PROPULSION METHODS IN ASTRONAUTICS

By W. E. MOECKEL

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio

INTRODUCTION

THERE is fairly general agreement that chemical rockets, particularly those using high-energy propellants, are capable of launching most of the unmanned scientific missions to the Moon and the nearby planets. These preliminary missions will involve one-way, instrumented trips with moderate payload. Even though some nonchemical propulsion systems could reduce the weight required, it appears that much of the data-gathering work needed before manned space flights are feasible will be well under way before nonchemical systems become available. This paper is therefore limited to a discussion of those nonchemical systems that appear feasible for manned expeditions to the Moon and nearby planets. For such expeditions, the payload weight will be at least an order of magnitude greater than for unmanned flights, and the initial gross weights required with chemical rockets become extremely large.

Because of the high payload weight required for adequate manned missions, or for large-scale freight transfers, it appears likely that such missions will require assembly in orbit. This means that a primary consideration in selecting propulsion systems for such missions will be the minimization of total initial weight in orbit, since each pound placed in orbit will require from 10 to 100 or more pounds in launching weight, depending on the type of launching rockets used.

During the past years, a wide variety of nonchemical propulsion systems for space missions have been suggested and analyzed (see, for example, refs. 1–4). It is not possible in a single paper to consider all of these systems in detail. Instead, this paper discusses the significant parameters that determine initial weight needed for space missions, and presents a descriptive survey and classification of feasible nonchemical propulsion systems.

GENERAL CONSIDERATIONS

The three basic performance parameters for space propulsion systems are specific impulse, thrust–weight ratio (or initial acceleration), and specific powerplant weight. The first two of these parameters determine the trajectory followed by a space vehicle in any gravitational field. They determine also the propellant weight required to perform a given mission. If the initial acceleration is low, more propellant will be needed for a given specific impulse due to gravitational losses. These losses result from the longer propulsion times required to attain a given transfer energy with low acceleration. It is possible, then, that a system with high specific impulse but low thrust may require more propellant to accomplish a mission than one with lower specific impulse but higher thrust. The magnitude of this effect is illustrated in Fig. 1, where the propellant-to-gross-weight ratio required for a minimum-energy round trip to Mars is plotted as function of specific impulse for several initial accelerations.

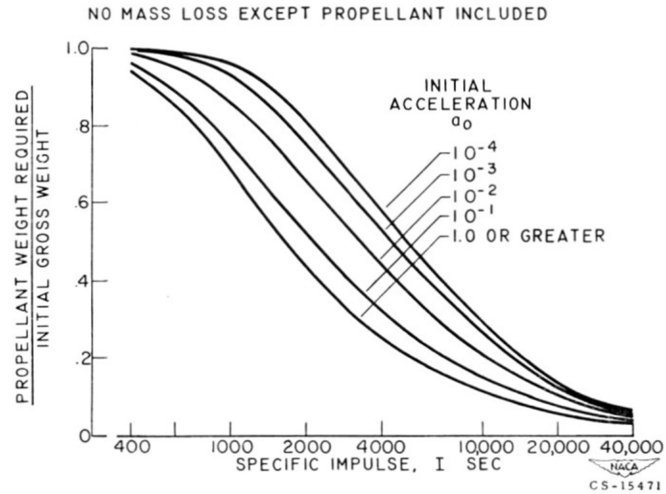


FIG. 1. Illustration of gravitational loss for Mars round-trip (satellite-to-satellite).

These curves are based on numerical integrations of constant-thrust trajectories for minimum-energy transfer from a circular orbit at a radius 1·1 times the Earth's radius to a circular orbit at a radius of 1·1 times the radius of Mars. The initial accelerations are given in terms of gravitational acceleration at the radius of the initial orbit. For this example, only the mass reduction due to propellant consumption was considered, that is, no propellant tanks or waste materials were disposed of during the trip. The propellant weight ratios are therefore somewhat pessimistic, since actual missions will have considerable disposable weight.

At the extremes of very low or very high specific impulse, there is little spread between curves for different initial accelerations. In the first case, the weight is almost all propellant, and in the other case, very little propellant is required for the mission. In the intermediate range, however, initial acceleration is a very significant parameter. Suppose, for example, that we have a propulsion system capable of producing an initial thrust-weight ratio a_0 of 1·0 with a specific impulse of 700 sec. This system could perform the mission with propellant-to-gross weight ratio of 0·8. Another propulsion system, with initial thrust-weight ratio of 10^{-4} g, would have to produce a specific impulse of at least 2000 sec to perform the mission with less propellant weight. A low-thrust system must therefore have a very considerable margin in specific impulse to make it superior to a high-thrust system on the base of propellant consumption alone.

Of course, propellant weight is not the only significant factor. A low-thrust system is also likely to have a heavier powerplant than a high-thrust system. The quantity of most interest, then, is the powerplant plus propellant weight ratio, since the difference between this weight ratio and unity is essentially the margin available for the payload weight ratio.

The powerplant weight can be represented in terms of the third significant parameter, the specific powerplant weight. This parameter α is the powerplant weight divided by the jet power produced. Although this parameter is applicable to all propulsion systems, it is particularly convenient for electric propulsion systems for which the weight of the propulsion system is in many cases almost directly proportional to its electric power output, at least for a considerable range of powers. The parameter α, in this case, includes any inefficiency in the conversion of electric power to jet power.

In terms of α, the powerplant weight is given by

$$\frac{W_{PP}}{W_0} = \frac{\alpha P_j}{W_0} \tag{1}$$

where α is in pounds per jet kilowatt and P_j is jet power in kilowatts. With these units, the jet power is

$$P_j = \frac{FI}{45{\cdot}9} \tag{2}$$

so that

$$\frac{W_{PP}}{W_0} = \frac{F}{W_0}\frac{\alpha I}{45{\cdot}9} \equiv \frac{a_0 \alpha I}{45{\cdot}9} \tag{3}$$

where F is thrust in pounds. Note that this weight ratio is directly proportional to each of the three basic performance parameters.

The propellant weight ratio (plotted in Fig. 1 for the Mars trip) is

$$\frac{W_{\text{prop}}}{W_0} = \frac{F}{W_0}\frac{\tau}{I} = \frac{a_0 \tau}{I} \tag{4}$$

where τ is total propulsion time required for the mission. The sum of powerplant and propellant weight ratios is therefore

$$\frac{W_{PP} + W_{\text{prop}}}{W_0} = a_0\left(\frac{\alpha I}{45{\cdot}9} + \frac{\tau}{I}\right) \tag{5}$$

This expression shows that, for each a_0, there exists an optimum specific impulse which minimizes the powerplant plus propellant weight, and therefore maximizes the payload ratio for a given mission. If propulsion time were independent of specific impulse, this expression could be differentiated with respect to I to find the optimum specific impulse for each a_0 and α. This process gives values of I_{opt} and payload weight that are not too far from those obtained by more exact analyses. However, for the mission being used for illustration, the propellant weight ratio has been calculated numerically (Fig. 1), so it is only necessary to add the powerplant weight ratio (Eq. 3) to find the powerplant plus propellant weight ratio as function of the three performance parameters. Some results are shown in Fig. 2, where the required powerplant plus propellant weight ratio for the mission of Fig. 1 is plotted as function of specific impulse for several combinations of initial accelerations and specific powerplant weight. Figure 2(a) contains curves for the lower range of initial accelerations ($a_0 = 10^{-3}$ and 10^{-4}), while Fig. 2(b) shows similar curves for the higher range ($a_0 = 1, 10^{-2}$).

As might be expected from Eq. (5), in the range of high specific impulse, where powerplant weight is dominant, the curves tend to become linear functions of I, with the product $a_0\alpha$ as parameter. However, in the range of low specific impulse, where propellant weight is dominant, the curves are more independent of α, and are functions principally of the initial acceleration, which determines propulsion time and gravitational loss.

Figure 2(a) shows that, for the low range of thrust–weight ratios, the Mars round-trip mission can be accomplished with payload weight ratios near 0·5 if the product $a_0\alpha$ is near 10^{-3} lb per kilowatt and if specific impulses in the range of 6000 to 20,000 sec are achieved. If the specific powerplant weight is changed by any factor, the initial acceleration can be changed by the reciprocal of the same factor without serious effect on payload ratio. The figure shows also that this mission cannot be accomplished with significant payload if $a_0\alpha$ is greater than about 4×10^{-3}.

For the higher-thrust range (Fig. 2b), the optimum specific impulses are lower, for a given $a_0\alpha$, than for the low-thrust systems. In the range of low specific impulse (less than 2000) the effect of specific powerplant weight is not as significant as the effect of initial thrust-weight ratio, provided that the product $a_0\alpha$ is of the order 3×10^{-3} or less.

To illustrate these results more concretely, consider two hypothetical propulsion systems. The first, due to material temperature limitations, has a fixed maximum specific impulse of 1500 sec. According to Fig. 2(b) this propulsion system could perform the assumed Mars mission with a powerplant plus propellant weight ratio of about 0·6 provided that the initial acceleration is near unity and the powerplant specific weight is less than about 0·003 lb per jet kilowatt. Higher accelerations would not reduce this weight, because initial accelerations of about 1·0 are nearly equivalent to impulsive velocity increments, which, of course, produce

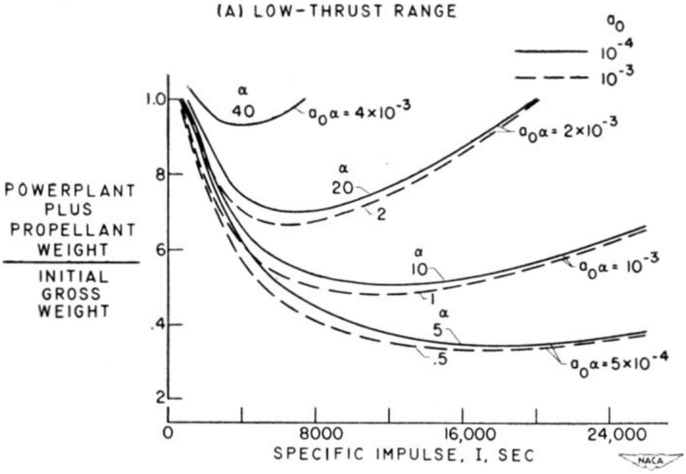


FIG. 2. Effect of powerplant parameters on weight required for minimum-energy Mars round-trip (satellite-to-satellite).

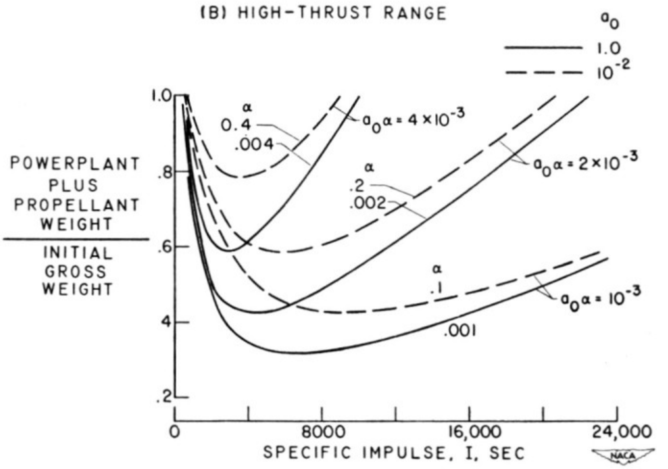


FIG. 2 (*cont.*). Effect of powerplant parameters on weight required for minimum-energy Mars round-trip (satellite-to-satellite).

zero gravitational loss. Further reductions in specific powerplant weight will also produce little increase in payload weight because the powerplant weight is already much less than the propellant weight. However, increases in powerplant weight above the 0·003 lb per kilowatt value soon force a reduction in initial acceleration, with progressive deterioration of the payload ratio. Thus, if $\alpha = 0{\cdot}01$, an initial acceleration of 1·0 will produce a product $a_0\alpha$ of 10^{-2}, which is too large to accomplish the mission. The initial acceleration must be reduced to maintain $a_0\alpha$ less than about 4×10^{-3}, that is, a_0 must be reduced to 0·4 g, and the resulting powerplant plus propellant weight ratio is increased to about 0·75. If α were as high as 0·1 lb per jet kilowatt, the acceleration would have to be reduced to about 0·04 g, and the powerplant plus propellant weight ratio would be about 0·85. Thus, for a propulsion system with specific impulse limited to 1500 sec, the powerplant specific weight must be less than 0·1 lb per kilowatt to have significant payload for the assumed Mars mission. With zero powerplant weight, this system could produce a maximum payload weight ratio of about 0·4.

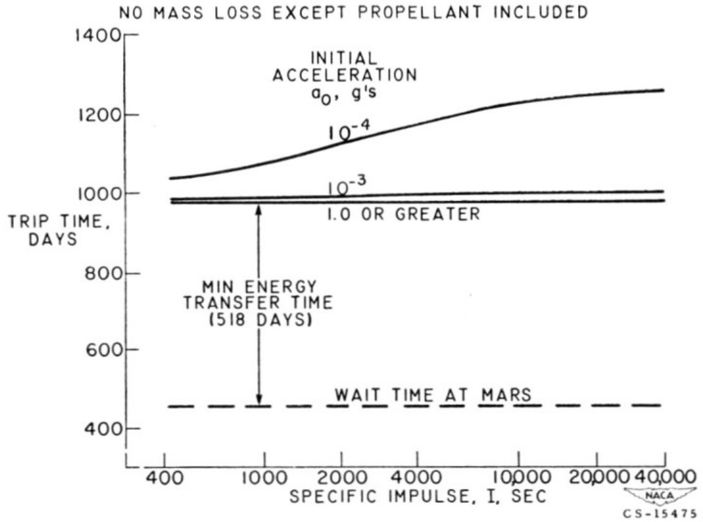


FIG. 3. Total time for minimum-energy Mars round-trip (satellite-to-satellite).

Consider now a second propulsion system, for which the specific impulse is not limited, but which requires a specific powerplant weight of the order of 10 lb per jet kilowatt. With such a system, obviously, the initial acceleration must be less than about 4×10^{-4} g in order to produce significant payload weight. Reducing the acceleration to 10^{-4} g will yield a payload of about 50% of gross initial weight with a specific impulse of about 10,000 sec. A further reduction to 5×10^{-5} g increases the payload weight ratio to about 0·65, using a specific impulse of about 18,000 sec. For this type of system, the payload ratio is determined principally by the product $a_0\alpha$, so that, for example, a payload ratio of 0·5 can be attained even with much higher specific powerplant weights if initial acceleration is correspondingly reduced. The limitation, of course, arises in the time required to perform the mission. For a manned expedition the payload itself increases as the time required increases, so that even though the payload ratio can be maintained, the total weight required to perform the mission increases.

The time required to complete a minimum-energy Mars mission is shown in Fig. 3 as function of specific impulse and initial acceleration.

For high-thrust systems, the total time, including waiting time at Mars, is about 970 days. There is little change in the total time for initial acceleration down to 10^{-3} g. With $a_0 = 10^{-4}$, however, the difference is quite appreciable—about 250 days for a specific impulse near 10,000. This difference is approximately inversely proportional to initial acceleration, so that for an initial acceleration of 10^{-5} g, the difference in time would be of the order of 2500 days. Obviously, accelerations less than about 10^{-4} g are not tolerable for an interplanetary mission starting from an orbit near the Earth. For $a_0 = 10^{-4}$ g, however, the difference is moderate enough so that it could be considered if the weight savings are great enough. For a Moon trip, the time required for a round trip with $a_0 = 10^{-4}$ is about 160 days, as compared with about 14 days with $a_0 = 10^{-3}$. Values of $a_0 = 10^{-4}$ therefore do not look attractive for manned Moon missions, although they might still be feasible for high-weight cargo transfer.

The total trip times shown in Fig. 3 can, of course, be reduced by providing excess energy above the minimum-energy value. Reduction in total trip time involves not only reductions in transit time, but also reduction in waiting time at the destination planet. Low-thrust systems are somewhat handicapped in this respect, because sufficient time must be allowed at the destination to spiral in and out of an orbit which is close enough to permit landing and take-off without excessively large auxiliary rockets. For excess-energy trips, therefore, the lower limit for allowable initial thrust-weight ratio may be considerably greater than the 10^{-4} g value permissible for minimum-energy journeys.

Although most of this discussion was based on requirements for a minimum-energy Mars mission, conclusions are almost equally valid for a minimum-energy Venus mission, since the transfer energies required for the two missions are not too different.

ELECTRIC POWER GENERATION

Many of the nonchemical propulsion systems to be described are electric systems, that is, they rely on electric power to ionize, heat, and/or accelerate the propellant. Moreover, for acceptable accelerator schemes, the power generating equipment is by far the heaviest portion of these systems, and will therefore largely determine the thrust-weight ratio allowable for a given mission. For this reason, a discussion of probable weights of electric power generating systems is pertinent before the thrust-generating systems themselves are described.

Weight estimates made by members of the NASA Lewis laboratory staff for several electric power generating systems are shown in Fig. 4.* Along the abscissa are shown the electric power levels appropriate for

*Particular credit is due to R. E. English, B. Lubarsky, and S. H. Maslen for the design studies which produced these weight estimates. These weights are believed to be achievable with reasonable extensions of current technology, but are not necessarily minimum possible weights.

various space propulsion missions. At the higher power levels required for manned interplanetary expeditions, only three of the many possible generating methods were found to be competitive on a specific weight basis—the nuclear-fission turboelectric, solar turboelectric, and the thermonuclear or fusion electric systems. The nuclear turboelectric system weights were based on use of a sodium-vapor cycle, with turbine inlet temperature of 2500° R and radiator temperature of 1800° R. The cycle efficiency for these temperatures was 20%. The design was based on minimization of radiator size without exceeding reasonable temperatures in other parts of the system. Even with this minimization, the radiator was the heaviest part of the generating system except at low power levels, where shielding weight became dominant.

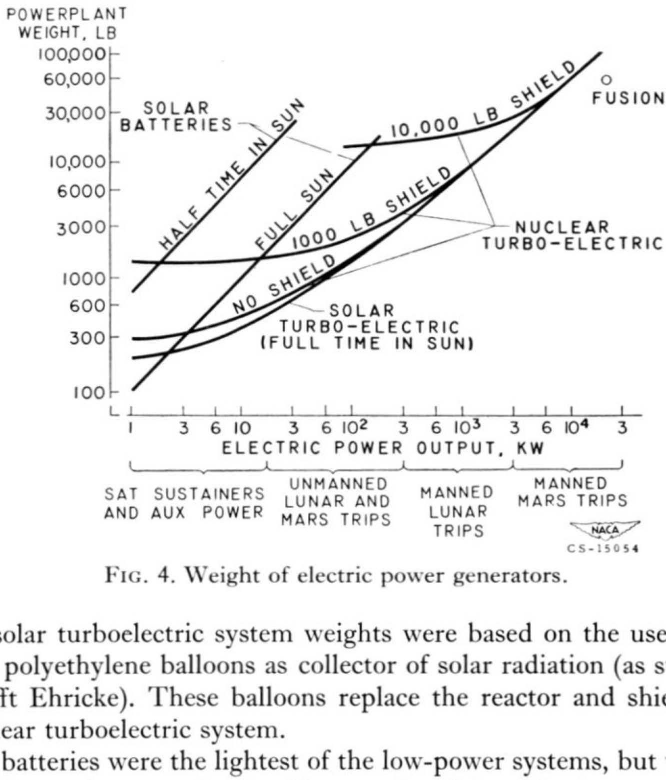


FIG. 4. Weight of electric power generators.

The solar turboelectric system weights were based on the use of half-silvered polyethylene balloons as collector of solar radiation (as suggested by Krafft Ehricke). These balloons replace the reactor and shielding of the nuclear turboelectric system.

Solar batteries were the lightest of the low-power systems, but were not competitive at the power levels of interest for high-payload space missions.

The solar turboelectric system was found to be very close in weight to the unshielded fission turboelectric system at all power levels. The straight line which represents both systems at high power levels corresponds to a specific powerplant weight of about 5·5 lb per electric kilowatt. If the electric power could be converted to jet power with 100% efficiency and no accelerator weight, the specific powerplant weight α for electric propulsion systems would then be 5·5 lb per jet kilowatt. Allowing for some inefficiency in conversion, and some accelerator weight, it appears that a specific powerplant weight between 5 and 10 lb per jet kilowatt should be attainable. These values, together with the curves of Fig. 2, indicate that electric propulsion systems should operate with initial acceleration between 10^{-4} and 2×10^{-4} g at a specific impulse near 10,000 sec to accomplish a round-trip satellite-to-satellite Mars mission with a payload ratio near 0·5.

A rough estimate was also made of the weight of an electric generator using thermonuclear fusion. This estimate was based on use of barium titanate condensers to generate a stabilized pinch, with direct extraction of electric power from the coils surrounding the pinch. If such a system eventually becomes feasible, it appears that 20 MW of electric power may be attainable with a weight of about 60,000 lb, giving a specific weight of about 3 lb per electric kilowatt. This would mean an increase by a factor of about 2 in the allowable acceleration with the same payload; however, this value is still too dubious to warrant its use in comparison of electric propulsion systems with other systems. The increase in weight with power output will probably be more gradual with fusion power systems than with fission or solar turboelectric systems, so that the weight advantage of fusion power may be much more striking at power levels higher than 20 MW, and may disappear at lower power levels.

CLASSIFICATION AND DESCRIPTION OF NONCHEMICAL PROPULSION SYSTEMS

The wide variety of nonchemical propulsion systems that might be feasible for high-payload space missions can be conveniently divided into three groups.

Group I (shown in Fig. 5) consists of those systems whose specific impulse is limited by material temperatures. This group includes the nuclear fission heat-transfer rocket, the electric arc rocket, and the solar heat-exchange rocket.

For the nuclear heat-exchange rocket, the specific impulse is limited to the value attainable with hydrogen at the temperatures tolerable for the fuel and moderator elements. Using an allowable temperature of 5000° R, the specific impulse will be limited to values less than 900 sec with a chamber pressure of 10 atmospheres and a value less than 1500 sec at a chamber pressure of 10^{-2} atmospheres. The higher specific impulse at low pressures results from the increased dissociation of hydrogen at a given temperature as pressure is reduced. The nuclear fission heat-transfer rocket is theoretically capable of take-off and launching operations. For such operations, the chamber pressure must be high, so that the specific impulse will be limited to values of 900 sec or less. For missions starting from assembly orbits, the chamber pressure can be much lower, so that the higher specific impulses may be achievable. With lower pressure, however, the thrust–weight ratio tends to decrease, so that acceleration near 1 g may not be possible with specific impulses in the range 1200 to 1500 sec.

Figure 2(b) shows that with a specific impulse of 1500 sec the round-trip Mars mission can be accomplished with payloads of the order of 30–40% of gross weight in orbit, provided that the powerplant specific weight is of the order of 0·001 to 0·004 lb per jet kilowatt. This is in the range of powerplant weights typical of high-thrust chemical rockets. With lightweight construction, such as is feasible at low chamber pressures with moderate acceleration, it may be possible to attain specific weights in this range with nuclear heat-transfer rockets. If the specific weights are higher, the initial accelerations must be reduced, and the payload ratios will suffer.

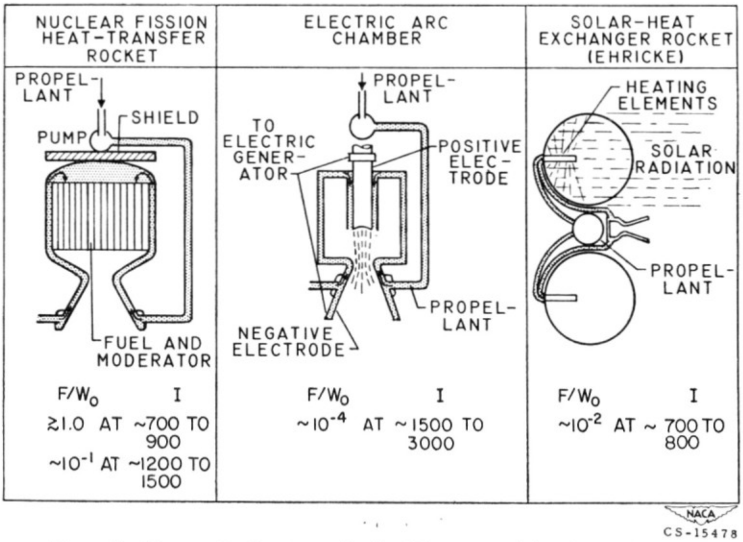


FIG. 5. Group I. Systems limited by material temperatures.

The electric-arc rocket utilizes electric power to heat the propellant. It must therefore have an electric generator system, which means that the powerplant specific weight will be of the order of 5–10 lb per jet kilowatt. Like all electric propulsion systems, it must therefore operate with an initial acceleration of the order of 10^{-4} g to produce appreciable payload for the assumed Mars round-trip. However, unlike other electric propulsion systems, the specific impulse for the electric-arc rocket is limited by factors such as electrode erosion rate, maximum nozzle heating rate, and over-all nozzle cooling requirements. Analyses conducted at the Lewis laboratory indicate that electrode erosion rate may limit this system to specific impulses less than 1500 sec. If this difficulty is overcome, over-all cooling requirements appear to limit specific impulse to values in the range of 2000–4000 sec.

According to Fig. 2(a), the ultimate payload capacity of the electric-arc rocket, for the mission considered, would be in the range of 15–30% of initial gross weights. The nuclear fission heat-transfer rocket has higher payload potential at much higher initial accelerations, and other electric propulsion systems have higher specific impulse and payload capacity at the same accelerations. For these reasons, the electric-arc rocket does not appear to be an attractive method of utilizing electric power for propulsion.

The solar heat-exchange rocket has been analyzed quite extensively by Ehricke. In addition to the temperature limitation on the heat-exchanger material, it appears that considerable heat losses will occur in the propellant lines, which must pass from the large polyethylene radiation collectors to the rocket chamber. With suitable high-temperature materials, Ehricke estimates that specific impulses in the range 700–800 sec may be attainable. Using polyethylene collectors, 1 mil in thickness, the specific weight for collectors alone is of the order of 0·35 lb per kilowatt of solar power collected. Allowing for some inefficiencies and rocket chamber weight, it appears that the powerplant specific weight will be of the order of 1 lb per jet kilowatt. With this specific weight, an initial acceleration of 10^{-3} g at a specific impulse of 800 yields a payload of about 5% of gross weight for the Mars mission of Fig. 2(a). If, by close coupling of rocket chamber and heat exchangers, a specific impulse of 1500 sec were attainable, the payload weight ratio would be about 15%. If the specific powerplant weight would be reduced by a factor of 10, to 0·1 lb per jet kilowatt, by using thinner collectors, a payload ratio of about 0·2 might be attainable with an initial acceleration of 10^{-2} g (Fig. 2b). These figures indicate that the solar heat-exchange rocket has less payload potential than the nuclear-fission heat-transfer rocket for the high-payload round-trip Mars mission.

Of the three systems in Group I, therefore, the nuclear fission heat-exchange rocket seems the most promising.

Group II of the nonchemical propulsion methods consists of systems which also rely on heating the propellant, but whose specific impulse is not limited by material temperatures (Fig. 6). The heated propellant is kept away from solid surfaces by confinement with a magnetic (or other) force fields. These systems are all theoretically capable of producing specific impulses in the range of 10,000 sec or higher. If hydrogen is used as propellant, the required temperature to attain such specific impulses is in the range of several hundred thousand degrees. With any propellant operating at these temperatures, the gas will be almost completely ionized, so that magnetic confinement is possible.

Shown in Fig. 6 are four propulsion systems which fall into this group. Two of them (the thermonuclear rocket and the gas-phase nuclear-fission rocket) are capable of self-generation of the electric power required for confinement or ionization. The other two require separate electric power systems and will therefore be limited in initial acceleration to values of the order of 10^{-4} g if payloads near 50% of gross weight are sought.

The feasibility of thermonuclear rockets or gas-phase nuclear-fission rockets has not yet been established. Consequently, little can be said about the probable powerplant weight or mode of operation. The schematic diagrams in Fig. 6 are intended only to suggest principles, and are not necessarily feasible schemes. The values of accelerations in the range 10^{-2} g indicated in Fig. 6 are purely conjectural, based chiefly on the fact that such systems would require no radiator, turbo-generator, and heat-exchanger systems, and would probably operate at higher power levels than electric systems. With an initial acceleration of 10^{-2} g, Fig. 2(b) indicates that a specific powerplant weight of about 0·1 lb per jet kilowatt is needed for payload ratios of the order of 0·5. This specific weight is only 1/30 of the 3 lb per electric kilowatt estimated for the fusion-electric system in Fig. 4, but the power level with $a_0 = 10^{-2}$ is of the order of 100 times that contemplated for electric systems ($a_0 = 10^{-4}$). As previously indicated, lower values of specific powerplant weight are likely for thermonuclear reactors as the power level increases.

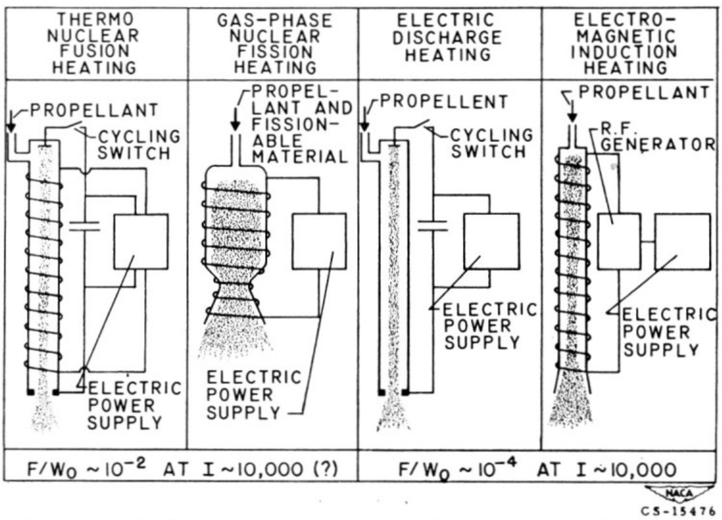


FIG. 6. Group II. Systems using magnetic (or other) containment of very high temperature plasma.

For the thermonuclear rocket, plasma temperatures of the order of 100 million degrees or more are required to maintain a reaction. If the reaction products were used directly as propellants, the specific impulse would then be of the order of 200,000 sec. Such values are much higher than necessary or desirable for missions to the near planets, and can only result in lower thrust for a given power level (Eq. 2). It would therefore be desirable to dilute the reaction products with additional propellant so that the net specific impulse is of the order 10,000 sec. It is not clear how this can be accomplished without quenching the reaction, but a cyclic scheme, or injection downstream of the reaction zone might be possible.

For the gas-phase nuclear-fission rocket, the temperatures need only be high enough to produce the desired specific impulse (of the order of several hundred thousand degrees). To generate the electric power required for containment and ionization, a cyclic operation is envisioned. This cycle might consist of injecting a mass of plasma containing appropriate amounts of fissionable material, compressing the plasma magnetically to critical size, and allowing the expanding plasma to react against the magnetic field to produce sufficient electric power for the next cycle. The excess expansion, over that required for electric power, would take place through a magnetic nozzle and would generate the required jet power. Again, it must be pointed out that no feasibility studies of the thermonuclear or gas-phase fission rockets have as yet been made.

For the other two systems in this group, the thrust–weight ratios indicated are somewhat more solidly based, since both require the separate electric power sources which were discussed earlier. In one case this power is used to heat the propellant with an electric discharge sufficiently strong to pull it away from the walls and to produce temperatures in the range of several hundred thousand degrees. These temperatures can apparently be attained, at least transiently, with an electric discharge, since reports published on thermonuclear research indicate that temperatures in the millions of degrees are being generated in this manner. At the lower temperatures required for this propulsion system, the plasma stability problem is likely to be much less severe than at thermonuclear temperatures.

The electromagnetic induction heating system has the advantage over electric-discharge heating that no electrode heating or erosion problem exists. This system, therefore, looks attractive from the endurance and reliability standpoint. The principal question is whether the propellant can be ionized and subsequently heated to sufficiently high temperature by induction alone. A preliminary analysis of this scheme by Rudolph C. Meyer of the Lewis laboratory indicates that this can be done, and that efficiencies of conversion of electric power to jet power of the order of 70% or higher may be attainable. The analysis indicates also that the oscillating magnetic field produces a net pinch pressure toward the axis, which contains the high-temperature plasma. An additional advantage over electric-discharge heating is that the operation is continuous, whereas discharge schemes will probably require cycling to attain the required temperatures.

The third group of nonchemical propulsion systems differs from the two preceding groups in that the acceleration process is accomplished by electric or magnetic means rather than by thermal energy. There is no particular advantage in this in itself, if it can be done thermally without over-heating problems. However, the problems associated with magnetic

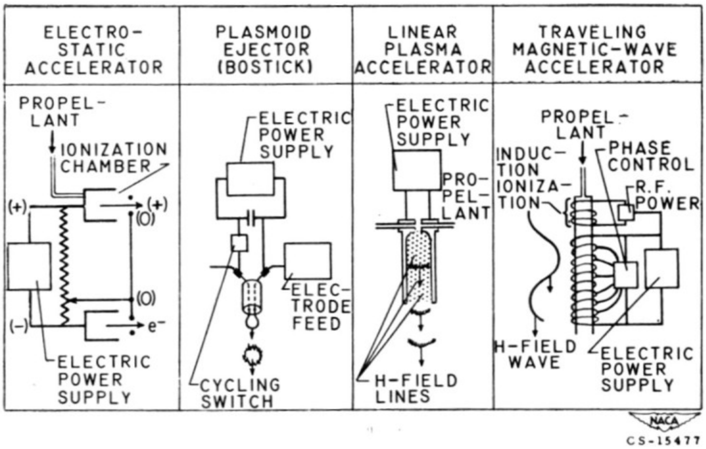


FIG. 7 (a). Group III. Systems using electric or electro-magnetic accelerations.

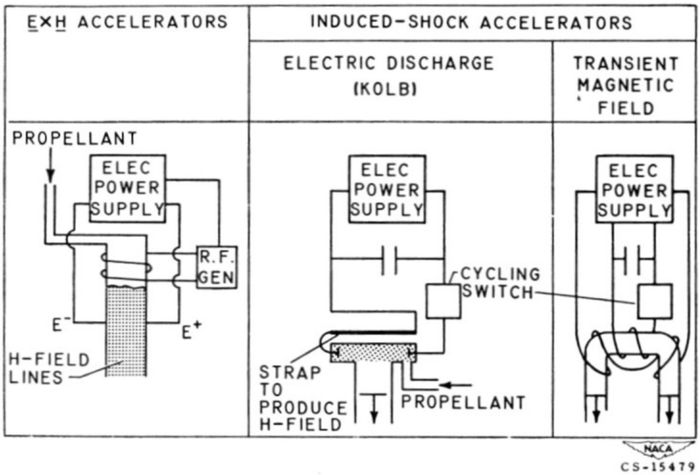


FIG. 7 (b). Group III (*cont.*). Systems using electric or electro-magnetic acceleration.
(All are capable of $I \sim 10{,}000$ sec with $F/W_0 \sim 10^{-4}$.)

or electrical acceleration may be easier to solve than those associated with very high temperature contained plasmas. In the present state of the art, the relative magnitude of the problems cannot be properly assessed.

There are a wide variety of ways in which ions or plasmas can be accelerated by magnetic and electric fields. Some of these accelerators are shown in Figs. 7(a) and (b). All of these accelerators are theoretically capable of producing specific impulses of the order of 10,000 sec and higher. The principal qualities that will determine which of these schemes is most promising are: (1) efficiency of conversion of electric power to jet power, and (2) reliability for long-duration operation. As mentioned earlier, by far the largest portion of the weight of these powerplants, as well as those in the previous groups which used electric power, is contained in the electric power generating equipment. Any inefficiency in converting electric power to jet power is reflected directly in the size and weight of this equipment. If the accelerator is only 50% efficient, the entire powerplant will weigh about twice as much, and the acceleration possible for the space vehicle will be reduced to about half as much, as if the conversion efficiency were near 100%. The weight of the accelerator is, therefore, secondary to the requirement of high efficiency. Of course, there are probably a number of schemes for which the accelerator weight is of the same order as the power generating equipment, in which case the preceding remarks would not apply, but by and large, among accelerators of the same order of weight, the most efficient would be selected even if it were somewhat heavier than the others. This required high efficiency in using the electric power must be accompanied by the capacity to operate for periods of time of the order of a year without serious breakdowns. With these requirements in mind, it may be that accelerators which utilize electric discharges either for the ionization process or for the acceleration process are basically not as attractive for space propulsion as those that manage without discharges.

The electrostatic accelerator is the only one of the systems in Group III that requires separate acceleration of ions and electrons. This is no particular disadvantage if the electrons and ions can be brought together again sufficiently rapidly after acceleration that the space charge is neutralized in a very short distance. If not, the jet area required to produce a given thrust becomes exorbitant, and the accelerator may become too heavy. However, with an efficient ionization technique such as the contact method (as suggested by Stuhlinger) or possibly an induction method, this acceleration method should work quite efficiently and reliably.

The remaining systems are a sampling of the many possible ways that electric fields, magnetic fields, and currents can be combined to accelerate plasma. The first one shown, due to Bostick[(5)], requires no separate ionization technique and uses the metal electrodes as propellant. The propulsive force arises from the curved discharge across electrodes imbedded in an insulator button. This curved discharge produces higher magnetic field, and, therefore, higher magnetic pressure, on the inner curve of the discharge than on the outer. Ejection velocities of the order of 10^7 cm/sec were obtained by Bostick. Further experiments are needed to determine methods of increasing the efficiency.

The next scheme, termed a linear accelerator, has a magnetic field normal to the discharge current. However, it does not require condensors, because the cycling is automatic. The force on the plasma element is normal both to the current and to the magnetic field, so that the discharge accelerates along the rails and is blown off the ends. The arc then restrikes at the narrowest gap and the process is repeated.

In the traveling magnetic wave schemes, the plasma is accelerated by bunching it together in a valley of the magnetic field and accelerating the entire magnetic field pattern to the desired exit velocity. This tends to involve rather elaborate circuitry, but seems to have no basic drawback. Any easily ionizable propellant can be used.

The so-called $E \times H$ system (Fig. 7b) is a steady-flow system which accelerates a plasma along a channel with perpendicular electric and magnetic fields. The scheme looks a little simpler than the travelling-wave scheme, but some difficulty has been encountered in providing a sufficiently strong E-field without breakdown across the channel. A plasma is a fairly good electric conductor, so this difficulty may be inherent in the scheme. If a discharge is obtained, this scheme looks similar to the linear accelerator.

The last two examples shown rely on large surges of current to generate shock waves traveling at speeds corresponding to specific impulse of the order of 10,000 sec. In the first scheme, developed by Kolb(6) the current passes through a heavy strap behind the T of the tube before it discharges between the electrodes. This produces a magnetic field normal to the discharge current, and projects the plasma down the leg of the T. In the second scheme, the large surge of current through the center coil generates enough transient magnetic flux to induce ionization potentials in the gas in the tube. The curved magnetic field lines around the coil provide the magnetic pressure gradients needed to propel the plasma in both directions, and the auxiliary coils turn the plasma in the desired direction to produce thrust.

There is as yet no sound way of deciding which of these, or similar, schemes is likely to be most efficient and reliable for converting electric power to jet power. It seems likely, however, that with so many possibilities, at least one will be found which fulfills the requirements for an adequate space propulsion system.

INITIAL WEIGHT COMPARISON FOR MARS JOURNEY

To illustrate the magnitude of weight savings that are possible with some of the nonchemical systems described in the preceding section, the total initial weights that must be placed in orbit near the Earth to undertake a rather elaborate Mars expedition are compared in Fig. 8. This comparison is for an eight-man expedition with landing and exploration equipment. The basic payload, which includes crew, cabin, navigation and communication equipment, environment control, etc., was assumed to be 50,000 lb. Exploration equipment, including glide landing vehicle and take-off rocket for six men and supplies, was estimated to be about 60,000 lb. Food, water, and oxygen allowance of 10 lb per man-day is included. The total-initial payload, which is basic payload, exploration equipment, and survival supplies, is, therefore, about 200,000 lb. With a four-stage chemical rocket having specific impulse of 300 sec and a structure and motor weight of 0·05 per stage, the total-initial weight required in orbit is about 6·5 million pounds. For the advanced chemical rocket, with specific impulse of 420 sec, the initial weight is reduced to about 2,000,000 lb. A nuclear heat-transfer rocket with specific impulse of 800 sec, a thrust-weight ratio of 1·0, a tankage weight equal to 8% of the propellant weight, and a powerplant weight of about 50,000 lb can accomplish the mission with an initial weight of about 800,000 lb. If the powerplant weight is 20,000 lb the initial weight required is about 600,000 lb.

Some improvement in initial weight is possible, for the same powerplant weights, if a specific impulse of 1400 sec is achieved with an initial acceleration of 10^{-1}, but the initial weight increases if the initial acceleration with $I=1400$ goes to 10^{-2} g.

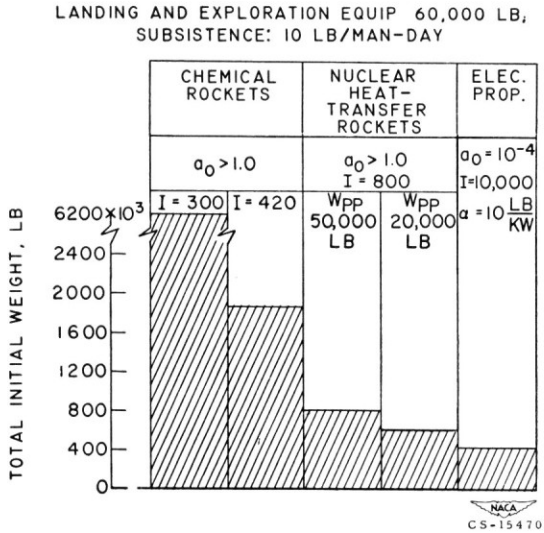


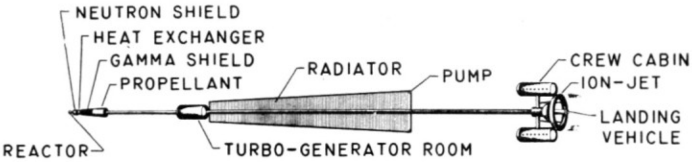
FIG. 8. Initial weight comparison for Mars round-trip eight-man crew, basic payload : 50,000 lb.

An electric propulsion system with initial acceleration of 10^{-4} g and a specific powerplant weight of 10 lb/kW can do the job with about 400,000 lb initial weight.

The nonchemical propulsion systems therefore offer the possibility of weight reductions by a factor of 10–15 over conventional chemical rockets and by a factor of 3–5 over high-energy chemical rockets. When it is remembered that each pound in orbit requires from 10 to 100 lb in launching weight, it seems clear that the development of nonchemical propulsion systems is essential for high-payload space missions.

HYPOTHETICAL SPACE VEHICLE

An example of the type of space vehicle that might be feasible using a nuclear turboelectric propulsion system is shown in Fig. 9. The mission for which this vehicle was designed is similar to that used in the comparison shown in Fig. 8. The components of the nuclear-electric system are scaled in accordance with the size and weight estimate made for the curves of Fig. 4. The design is based on shadow shielding, with rigid tubular separation of the major components. The entire vehicle is to be spun about its axis to provide artificial gravity both for the crew and for the separation of liquid and vapor phase in the radiator.



PAYLOAD FOR 8 MAN CREW			
	LB	LENGTH	600 FT
		GROSS WT	322,000 LB
10 LB/DAY MAN	88,000	POWERPLANT WT	60,400 LB
CABINS	50,000	PROPELLANT	60,400 LB
LANDING VEHICLE	40,000	THRUST	32.2 LB
TOTAL	178,000	POWER	11 MW

EXPLORATION TIME AT MARS, 329 DAYS
TOTAL ROUND TRIP TIME, 1100 DAYS

NACA CS-15474

FIG. 9. Electric spacecraft for round-trip to Mars.

Needless to say, there are some unattractive features in this design, particularly in the magnitude of the radiator required. This size, and the need for light construction make it vulnerable to meteoroid damage. The frequency of penetration can not yet be accurately predicted, but rough estimates based on available meteoroid distribution data indicate that penetrations of the order of several per week may be expected with the thickness of tubing assumed. The weight penalty to insure against these penetrations would be prohibitive. Most penetrations, of course, will be microscopic, but they will nevertheless have to be repaired to avoid significant mass loss of the heat-transfer fluid.

Assuming that the engineering problems associated with such a space vehicle can be solved, the propulsion system is seen to be a very good one indeed on the basis of payload capacity. The weight that must be launched into orbit is less than twice the initial payload weight. Similar weight ratios seem feasible with nuclear heat-transfer rockets. If launching rockets are developed which are capable of placing 100,000 lb of payload into orbit, only four launchings would be required to start the expedition on its way. In terms of current technology, this is a staggering undertaking, but certainly few will deny that such projects are well within the limits of future capability.

CONCLUDING REMARKS

This paper has dealt almost entirely with nonchemical propulsion systems suitable for minimum-energy, or near minimum-energy, high payload missions to the near planets. There are, of course, many other missions for which nonchemical propulsion systems may be suitable. Some of them are in the more distant future, and some may be nearer at hand. Nuclear-fission heat-transfer rockets, for example, may be used for manned Moon expeditions, or for launching a sizeable space platform, before expeditions to Venus or Mars take place. Electric rockets of a lower power level than those discussed herein would be useful for controlling or altering the orbits of permanent satellites.

Reduction of round-trip time to Venus or Mars by following excess-energy trajectories is, to a limited extent, possible both with nuclear heat-transfer rockets and electric rockets. Unfortunately, the reduction in round-trip time is not directly proportional to reductions in transit time, due to the orbital motion of the planets. Consequently, substantial reductions in transit time are needed before significant reductions in total time are attained. With a nuclear heat-transfer rocket, the Mars round-trip time can be reduced to about 400 days, and the Venus trip time to about 180 days without much increase in initial weight for a manned expedition, because the reduction in required payload to some extent balances the required increase in propellant weight ratio. Reduction of the Mars round-trip time to about 160 days, however, requires velocity increments over four times those needed for a minimum-energy satellite-to-satellite mission. With such velocity increments, the nuclear heat-transfer rocket is near its limit, and ratios of initial weight-to-payload weight approach those required for a minimum-energy trip with high-energy chemical rockets. Electric rockets with specific powerplant weights in the range of 10 lb per kilowatt are incapable of completing such missions because the wait time at the destination is insufficient to allow for the slow inward and outward spiral. Reductions in specific powerplant weight by at least a factor of 10 are needed to make electric rockets suitable for these fast missions. The best hope for accomplishing such very-high-energy missions with reasonable payload ratio appears to lie in the development of thermonuclear rockets or gas-phase nuclear fission rockets.

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