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Electric Propulsion

For space exploration we need rockets that are better, not necessarily bigger, than those being developed.

W. E. Moeckel

Propulsion is a means of producing motion, or, more accurately, changes in momentum. Propulsion is also required to maintain momentum in the presence of resistive or retarding forces. These functions of a propulsion system require the expenditure of energy. If this propulsive energy must be carried by the vehicle, as is the case with most systems, then the amount of change in momentum, or the length of the period during which a given momentum can be maintained, is determined by the form and amount of the energy carried along. Ideally, this energy should be in a form which yields the maximum useful propulsive energy per unit of weight.

Other factors, however, such as development and operating problems and costs, or the adequacy of existing systems, may argue against use of systems having the highest possible energy per unit weight. In the case of aircraft, for example, the use of nuclear power would have greatly increased the energy per unit weight, but it involved many difficult and costly problems of development and operation. Its advantages, in terms of much greater range and duration of flight, were overbalanced by these disadvantages, and by the fact that kerosene-burning aircraft were already available with range, speed, and flight durations which were adequate for most applications of interest.

For space missions, if man would be satisfied with exploring the moon, or even with settling a colony on it, there would be little incentive to develop propulsion systems with much greater energy per unit weight than the chemical rockets now in use or under development. Some economies might eventually be realized, it is true, if nuclear rockets or electric rockets were to be developed, but on the other hand, there are many improvements that might be made in chemical rockets, such as using the air through which the vehicle passes to enhance the useful energy, recovering and re-using the launching vehicle, or developing chemicals of higher energy. The net gain in using nonchemical energy sources for spacecraft on lunar missions would therefore be marginal, as it is for nuclear-propelled aircraft on terrestrial missions.

However, if man wants to travel beyond the moon, the need for propulsion systems with more energy per unit weight than chemical rockets becomes painfully obvious. To launch the Mercury spacecraft into orbit required a launching vehicle with about 163,000 kilograms (~ 360,000 lb) of thrust, standing about as high as a seven-story building. To launch the Apollo spacecraft to the moon will require a launching vehicle (a Saturn 5) with about 3.4 million kilograms of thrust, standing about as high as a 30-story building. To send an expedition of seven men to explore Mars and return would require a launching vehicle with thrust of about 68 million kilograms, standing higher than a 70-story building.

Now, no one is seriously proposing to launch an expedition to Mars with a single monstrous launching vehicle. Instead, parts of the interplanetary vehicle could be launched into a low orbit around the earth, and the pieces could be assembled in orbit. Even this, however, becomes a project of tremendous magnitude, even if high-energy reactants such as hydrogen and oxygen are used. It would involve 20 to 40 launchings with the huge Apollo booster (Saturn-5), and the total weight of the orbiting vehicles would be between 2 and 4 million kilograms. The propellant would account for more than 90 percent of the initial weight at launch and also for more than 90 percent of the weight of the orbiting vehicle. To reduce this weight significantly, the useful propulsive energy of the propellant must be substantially increased.

Useful Propulsive Energy

To get the most useful propulsive work out of each bit of propellant, the propellant must be ejected rearward at the highest possible speed, because high rearward momentum of the propellant implies, according to Newton's law, a high forward thrust on the vehicle. Useful propulsive energy is, therefore, the kinetic energy of the ejected propellant, and this energy depends only on the exit velocity.

As we have seen, the chemical energy of combustion is not enough to produce, with a reasonable weight of propellant, the ejection speeds (more commonly called jet velocities) needed for manned interplanetary missions. There are two alternative energy sources: nuclear energy and solar energy.

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Each of these can be used in at least two ways to eject propellant at higher velocities. One method involves direct heating of the propellant; the other, electric propulsion.

Propellant-Heating Systems

With nuclear or solar energy, considerably more heat can be added to the propellant than is possible with chemical energy. A major development program is, in fact, under way to produce a nuclear rocket in which hydrogen is passed through tubes in a nuclear reactor. A rocket propelled by hydrogen heated by solar energy is also a possibility, although the collector needed to concentrate enough solar energy (sunlight) for the purpose would be huge. In these heating methods, the limit beyond which no more energy can be added to the propellant is determined by the temperature at which the heating tubes soften or melt. Hydrogen is the best propellant to use in such a system because it has the lowest molecular weight, and therefore the greatest energy per gram when heated to a given temperature.

Much better performance is possible with such systems than with chemical rockets. The energy in each bit of propellant can be quadrupled, and the exit velocity of the propellant can be about doubled, before the temperature limit of the heater is reached. This doubling of the jet velocity reduces the manned Mars mission to somewhat more manageable size. Instead of having to launch 2 to 4 million kilograms into orbit, only 700,000 to 1½ million kilograms would be needed. The required number of Saturn-5 boosters would be correspondingly reduced by a factor of 3. Rockets propelled by hydrogen heated by nuclear or solar energy are therefore extremely attractive propulsion systems. Not only are they attractive for the manned Mars mission; they would also be useful in carrying the large payloads needed for establishing a permanent base on the moon.

Propellant Velocities Achieved with Electric Propulsion

The ejection speed of the propellant is limited, in the case of chemical rockets, by the energy of the chemical reactions, and in the case of the rockets propelled by heated hydrogen, by the temperature of the heater materials. Electric propulsion is one means of avoiding these limitations. If the atoms or particles of the propellant are electrically charged, all that is needed to accelerate them is a voltage. By applying the proper voltage, the particles of the propellant can be accelerated to any exit speed desired. Or, if the propellant is changed into an electrically conducting gas (or plasma), it can be accelerated, through application of an electric current and a magnetic field, in the same way that solid conductors are moved in an ordinary electric motor. The possibilities for adding as much energy as we may wish to the particles of the propellant are many and various.

It would seem, then, that the amount of propellant needed for space missions can be reduced to very small amounts indeed. This is quite true, but there is just one problem: equipment for generating electric power must be carried along, and this can be quite heavy.

Thrust and Power Relations

Three equations are needed to show what happens. The first is the equation for thrust T:

$$T = \frac{mv}{g} = \dot{m}I$$

where m is the rate of flow of the propellant, v is the ejection velocity of the propellant, and g is the gravitational acceleration constant (980 cm/sec^2), which relates mass to weight. The quantity I is the *specific impulse*—perhaps the most common "figure of merit" for a rocket. It is a measure of the amount of thrust produced by a given rate of flow of the propellant and is directly proportional to the ejection velocity.

The second equation is that for the power P in the ejected propellant:

$$P = 1/2\, \dot{m}v^2 \qquad (2)$$

Combining Eqs. 1 and 2 gives the third equation of interest:

$$P = \frac{v}{2g} \times T = 1/2\,(I \times T) \qquad (3)$$

These equations show that the required power (Eq. 2) goes up as the square of the propellant velocity, while thrust (Eq. 1) goes up only as the first power of propellant velocity. Consequently, for a fixed power (Eq. 3), the thrust must go down as the ejection velocity (or the specific impulse) goes up.

This result can also be stated as follows: Although the required weight of propellant can be reduced by increasing the jet velocity indefinitely, the electric power required for a given thrust goes up, and therefore larger and heavier power-generating systems are required. What must be done is to find the ejection speed (specific impulse) for which the *sum* of the weights of the propellant and the power generating equipment is lowest. This ejection velocity will yield the greatest total energy per unit weight and the least total weight for the mission. It turns out that, for the missions most likely to be undertaken and the power plants most likely to be used in the future, propellant velocities of 1 to 10 million centimeters per second are desirable. These velocities correspond to specific impulses of about 1000 to 10,000 seconds, as compared to the specific impulses of 450 seconds typical of high-energy chemical rockets, and the 900 seconds which appear to be attainable with nuclear rockets.

Obviously, the weight of the electric power system plays a crucial part in determining the extent to which mission capability can be increased by removal of limitations on propellant velocity. An increase in the useful payload can be achieved only if the weight added in the propulsion system is less than the weight saved in propellant. Mission studies for a manned expedition to Mars have shown that an electric rocket weighing less than about 10 kilograms for each kilowatt of jet power produced would provide a net gain in useful load-carrying ability over nuclear rockets. This ratio of rocket weight (in kilograms) per kilowatt of jet power is called the *specific weight* of the electric propulsion system and is one of its most important figures of merit. Detailed studies indicate that it should be possible to build electric-propulsion systems with specific weight as low as 3 kilograms per kilowatt in the sizes (greater than 1000 kilowatts) needed for manned interplanetary missions. Possibly even lower specific weights may be attainable with novel types of power generators now only in the conceptual stage.

Need for Electric Propulsion

With an electric propulsion system weighing about 5 kilograms per kilowatt, a manned expedition equipped to land on and explore Mars would require an initial vehicle weight (that is, the weight after the vehicle is launched into a low earth orbit) of only about

225,000 to 450,000 kilograms. (A fully loaded jet liner, by comparison, weighs about 145,000 kilograms.) A conceptual design for such a vehicle is shown in Fig. 1. The vehicle could be launched into orbit by two to four of the Saturn-5 launching rockets now under development. Therefore, if suitable electric propulsion systems can be developed, there will be no need to build launching vehicles larger than, at most, four times the size of Saturn-5. With some capability of assembly in orbit, Saturn-5 would be quite adequate for launching manned interplanetary missions.

In addition to greatly reducing the size and cost of the boosters for full-scale, manned planetary expeditions, development of light-weight and reliable electric propulsion systems would simplify the scientific probing of all regions of our solar system by unmanned spacecraft. This is illustrated in Table 1, where the payload that can be carried to various destinations is given for three space vehicles: one propelled by electric rockets (of two different specific weights), one, by a nuclear rocket, and one by chemical rockets with hydrogen and oxygen as the propellant. The data in Table 1 show that the vehicles propelled by nuclear and chemical energy, even though they are much heavier than the electrically propelled vehicle, could successfully carry out only some of the indicated missions. The electrically propelled vehicle has enough payload for all of the missions shown and could be launched by the medium-sized Saturn-1B rocket now under development, while the vehicles propelled by nuclear and chemical energy would require a much larger booster. There is, therefore, a strong incentive to develop an electric propulsion system in the weight and power range indicated in Table 1, in order to reduce the number and size of the vehicles that must be developed for sending scientific probes throughout the solar system.

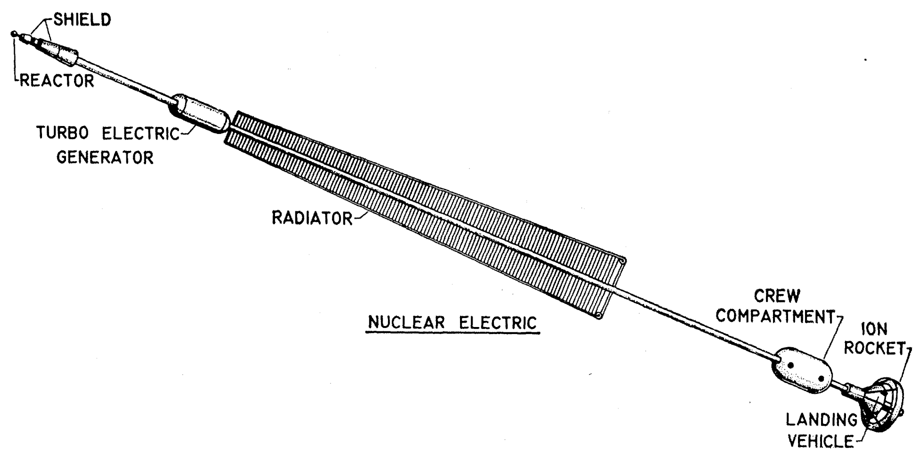

Fig. 1. Conceptual design of a space vehicle, with nuclear turbo-electric propulsion system, for a manned Mars mission.

Development of Electric Propulsion Systems

What is the likelihood that electric propulsion systems in the sizes and weight ranges needed can be developed? As yet, no clear guarantee of success is possible. Most engineers and scientists concerned with the problem believe that success will ultimately be achieved, but the estimated times required vary from a few years to more than 15. The major unresolved problems relate to the development of suitable systems for generating electric power. The present stage of development of the other major component, the thrustor, in which electric power is used to eject the propellant, is fairly promising. There is considerable confidence that thrustors with relatively high efficiency, low weight, long lifetime, and reliability for interplanetary missions can be developed in the next few years, even though there is still a great deal to be done before this is accomplished.

Thrustors

The basis for the rather general belief that suitable thrustors can be developed is in the fact that a number of ion accelerators with characteristics similar to those needed for space missions have been conceived, designed, built, and extensively tested in vacuum chambers. Their operation and performance, and the attendant problems, are consequently quite well understood. The most successful of the ion thrustors so far developed is one conceived and investigated at the NASA Lewis Research Center (*1*). This thrustor (Fig. 2) ionizes the atoms of the propellant by bombarding the propellant vapor with electrons in the ionization chamber. A weak magnetic field is provided, in the ionization chamber, to make the electrons spiral around on their way to the chamber wall and thus to increase the probability that they will collide with atoms of the propellant. The resulting ions are extracted from the chamber by means of an accelerating grid, to which an appropriate voltage is applied. A second electron emitter, in the ion beam, provides the electrons needed to neutralize the ion beam so that is does not break apart as a result of mutual repulsion of like charges, and so that the vehicle will not accumulate a net negative charge. Experimental efficiencies in converting electric power into jet power currently range from 60 to 75 percent at specific impulse of 4000 to 8000 seconds.

With other ion thrustors, based on a technique called contact ionization (the technique just described is called electron bombardment ionization), somewhat lower performance levels have been achieved. With these thrustors, cesium is used as the propellant, because cesium is the element most easily ionized. If cesium is passed through a hot, porous tungsten material, the cesium atoms lose an electron to the hot surface of the tungsten, thereby becoming ionized and subject to electrostatic acceleration. Thrustors of this type are being developed for the Air Force and NASA by Electro-Optical Systems, Inc., and by Hughes Research Laboratories, respectively.

It is planned that brief space-flight tests of these thrustors will be made within a year to make sure that performance in space is the same as performance in laboratory vacuum chambers.

Although the use of ion thrustors for primary propulsion would require large electric power systems, small units, requiring only a few kilowatts of power, are under development to serve another function—that of maintaining the position and orientation of communication and weather satellites, and satellites for making astronomical observations, for very long periods. Ion thrustors are well suited for this function,

Table 1. Data on payloads for space missions. [After Spencer *et al.* (5) and Moeckel (6)]

Mission	Payload (kg) *			
	Electric rocket (initial weight, 11,000 kg; power plant weight, 2,800 kg)		Nuclear rocket (initial weight, 34,000 kg)	Chemical rocket (initial weight, 22,000 kg; propellant, H_2–O_2)
	Specific weight: 6 kg/kw	Specific weight: 12 kg/kw		
Place payload in 500-mile Mars orbit; trip time, 230 days	4600	3000	8400	4000
Jupiter flyby; trip time, 500 days	5500	3700	6800	3200
Send payload 30° out of ecliptic plane; trip time, 232 days	3900	2000	1800	No mission
Pluto flyby; trip time, 1100 days	2800	900	No mission	No mission
Place payload in 2000-mile Saturn orbit; trip time, 1000 days	2300	500	No mission	No mission
Place payload in 2000-mile Jupiter orbit; trip time, 900 days	1800	136	No mission	No mission

* "No mission" indicates that payload is too small

because of their small thrust and very low rate of consumption of propellant.

Other types of electric thrustors are currently being studied, which, it is hoped, will either be more efficient than ion thrustors in performing the conversion to jet power at the lower jet velocities (10^6 to 5×10^6 cm/sec) or will increase the thrust attainable with a given size and weight of thrustor. These other types are (i) electrothermal thrustors, in which electric power is used to heat the propellant; (ii) plasma thrustors, which generate and accelerate an electrically conducting gas; and (iii) colloidal-particle thrustors, which operate like ion thrustors, but with charged particles that are much heavier than atomic ions.

Atomic ion thrustors are less efficient in the lower range of jet velocities because a certain fixed amount of energy is required to strip an electron from the atom. If this nonuseful energy is large compared with the useful energy (the ejection velocity), then the efficiency of converting electric power into jet power is low. At the lower ejection velocities required for some missions (specific impulses of 1000 to 5000 seconds), the efficiency of ion thrustors is therefore rather low, whereas at higher ejection velocities the useful energy becomes much greater than the ionization energy and the efficiency is higher.

At the lower jet velocities the efficiency can also be increased by (i) reducing the requirement for ionization energy, or (ii) increasing the amount of useful energy per charged particle, at a given velocity, by increasing the mass of the charged particle (this is the approach used in the colloidal-particle thrustor). Both approaches are being investigated. The electrothermal devices require only enough ionization energy to carry the arc current. These devices heat the propellant and force it out through a nozzle, as in the usual chemical or nuclear rocket. This heating yields specific impulses only in the lowest range of interest (from 1000 to perhaps 2000 seconds). These specific impulses are higher than those obtainable in a solid-core nuclear rocket because the temperature generated in an electric arc discharge can be much higher than the temperatures tolerated by solid tubes or chamber walls.

Although the electrothermal thrustors require only partial ionization of the propellant, other power losses limit the efficiency. Some of the energy goes into breaking up the atoms and molecules of the propellant, and some goes into producing more ionization than is needed. Consequently, although electrothermal thrustors are usable, their efficiency in this lower specific-impulse range is not as high as the efficiency of ion thrustors in the higher specific-impulse range.

A considerable amount of developmental work on electrothermal thrustors is being sponsored by NASA and the Air Force, and it seems likely that operational units can be developed within a few years.

The other possible improvement over atomic-ion thrustors—production of more thrust for a given size or weight

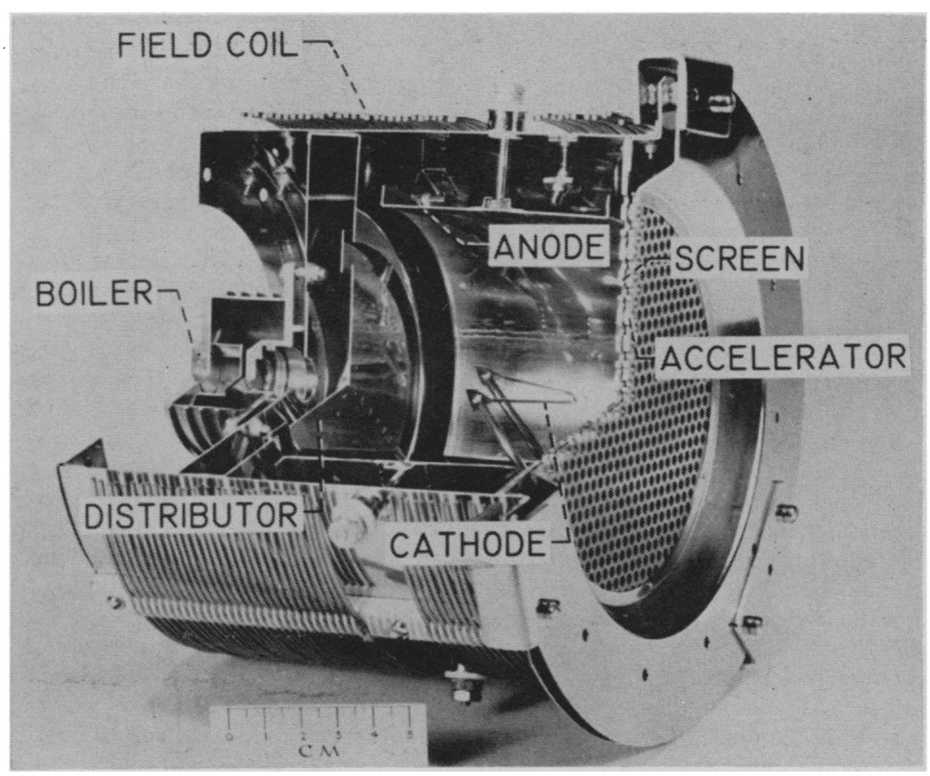

Fig. 2. Cutaway photograph of electron-bombardment ion thrustor. Using about 1 kilowatt of power, this unit produces a thrust of 2.5 grams at specific impulse of 5000 seconds.

of thrustor—may be achievable with the plasma thrustors. (The electrothermal thrustors already have a high thrust density in the lower specific impulse range). The limitation arises, in the ion thrustors, from limitations on the ion current attributable to space-charge. This current can be increased only by increasing the voltage or by reducing the distance between the accelerating electrode and the ion source. The voltage is approximately fixed by the desired ejection velocity (or specific impulse), and the accelerator spacing cannot be reduced below a certain limit without encountering either electrical breakdown or severe design problems that arise from the need to maintain accurate spacing in the presence of thermal and other stresses. Another limitation on the thrust-density ratio—one which may be even more severe than the space-charge limitation—is imposed by the lifetime requirement. As the current density is increased, there is greater impingement of ions on the accelerator electrode, and erosion of the electrode increases. Present estimates indicate a limit, for thrust, of about 40 milligrams per square centimeter (1 oz/ft^2) of exit area of thrustor at a specific impulse of 5000 seconds. Plasma thrustors, since they operate on the principle of moving an electrically conducting but neutral gas, are not subject to the space-charge limitation, and they require no accelerating electrode. Thus, with plasma thrustors a higher thrust-density ratio may be attainable. At present, however, none of the proposed plasma thrustors has achieved sufficiently high efficiencies in continuous operation, and much research is required before their usefulness can be demonstrated.

Power-Generation Systems

For use with electric thrustors to form a primary propulsion system for interplanetary missions, a system for generating electric power must fulfill the following requirements: (i) it must operate in space; (ii) it must have an operational lifetime of 1 year or more; (iii) its specific weight must be low (preferably less than 10 kg/kw); and (iv) its total output of electric power must be in the range of several hundred to several thousand kilowatts.

The first requirement immediately poses two rather severe problems: (i) there is no way to get rid of waste heat in space except to radiate it away, and (ii) there are micrometeoroids in space capable of puncturing thin, lightweight containing walls.

Problem (i) is bothersome because one of the most straightforward ways to generate large amounts of electric power in space is with a nuclear turboelectric system (see Fig. 1). In such systems the turbine extracts from the working fluid only a fraction of the heat added by the reactor. For steady-state operation, the remaining fraction of the heat picked up must be disposed of before the fluid can recirculate through the system. For high efficiency, it is desirable to have the turbine-inlet temperature as high as possible and the turbine-outlet temperature as low as possible, so that as much of the heat as possible is extracted in the form of useful work. When it is necessary to radiate away the remaining heat, the size of the radiator goes up as the inverse fourth power of the turbine-outlet temperature but goes down only linearly with reduction of this temperature. Consequently, there is an optimum turbine-outlet temperature at which the size of the radiator is a minimum. This optimum temperature is approximately three-fourths of the turbine-inlet temperature, and this ratio corresponds to an overall efficiency in converting heat into electricity of less than 25 percent. Thus, about 75 percent of the heat generated by the reactor must be radiated away. To achieve reasonable specific weights for the entire system, it is therefore necessary to achieve the highest possible temperatures. For example, to achieve total specific weights of less than 10 kilograms per kilowatt, the temperature of the radiator should be higher than 900°K and the temperature of the reactor outlet should be higher than 1200°K. Furthermore, the working fluid should be condensable into a liquid at the temperature of the radiator, so that radiation will take place at a uniformly high temperature. Moreover, if the working fluid is a liquid it can be pumped around the system, whereas, if it is a gas, it must be circulated with a gas compressor. Both of these features effect substantial savings in weight.

The only good prospective working fluids appear to be the alkali metals, such as sodium and potassium, which condense and vaporize in about the right temperature and pressure range. These fluids, however, are highly corrosive under some circumstances, so that their use, for reducing the radiator to reasonably small size, introduces difficulties in the matter of finding suitable containment materials.

Problem (ii) is related to problem (i) in that the radiator, because of its large area, is particularly vulnerable to puncture by micrometeoroids. The working fluid cannot be allowed to leak out, because the amount that can be carried along is limited. Fortunately, recent data on hazard of micrometeoroids indicate that the radiator-tube thicknesses that are needed for strength may provide enough protection to make the probability of puncture by micrometeoroids negligible. If this conclusion is verified, then no armor or increase in thickness may be required.

Requirement (ii)—that the generator be capable of continuous operation for a year or more—imposes severe problems of reliability and durability for a nuclear turboelectric system that operates at high temperatures and contains corrosive materials and high-speed rotating machinery. The problem is perhaps less severe for manned than for unmanned missions, since the crew could, presumably, make minor repairs in the event of noncatastrophic failures. However, the corrosion and wear problems must be solved before the practicability of such systems is clear.

The long-time continuous operation is required for electric propulsion primarily because the specific weight, in terms of kilograms per kilowatt of jet power, is much higher for electric rockets than for chemical or nuclear rockets —in fact it is of the order of 1000 times as high. Thus, even though the total useful energy per unit weight is higher with electric propulsion, the energy must be released at a much lower rate if the propulsion system is to be of reasonable weight. This low rate of energy release, which is the jet power (Eq. 3), means that the thrust that can be generated per unit weight is very low, since the jet velocity must of course be kept high. Consequently, whereas in chemical or nuclear rockets the thrust is applied in relatively short bursts of very high power, in electric rockets a relatively low thrust must be applied continuously for long periods. Typically, the thrust is of the order of 2000 to 10,000 times lower than the weight of the vehicle to be propelled. This means that the thrust must be applied 2000 to 10,000 times longer than in a system in which thrust is equal to vehicle weight. For a round-trip mission to Mars, for example, with a total round-trip time of about 450 days, with electric propulsion the thrust would be applied continuously throughout the trip except for the period when the planet was being explored.

Thus, by decoupling the power source from the thrustor, the restrictions on propellant velocity and specific impulse are eliminated, but at the cost of an increase in specific weight and in the required operational lifetime of the propulsion system.

At present the only power-generation system under development which is of possible importance for major missions with electrically propelled vehicles is the SNAP-50 system, jointly sponsored by the Air Force and the Atomic Energy Commission. This system, if successful in achieving its goals, could provide the electric propulsion power needed for the space-probe vehicle represented by the data of columns 2 and 3 of Table 1 and would, consequently, be extremely useful.

The SNAP-50 system is a nuclear turbo-electric system, with potassium as the working fluid for the turbine-radiator system. Liquid lithium, however, is to be used to transfer the heat from the nuclear reactor to the potassium. The two-loop system avoids the problems of boiling and vaporizing in the reactor itself, since lithium remains liquid at the highest temperatures of the cycle.

In addition to the SNAP-50 program, extensive research on high-temperature materials, on corrosion, on bearings and seals, and on a multitude of other problems related to the generation of power by the nuclear turbo-electric system are being sponsored by NASA, the Air Force, and the Atomic Energy Commission. These advanced technological programs should lead to power-generation systems which meet the four requirements stated at the beginning of this section.

Other Methods of Power Generation

In the meantime, a number of promising improvements and alternative methods of meeting the electric-propulsion power requirement are being studied. In one method, which is closely related to the liquid-metal nuclear-electric system described, a nonreacting gas, such as argon or helium, is used instead of the liquid-and-vapor cycle. This method eliminates problems of corrosion at high temperatures but requires a larger radiator for the same maximum temperature of the cycle. The overall specific weight is therefore higher than the liquid-and-vapor cycle.

A promising method which eliminates all rotating equipment is the so-called magnetohydrodynamic method of power generation. In this method the turbine and generator are replaced by a duct surrounded by an electromagnet. The hot vapor of the working fluid is made to be electrically conducting and is forced through the duct and its magnetic field. The conductive vapor serves the function of the solid conductors (copper wires) in an ordinary electric generator, and a voltage and current are produced in the vapor as in the copper wires. The resulting generated power is thus obtained with a system that has no moving parts except the working fluid itself. After condensation in the radiator, the fluid is recirculated through the reactor by an electromagnetic pump which, again, has no moving parts.

If this method turns out to be successful, the problems of developing bearings and seals for high-temperature, high-speed turbines and generators will have been eliminated. Furthermore, it might be possible to operate the system at somewhat higher temperatures than are feasible with the turbo-generator, thereby reducing the specific weight. Such low specific weights appear attainable, however, only if superconducting electromagnets can be used to provide the required magnetic field.

In another method for eliminating rotating equipment, the reactor heat is converted directly into electricity. This can be accomplished by building thousands of thermionic cells into the reactor itself. A thermionic cell consists basically of a hot element and a cooler element. Electrons are boiled off the hot element at the temperatures attainable in the reactor. These electrons have enough energy to travel against a small voltage to the cold element. The cell thereby produces an electric current at a small voltage (of the order of 1 volt). By connecting many of these elements together, a sizable amount of power can be generated.

Among the difficulties in this method is the difficulty of designing a reactor containing thousands of these cells (each with electrical connections), with flow of coolant over the cooler element and very close spacing between elements. The problems appear formidable indeed. Mounting the thermionic cells outside the reactor and carrying heat to them by means of a liquid metal is an alternative approach but one that is less promising from the standpoint of reducing specific weight. A successful magnetohydrodynamic system would probably be preferable, because it would be more rugged and would have a sizable voltage output.

The systems that have been discussed are all modifications (possibly improvements) of the basic nuclear-reactor thermodynamic-cycle approach to power generation. It is possible that solar power could be substituted for power from a nuclear reactor in these systems. There appears to be no real advantage in such an approach; the same problems—of high temperature, for example—would exist. The only benefit would be the elimination of the nuclear reactor, with its need for radiation shielding. New problems would be the need for a very large, extremely lightweight solar-energy collector, which would concentrate and focus the solar rays on a heater. Little, if any, decrease in specific weight appears likely with this approach.

There is, however, one possible method of using solar energy that could eliminate the need for radiator, turbine, generator, pumps, working fluid, and heat exchangers as well as for nuclear reactor and shielding. This method would use thin-film photovoltaic cells. Solar cells have been used extensively on a wide variety of satellite and space probes, but those used have been much too heavy and expensive to be considered as a means of generating the hundreds of kilowatts needed for electric propulsion. Recently, however, it has been found that solar cells can be made by coating thin solid layers of materials such as cadmium sulfide on a suitable thin substrate (*2*). The resulting cell is flexible, quite rugged, and resistant to radiation damage. As in other photovoltaic cells, a voltage is generated simply by the action of light rays on the cell surface. The present specific weight of these cells is about 35 kilograms per kilowatt; this is still somewhat too high for use in the cells of major electric propulsion systems, but there are indications that specific weights of 10 kilograms or less per kilowatt may be attainable. If such weights are attained, there will still be the problem of mounting the cells on a suitable inflatable structure to be unfurled after launch into space. Such a structure would have to be quite large—of the order of 46 square meters (500 ft^2) per kilowatt of electric power produced—but this problem appears simple relative to those presented by the nuclear turbo-electric system.

Although it seems probable that, with most of the systems described; specific weights as low as about 5 kilograms per kilowatt can be achieved, it does not seem likely that the value can be reduced below about 2.5 kilograms per kilowatt with any of them. Two systems have been suggested with which, at least conceptually, specific weights below 1 kilogram per kilowatt could be achieved. These are a radioisotope cell system (*3*) and a variable-temperature dielectric system (*4*). Both of these systems require thin-film techniques. Both methods appear worthy of further investigation. Possibly other methods based on new uses of thin films will be conceived in the future.

Conclusion

The development of electric-propulsion systems with sufficiently low specific weights and sufficiently long lifetimes for interplanetary missions is difficult, but with good engineering judgment and stubborn perseverance, success appears likely. The ultimate benefits, in terms of man's ability to travel greater distances through space in less time and with vehicles of less initial weight and size than those currently visualized, appear to warrant an effort at least as great as that now being made to develop such systems.

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Ultrahigh Vacuum Instrumentation

The general subject of the measurement techniques associated with ultrahigh vacuum are reviewed.

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In the past decade there has been an enormous increase in the use of ultrahigh vacuum equipment, at first in the laboratory and more recently in commercial production facilities (*1*). For many applications, pressures of the order of 10^{-9} atmosphere—that is, approximately 10^{-6} torr (1 torr=mm-Hg =1/760 atm)—are entirely adequate. Even at these pressures, surfaces that are originally free of absorbed gases become covered with absorbed gas in only a few seconds. With pressures of 10^{-9} torr and lower, significant recontamination of the surface does not occur for several hours. Pressures of 10^{-8} torr and below are generally called the ultrahigh vacuum range.

Ultrahigh vacuum is indispensable in many investigations of the reaction between a surface and a gas, or of the properties of the surface itself. The success of an experiment in the laboratory depends to a large extent on knowledge and control of the variables affecting the experiment. In investigations of properties of surfaces, for example, the surfaces are often contaminated by unknown amounts of unknown impurities. When two surfaces are rubbed together in air, one can ask to what extent the surfaces are in contact or to what extent the absorbed impurities are in contact. In many investigations of dry sliding, the surface films may act as lubricants, and it is clear that sliding in the absence of absorbed films would lead to entirely different results. Pressures in outer space are very low, and any clean surfaces formed by frictional movement remain free of absorbed layers.

Since clean surfaces tend to seize,

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