Comparison of Advanced Propulsion Concepts for Deep Space Exploration

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Equations and charts are presented which permit rapid estimation of propulsion system performance requirements for some typical deep space missions. The simplicity results from use of gravity-free equations of motion, which are shown to yield good approximations to trip times obtained with solar gravity and planetary motion included. The agreement is satisfactory for missions that do not enter or depart from low orbits about the major planets. A number of advanced propulsion concepts for which performance estimates are available are compared with respect to their capability for fly-by, rendezvous, and round-trip planetary missions. Based on these estimates, the gas-core nuclear fission rocket and the pulsed fusion rocket yield the fastest trip times to the near planets. For round trips to Jupiter and beyond, the controlled fusion rocket shows progressively superior capabilities. Several propulsion concepts based on use of impinging laser beams are found to be noncompetitive with the other advanced concepts for deep space missions.

Nomenclature

F = thrust
g_0 = gravitational acceleration at Earth's surface, 9.8 m/sec^2
I = specific impulse
J = mission difficulty parameter, type II systems
k = ratio of tankage mass to propellant mass
m = mass
N = number of stages
P_j = exhaust jet power
T = total trip time
T_p = propulsion time
t = time
v = velocity
Δv = velocity increment
α = specific mass, k/gw
α' = specific mass, kg/kw
γ = ratio of propulsion system mass to initial total mass

Subscripts

c = circular
j = jet
n = nth stage
0 = initial values
pay = payload
pr = propellant
ps = propulsion system
s = structure
t = tankage

Introduction

THE relationship between propulsion system performance and mission capability was discussed in Ref. 1 for current and future systems that seem suitable for manned exploration of the solar system. For the future systems the principles of operations and the performance to be expected were outlined. Subsequently, some changes have been made in estimated performance of gas-core nuclear fission rockets[2] and controlled thermonuclear fusion rockets.[3,4] Studies have also been published on the performance possibilities of fusion microbomb propulsion systems[5] and laser-powered systems.[6,7] One purpose of this paper is therefore to update the discussion in Ref. 1. Another purpose is to extend the comparison in Ref. 1 to the use of advanced propulsion concepts for high-payload unmanned exploration of the planets, since such missions will probably precede manned flights. A third purpose is to illustrate the usefulness of simple field-free space trajectories for preliminary comparisons of advanced propulsion concepts.

Received February 28, 1972; revision received August 10, 1972.
Index categories: Electric and Advanced Space Propulsion; Nuclear Propulsion; Spacecraft Mission Studies.
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Classification of Propulsion Systems

As in Ref. 1, the basis for comparison of propulsion systems is the trip time required for a given mission and payload ratio. The propulsion systems are again divided into the two types described in Ref. 1. Type I consists of those systems whose mission performance is primarily limited by the "maximum specific impulse" I attainable. For type II system, mission performance is limited primarily by the "minimum specific mass" α attainable (α is the ratio of propulsion system mass to jet power produced).

The main reason for defining these two types of propulsion system is that they produce two different types of space trajectory because of their difference in thrust/weight ratio. In terms of specific impulse and specific mass, this ratio is

$$F/m_{ps}g_0 = 2000/\alpha' I g_0^2 \tag{1}$$

where α' is specific mass in the usual units of kilograms per kilowatt and I is in seconds, m_{ps} is propulsion system mass, and g_0 is gravitational acceleration at Earth's surface (9.8 m/sec^2). Equation (1) results from the following expression for jet power P_j

$$P_j = \tfrac{1}{2}\dot{m}v_j^2 = Fv_j/2 = FIg_0/2 \tag{2}$$

where $\dot{m}$ is propellant ejection rate and v_j is jet exhaust velocity.

Figure 1 shows the range of specific mass and specific impulse covered by the major type I and type II systems. Also shown are lines of constant thrust/weight ratio from Eq. (1). This figure shows that the available studies of type I propulsion systems yield thrust-weight ratios generally above 0.05. For the entire vehicle, these ratios could be much lower if propulsion system mass is only a small part of the total vehicle mass. However, the accelerations produced are generally sufficiently high that the propulsion time is very much smaller than the trip time for planetary missions, so that the thrust can be considered to take place in impulsive bursts. This permits use of unpowered (free-fall) trajectory equations.

Type II systems, since they are not specific-impulse limited, can produce the optimum specific impulse needed to minimize the sum of propellant and propulsion system mass. However, as Fig. 1 shows, for high specific impulse they have

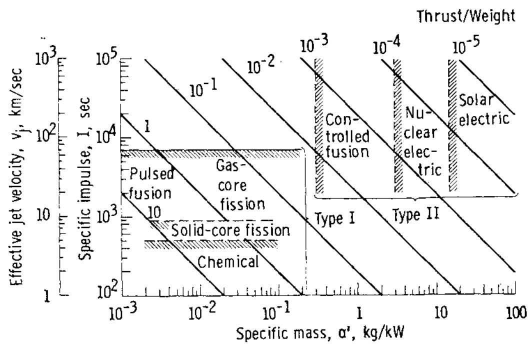

Fig. 1 Propulsion system performance parameters.

thrust/weight ratios less than 0.01. For such low accelerations, propulsion time may be of the same order of magnitude as trip time. In fact, for gravity-free space, the optimum propulsion time for type II systems is readily shown to be two-thirds of the total trip time. The separation into types I and II can be regarded as equivalent to the more conventional division into high-thrust and low-thrust propulsion systems.

The performance estimates shown in Fig. 1 differ from the values of Ref. 1 only for the most advanced concepts. For the controlled-fusion rocket, further optimization of system masses and consideration of higher power has reduced the estimated specific mass from 1 to approximately 0.3 kg/kw (Ref. 3).

For the gas-core fission rocket, more recent analysis using cooled chamber walls and radiative heat rejection[2] resulted in an increase in maximum specific impulse from about 2500 sec to as high as 7000 sec, with thrust/weight ratios as high as 0.17.

The pulsed fusion system, propelled by a series of laser-ignited fusion-powered explosions, was not included in Ref. 1, because no system mass and performance studies were available. Since then, Ref. 5 has appeared, with estimated maximum specific impulse of approximately 5000 to 7000 sec. This (as yet uncertain) limit is imposed primarily by ablation of the "pusher" used to transform the shock impulses into a moderate acceleration of the vehicle and by ability to direct the blast within small angles. Thrust/weight ratios as high as 3.6 are estimated in Ref. 5 for this propulsion system. Its projected performance is therefore superior to that of the gas-core fission rocket. However, the feasibility studies of the pulsed fusion system are less advanced that those for the gas-core fission rockets, and the performance estimates may have correspondingly greater uncertainty.

Other advanced propulsion concepts that have been proposed are not included in Fig. 1 either because they lack sufficient physical realism to estimate performance or they can be considered as special cases of the systems shown. Thus, photon rockets are type II systems whose specific mass depends on the power source selected to generate the photons. Their specific impulse ($I \approx 3 \times 10^7$ sec) is far beyond the optimum for any mission achievable with specific masses shown in Fig. 1. Although lower specific masses may eventually be possible by some as yet unknown method (such as direct mass annihilation) no basis for estimates is now available.

Other proposed forms of propulsion involve the use of powerful lasers stationed on the Earth, moon, or an Earth satellite. These lasers would be directed at the vehicles to be propelled. The impinging laser power would be used in one of three ways: 1) to heat hydrogen propellant emerging from a rocket nozzle,[6] 2) to generate onboard electric power for electric propulsion, or 3) to push the spacecraft directly by reflecting or absorbing the laser radiation.[7,8]

Each of these methods shares the technological problems associated with generating very high power highly-collimated laser beams and transmitting them with high pointing accuracy over long distances. If the required huge power stations become available and if nearly diffraction-limited laser beam divergence is achievable in giant beams, transmission distances of several thousand kilometers may become feasible without excessive intensity drop using optical frequencies and very large optics. Even greater transmission distances could be visualized if far ultraviolet or x-ray lasers turn out to be feasible, but no plausible methods of generating highly-collimated beams using such high-energy photons have as yet been proposed. Use of laser transmitting stations for propulsion is limited to an initial propulsion period on departure from the vicinity of the station. Space mission capability is therefore limited to flyby missions unless other forms of propulsion are carried along.

Method 1 (a type I system) could theoretically produce specific impulses comparable to those of a gas-core fission rocket, but without the nuclear problems. It could therefore conceivably be used to launch from the Earth's surface.[6]

Method 2 would correspond to a type II system (electric propulsion) with specific mass dependent on the type of conversion system. If photo-voltaic cell arrays are used, specific masses comparable to those of solar-electric propulsion systems should be achieved, with perhaps some (as yet unknown) advantage in conversion efficiency due to monochromaticity and higher intensity. However, no great increase in deep space mission capability seems likely from this concept.

Method 3 is similar to the "photon sails" frequently discussed for using solar radiation pressure in planetary space, but the laser could produce higher initial radiation pressure. Since the specific impulse is infinite (no onboard propellant is ejected), such systems are neither type I or type II. Their mission performance is limited instead by thrust/weight ratio attainable, which in turn is determined by the tolerable or attainable intensity of impinging radiation and the allowable thinness of the reflecting "sail."

Performance parameters for methods 1 and 2 are thus included in the systems shown in Fig. 1 while method 3 requires a different type of analysis. Such an analysis was made in Ref. 8 and leads to the conclusion that the method is not competitive with the systems in Fig. 1 for beam power levels that seem achievable in the foreseeable future.

Missions in Field-Free Space

The simplest approach to estimating the mission capability of propulsion systems is to evaluate their performance in gravity-free space. The resulting relationship between the distance travelled and trip time shows primarily relative propulsion system effectiveness, but it also yields a good approximation to accurately computed planetary trip times for certain types of missions. The limitations of these field-free mission studies are considered in a later section.

Three types of mission are considered: flyby, rendezvous, and round trip. For each mission, the field-free equations approximate the energy requirements for that portion of the mission which excludes escape from the departure planet and capture by the destination body. Thus, for a flyby mission, only a single propulsion period is required, and the energy increment corresponds to that needed after achievement of escape velocity from the Earth. For the (one-way) rendezvous mission, two propulsion periods are needed, the first to achieve additional velocity beyond Earth escape for transfer to the destination, and the second to match the velocity of the destination planet or object. For the round-trip mission, four propulsion periods are needed, the first two corresponding to those of the outgoing rendezvous mission and the second two corresponding to the return rendezvous mission. In the field-free approximation, the vehicle begins and ends its rendezvous and round-trip mission legs at zero velocity. Consequently, each propulsion period provides the same velocity increment.

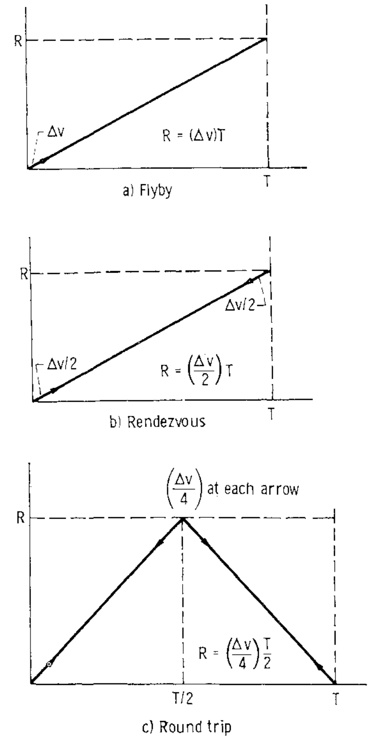

Fig. 2 Field-free space missions for type I systems.

Type I Propulsion System

For type-I propulsion systems, the velocity increments are produced impulsively and the three mission types are represented in Fig. 2. For each mission type, R is the distance to the destination, T is the total trip time, and Δv is the total velocity increment needed for the mission. Thus, the rendezvous mission requires twice the Δv of the flyby mission, and the round-trip mission requires eight times the Δv of the flyby mission for a given trip time.

To find Δv achievable in terms of specific impulse (the primary type I performance parameter), consider a multistage vehicle, with all mass ratios identical for each stage. Then the total initial mass m_0 of the $(n+1)$ stage can be written

$$m_{0,n+1} = m_{0,n} - m_{pr,n} - m_{t,n} - m_{ps,n} \qquad (3)$$

where m_{pr} is propellant mass, m_t is propellant tankage mass, and m_{ps} is the mass of the propulsion system and any remaining vehicle structure. The tankage mass is generally considered to be proportional to propellant mass ($m_t = km_{pr}$) so that Eq. (3) can be written

$$m_{0,n+1}/m_{0,n} = 1 - (1+k)m_{pr,n}/m_{0,n} - \gamma \qquad (4)$$

where $\gamma = m_{ps,n}/m_{0,n}$. The propellant mass is given by the rocket equation

$$m_{pr,n} = m_{0,n}[1 - e^{-\Delta v_n/v_j}]$$

where v_j is the effective exhaust velocity of the rocket. Hence, Eq. (4) becomes

$$m_{0,n+1}/m_{0,n} = (1+k)e^{-\Delta v_n/v_j} - k - \gamma \qquad (4a)$$

The net payload of the mission m_{pay} can be regarded as the initial mass of the $(N+1)$ stage where N is the total number of propulsion stages. Thus,

$$m_{pay}/m_{0,1} = [(1+k)e^{-\Delta v_n/v_j} - (k+\gamma)]^N \qquad (5)$$

From Eq. (5), the total Δv achievable is

$$\Delta v \equiv N\Delta v_n = Ng_0 I \ln\left[\frac{1+k}{(m_{pay}/m_{0,1})^{1/N} + k + \gamma}\right] \qquad (6)$$

where $v_j = g_0 I$ has been substituted.

Type II Propulsion System

For type II systems, the propulsion time is a major portion of total trip time, so that simple impulsive velocity increments cannot be assumed. The mass ratio per stage, as for type I, is given by Eq. (4). Since the propulsion system mass ratio γ, is generally much larger than k for these systems, k can be neglected. As shown in Ref. 1 and elsewhere, Eq. (4) can be written

$$m_{0,n+1}/m_{0,n} = [1 + \alpha J_n/2\gamma]^{-1} - \gamma \qquad (7)$$

and the propellant mass ratio is

$$m_{pr,n}/m_{0n} = 1 - [1 + \alpha J_n/2\gamma]^{-1} \qquad (8)$$

where α is the propulsion system specific mass (kg/w) and

$$J_n = \int_0^{T_{pn}} a^2 dt \equiv a_0^2 T_{pn} \quad \mathrm{m^2/sec^3} \qquad (9)$$

The quantity J_n is called the mission difficulty parameter (per stage), a_0 is the mean acceleration, and T_{pn} is the propulsion time during each propulsion period. Optimization of the mass ratio in Eq. (7) then yields

$$\gamma_{opt} = (\alpha J_n/2)^{1/2}[1 - (\alpha J_n/2)^{1/2}] \qquad (10)$$

Using this optimum value in Eq. (7) yields

$$m_{0,n+1}/m_{0,n} = [1 - (\alpha J_n/2)^{1/2}]^2 \qquad (11)$$

and the payload ratio with N stages becomes

$$m_{pay}/m_{0,1} = [1 - (\alpha J_n/2)^{1/2}]^{2N} \qquad (12)$$

The total mission difficulty parameter in terms of specific mass and payload ratio is then

$$\alpha J \equiv N\alpha J_n = 2N[1 - (m_{pay}/m_{0,1})^{1/2N}]^2 \qquad (13)$$

In terms of total propulsion time T_p, Eq. (9) yields

$$J \equiv NJ_n = Na_0^2 T_{pn} = a_0^2 T_p \qquad (14)$$

The total velocity increment is

$$\begin{aligned}\Delta v &= a_0 T_p = (JT_p)^{1/2} \\ &= (2NT_p/\alpha)^{1/2}[1 - (m_{pay}/m_{0,1})^{1/2N}] \qquad (15)\end{aligned}$$

Equations (13–15) enable evaluation of distance vs trip time relationships in terms of the payload ratio and the primary performance parameter α.

The distance vs time relationship for the three types of mission are illustrated in Fig. 3 and are derived by simple addition of constant-acceleration and coasting periods. For flyby,

$$\begin{aligned}R &= \tfrac{1}{2}a_0 T_p^2 + a_0 T_p(T - T_p) \\ &= (J/T_p)^{1/2}(TT_p - \tfrac{1}{2}T_p^2)\end{aligned}$$

Differentiation yields $(T_p)_{opt} = \frac{2}{3}T$, so that, with Eqs. (13) and (15)

$$\begin{aligned}R &= (2N/\alpha)^{1/2}\left[1 - \left(\frac{m_{pay}}{m_{0,1}}\right)^{1/2N}\right](\tfrac{2}{3}T)^{3/2} \\ &= \Delta v(\tfrac{2}{3}T) \qquad (16)\end{aligned}$$

Similar derivations yield the expressions shown in Fig. 3 for rendezvous and round trip missions.

As for type I systems, the distances that can be travelled for a given trip time, payload ratio, and propulsion system parameter are in the ratio $1 : \frac{1}{2} : \frac{1}{8}$ for flyby, rendezvous, and round trip, respectively. The range of validity of the above expressions is again limited to nonrelativistic velocities.

Summary of Equations

The preceding derivations have been carried out in the international system of units. For astronomical distances and long trip times, it is convenient to express R in astronomical units (1 a.u. = 1.495×10^{11} m) and T in years (1 yr = 3.155×10^7 sec). Furthermore, the specific mass is generally expressed

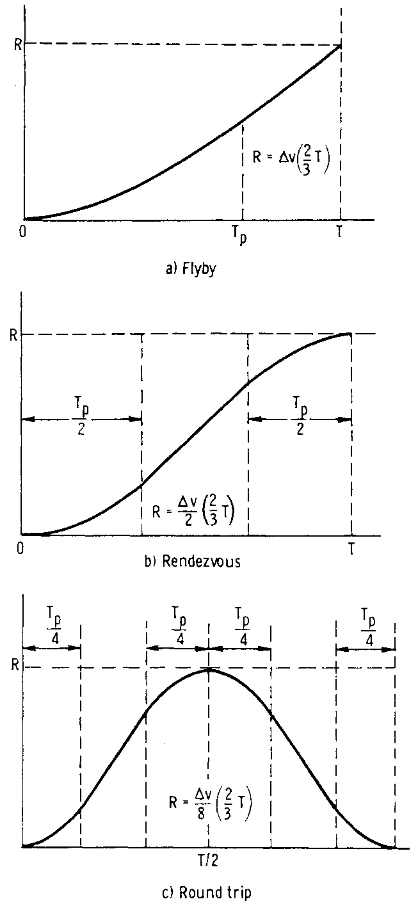

Fig. 3 Field-free space missions for type II systems, (Optimum propulsion system mass ratio and coast times).

n kilograms per kilowatt (denoted by α') rather than kilograms per watt α. In these units, the equations for distance vs trip time are as follows:

Type I propulsion, flyby mission

$$R_F = 4.8 \times 10^{-3}\, NIT \log_{10}\left[\frac{1+k}{(m_{pay}/m_{0,1})^{1/N} + k + \gamma}\right] \quad (17)$$

Type II propulsion, flyby mission

$$R_F = 29(N/\alpha')^{1/2}\, T^{3/2}[1 - (m_{pay}/m_{0,1})^{1/2N}] \quad (18)$$

For both type I and type II propulsion, distances for rendezvous and round-trip missions are related to the flyby distance by

$$R_R = \tfrac{1}{2}R_F \quad (19a)$$

and

$$R_{RT} = \tfrac{1}{8}R_F \quad (19b)$$

For the following comparisons of relative mission performance, the mass ratio per stage, $m_{0,n+1}/m_{0,n}$, is assumed to be equal to 0.1. Consequently, in Eqs. (17) and (18), $(m_{pay}/m_0)^{1/N} = 0.1$. Also for type I systems, k and γ are assumed to be zero to illustrate best possible performance. Equations (17) and (18) then become

$$\text{Type I:} \quad R_F = 4.8 \times 10^{-3}\, NIT \quad (17a)$$

$$\text{Type II:} \quad R_F = 19.8(N/\alpha')^{1/2}\, T^{3/2} \quad (18a)$$

Mission Capability

Interstellar Distances

Results from Eqs. (17a) and (18a) are shown in Fig. 4a and 4b, respectively, for values of distances, trip times, and performance parameters to the limits of validity of nonrelativistic equations. The plots are for 4-stage vehicles, with over-all payload ratio of 10^{-4} (0.1 per stage).

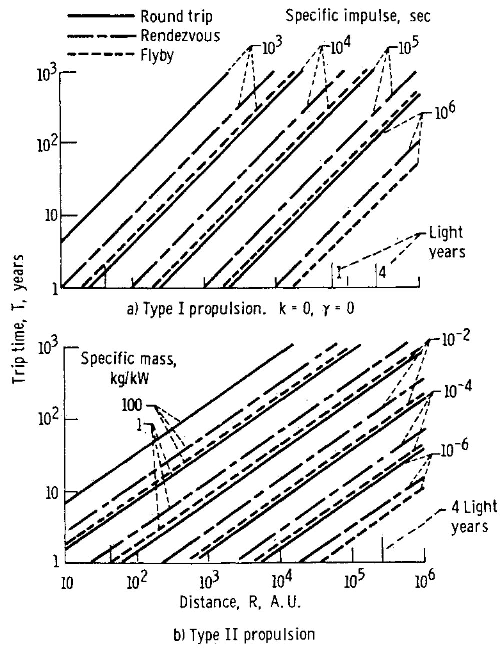

Fig. 4 Interstellar distance vs trip time in field-free space, $N = 4$.

Figures 4a and 4b show that flyby or rendezvous missions to the nearest star (4 light years) in trip times less than 20 yr will require specific impulses of the order of 10^6 sec for type-I systems, or specific masses of the order of 10^{-4} kg/kw for type-II systems. These parameters are obviously beyond those attainable with any known propulsion concept. The required optimum specific impulse for type-II systems is greater than 2×10^6 sec. The corresponding exhaust velocity is about one-tenth the speed of light.

Interplanetary Distances

Results from Eqs. (17a) and (18a) for interplanetary distances are shown in Figs. 5a and 5b.

The figures show that for single-stage, round-trip missions

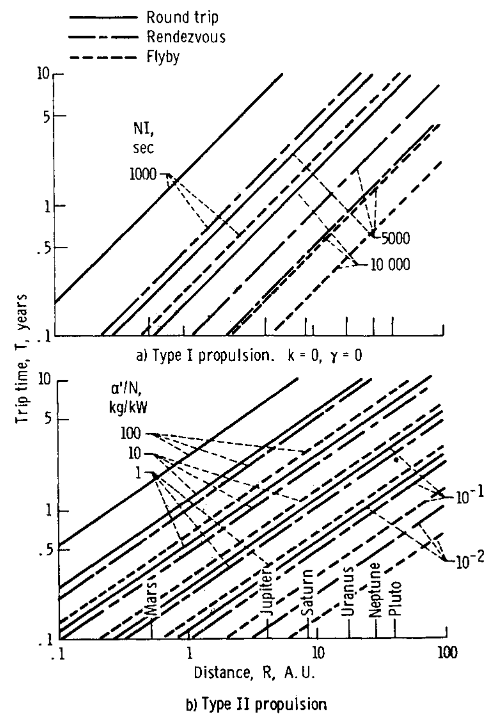

Fig. 5 Interplanetary distance vs trip time. Field-free space.

to the outer planets (Neptune, Pluto) in trip times less than 5 yr, a specific impulse greater than 10,000 sec is needed with type I systems, and a specific mass less than 1 kg/kw is needed for type II systems. Figure 1 shows no type-I system with this capability, but a type-II system may be able to do it. Flyby missions to Pluto in less than 5 yr require a specific impulse of about 1600 sec with type I, and a specific mass of about 30 kg/kw for type-II systems. As Fig. 1 shows, these performance values should be achievable with future systems of both types.

Comparison of Advanced Propulsion Systems

Figure 6 compares trip-time requirements for advanced type I and type II systems that have performance parameters in the orders of magnitude shown in Fig. 1. Round-trip missions are compared in Fig. 6a and one-way rendezvous missions in Fig. 6b. For the round-trip mission, a type-II system with the lowest specific mass shown in Fig. 1 (0.3 kg/kw) would produce lower trip time than a type-I system with the highest specific impulse ($I = 7000$ sec) for all planets beyond Jupiter. For the Jupiter trip, these systems yield about the same capability, while for the Mars trip the type-I system provides faster trip times. These capabilities agree qualitatively with those of Ref. 9, which contains comparisons of several nuclear fission rockets and a nuclear fusion rocket for several specific missions. For the farthest planets, even a relatively heavy type-II system ($\alpha' \approx 10$ kg/kw) is competitive with the most advanced type-I systems of Fig. 1 ($I \approx 7000$ sec). For the rendezvous mission (Fig. 6b), the crossover point (where the best type-II system of Fig. 1 is faster than the best type-I system) lies at a distance beyond Saturn.

Since the feasibility of these advanced systems remains to be demonstrated, one cannot tell which of the performance estimates shown in Fig. 1 is most likely to survive without major deterioration, or whether some unforeseen problem will arise which eliminates a system from the competition. It is obviously much too soon to try to decide which of the major advanced propulsion concepts represented in Fig. 1 will turn out to be most useful. One can only conclude that they are competitive for planetary missions.

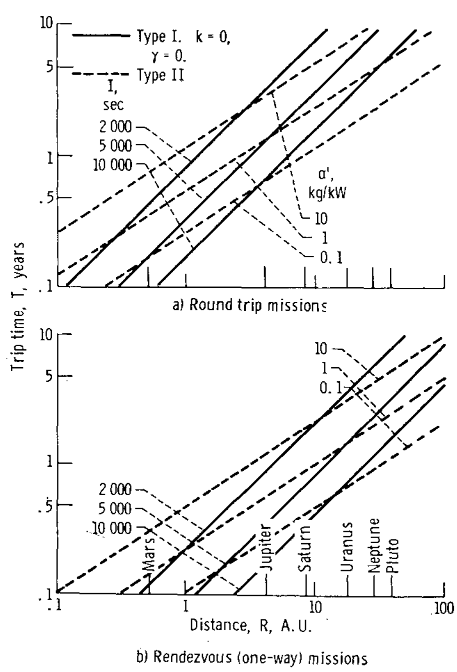

Fig. 6 Comparison of type I and type II propulsion for planetary distances, $N = 1$.

Validity of Gravity-Free Approximation

One might expect that gravity-free mission calculations would agree closely with those calculated with gravity fields included when the kinetic energy needed for the mission becomes much greater than the maximum gravitational potential energy involved. This potential energy (per unit mass) in any gravity field is equal to $-GM/r = -v_c^2$ where v_c is the circular orbital velocity at radius r from the center of the gravitational mass M. The condition for validity of the field-free approximation should therefore be

$$\Delta v / v_c \gg 1 \qquad (20)$$

where v_c is the maximum circular velocity appropriate to the mission. The values of Δv to be compared with v_c are given by Eqs. (6) and (15) for type I and type II systems, respectively. For $k = \gamma = 0$ and $m_{0,n+1}/m_{0,n} = 0.1$, these equations become

$$\text{Type I:} \qquad \Delta v = 22.6\, NI \text{ m/sec} \qquad (21)$$

and

$$\text{Type II:} \qquad \Delta v = 1.4 \times 10^5 \left(\frac{NT_{yr}}{\alpha'}\right)^{1/2} \qquad (22)$$

If $(\Delta v / v_c) > 10$ is used as the criterion for negligibility of gravity fields, then these equations yield

$$NI \gtrsim 0.44 v_c \qquad (23)$$

and

$$\alpha'/N \lesssim (14 \times 10^3 / v_c)^2\, T_{yr} \qquad (24)$$

where v_c is in meters per second.

Alternatively, if Δv is expressed in terms of the field-free mission distance and time (equations in Figs. 2 and 3) these conditions become

$$\begin{aligned} &\text{Flyby:} && R/T \gtrsim 10 f v_c \\ &\text{Rendezvous:} && R/T \gtrsim 5 f v_c \\ &\text{Round trip:} && R/T \gtrsim 1.25 f v_c \end{aligned} \qquad (25)$$

where $f = 1.0$ for type I systems and $f = \frac{2}{3}$ for type II systems. Some pertinent values of v_c are shown in Table 1.

As might be expected, the conditions for neglibility of solar and planetary gravity fields are amply satisfied for the range of parameters required for interstellar distances (Fig. 4). However, for missions within the solar system, and for the propulsion concepts shown in Fig. 1, these conditions are not necessarily satisfied. For example, for a type-I propulsion system with $NI \approx 5000$, condition (23) requires $v_c \lesssim 10$ km/sec. For a type-II system with $\alpha'/N = 1$ kg/kw, condition (24) requires $v_c \lesssim 14(T_{yr})^{1/2}$ km/sec. Table 1 therefore indicates that one might expect substantial errors in gravity-free calculations for missions that require departure from or capture into low orbits about the major planets (Jupiter to Neptune) and perhaps for any missions within the orbit of Jupiter. The magnitude of these errors was evaluated by comparing the gravity-free calculations with results of more accurate mission analysis in Refs. 1 and 10–13. These comparisons showed that interplanetary missions within the orbit of Jupiter,

Table 1 Typical circular velocities in solar system

Planet	Circular velocity at 1.1 times planet's radius		Orbital velocity around Sun at planet's distance	
	km/sec	a.u./yr	km/sec	a.u./yr
Earth	7.5	1.6	29.8	6.3
Mars	3.5	0.74	24.1	5.1
Jupiter	41.2	8.7	13.1	2.8
Saturn	25.5	5.4	9.7	2.0
Uranus	15.6	3.3	6.8	1.4
Neptune	14.0	3.0	5.4	1.1
Pluto	7.4	1.6	4.7	1.0

as well as those outside the orbit, are estimated with good accuracy by the curves of Figs. 4–6. However, escape from or descent into deep gravitational wells, as expected, produces substantial differences. As an example, for a type-II system with $\alpha' = 1$ kg/kw, $T \approx 0.5$ yr, and $N = 1$, the lowest allowable Jupiter orbit is about 16 Jupiter radii if about 10% accuracy in the field-free estimate is desired [Eq. (24) and Table 1]. If the vehicle descends to 1.1 Jupiter radii, the trip time is increased by about 50%. Escape from low Earth or Venus orbit (1.1 R), however, only increases trip time by about 5% for this specific mass. These increases are greater for higher values of α'. The general conclusion is that for high-specific impulse type I systems and the low specific mass type II systems of Fig. 1, the gravity-free results are good approximations for all solar-system missions that do not involve escape from or descend to low orbit about the major planets.

Effect of Variations in k, γ, and Payload Ratio

For type-I missions, the effect of variations in k, γ, and assumed payload ratio are all contained in the logarithmic term of Eq. (17). For type-II missions, since γ is optimized and k is assumed negligible, only the effect of payload ratio appears in the mission equation [Eq. (18)]. The effects on mission distance for a given trip time obtained from these equations are shown in Fig. 7, where R_0 is the distances for $k = \gamma = 0$ with type I and for payload ratio per stage of 0.1 for both types. Mission capability is seen to be quite sensitive to values of tankage and propulsion system mass ratio for type-I systems. The trip times plotted for $k = \gamma = 0$ on previous figures are therefore somewhat optimistic, particularly for relatively heavy propulsion systems such as gaseous-core fission rockets, where minimum values of m_{ps} in the range of 10^5 kg or more are estimated for the specific impulse range near and above 5000 sec (Ref. 2).

Conclusions

Comparison of missions in gravity-free space with more realistic mission studies shows that the gravity-free approximation provides good estimates of trip time as function of propulsion system parameters, not only for interstellar distances but also for planetary missions. The agreement, however, is limited to these missions that do not require escape from or descent to low orbits about the major planets. The field-free equations are therefore useful for preliminary evaluation of the mission capability of advanced propulsion system concepts.

Comparisons show that, among the most advanced concepts for which performance estimates are available, the pulsed (microbomb) fusion rocket and the radiation-cooled gaseous-core fission rockets provide the fastest trip times to the near planets. Of these two systems, the pulsed fusion system, if its estimated higher thrust to weight ratio prevails would be superior. For round trips to Jupiter and beyond, the controlled-fusion system shows progressively greater trip time advantages. For mission involving the establishment of low orbits about major planets, this advantage exists only for more distant planets. Because of the very large uncertainties in current performance estimates the only conclusion warranted at present is that these advanced concepts are competitive for future planetary missions. Concepts based on the use of impinging laser beams (remotely generated) are limited to flyby propulsion, and require vast amounts of power for modest payload. They do not, therefore, seem competitive with the advanced onboard propulsion concepts for the range of missions considered.

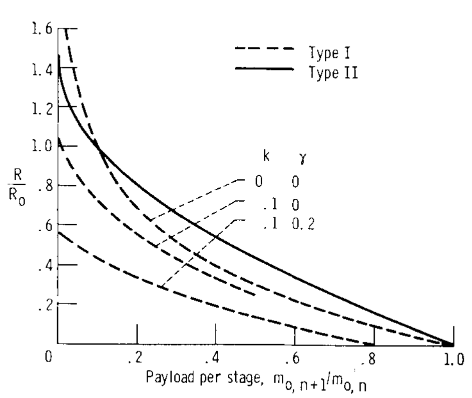

Fig. 7 Effect of payload ratio, tankage, and propulsion system mass on mission distance. Field-free space.

References

[1] Moeckel, W. E., "Propulsion Systems for Manned Exploration of the Solar System," *Astronautics and Aeronautics*, Vol. 7, No. 8, Aug. 1969, pp. 66–67; also TM X-1864, 1969, NASA.

[2] Ragsdale, R. G., "Status of Open-Cycle Gas-Core Reactor Project through 1970," TM X-2259, 1971, NASA.

[3] Reinmann, J. J., "Fusion Rocket Concepts," TM X-67826, 1971, NASA.

[4] Reinmann, J. J. and Rayle, W. D., "Deuterium-Helium-3 Fusion Power Balance Calculations," TM X-2280, 1971, NASA.

[5] Boyer, K. and Balcomb, J. D., "Systems Studies of Fusion Powered Pulsed Propulsion Systems," AIAA Paper 71-636, Salt Lake City, Utah, 1971.

[6] Rom, F. E. and Putre, H., "Laser Propulsion," TM X-2510, 1972, NASA.

[7] Marx, G., "Interstellar Vehicle Propelled by Terrestrial Laser Beam," *Nature*, Vol. 211, July 2, 1966, pp. 22–23; also *Nature*, Feb. 11, 1967, pp. 588–589.

[8] Moeckel, W. E., "Propulsion by Impinging Laser Beams," *Journal of Spacecraft and Rockets*, Vol. 9, No. 12, Dec. 1972, pp. 942–944.

[9] Fishbach, L. H. and Willis, E. A., Jr., "Performance Potential of Gas-Core and Fusion Rockets: A Mission Applications Survey," TM X-67940, 1971, NASA.

[10] Sauer, C. G., Jr. and Melbourne, W. G., "Optimum Earth-to-Mars Roundtrip Trajectories Utilizing a Low-Thrust Power-Limited Propulsion System," *Advances in the Astronautical Sciences*, Vol. 13, edited by E. Burgess, Western Periodicals, North Hollywood, Calif., 1963, pp. 547–570.

[11] Dugan, J. F., Jr., "Analysis of Trajectory Parameters for Probe and Round-Trip Missions to Mars," TN D-281, 1960, NASA.

[12] Dallas, S. S., "High-Energy Trajectories from Earth to Mars and Return," JPL-TR-32-803, NASA CR-69221, Dec. 1965, Jet Propulsion Lab., Pasadena, Calif.

[13] Fishbach, L. H., Giventer, L. L., and Willis, E. A., Jr., "Approximate Trajectory Data for Missions to the Major Planets," TN D-6141, 1971, NASA.