2000	4.1
Callisto (Jupiter IV)	9.64	2389	1.7
Moon	7.35	1738	3.343
Io (Jupiter I0	7.22	1800	3.0
Europa (Jupiter II)	4.70	1549	3.0

Two of the satellites listed in Table 4, Ganymede and Titan, are intermediate in size (radius) between Mercury and Mars. Three other satellites, Callisto, Triton and Io, are intermediate in size between the Moon and Mercury. Other conclusions emerging from Table 4 are as follows:

1. The two largest satellites, Ganymede and Titan, are closely similar in mass, radius and mean density.
2. Io is closely similar to the Moon, in mass, radius and mean density.
3. Europa, although only about half as massive as the Moon, has very nearly the same density.
4. Callisto, intermediate in size between the Moon and Titan, has the lowest density of the six largest satellites.
5. The mean densities of the Galilean satellites Io, Europa, Ganymede and Callisto, decrease with increasing distance from Jupiter.
6. Apart from Triton (whose radius is uncertain by a factor of ~ 2) the mean densities of the satellites are all lower then that of the Moon, in some cases substantially.

Leaving aside Triton, because of uncertainty in its radius, the other 5 largest satellites may be divided into two sub-groups on the basis of their mean densities: 1) Io and Europa, with mean densities ~ 3 grm cm^{-3}, and 2) Ganymede, Titan, and Callisto, with mean densities ~ 2.2 grm. cm^{-3}. Io and Europa are probably rocky bodies similar to the Moon; Ganymede, Titan, and Callisto may be very different, composed perhaps of a mixture of rock and ice.

The six largest Regular satellites are of substantial size and mass, similar to the smaller planets in the solar system. They should, therefore, be treated as distinct planetary bodies in their own right. Determination of their chemical composition and internal structure would provide data essential for studies of both the mode of formation of the smaller planetary bodies in the solar system, and the origin of satellite systems. Comparison of the physical properties of the Regular satellites with those of the Moon and smaller planets could provide a link between studies of the Inner and Outer planetary groups.

The Irregular satellites, which are supposedly captured asteroids, may be compared with asteroids in solar orbit. The four largest Minor Planets are, in order of decreasing mean radius, Ceres, Pallas, Vesta and Juno. Their respective radii, given by Allen (1963), are 370, 240, 190 and 100 km. Smaller asteroids range downwards in size through bodies the size of Eros (radius 10 km) to boulder - sized objects, and finally down to dust particles. The Irregular satellite group may be divided into two apparently distinct sub-groups, the division being by planet, 1) Jupiter, and 2) Saturn. The seven Irregular Jovian satellites are similar in size to the smaller asteroids, for example Eros. However, the two Irregular satellites of Saturn are comparable in size to the four largest asteroids. Iapetus (Saturn VIII) may, in fact, be somewhat larger than Ceres, while Phoebe (Saturn IX) appears to be much the same size as Juno.

2.2 The Satellites as Bases for the Remote Observation of their Parent Planets

The surfaces of the satellites provide extremely stable platforms from which remote observations of their parent planets could be made. Of the two satellites groups, the Regular satellites would be far more useful for this purpose because of their greater proximity to the parent planet. However, the usefulness of any particular satellite depends not only on its orbital parameters but on its rotation parameters also.

Strong evidence indicates that for both groups of satellites, Regular and Irregular, the periods of rotation of the satellites are generally equal to their periods of revolution.

For Jupiter and Saturn, the main evidence comes from the published work on the variation in the brightness of several of their satellites with position in their orbits, reviewed by Harris (1961). In the case of Jupiter the evidence is provided by observations of the Regular satellites Io(I), Europa (II), Ganymede (III), and Callisto (IV); for Saturn it is provided by observations of the Regular satellite Rhea (V) and the Irregular satellite Iapetus (VIII). For Jupiter, additional evidence is provided by visual observations of surface markings on the Galilean satellites, reviewed by Dolfus (1961). Not only do the satellites appear to keep the same faces permanently turned towards their parent planets, but their rotation axes seem to be closely perpendicular to the plane of their orbits. In the case of Uranus, no studies have been made to determine any possible variation in the brightness of its satellites with position in their orbits. Consequently, we have no direct knowledge of their rotation parameters. However, in view of the regularity of the satellite system - the circularity of the direct orbits, their lack of inclination to the equatorial plane of Uranus, and the relative proximity of the satellites to the planet - it is probably a fair assumption that the satellites do indeed keep their same faces permanently turned towards the parent planet. It is almost certain that in the lifetime of the solar system the tidal influence of Uranus would have produced just such an effect. In the case of Neptune, however, we are not justified in assuming that its two satellites also keep their same faces permanently turned towards the planet. The apparent disruption of the system, discussed in Section 2.1.3, may have been too recent, on an astronomical time scale, for the tidal influence of the planet to have re-asserted itself, and enforced equality between the periods of rotation and the new periods of revolution.

Since the periods of rotation and revolution of each of the Regular satellites are apparently equal, and their orbits are very nearly circular, it follows that as viewed from their parent planets libration is negligible. In addition, the circularity of the orbits ensures that variation in the apparent angular size of the parent planet as seen from each satellite will be small. In the case of the Irregular satellites, although their periods of rotation and revolution may also be equal, the eccentricity of their orbits means that libration is non-negligible. In addition, the apparent angular size of the parent planet will depend on the position of the satellite in its orbit.

The orbital data for the satellites, listed in Table 1, have been used to calculate the apparent equatorial angular diameters of the parent planets, as seen from each of the satellites. In all cases it has been assumed that the distance of the satellite from the parent planet is equal to the semi-major axis of its orbit, and that the satellites all revolve exactly in the equatorial plane of the planet. The linear distances at the sub-satellite points on the parent planets which correspond to an angular resolution of 1 second of arc (1") have also been calculated. The results are listed in Table 5. Data on the equatorial radii of the parent planets were taken from Allen (1963). The apparent angular diameters of the parent planets cover a broad range from nearly 45 degrees, for Jupiter V and Saturn X, to somewhat less than 0.5 degrees for the Irregular satellites of Jupiter and Saturn. For comparison, note that, as seen from the Earth, the Moon subtends an angle of close to 0.5 degrees.

In almost all cases the parent planet, as seen from one of its satellites, will appear to revolve rapidly because its rotation period is much shorter than the periods of revolution of its satellites. It would not be possible to use them to observe individual features on the planetary disk for periods longer than about one-half the rotation period of the planet. This period would range from about 5 hours in the case of Jupiter to about 8 hours in the case of Neptune. The only exception is the case of Jupiter V which revolves around Jupiter in a sidereal period of 0.498179 days at a distance of 2.537

TABLE 5

THE FOUR GIANT OUTER PLANETS AS VIEWED FROM THEIR SATELLITES

PLANET	SATELLITE NUMBER	SATELLITE NAME	ANGULAR EQUATORIAL DIAMETER OF PLANET (IN DEGREES)	DISTANCE AT SUB-SATELLITE POINT ON PLANET CORRESPONDING TO 1" OF ARC RESOLUTION (IN KM)
JUPITER	V	AMALTHEA	45.2	0.5
	I	IO	19.4	1.7
	II	EUROPA	12.2	2.9
	III	GANYMEDE	7.6	4.8
	IV	CALLISTO	4.3	8.8
	VI		0.7	55.3
	VII		0.7	56.6
	X		0.7	57.1
	XII		0.4	102.4
	XI		0.4	109,0
	VIII		0.4	113.6
	IX		0.3	114.6
SATURN	X	JANUS	43.3	0.5
	I	MIMAS	37.2	0.6
	II	ENCELADUS	29.1	0.9
	III	TETHYS	23.5	1.1
	IV	DIONE	18.4	1.5
	V	RHEA	13.1	2.3
	VI	TITAN	5.7	5.6
	VII	HYPERION	4.7	6.9
	VIII	IAPETUS	1.9	17.0
	IX	PHOEBE	0.5	62.5
URANUS	V	MIRANDA	21.3	0.5
	I	ARIEL	14.2	0.8
	II	UMBRIEL	10.2	1.2
	III	TITANIA	6.2	2.0
	IV	OBERON	4.7	2.7
NEPTUNE	I	TRITON	7.2	1.6
	II	NEREID	0.5	27.0

Jovian equatorial radii from the center of the planet. The satellite is close to being in a synchronous orbit, the sidereal periods of rotation of the equatorial/temperate regions of the Jovian cloud layer being $9^h\ 50^m.5$ (0.410060 days)/$9^h 55^m.4$ (0.413472 days), respectively. Because of the particular location of Jupiter V, individual features on Jupiter will remain below the satellite for a period very substantially longer than 5 hours. It can be shown that the rotation periods of Jupiter with respect to an observer on Jupiter V are 2.318554/2.4317126 days. Consequently, individual equatorial/temperate features could be monitored for up to 1.159277/1.215856 days (i.e. for about 2.5 orbits of the satellite), provided solar illumination is not necessary for observation. On the basis of geometrical considerations, the satellites of the Regular group, particularly the innermost, would be especially useful as bases for remote study of their parent planets. Landings of remote sensing instruments should be made on the planet-turned face of each satellite, ideally at the sub-planet point for planetary observations. Here, the planet will be continuously in the zenith with a large apparent angular diameter, and observing conditions will be optimum.

Precisely which of the innermost Regular satellites would be most appropriate for remote study of their parent planet depends also on the intensity of the radiation environment of the planet. If the planet possesses radiation belts, and the satellite under consideration for a lander mission is located within their most intense regions, the lifetime of the electronic components on-board a landed spacecraft could be shortened substantially by radiation damage. Anticipating the discussion in Section 2.3, it appears that only landers on Jupiter V would experience significant levels of radiation. In planning landings on that satellite consideration must be given to shielding the electronic components on-board.

2.3 Measurement of the Physical Properties of the Magnetospheres of the Outer Planets from the Surfaces of their Satellites

Of the four giant outer planets only Jupiter is known to possess a magnetosphere. Warwick (1967) has reviewed present knowledge of the interplanetary environment in the immediate vicinity of Jupiter. Results from radio astronomy indicate that Jupiter has van Allen-type radiation belts surrounding it, which suggests that the planet may have an essentially dipole magnetic field with probable strength at the Jovian cloud layer on the order of 10 gauss. Observations indicate that the axis of the dipole is inclined about 10 degrees to the rotation axis of the planet. The maximum density of the charged particles in the belts appears to occur at approximately 2 Jupiter radii from the center of the planet. Almost all the accumulated particles lie within about 5 Jupiter radii from the planet.

Of the twelve known moons of Jupiter the five Regular satellites appear to be most suitable for use as observation stations for monitoring the physical properties of the Jovian magnetosphere, because of their relative proximity to the planet. Since the periods of revolution and rotation of the Regular satellites are apparently equal, each body presents the same face permanently turned towards the direction of its orbital motion. The ideal location for soft-landing particle/field detectors to monitor the environment of Jupiter would, therefore, appear to be on the "forward" face of the satellite, 90 degrees from either pole of rotation and from the sub-Jupiter point.

There are, however, several reasons why the Regular satellites are not particularly suitable as stations for the study of the Jovian magnetosphere:

1. With the exception of Jupiter V, the other four - the Galilean moons - revolve around the planet well outside the most intense regions of the radiation belts.

2. Since their orbits are essentially circular it would not be possible to study variations in particle densities and field strengths as a function of distance from the center of the planet.
3. Even if the particle/field detectors were soft-landed at the ideal spot on the "front" face of the satellite interpretation of their measurements would be difficult. The instruments could not make "pure" measurements of the radiation belts because interaction of the charged particles with the satellite itself would almost certainly drastically modify the local and non-local particle/field environment of the lander. The severity of the interaction would, of course, depend on both the electrical conductivity of the satellite and the magnitude of its magnetic field. Very likely the instruments would be measuring the interaction of the satellite with the radiation belts, rather than the intricsic properties of the undisturbed belts. Such information would, of course, be of interest, but only after a better physical understanding of the belts has been obtained.

Spatial and temporal measurements of particle densities and field strengths in the vicinity of Jupiter would be better made using either a spacecraft in a highly elliptical orbit in the plane of the Jovian magnetic equator or a series of Pioneer-type fly-by missions. Should magnetospheres be discovered to surround Saturn, Uranus and Neptune, very similar conclusions would apply.

3. MISSION SYNTHESIS

3.1 General Remarks

Up to this point all 29 satellites of the outer planets have received more or less equal emphasis. For payload analysis of initial lander missions, however, it was appropriate to limit consideration to the larger regular satellites. As distinct bodies they are almost certainly the most interesting scientifically and have undoubtedly played a significant role in the origin and evolution of the outer planet systems. All are sufficiently close to their parent planets to make them useful as observation bases for remote measurements of the parent - a secondary consideration for lander missions. Six specific satellites were chosen for the payload analysis. They are Io (Jupiter I), Europa (Jupiter II), Ganymede (Jupiter III), Callisto (Jupiter IV), Titan (Saturn VI), and Triton (Neptune I). Amalthea (Jupiter V) has been discussed at some length as an observing base of Jupiter. It was not considered further here for the following reasons: 1) its small size (~ 70 km) makes it less interesting than its larger neighbors, the Galilean satellites, for satellite exploration, 2) its orbital location, apparently in the Jovian radiation belts, make missions to it a special hazard, 3) its tight circular orbit implies high energy requirements for a lander mission which means less payload capability (this also holds true for the several smaller regular satellites inside of Titan at Saturn).

The primary objectives of the lander mission are concerned with investigation of the satellite itself. Maximum science return almost certainly will require complete freedom of site selection. Although preliminary knowledge of satellite surface characteristics would probably be available from previous flyby and orbiter missions, real-time site selection could best be achieved by preceeding the landing phase of the mission with orbital reconnaissance. Obtaining a complete surface map of the satellite obviously also improves the mission's scientific value. A low (~ 100 km) polar orbit about the satellite would provide the geometrical and operational flexibility necessary for final site selection and landing.

After the final site selection has been made, two choices are available for the landing phase. Either the entire orbiting spacecraft can be landed, or a separate lander can be detached from an orbiter.

Landing the entire spacecraft reduces occultation problems permitting longer interrupted periods of communication with the Earth. At best, the communication periods would be half of the satellite's period of revolution. On the other hand, orbiter/lander type missions allow continued mapping of the satellite surface, and avoid landing the large spacecraft-to-Earth communication subsystem. Also, orbiter-to-Earth communication can be accomplished at any time except during brief periods of occultation by the satellite. The lander-to-orbiter communication link occurs twice per satellite revolution.

For the purpose of this analysis, it was concluded that the latter scheme is preferred. The advantages of continuous orbital mapping and smaller lander weight probably outweigh the difficulties imposed by periodic blackouts in orbiter/lander communication. Using the TOPS spacecraft as a benchmark, the weight of the orbiter was selected to be 1500 lbs.

3.2 Lander Alternatives

Assuming a fixed payload budget is available for the landed portion of the total spacecraft, two alternative lander designs are relevant, 1) a single soft-lander, and 2) multiple rough-landers.

A single soft-lander, patterned after Viking, represents a comprehensive surface investigation capability, limited, however, to a very small area about the landing point. Soft-lander missions to the satellites of Jupiter have already been discussed briefly by Price and Spadoni (1970). In their preliminary feasibility study direct approach trajectories were chosen for soft-landing a payload of 1000 lbs on each satellite. Vertical landings only, direct from the interplanetary trajectory, were treated to simplify the trajectory and payload analysis. Such a direct approach, however, requires a priori site selection which severely limits the mission flexibility necessary to produce maximum scientific benefit. For the payload analysis presented below, a nominal landed weight of 1500 lbs will be used as a guideline to insure an exploration capability similar to that envisioned for first landing missions to Mars, i.e., Viking.

Multiple rough-landers, similar to the early Ranger impacters (impact velocity ~ 100 ft/sec), represent a very limited investigation capability, traded for a distribution of that capability over many sites on the satellite surface. A comparison of soft-landers versus rough-landers produces the following conclusions:

1. Rough lander instruments must be more rugged in design to withstand the high-g impact, yet provide comparable sensitivity to soft-landed

2. The soft-lander emphasizes a complete investigation of science objectives at one site, the rough-landers emphasize investigation of one or two objectives at many sites. For example, rough-landers could be used to set up an extensive array of seismic detectors over the surface.

3. Choice of rough-lander sites is less critical from the standpoint of operational hazards. Sites may be chosen primarily for their potential scientific merit rather than for their topographic

nature.

4. Design development and flight operations of multiple rough-landers are less complex and perhaps more reliable (one lander failure is not a total failure) than for a soft-lander.

Several independent design and development studies have been performed concerning rough-lander capsules, most notably by Philco Aeronautronic (1965), Space-General Corporation (1965), and the Jet Propulsion Laboratory (1968). The results of these studies vary due to dependence upon mission objectives and operations.

The results of the Space-General study will be used as a benchmark since their inputs appear to be most nearly compatible with the requirements of an outer planet satellite rough-lander.

3.3 Science Payload Definition

A number of appropriate experiments can be defined for an exploratory satellite lander mission. The first step is to identify the satellite measurables which are important. These are as follows:

1. composition of surface material
2. internal physical structure
3. intrinsic magnetic field
4. charged particle flux at surface
5. thermal balance and radioactive content
6. atmospheric composition and structure
7. surface topography and physical structure.

Instrument selection and definition is presented in Table 6 for this set of measurables.

Chemical composition and volatile-content of the surface material would be studied using gas chromatography and mass spectrometry, coupled with a scanning colorimeter. Because of the low densities of the satellites (Table 4), the chemical composition of the surface material would be of considerable scientific interest. The composition may turn out to be "rocky-ice", which would be of direct relevance to selection of a theory for the origin of the satellites. To aid study of the origin of the satellites, the internal physical and thermal structures would also be determined using three basic instruments - a seismometer, a magnetometer, and a thermal flow meter. Information on the interaction of the satellites with the interplanetary gas, and/or with trapped charged particles near the parent planets, would be provided by Plasma, and high-energy particle, detectors.

Study of any atmospheres surrounding the satellites would be of considerable scientific interest, since their existence is related to the thermal history of the surface material. Titan

TABLE 6

OUTER PLANET SATELLITE LANDER INSTRUMENT CANDIDATES

MEASURABLE	INSTRUMENT	WEIGHT LBS	POWER WATTS	SOURCE
Surface Composition	Gas Chromatograph/Mass Spectrometer	23	54-79	Viking
	Scanning Calorimeter	1	10-28	Viking
Internal Structure	Seismometer	3	1	Viking
Magnetic Field	Magnetometer	7-10	3	Pioneer F/G
Charged Particle Flux	Plasma Detectors	8-10	3-4	Pioneer F/G
	High-Energy Particle Detectors	3-6	1-3	Pioneer F/G
Thermal Balance	Heat Flow Meter	10	11	Apollo ALSEP
	Drill (3 Meter Penetration)	24	see text	Apollo ALSEP
Atmosphere	Mass Spectrometer	Incl. in Soil Sampler		Viking
	Anemometer, Press/Temp Sensors	8	11	Viking
Surface Topography	Facsimile Camera/Spectral Photometer	13	10-12	Viking
	TOTALS	99-107	104-152	

(Saturn VI) is known to possess a thin atmosphere of methane (Kuiper 1944), and weak evidence for the presence of atmospheres on Io (Jupiter I) and Europa (Jupiter II) has been presented by Binder and Cruickshank (1964, 1966). An atmospheric monitor to measure chemical composition, surface pressure and temperature, and velocity of the winds has been included in the instrument selection.

The imaging facsimile camera would examine the surface topograph in the vicinity of the landing site. The formation and melting of any "hoarfrost" during and after solar eclipse and/or the satellite night would also be monitored with the camera. The associated spectral photometer would study specular reflections of surface material in the visible area surrounding the landing site. These data would be correlated with the in situ soil analysis conducted within the reach of the lander.

A total science package of 100 lbs requiring 150 watts of power was adopted for the soft-lander mission. All equipment except the drill (see Table 6) may operate simultaneously. To provide power for the drill while the thermal flow meter is set up on the surface, all other experiments must be turned off.

A science package of 10-15 lbs. (one or two instruments), requiring less than 15 watts, is budgeted for an individual rough-lander. The instruments in Table 6 which are applicable include the seismometer, magnetometer, particle detectors, atmospheric monitor and the facsimile camera. Rough-lander experiments emphasize the investigation of one or two specific measurables on a global scale.

3.4 Definition of Lander Modes

In the preceeding discussion, it was seen that a composite orbiter/lander type mission is more practical, from an operational standpoint, than landing the entire spacecraft. The following discussion pertains to a more complete definition of this mission mode.

Mission payload feasibility can most easily be discussed in terms of total spacecraft weight delivered into orbit about the satellite for two reasons: 1) total weight in orbit is the common factor which links the discussion of various interplanetary transfer modes with that of various landing systems, and 2) in-orbit weight prior to separation of the lander(s) best reflects the near satellite mission performance requirements (after separation the operations of two separate spacecrafts are considered). In order to adequately define the total spacecraft weight in orbit, propulsion system requirements for the overall landing system were first examined.

3.4.1 Deorbit and Retro Maneuvers

After the surface has been surveyed and the landing site selected, a sequence of maneuvers is required to deploy the lander. This sequence is illustrated in Figure 1. The first impulse (No. 2 in the figure) places the lander (after separation from the orbiter) on a descent orbit to bring it much closer to the surface. Operationally, this impulse occurs approximately 180 degrees from the selected landing site. The periapse altitude of the descent orbit is dependent upon the operational mode of the lander and the size and mass of the satellite. For a soft-lander the periapse altitude must match the altitude of initiation of the soft-landing maneuver (braking and terminal descent), which ranges from 1.6 km to 3.2 km for the six satellites. In the case of a rough-lander, the periapse altitude was constrained to match the height above the surface required for the lander to impact with the desired velocity (100-200 ft/sec). This altitude ranges from 0.8 km to 1.6 km for the six satellites. These low altitudes almost certainly infer guidance difficulties for the rough-landers, a problem which has not been considered here.

Near the low point of the descent ellipse a fixed-thrust braking maneuver (No. 3 in Figure 1) is performed to reduce the

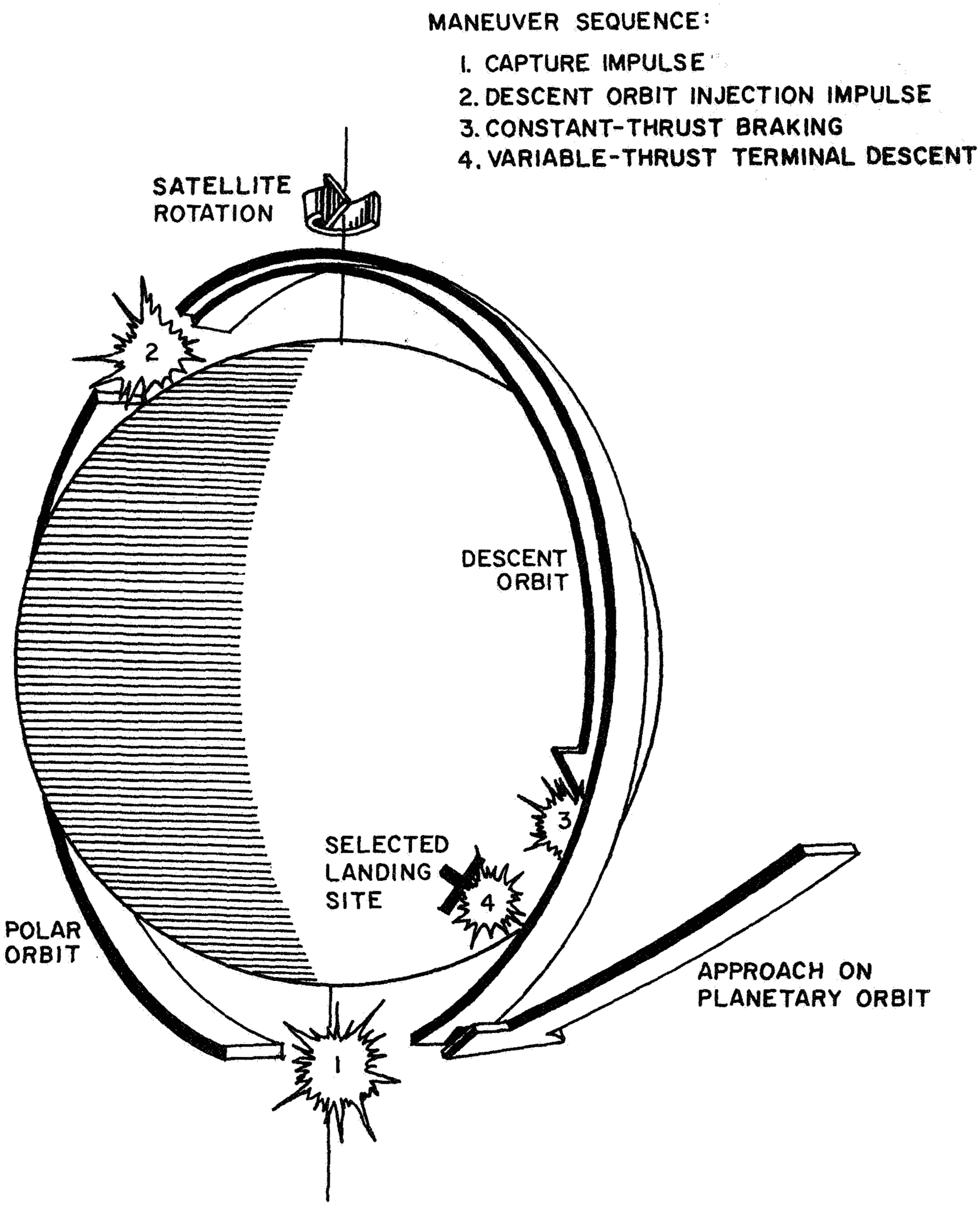

FIGURE 1. SATELLITE LANDING SEQUENCE

lander's velocity. The final velocity of this retro burn is also dependent on the operational mode of the lander. The soft-lander's velocity must be reduced to match the initial velocity of the terminal descent maneuver, whereas the forward velocity (horizontal to the surface) of a rough-lander is reduced to zero so that its payload package can drop as nearly vertical to the surface as possible.

A number of exploratory trajectory computations have led to a nominal set of orbit and propulsion parameters for this phase of the landing maneuver. These include a 100 km altitude circular polar parking orbit and an initial thrust level for the braking maneuver of 1g. The altitude at thrust initiation has already been mentioned above.

3.4.2 Landing Maneuver

The final maneuver (No. 4 in Figure 1) is the terminal descent maneuver. For a soft-lander a variable-thrust, gravity-turn descent trajectory was chosen. Gravity-turn steering implies that the thrust direction is always aligned opposite to the velocity vector. Exploratory computations have led to a set of nominal parameters for this maneuver, including an initial velocity of 300 m/sec, an initial steering angle of 90 degrees (implying motion horizontal to the surface) and a 4:1 throttle ratio.

The landing phase for a rough-lander requires no terminal propulsion. Since landing is initiated at periapse of the descent ellipse, the vertical velocity is essentially zero, and upon completion of the retro burn the horizontal velocity is also eliminated. Thus the rough-lander begins to fall with zero velocity relative to the surface. Space-General Corporation (1965) indicates an impact velocity on the order of 200 ft/sec and an impact shock of up to 3000 earth gravities as reasonable values for a 100 lb rough-lander. Note that no allowance or consideration has been given in this study for the effect of an atmosphere on the landing maneuvers (see Figure 1).

3.5 Payload Scaling

Scaling of the terminal descent propulsion system and landing hardware for the soft-lander was accomplished by use of the following equation, derived from existing soft-lander designs including Surveyor, Apollo and Viking:

$$W_L = \frac{W_{PL} + 50}{1 - 0.2\left(\frac{1 - R}{R}\right) - 0.1\left[1 + \frac{1.5\ g_{sat}}{g_{Moon}}\right]}$$

W_L is the total landed weight, W_{PL} is the total useful landed payload, $R = W_L/W_o = \exp(-\Delta V/g\ Isp)$ is the characteristic mass ratio, g_{sat} and g_{Moon} are the surface gravities of the satellite and Moon respectively, and 50 lbs. has been included for an attitude/velocity control unit. The propellant assumed was Earth-storable N_2O_4- Aerozine 50 with an Isp of 310 secs.

Useful landed weight (W_{PL}) is defined as the weight of the scientific instrument package, subsystems and structure, while total landed weight (W_L) also includes the landing gear, descent powerplant and terminal guidance unit. For this study, the useful payload (W_{PL}) was taken to be 1000 lbs., and the total landed weight was calculated for each satellite accordingly. Total landed weights (W_L) range from 1364 lbs. at Callisto to 1576 lbs. at Triton.

To successfully place a 10-15 lb. science package on the surface via a rough-lander, approximately 50 lbs of internal support equipment are required, and the limiter (the external shell which absorbs the initial impact shock) should weigh about 40 lbs. The total rough-lander package weight is, therefore, about 100 lbs.

Payload scaling for the deorbit and braking maneuvers was accomplished by use of methods developed by Chadwick (1968) for those cases involving liquid chemical propellants, and from the Launch Vehicle Estimating Factors handbook (1970) for those cases using solid propellants. The various propellants studied for these maneuvers, and their respective Isp's are: cryogenic (468), space-storable (400), beryliumized solid (315), and Earth-storable (310).

Table 7 shows the total in-orbit spacecraft weight (i.e. before separation of orbiter and lander) for the various retro propulsion systems. For the liquid propellant systems, the same propulsion unit can perform both the deorbit impulse and the fixed-thrust braking maneuver. The solid propellant motor has been scaled only for the braking maneuver with extra weight allocated for the vernier rockets to perform the deorbit maneuver and attitude control during firing of the solid motor.

Only solid propellant propulsion was considered for the rough-landers. The 100 lb. package was first scaled for the braking maneuver just prior to the rough landing, establishing solid propellant rocket weights ranging from 100 lbs. to 150 lbs., depending on the mass of the target satellite. The combined lander plus retro rocket weight was then scaled for the deorbit impulse, leading to a second, smaller solid rocket weighing approximately 20 lbs. The total weight of a single rough-lander ranges from about 220 lbs. at Europa to about 270 lbs. at Triton.

In a study on multiple satellite configurations, Philco Aeronutronic (1967) has indicated that the weight of the orbiter support equipment for the rough-landers should be approximately 10 percent of the total lander weight. From the in-orbit weights presented in Table 7, it can also be seen that the weight of a soft-lander with its retro propulsion system ranges from about 2300 lbs. to about 3900 lbs., depending on the satellite and type of propulsion system. Therefore, it was assumed that any number of rough-landers up to ten, plus their

TABLE 7

REQUIRED IN-ORBIT PAYLOAD[1] WEIGHTS (LBS) FOR ALTERNATIVE LANDER SYSTEMS

SATELLITE	SINGLE SOFT-LANDER				TEN ROUGH-LANDERS
	CRYOGENIC RETRO SYSTEM	SPACE-STORABLE RETRO SYSTEM	SOLID ROCKET RETRO SYSTEM	EARTH-STORABLE RETRO SYSTEM	
IO	4040	4140	4240	4500	3980
EUROPA	3810	3880	3940	4150	3920
GANYMEDE	4250	4390	4590	4890	4300
CALLISTO	3840	3930	4030	4250	3980
TITAN	4160	4390	4560	4750	4250
TRITON	4650	4840	5070	5480	4470

1) PAYLOAD CONSISTS OF ORBITER, LANDER(S), AND RETRO PROPULSION SYSTEM

support equipment, could be used in place of one soft-lander at any particular satellite for the same total weight in-orbit requirement.

Although no spacecraft design considerations are made within the scope of this report, a note briefly describing one possible sequence of operations for rough-landers is appropriate. A rough-lander should be of the simplest design, probably spin-stabilized. Assuming this to be the case, and further assuming that no active attitude control is provided, the following operations are proposed to deploy rough-landers:

1) proper alignment of lander's spin axis by orbiter's attitude control unit,
2) spin-up of lander package,
3) separation of lander from orbiter, 180° in transit from landing site,
4) ignition of deorbit rocket (thrust opposite to direction of motion),
5) jettison of deorbit rocket,
6) descent along coast ellipse,
7) ignition of braking rocket at proper altitude (thrust is again opposite to direction of motion),
8) jettison of retro rocket,
9) descent to surface and impact.

Although this is a relatively simple approach to the problem, questions of practicality and reliability arise, such as the use of automatic timing devices to trigger each step of the procedure, the stability of the spin-axis during the 180° descent transit, and accurate control of the impact-sensitive final periapse altitude. These questions are as yet unanswered and would have to be carefully examined should the above procedure be utilized.

4. TRAJECTORY AND PAYLOAD ANALYSIS

In the preceeding section composite orbiter/lander type missions to each of the six largest regular satellites of the outer planets were synthesized, and the payloads to perform these missions were developed. Before the question of mission payload feasibility can be answered, methods of delivering the payload to its target must first be examined.

Two types of interplanetary transfer, ballistic and low-thrust, are considered, and two methods of capture into orbit about the satellite, one from ballistic approach, the other from low-thrust approach, are examined. The performance of several launch vehicles, ranging from the Titan 3F (seven segment) to the Saturn V/Centaur, is integrated with the trajectory data to determine payload capabilities. Although several of these launch vehicles, and certain high- and low-thrust propulsion systems studied have not yet been fully developed, it is felt that they could be available by the time such missions are undertaken (probably 1985 - 1990 time frame).

4.1 Interplanetary Transfer Modes

Data for ballistic transfers from earth to the three target planets under consideration are presented in Table 8. The launch energies and approach conditions are respresentative values over the synodic period of launch opportunities for each target planet. The characteristic velocities shown reflect an allowance for a ten day launch window at earth. Precursory examination of the characteristic velocities indicate that launch vehicles of the Saturn class will be necessary to deliver the required payload to the target from a ballistic transfer.

Table 9 presents data for solar-electric low-thrust transfer to Jupiter and Saturn and solar-electric propulsion system parameters for a range of flight times. Solar-electric

TABLE 8

TYPICAL BALLISTIC EARTH-TARGET PLANET TRANSFER DATA

PLANET (LAUNCH OPPORTUNITY)	FLIGHT TIME (DAYS)	LAUNCH ENERGY		ARRIVAL CONDITIONS		INJECTED PAYLOAD WEIGHT[2] (LBS)		
		V_c (FT/SEC)	C_3 (KM^2/SEC^2)	VHP (KM/SEC)	DECLINATION[1] (DEG)	SATURN V/ CENTAUR	SATURN V	INTERMEDIATE-20 / CENTAUR
JUPITER (1984)	500	49,600	107.4	12.76	1.05	30,000	16,300	12,400
	600	48,500	90.0	9.63	1.96	35,000	20,000	14,000
	700	47,025	83.9	7.62	3.56	40,000	27,000	17,000
	800	47,200	85.3	6.50	5.65	39,000	26,500	16,500
	900	48,060	93.0	6.07	7.12	36,000	22,500	15,000
SATURN (1992)	1,250	52,340	132.9	9.67	1.84	23,000	7,000	8.750
	1,625	52,540	134.0	6.80	4.74	22,000	6,500	8,600
	1,825	53,060	140.2	6.08	6.04	21,000	5,100	8,000
NEPTUNE (1986)	3,440	57,480	185.4	12.96	28.8	13,000	--	4,200
	3,840	55,990	169.6	11.29	28.8	15,000	--	5,300
	4,240	55,010	159.6	9.95	28.8	17,000	--	6,250
	4,640	54,360	152.9	8.85	28.7	18,000	2,000	6,750

1) DECLINATION SHOWN IN PLANETOCENTRIC EQUATORIAL COORDINATES

2) FROM OSSA LAUNCH VEHICLE ESTIMATING FACTORS HANDBOOK (1970)

TABLE 9

SOLAR-ELECTRIC PROPULSION TRANSFER DATA AND SYSTEM[1] PARAMETERS

PLANET (LAUNCH OPPORTUNITY)	FLIGHT TIME (DAYS)	LAUNCH ENERGY, C_3 (KM^2/SEC^2)	VHP (KM/SEC)	DELIVERED PAYLOAD[2] (LBS)	INJECTED PAYLOAD (LBS)	POWER AT 1 AU (KWE)
JUPITER (1977 - 1986)	600	39.5	9.85	16,865	30,050	119.8
	800	25.2	6.24	20,196	36,475	142.0
	1000	18.5	4.59	20,194	39,925	160.3
	1200	13.9	3.87	20,795	42,587	178.1
SATURN (1979 - 1990)	1480	46.3	7.65	13,297	27,050	114.4
	1680	41.5	6.39	14,217	29,217	123.5
	1880	38.9	5.51	14,669	30,325	127.8
	2080	36.6	4.93	15,012	31,925	131.9

1) SYSTEM CONSTANTS: PROPULSION SYSTEM SPECIFIC MASS, α : 66.15 LBS/KWE
LOW-THRUST PROPELLANT TANKAGE FRACTION: 0.03

2) LAUNCH VEHICLE: INTERMEDIATE-20/CENTAUR
PAYLOAD IS TOTAL DELIVERED TO PLANET PRIOR TO CAPTURE MANEUVERS.

missions to Neptune were not examined due to excessively long (19 to 25 years) flight times necessary to deliver required payloads.

Again, the data presented are values averaged at constant flight times typical of a period of launch opportunities. The payload and power data were scaled to the Intermediate-20/Centaur launch vehicle from data generated for Titan 3D/Centaur missions to Jupiter and Titan 3F/Centaur missions to Saturn (Friedlander, 1970). Thus the payload and propulsion system parameters are not necessarily optimum values for the flight times and launch conditions under consideration, but good approximations thereof. Impulsive capture maneuvers (explained below) in the planet-satellite system are employed with both the ballistic and solar-electric low-thrust flight modes.

An optimum Titan 3F/Centaur/SEP mission to Callisto was briefly examined to determine the capability of this launch vehicle. The mission characteristic included a 2000 day flight time, indirect heliocentric transfer mode, an initial power at 1 AU of 29.4 kw, a VHP of 3.697 km/sec and a delivered payload weight of 8950 lbs prior to capture into orbit about Callisto.

Nuclear-electric low-thrust transfer data and propulsion system parameters are shown in Table 10. The data were generated from a method developed by Mascy (1970). The payload and propulsion system parameters were optimized for the values of specific mass and power indicated, which assume a nominal technology level. Unlike the ballistic and solar-electric transfers, the nuclear-electric transfers include a spiral escape from a 300 n.m. earth parking orbit and a spiral approach to the orbit of the target satellite.

4.2 Orbit Insertion Maneuver

For a ballistic planet approach (i.e., both ballistic and solar-electric flight modes) it is generally not possible to

TABLE 10

NUCLEAR-ELECTRIC PROPULSION PAYLOAD AND SYSTEM[1] PARAMETERS

TARGET SATELLITE	FLIGHT TIME (DAYS)	DELIVERED PAYLOAD[2] (LBS)	POWERPLANT MASS FRACTION	PROPELLANT MASS FRACTION	EFFICIENCY	INJECTED PAYLOAD[3] (LBS)	LAUNCH VEHICLE
IO	900-1600	1150-10370	0.319	0.632-0.398	0.735-0.806	38240	TITAN 3F
EUROPA	800-1600	600-12280	0.319	0.646-0.349	0.716-0.809	38240	TITAN 3F
GANYMEDE	800-1600	2040-13880	0.319	0.609-0.309	0.732-0.812	38240	TITAN 3F
CALLISTO	800-1600	3540-15410	0.319	0.571.0.270	0.729-0.815	38240	TITAN 3F
TITAN	1100-2500	480-15650	0.319	0.649-0.264	0.743-0.824	38240	TITAN 3F
TRITON	2300-3500	200-10450	0.253	0.721-0.515	0.773-0.809	48270	TITAN 3F/ CENTAUR

1) SYSTEM CONSTANTS: POWERPLANT SPECIFIC MASS, α = 48.8 LBS/KWE
POWER (DELIVERED TO THRUSTOR SYSTEM) = 250 KWE
LOW-THRUST PROPELLANT TANKAGE FRACTION = 0.03

2) PAYLOAD PRIOR TO CAPTURE INTO 100 KM ALTITUDE ORBIT ABOUT TARGET SATELLITE, AFTER JETTISONING NUCLEAR-ELECTRIC STAGE

3) PAYLOAD INJECTED INTO 300 NM ALTITUDE EARTH PARKING ORBIT

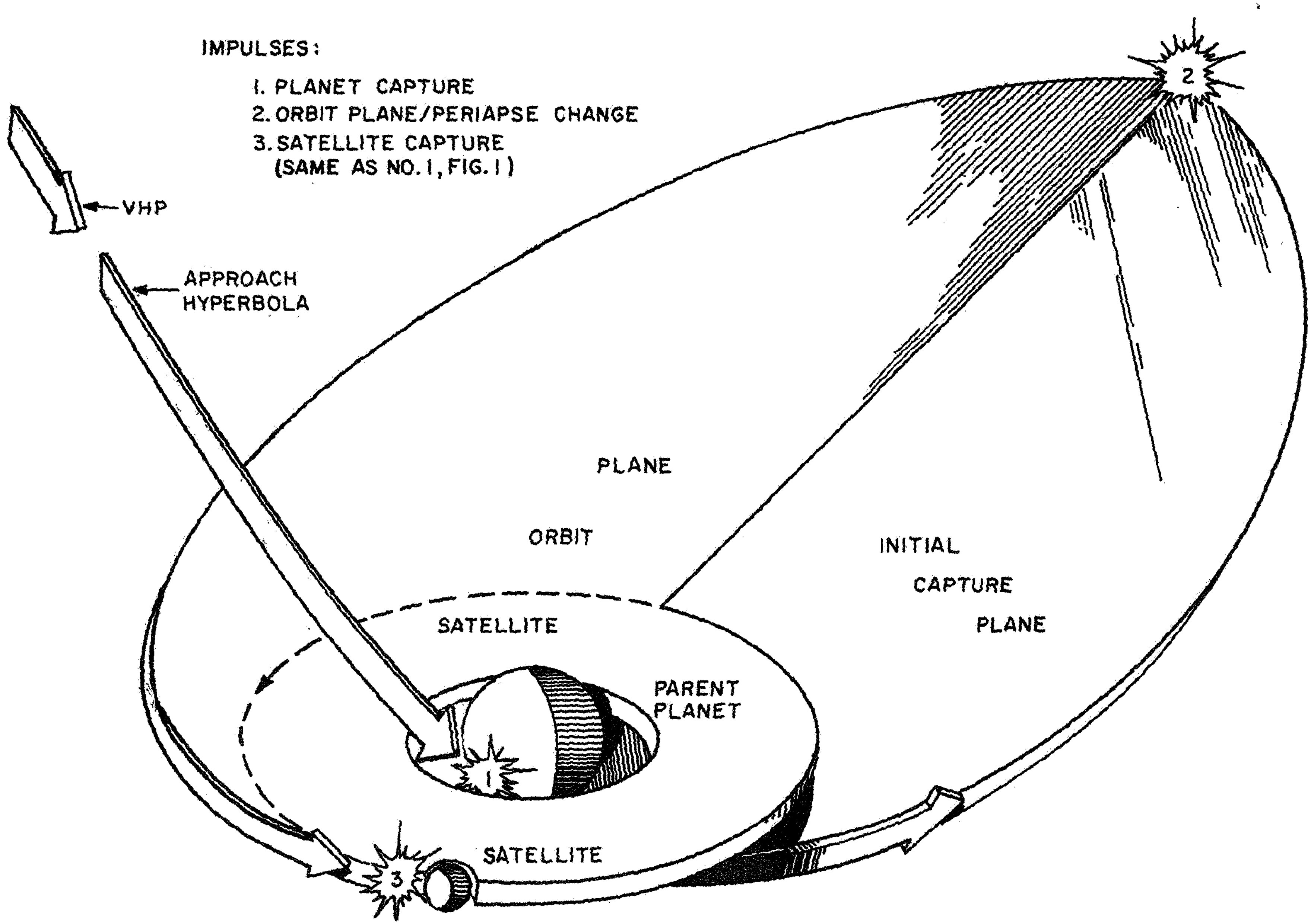

FIGURE 2. SATELLITE CAPTURE SEQUENCE.

efficiently establish a spacecraft orbit about a satellite with a single impulse. Figure 2 illustrates a sequence of three impulses which was adopted for satellite capture. The sequence is a derivative of the bi-elliptic transfer.

The first impulse establishes a loose elliptical orbit about the parent planet in a plane which contains the hyperbolic approach velocity (VHP) and intersects the satellite's orbit at the periapse of the approach hyperbola (Note: this constraint fixes the initial periapse radius). The second impulse occurs at the apoapse of the initial orbit. Its purpose is to change the orbit plane to coincide with that of the satellite and raise the periapse radius to match the satellite orbit. The amount of plane change at the second impulse is a function of both the VHP vector (direction and magnitude) and the radius of closest approach to the parent planet. The third impulse, performed at periapse of the second orbit, is the actual satellite capture maneuver which establishes the desired orbit about the satellite. It is assumed that the spacecraft's periapse passage is matched with the position of the satellite so that capture can occur. Since it is assumed that the satellite's equatorial and orbital planes coincide, there is no penalty in establishing a polar orbit with the third impulse.

Using a low-thrust spiral approach from the nuclear-electric interplanetary transfers, the capture maneuver is simpler than for the direct approach. Upon completion of the spiral to the satellite's orbit, the spacecraft's velocity is assumed to match the satellite's circular velocity about the parent planet. A single impulse can then be used to establish the desired orbit about the satellite. Again, it is also assumed that the spacecraft's position at the end of the spiral maneuver matches the position of the satellite in its orbit, and that the satellite's equatorial and orbit planes coincide so that the satellite's equatorial and orbit planes coincide so that

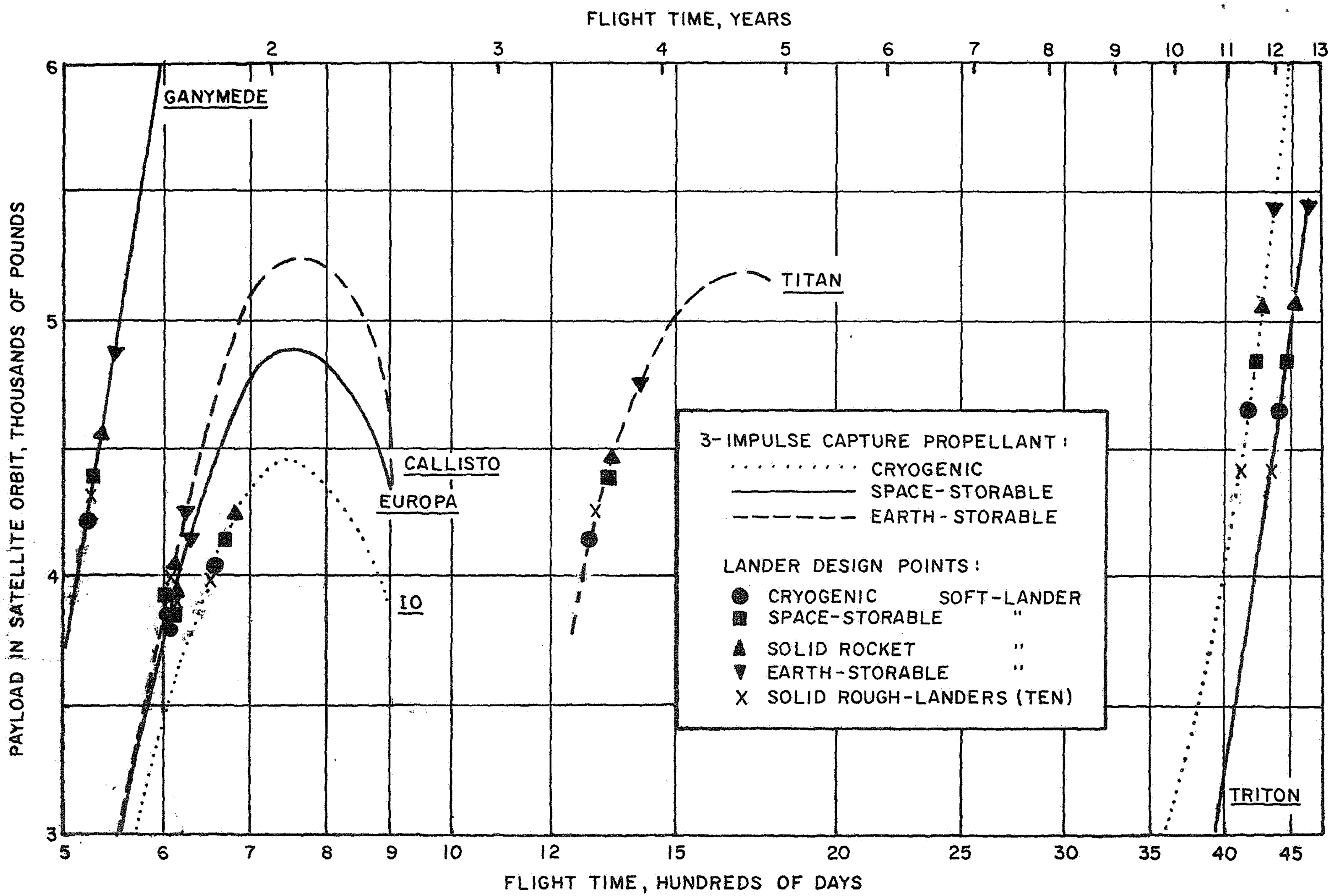

FIGURE 3. SPACECRAFT WEIGHT PLACED IN SATELLITE ORBIT (BALLISTIC TRANSFER MODE- SATURN V/ CENTAUR LAUNCH VEHICLE).

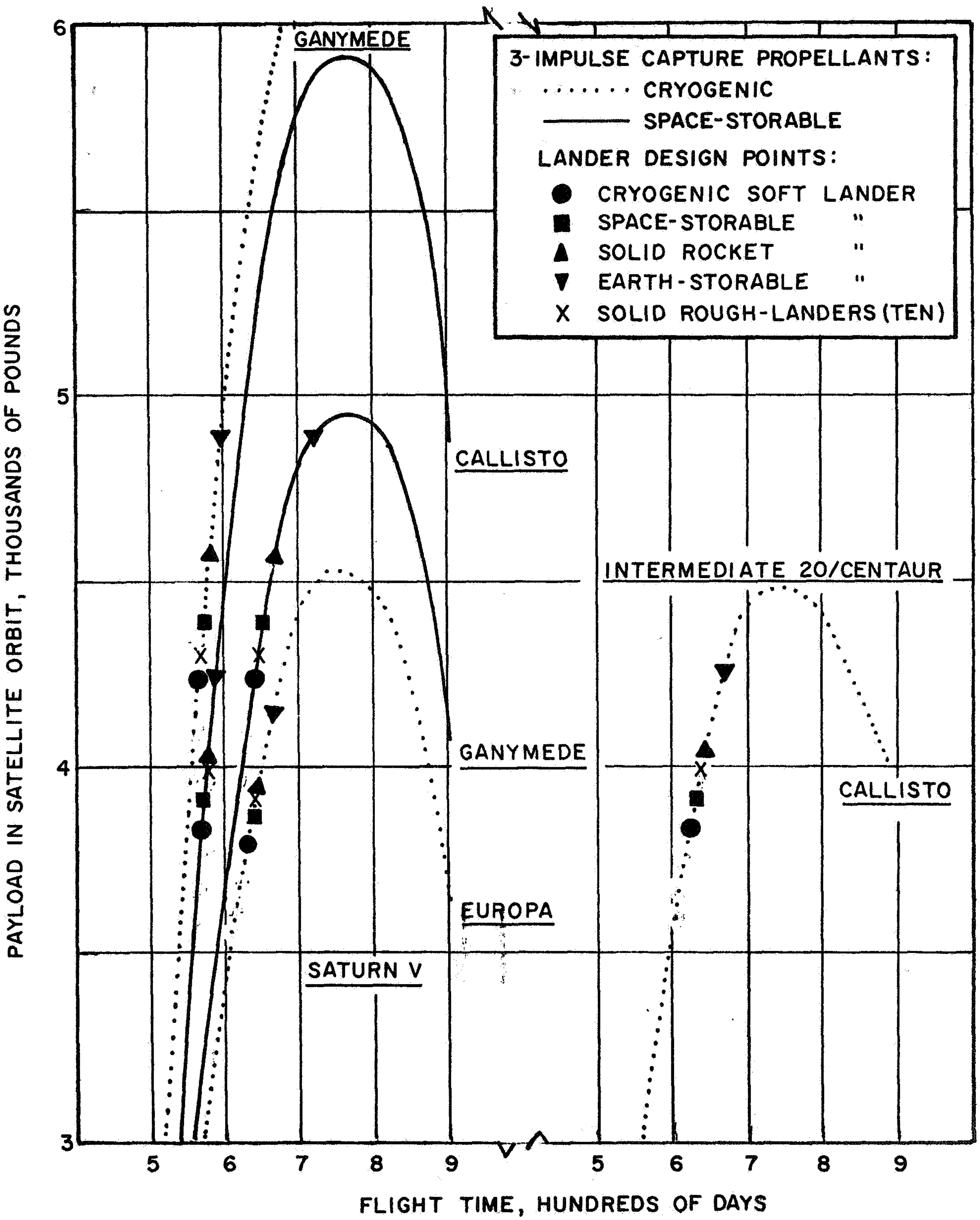

FIGURE 4. SPACECRAFT WEIGHT IN ORBIT (BALLISTIC TRANSFER MODE-SATURN CLASS LAUNCH VEHICLES).

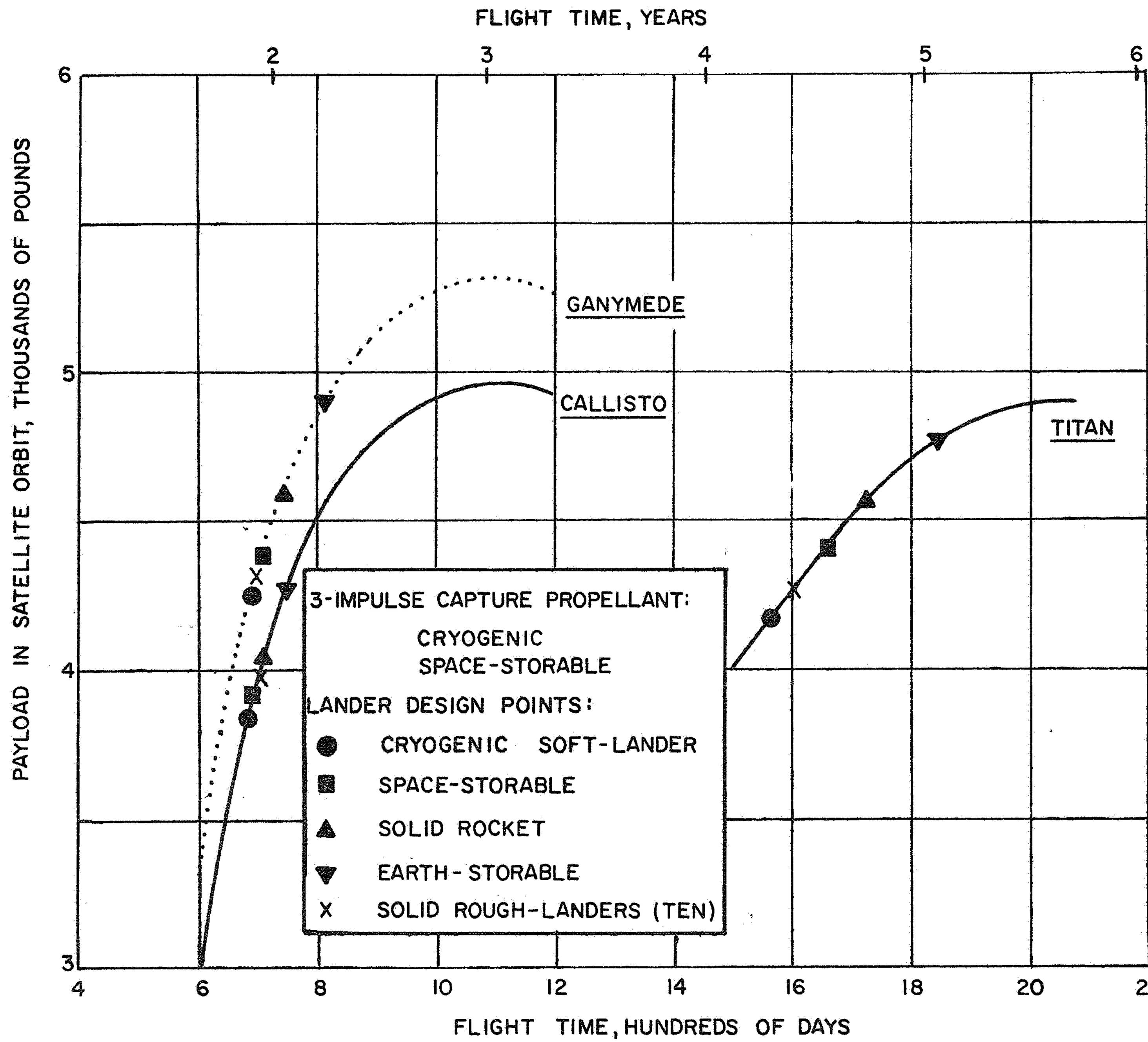

FIGURE 5. SPACECRAFT WEIGHT PLACED IN SATELLITE ORBIT
(SOLAR-ELECTRIC TRANSFER MODE - INTERMEDIATE-20/CENTAUR LAUNCH VEHICLE)

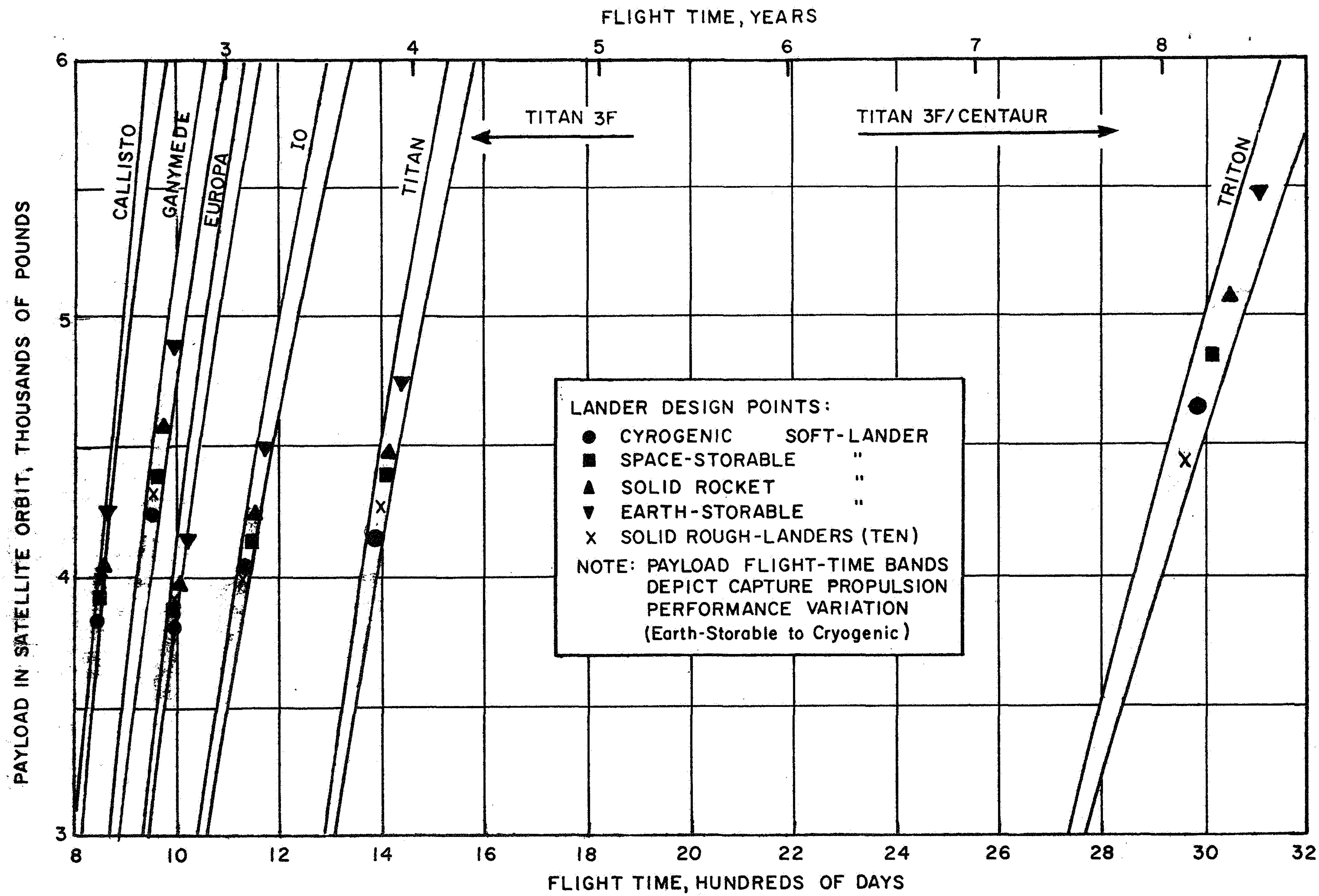

FIGURE 6. SPACECRAFT WEIGHT PLACED IN SATELLITE ORBIT (NUCLEAR-ELECTRIC TRANSFER MODE-TITAN CLASS LAUNCH VEHICLES)

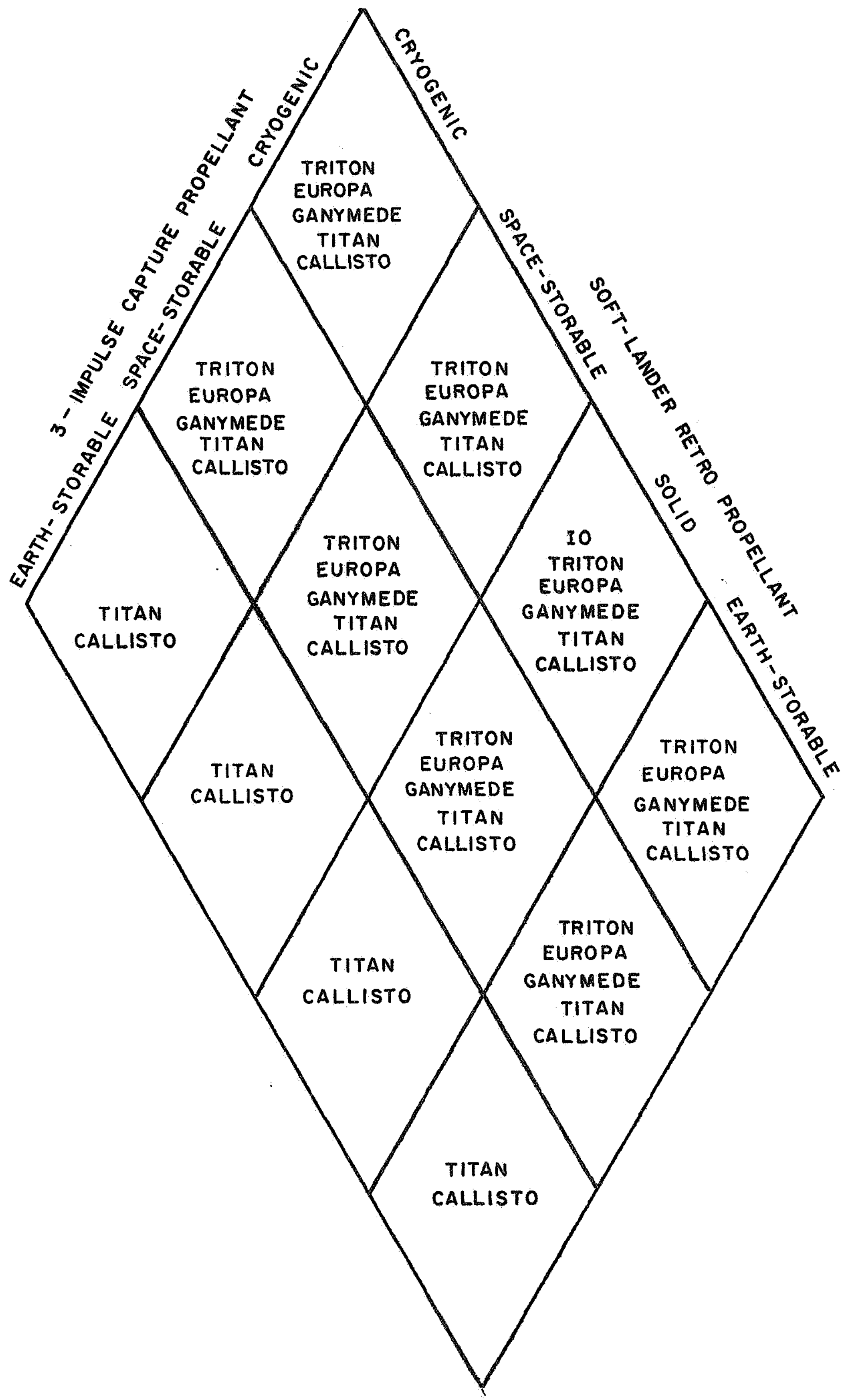

FIGURE 7. MISSION FEASIBILITY USING VARIOUS PROPULSION SYSTEM COMBINATIONS WITH BALLISTIC TRANSFER MODE AND SATURN V/CENTAUR LAUNCH VEHICLE.

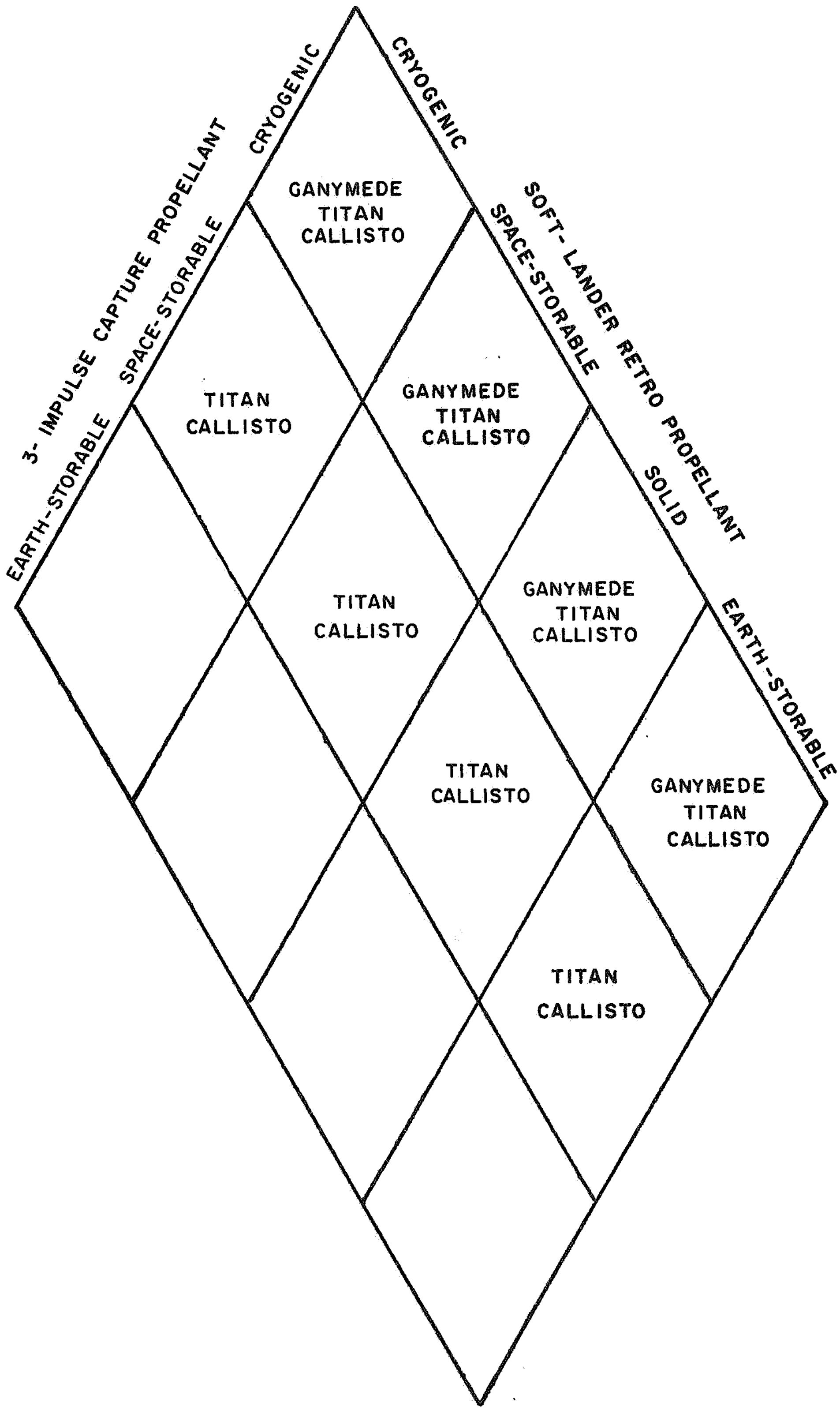

FIGURE 8. MISSION FEASIBILITY USING VARIOUS PROPULSION SYSTEM COMBINATIONS WITH SEP TRANSFER MODE AND INTERMEDIATE-20/CENTAUR LAUNCH VEHICLE.

polar orbits are possible without penalty.

4.3 Feasible Missions

The required in-orbit spacecraft weights for each satellite (Table 7, Section 3.5) were taken as design points on the payload versus flight time data developed in Section 4.1 and 4.2. Figure 3 presents payload versus flight time curves for ballistic transfers using a Saturn V/Centaur launch vehicle. The payload curves are in-orbit weight, i.e., after the 3-impulse capture maneuver, for the various propulsion systems considered. Note that only data for certain of the missions considered are presented. Those not shown are either not possible, such as Io using a space-storable capture, or unnecessarily powerful, such as Titan using a space-storable capture propulsion system. Figure 4 presents similar ballistic transfer data for the Saturn V without Centaur and the Intermediate-20/Centaur launch vehicles. Figure 5 presents similar data for solar-electric transfers combined with the Intermediate-20/Centaur launch vehicle.

The optimum Titan 3F/Centaur solar-electric mission can deliver a payload of 2490 lbs (8950 on approach) into orbit about Callisto. This does not meet the nominal payload requirement, but would be sufficient to perform a marginal mission consisting of a 1500 lb orbiter and two or three rough-landers. Although this mission profile may qualify for more study, it was not considered further in this report.

Figure 6 presents payload versus flight time data for the nuclear-electric low-thrust flight mode. This data is relatively insensitive to the type of propulsion system used for the satellite capture maneuver. The bands in the figure indicate the extremes assumed for propulsion system performance, i.e., cryogenic (left-side of band) to earth-storable (right-side of band) propellant, at each satellite.

To clarify the large number of useable options, Figures 7 and 8 present possible missions for various combinations of propulsion systems for the capture and braking maneuvers, for the ballistic and solar-electric low-thrust transfer modes, respectively, Moving from top to bottom on the figure implies a decreasing level of propulsion system technology, or decreasing level of performance. The target satellites are also listed for each propulsion system combination (diamond), in approximate order of decreasing mission difficulty. Note that the dominating effect of the Jovian gravitational field causes missions to certain of Jupiter's closer satellites to be more difficult to perform than missions to the satellites of the two other outer planets. No figure is necessary for the nuclear-electric low-thrust flight modes since all possible missions are feasible, irregardless of the propulsion system combinations or satellite selected.

5. CONCLUSIONS AND RECOMMENDATIONS

The major scientific objectives envisioned for composite orbiter/lander missions to the six largest satellites of the Outer planets are:

1. To obtain data essential for studies of:
   (a) the mode of formation of the smaller planetary bodies in the solar system
   (b) the origin of planet/satellite systems
   (c) the origin of the solar system itself

2. To develop a possible link between studies of the Inner and Outer planetary groups by comparing the physical properties of the satellites with those of the smaller planets.

A further objective would be to use the satellites as bases for the remote observation of their parent planets. A satellite base has the inherent advantage of platform stability compared with an orbiting spacecraft. Also, if the satellite's rotation periods are locked to their orbital periods, as is predicted, then the parent planet is continuously observable from any landing site on the "front-face" of the satellite. Since the six satellites selected all apparently revolve well outside the intense regions of planetary radiation belts, radiation hazards should not be a major concern. The constant altitudes of the Regular satellites above their parent planets would simplify imagery requirements for planetary observations.

The flight time requirements for soft-landing 1000 lbs. useful payload, or rough-landing 10 50-lb. instrument packages, on each of the Outer planet satellites under consideration are summarized in Table 11. Implicit in the flight time range presented is the performance of the various propulsion systems considered for the capture and braking maneuvers at the satellite.

TABLE 11

MISSION SUMMARY

Interplanetary Transfer Mode Launch Vehicle Combination	Io	Europa	Ganymede	Callisto	Titan	Triton
BALLISTIC:						
Intermediate-20/Centaur	No	No	No	$600\text{-}700^d$	No	No
Saturn V	No	$600\text{-}700^d$	$500\text{-}700^d$	$500\text{-}600^d$	No	No
Saturn V/Centaur	$600\text{-}700^d$	$500\text{-}700^d$	$500\text{-}700^d$	$500\text{-}700^d$	$3\frac{1}{2}\text{-}4^y$	$11\text{-}12^y$
SOLAR-ELECTRIC:						
Intermediate-20/Centaur	No	$900\text{-}1100^d$	$900\text{-}1000^d$	$600\text{-}700^d$	$4\text{-}4\frac{1}{2}^y$	No
NUCLEAR-ELECTRIC:						
Titan 3F	$1100\text{-}1200^d$	$900\text{-}1000^d$	$900\text{-}1000^d$	$800\text{-}900^d$	$3\frac{1}{2}\text{-}4^y$	No
Titan 3F/Centaur	NA	NA	NA	NA	NA	$8\text{-}9^y$

These propulsion systems consisted of combinations of liquid propellants (earth-storable, space-storable, and cryogenic), and in some instances, solid propellant, to perform the various maneuvers. The applications of various propulsion combinations to each of the considered flight modes are summarized in the following paragraphs.

Ballistic Transfer Mode

Missions with ballistic trajectories to the six target satellites are conceptually possible using Saturn-class launch vehicles. A mission to Callisto is feasible with the Intermediate-20/Centaur if cryogenic propulsion systems are available to perform the capture and braking maneuvers. Use of the Saturn V launch vehicle provides mission capability to Europa, Ganymede and Callisto without regard to the type of propulsion system used at the satellite. Addition of the Centaur upper stage to the Saturn V includes missions to all four Galilean satellites of Jupiter and the more distant satellites, Titan and Triton, with flight time requirements ranging from about 2 years at Jupiter to 11 years at Neptune. Availability of the Saturn launch vehicle and the 3-impulse maneuver required because of direct approach conditions raise unanswered feasibility questions concerning the practicality of the ballistic mission mode.

Solar-Electric Transfer Mode

Solar-electric low-thrust flight mode missions are possible (excluding Io, Europa and Triton) using the Intermediate-20/Centaur launch vehicle. Flight times are comparable to the ballistic mode at Callisto, and somewhat longer at Europa, Ganymede and Titan. The reservations noted above still apply, and in addition, the development of a high-powered (125-150 Kwe at 1 AU) solar-electric stage is questionable.

Nuclear-electric Transfer Mode

Missions using nuclear-electric propulsion are possible to all six satellites with use of the Titan-class launch vehicles. Flight time requirements to the Galilean satellites are somewhat longer than for the ballistic and solar-electric modes, this being mainly due to use of a smaller launch vehicle and spiral departure and approach modes. Whereas the flight time requirement to Titan is comparable to the ballistic mode, the time to Triton is on the order of 2 to 4 years shorter than that for the ballistic mode. The primary feasibility questions center on the development of the nuclear-electric low-thrust stage.

Missions, each consisting of ten rough-landers instead of a single soft-lander, are possible with approximately the same flight time requirements as a soft-lander for the various interplanetary transfer modes and propulsion systems under consideration. Although many of the interplanetary transfer mode/launch vehicle/ propulsion system combinations considered have yet to be developed, it is concluded that the nuclear-electric low-thrust mode is most effective in regard to minimizing launch vehicle and flight time requirements.

Further study of the technical feasibility of satellite lander missions is desirable. Specifically, a mission study is recommended for composite orbiter/lander missions to Ganymede (Jupiter III) and to Titan (Saturn VI), the two largest satellites in the solar system. A mission to Titan may be of special scientific interest, since to date it is the only satellite positively identified to possess an atmosphere. In addition to defining more completely the science objectives, instruments, mission operations and subsystem requirements of these selected missions, the study should also identify the relative scientific importance of Ganymede and Titan, compared with the other 27 satellites of the Outer planets. Also, the study should take into consideration the effect of satellite atmospheres (especially for Titan) on the lander design and operations.

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