NUCLEAR DIVISION

Report No. RN-NC-0017

PERFORMANCE OF THREE SIZES OF SOLID CORE

NUCLEAR ENGINES FOR SPACE MISSIONS

MASTER

May 1968

NUCLEAR ROCKET OPERATIONS

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W. F. Heydorn

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PERFORMANCE OF THREE SIZES OF SOLID CORE NUCLEAR ENGINES FOR SPACE MISSIONS

D. F. Heyman
Advanced Concepts/Technical Staff
Nuclear Rocket Operations

J. J. Peterson, Manager
Advanced Concepts/Technical Staff
Nuclear Rocket Operations

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ABSTRACT

Three NERVA-type solid-core nuclear engines are compared on the basis of primary propulsion for a representative group of space missions. The mission opportunities occur in the 1975-86 time period and represent applications desirable for solid-core systems on the basis of the NASA space exploration goals, engine performance and development schedules, and growth in nuclear-stage operational experience.

Comparison begins with unmanned planetary flybys in the 1975-79 time period, through unmanned orbiters in the 1978-83 time period, and then manned orbiters to Mars, Venus and Mercury in 1982 through 1985. The final mission considered is a landing on Mars in 1986.

The study shows that large Voyager-size (20,000 lb and up) payloads can be placed in orbits around Mercury, Venus, and Mars. Payloads of similar size can be achieved for flybys of Jupiter and to the planets beyond Jupiter if the Jupiter swing-by is used. The effect of a reduction in thrust of 20% has a negligible effect on missions to Mars and the interior planets. The effect becomes appreciable to planetary bodies beyond the asteroids. Increasing the thrust 67% and the I_{sp} to 1000 sec results in a marked increase in performance for all missions considered.

The intent of the information presented is to provide basic mission capability data for use in summary documents and presentations.

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I. INTRODUCTION

Solid-core nuclear propulsion systems have been applied to the exploration of space in numerous studies under the cognizance of both industry and the Government (References 1, 2, and 3). In general, these studies are parametric in nature and consider an array of vehicle characteristics and mission opportunities. The information reported here is considered supplemental data to these studies.

Three NERVA-type solid-core nuclear engines are compared for missions ranging from unmanned planetary probes to a manned Mars landing in 1986. The missions conform to general recommendations of Agencies in the Government concerned with space exploration. (References 4 and 5). Additional considerations include NERVA development schedules and the estimated time required to accumulate nuclear flight stage operational experience. Comparison of the engines is made on the basis of identical missions and vehicle constraints.

II. MISSION ANALYSIS

A. MISSION SELECTION

Mission sequence and selection was based on Figure 1, which was constructed from an evaluation of References 4 and 5, from material provided by the Atomic Industrial Forum, and from conversations with the OMSF, NASA Headquarters. Additional screening of requirements included effects of nuclear engine development schedules (Reference 6), and the time necessary to acquire nuclear-stage operational experience (Reference 7).

Essentially, scientific exploration of the solar system, after initial unmanned probes to Venus and Mars, should begin in earnest about 1975. As transfer distances become greater, both velocities and payload requirements become larger. Trip times become increasingly important not only from on-board

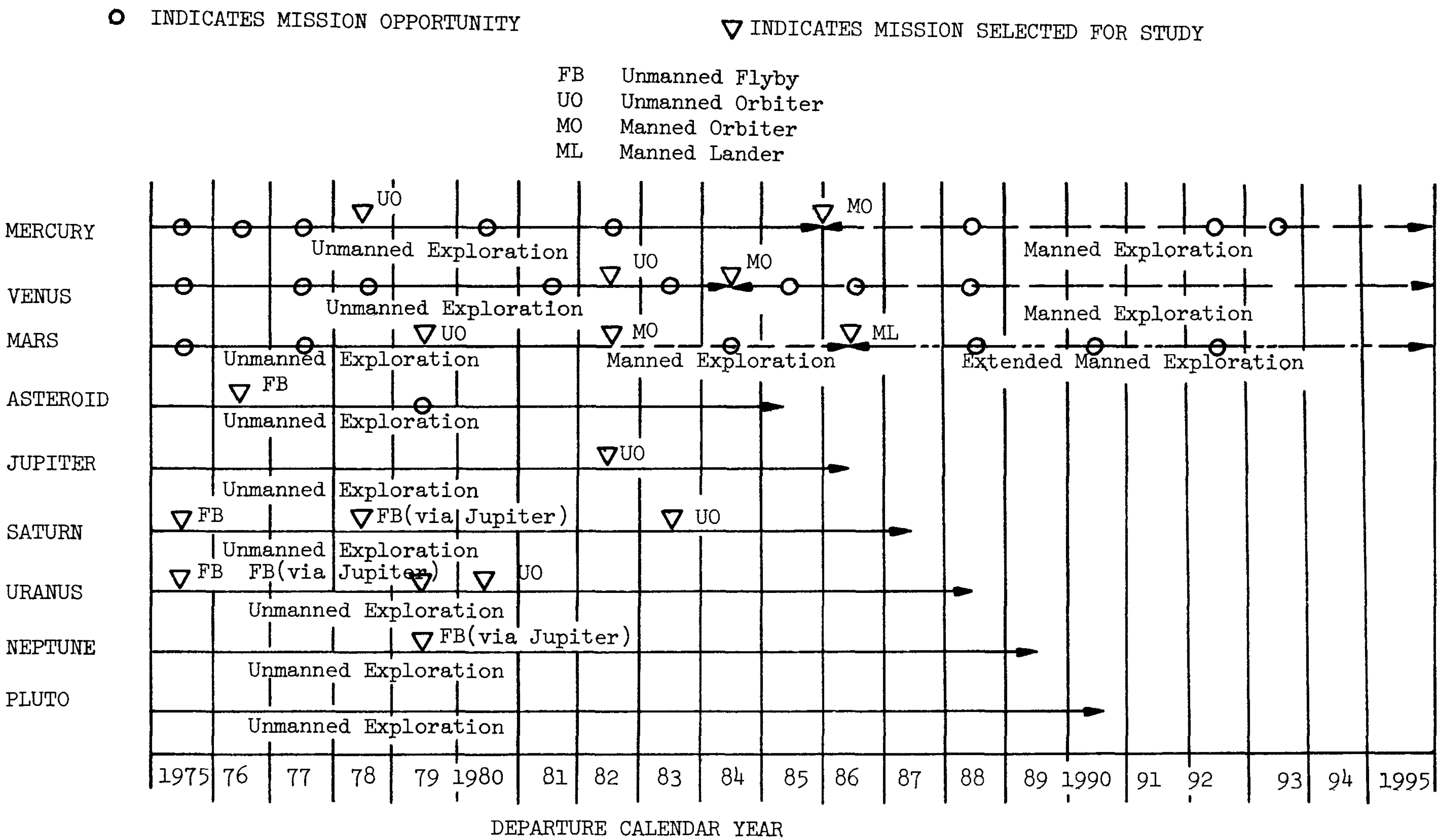

Figure 1 - Estimated Post-1975 Solar System Exploration Schedule

power requirement but also from increased difficulty in maintaining contact with the spacecraft. An experiment lasting upward of 10 years may also suffer from loss in motivation by the experimenters.

The missions studied begin with the use of unmanned planetary flybys to the far planets. Table 1 lists the departure year and trip times for these opportunities.

TABLE 1

SCIENTIFIC EXPLORATION OF THE SOLAR SYSTEM WITH UNMANNED PLANETARY FLYBYS

1975 - 1979

Flyby Mission	Departure Year	Approximate Trip Time	Cognizant NASA Agency
Saturn	1975	4 years	OSSA
Uranus	1975	10 years	OSSA
Neptune	1976	10 years	OSSA
Asteroid Ceres	1976	180 days	OSSA
Galactic Probe to 18.2 AU	1976	6 years	OSSA
Saturn via Jupiter	1978	3 years	OSSA
Uranus via Jupiter	1979	5 years	OSSA
Neptune via Jupiter	1979	8 years	OSSA

Continued unmanned solar system exploration is assumed to be the role of probes having planetary orbiting capability. The missions selected are listed in Table 2.

TABLE 2

CONTINUED EXPLORATION OF THE SOLAR SYSTEM WITH UNMANNED PLANETARY ORBITERS

1978 - 1983

Orbiter Mission	Departure Year	Approximate Trip Time	Cognizant NASA Agency
Mercury	1978	120 days	OSSA
Mars	1979	218 days	OSSA/OMSF
Uranus	1980	8.2 years	OSSA
Venus	1982	120 days	OSSA/OMSF
Jupiter III (Ganymede)	1982	1.9 years	OSSA
Saturn	1983	3 years	OSSA

Manned exploration of the solar system may be found desirable from data obtained during unmanned exploration. During the time period considered only the near planets (Mars, Venus, and perhaps Mercury) appear to be technologically feasible for visits by man. The missions selected are listed in Table 3.

TABLE 3

MANNED EXPLORATION OF THE SOLAR SYSTEM WITH PLANETARY ORBITERS AND LANDERS

1983 - 1986

Mission	Requirement	Departure Year	Approximated Round Trip Time, days	Cognizant NASA Agency
Mars	Orbiter	1982	450	OMSF
Venus	Orbiter	1984	410	OMSF
Mercury	Orbiter	1985	370	OMSF
Mars	Lander	1986	450	OMSF

B. TRAJECTORY DATA

Velocity requirements for the selected missions are shown in Figure 2 for unmanned probes, in Table 4 for unmanned orbiters, and in Table 5 for manned orbiters and landers. The data were selected from References 3, 8, 9, 10, 11, and 12.

In general, departure times are selected that result in fairly realistic trip time within the mission opportunities considered. All missions start from a circular earth orbit with the capability to meet the time limitations imposed by the departure window. Earth/vehicle communication conditions are assumed within the capabilities of the DSIF network.

For manned missions to Venus, Mars, and Mercury a constant midcourse correction of 100 m/sec is assumed for each transfer leg. The same value is assumed for unmanned orbiters. Payloads given for flybys should be further adjusted to reflect this and any other maneuver required for completion of the mission.

A direct reentry back to earth is assumed for manned missions. The reentry velocity is equal to or less than 20 km/sec.

TABLE 4

UNMANNED ORBITER TRAJECTORY DATA

Mission	Earth Holding Orbit, km	Earth Departure Velocity, $C_3(km^2/sec^2)$	Planet Arrival Velocity, $C_3(km^2/sec^2)$	Trip Time, days
1978 Mercury	185	58	42	120
1979 Mars	185	13	14	218
1980 Uranus	185	142	66	3000
1982 Venus	185	16	14	120
1982 Jupiter III	185	81	41	700
1983 Saturn	185	136	138	1100

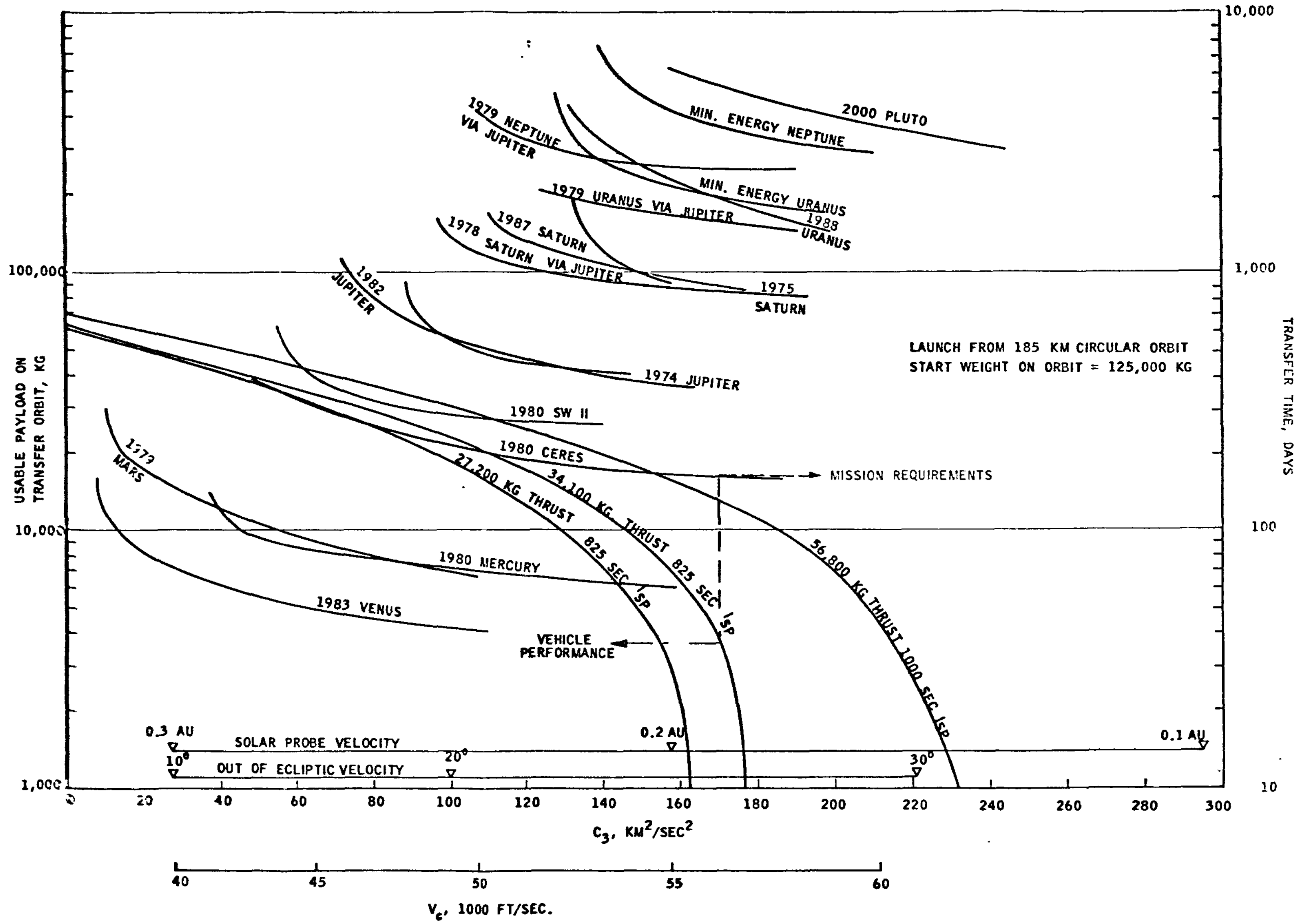

Figure 2 - Nuclear Vehicle Performance and Mission Data for Unmanned Planetary Probes

TABLE 5

MANNED ORBITER AND LANDER TRAJECTORY

	Mars Orbiter	Venus Orbiter	Mercury Orbiter	Mars Lander
Departure Year	Feb. 1982	Dec. 1984	Mar. 1985	Mar. 1986
Outbound Trip Time, days	200	140	110	160
Stopover Time, days	50	20	80	50
Inbound Trip Time, days	200	250	180	140
Maximum Time on Earth Orbit, days	30	30	45	90
Perihelian Passage, AU, Minimum	1.0	0.7	0.4	1.0
Earth Orbit Altitude, km	385	385	385	385
Earth Departure V_∞, C_3	15	16	55	15
Outbound Midcourse ΔV, m/sec	100	100	100	100
Planet Arrival V_∞, C_3	14.5	17	93	18
Planet Departure, V_∞, C_3	41	20	93	26
Inbound Midcourse ΔV, m/sec	100	100	100	100
Earth Arrival Velocity, C_3	336	196	314	241
Solar Activity	Intermediate	Intermediate	Quiet	Quiet

III. VEHICLE ANALYSIS

A. ENGINE PERFORMANCE

Engine performance characteristics are given in Table 6.

TABLE 6

NUCLEAR ENGINE PERFORMANCE CHARACTERISTICS

Thrust, kg	27,200	34,100	56,800
T_c, °R	4,500	4,500	6,000
P_c, kg/cm^2	31.6	31.6	34.6
Nozzle Area Ratio	70:1	70:1	100:1
Turbine Exhaust Ratio	8:1	8:1	10:1
Thermal Power, Mw	1,260	1,550	3,500
Engine I_{sp}, kg sec/kg	825	825	1,000
Weight, kg	7,260	7,260	11,200
Flow Rate, kg/sec	32.1	41.2	56.7

The 27,200-kg thrust engine (60,000 lb thrust) is compatible with minimum changes to ETS-1. The 34,100-kg thrust (75,000 lb) and 56,800 kg thrust (125,000 lb) represent the nominal and maximum engine performance based on the same reactor and within foreseeable technology improvements.

Engine weights include engine component and tank-bottom shielding only. Mission-oriented payload or crew biological shields are added when the stage is sized either by scaling laws or payload considerations (Section III-C).

B. LAUNCH VEHICLE CONSIDERATIONS

A product-improved Saturn V is used as the earth-launch vehicle. The vehicle is assumed to be capable of placing 125,000 kg (275,000 lb) into a 185 km circular orbit using two stages. Nuclear space stages and payloads are launched to orbit and assembled as required. Single launches are used for the flyby missions and single and multiple launches are used for the orbiters and landers. Automatic rendezvous and docking is assumed for unmanned orbiters requiring multiple earth launch. Suborbital start of the nuclear stage is not considered.

C. NUCLEAR STAGE CONSIDERATIONS

1. Modular Considerations

Planetary flybys are accomplished by a vehicle having a total gross weight of 125,000 kg on 185 km circular earth orbit. The vehicle consists of an earth-depart stage, instrument unit, interstage structure, and payload. No tank insulation or stage meteoroid armor is assumed to be required. Shielding for payload protection beyond that provided by engine shielding must be subtracted from payload usable weight.

Multiple nuclear stages are assumed for orbiters and landers. Identical engines are used in earth-depart, planetary-arrival, and planetary-departure stages as required. The earth-departure stage may contain multiple tank-engine modules. For unmanned orbiters modules are sized to the capability of the launch vehicle. The number of earth launches is a primary constraint on this type of mission. For manned orbiters and landers the launch vehicle capability, although considered in the sizing calculations, is assumed to be a constraint. The large size of these vehicles provides considerable flexibility in the packaging of the earth-launch-vehicle payloads.

2. Scaling Laws

a. Tank Structure and Accessories

Stage tankage and associated subsystems, less engine and shield are sized using the following equation from reference 3.

$$W_t = \frac{AW_p^{0.9}}{\sigma^{0.533}} + 500 \quad kg$$

A is assumed 0.10 nominal

W_p = total propellant load, kg

σ = propellant average specific gravity

b. Meteoroid Protection Scaling Laws

Meteoroid armor is determined by the solution given in Reference 1.

$$W_m = 1.492 \times 10^{-2} T^{1/2} A_m^{4/3} + 2{,}040 \ lb$$

T = exposure time, days

A_m = exposed area, sq ft

assuming a hydrogen tank with an L/D of approximately 3 and hemispherical ends;

$$W_m = 9.27 \times 10^{-3} T^{1/3} W_p^{8/9} + 4500 \ kg$$

where T is in days and W_p is in kilograms

c. Boiloff

Boiloff is calculated from equations developed in Reference 3. Propellant loss and insulation weight are considered separately.

$$W_{bo} = \frac{KA_m \, t \, \Delta T}{\gamma_L} \quad kg$$

K = thermal conductivity of the insulation. Assumed equal to 4.40×10^{-5}k cal/hr mK

A_m = exposed area, sq meters

t = exposure time, hours

ΔT = temperature differential across tank wall, °K

γ = insulation effective thickness, meters

L = heat of vaporization, k cal/kg

the expression for optimum insulation thickness is given by;

$$\gamma_{opt} = \frac{K\ t\ \Delta T^{1/2}}{L_{\rho}} \quad \text{meters}$$

ρ assumed to equal 80 kg/m^3

combining the above two equations and solving for a hydrogen tank in terms of propellant weight

$$W_{bo} = 3.5 \times 10^{-4}\ W_p^{\ 2/3}\ (t\ \Delta T)^{1/2} \quad \text{kg}$$

d. Thermal Insulation

Thermal insulation weight for propellant storage is calculated from the equation given in Reference 3.

$$W_i = \rho\ \gamma\ A_m$$

solving in terms of hydrogen weight;

$$W_i = 4.34 \times 10^{-3}\ W_p^{\ 2/3}\ (t\ \Delta T)^{1/2} \quad \text{kg}$$

3. <u>Manned Payload Weights</u>

Manned payloads to Mars, Venus, and Mercury are defined as follows:

Crew size: 8 man for both orbiters and landers

Life support expendables: equation is taken from Reference 3:

$$W_{1s} = 13.02\ (T-200) + 4370 \quad \text{kg}$$

$$T = \text{trip time, days}$$

Mission Module: 4,600 kg without solar flare shield engine biological shield, or Command Module.

Planet Lander: 35,500 kg includes descent and ascent stages and an Ascent Module of 1000 kg.

Earth Entry and Command Module: 9,000 kg just prior to entry. Assumes an entry velocity $\leqq$ 20 km/sec.

Solar Flare Shield: this weight is based on data given in Reference 1, (Page II-19). An average perihelion distance is assumed.

Engine Biological Shield: an added engine shield of 4535 kg is assumed for crew protection, limiting dosage to approximately 3 rem/year (Reference 3).

IV. VEHICLE PERFORMANCE

A. PLANETARY FLYBY MISSIONS

Results of analysis of the three nuclear propulsion systems for the assumed planetary flyby missions are given in Table 7 and Figure 3. Figure 2 is included for general information and as an aid in determining

TABLE 7

USABLE PAYLOADS FOR UNMANNED PLANETARY PROBES

Single-Launch, Improved Saturn V with Orbit Start of Nuclear Third Stage

(Kilograms)

Mission	27,200 kg thrust 825 sec, I_{sp}	34,100 kg thrust 825 sec, I_{sp}	56,800 kg thrust 1000 sec, I_{sp}
1975 Saturn, 4 years	8,200	12,700	20,800
1975 Uranus, 10 years	9,500	14,000	22,000
1976 Neptune, 10 years	0	4,000	13,100
1976 Asteroid Ceres, 180 days	10,600	15,000	23,000
1976 18.2 AU Galactic Probe, 6 years	3,400	7,400	16,000
1978 Saturn via Jupiter, 3 years	14,000	18,100	26,000
1979 Uranus via Jupiter, 5 years	7,000	11,300	19,800
1979 Neptune via Jupiter, 8 years	9,700	14,200	22,200

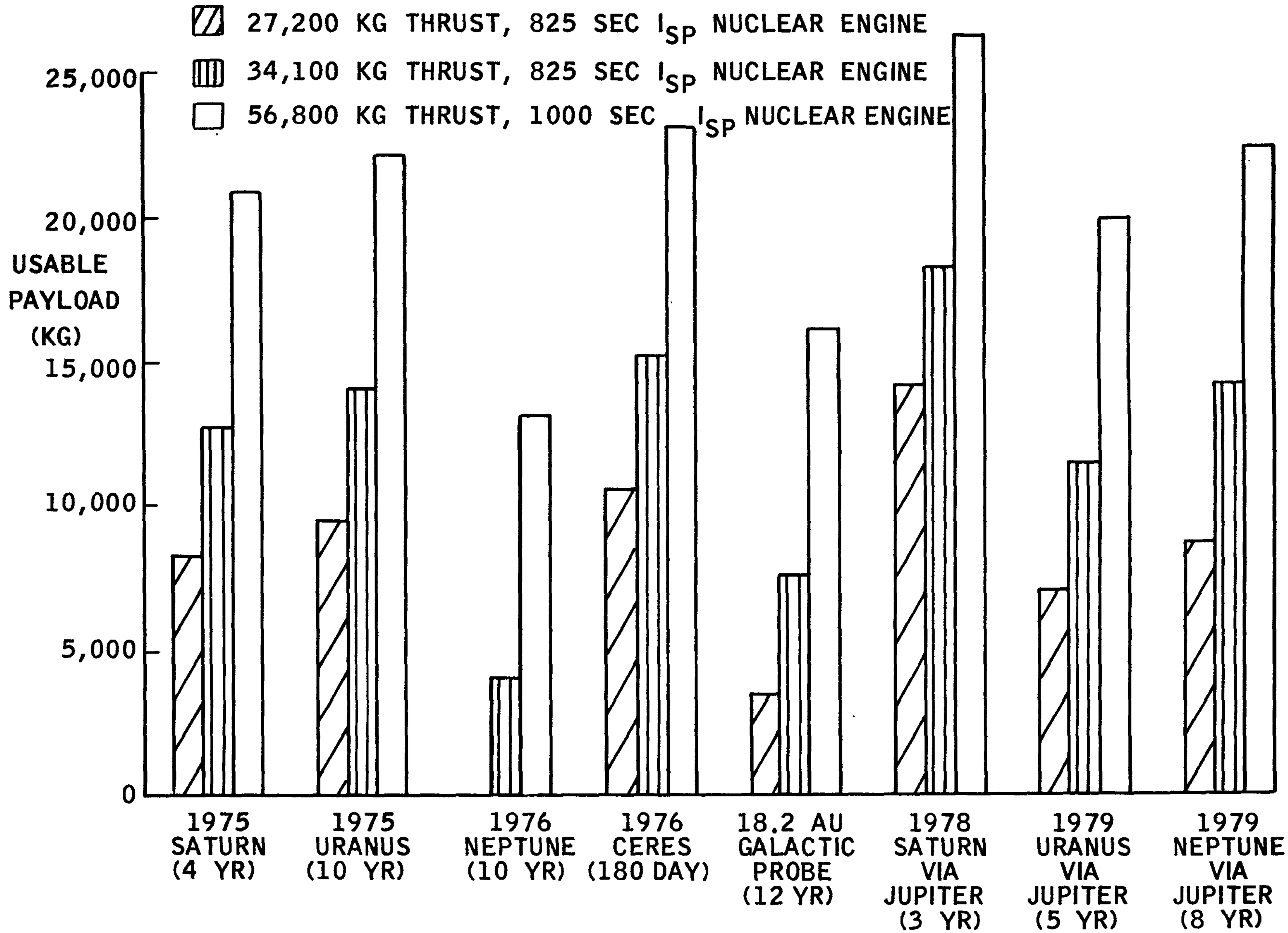

Figure 3 - Payloads for Unmanned Planetary Probes

maximum payload capability associated with the engines over a broad range of velocities and transfer times.

Also of importance in flyby missions is the earth-to-planet transfer time. Assuming a constant Voyager-size payload of 9070 kg (20,000 lb), Table 8 and Figure 4 show the relationship between the transfer times.

B. UNMANNED PLANETARY ORBITERS

The performance objective of an unmanned orbiter is to obtain maximum usable weight on the target-planet orbit for a given number of earth launches. Table 9 presents the results of this portion of the analysis, and Figure 5 shows the comparison in graph form.

C. MANNED ORBITERS AND LANDERS

For the relatively large vehicle weights required for manned planetary missions the start weight on earth orbit is generally used as a basis for comparing vehicle variables. Start weight is particularly sensitive to the weight of the vehicle used to return from space to the earth's surface. The variation in start weight for the five manned missions considered over a range of earth re-entry vehicle weights is given in Figures 6 through 9. Earth-entry velocity is equal to or less than 20 km/sec in all cases.

Table 10 and Figure 10 show start weight requirements if a constant weight of 9,000 kg is assumed for the earth-entry vehicle.

V. CONCLUSIONS

Performance of the three propulsion systems is summarized in Table 11. The table considers the 34,100 kg (25,000-lb thrust) 825 sec I_{sp} engine as the baseline for comparison.

TABLE 8

MISSION TRANSFER TIMES FOR UNMANNED PLANETARY PROBES

Single-Launch, Improved Saturn V with Orbit Start of Nuclear Third Stage

9070 kg Usable Payload at Transfer, (Days)

	27,200 kg Thrust 825 sec, I_{sp}	34,100 kg Thrust 825 sec, I_{sp}	56,800 kg Thrust 1000 sec, I_{sp}
1975 Saturn	1,900	1,000	780
1975 Uranus	3,350	2,360	1,730
1976 Neptune	10,000	4,470	3,160
1978 Saturn via Jupiter	940	860	800
1979 Uranus via Jupiter	1,930	1,730	1,450
1979 Neptune via Jupiter	2,870	2,620	2,500

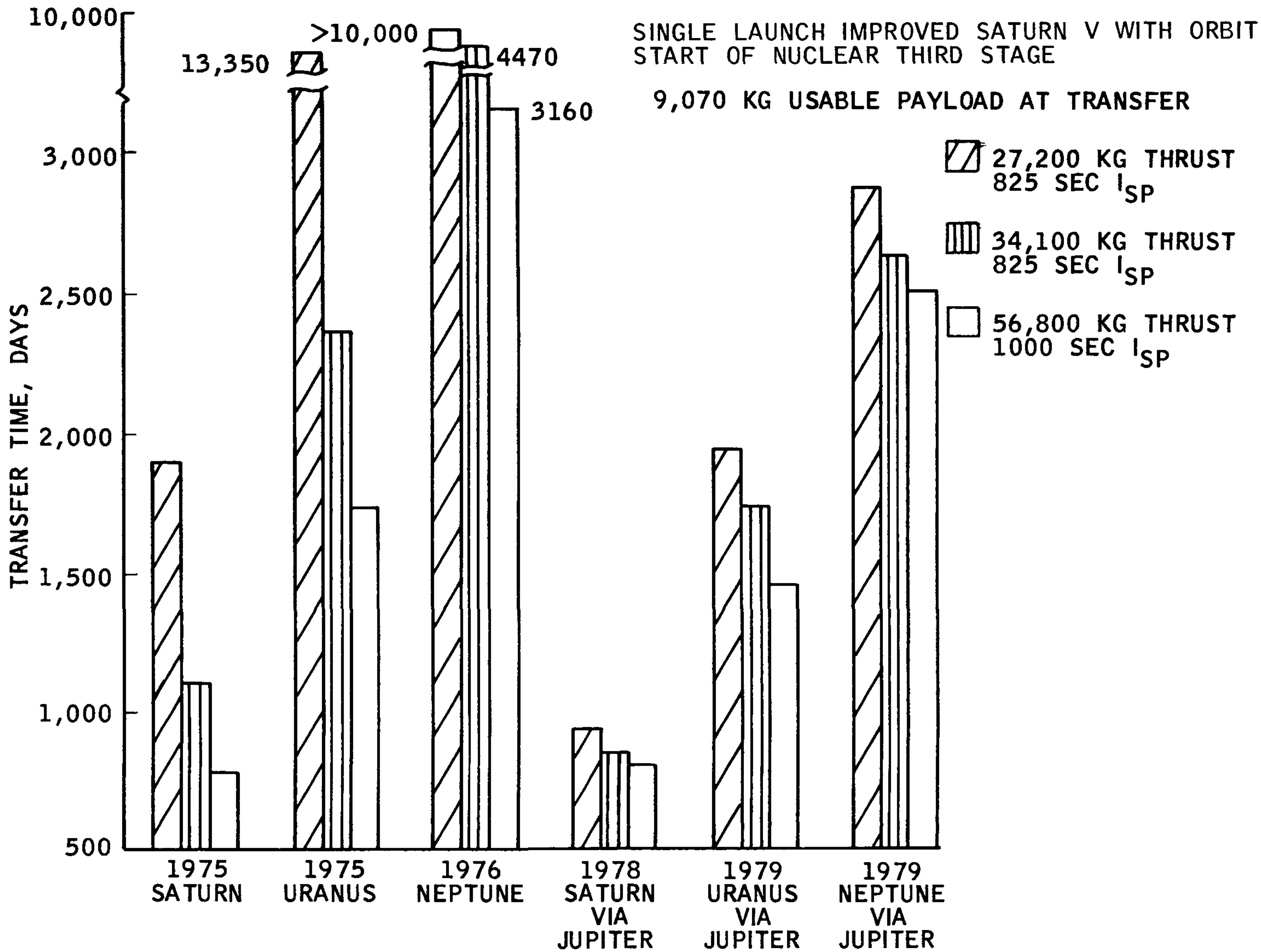

Figure 4 - Transfer Times for Unmanned Planetary Probes

TABLE 9

USABLE WEIGHT ON PLANETARY ORBIT FOR UNMANNED ORBITERS

(Kilograms)

	27,200 kg Thrust 825 sec I_{sp}			34,100 kg Thrust 825 sec I_{sp}			56,800 kg Thrust 1000 sec I_{sp}		
Mission / ELV's	1	2	3	1	2	3	1	2	3
1978 Mercury	4,630	23,060	---	5,170	24,340	---	7,840	33,920	---
1979 Mars	16,960	47,350	---	17,310	48,770	---	20,140	58,260	---
1980 Uranus	0	0	5,830	0	1,340	7,800	0	6,930	18,810
1980 Venus	20,520	53,620	---	21,310	55,050	---	24,040	64,960	---
1982 Jupiter III	0	12,760	23,360	0	18,510	30,030	0	23,120	41,450
1983 Saturn	0	0	0	0	0	0	0	0	7,060

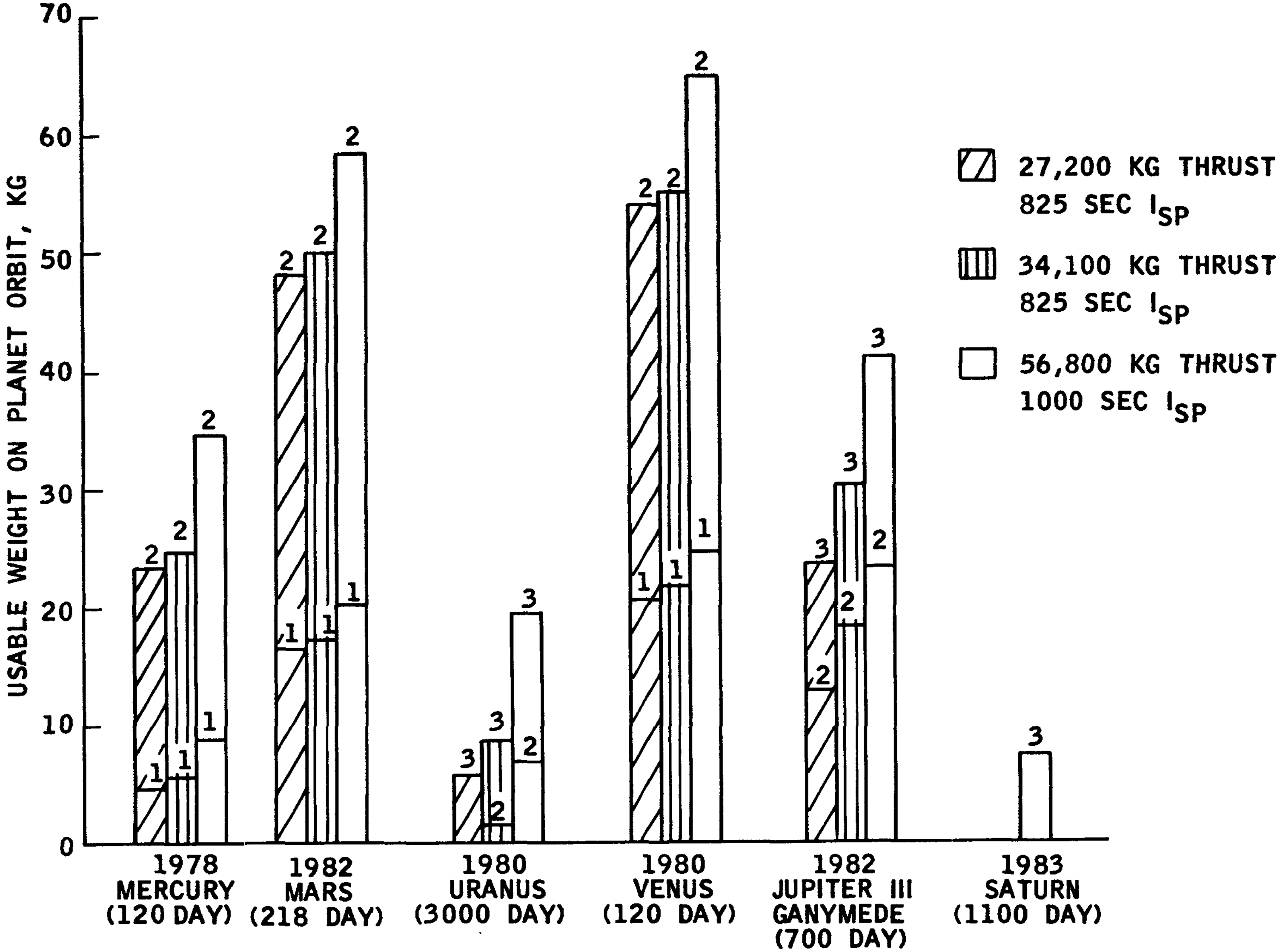

Figure 5 - Planet Orbit Payloads for Unmanned Orbiters

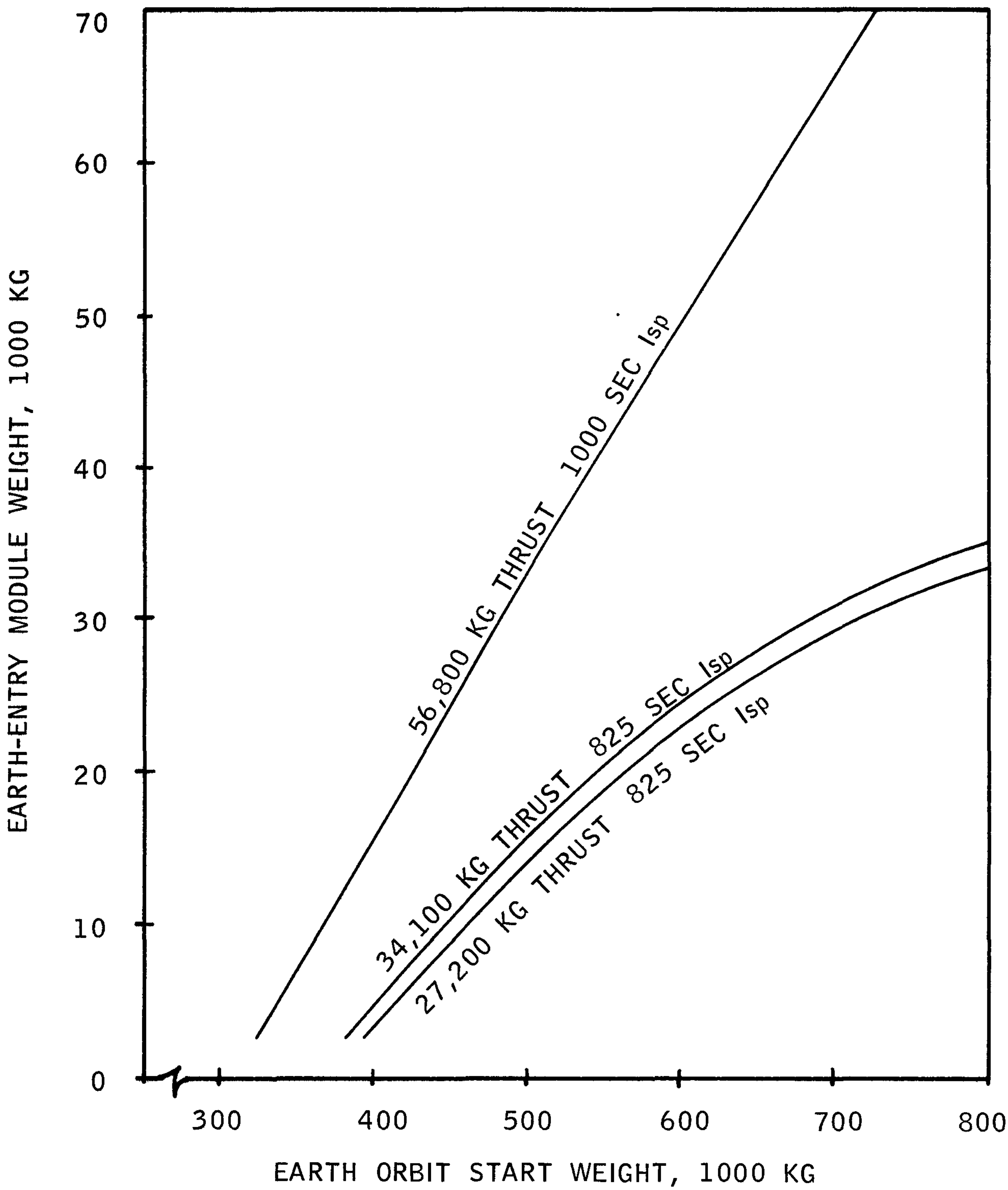

Figure 6 - 1982 Manned Mars Orbiter-Start Weight Requirements

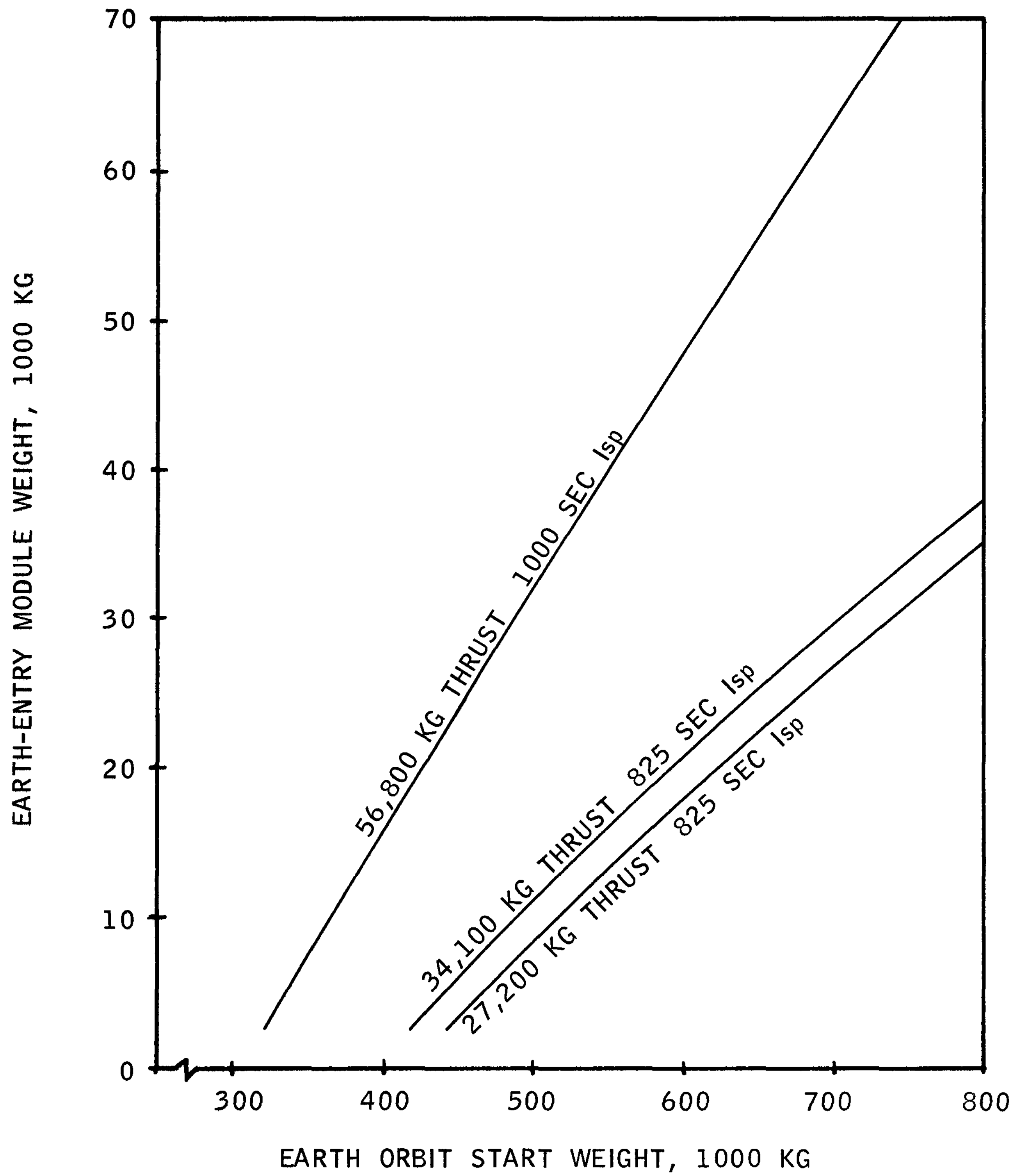

Figure 7 - 1984 Manned Venus Orbiter-Start Weight Requirements

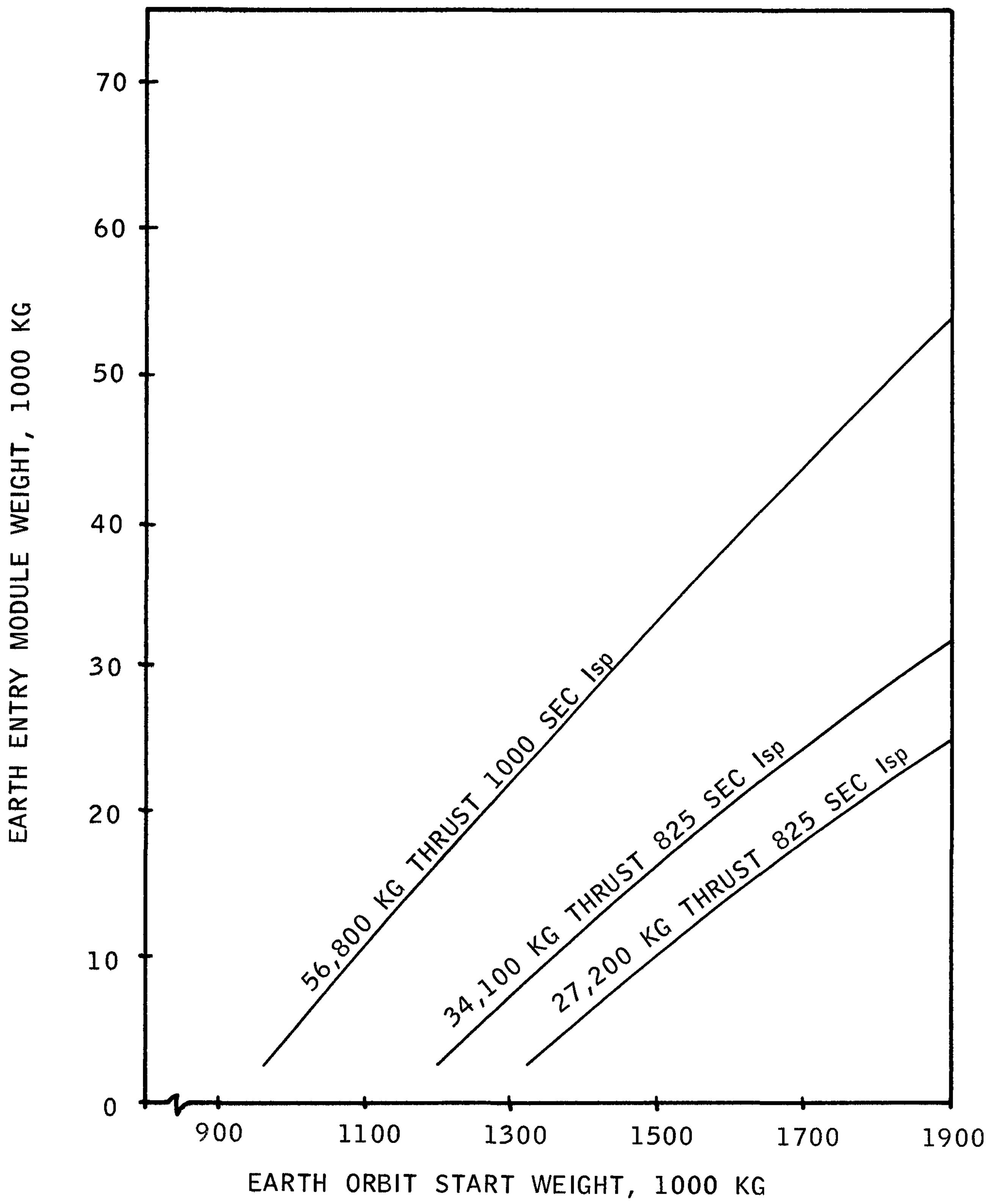

Figure 8 - 1985 Manned Mercury Orbiter-Start Weight Requirements

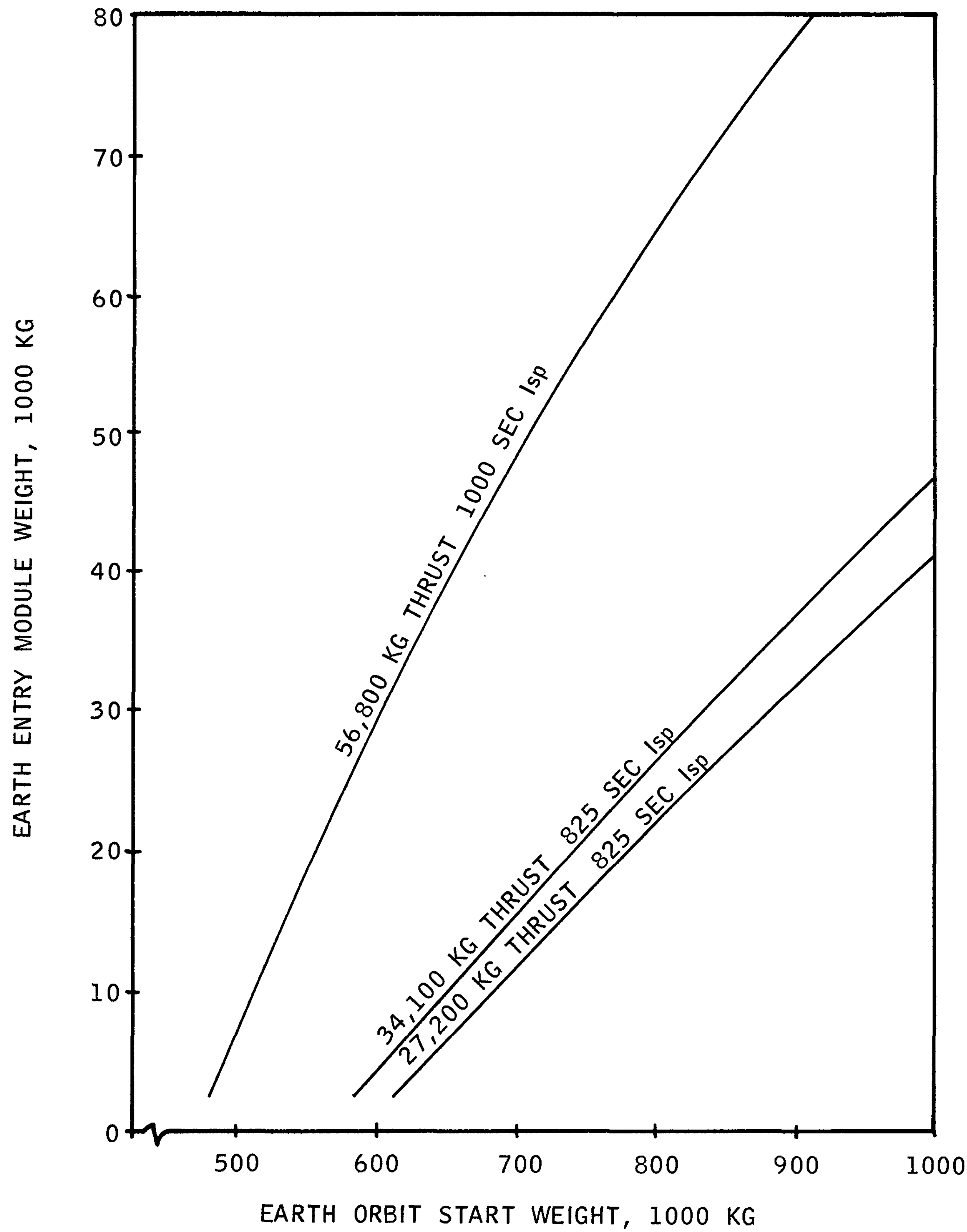

Figure 9 - 1986 Manned Mars Lander-Start Weight Requirements

TABLE 10

EARTH-START WEIGHT REQUIREMENTS FOR MANNED PLANETARY MISSIONS

9000 kg Earth-Reentry Module

(Kilograms)

Missions	27,200 kg Thrust 825 sec I_{sp}	34,100 kg Thrust 825 sec I_{sp}	56,800 kg Thrust 1000 sec I_{sp}
1982 Mars Orbiter	453,000	438,000	361,000
1984 Venus Orbiter	509,000	477,000	359,000
1985 Mercury Orbiter	1,475,000	1,342,000	1,075,000
1986 Mars Lander	673,000	646,000	510,000

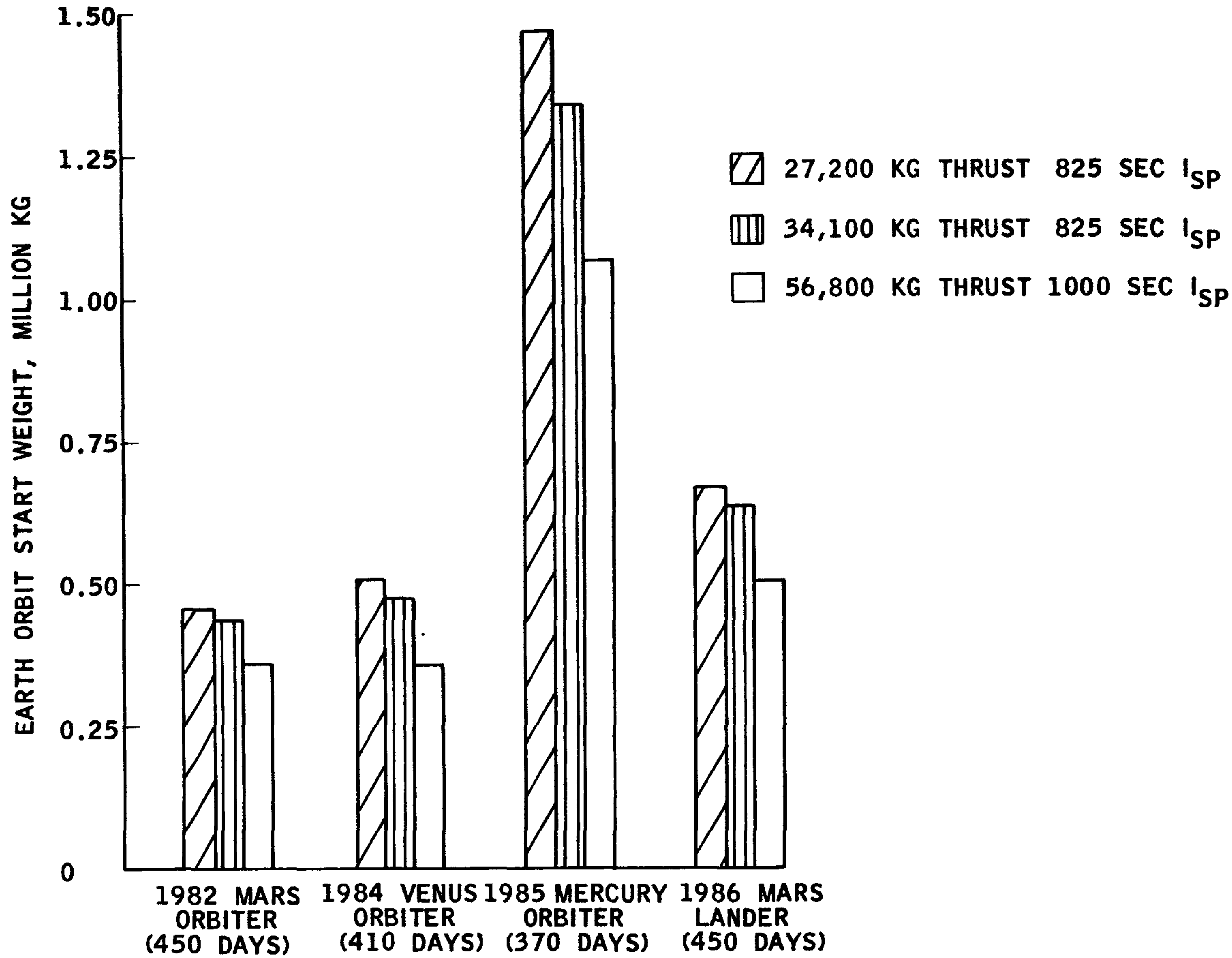

Figure 10 - Earth-Start Weight Requirements for Manned Planetary Missions

TABLE 11

ENGINE COMPARISON SUMMARY

(The 34,100 kg thrust, 825 sec I_{sp} engine payload capability is used at the 100% baseline for comparison)

Mission	27,200 kg thrust 825 sec I_{sp}	56,800 kg thrust 1000 sec I_{sp}
Unmanned Flybys (% Transfer Weight from Earth Orbit)		
1975 Saturn, 4 years	65	164
1975 Uranus, 10 years	68	157
1976 Neptune, 10 years	0	328
1976 Ceres, 190 days	71	154
1976 18.2 AU Probe	46	217
1978 Saturn via Jupiter, 3 years	77	144
1979 Uranus via Jupiter, 5 years	62	175
1979 Neptune via Jupiter, 8 years	68	156
Unmanned Orbiters (% Weight on Planetary Orbit)		
1978 Mercury (2 launch)	95	140
1979 Mars (1 launch)	98	116
1980 Venus (1 launch)	96	113
1982 Jupiter III (2 launch)	69	125
1980 Uranus (3 launch)	75	241
Manned Orbiters & Lander (% Weight on Earth Orbit)		
1982 Mars Orbiter	104	82
1984 Venus Orbiter	107	75
1985 Mercury Orbiter	110	80
1986 Mars Lander	104	79

For unmanned flybys the payload transferred from earth orbit is reduced from about 13% to no payload at all for the lower thrust engine and increased from 44 to over 300% for the higher performance engine. As might be expected, the gain or loss is largely a function of the velocity requirement and I_{sp} developed.

Weight on planetary orbit for unmanned orbiters is increased from 16 to 241% for the 56,800 kg thrust engine and reduced from 2 to 31% for the 27,200 kg thrust engine. As vehicles become larger the effect of the differences in engine thrust become smaller.

Weight on earth orbit required for manned missions increases 4 to 10% when the lesser thrust engine is used. The weight decreases 8 to 25% when the higher performance engine is employed. Although these variations are considerably smaller than those shown for unmanned missions, the effects of the earth launch vehicle are significant. A decrease of 10% in earth-start weight for a 1986 Mars lander, for example, results in a 65,000-kg reduction (approximately 143,000 lb).

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