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SUMMARY AND CRITICAL EVALUATION OF THE
MISSION CAPABILITIES OF ELECTRIC PROPULSION

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ABSTRACT

The literature on electric propulsion abounds with the results of mission studies that have been performed in various laboratories, often with such a degree of independence that comparisons between them are extremely difficult. This paper presents a method of correlating many of these results by plotting payload against a mission difficulty parameter defined as the velocity increment from an earth orbit that would be required for a minimum-energy impulsive transfer.

The location of some of the boundaries are discussed, and propulsion system selection maps are presented for low time and cost missions and for low cost slow missions. The published results of several authors are extrapolated and plotted on the maps, showing fair agreement.

Care has been taken to be objective with respect to the favorable aspect of chemical and nuclear-thermal propulsion. The superiority of electric propulsion became obvious for all manned missions beyond the moon and for a large class of unmanned missions to the planets.

A very significant conclusion is reached from the analysis that even with relatively heavy power supplies the electric rocket has a much higher potentiality than is generally believed, and the current paper will serve as strong evidence in the case for electric propulsion.

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1. INTRODUCTION

In selecting a propulsion system for use in a space mission, there are many parameters to be considered. However, if we assume that the necessary reliability and lifetime requirements, and the necessary availability are met, we are then primarily concerned with the cost of delivering a certain payload in a given time.

This paper presents a comparison of chemical, nuclear heat-exchanger, and electric propulsion systems on the basis of payload to be delivered on one-way missions and on the out-going part of round-trip missions. The comparisons are made on the basis of initial weight in orbit (which reflects launch vehicle weight and therefore cost), for identical mission times where time is an essential parameter, and on initial weight in orbit alone for missions where short flight time is not required.

2. PREVIOUS COMPARISONS

Several detailed analyses of the capabilities of electric propulsion have been performed at JPL (see Refs. 4, 9-12). These are largely concerned with selecting a launch vehicle and nuclear-electric power system, and determining the gross payload and terminal-to-initial mass ratio for various missions as a function of time, thrust, specific impulse. The analyses have generally avoided comparison or contrasts with chemical and nuclear rockets, but in Ref. 3 JPL reported some selected comparative data, which will be utilized in the summaries that follow.

Edelbaum, in Ref. 13, reported again on the basis of payload delivered by a fixed launch vehicle, and included some selected comparisons. Other authors, including Moeckel (Ref. 2) and Stuhlinger (Ref. 1) have presented results of calculations of initial mass in orbit needed to deliver a fixed payload, but only a few such results have been published. The authors are unaware of any general attempt to determine the regimes (the values of payload) for which various propulsion systems are most clearly preferable, or the boundaries on which they are competitive.

3. PERTINENT PARAMETERS

It is convenient to characterize the requirements of a mission in terms of the payload to be delivered, the mission time, and the cost expressed as initial mass in a parking orbit. In selecting values for comparison, we shall need to consider the conditions under which each parameter is expressed.

(a) Payload Mass

Nuclear reactor power-to-weight ratio and power supply specific mass both decrease as power increases. Consequently for a given mission, there can be levels of payload above which it is appropriate to change from a chemical propulsion system to one using nuclear or nuclear-electric propulsion. It is therefore particularly appropriate to examine the missions in terms of actual payload, rather than using the payload ratio. This has additional value in that the consideration of real payload gives a greater insight into the magnitude of the systems involved, such as boosters and power supply.

We shall ordinarily be concerned with the net useful payload, where numbers exist for this. For a scientific mission, the scientific package plus the communication system and sufficient power for telemetry and facsimile transmission will be included, but additional power such as might be available from a large nuclear-electric powerplant used for electric propulsion will not be included. In some calculations, such a powerplant would be substituted by a lesser powerplant sufficient for communications. For a manned mission the payload will include crew quarters and life-support systems, communication and navigation equipment, landing and logistic supplies, and that part of the power supply and shielding necessary for continued life and operation at the destination.

One-way payloads only are considered because of the complexity of correlating return payload with initial mass in earth orbit.

(b) Mission Time

The mission time considered for low time and cost missions will be that for a minimum-energy impulsive transfer. In general there will be some advantage to be gained for impulsive systems by choosing shorter times than this, and advantages for continuous thrust by choosing longer times, but these will be offset by increase in cost.

For some missions, the mission time need not be kept low. These include logistic missions such as the lunar ferry, and mapping missions in which the time may be deliverately extended.

(c) Mission Cost

The cost of a mission will depend largely on the cost of the launch vehicle, providing that the development costs of the nuclear rocket reactor and the nuclear powerplant are amortized over a sufficient number of missions. For comparison purposes we may use the mass in an initial orbit, even though many of the values have been calculated for 300 n. mi rather than the 600-700 n. mi minimum altitude that may be mandatory for reactor start-up.

4. METHOD OF PRESENTATION

It is the purpose of the present paper to provide a few simple charts on which the selection of a propulsion system for a given mission may be readily made. In order to define propulsion regimes we shall use two plots of payload versus mission, using the following criteria.

(a) Low Time and Cost Missions

A given point (payload and mission) belongs in propulsion region A when that propulsion system requires less initial mass in orbit than propulsion systems B or C to carry the payload on the mission in the minimum energy transfer time.

(b) Low Cost Missions

A given point belongs in propulsion region A when that propulsion system requires less initial mass in orbit than propulsion systems B or C to carry the payload on the mission, even if it takes longer.

5. MISSION DIFFICULTY PARAMETERS

The ordering of missions in terms of difficulty poses a problem. There is no parameter available which will characterize the difficulty of a mission for impulsive and continuous thrusting systems unambiguously. Bussard (Ref. 15) used distance from the earth on a logarithmic scale to show near-earth, minor-planet and major-planet missions, but such a scheme is valid only for probes.

The use of minimum velocity increment, corresponding to a minimum energy impulsive transfer from an initial earth orbit (usually 300 n mi) has been used extensively by many authors (Refs. 5, 14, 16, 17) and it serves to arrange the missions in a semblance of order. The main anomalies are to be found in (a) the small gravitational pull of Mercury making it difficult to orbit around Mercury and (b) the atmospheres of Mars, Venus and the larger planets, making it easy to land on them, saving on Δ V by using atmospheric braking.

The values of velocity increment using continuous thrust are functions of thrust, acceleration and specific impulse, there being generally no minimum for a given mission except for a lower bound of $(V_1 - V_2)$ for a low-acceleration transfer between circular orbits having orbital velocities V_1 and V_2. The value $(V_1 - V_2)$ also approximates to the impulsive transfer case where the change in orbital radius is small, but for interplanetary missions the continuous thrust will require a larger characteristic velocity (defined by $V_c = g_o I_{sp} \ln \frac{M \text{ initial}}{M \text{ final}}$) over the impulsive case, by factors of 2 or 3.

For the purposes of plotting the missions in order to define propulsion regimes, we shall use the minimum energy impulsive velocity increment. This will provide a mission difficulty parameter, and it is not important that the Δ V shown does not represent the Δ V necessary for electric (continuous) propulsion. Values of Δ V for many interplanetary missions are given in Refs. 5 and 17, and have been used to prepare Fig. 1.

Just as for the payload, the one-way Δ V only is plotted, so that ambiguities can be avoided for all except, possibly, the missions to Mercury.

6. DATA ESTABLISHING PROPULSION REGIMES

(a) Minimum Scientific Payload

If we use the approach of Stearns as reported in Ref. 5 , we can plot the minimum initial mass in low earth orbit necessary to carry a minimum scientific spacecraft on the probe and capture missions of Fig. 1. Stearns' graph is reproduced in Fig. 2 with structure factors of 10-20 percent used to give the spread for the chemical and nuclear systems, and 10-30 percent used for the electric systems. Specific impulses of 430, 1000, and 6000 sec are used as shown. The initial mass scale is logarithmic to show greater detail and a minimum spacecraft mass of 1000 lb for each 10,000 ft/sec is the relationship used. The linearity of this relationship is not essential to the general validity of the plot, since the curves for electric and nuclear propulsion are fairly flat.

The importance of Fig. 2 to a general plot is that it shows nuclear propulsion to be non-competitive for the smaller payloads. This is primarily due to the 10,000 lb initial weight of the shielded nuclear reactor heat-exchanger system, which must be lifted through the same ΔV as the payload. It will take a Saturn C-1B to launch the nuclear rocket spacecraft even for a Mars or Venus probe, yet with this same launch weight and a SNAP-50 nuclear electric rocket a 4000 lb payload can be placed in a capture orbit about any of the major planets. Figure 2 does not show time relationships, but it is evident that the chemical rocket is competitive with the nuclear rockets for missions up to 20,000 ft/sec, above which the electric rocket is competitive in time as well as in initial mass required.

(b). Minimum ΔV for Electric Propulsion

For low time and cost missions electric propulsion will not be competitive for low values of ΔV. In fact the lowest ΔV for which electric rockets should be used will always be greater than about 13,000 ft/sec (that for the moon), and this determines an asymptote for large payloads.

(c) Maximum Δ V for Impulsive Propulsion

For small payloads (< 100 lb) on low time and cost missions there will be a preference for chemical propulsion, but the Δ V achievable does not increase without limit. Staging can be carried up to 10 or 12 stages, but the vehicle design and reliability problems become formidable. There will therefore be some Δ V, perhaps less than 40,000 ft/sec, beyond which electric propulsion will always be used, even for negligibly small payloads.

(d) Solar Electric Propulsion

The role played by solar power in electric propulsion is a restricted one. For small payloads and large velocity increments, the missions for which electric propulsion should be used will carry the vehicle far from the earth into regions where the solar flux is greatly reduced (or increased, for solar probes). Solar power systems cannot accommodate the large changes involved, and reactors or isotope sources will be used.

However, for near-earth missions where time is either unimportant or a long mission time is desired (as in mapping) the solar-electric system is competitive with nuclear-electric systems for power levels below about 50 kw, and is competitive with chemical rockets for all but the smallest values of Δ V.

(e) The Choice Between Chemical and Nuclear Propulsion

We can define the boundaries where chemical and nuclear propulsion have equivalent capabilities on a payload/velocity increment plot by taking suitable assumptions. Following Stearns (as reported in Ref. 5) we can write, for the nuclear rocket,

$$\frac{M_L + \zeta M_p + M_{pp}}{M_L + M_p + \zeta M_p + M_{pp}} = \exp\left(- \frac{\Delta V}{I_{sp}\, g_o}\right)$$

where

ΔV = the velocity increment to be gained

I_{sp} = specific impulse

g_o = gravitation acceleration at earth's surface

M_L = payload mass

M_p = booster propellant weight

ζ = inert mass factor, applied in particular to tankage, controls, structure, etc., excluding the nuclear powerplant

M_{pp} = powerplant mass.

By putting $M_{pp} = 0$ we get the usual rocket equation, and Fig. 3 shows the results of choosing some selected values for I_{sp}, M_{pp}, and ζ, and setting the payloads delivered by the chemical and nuclear rockets to be equal (curves A, D, and E) or to differ by 10 percent (curves B and C). The values taken are given in the following table.

TABLE I

Curve	A	B	C	D	E
(I_{sp}) Nuclear, sec	1000	1000	1000	850	1000
Powerplant weight, lb	10,000	10,000	10,000	10,000	7500
Payload Advantage	0	10 % for chem.	10 % for Nucl.	0	0

(I_{sp}) chemical = 430 sec

(ζ) chemical = 10 percent

(ζ) nuclear = 15 percent

It will be seen that the spread is quite small. Curves A, B, C show that the deviation from a boundary to gain a clear 10 percent advantage gives at most a factor of two on the payload. Curves D and E show how little effect a change in I_{sp} and M_{pp} will have. Curves have also been calculated for other values of ζ, up to 25 percent for both, and these lie very close to Curve A.

(f) Regimes for Slow Missions

As an example of a non-optimal analysis of a mission consider the case where the propellant supply mass, M_p, is negligible compared to the power supply mass, M_{pp}. Hence, the mass ratio, R, is unity.

$$R = \exp\left(\frac{\Delta V}{I_{sp}g}\right) \sim 1 + \frac{\Delta V}{I_{sp}g} \tag{1}$$

$$R = \frac{M_{pp} + M_L + M_p}{M_p + M_L} \tag{2}$$

$$\frac{M_p}{M_{pp} + M_L} = \frac{\Delta V}{I_{sp}g} \tag{3}$$

let $M_p = f\, M_{pp}$ where $f \ll 1$

$$\frac{M_L}{M_{pp}} = \frac{f\, I_{sp}g}{\Delta V} - 1 \tag{4}$$

When

$\frac{f\, I_{sp}g}{\Delta V} = 1$ the payload is zero.

If f is chosen to be 0.1, an upper boundary for the plot of M_L versus ΔV is defined, given a nominal value of I_{sp}. For I_{sp} = 6000 sec and f = 0.1 a plot of M_L versus ΔV is shown in Fig. 5, where curves A, B, and C represent different power supplies. From Equation (4) it can be seen that the value of M_L varies in direct proportion to f. To get the most payload for a negligible value of propellant weight, f should be chosen in a narrow region about its maximum value, say 0.1. The region about the curves in Fig. 5, then is the region for operation of such a propulsion system.

Equation (4) is used to define the regions of operation for SNAP VIII and SNAP 50 in the mapping shown in Fig. 6. Note that the payload is not a strong function of power but is mainly dependent upon the power supply mass M_L. The power level primarily determines the mission time.

The curve or region which is minimal for electrical propulsion is set by the solar electric system.

(g) Specific Mission Points

Extrapolations have been made for results taken from several references, and the values plotted in the maps of Figs. 4 and 6.

For the low time and cost missions (Fig. 4) all the points plotted represent payload levels at which electric propulsion is reported to be superior to nuclear or chemical propulsion. In Ref. 1, Stuhlinger considers ion propulsion for a manned Mars mission, and we can take the useful payload delivered one way to be the sum of the landing vehicle, the landing payload, and about one-third of the propulsion system propellant and shielding, so this point is plotted at about 380,000 lb. From Ref. 2, the effective payload delivered one way has been extracted from Moeckel's equations to be 94,000 lb, at which the electric rocket is superior to the nuclear rocket.

Ref. 3 gives the gross payloads for several missions and to use these for Fig. 4 several artifices have been resorted to. Only those missions for which electric rockets (with power supply specific weights of 13 lb/kw) complete the one-way trip in the same time, or faster than the impulsive systems have been used. The payloads delivered by the Nova and the Saturn-Rover combinations have been scaled down to give equal initial weights in orbit (which gives a conservative comparison) yielding payloads L_N and L_C. The value of payload for which electric propulsion would be the choice has been taken to be the larger of L_N, L_C, providing this is less than L_E, the payload delivered by the electric rocket. For if

$$L_E > L_N > L_C$$

for the same initial weight, we can infer that the electric rocket would be the choice to deliver a payload L_N or larger, but we cannot say what choice would be made to deliver L_C.

In Ref. 4, Beale and Speiser give the terminal mass for a Mars orbiter. From their Fig. 1 we can estimate the terminal mass for a 230 day transfer to be 10,500 with a power supply specific weight of 15 lb/kw. To improve the estimate we can take 25 lb/kw for a shielded SNAP 50 and still deliver about 2000 lb plus SNAP 50, compared with about 3000 lb for a chemical rocket. The placing of this point was difficult because the 3000 lb delivered by the chemical system will include about 1000 lb of communications, power, and other inert weight. The point must therefore be plotted somewhat above 2000 lb, because for the same launch weight the chemical system would be cheaper than the electric system (by at least the price of the SNAP 50) unless the SNAP 50 is required and forms part of the useful payload. The maximum useful payload for this example is about 9,000 lb, and the point has been plotted at that value.

The payload for the Jupiter orbiter mission of Ref. 5 was taken as 4000 lb, to include the scientific payload, communications system, and a share of the power supply system, and the Mars mission of Ref. 6 has been plotted directly.

For the low cost slow missions, (Fig. 6) data points are given for the lunar ferry propulsion capabilities reported by Stuhlinger (Ref. 5) and Currie (Ref. 8), and a point is plotted for the station-keeping and attitude control system for the stationary satellite, for which electric propulsion is competitive for periods over about 3 years (Ref. 7).

7. RESULTS - PROPULSION SELECTION MAPPING

(a) Low Time and Cost Missions

The compilation of data from references on minimum energy transfer studies results in the propulsion regime mapping shown in Fig. 4. The boundary between chemical and nuclear propulsion systems is fairly well defined by the results of the analysis described in Section 6e. The boundary between nuclear and electric-nuclear systems is not as easy to define analytically. It is set as asymptotic to the ΔV value of 13,000 ft/sec, the lunar transfer. It approaches the asymptote from the high ΔV side because at higher power levels the electric-nuclear systems become more economical (lower specific weight).

The boundary between chemical and electric propulsion systems for low payload weights is set near 40,000 ft per second since the chemical systems require exceedingly long flight times for missions to the outer planets and the payload ratios become relatively small.

The data points discussed in Section 6g are superimposed on the mapping in Fig. 4. There is also a band which is estimated to be the region of minimum payloads which would be sent on missions as a function of ΔV (Ref. 5).

All the boundaries which have been described are necessarily diffuse because of the multiplicity and variability of the needs for specific missions. However, the agreement between the placement of regimes and actual data points is evidence that the mapping criteria are realistic.

(b) Low Cost Slow Missions

Figure 5 shows a set of curves calculated from Equation 4. These curves show the effect on payloads delivered using several power levels, two types of engines, and two propellant fractions. These cases are summarized in Table II. This data is used to define the regions applicable to four propulsion classifications in Fig. 6, (electric-nuclear greater than 1 mw power level, electric-nuclear for the 30 kw to 1 mw power levels, electric-solar and chemical). The boundary between chemical and solar electric is arbitrarily set by a single data point which corresponds to the requirements for the attitude control and station-keeping systems for a synchronous satellite (Ref. 7). The boundary between electric solar and electric nuclear is set by the weight of the SNAP 8 power system. The boundary between the two electric-nuclear power regimes is set by the estimated SNAP 50 weight and a 20 percent propellant fraction as indicated in Table II, below.

TABLE II

Curve	A	B	C	D	E
Power Supply	SNAP 50	SNAP 8	1 Mw	SNAP 50	SNAP 50
Weight, lb.	9,000	6,000	15,000	9,000	9,000
Specific Impulse, sec	6,000	6,000	6,000	6,000	1,000
f	0.1	0.1	0.1	0.2	0.2

The boundaries themselves are shown to be diffuse since precise values of parameters cannot be specified. For example, a value of 6000 seconds was chosen for the specific impulse although it may range from 3000 to over 10,000 seconds for an electrostatic propulsion device. For the class of missions represented in Fig. 6 it was assumed that the specific impulse would be set at the value which minimizes the power to thrust ratio, thereby maximizing thrust. Thus, although flight time is of secondary concern in these missions, such a choice of I_{sp} will minimize it. A few sample data points are shown in Fig. 6 which were taken from specified references. There are three types of application represented; lunar ferry, earth space mapping probes, and an orbit control system.

Data pertinent to the type of mission covered by Fig. 6 is quite sparse since interest in the relatively short duration flights has been predominant until recent time.

Nuclear propulsion is considered to have little application since the advantage of the higher specific impulse is offset by the payload advantage in using an electric solar propulsion system of much higher specific impulse.

(c) General Results

The two mappings presented in Figs. 4 and 6 show that electric propulsion is superior for all missions in the limit of high Δ V beyond a Mars flight and large payloads except for the restriction of Δ V > 13,000 ft/sec (a moon capture) for the faster missions. Chemical propulsion is suited to the low Δ V values with decreasing value in favor of other forms of propulsion at high payloads. Nuclear propulsion is useful in the intermediate regions of Δ V for payloads above 5000 pounds.

8. SUMMARY AND CONCLUSIONS

The mapping of missions in terms of payload and Δ V (Figs. 4 and 6) is intended to be an aid to rounding out regimes which belong to specific propulsion systems. It also shows the boundary regions where the choosing between systems would be difficult or where the choice may be made on the basis of other criteria such as convenience or availability of components. The advantage of such a representation is that it directly suggests the system size and weight, and power level. The form of this mapping is tied to the weight of known or anticipated components and is not subject to the generalities of ratios.

The risk in such a representation, of course, is in oversimplification. The variation of the parameter I_{sp} has been ignored in favor of specifying it for a particular power to thrust ratio. A somewhat arbitrary value of the propellant fraction has been chosen. However, it is expected that further studies will show that a wide range of application can be included in this class. The choice of minimum energy missions is arbitrary so that a study of tradeoffs between mission time and non-minimum energy flights is warranted. Other classes of missions need to be considered, such as those in which payload per unit time is to be minimized as may be required where rapid supply buildup is needed. Cost of supply per unit time is another parameter which may be important where a long sustained supply rate is required with a minimum of expenditure.

This work cannot be regarded as complete or conclusive. The data used confirms the existence of some regions in which the choice of propulsion system is clear, and transition regions which are not yet precisely defined. There is a need to incorporate more data points and more critical use of comparative information and classes of mission. However, this is believed to be an important beginning in providing a useful aid for the planning of space missions.

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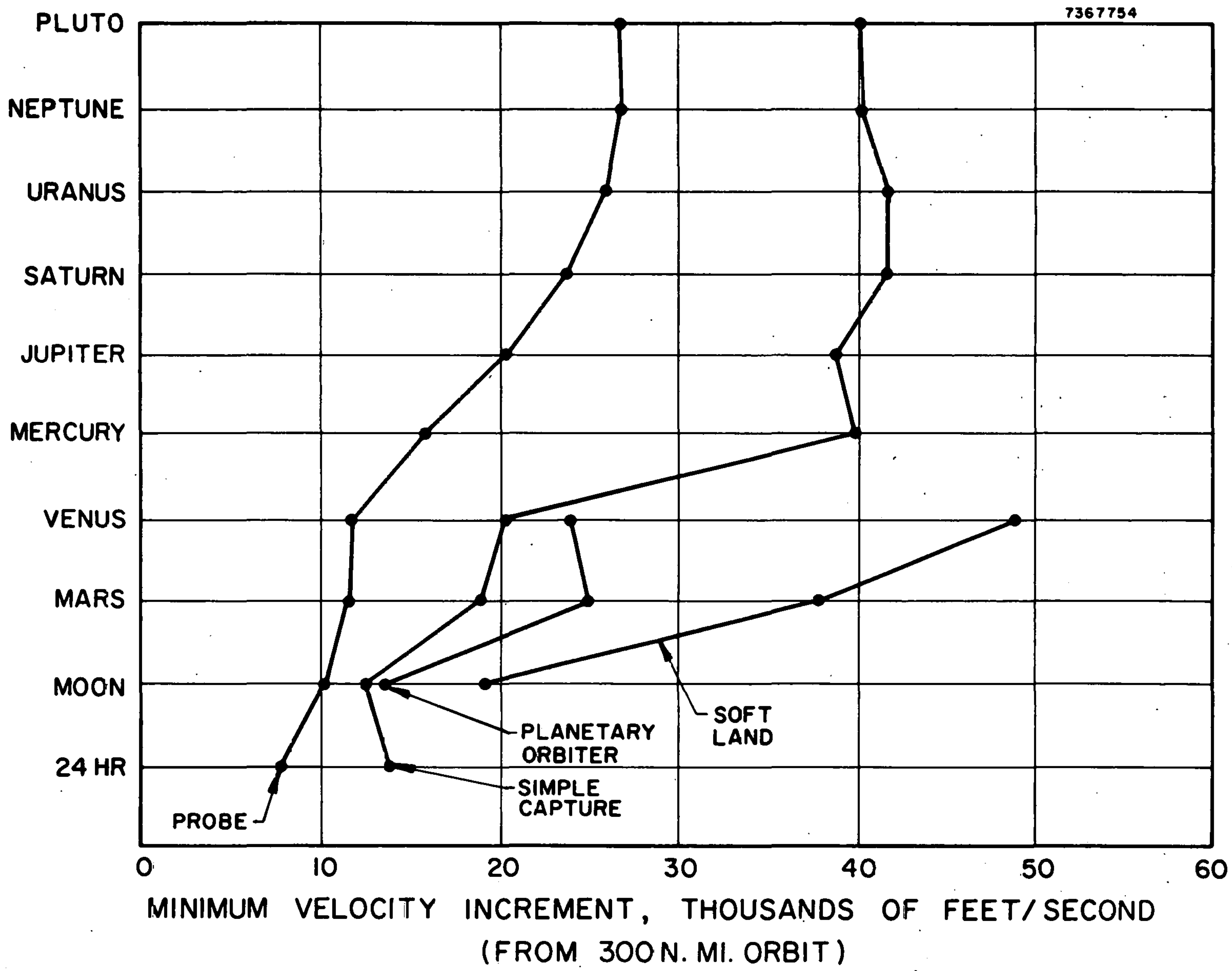

FIG. 1 MISSION DIFFICULTY PARAMETER FOR PLANETARY MISSIONS

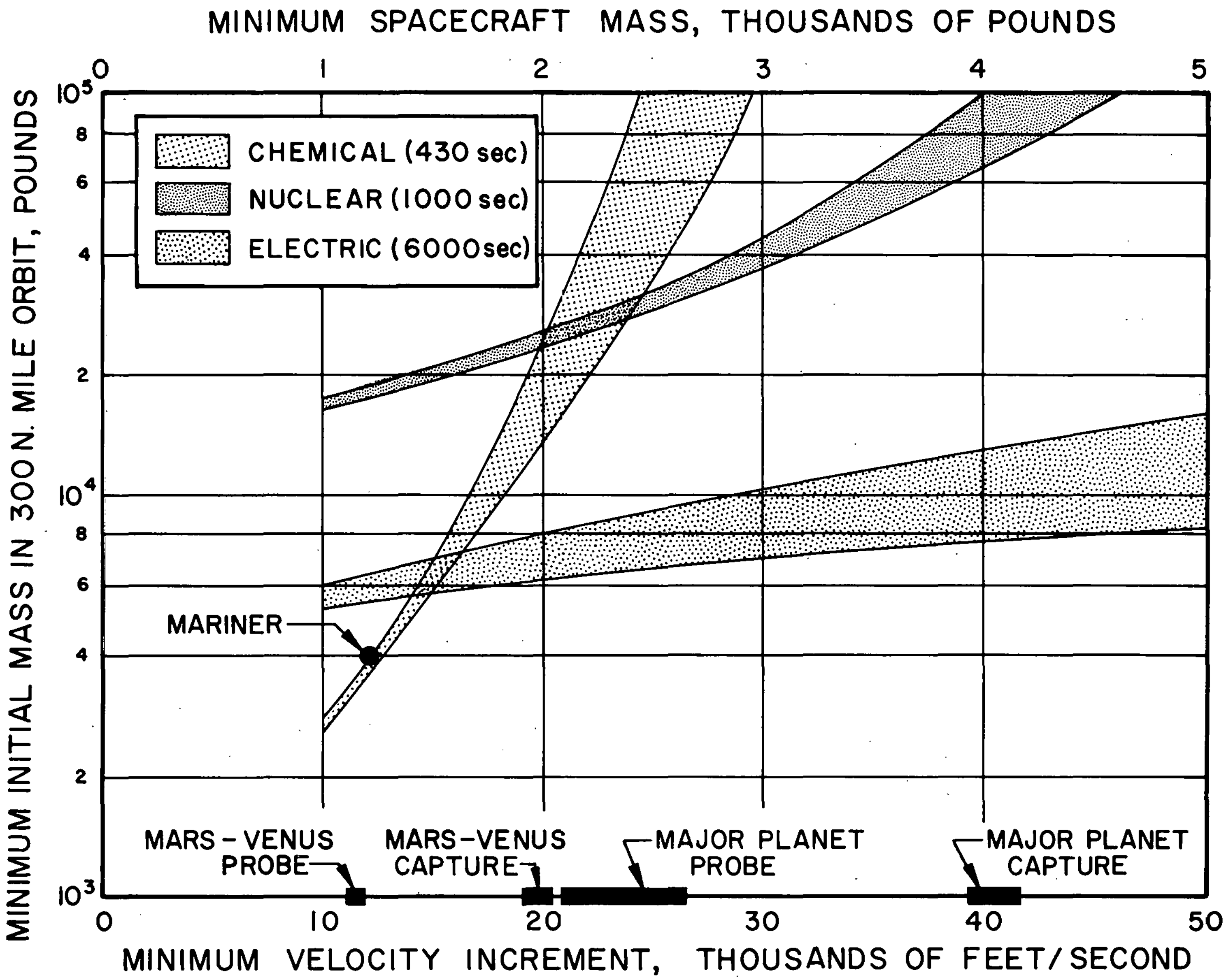

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FIG. 2 PROPULSION FOR MINIMUM SCIENTIFIC MISSIONS

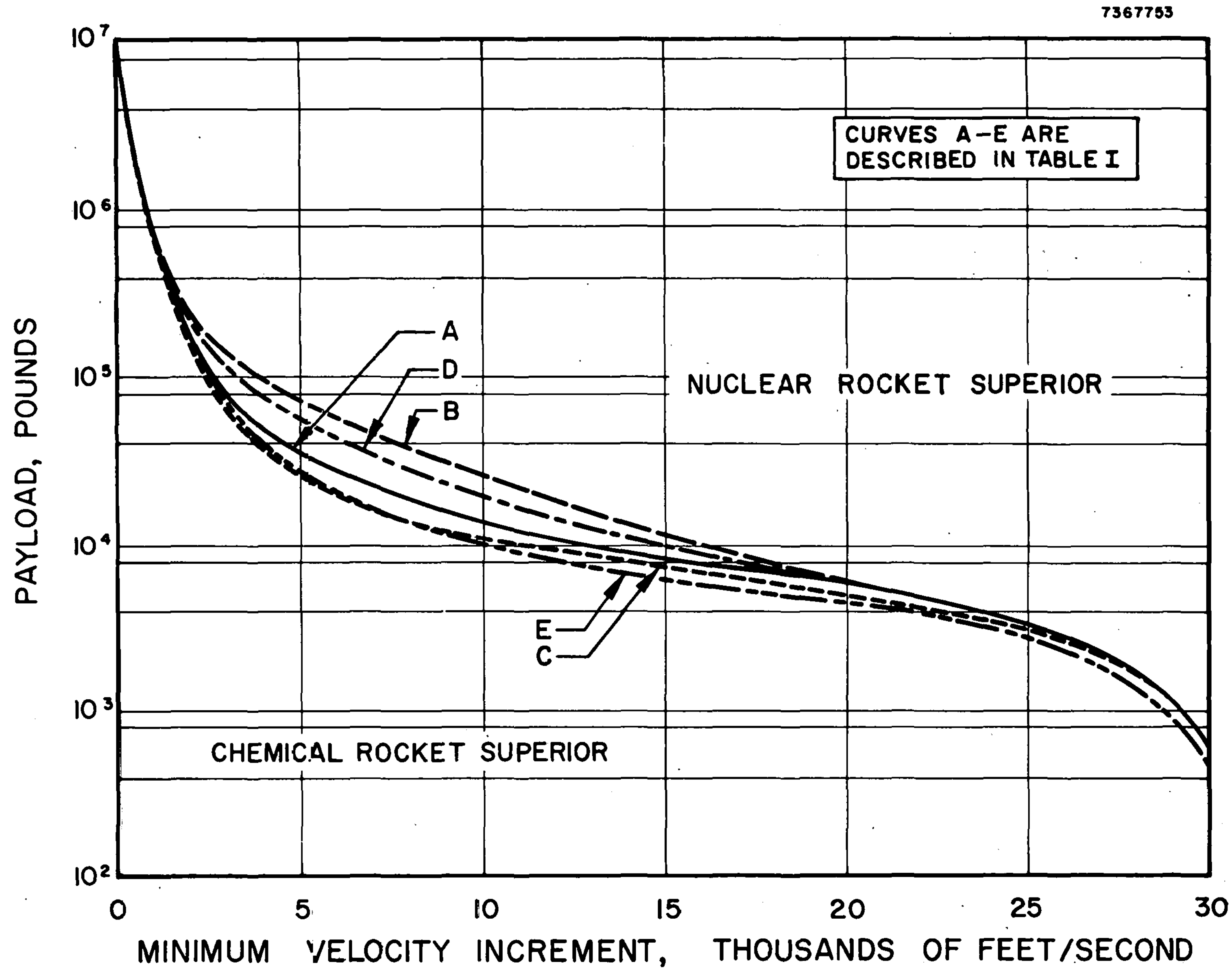

FIG. 3 CHOICE BETWEEN NUCLEAR AND CHEMICAL PROPULSION

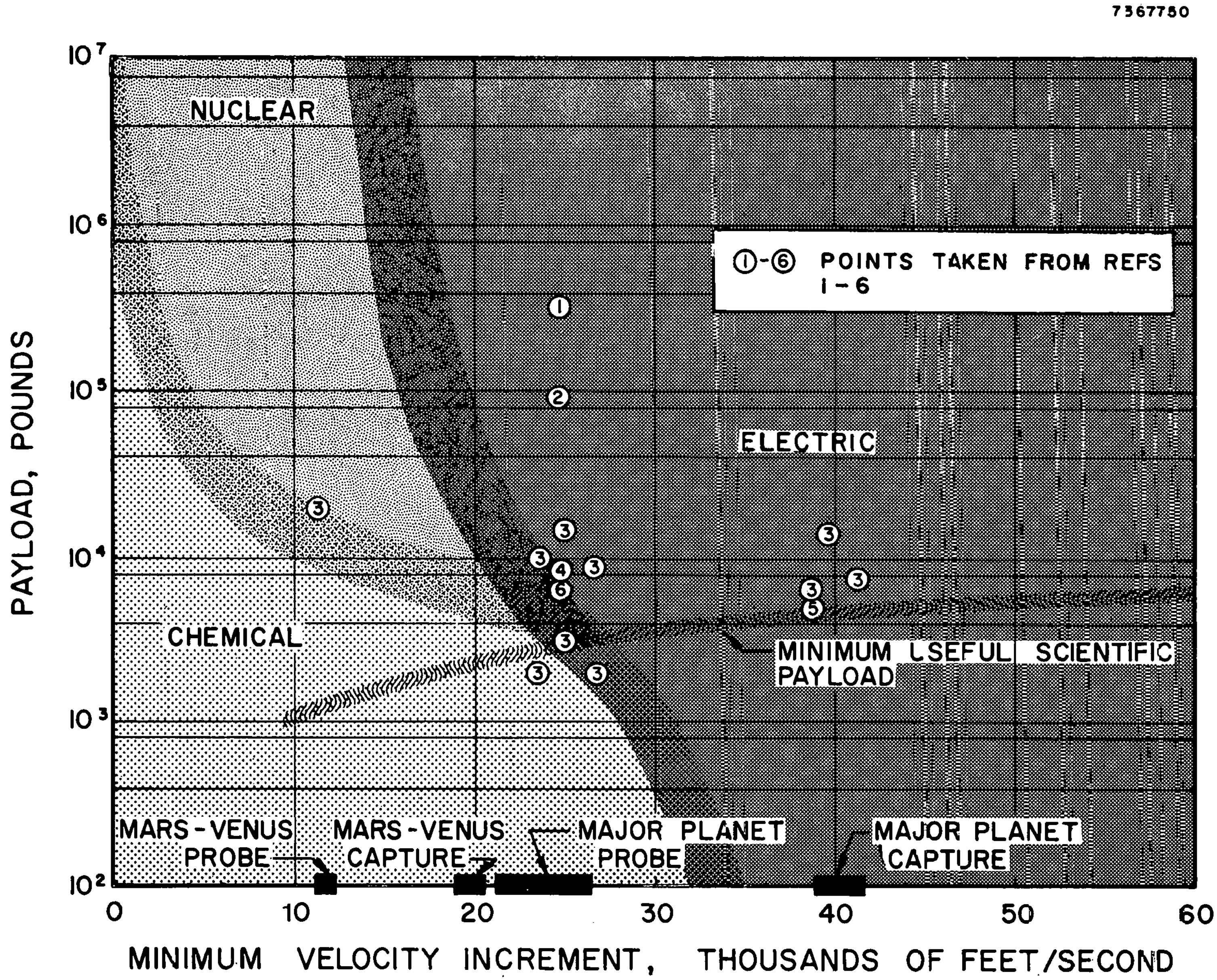

FIG. 4 PROPULSION SELECTION FOR LOW TIME AND COST MISSIONS

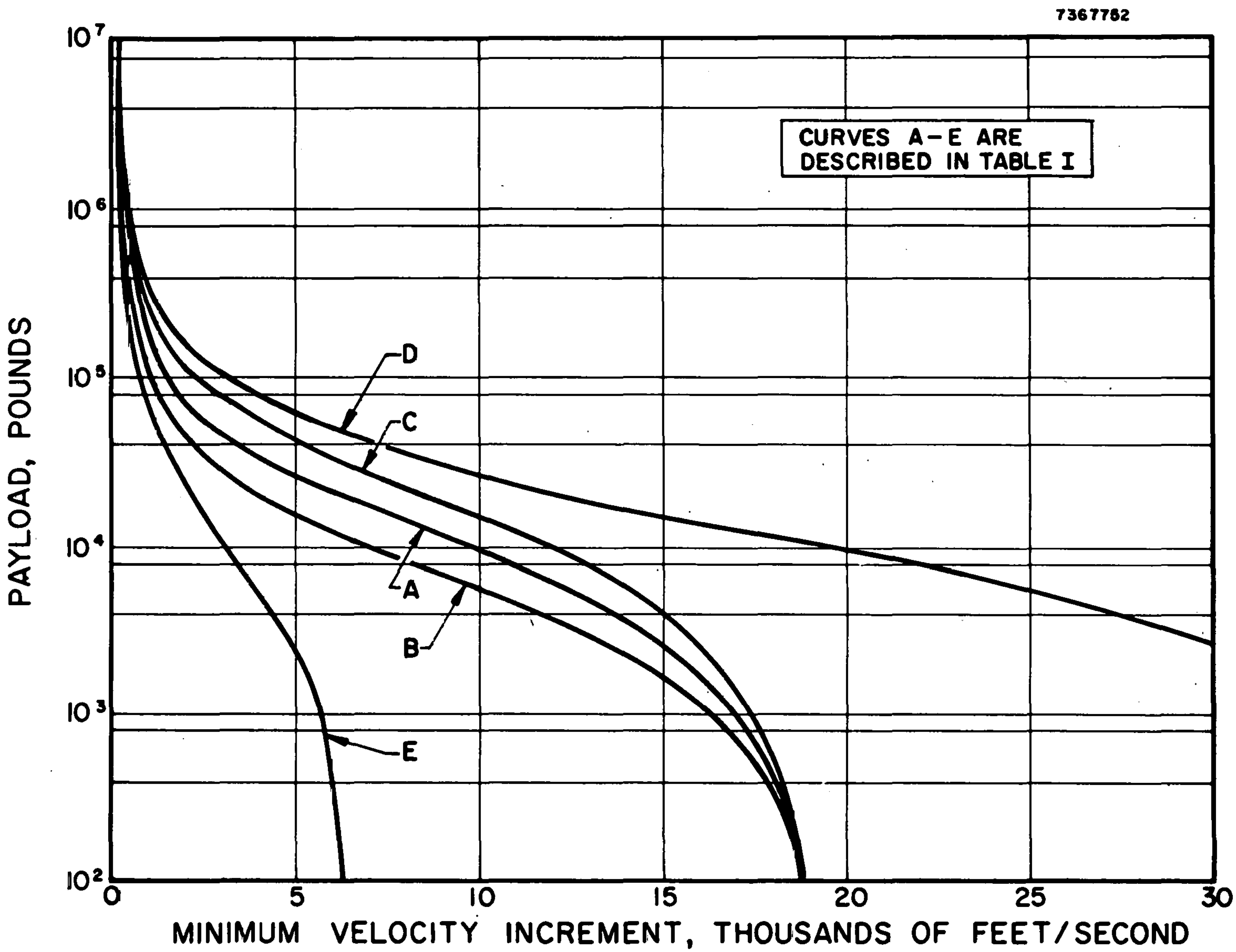

FIG. 5 ELECTRIC PROPULSION FOR LOW COST SLOW MISSIONS

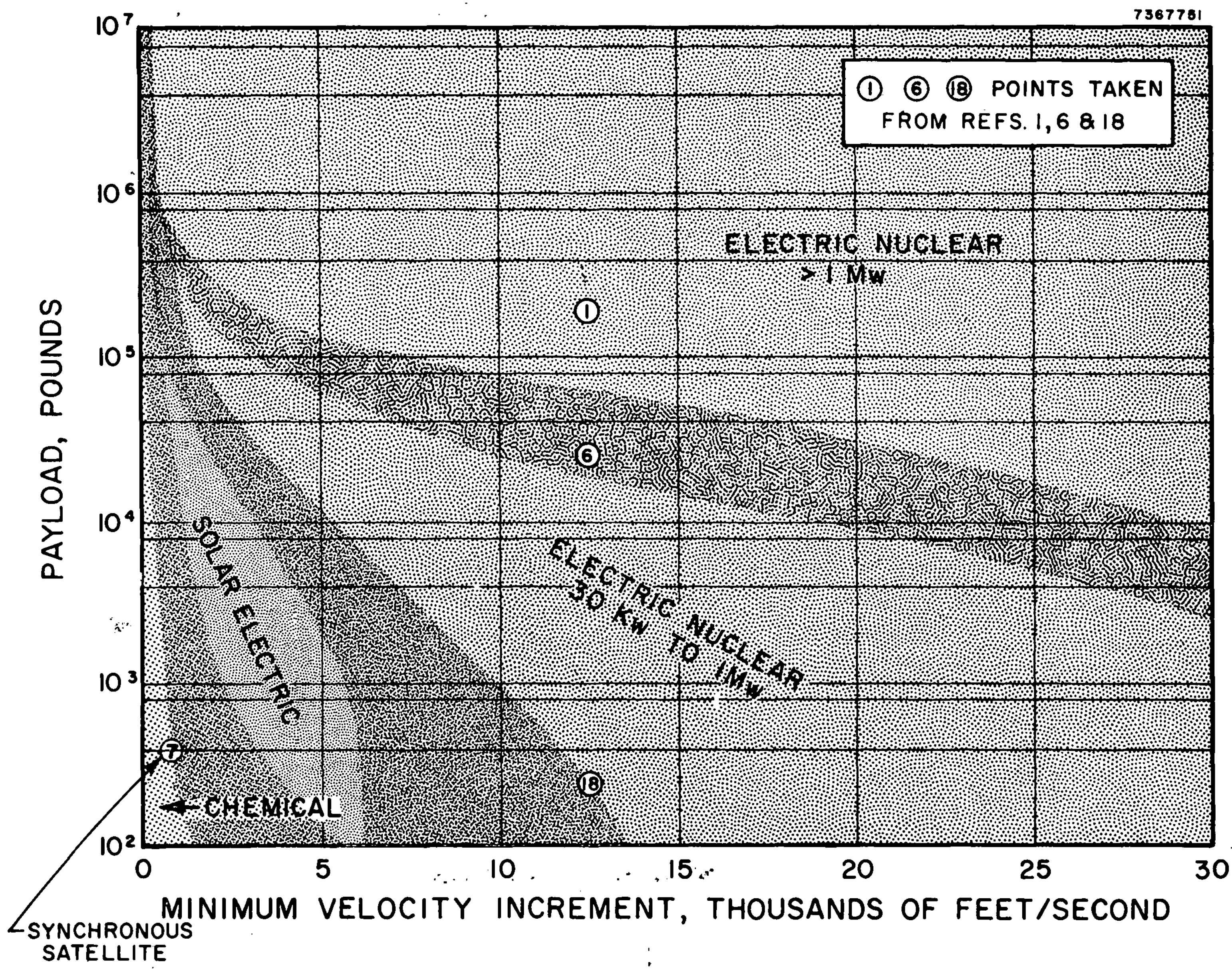

FIG. 6 PROPULSION SELECTION FOR LOW COST SLOW MISSIONS