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PROPULSION SYSTEMS FOR MANNED EXPLORATION OF THE SOLAR SYSTEM

by W. E. Moeckel

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

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SUMMARY

What propulsion systems are in sight for fast interplanetary travel? Only a few show promise of reducing trip times to values comparable to those of 16th century terrestrial expeditions. The first portion of this report relates planetary round-trip times to the performance parameters of two types of propulsion systems: type I is specific-impulse limited (with high thrust), and type II is specific-mass limited (with low thrust). The second part of the report discusses advanced propulsion concepts of both types and evaluates their limitations. The discussion includes nuclear-fission rockets (solid, liquid, and gaseous core), nuclear-pulse propulsion, nuclear-electric rockets, and thermonuclear-fusion rockets. Particular attention is given to the last of these, because it is less familiar than the others. A general conclusion is that the more advanced systems, if they prove feasible, will reduce trip time to the near planets by factors of 3 to 5 and will make several outer planets accessible to manned exploration.

INTRODUCTION

Man seems bound by his nature to explore the universe to the limits of his capability. This is the age when those limits are expanding to include the planets of our solar system. To a much greater degree than in terrestrial explorations, propulsion capability will determine the extent and frequency of man's space excursions. For this reason, and despite the current low level of interest in funding manned planetary exploration,

*This report is an expanded version of an invited article for Astronautics and Aeronautics, August 1969.

research should continue on systems and concepts that will eventually make such journeys easier, faster, and cheaper than is possible with existing systems.

Propulsion, in common with other fields of technology, is participating in a rapid increase in capability. Less than 15 decades have elapsed since man ceased to rely entirely on muscle power and the natural movements of air and water for propulsion. During this brief period in human history, propulsion systems have helped to produce vast changes in human life. Railroads, automobiles, powered ships, aircraft, and space vehicles are all based on proliferation of propulsion systems. Although an exponential growth such as this must eventually saturate, no reduction in growth rate for propulsion is as yet in sight. In fact, the application of nuclear energy to propulsion promises to yield increments in capability that exceed those due to applications of chemical energy during the past century. Whether this potential will be fully realized may depend as much on human energy and persistence as on scientific or technical limitations.

Nuclear energy is already in use for ocean travel and may become useful for very large aircraft in the future. The main gains in capability, however, will result from its use for space propulsion. Studies have shown that both solid-core nuclear rockets and nuclear electric rockets, with performance characteristics that appear to be attainable in the next decade, can produce major gains over chemical rockets for manned trips to Mars or Venus. They can also increase the unmanned payloads that can be carried to more distant parts of the solar system. But even these high-performance systems are inadequate for manned missions beyond Mars, because the round-trip travel times are too long. Even for Mars and Venus missions, trip times would be comparable to those needed by sailing ships to circumnavigate the Earth in the 16th century. There are only a few propulsion concepts in sight that seem capable of substantially reducing these travel times. These concepts are still very nebulous with regard to technical feasibility and performance limitations, but if their anticipated capabilities can be realized, another major step in man's power to traverse space will follow. On the basis of past technological experience, such advances in propulsion can be expected to have profound effect not only on interplanetary travel but on near-Earth and Lunar travel as well.

This report examines, first, the relationship between propulsion system performance and planetary mission capability. Then the anticipated performance is discussed for a variety of advanced propulsion concepts, ranging from fairly well-defined systems such as solid-core nuclear-fission rockets to quite speculative systems such as the thermonuclear-fusion rocket. We omit, however, such concepts as mass-annihilation photon rockets, for which no conceptual basis exists to make performance estimates. Finally, the estimated performance capabilities of the propulsion systems are related to mission requirements to indicate how far and how fast humans may go in the foreseeable future.

WHAT PERFORMANCE IS NEEDED?

To establish a basis for evaluating propulsion system requirements, we estimate the time required to travel to the planets beyond Earth and return. For this purpose we define two types of propulsion systems. Type I consists of systems (such as chemical or nuclear-fission rockets) which are limited in specific impulse but can generate thrust comparable to their Earth weight. These systems will be denoted specific-impulse limited. They are also frequently called high-thrust systems. Type II consists of systems (such as electric rockets and perhaps nuclear-fusion rockets) which can produce very high specific impulses but are limited in thrust to values much less than their Earth weight. These systems are designated specific-mass limited, where specific mass is the ratio of the propulsion system mass to the jet power produced. They have also been called low-thrust systems and power-limited systems. The reason for defining these two types is that they propel space vehicles along two different classes of space trajectory.

To further simplify the mission analysis, we consider only the interplanetary portion of the trip. The initial mass of the vehicle is the mass that has been launched to escape velocity relative to the Earth. Starting at the Earth's orbit with the Earth's orbital velocity, the vehicle transfers to the orbit of the destination planet and matches the orbital velocity of that planet. No stopover time or descent to low orbit around the destination planet is considered. The return trip is symmetrical with the outward trip with respect to trip time and propulsion energy.

For both types of propulsion, two vehicles are considered: a four-stage vehicle with a ratio of returned payload to initial gross mass of 10^{-4} and a single-stage vehicle with a ratio of returned payload to initial gross mass of 10^{-1}. The 10^{-4} payload ratio seems near the practical lower limit. Thus, for a payload of 10^4 kilograms (which is certainly small for a manned planetary expedition), an initial mass of 10^8 kilograms would have to be launched to escape velocity. On the other hand, the single-stage vehicle payload ratio of 10^{-1} provides a reasonable initial mass.

Type I Propulsion

For specific-impulse limited systems, the propulsion system mass is, by definition, a small portion of the total vehicle mass. Most of the initial mass is propellant mass. The four thrust periods of the mission are assumed to be impulsive in nature, and staging takes place after each thrust period. For each stage, the mass of tankage, engine, and structure discarded after the thrust period is assumed to be a fraction k of the propellant mass. The ratio of empty to initial mass for the n^{th} stage is

$$\frac{m_{bn}}{m_{an}} = e^{-\Delta v_n/v_j} \tag{1}$$

where Δv_n is the velocity increment produced by the n^{th} stage, and v_j is the exhaust velocity of the propellant.

The initial mass of the next stage is

$$m_{a,n+1} = m_{bn} - k(m_{an} - m_{bn}) \tag{2}$$

and the ratio of this mass to the initial mass is (with eq. (1))

$$\frac{m_{a,n+1}}{m_{an}} = (1 + k)e^{-\Delta v_n/v_j} - k \tag{3}$$

This ratio is the payload ratio of the n^{th} stage. The assumption that Δv_n is the same for each stage is sufficiently accurate for the present estimates. Hence, the ratio of net payload m_L to initial mass m_{a1} for a four-stage vehicle is

$$\frac{m_L}{m_{a1}} = \left[(1 + k)e^{-\Delta v_n/v_j} - k\right]^4 \tag{4}$$

For the assumed payload ratio of 10^{-4}, and a value of k equal to 0.05, equation (4) yields:

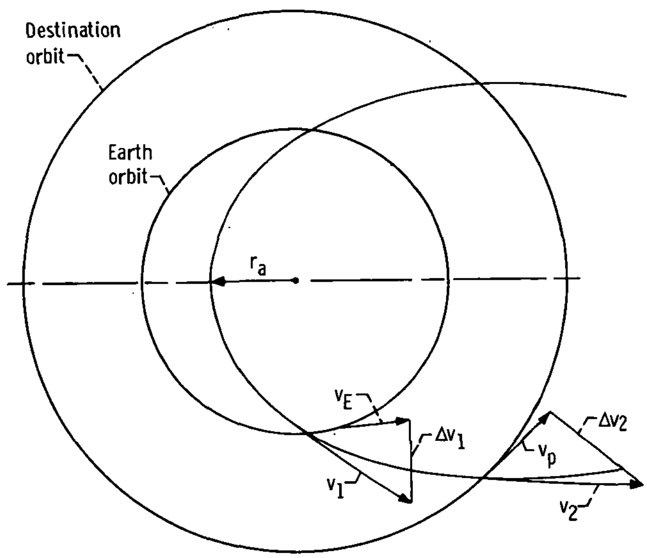
$$(\Delta v)_{tot} \equiv 4\Delta v_n = 7.8\ v_j \qquad \text{(m/sec)} \tag{5}$$

In terms of kilometers per second and specific impulse ($I = v_j/g_o$) equation (5) becomes

$$(\Delta v)_{tot} = 0.076\ I \qquad \text{(km/sec)} \tag{6}$$

Equation (6) gives the total velocity increment capabilities in terms of specific impulse for a four-stage vehicle with a net payload ratio of 10^{-4}. For the single-stage vehicle with payload ratio of 10^{-1}, Δv_1 is $(\Delta v)_{tot}$, so that the right side of equation (6) is simply divided by 4. Hence, the single-stage vehicle requires four times the specific impulse of the four-stage vehicle for the same trip.

The total velocity increment $(\Delta v)_{tot}$ required for transfer to each of the planets beyond Earth was evaluated as a function of trip time for symmetrical out-and-return trips, with no stopover and no consideration of planetary rendezvous. The resulting trip times are therefore minimum for reaching the planets and are only approximately achievable in practice. The total velocity increment consists of twice the sum of the hyperbolic excess velocity[1] at Earth and the hyperbolic excess velocity at the destination (see sketch (a)). This sum was minimized for each trip time by variation of the conic-section parameters of the trajectory (perihelion r_a and eccentricity).



$\Delta v_{tot} = 2(\Delta v_1 + \Delta v_2)$

(a)

The total trip time is twice the one-way time. From equation (6) and the calculated variation of $(\Delta v)_{tot}$ as a function of trip time, the time required for out-and-back trips to the planets was determined for a number of specific impulses. The results are plotted in figure 1.

The curves of distance as a function of time for a given specific impulse are straight lines within the accuracy of the optimization. As a matter of interest, this linear variation agrees with the equation for out-and-back travel to a distance R in gravity-free

[1]Defined as the velocity in excess of escape velocity from the planet, and hence also the difference between vehicle velocity and the orbital velocity of the planet.

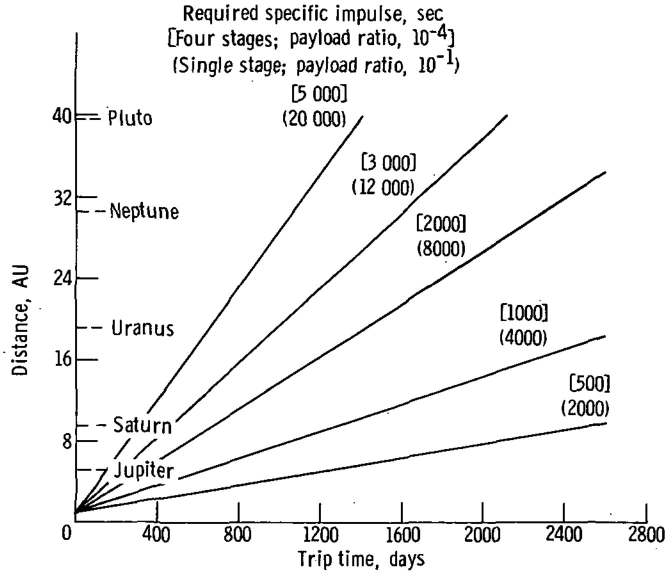


Figure 1. - Round-trip time to planets for type I propulsion systems.

space. For the same vehicle assumptions in field-free space, the distance-time equation is

$$R = \Delta v_n \frac{T_{tot}}{2} \qquad (m) \tag{7}$$

or

$$R = 5.5\times10^{-6}\, I\, T_{tot} \tag{8}$$

where R is in astronomical units and T_{tot} is in days. Equation (8) yields a longer trip time for a given distance than the solar-field calculations shown in figure 1. This is because the Earth's orbital motion provides an initial kinetic energy for the vehicle in the solar field case.

Type II Propulsion

For specific-mass limited systems, the propulsion system mass is a sizable fraction of the total vehicle mass, comparable to the propellant mass. The elimination of restriction on specific impulse permits optimization of the distribution between propellant and propulsion system mass. But the specific mass of presently conceived systems is so large that the acceleration attainable is quite low. Hence the propulsion periods are major portions of the transit time. Consequently, trajectories must be numerically integrated and are not portions of conic sections as with type I propulsion. Mass ratios for such systems are derived (in the manner introduced in ref. 1) from the differential form of Newton's law:

$$dm = -\frac{F dt}{v_j} = -\frac{m^2 a^2}{2P_j} dt \tag{9}$$

where F is thrust, m is the mass of the vehicle, a is F/m, P_j is the jet power in the expelled propellant (equal to $Fv_j/2$), and v_j is the jet exhaust velocity.

Integration of equation (9) yields

$$\frac{m_{an}}{m_{bn}} - 1 = \frac{m_{an}}{2P_j} \int_0^{T_{pn}} a^2 dt = \frac{m_{an}}{2P_j} J_n \tag{10}$$

where J_n (the time integral of acceleration squared) is the so-called mission difficulty parameter and T_{pn} is stage propulsion time. Let m_{sn} be the propulsion system mass of the n^{th} stage (including structure and other masses discarded at staging). Define the specific mass α as

$$\alpha = \frac{m_{sn}}{P_j} \qquad \text{(kg/W)} \tag{11}$$

and the propulsion system mass ratio γ as

$$\gamma = \frac{m_{sn}}{m_{an}} \tag{12}$$

Both α and γ are assumed to be the same for each of the four stages. Then equation (10) becomes

$$\frac{m_{bn}}{m_{an}} = \left(1 + \frac{\alpha J_n}{2\gamma}\right)^{-1} \tag{13}$$

and

$$\frac{m_{a,n+1}}{m_{a,n}} = \frac{m_{bn} - m_{sn}}{m_{an}} = \left(1 + \frac{\alpha J_n}{2\gamma}\right)^{-1} - \gamma \tag{14}$$

Optimization of each stage payload ratio $m_{a,n+1}/m_{a,n}$ with respect to γ yields

$$\gamma_{opt} = \sqrt{\frac{\alpha J_n}{2}}\left(1 - \sqrt{\frac{\alpha J_n}{2}}\right) \tag{15}$$

(A discussion of this and other optimizations for low-thrust propulsion is given in ref. 2.) Substitution of this optimum value into equation (14) yields

$$\frac{m_{a,n+1}}{m_{a,n}} = \left(1 - \sqrt{\frac{\alpha J_n}{2}}\right)^2 \tag{16}$$

The net payload ratio for the four-stage vehicle is therefore

$$\frac{m_L}{m_{a1}} = \left(1 - \sqrt{\frac{\alpha J_n}{2}}\right)^8 \tag{17}$$

where J_n is assumed to be equal for each stage. For $m_L/m_{a1} = 10^{-4}$, equation (17) yields

$$J_{tot} \equiv 4J_n = \frac{3.74}{\alpha} \qquad m^2/sec \tag{18}$$

Equation (18) expresses the required specific mass α of the propulsion system in terms of the overall mission difficulty parameter J_{tot}.

The mission difficulty parameter J for a given mission (eq. (10)) is generally lower if variations in acceleration (and therefore thrust and jet velocity) are permitted during the thrusting period. However, the gains in J over use of constant thrust and specific impulse are usually of the order of 10 percent. The magnitude of specific impulse and thrust required for a mission can therefore be estimated by assuming that these quantities are constant during each thrust period. For this assumption, the propellant mass ratio per stage becomes

$$\frac{m_{pn}}{m_{an}} = \frac{FT_{pn}}{m_{an}v_j} = \frac{2P_{jn}T_{pn}}{m_{an}v_j^2} = \frac{2\gamma}{\alpha}\frac{T_{pn}}{v_j^2} \tag{19}$$

From the relation $m_{pn}/m_{an} = 1 - (m_{bn}/m_{an})$, and using equations (13), (15), and (18), equation (19) yields the following expressions for the optimum jet velocity:

$$v_{j,opt} = \sqrt{0.632 \frac{T_{pn}}{\alpha}} \qquad (m/sec) \tag{20}$$

where $T_{pn} = T_{tot}/4$ for this mission (continuous propulsion). The optimum specific impulse is therefore

$$I_{opt} = 377\sqrt{\frac{T_{tot}}{\alpha}} \tag{21}$$

where T_{tot} is in days and α' is in kilograms per kilowatt.

Equations (10) to (18) can be used directly for the single-stage mission by letting n equal 1, and using J_{tot} in place of J_n in all equations. The ratio m_{a2}/m_{a1} (eq. (16)) is then the payload ratio for the mission. Using the value 0.1 for this ratio, equation (16) yields

$$J_{tot} = \frac{0.936}{\alpha} \qquad (m^2/sec) \tag{18a}$$

which is just the single-stage value from equation (18). This shows that for a given mission difficulty, J_{tot}, the propulsion system specific mass required with the single-stage vehicle, is one-fourth the value required with the four-stage vehicle. Equations (20) and (21) become

$$v_{j,opt} = \sqrt{0.632 \frac{T_{p,tot}}{\alpha}} \tag{20a}$$

and since $T_{p,tot} = T_{tot}$,

$$I_{opt} = 754\sqrt{\frac{T_{tot}}{\alpha'}} \tag{21a}$$

where T_{tot} is in days and α' in kilograms per kilowatt.

Values of J for orbit-to-orbit transfers from Earth to all planets are contained in reference 3 for a large range of trip times and trajectory parameters. These trajectories start with the Earth's orbital velocity and end with the planet's orbital velocity. The values of J for these trajectories thus correspond to $2J_n$ in equation (18). Using the trajectory with the lowest value of J from reference 3 for each trip time yielded curves of J_{tot} as function of total trip time. Equations (18) and (18a) then yielded the values of specific mass required as a function of trip time. The results are shown in figure 2 for both the four-stage and the single-stage vehicle.

For comparison purposes, the gravity-free equivalent mission for this case is illus-

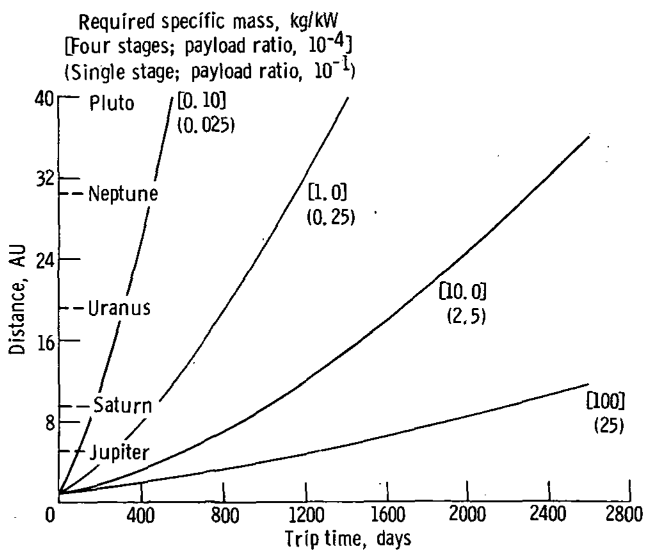
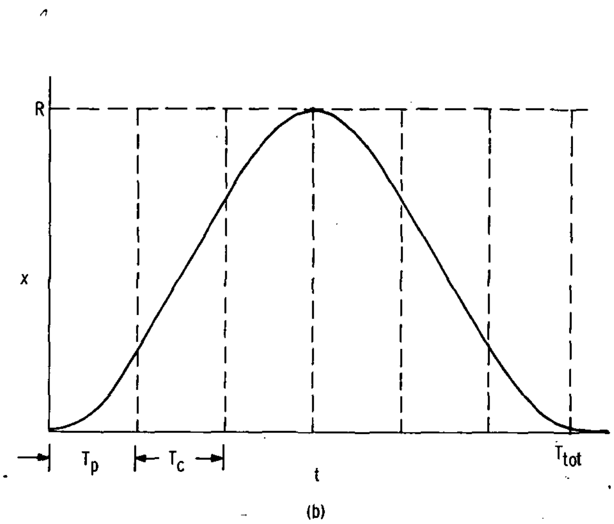


Figure 2. - Round-trip time to planets for type II propulsion systems.



(b)

trated in sketch (b). The mission consists of four constant-acceleration propulsion periods of duration T_p and two coast periods of duration T_c. The distance traveled is given by

$$R = a_0 T_p (T_c + T_p) \tag{22}$$

and the mission-difficulty parameter per stage is

$$J_n = a_0^2 T_p \tag{23}$$

Since

$$T_{tot} = 4T_p + 2T_c$$

equation (22) can also be written

$$R = a_0 T_p \left(\frac{1}{2} T_{tot} - T_p \right) \tag{24}$$

For the payload ratio of 10^{-4}, $J_n = 0.936/\alpha$ (eq. (18)). Hence

$$a_0 = \sqrt{\frac{0.936}{\alpha T_p}} \tag{25}$$

so that equation (24) becomes

$$R = \sqrt{0.936 \frac{T_p}{\alpha}} \left(\frac{1}{2} T_{tot} - T_p \right) \tag{26}$$

Equation (26) yields an optimum value of propulsion period T_p of (ref. 4)

$$T_{p,opt} = \frac{1}{6} T_{tot}$$

For this optimum value, equation (26) becomes

$$R = 4.16 \frac{T_{tot}^{3/2}}{\sqrt{\alpha'}} \tag{27}$$

or, with R in astronomical units, T_{tot} in days, and α' in kilograms per kilowatt,

$$R = 7\times10^{-4} \frac{T_{tot}^{3/2}}{\sqrt{\alpha'}} \qquad \text{(A.U.)} \tag{28}$$

As in the type I calculation, the gravity-free trip time for a given range R is greater than for the solar-field calculation. But the curves of figure 2 follow very well the $T_{tot}^{3/2}$ and $\sqrt{\alpha'}$ variation of the gravity-free equation (28) if the Earth's orbit ($R = 1$ in fig. 2) is taken as the $R = 0$ of equation (28).

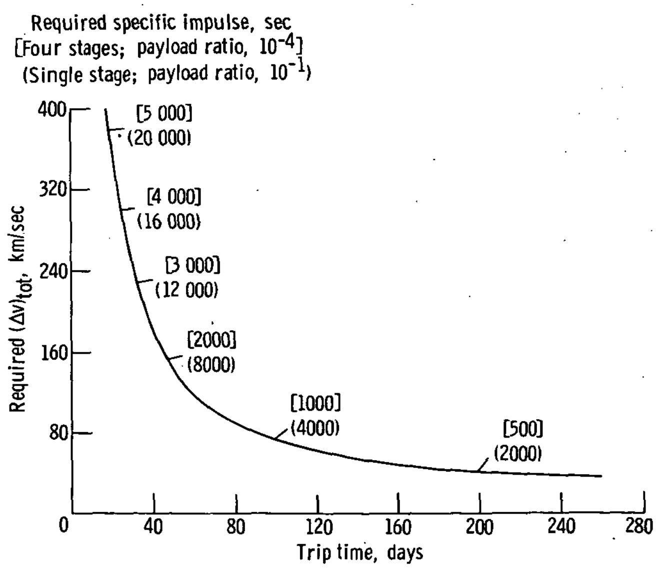
This $T^{3/2}$ variation of distance with time implies that type II systems become increasingly effective relative to type I systems as travel distance increases. This can be illustrated by comparing the requirements for the two types of propulsion systems for Jupiter and Neptune trips in figures 1 and 2. For the four-stage vehicles, a type II system with α' = 10.0 kilograms per kilowatt produces a trip time of about 600 days for the Jupiter trip and 2300 days for the Neptune trip. A type I system can produce a comparable trip time for the Jupiter mission with a specific impulse of 1000 seconds, but requires about 2000 seconds to match the Neptune trip time.

Mars "Quick-Trip" Requirements

Exploration of Jupiter or Neptune and their satellites is not very high on the list of national priorities at present. But most people agree that manned trips to Mars and Venus should take place in the not-too-distant future. For such trips, a reduction in trip time from the 400-day level now contemplated would make the mission simpler and more attractive.

For these relatively near planets, planetary motion and rendezvous requirements are important in determining the mission difficulty parameters $(\Delta v)_{tot}$ and J_{tot} for a given trip time. These parameters might be expected to be much larger than the simple out-and-back values, but the use of indirect trajectories (passing inside Earth orbit for a Mars trip, for example) can reduce them quite substantially. Consequently, the results of the previous section are not bad approximations to values obtained with more sophisticated mission studies.

Figure 3(a) shows the variation with round-trip time of $(\Delta v)_{tot}$ and required specific impulse from equation (6) for type I propulsion for the direct out-and-back flights. These values agreed well with calculations in references 5 and 6, wherein round trips with



(a) Type I propulsion.

Figure 3. - Propulsion requirements for Mars "quick trips."

rendezvous and trajectory optimization were studied. Results in references 5 and 6 were available down to 100 days and 67 days trip time, respectively.

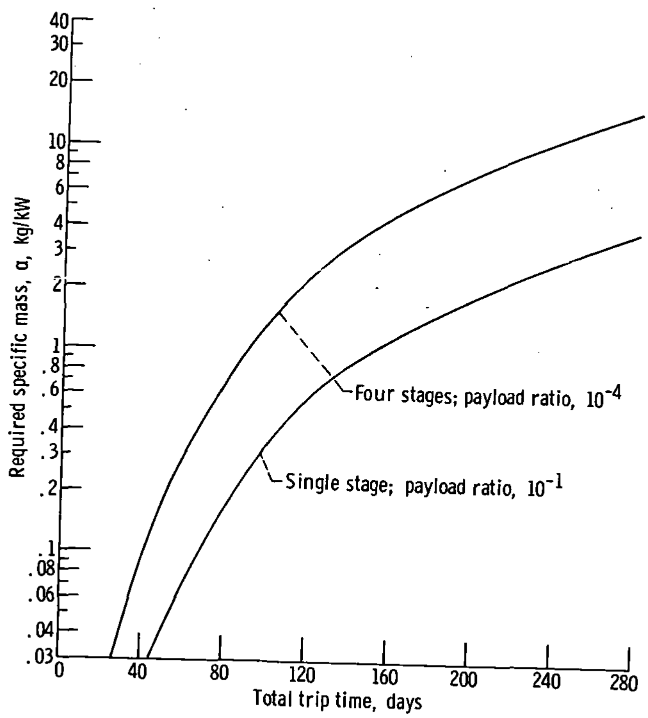
As the mission time is reduced to very small values, the direct out-and-back trajectories eventually become fast enough to keep up with the angular velocity of Earth and are therefore valid planetary trajectories if the stay-over time at Mars is short. This Earth-rendezvous capability for out-and-back trips occurs in figure 3(a) for total trip times less than 30 days. The fact that the simple out-and-back results are valid rendezvous missions for the short trip times and agree with the Δv values of references 5 and 6 for the longer trip times in figure 3(a) indicates that the curve of figure 3(a) is a reasonably good estimate.

For type II systems, effects of planetary motion and rendezvous requirements can be evaluated directly from the data of reference 3, because the angular distance traveled is presented as one of the trajectory parameters. For symmetrical direct out-and-back trips, the condition that the Earth can be overtaken by the spaceship is

$$\theta_{T,1} > \dot{\theta}_E T_1$$

where $\theta_{T,1}$ is the polar angle covered during the transfer time T_1 from Earth orbit to destination orbit and $\dot{\theta}_E$ is the angular velocity of the Earth (0.985°/day).

Figure 3(b) shows the values of α required when planetary motion is considered.



(b) Type II propulsion systems.

Figure 3. - Concluded.

These values are somewhat lower than those for simple out-and-back trips. However, the difference can be reduced to some extent by use of unsymmetrical and indirect trips, so again the simple out-and-back mission are a good approximation to actual requirements.

Some Conclusions on Performance Needs

A useful long-range planning approach is to enumerate things that should be done and then see what performance is needed to do them. If that performance is beyond anything foreseeable, then the desired goals must be scaled down accordingly. For planning solar-system exploration, a seemingly reasonable goal might be to try to land a team of scientists on each of the planets (or a moon thereof) during the next few decades. Since scientists are generally not keen on hardships and deprivation, these voyages should include adequate accommodations, and should not require more than a few years out of their lives (say 3 at most).

To achieve this trip time for Jupiter, figure 1 shows that a type I system would require a specific impulse between 500 seconds for the four-stage low-payload vehicle and 2000 seconds for the single-stage, high-payload vehicle. For a 3-year Pluto trip, these values are 5000 to 20 000 seconds. A single-stage type II system would require (fig. 2) a specific mass of about 20 kilograms per kilowatt for the Jupiter trip and about 0.2 kilogram per kilowatt for the Pluto trip. As shown in the next section, these performance requirements, for both propulsion types, range from values that should be attainable with systems now under development to values that are beyond the foreseeable capabilities of any known propulsion concept.

Another desirable goal is to establish scientific bases on Mars (and possibly Venus, if the environment permits), so that thorough studies could be made of our neighboring planets. The maintenance and personnel rotation requirements for such a base would make trip times of less than a month extremely desirable. Figure 3 shows that this trip time would require specific impulses between 3000 and 12 000 seconds for a type I system, and specific masses less than 0.1 kilogram per kilowatt for type II systems. Again these figures are beyond the anticipated capabilities of any known propulsion concept. If we relax the trip-time requirement to a year, some of the type I and type II systems now under development may do the job adequately. But this would reduce greatly the attractiveness and usefulness of such a base, and perhaps would rule out entirely the idea of establishing one. Intermediate trip times, however, should be attainable with future systems.

Figures 1 to 3 give a good idea of the relationship between propulsion system capability and mission difficulty. The next section considers the propulsion concepts that look promising, the performance parameters that seem achievable, and the nature of the limitations on these parameters.

WHAT PERFORMANCE IS ATTAINABLE?

The major space propulsion concepts that can presently be defined well enough in principle to make some performance estimates can all be considered to belong within either type I or type II, depending on whether the specific impulse limitation or the specific mass limitation is more serious.

Type I Systems

For primary propulsion (as distinguished from orientation or orbit control) chemical rockets are currently our only operating space propulsion systems. They clearly belong to type I, and their specific impulse is limited by chemical energy per unit mass to less than 500 seconds. Some free-radical systems with higher theoretical specific impulses have been proposed, but no workable methods for making and storing such propellants have materialized. Because of their low specific impulse limit, chemical rockets are not suitable for the class of missions considered in this report.

The anticipated capabilities of nuclear fission rockets have recently been reviewed in reference 7. Performance limits will therefore be only briefly summarized herein.

For solid-core nuclear-fission rockets, such as those now under development (NERVA program) specific impulses higher than about 850 to 900 seconds are not likely, due to core temperature and heat transfer limitations. If hydrogen could be heated to the melting point of certain refractory materials (4000 K), a specific impulse of about 1200 seconds would result, but 900 seconds is a practical limit. This is the only type I system other than the chemical rocket which has a demonstrated technical feasibility. A rocket with a thrust of 330 000 newtons (75 000 lb) at a thermal power of 1500 megawatts is now under development. A specific impulse of 825 seconds and thrust-to-weight ratio of 10 to 20 are anticipated for flight versions.

Liquid-core and gaseous-core nuclear-fission rockets have been studied analytically, with some experimental evaluation of critical processes. For liquid-core rockets, specific-impulse limits of 1300 to 1500 seconds are calculated with thrust-to-weight ratios of 2 to 10. The specific impulse is limited by increasing vaporization of the fuel with increasing temperature and the entrainment of the fuel by the hydrogen propellant. The entrainment increases the mean molecular weight of the propellant so that further increases in temperature produce little increase in specific impulse. The large fuel loss rates with liquid-core systems would be a major economic problem for large rockets, and there is currently little interest in further studies of this system.

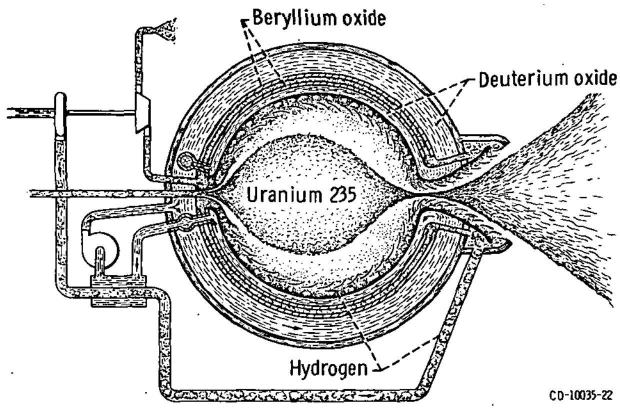


Figure 4. - Coaxial-flow gaseous-core nuclear rocket (ref. 7).

For gaseous-core nuclear-fission rockets, specific impulses in the range of 1500 to 2500 seconds seem to be theoretically feasible, with engine thrust-to-weight ratios of 1 or more. These figures are estimated in reference 7 for a coaxial flow system shown in figure 4. The limitation on specific impulse is in this case due to the need to transfer heat from the gaseous nuclear fuel to the hydrogen propellant by radiation. As for the liquid-core rockets, direct heating (i.e., passing the propellant through the fuel) is not feasible because of both the excessive fuel loss and the difficulty of achieving high specific impulse when heavy fuel particles become a substantial part of the propellant. The latter difficulty is illustrated by the equation for ideal (vacuum) specific impulse for a gaseous propellant. From the energy equation

$$\frac{1}{2}\,\overline{m}v_j^2 = \left(1 + \frac{f}{2}\right)kT$$

we obtain

$$I = \frac{1}{g}\left[\frac{(2 + f)kT}{\overline{m}}\right]^{1/2}$$

where f is the number of degrees of freedom of the propellant particles, k is Boltzmann's constant, T is the initial gas temperature, and $\overline{m}$ is the mean particle mass. For a mixture of atomic hydrogen and a mass fraction r of atomic uranium

$$\overline{m} = \frac{m_H m_u}{(1 - r)m_u + rm_H} \approx \frac{m_H}{1 - r}$$

and I becomes (for $f = 3$)

$$I \approx 20.6 \sqrt{(1 - r)T} \tag{29}$$

Equation (29) shows that the temperature required for a given specific impulse varies inversely with the concentration of hydrogen (1 - r) in the propellant.

For radiative heating, the upper limit on temperature is determined by the absorptivity of hydrogen and the heating of the chamber walls. The absorptivity drops rapidly beyond 60 000 K due to ionization. The need to keep heat transfer to the walls at a reasonable value (less than about 1 kW/cm^2) means that the average temperature of the propellant must be much less than the maximum attained near the fuel core. Figure 5 (from ref. 7) shows a calculated temperature distribution for a coaxial flow concept. Although fuel losses of the order of 1 to 10 percent of the propellant mass flow are anticipated, the prospect of attaining specific impulses of the order of 2000 with high thrust-weight ratio makes further research on gas-core reactors desirable. Reference 8 describes encouraging results of recent experiments to reduce mixing between coaxial streams.

A serious research and development difficulty for gaseous-core reactors lies in the fact that the minimum critical size is quite large at tolerable gas pressures. Reference 7 quotes a minimum expected thrust level of 10^6 newtons at a pressure of 1000 at-

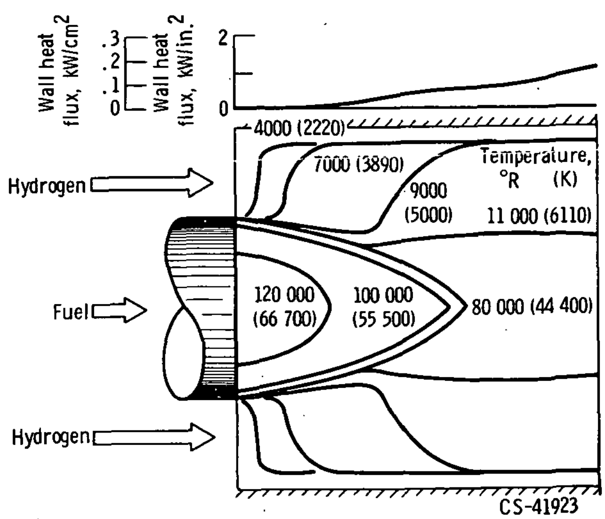


Figure 5. - Temperature field in coaxial-flow engine (ref. 7).

mospheres. This corresponds to a minimum reactor mass of over 45 000 kilograms. This is not too large for the first planetary-transfer stage of a major manned interplanetary mission, but problems of contamination with radioactive fission products in the exhaust jet would make a ground test program more difficult than that of the solid-core nuclear rocket. A thorough exhaust-gas cleaning system would be needed.

An alternative to the coaxial flow concept of a gas-core fission reactor is the so-called "light-bulb" concept studies by United Aircraft Corporation (ref. 9). In this concept the fuel is contained in a thin shell whose walls are as transparent as possible to radiative energy. This concept avoids fuel loss and consequent economic and test problems. Here the fragility of the shell with the necessary thinness to avoid melting is a major limitation. The minimum size is, of course, the same as for the coaxial flow concept.

One of the first gas-core reactor concepts proposed was to heat the hydrogen propellant by diffusion through the hot fissioning fuel and to contain the fuel with centrifugal force. Various vortex arrangements were studied both experimentally and theoretically (ref. 10). The net result of these studies was that mixing between propellant and fuel, and the consequent fuel loss rate, would probably be excessive for specific impulse above 1500 seconds and for reasonable reactor size.

Magnetic containment or stabilization of the uranium fuel (which ionizes at a lower temperature than hydrogen) has also been proposed. But the conductivity of uranium plasma at the temperatures suitable for a gaseous-core fission rocket is low relative to, for example, a controlled thermonuclear-fusion plasma. This means that the interaction forces between magnetic field and plasma would be smaller. However, unlike the fusion case, the presence of the hydrogen around the uranium core produces a gas pressure containment, so that the problem is one of stabilizing the interface between hydrogen and uranium rather than magnetic containment of a plasma in a surrounding vacuum. In any event, possible benefits resulting from the presence of a magnetic field would tend to increase as the attainable strength of the field increases.

The need for large-volume, high-intensity magnetic fields is common to several advanced propulsion and power concepts. For space use, these fields must be generated with superconducting materials or with cryogenically cooled conductors. Otherwise the mass and power requirements would make space application questionable. During the past decade, substantial improvements have been made in the volume and intensity of fields achieved (ref. 11). With cryogenically cooled magnets, fields of 20 teslas (200 000 G) have been developed in magnets with an 11-centimeter bore and a 50-centimeter length. With superconducting magnets, a field of 14 teslas was achieved with a 15-centimeter bore and 29-centimeter length. But further substantial reductions in mass are needed. Also, an increase in the critical temperature of superconductors is desirable to reduce the magnitude of the cooling problem.

Stabilization of the interface with radio frequency power may also be a possibility. This is suggested by one of the experiments described in reference 8, where inductive heating of a heavy gas was used to simulate the fuel core of a reactor. The mixing between the heavy gas and the surrounding gas appeared much reduced when the radiofrequency power was applied. If the frequency is properly selected, a radiofrequency magnetic field can produce an inward pressure on an electrically conducting gas. Radio frequency containment was investigated to some extent in controlled fusion research, but the power requirements are excessive for containment with a surrounding vacuum. For stabilization of an interface, however, a weaker field may suffice. The magnitude of the required field and the resulting mass and power penalties can only be determined experimentally.

Another type I propulsion concept proposed and studied some years ago is the nuclear-pulse propulsion system (Project Orion, ref. 12). This system proposed to deploy and fire a succession of small nuclear bombs at the rear of the vehicle at intervals of about 1 second. The blasts would be absorbed by a thick mass (called the pusher) at the rear of the vehicle, and the impulses would be transmitted to the vehicle through a shock-absorbing system.

This concept was subjected to some derision at first, but engineering design studies, together with chemical-explosion simulations, indicated that the system might be feasible. The production of thrust by high-velocity impingement of propellant (the bomb material) instead of rearward ejection of propellant makes the concept unique among space propulsion systems. (Other impingement concepts, such as solar sailing, use externally produced particles or radiation.) The studies (summarized in ref. 12) indicated that ratios of thrust to Earth weight greater than 1 were feasible, with specific impulses up to 2500 seconds. The specific impulse was limited by the mean velocity of the impinging explosion-produced particles. If the explosions are too near, the ablation of the pusher mass and the shock of the explosions are excessive. Although the research program on this concept revealed no technical obstacle to the eventual achievement of the estimated performance, further development posed many economic and political problems. The Nuclear Test Ban Treaty could prevent development of such systems on Earth unless special dispensations were to be agreed upon by participating nations. Development of the system in space would, of course, be very costly and would probably also be politically undesirable, particularly if alternative concepts with comparable performance turn out to be feasible. No further research on nuclear-pulse propulsion is now under way.

Type II Propulsion Systems

Among the specific-mass limited systems, several types of electric rocket can be regarded as having demonstrated technical feasibility. The primary question is the magnitude of specific mass that can ultimately be achieved. The specific mass of electric rockets is determined primarily by the specific mass of the electric power generation system, although the power conditioning, thruster efficiency, and thruster mass are also significant contributions. The current status of electric thruster development is reviewed in references 13 and 14.

The power generation system closest to application for electric propulsion is the solar-cell array. Specific masses of 20 kilograms per kilowatt seem assured with current technology, and values as low as 10 kilograms per kilowatt (at 1 AU) may soon be possible (ref. 15). These are very attractive values for unmanned planetary exploration vehicles out to the orbit of Jupiter. They may also be attractive for manned missions in the near solar system, if the problems of deployment and the dynamics associated with thin flexible arrays of very large area (more than 2 acres/MW) can be solved.

For multimegawatt systems such as those required for manned planetary exploration, nuclear-fission reactors are considered to be the most promising power source. Possible power conversion systems include turboalternators, thermionic cells, and magnetohydrodynamic (MHD) generators. Of these, the turboalternator is the most highly developed and is suitable for use with reactors whose maximum temperature is 1500 K or less. Both thermionic and MHD systems become more attractive at higher reactor temperatures, while turboalternators become less attractive due to turbine stress and cooling problems. At a 1420 K reactor temperature, an estimate (ref. 16) of the specific mass of a complete nuclear turboalternator system at 1 megawatt electric power level is about 7.5 kilograms per kilowatt without reactor shielding. Of this mass, about 1.2 kilograms per kilowatt is reactor mass, 3.1 kilograms per kilowatt is radiator mass (radiator temperature of 1000 K), and 3.2 kilograms per kilowatt is turboalternator and miscellaneous mass. Another study of the vapor chamber radiator concept yielded radiator specific mass down to about 1 kilogram per kilowatt (ref. 17) for radiator temperature of 945 K.

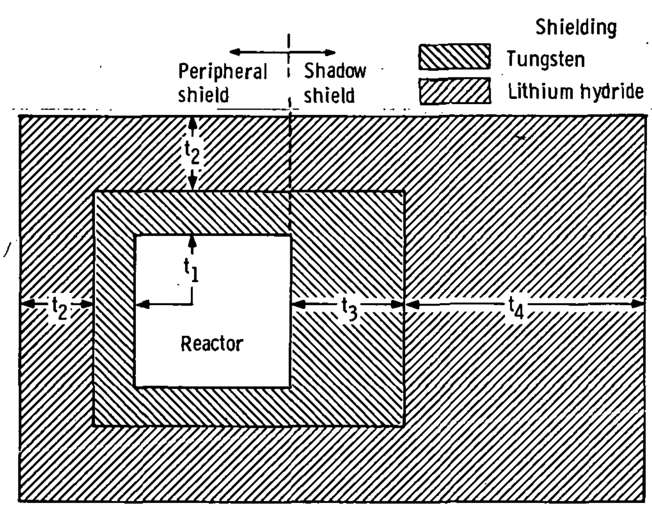
For a thermionic or MHD system at a reactor temperature of 2000 K, the specific mass of the radiator could be reduced from the above values to between 1.0 and 0.3 kilograms per kilowatt because radiator temperature could be about 1300 K. The radiator areas are not excessive at these temperatures. The formula for radiated power per unit area is

$$\frac{P_r}{A_r} = 5.67\times10^{-8}\,\epsilon T_r^4 \qquad (\mathrm{W/m^2}) \tag{30}$$

For an emissivity ϵ of 0.9 the radiator area is found to be 7 square meters per megawatt of radiated power at 1300 K. For an assumed conversion efficiency of 15 percent this area would be about 40 square meters per megawatt of electric power, which is a manageable value even for a 10-megawatt electric propulsion system.

The reactor and heat exchanger masses, with either thermionic systems or MHD systems, would be larger than for the turboalternator system because of additional volume needed for the cells, leads, and coolant passages in the thermionic and the need to heat a gas or vapor (rather than a liquid metal) in the MHD system (ref. 18). An unshielded reactor specific mass of 2 kilograms per kilowatt may be achievable for these systems (ref. 19). For the MHD systems, acceptable specific mass can be achieved only if superconducting electromagnets are used to produce the required high magnetic fields. With these superconducting windings, the total magnet system, including coils, support structure, and cooling system is a relatively small part (about 1 kg/kW) of the complete power system (ref. 18).

One of the largest contributions to the specific mass for all nuclear power generation systems is the shielding needed for manned missions. Because the shielding thickness tends to be independent of reactor power level, the specific mass of the shielding is particularly severe at low power levels. To see how the shielding mass varies with power level, consider the cylindrical reactor shown in sketch (c), where an inner



(c)

tungsten shield is used to reduce the dose from γ-radiation and an outer lithium hydride shield is used to attenuate the neutron dose. The shielding is divided into shadow shield and peripheral shield sections.

Approximate formulas for the shielding thickness as a function of dose rate are obtainable from calculated curves in reference 20.

Neutron dose

$$\log_{10} r_N = 3.9 - 0.05\, t_{LiH} \qquad \text{(rem/hr)} \tag{31}$$

γ-ray dose

$$\log_{10} r_\gamma = 3.5 - 0.01\, t_{LiH} - 0.35\, t_W \qquad \text{(rem/hr)} \tag{32}$$

where t_{LiH} and t_W are the thicknesses of LiH and tungsten in centimeters.

These simple formulas do not account for the secondary gamma radiation induced by the interaction of the neutrons with the shield material. This secondary radiation substantially increases the shielding thickness needed for a given dose rate. However, use of alternate layers of W and LiH (lamination of the shield) tends to reduce the required additional thickness (ref. 21). As a result, comparisons with more exact calculations for specific configurations indicate that the shield masses calculated by these formulas are reasonable estimates. The total shielding specific mass for the geometry of sketch (c) can be written

$$\alpha_s = \frac{m_s}{P_j} \tag{33}$$

where

$$m_s = \frac{\pi D^2 L}{4} \rho_W \left[\left(1 + \frac{2t_1}{D}\right)^2 \left(1 + \frac{t_1 + t_3}{L}\right) - 1\right]$$
$$+ \frac{\pi D^2 L}{4} \rho_{LiH} \left[\left(1 + \frac{2t_1 + 2t_2}{D}\right)^2 \left(1 + \frac{t_1 + t_2 + t_3 + t_4}{L}\right) - \left(1 + \frac{2t_1}{D}\right)^2 \left(1 + \frac{t_1 + t_2 + t_3}{L}\right)\right]$$

where ρ_W and ρ_{LiH} are 19.3 grams per cubic centimeter and 0.82 gram per cubic centimeter, respectively. Let P_T be the total thermal power of the reactor and p_T the thermal power density. Then the reactor diameter is obtained from

$$\frac{\pi D^3}{4}\left(\frac{L}{D}\right) = \frac{P_T}{p_T} \tag{34}$$

The reactor specific mass is

$$\alpha_R = \frac{\rho_R}{p_T \eta}$$

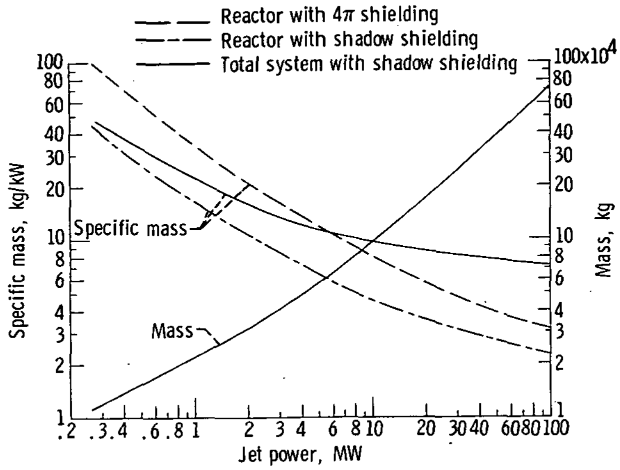


Figure 6. - Variation of mass with power for nuclear-electric propulsion systems. Reactor thermal power density, 100 watts per cubic centimeter; overall efficiency, 10 percent.

where ρ_R is the mean density of the reactor (assumed to be 10 g/cm^3) and η is overall conversion efficiency of thermal power to jet power. For a reactor L/D of 1.0 equations (31) to (34) were used to calculate shielding and reactor specific mass as functions of power level for several power densities and assumed radiation doses. Results are shown in figure 6 for a reactor thermal power density p_T of 100 watts per cubic centimeter and an overall efficiency η for conversion of thermal to jet power of 10 percent. One curve is for complete shielding (4π shielding) to a dose rate of 10^{-2} rem per hour, which for a 1-year trip without additional dose reductions due to distance or cabin shielding, yields a dose of about 100 rem. A second curve is for shadow shielding to 10^{-2} rem per hour and peripheral shielding to 1 rem per hour. The pair of solid curves is for complete system mass with shadow shielding for which 5 kilograms per kilowatt was added for all other components including thrusters and power conditioning.

These specific masses show that the shielding requirement for manned use is indeed a severe penalty at power levels up to 10 megawatts. Beyond this level, the shield and reactor specific mass becomes comparable with the sum of the other components of the system. From these calculations, the specific mass of the entire system can probably be assumed to approach a minimum of about 7 kilograms per kilowatt at very high power levels for advanced nuclear-electric systems, of which about 3 kilograms per kilowatt would be for reactor and shield.

The range of power levels of interest for manned planetary expeditions is dependent on the scale of the expedition we wish to imagine. The optimum propulsion system mass from equations (15) and (18a) is about 20 percent of the initial interplanetary vehicle mass. An initial payload of 50 000 kilograms (counting crew, supplies, environmental control, life support, and possibly a landing vehicle) is not excessive for such a mission. For a payload ratio of 0.1 (single stage) this implies an initial vehicle mass 5×10^5 kilograms and a propulsion system mass of 10^5 kilograms. Using the total-system curves of figure 6 for shadow shielding yields a power level of the order of 10 megawatts and a total specific mass of about 10 kilograms per kilowatt. A larger vehicle would have an advantage in lower specific mass (down to 7 kg/kW) and thus lower trip time for a given payload ratio. These specific masses seem to be the lowest that can reasonably be anticipated for electric propulsion systems.

Another major type II propulsion concept, one that is much more nebulous than electric propulsion with regard to technical feasibility, is the thermonuclear-fusion rocket. This concept will be discussed at greater length than the others in this report because it is relatively unfamiliar. A few studies have been made of the specific mass that might be achievable when and if controlled thermonuclear fusion is realized (refs. 22 to 26). For the configurations analyzed, values of α of the order of 1 kilogram per kilowatt were obtained (refs. 24 and 25). These studies were based on the assumption that diffusion of plasma across magnetic field lines could be reduced to the classical value, which considers only binary interactions. This high degree of magnetic containment has as yet been realized only in a few experimental plasmas at conditions far from those needed for fusion. The adequate containment of high-temperature, high-density plasma has long been the main problem in the world-wide program to achieve controlled fusion power. Consequently, the potential size and specific mass of a fusion reactor are analyzed parametrically in this report, with cross-field diffusion rate as one of the primary parameters.

The nuclear and plasma physics of controlled thermonuclear fusion, and the various experimental approaches taken to achieve it are described in numerous articles and books (e.g., refs. 27 and 28). Recent studies of fusion feasibility for ground power stations are given in references 29 and 30. Much research has been conducted with magnetic-mirror systems, which consist of a pair of solenoids, with the hot plasma contained in the volume between and inside them. The propulsion system studies in references 24 and 25 assumed such magnetic-mirror containment, with one mirror weaker than the other to permit preferential escape of the reaction products through the weaker mirror. These escaping reaction products were then to impinge on a stream of hydrogen which would thereby be accelerated to the desired jet velocity. Mixing with hydrogen is required because the escaping fusion reaction products by themselves would have a specific impulse of the order of 200 000 seconds, which is far beyond the optimum value for planetary propulsion times and estimated specific masses (eq. (21)). The resulting thrust-to-mass ratio would be too small for planetary missions. A study in reference 26 showed that the fusion reaction products could theoretically transfer their energy to the hydrogen propellant with an efficiency of about 25 percent at a specific impulse of 2500 seconds. This proposed method of propellant acceleration may therefore be feasible, but it requires experimental evaluation.

During the last few years, studies of instabilities and loss rates from the ends of mirror machines have indicated that the probability of achieving containment time long enough for sustained fusion is small with such systems (ref. 31). Reference 32 showed that to reduce end losses adequately for such an open-ended configuration, even assuming a quiescent plasma, minimum length of the order of 1 kilometer may be required. Consequently, emphasis has shifted to toroidal geometries, which although they introduce magnetic field gradients and other difficulties, at least eliminate the very large end losses. For propulsion applications, such a closed geometry is somewhat less convenient, but it should still be possible to achieve preferential ejection by providing a suitable hole in the torus. If the cross-field plasma diffusion loss remains large, as seems likely, it may be desirable to use the hydrogen propellant as a cool-gas blanket around the toroidal reaction volume to absorb some of the power that would otherwise flow to the magnetic field coils and also to intercept the impurities that would flow into the plasma from the container walls. The advantages of such screening are examined in reference 33.

The fusion rocket concept has some similarities to the gaseous-core fission rocket, but there are several major differences in addition to the type of nuclear reaction involved. For fusion, good fuel containment is important to produce a net power output with a reasonable reactor size and power level, and not so much for reasons of economy or for preservation of a natural resource. Also, the fusion reactions produce no radioisotopes. Consequently, direct mixing of fuel and propellant (outside the reaction volume) can be used for heating and accelerating the propellant. This permits attainment of higher specific impulse with the fusion rocket, which makes it a type II rather than a type I system. The higher specific mass of the fusion rocket results primarily from the need for magnetic containment rather than simple structural containment. Magnetic containment, in turn, is needed to reduce the outward mass flux rate, which is very rapid at thermonuclear-fusion temperatures.

To estimate the minimum size, specific mass, and power output of a fusion reactor as function of the achievable cross-field diffusion rate, consider the toroidal reactor geometry shown in figure 7, where the magnetic field is provided by a superconducting coil of uniform thickness $(r_2 - r_1)$ wound around the entire torus. The major radius is assumed to be twice the plasma radius r_0. A thickness $(r_1 - r_0)$ is allowed for insertion of shielding and cooling to protect the superconducting magnet windings and structure from the neutron and bremsstrahlung radiation and from the diffusion of hot plasma to the walls. The thickness $(r_2 - r_1)$ includes both superconducting winding and structure. The structure may be separate from the winding or it may be included as a substrate for the superconducting material. To estimate the power output corresponding to the radius r_0, one can select the reaction temperature as 10^9 K (10^5 eV), for which the fusion reaction cross sections are near maximum for the reactions of interest (ref. 28).

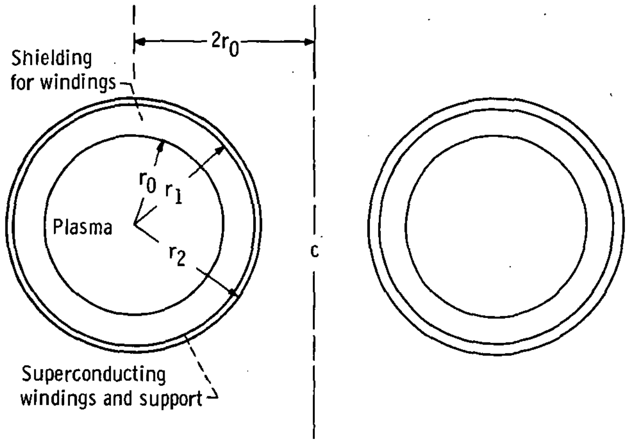


Figure 7. - Toroidal fusion reactor geometry.

For this temperature, a fusion power density of about 10^8 watts per cubic meter is obtained at a number density n of reacting particles of 10^{21} per cubic meter. The total fusion power density is proportional to the square of number density and can therefore be written

$$p_T = 10^8\left(\frac{n}{10^{21}}\right)^2 \qquad (W/m^3) \tag{35}$$

The total volume of the plasma is

$$V = 4\pi^2 r_o^3 \tag{36}$$

and the total thermal power is the product $p_T \times V$.

We consider now the minimum value of r_o needed to achieve net power output. This can be determined from the so-called Lawson parameter $n\tau$, where τ is the mean residence (containment) time of the reacting particles. A minimum value of this parameter expresses the condition that fusion power generated per unit volume must exceed the sum of the power required to heat the fuel to reaction temperature and the power radiated out by bremsstrahlung. The requirement $n\tau > 10^{20}$ is obtained for an assumed efficiency of one-third for the conversion of total energy into useful energy and for a deuterium-tritium (D-T) reaction. This is the value generally used in studies of ground power stations. For propulsion applications, however, references 24 and 25 show that a deuterium - helium-3 (D-He^3) reaction is more suitable, because the reaction products (protons and alpha particles) are charged particles subject to magnetic containment. This is of great importance when a direct fusion rocket (rather than a fusion-electric rocket) is considered, because the large neutron flux from the D-T reaction would require a heavier shielding and heat-disposal system for the superconducting magnet coils. This large neutron energy could be utilized as part of the heating cycle in a nuclear-fusion-electric propulsion system, as it is in ground-power station concepts, but such a system is unlikely to have specific mass much less than a nuclear-fission-electric system. Consequently, a D-He^3 reactor would be preferable for propulsion. For this reaction, the Lawson criterion becomes

$$n\tau > 10^{21} \qquad ((m^{-3})(sec)) \tag{37}$$

This value will be used in this analysis.[2]

The containment time τ can be expressed as

$$\tau \approx \frac{r_o}{\bar{v}} \tag{38}$$

where $\bar{v}$ is the mean diffusion velocity of particles normal to the magnetic field lines. It is the magnitude of this diffusion velocity that is critical in determining the minimum value of r_o.

Experimental measurements in the controlled fusion research program and in other plasma physics studies have frequently been in approximate agreement with a diffusion formula proposed by Bohm many years before fusion research began. This formula can be written

$$\bar{v}_B = \frac{10^{-4}T}{16Br_o} \qquad (m/sec) \tag{39}$$

where T is in K and B in tesla. This formula is in disagreement with the classical diffusion formula for particles in a magnetic field (refs. 28 and 34):

$$\bar{v}_c = \frac{10^9\beta}{r_o T^{3/2}} \qquad (m/sec) \tag{40}$$

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[2]An alternative expression derived in ref. 29 considers the fraction of the charged reaction-product energy which may be radiated away and hence be unavailable to heat the incoming reactants. For a rather high value of 0.8 for this radiated fraction, $n\tau$ is about twice the value given in eq. (37). The mechanisms of energy transfer from the high-velocity reaction products (protons and alpha particles) to the plasma are not yet clear enough to establish this fraction with confidence.

where β is the ratio of plasma pressure magnetic pressure.[3]

It appears that formula (40) is obeyed quite well if the plasma is maintained in a "quiescent" state. However, most plasmas generated in the fusion program are not in this benign state. The strong heating required together with the strong magnetic fields and currents may, in fact, be incompatible with such a state. The distinction is analogous to that between laminar and turbulent diffusion rates in fluid mechanics. Bohm diffusion is associated with instabilities and resulting mass motion of the plasma. The transition conditions in the plasma case, however, are not well established.

Since diffusion rates below the Bohm value have been difficult to achieve or maintain, it is useful to express the diffusion velocity in equation (38) as a fraction of the value that would result from Bohm diffusion. Let K be this fraction of Bohm diffusion. Then

$$\overline{v} = \frac{10^{-4} KT}{16 B r_o} \qquad (\mathrm{m/sec}) \tag{41}$$

which yields for the Lawson criterion:

$$n\tau \equiv 16 \times 10^4 \, \frac{n B r_o^2}{KT} > 10^{21} \qquad ((\mathrm{m}^{-3})(\mathrm{sec})) \tag{42}$$

One might assume that this condition could be satisfied regardless of τ by increasing n sufficiently. But another condition limits n, namely, the requirement that, for magnetic containment, the plasma pressure must be less than the magnetic pressure. This condition is

$$p \equiv nkT \leq \beta p_B \equiv \frac{10^7 B^2 \beta}{8\pi} \tag{43}$$

With $k = 1.38 \times 10^{-23}$ joule per K and $T = 10^9$ K, relations (42) and (43) yield the desired condition on minimum r_o:

$$r_o \geq 464 \left(\frac{K}{\beta B^3} \right)^{1/2} \qquad (\mathrm{m}) \tag{44}$$

[3]The formula for magnetic pressure (or energy density) is $p_B = 10^7 B^2 / 8\pi$, where B is in T and p_B is in N/m^2 or J/m^3.

For comparison, the equivalent relations for classical (collisional) diffusion are presented. Equation (40) leads to the criterion

$$n\tau \equiv \frac{n r_o^2 T^{3/2}}{10^9 \beta} > 10^{21} \tag{45}$$

The conditions bracketing n are then, for $T = 10^9$ K,

$$3.16\times10^{16} \frac{\beta}{r_o^2} \leq n \leq 2.9\times10^{19} B^2 \beta \tag{46}$$

or

$$r_o \geq \frac{0.033}{B} \qquad \text{(m)} \tag{47}$$

Obviously, the minimum size limitation is not a problem if classical plasma diffusion can be realized.

Condition (44) determines the minimum size of the reactor for a given diffusion rate, magnetic field strength, and plasma pressure. An additional condition, which determines the maximum allowable diffusion rate for a given reactor size, is the heat flux carried to the walls by the plasma. This heat must be absorbed by a coolant and transported to a radiator for rejection to space. However, the radiator temperature could be near the materials limitation (say 2000 K), since no thermodynamic power conversion cycle is desired. At this temperature, the radiator specific mass can be about one-sixteenth of the values (1 to 3 kg/kW) given in the preceding section for the electric propulsion system radiating at about 1000 K. It might therefore be only about 0.12 kilogram per kilowatt based on useful power output. Thus, if waste heat can be rejected near 2000 K, the specific mass of the radiator is not a major mass component.

The magnitude of the heat flux rate to the walls, however, limits the allowable diffusion rate. This magnitude should be less than about 1 kilowatt per square centimeter (10^7 W/m^2), which is of the order of the value experienced in chemical rocket nozzles. If 80 percent of the total reactor power flows to the walls (corresponding to 20 percent efficiency), then the heat flux rate (from eqs. (35), (36), and (44)) can be expressed as

$$\frac{P_H}{A} = 1.56\times10^7 \beta^{3/2} K^{1/2} B^{5/2} < 10^7 \qquad (\text{W/m}^2) \tag{48}$$

where A is the inner surface area of the torus, $8\pi^2 r_o^2$.

This condition together with condition (44) can be used to bracket the values of K and B required to produce a self-sustaining fusion reactor:

$$\left(\frac{464}{r_o}\right)^{2/3}\left(\frac{K}{\beta}\right)^{1/3} < B < 0.84(K\beta^3)^{-1/5} \tag{49}$$

The two end terms in this relation yield a condition relating K and β:

$$K < \frac{r_o^{5/4}}{3000\sqrt{\beta}} \tag{50}$$

This relation shows that for a plasma radius r_o of 1 meter and for values of β between 0.1 and 1.0, the diffusion rate fraction K must be between 10^{-3} and 3×10^{-4} of the Bohm value. The magnetic field strength for this range varies from 18 to 6 teslas.

Table I compares required diffusion rates and other parameters for r_o = 0.1, 1, and 10 meters and for $P_H/A = 10^7$ watts per square meter.

TABLE I. - SOME REQUIRED PARAMETERS FOR

THERMONUCLEAR FUSION REACTORS

$n\tau$	Plasma radius, m	β	K_{max}	Magnetic field, T	Output power (η=20 percent), W
10^{21}	0.1	0.10	5.9×10^{-5}	23.2	2×10^{6}
	.1	1.0	1.9×10^{-5}	7.4	2×10^{6}
	1.0	.1	1.0×10^{-3}	12.9	2×10^{8}
	1.0	1.0	$.3\times10^{-3}$	4.1	2×10^{8}
	10.0	.1	1.8×10^{-2}	7.2	2×10^{10}
	10.0	1.0	$.6\times10^{-2}$	2.4	2×10^{10}
10^{20}	10.0	0.1	.18	7.2	2×10^{10}
	10.0	1.0	.06	2.4	2×10^{10}

These sizes correspond, respectively, to a research reactor, a propulsion reactor, and a large ground power reactor. For the last of these, a D-T reaction is more appropriate than the D-He^3 reaction assumed for the preceding derivations. This permits use of the more liberal Lawson criterion ($\eta\tau = 10^{20}$ instead of 10^{21}). The last two rows show that the diffusion requirement is moderated by a factor of 10 when this lower value is used. The resulting values of K_{max} are large enough to make such a reactor feasible with diffusion rates that have already been attained under some conditions. Furthermore, with this factor-of-10 increase in allowed K, a fusion reactor with $\beta = 0.1$ and $r_0 = 17$ meters is feasible, even with the Bohm diffusion rate (K = 1.0). The power output, however, would be 2×10^{11} watts. This power level corresponds to roughly 10 percent of the power output of the United States. Thus, one may say that controlled thermonuclear-fusion power is feasible from the standpoint of attained plasma loss rates, provided that the reactor is large enough!

We conclude that, for propulsion applications (and also for research and development leading to large power stations), the diffusion rate of plasma across magnetic field lines must be reduced by a factor of about 1000 below the Bohm diffusion rate. Whether this rate is achievable under the conditions of a reacting thermonuclear plasma remains to be determined. A world-wide research effort is under way to answer this question.

The preceding discussion showed how the plasma loss rate, and its associated rate of transport of heat to the walls determines the minimum size, maximum allowable values of K, and the required values of magnetic field strength.

We next consider the problem of shielding the superconducting material from the neutrons and bremsstrahlung (X-rays) emanating from a fusion plasma. Although the $D\text{-}He^3$ fusion reaction produces only an alpha particle and a proton, the deuterium particles also react with each other to significant extent. The D-D side reactions produce neutrons with energies of millions of electron volts. The bremsstrahlung power load, however, is more serious than the neutron heating. Component masses have been esti-

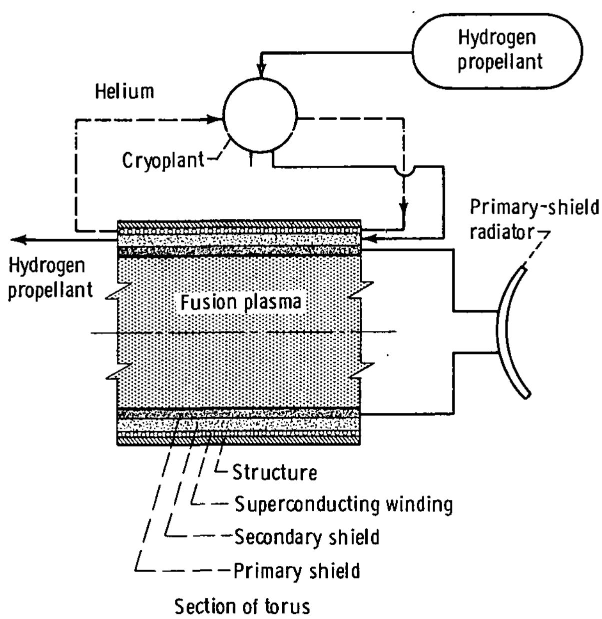

Figure 8. - Schematic diagram of toroidal fusion rocket propulsion system (Englert).

mated by Englert for the toroidal configuration of figure 7 in a manner similar to that in reference 25 for the magnetic mirror configuration. Figure 8 shows a schematic diagram of the propulsion system components considered. The hydrogen propellant is used to absorb part of the heat generated in the shielding material by the radiation from the plasma. The calculations show that the shielding and the refrigeration plant (cryoplant) are the heaviest parts of the system and can considerably outweigh the magnet and structure. For example, for the 1-meter reactor of table I the combined cryoplant and shielding specific masses are about 3.5 kilograms per kilowatt for the $\beta = 0.1$ case to about 1.5 kilograms per kilowatt for the $\beta = 1.0$ case. The superconducting windings were assumed to be separate from the structural backing, so that the entire structure does not have to be maintained at liquid helium temperature. A current density of 10^9 amperes per square meter was assumed, since this has been achieved in (unstabilized) small coil samples at 10 teslas (ref. 35). It has not yet been achieved, however, for stabilized[4] large coils, where values of the order of 3×10^8 amperes per square meter are more common (ref. 36).

These estimates of specific mass illustrate that, to make thermonuclear fusion attractive for space propulsion, we need large superconducting magnets with very high current density to minimize the mass that must be cooled to 4.2 K. Any improvement in raising the critical temperature above 4.2 K at high current density will also help greatly in reducing specific mass, because refrigeration plant mass is very sensitive to the minimum required temperature.

Another way to reduce the specific mass might be to produce the magnetic containment field with a group of equally spaced coils, distributed around the torus, instead of continuous winding. Such a configuration, called a "bumpy torus" because of its nonuniform field, has other attractive features, one of which is that charged-particle drift directions tend to be alternated so that the particles are theoretically contained within the torus longer (ref. 28). However, to maintain the same average value of magnetic field strength, the current density would have to be greater than for the continuous winding. If such higher current density is possible, it could also be used with the continuous winding. The net reduction in total mass of cooling winding, and hence in required

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[4]Stabilization is the process of counteracting the effect of localized heating in the superconducting material, which may drive the coil into the normal resistive state. These local "hot spots" may be produced by sudden jumps in magnetic flux. Stabilization can be accomplished by providing enough high-conductivity material in electrical contact with the superconducting material so that the current can be carried around the local hot spot until it cools down again (ref. 37). Another method (ref. 38) is to use superfluid helium (attained at reduced pressure by cooling normal liquid helium). The superfluid acts as a superconductor of heat and is therefore very appropriate as a coolant for superconductors of electricity.

shielding and cryoplant mass, may therefore not be significant.

Even more attractive from the standpoint of mass reduction is the prospect of eliminating entirely the coils external to the plasma and instead generating the containing field with huge currents through the center of the plasma, either through superconductors or through the plasma itself. Early experiments on the latter method, however, were discouraging because the plasma quickly broke up. Theory also indicated that such a containment field is highly unstable. Nevertheless, new concepts of confinement and heating without external magnet coils may still materialize.

In the meantime, research on superconducting materials and coil configurations may produce less need to shield and cool the material. Of course, any improvement in diffusion loss rate would increase the usable power and hence reduce the specific mass for a given confinement system.

In summary, if thermonuclear fusion becomes feasible for about 100 megawatts of output power, specific masses (without crew shielding) of the order of 1 kilogram per kilowatt may be possible. This compares with the value of about 6 kilograms per kilowatt achievable (without shielding) for a comparable nuclear-electric system mass (about 10^5 kg, fig. 6). The improvement over nuclear-electric propulsion results from several factors: the replacement of a solid-core reactor with a plasma core, the elimination of the thermodynamic cycle, which permits high radiator temperature, and the elimination of separate thrusters and power conditioning.

The problem of crew shielding may be as significant for the fusion rocket as for the nuclear-electric rocket. Although the neutron flux is reduced by the magnet shielding and structure, it is still far too large for unshielded continuous human exposure near the reactor. Rough estimates indicate that an additional 200 grams per square centimeter of total shielding thickness would be required to reduce the dosage to a level suitable for continuous human exposure. The projected area of the torus of figure 7 with an r_0 of 1 meter and an r_2 of 1.3 meters is about 30 square meters face-on (viewed along the axis of symmetry) or about 17 square meters end-on (parallel to the plane of the torus). The shadow-shielding mass for these views would therefore be 6×10^4 and 3.4×10^4 kilograms, respectively. With the power output of 200 megawatts, the specific masses of these shields are 0.3 and 0.17 kilogram per kilowatt. Although these specific masses are not prohibitive, they are still a very sizable fraction of the total, particularly if improvements in superconducting materials result in lower magnet shielding requirements. A possible reduction in shielding mass could result from shielding the crew cabin, rather than the fusion reactor, and by increasing the separation distance between crew and reactor. A significant advantage of fusion reactors over fission reactors is that no radioactive products are produced. Consequently, when the fusion reactor is turned off, no radiation hazard should exist. A shadow shield or crew shield should therefore be adequate for the deep-space missions contemplated for the system. When human access is necessary, the reactor could be turned off.

The need for a restart capability implies an additional mass component of the system which has not yet been discussed. Since most of the fusion research now under way is directed specifically at determining what it takes to ignite a fusion reaction, little can be said about the required mass of a starting system, except that it will tend to be lower as the plasma containment gets better. Starting concepts being investigated include very fast condenser-bank discharges, high-energy plasma and ion injection systems, and radiofrequency heating systems. If we assume that a condenser-bank discharge will work, then the mass of the starting system is directly proportional to the required energy storage. A figure of about 0.01 kilogram per joule has been estimated to be possible for capacitor banks designed for low mass. Values near 0.05 kilogram per joule have already been achieved (ref. 39). If a megajoule bank turns out to be adequate, and if 0.01 kilogram per joule is attained, the mass would be 10^4 kilograms, which is not a major problem at the 200 megawatt reactor power output level. But a 10-megajoule rerequirement would add more than 0.5 kilogram per kilowatt, and thus would be a serious handicap unless lighter weight energy storage systems can be developed.

Obviously, the status of controlled fusion research is such that no great confidence can be placed in the sizes and masses estimated herein. The best that can be said is that, so far, nothing has turned up that rules out the eventual achievement of controlled fusion systems with specific masses considerably lower than those achievable with nuclear-electric propulsion systems. Improvements in superconducting materials, energy storage systems, structural materials, cryoplants, and other technologies will all help to achieve lower specific mass.

HOW FAR AND HOW FAST CAN WE GO?

Table II summarizes the performance limits associated with each of the propulsion concepts, and the planetary round-trip times that would be attainable if these performance parameters can be achieved. Trip times longer than 2500 days have not been included.

As pointed out before, vehicle sizes and payload ratios closer to the one-stage values rather than the four-stage values constitute reasonable launching requirements. Table II shows that type II systems tend to be superior to type I systems for trips to Jupiter and beyond. This statement assumes that roughly comparable levels of technology are represented by the listing order of the two types. In actuality, the type I system is in each case somewhat ahead of the corresponding type II system with regard to research or development status.

Not shown by the table is the effect of combining type I and type II systems for a given mission. Studies for Mars round trips have shown that such a combination consist-

TABLE II. - SUMMARY OF PERFORMANCE AND MISSION CAPABILITY

(a) Type I systems

System	Maximum specific impulse, sec	Round-trip time, days									
		Mars		Jupiter		Saturn		Uranus		Pluto	
		Single stage	Four stages	Single stage	Four stages	Single stage	Four stages	Single stage	Four stages	Single stage	Four stages
Chemical	500	---	200	----	1250	----	2500	----	----	----	----
Solid-core fission	900	---	120	----	700	----	1500	----	----	----	----
Gas-core fission Nuclear pulse	2500	160	40	1100	280	2200	560	----	1200	----	2500

(b) Type II systems

System	Minimum specific mass, kg/kW	Round-trip time, days									
		Mars		Jupiter		Saturn		Uranus		Pluto	
		Single stage	Four stages	Single stage	Four stages	Single stage	Four stages	Single stage	Four stages	Single stage	Four stages
Solar electric	10 (1 AU)	450	230	960	600	1600	1050	----	----	----	----
Nuclear-fission electric	7	350	190	800	500	1200	800	2400	1400	----	2300
Thermonuclear fusion	1	150	90	480	280	780	480	1250	800	2100	1500

ing of a solid-core fission rocket stage and a nuclear-electric rocket stage can reduce trip time more for a given launch weight than either type alone (refs. 40 and 41). Similar improvements should be possible for more distant planets.

One thing that we can see from table II is that each advance in propulsion capability not only reduces the time of the missions that can be accomplished with lower-performance systems, but makes other missions possible. A general conclusion is that the more advanced systems, if they prove feasible, can reduce trip times to the near planets by factors of 3 to 5 and can make several outer planets accessible to manned exploration.

Not shown here are the improved capability for less difficult missions, such as unmanned, one-way solar-system probes, high-payload lunar transports, and various manned and unmanned Earth-orbital missions.

But even with a gaseous-core nuclear-fission rocket or a thermonuclear fusion rocket, we still will not be able to get to Mars and back in a month, or to Pluto and back in a year. Is there any possibility of getting the required specific impulse of 10 000 seconds in some type I system, or a specific mass of 0.03 kilogram per kilowatt in some type II system? Perhaps we need more efficient conversion of mass into energy. The big advantage of nuclear energy over chemical energy is the much greater energy release per unit mass. But the fission and fusion processes convert less than half of 1 percent of the mass into energy. Perhaps if more complete mass annihilation someday becomes controllable, we could achieve both very large specific impulse and extremely small specific mass.

The main problems with high performance propulsion systems, however, are not associated with limited mass-energy conversion. The problems are really those of power containment and the conversion of isotropic power to directed jet power. For the gaseous-core nuclear-fission rocket, the minimum critical size is determined by the maximum gas pressure that can be tolerated by the chamber walls, and the maximum specific impulse is limited by the process of transferring energy from the fuel to the propellant. For the thermonuclear-fusion rocket, both the minimum size and minimum specific mass are limited by the plasma containment capability of magnetic fields and the resulting energy flux to the walls. More complete mass-energy conversion will tend to accentuate these containment and conversion problems. No promising method has been suggested, for example, to contain and redirect the high-energy photons resulting from mass annihilation, without appreciable loss of energy to a containing chamber.

The best approach for achieving advanced propulsion capabilities seems to be to continue working toward better fission and fusion systems. They may not look good relative to the warp drive, and they have many difficult problems, but they are nevertheless the best that we can now visualize for the space journeys of the future.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, June 9, 1969,
120-27-06-04-22.

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