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Performance of Nuclear-Electric Propulsion Systems in Space Exploration

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ABSTRACT

This paper presents an analysis of the payload capabilities of nuclear-electric spacecraft for interplanetary exploration. Results are presented in terms of vehicle terminal mass at its destination as a function of flight time for the mission. The missions which have been studied include probes and orbiters to most of the planets in the solar system, plus solar probes and flights out of the plane of the ecliptic.

For a given mission, flight time is chiefly determined by the initial acceleration of the spacecraft, whereas terminal mass is chiefly determined by the specific impulse of the thrust device. Some generalized curves are presented which indicate the initial accelerations required for several missions. The specific impulses required for various ratios of terminal mass to initial mass are shown for the same missions.

Once a particular set of mission flight time and payload requirements has been established, the optimum combination of ion engine characteristics, powerplant weight and power level, and launch vehicle can be specified to satisfy these requirements.

I. INTRODUCTION

This paper presents the results of studies which JPL has made of the capabilities of nuclear-electric powered spacecraft for solar system exploration. Missions to Mercury, Venus, Mars, Jupiter, Saturn, and Pluto have been studied, as well as solar probes and missions out of the plane of the ecliptic. Results are presented in terms of terminal mass or gross payload as a function of total flight time for the missions considered. Gross payload is defined as the sum of the weights of scientific instrumentation, telecommunications, thrust device, guidance and control equipment, structure, and tankage. The terminal mass consists of the gross payload plus the powerplant mass; although the powerplant is not included in the payload, it is expected to be available on arrival to power communications and space sciences experimental equipment.

An attempt has been made to optimize the relationship between the three major system components—boost vehicle, thrust device, and powerplant—as a function of the requirements specified for a mission. The assumptions necessary for each of these components are discussed below. For a specific mission, the booster availability, powerplant specific weight and power level, and thrust device performance must all be considered before a preferred system can be selected. It is possible, however, to present some reasonably generalized studies of mission capabilities at this time, and from these to select the ranges of operating parameters which appear both most useful and most likely to be achieved. In addition, specific results can be examined for a system utilizing a *Snap-8* type powerplant.

II. BOOST VEHICLE SELECTION

The performance of a boost vehicle is defined here as the payload weight which the booster can place in a 300-nm Earth orbit. This then becomes the initial spacecraft weight for the nuclear-electric spacecraft system. There are a large number of boost vehicles suitable for nuclear-electric spacecraft, from the *Atlas-Centaur* to the *Saturn* series and possibly the *Nova*, with or without nuclear boost stages. The performance of these vehicles is, at this date, still subject to change; therefore, rather than present results for several specific boost vehicles, it is preferable to normalize results with respect to the initial spacecraft weight and to speak of gross payload, powerplant, and expellant weights as fractions of the initial spacecraft weight M_o, so that results will be applicable to boosters of any size. To do this it is necessary to introduce a new parameter, the specific power level P°, defined as kilowatts of power per unit mass of spacecraft. The behavior of terminal mass with flight time can then be examined for various values of P°, and the most suitable power level for any boost vehicle can then be determined.

III. THRUST DEVICE SELECTION

At this point some assumptions are introduced concerning the efficiency of the thrust device to be used for spacecraft propulsion. The results presented show that a thrust device with efficient operation in the range of 5,000 to 15,000 seconds specific impulse will permit satisfactory payloads to be delivered to almost any destination within the solar system. Electrostatic ion motors are currently the most promising thrust devices for this operating region, with both cesium surface-ionization and mercury bombardment-type engines presently under intensive development. The relationship between efficiency, thrust, power, and specific impulse is characteristic of any power-limited thrust device, and is defined in the following manner:

$$\eta \equiv \frac{F I_s g}{2 P_0} \text{ and } I_s \equiv \frac{F}{\dot{M}} \tag{1}$$

where

F = thrust

g = gravitational acceleration at sea level

I_s = specific impulse

$\dot{M}$ = mass flow rate

P_0 = input power to thrust device

η = thrust device total efficiency

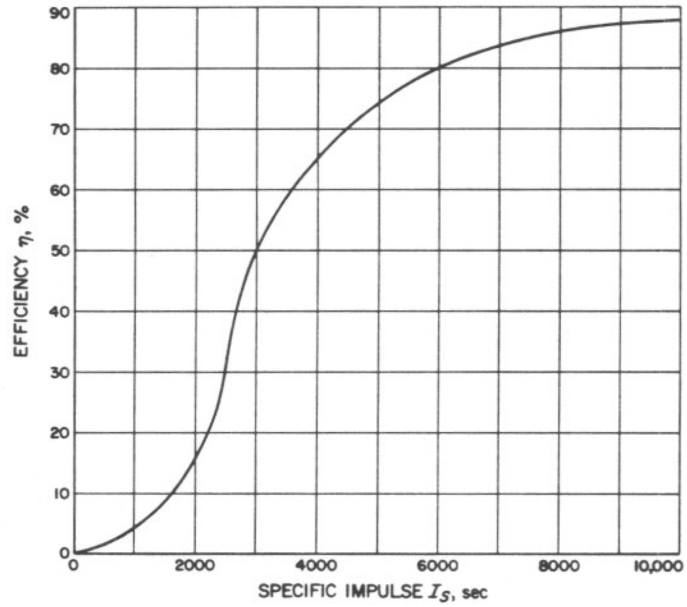

Fig. 1. Assumed ion motor efficiency

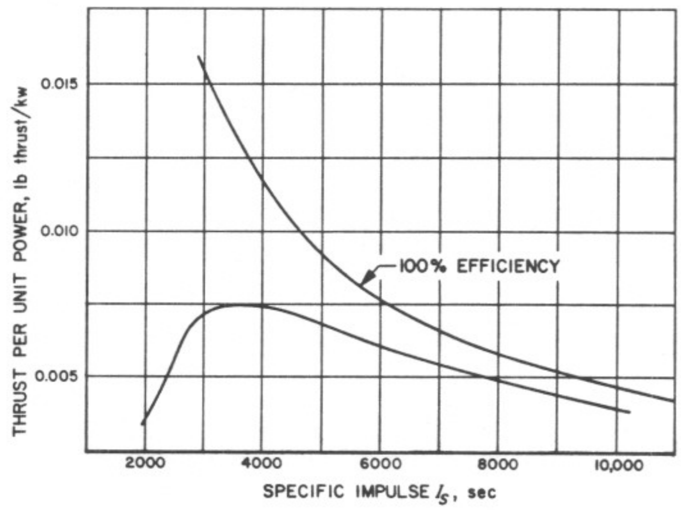

Fig. 2. Assumed thrust per unit power

The estimated variation of efficiency with specific impulse which has been assumed for these studies is shown in Fig. 1 (Ref. 1). A more useful representation is the variation of the thrust per unit power, F/P_0, as a function of I_s; this is shown in Fig. 2, in which the thrust per unit power is shown to have a definite peak value for this engine. Operation at peak thrust will determine the shortest possible flight time for any mission; but, as will be shown later, it is usually desirable to operate at a higher specific impulse than that corresponding to peak thrust. Clearly, there is no point in considering a specific impulse less than that corresponding to peak thrust. There are several other thrust devices which have potentially high efficiency (at least 50%) in the range of 1500 to 3000 sec I_s, but these are better suited to lower energy missions than to interplanetary exploration and so have not been considered here.

IV. TRAJECTORY OPTIMIZATION

The low-thrust, power-limited trajectories to be described here are taken from the analyses of Ref. 2 and 3.

All of the interplanetary and solar missions considered start from a low-altitude, circular Earth orbit (the initial altitude is 300 nm unless otherwise specified) and may terminate in one of three ways: (1) a "flyby" or probe mission, in which the vehicle intercepts the orbit of the destination planet but does not match velocities with the planet; (2) a capture mission, in which the vehicle arrives at the destination planet at the same time and with the same velocity as the planet and therefore may achieve an elliptical planetary orbit with only negligible additional fuel expenditure; or (3) an orbiter mission, in which the vehicle terminates in a stable circular orbit at some desired altitude above the destination planet.

Each mission has been divided into separate phases, and each of these is treated as a two-body problem. In the first phase, the vehicle spirals out from its initial parking orbit until Earth-escape energy is reached. The second phase is a heliocentric transfer of the vehicle from the orbit of Earth to the orbit of the destination planet (or to the desired ecliptic inclination). The third phase, for orbiter missions only, consists of a slow spiral inward around the destination planet until the final planetary altitude is reached. Dividing the mission into a series of two-body problems gives results which are accurate to a few percent and generally conservative compared to the actual three-body situation near planetary escape or capture.

The Earth-escape and planetary-capture spirals are assumed to be performed with constant tangential thrust. This has been shown to be very close to the optimum thrust program for planetary escape or capture, where the ratio of thrust to local gravitational field is low; and it is also probably the simplest thrust program to follow (Ref. 2).

The heliocentric transfer thrust program may be optimized in any one of several ways, depending on the constraints placed upon the vehicle acceleration vector. The rocket equation describing power-limited flight is

$$\frac{1}{M_t} = \frac{1}{M_0} + \frac{1}{2\eta P_0}\int_0^t [a(t)]^2\,dt \qquad (2)$$

where

M_0 = initial vehicle mass

M_t = terminal vehicle mass

P_0 = output of the powerplant (assumed equal to thrust device input power)

$a(t)$ = vehicle acceleration at time t

From Eq. (2) it is clear that the terminal mass may be maximized (or fuel consumption minimized) by minimizing the quantity $\int a^2dt$, for any efficiency and power level. This procedure was originally described by Irving and Blum (Ref. 4) and has been discussed for several planetary missions over a large range of flight times in Ref. 2 and 3. The absolute minimization of $\int a^2dt$ for a heliocentric transfer maneuver is attained when no arbitrary constraints are placed on either the magnitude or the direction of the spacecraft acceleration. The prescribed optimum acceleration program which results from this procedure requires a peak acceleration a_p at the beginning of the heliocentric transfer, then a gradually decreasing acceleration (and increasing specific impulse) until the middle of the transfer when the acceleration magnitude begins to rise again until it reaches a final absolute value of a_p. A typical transfer of this type is illustrated in Fig. 3 for a 160-day Mars rendezvous mission (Ref. 3). This variable-thrust, variable specific impulse program yields the minimum fuel consumption for any given trip time.

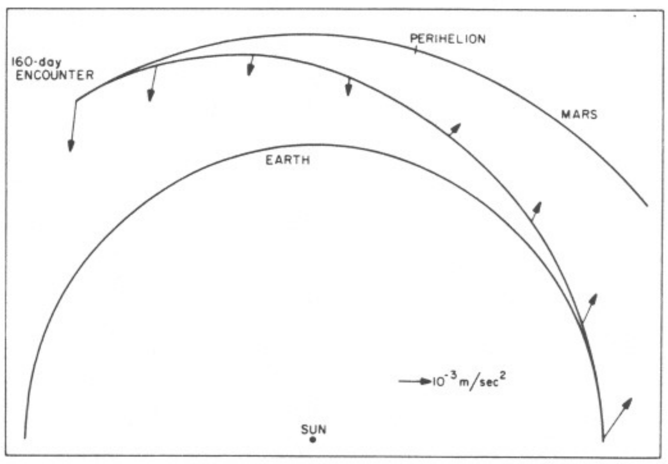

Fig. 3. Heliocentric transfer trajectory, variable thrust program

It does not seem realistic at this time, however, to design an ion motor that will be capable of operating over a continuously variable range of specific impulses (typical values from 2,000 to 20,000 sec). Therefore, the case of an optimized transfer in which the thrust vector is constrained to either have a constant magnitude or be zero at all times, with unspecified direction, has also been investigated (Ref. 3). The resultant thrust program consists of two periods of powered flight, at the beginning and end of the trip, with an interim coast period. The length of the coast period is optimized to yield the minimum value of $\int a^2 dt$. This is not precisely equivalent, in the constant thrust case, to maximizing the vehicle mass, but it has been shown (Ref. 3) that the minimum value of the integral is very nearly independent of specific impulse in most of the range of flight times and specific impulses of interest. As long as this is true, using the criterion of minimum $\int a^2 dt$ is equivalent to maximizing the final mass for a given flight time to within one or two percent.

It is very desirable to determine the extent to which this constant-thrust-plus-coast trajectory increases the fuel consumption, since the simplest way to operate any single thrust device will be at one level of thrust or specific impulse only. The results indicate that the amount of propellant required for the heliocentric part of the trip is in this case roughly 10 to 12% greater than the absolute minimum, using variable thrust. A typical constant-thrust-plus-coast trajectory is illustrated in Fig. 4 (Ref. 3) again for a 160-day Mars transfer.

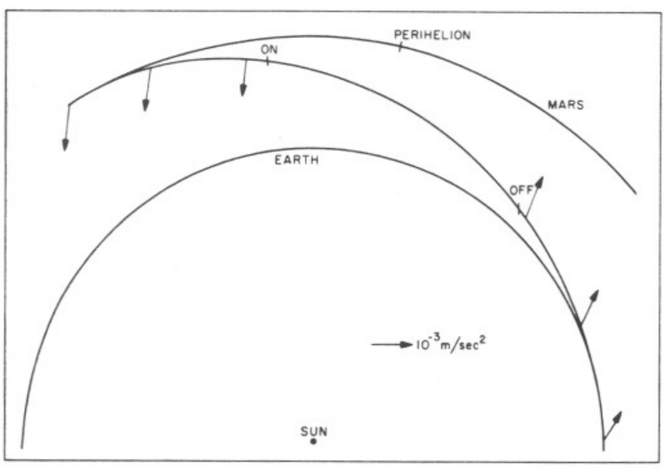

Fig. 4. Heliocentric transfer trajectory, constant thrust plus optimum coast

A third type of interplanetary transfer investigated is the so-called minimum time trip (Ref. 3). This requires the minimization of the integral in Eq. (2) subject to the constraint that the engine operate at a continuous constant thrust level with no coast period, and corresponds to the shortest possible transfer time at a given initial acceleration.

A comparison of these three transfer programs has been made for a complete Mars capture mission (Fig. 5) with a 60-kw powerplant on a spacecraft of 8500 lb initial mass. For the variable-thrust curve, it was assumed that the maximum acceleration required during transfer a_p was equal to the vehicle acceleration at Earth escape, and that the engine efficiency remained constant during the entire trip. For the other two cases, the thrust level during powered flight is constant. From Fig. 5 it is clear that the payload penalty in flying with constant thrust plus coast will not be severe. The minimum time trip, however, requires a fairly large payload penalty for a small decrease in flight time, compared to an optimum coast trip at the same specific impulse. In general, it is only near the region of peak engine thrust (low specific impulse) that an actual time decrease for a given payload can be obtained by operating the engine with no coast period. Therefore, the no-coast trajectory might be considered if time reduction is so critical that it is necessary to operate near peak thrust, or if difficulties are encountered in designing a restartable thrust unit.

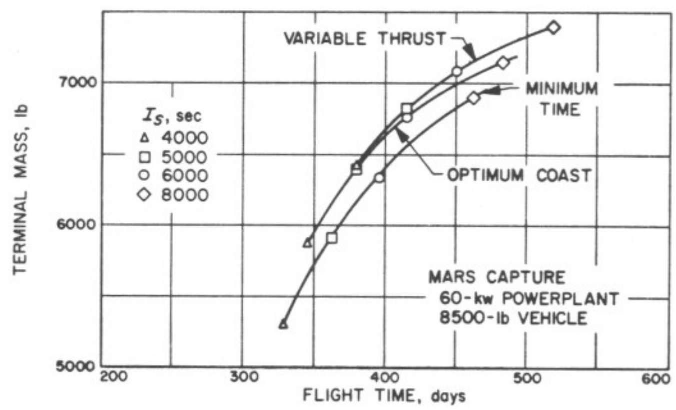

Fig. 5. Effect of thrust program on Mars capture mission

Figure 6 compares the three types of transfer programs for a Venus capture mission. The difference between variable and constant thrust is even less marked in this case, since the vehicle is operating in a higher gravitational field and, as in the case of planetary escape, the higher the local field the closer a constant-thrust program approaches the optimum. For missions to Jupiter and beyond, it is expected that the variable-thrust program will require about 12% less propellant than the constant-thrust-plus-optimum-coast program.

In summary, the three thrust programs do not differ greatly in performance, and a choice between them may be made largely on the basis of best operating conditions for the system components.

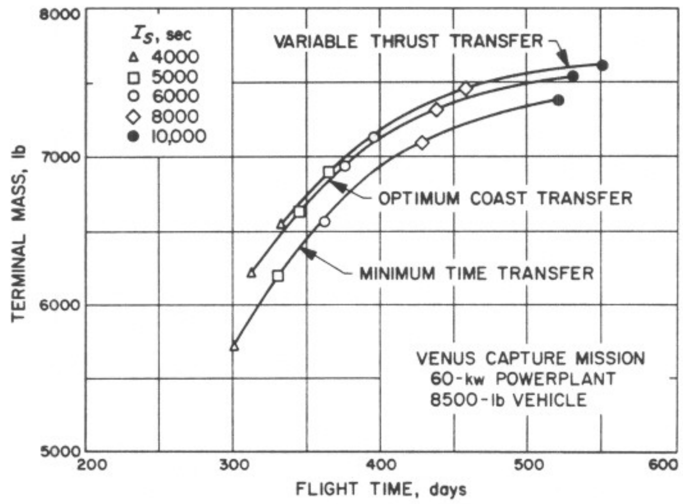

Fig. 6. Effect of thrust program on Venus capture mission

V. EFFECT OF INITIAL ALTITUDE

The effect of the initial Earth-orbital altitude on a typical mission (again a Mars orbiter) has also been investigated, and is shown in Fig. 7. It was necessary to assume, for this purpose, an expected variation in booster capability as a function of orbital altitude; for 8500 lb in a 300-nm orbit, a linear payload decrease of 200 lb per 100 nm of additional altitude has been assumed. Two effects on the nuclear-electric spacecraft result from a change in initial altitude. First, the energy required to reach Earth escape decreases slightly. In going from a 300-nm to a 1000-nm initial altitude, the velocity increment required to reach escape energy decreases by about 6%. Since the escape time for this mission is on the order of 1/3 of the total mission time, the total time is reduced by only 2%; this effect is therefore minor. The initial thrust level required for a given acceleration is, however, decreased by 20% (corresponding to the decrease in spacecraft weight), and this permits operation at a lower propellant mass flow rate and at higher efficiency for the nuclear-electric portion of the trip. These effects are discussed in Section VI. Qualitatively, for very long flight times, the lowest altitude available is best, but as flight times approach the shortest possible, with a given system, a higher altitude is preferable. The exact crossover points of the three curves shown depend critically on the variation of ion engine efficiency with specific impulse and on the variation of booster performance as a function of altitude; therefore Fig. 7 should be used only as an example of general behavior as a function of altitude; choice of a best initial altitude for a mission will clearly depend on precise information about the booster and spacecraft system capabilities.

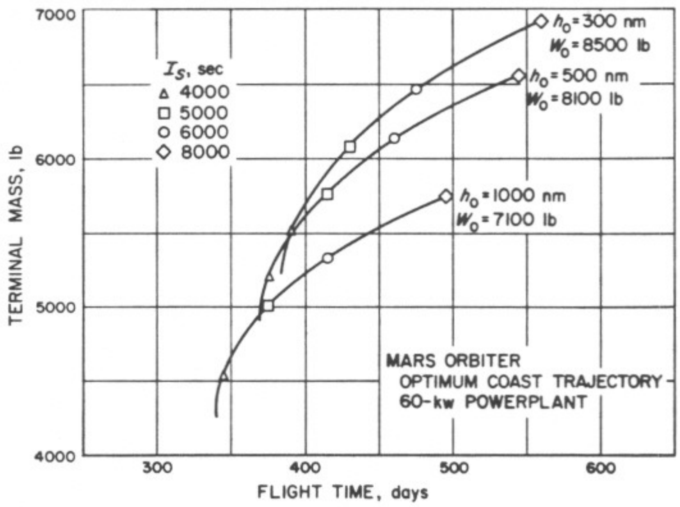

Fig. 7. Effect of initial altitude on performance

VI. GENERALIZED MISSION STUDIES

A good deal of information may be obtained about the requirements for a planetary mission in a generalized manner, without making any assumptions about the efficiency of the thrust device; this may be seen by combining Eq. (1) and (2) to eliminate ηP_0. Then, for any mission, the terminal mass as a function of flight time can be described in terms of a_0 and I_s only (where a_0 is the initial vehicle acceleration) without specifying any relationship between the two (i.e., without specifying a power level), as long as the criterion of minimizing $\int a^2 dt$ independent of specific impulse remains valid. This is illustrated in Fig. 8, which shows Mars orbiter missions with a constant-thrust-plus-optimum-coast type of heliocentric transfer. The two Mars missions labeled "Best Encounter" and "Worst Encounter" result from including the ellipticity of the Martian orbit in the terminal conditions of the heliocentric flight. It is probable that transfer opportunities will fall somewhere between the two extremes.

Several important conclusions can be drawn from a generalized plot such as this. It is clear that the flight time is almost completely dependent on the initial vehicle acceleration, while the propellant consumption (and hence the terminal mass) is almost entirely a function of the specific impulse of the thrust device. If, then, a given flight time is selected for a mission, the best thrust device operating point will be that which produces the required acceleration at the highest possible specific impulse. From this curve it can also be seen that if the ion engine performance assumed in Fig. 1 is in error, the effect will be to raise (or lower) the flight time at a given specific impulse, without a significant change in payload.

Each net point on these curves represents a certain specific power level P^*. If the thrust device efficiency is included, it is possible to superimpose curves of different specific power levels on this generalized mission curve as in Fig. 9. It is notable that the effect of increasing specific power level is to allow the delivery of a greater terminal mass in the same flight time or to shorten the flight time for the same terminal mass.

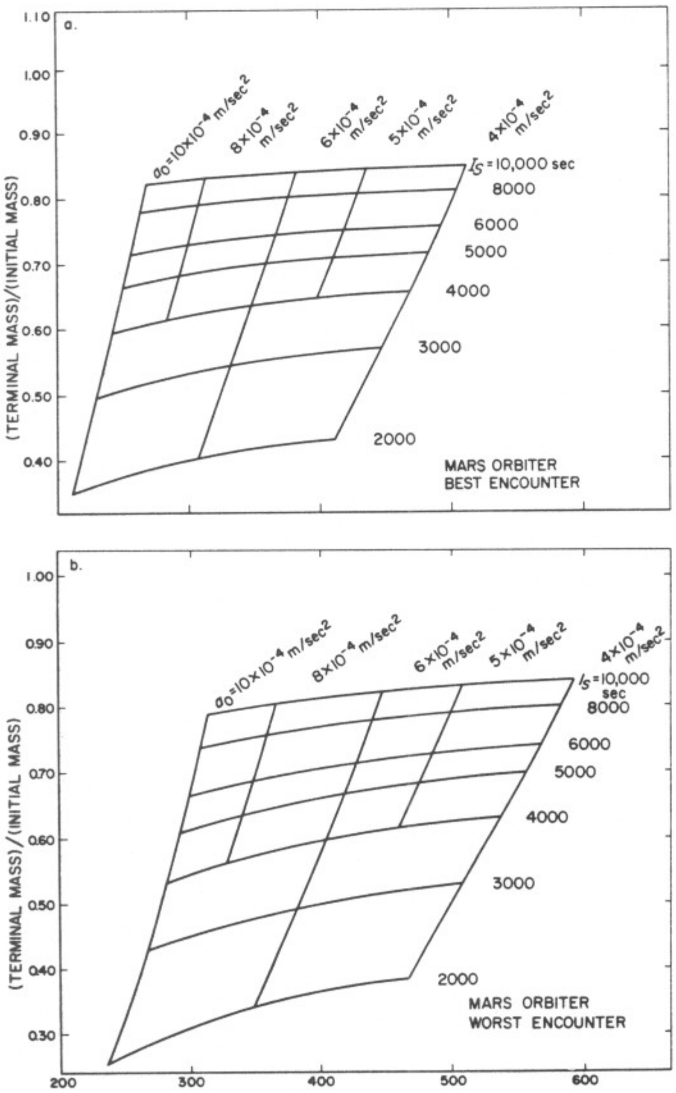

Fig. 8. Generalized Mars orbiter mission

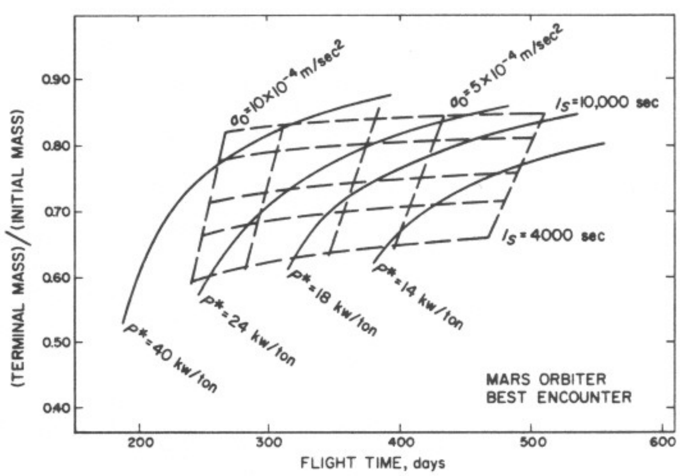

Fig. 9. Mission performance as a function of specific power level

VII. PERFORMANCE OF A 60-KW SYSTEM (SNAP-8 TYPE)

The first available nuclear-electric system suitable to interplanetary exploration will probably consist of a 3000-lb, *Snap-8* type power source, delivering 60 kw to the thrust unit, and carried on an 8500-lb spacecraft. The gross payloads (terminal mass minus powerplant mass) obtainable with such a configuration, for several missions are indicated in Fig. 10. The lunar orbiter mission shown is performed at constant thrust, with approximately a week of coast time near the end of the mission (Ref. 5). The terminal orbit is circular, 100 miles above the lunar surface. This is the least difficult mission considered herein, although lower-energy missions can of course be performed with this system. Both the Mars and Venus missions have been computed on the basis of constant-thrust plus optimum-coast heliocentric transfer. The Mercury flyby mission shown in Fig. 10 has a variable-thrust heliocentric transfer; therefore, payloads shown may be slightly high for this mission.

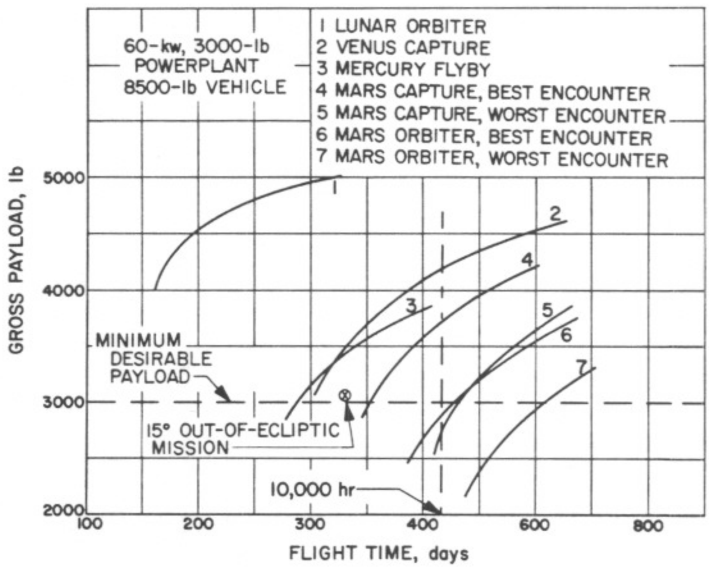

Fig. 10. Summary of 60 kw/ton spacecraft capability

Preliminary weight breakdowns of the gross payload have indicated that it is desirable to have at least 3000 lb of gross payload at the destination for a significant scientific experiment with high-power communication equipment, utilizing the power available from the reactor. This payload has been indicated by a horizontal line as the minimum desirable payload and includes several hundred pounds of scientific instrumentation and a wideband transmitter. The transmitter requires roughly 10 kw to the antenna for transmission of a good-quality video picture at an information rate of 10^6 bits per sec from about the distance of Venus. Assuming a transmitter efficiency of 20%, an input power of 50 kw is required, which would be available from the powerplant after the propulsion system is turned off. The remainder of the gross payload is allocated to guidance and control equipment, power conditioning equipment, instrumentation, structure, tankage, and the ion motor. If flight times of approximately 500 days were available, all of the missions indicated here would be feasible, with payloads above 3000 lb. However, since the lifetime of the *Snap-8* supply is expected to have an upper limit of 10,000 hours, and since it has no restart capability, only missions with a total flight time of less than 10,000 hours can be performed. (One additional mission which is an exception to this rule—a Jupiter flyby mission in which the propulsion time is less than 10,000 hours—can also be performed. The power supply is, of course, inoperative at arrival, and hence cannot be used for communications; nevertheless, a gross payload of 2900 lb can be delivered to Jupiter in 800 days.)

If the regions of the performance plot in Fig. 10 corresponding to excessive flight times or insufficient payloads are disregarded, only those missions remaining in the upper-lefthand region of the plot are within the capability of the system. Note that all such missions are well within the allowable area, so there will be some leeway if the postulated engine, booster, and powerplant performance levels are not met. In addition, all of the allowable missions have the 60-kw power supply still operative on arrival.

VIII. PERFORMANCE OF SYSTEMS OF HIGHER SPECIFIC POWER LEVEL

Inasmuch as it is not clear at the present time what the next generation of boost vehicles suited to advanced nuclear-electric propulsion missions will be, performance is now examined in terms of the parameter P^*, or kilowatts of power per unit mass of initial weight, which was introduced earlier. Since P^* suffices to determine performance, the missions discussed for the 60-kw system can be applied to any vehicle with the same P^* (14 kw/ton). The same missions as Fig. 10 are shown in Fig. 11, in which terminal mass ratio is shown as a function of flight time for a specific power level of 14 kw/ton (i.e., the capability of a 20,000-lb spacecraft with 140 kw or a 50,000-lb spacecraft with 350 kw as well as of the 60 kw, 8500-lb spacecraft).

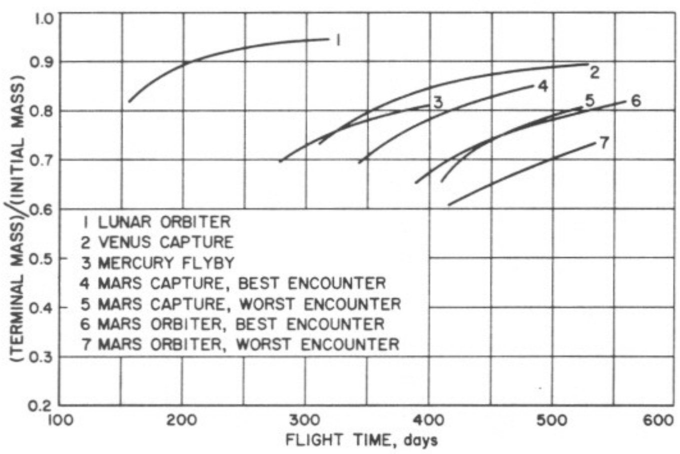

Fig. 11. Summary of 14 kw/ton vehicle capability

In this and the following mission curves the Mars and Venus missions have a constant-thrust-plus-optimum-coast thrust program; all others have a variable-thrust, variable-specific-impulse thrust program, and therefore the performance values for them are slightly optimistic.

For many of the missions examined, the flight times at a specific power level of 14 kw/ton appear to be excessively long. In general, payloads are satisfactory, but it is desirable to investigate further the requirements of a higher powered spacecraft. Additional reliability and lifetime may be obtained by completely or partially deactivating the spacecraft during coast on missions to the outer planets. For Mercury, Venus, and Mars missions, as well as solar probes, with an optimum coast type of trajectory, the powered flight time is generally between 65 and 85% of the total flight time. It is questionable whether it is worth while to attempt a powerplant shutdown during such a brief coast. For Jupiter, Saturn, Uranus, Neptune, and Pluto missions, however, the powered flight time is on the order of 20 to 35% of the total flight time, and it may be reasonable to operate the powerplant at perhaps 10 to 15% of capacity during the long coast periods which result. This may effect a considerable saving in total megawatt hours and, thus, in powerplant weight.

Figures 12 and 13 show the performance of a vehicle with a specific power level of 40 kw/ton (a 20,000-lb spacecraft with a 400-kw or a 50,000-lb spacecraft with a 1-megawatt power supply).

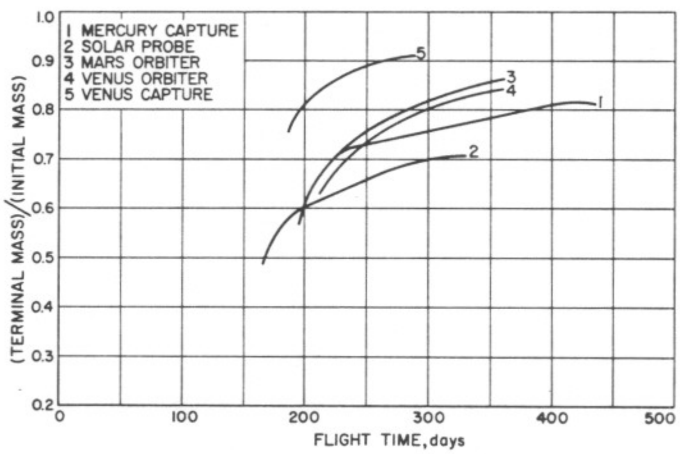

Fig. 12. Summary of 40 kw/ton vehicle near-planet capability

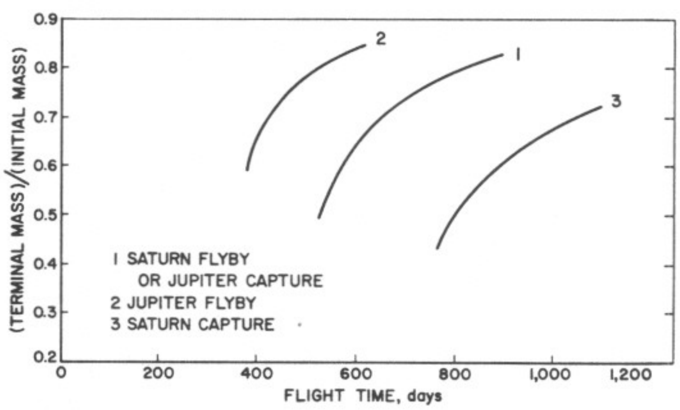

Fig. 13. Summary of 40 kw/ton vehicle outer-planet capability

Figure 12 shows the near-planet missions—Mars, Venus, Mercury—and a solar probe mission which approaches tangentially to 20 solar radii. All have flight times of less than a year and will deliver 60 to 90% of the initial weight to destination.

Figure 13 shows several outer planet missions for a specific power level of 40 kw/ton. Another mission, not shown here, is a Pluto flyby. This system can deliver 40% of its initial weight to Pluto in 3½ years. (Any practicable chemical system will require 20 years to reach Pluto.) Other missions which can be performed with a 40 kw/ton vehicle are probes out of the plane of the ecliptic. Here, the vehicle can carry 62% of its initial weight 15 deg out of the ecliptic, 47% to an inclination of 30 deg, and 35% to an inclination of 45 deg.

Table 1. Nuclear-electric missions

Mission	Terminal mass / Initial mass	P° kw/ton	Flight time days
Mars orbiter	0.70	200	100
		40	225
		14	430
Mercury capture	0.70	200	108
		40	225
Solar probe (to 20 solar radii)	0.70	200	85
		40	310
Jupiter flyby	0.70	40	425
		14	800
Jupiter capture or Saturn flyby	0.70	200	365
		40	660
Saturn capture	0.70	200	550
		40	1050
Pluto flyby	0.70	40	2000
	0.38	40	1250
	0.87	200	1250

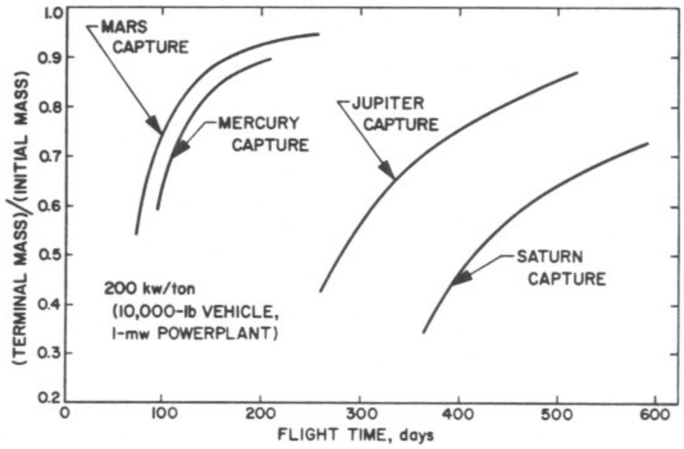

Fig. 14. Summary of 200 kw/ton vehicle capability

Finally, Fig. 14 shows a very-high-powered spacecraft, operating with 200 kw/ton of initial weight. This spacecraft will require an extremely lightweight powerplant (of perhaps 2 to 3 lb per kw), and so is shown here purely for comparison purposes, not as a system which is likely to exist in the near future.

A summary of electric propulsion performance for specific power levels of 14, 40, and 200 kw/ton is given in Table 1.

The time required for escape from the Earth's gravitational field is nearly proportional to the initial acceleration, in the range of a_0 which is under consideration—between 4×10^{-4} m/sec and 5×10^{-3} m/sec^2 (Ref. 2). Therefore, any increase in specific power level is reflected as a proportionate decrease in escape time (at the same specific impulse). However, for a heliocentric transfer maneuver, the time required varies more slowly than the corresponding increase in power. Therefore, increasing

For near-planet missions, where the heliocentric transfer times are fairly short, the escape time is a significant fraction of the total time. When this condition holds (as it does in the case of $P^* = 14$ kw/ton), increasing power level is very effective in reducing the total flight time. However, once the planet-centered portion of the trip becomes small compared to the heliocentric transfer time, it becomes much more difficult to reduce flight time. This is true both for flyby and capture missions, although the times involved are, of course, different for each case.

The effect of a proper choice of specific power level on flight time is illustrated in Fig. 15. Flight times for several missions (using variable thrust) are shown as a function of P^*. A single specific impulse of 8000 sec has been used; changing this would raise or lower the curves. However, it is clear that there is a region (the knee of each curve) in which, for a given mission, flight time and power have a reasonable interdependence, and this is the region where it is most advantageous to make tradeoffs between

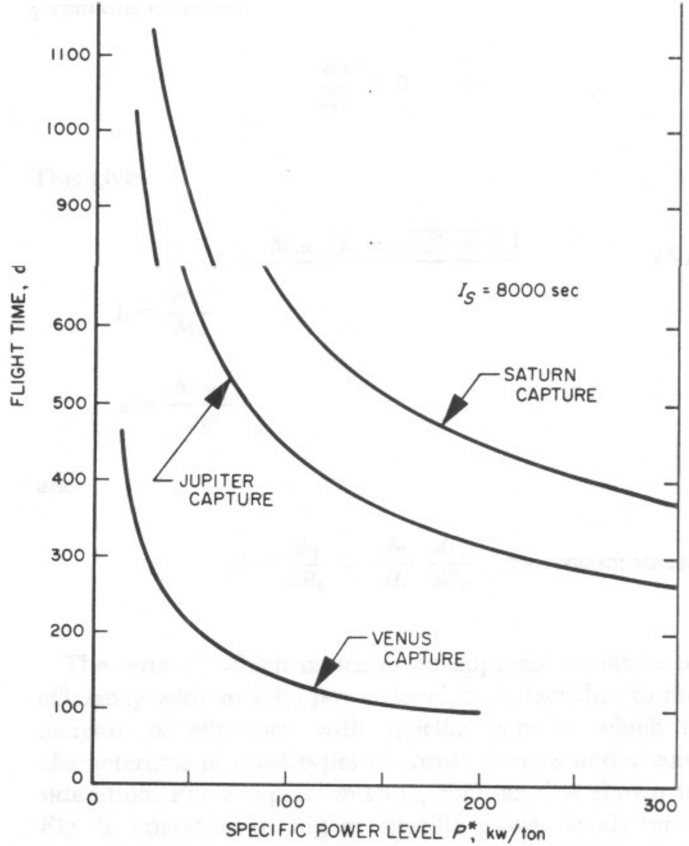

Fig. 15. Variation of flight time with specific power level