NASA TECHNICAL MEMORANDUM

NASA TM X-52789

NASA TM X-52789

LOW THRUST (HIGH-IMPULSE) NUCLEAR AND ELECTRIC SPACE PROPULSION SYSTEMS

by W. E. Moeckel
Lewis Research Center
Cleveland, Ohio

TECHNICAL PAPER proposed for presentation at
Sixth Propulsion Joint Specialists Conference sponsored by
the American Institute of Aeronautics and Astronautics
San Diego, California, June 22-26, 1970

LOW THRUST (HIGH-IMPULSE) NUCLEAR AND ELECTRIC SPACE PROPULSION SYSTEMS

by W. E. Moeckel

Lewis Research Center
Cleveland, Ohio

TECHNICAL PAPER proposed for presentation at

Sixth Propulsion Joint Specialists Conference sponsored by
the American Institute of Aeronautics and Astronautics
San Diego, California, June 22-26, 1970

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

LOW THRUST (HIGH-IMPULSE) NUCLEAR AND ELECTRIC SPACE PROPULSION SYSTEMS

by W. E. Moeckel*

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio

E-5646

ABSTRACT

This paper briefly reviews the research and development status of a class of space propulsion devices commonly called low thrust systems, and discusses their potential performance capability for space missions. Some conclusions are drawn concerning the most expeditious way to realize the potential benefits of these systems for space exploration.

INTRODUCTION

The common classification of space propulsion systems as "high thrust" or "low thrust" is really not very meaningful. Mission analysts recognize that what we really mean is "high acceleration" or "low acceleration," and that one can usually get away with using conic-section trajectories with the former, but must generally resort to numerical trajectory integration with the latter. The the uninitiated, however, these names may imply an unwarranted deficiency in some systems. "High thrust" sounds fine, whereas "low thrust" sounds weak and undesirable.

To avoid such unfair discrimination, a better classification is one based on the main limitation of each class. The "high thrust" systems are unable to produce high propellant exhaust velocities (or specific impulse). A good name for such systems is therefore "specific-impulse limited" systems. This name is especially suitable because the attainable

*This report is a portion of a paper presented by David S. Gabriel at the Sixth Propulsion Specialist Conference, San Diego, California, June 15-19, 1970.

TM X-52789

specific impulse is the parameter that mainly determines the mission capability of such systems.

The reason that "low thrust" systems cannot produce high acceleration is that they have removed the restriction on attainable specific impulse, with the penalty that the propulsion system mass per unit of jet power (specific mass) is greatly increased. Hence, "specific-mass limited" is a good designation for these systems because attainable specific mass is the main parameter that determines the mission capability of these systems.

If these names are too cumbersome for common usage, an alternative designation can be based on the fact that power in the exhaust jet is proportional to both thrust and specific impulse. For a given power level, therefore, thrust and specific impulse are inversely related. Hence we can with good justification replace "high thrust" and "low thrust" with "low impulse" and "high impulse." Or we can simply call them Type I and Type II as in reference 1. Using these more palatable classifications, this paper is concerned with "high impulse" or Type II space propulsion systems.

There are two main candidates for discussion in this class: electric rockets and thermonuclear fusion rockets. The electric rockets can be considered in two subclasses: solar-electric rockets and nuclear-electric rockets. Some other high-impulse concepts, such as mass-annihilation photon rockets, are not sufficiently defined to estimate even in a preliminary way the potentially attainable specific masses.

SOLAR AND NUCLEAR ELECTRIC ROCKETS

Of the two power sources for electric propulsion, solar power is by far the most advanced toward application. In fact, one solar-electric propulsion system - the SERT II system - is currently operating in space in a long-duration test of an ion thruster. The SERT II solar-cell array is not sufficiently lightweight for mission use, but arrays with specific mass low enough to be interesting for several electric rocket missions are under active development (ref. 2).

Nuclear-electric rockets, on the other hand, do not yet exist except in the minds and publications of systems and mission analysts. Furthermore, there is no national program to develop such a system. Nuclear-electric rockets, therefore, remain in the purgatory of "advanced technology." This is in contrast to the nuclear rocket program, which has recently received Presidential blessing, and is thus elevated to the paradise of a flight development program.

Since no decisions have yet been made on the form that a nuclear-electric rocket will take, or indeed whether such a system should even be developed, many alternatives with respect to energy source, conversion method, and thruster types are still being studied. Perhaps more than one combination of these components will be found desirable, depending on the required power level and application. It is worthwhile to review these potential applications of electric propulsion in order to see where both nuclear and solar energy sources fit in.

The most imminent applications for electric rockets are for satellite orbit control (station keeping) and space vehicle attitude control. The power requirement is very small (a few hundred watts at most) and will usually be supplied by existing on-board electric power supplies. Thrust levels are a few tens of micronewtons (1 newton = 0.225 pounds force), and the thruster efficiency and specific impulse are rather unimportant, so long as they permit reduction of propellant mass to values much lower than those used by cold gas jets or small chemical thrusters. No special need for solar or nuclear power exists in this application.

The next level of power for electric rocket applications is in the range 5 to 50 kilowatts, corresponding to primary propulsion of small interplanetary exploration vehicles (including fly-by and orbiter missions) to all parts of the solar system. These electrically-propelled vehicles have masses between 500 and 5000 kilograms, and would require specific masses of about 30 kilograms per kilowatt of electric power (or less) to be of real interest. At present, large lightweight solar cell arrays with specific mass less than 15 kilograms per kilowatt at the Earth's orbit are under development (ref. 2). These would be suitable for missions to the nearer planets (perhaps as far as Saturn). However, nuclear power systems with the required specific mass would be more desirable, not only for the more distant missions, but also for near-planet missions, because they are more compact and are independent of Sun orientation and planetary shadowing.

Another mission in this power range, for which electric propulsion looks attractive, is the raising of satellites from low orbits to higher orbits, particularly to the 24-hour Earth-synchronous orbit. Solar cell arrays are currently being considered as power sources in this application also, but a nuclear power source with comparable specific mass would again be preferred because of the complicated rotations required to maintain tangential thrust while simultaneously keeping the solar array facing the sun.

The next level of power is associated with unmanned planetary exploration vehicles with high-resolution real-time television and/or radar mapping. These missions could also include small landing vehicles. The high power requirements for television transmission and mapping of distant planets matches the power needed to propel these larger vehicles. If laser communication systems become feasible, this power need will be reduced, but the payload advantages over chemical systems would remain. Power levels of a few hundred kilowatts, with spacecraft mass of 5,000 to 30,000 kilograms are typical for these missions. These power levels would be very awkward to achieve with solar cell arrays, particularly for the distant planets. As for the lower power level, specific masses less than 30 kilograms per kilowatt would be required.

The highest levels of power are those for manned planetary exploration missions. Here we need power levels of the order of 10 megawatts to propel vehicles with total mass of the order of 500,000 kilograms. For trips to the near planets, the specific mass of the powerplant should be 10 kilograms per kilowatt or less to provide significant gains for use in combination with, or in place of, nuclear rockets.

Until the time arrives when large planetary and lunar bases are under active consideration, the major use for space power levels greater than about 50 kilowatts seems to be for electric propulsion. Consequently, research and development programs for larger space power levels should be directed toward achieving the specific masses, operating lifetimes, currents and voltages required for the contemplated electric propulsion applications.

Electric propulsion systems can be considered to consist of two major parts: the power generations system and the thruster system. The former makes up most of the mass of the system and therefore determines the minimum specific mass of the entire propulsion system. The thruster system determines, through its efficiency of conversion of electric power into jet power, how nearly this minimum value can be attained. The current status of each of these subsystems will be described, and future prospects for improvement will be outlined.

Electric Thruster Systems

For satellite attitude control and station keeping, several types of small electric thrusters are being developed and some have already been flow on satellites (ref. 2). For this low-power application, niether high efficiency nor high specific impulse are critical; one merely needs long lifetime, with low propellant consumption relative to chemical rocket or cold-jet systems. Efficiencies are in the range of ten percent or less.

For primary electric propulsion both high efficiency and the achievement of an optimum specific impulse are important. Only the electron-bombardment ion thruster with mercury propellant has been developed to flight status. Such thrusters are now undergoing flight test on the SERT II spacecraft (fig. 1) which was launched into a polar orbit on February 3, 1970. The vehicle has two thrusters, each of which is designed to operate at a power level of one kilowatt with a thrust of 26 millinewtons (6 millipounds). The power source is a 1500-watt solar cell array. The major goal of this flight is to demonstrate mission-readiness of electric rockets by continuous operation in space for a period of six months. The test has proceeded according to plan, and has been completely successful as of this writing. One thruster was brought to full thrust and power level and then shut off. The second thruster was then started for the six-month duration test. Except for a few hours during the March 7 solar eclipse, when the thruster had to be shut down because of inadequate solar power level, this thruster has been operating continuously at full thrust.

The SERT II thruster system (fig. 2) is the culmination of some ten years of research and development on electron bombardment ion thrusters at NASA-Lewis Research Center. Further substantial improvements in efficiency have been made during the last two years (since SERT II design was frozen). The improvement is due primarily to use of a single glass-coated accelerator grid instead of the previous two-grid system (fig. 3). The improved efficiency, however, has not yet been accompanied by the long operating lifetime attainable with the SERT II system. Further research on glass-coated grids is needed to improve their durability.

Mercury electron-bombardment ion thrusters have been and are being studied at Lewis and by contractors at other power levels and sizes than the SERT II values. The largest has a diameter of 1.5 meters (fig. 4) and is designed to produce about one pound of thrust at a power level of about 150 kilowatts. This is a size that might be suitable as a module of a multi-megawatt thruster array for nuclear-electric propulsion of manned planetary vehicles.

Studies are being made to optimize the dimensions (length-diameter ratio), magnetic field distribution, propellant feed distribution and other variables to obtain better efficiencies at each power level.

Although ion thrusters have shown the most promising efficiencies, and have therefore received most emphasis in electric propulsion research, studies are continuing on magnetoplasmadynamic (MPD) thrusters (fig. 5) at very low, medium, and high power levels (ref. 3). Potential advantages of MPD thrusters are that they are much more compact for a given power level, and they operate at relatively low voltage (of the order of 100 volts instead of several thousand volts). The former advantage could produce lower mass and easier packaging, while the latter implies less power conditioning equipment. MPD thrusters may, in fact, be able to operate directly from low-voltage supplies such as solar-cell arrays or thermionic cell systems. These advantages can compensate for somewhat lower efficiency relative to ion thrusters, but generally not for as large a difference as is currently attained.

Figure 6 shows the efficiencies that have been achieved with mercury bombardment ion thrusters and MPD thrusters. These efficiencies are the ratio of jet power to electric power used. At specific impulses below 1500 seconds, efficiencies for ion thrusters are not shown, but they are currently at about the same level as MPD thrusters. The increasing gap between efficiencies for the two thruster types at higher specific impulse is due mostly to the large percentage of electrical energy that goes into anode heating for the MPD thrusters (ref. 3). Higher magnetic fields and higher power levels should reduce this loss fraction but the extent of the reduction attainable cannot yet be estimated. Further studies are needed to see whether the efficiency can be raised sufficiently to be competitive with those of ion thrusters. Long lifetime has also not yet been demonstrated with MPD thrusters.

In concluding this section, one can say that thruster systems, particularly electron bombardment ion thrusters, are in a more advanced state of development for primary propulsion than are the power generation systems needed to operate them.

Solar-Electric Space Power Systems

Operational solar arrays, such as that in use on SERT II, have a specific mass of about 100 kilograms per kilowatt. The power conditioning to convert this power to the form needed by the thrusters has a specific mass of about 9 kilograms per kilowatt. These numbers represent the state of the art several years ago, before much emphasis was placed on low specific mass for electric propulsion use. Since then, several development programs have been directed toward achieving solar arrays with specific mass less than 15 kilograms per kilowatt (ref. 2). These include both fold-out and roll-up arrays, using 8-mil silicon solar cells. Both

Air Force and NASA have programs that are scheduled to produce flight-ready roll-up arrays at the 15 kilograms per kilowatt level within about one year. Success of these programs, together with SERT II, will signal the readiness of solar electric rockets as candidates for space missions.

Nuclear-Electric Space Power Systems

Research and development activity is underway on components of nuclear-electric systems suitable for electric propulsion and much analysis has been done, but no nuclear power system with the required specific mass has yet been built. SNAP-8, the only nuclear reactor space power system now under development, has a specific mass of about 100 kilograms per kilowatt without shielding. Of course, the low power level of SNAP-8 (35 to 60 electric kilowatts), its low reactor temperature (950 K), and its consequent low efficiency (8 percent) all mitigated against achievement of low specific mass. The system is mainly directed toward early attainment of a reliable, long-lifetime nuclear power supply suitable for nonpropulsion space applications such as manned space stations or a lunar base.

In the years since the SNAP-8 system was designed, much work has been done on alternative methods of energy conversion, heat transport, and reactor design. Consequently, the prospects for future nuclear-rocket space power systems with low enough specific mass to be interesting for electric propulsion are not too bad, particularly for the relatively high power levels needed for large manned and unmanned planetary missions.

Solid-core nuclear-fission reactors are the most feasible energy sources suitable for nuclear-electric systems. Other much more speculative future sources are gas-core fission reactors and thermonuclear fusion reactors. Although these alternative sources have been considered mainly in the form of direct rockets, rather than electric rockets, they could perhaps improve the latter by providing higher initial temperatures for energy conversion systems. The possible gains, however, are not as large as one might at first expect because solid-core fission reactors can theoretically provide temperatures that are about as high as can be handled by feasible conversion systems. These converters are turbo-alternators, thermionic cells, and magnetohydrodynamic (MHD) generators. Of these, the MHD can handle the highest inlet temperature because no solid material need be at the peak temperature. But MHD generators will extract only a fraction (perhaps 10 to 20 percent) of the thermal power of the gas that passes through them. This limit is due primarily to boundary layer separation (stall) due to adverse pressure gradient. Consequently the gas exit temperature is not too much lower than the inlet temperature. This exit gas must be handled by a regenerator with surface temperature close to the exit temperature of the generator ducts to produce adequately high cycle efficiency. Although the net gain in specific mass by using gas-core fission or fusion reactors cannot yet be estimated with any confidence, it is clear that the maximum cycle temperature of a closed-cycle conversion system would be much lower than values theoretically achievable by such reactors and not much higher than those possible with solid-core reactors. Furthermore, the reactor and shield combination for the alternative sources may be heavier than for the solid-core reactor, thereby negating possible gains due to higher cycle temperatures. Consequently, these future energy sources seem best suited for direct open-cycle rocket thrust generation rather than nuclear-electric propulsion.

Another concept, termed the hybrid system, consists of using part of the power of an open-cycle solid-core or gas-core rocket to generate electric power, via an MHD generator. This power would then be used to accelerate the propellant to higher exhaust velocity (higher specific impulse) than is attainable by thermal nozzle expansion alone. The difficulty with this concept is that increasing the exhaust velocity by, say 10 percent, means that the power in the exhaust jet must be increased by more than 20 percent. Because of inefficiencies in the generation of MHD power from a gas stream, and further inefficiencies in getting the resulting electric power back into jet power, it is likely that more than 50 percent of the jet power would be needed to produce the 10 percent increase in specific impulse. With the high specific mass of the electric conversion system, this concept is probably a losing proposition - little gain in specific impulse with much gain in total mass.

Thus, solid-core fission reactors appear to be the only primary energy sources for nuclear electric rockets in the foreseeable future. The remainder of this section briefly discusses performance estimates for such reactors and the three power conversion concepts with which they may be used.

A major mass component for nuclear-electric space power systems, particularly for use with manned missions, is the shielding required. This is particularly severe at low power levels, because shielding thickness tends to be independent of reactor power level. An estimate of shielding specific mass as function of power level was given in reference 1, and is repeated in figure 7. About 1 kilogram per kilowatt of the combined reactor and shielding specific mass shown was assumed to be due to the reactor and the rest is attributed to the shielding. One curve is for complete (4π) shielding to a dose rate of 10^{-2} rem per hour just outside of the shield, and a second curve is for shadow shielding to 10^{-2} rem per hour forward and peripheral shielding (sides and rear) to 1 rem per hour. The peripheral shielding accounts for about half of the shield specific mass. Although the level of these curves may vary with assumed dose rate and shielding configuration, the variation with power level is rather independent of such assumptions. The pair of solid curves is for a complete system with shadow shielding for which a value of 5 kilograms per kilowatt was added for all other components, including thrusters and power conditioning. This is a somewhat arbitrary number, representing the lower range of values obtained in studies made during the last few years in the power range of 300 to 1000 electric kilowatts. Values obtained in these studies range from less than 3 to more than 20 kilograms per kilowatt, depending on the power level, the type of conversion system assumed, the level of maximum cycle temperature, and the degree of conservatism of the system analyst.

With the above qualifications, figure 7 shows that power levels of 10 megawatts or higher may be needed to achieve specific masses of

10 kilograms per kilowatt of jet power (the parameter that determines mission performance for these Type II systems). This power level, however, is not excessive for manned planetary missions. The optimum propulsion system mass is typically about 20 percent of the mass of the propelled vehicle. The 10 megawatt power level (with 10 kg/kw) thus implies a vehicle mass of 500,000 kilograms, which is about right for a relatively comfortable manned planetary trip.

For the much more modest shielding needed by unmanned planetary exploration vehicles (at power levels of a few hundred kilowatts), the shielding specific mass is sufficiently low that overall systems specific masses less than the 30 kilograms per jet kilowatt should be realizable. Nearly all system studies have yielded estimates below this value, and several are below 10 kilograms per kilowatt.

For the even lower power levels (5 to 50 kw), needed for satellite orbit raising or smaller planetary probes, nuclear electric systems with specific mass less than 30 kilograms per kilowatt will be difficult to achieve, even with the modest shielding needed for unmanned vehicles. Component specific masses other than shielding and reactor also tend to increase as the power is reduced to these low values. There seems to be little or no future for nuclear-electric propulsion in this low power range. It is instead the primary regime of solar-electric rockets.

The above survey of the ranges of specific mass that seem possible for nuclear-electric propulsion systems is based on a number of studies of reactors and conversion systems. Some representative ones are as follows: The Space Power Systems Division at NASA-Lewis Research Center has made a detailed study of an advanced Rankine cycle nuclear turbo-alternator system capable of generating about 375 kilowatts of electric power (ref. 4). Testing of some components of such a system is being carried out as part of an advanced technology program. The reactor outlet temperature assumed is 1475 K. Lithium is assumed as reactor coolant, and potassium in the conversion loop. A specific mass of 19.6 kilograms per electric kilowatt was estimated without shielding. With shielding, these specific masses go to about 28.5 kilograms per kilowatt for unmanned and over 40 kilograms per kilowatt for manned missions. This latter shielding specific mass (about 20 kg/kw) is lower than that in figure 7 for this power level. It is based on 10-degree shadow shielding with no peripheral shielding. This may be adequate if no activity or access outside the shadow shield area is considered necessary. Technology related to this system is sufficiently far advanced that there is considerable confidence that such a system could be built at the estimated specific mass and with very long lifetime (several years).

For a power level of 1 megawatt electric, another study of a nuclear turbogenerator system (ref. 5) yielded a specific mass of 7.5 kilograms per kilowatt without shielding. A reactor temperature of 1420 K was assumed. With shielding for manned use, the specific mass is close to that shown in figure 7.

Although the turbo-alternator conversion system is the most advanced of the three in terms of technology readiness, it is limited to use with reactors whose maximum temperature is about 1600 K. In this range of temperatures, the turbo-alternator system is difficult to beat. Neither thermionic nor MHD systems would be very efficient at these temperatures, and would therefore require larger reactors with higher thermal power. The resulting shielding penalty (for manned use) would probably outweigh any gains in lower weight of the conversion system.

At higher reactor temperatures (1800 K and above) thermionic conversion becomes more attractive, and systems analyses generally yield lower specific mass than for turbo-alternator systems. Unshielded specific masses from 2 to 5 kilograms per kilowatt are quoted (ref. 6). These are for thermionic conversion systems in which the thermionic elements are embedded in the reactor (in-core system). Although much work has been done to develop suitable fission-fueled thermionic elements, the technical problems of in-core thermionic systems are extremely challenging to say the least. Problems include high-temperature materials compatibility, thermal gradients, nuclear environment, liquid metal cooling, electrical insulation requirements, fuel-element swelling, and close spacing tolerances.

Early studies of out-of-core thermionic systems generally yielded much higher specific mass, because of the temperature loss associated with carrying the heat from the reactor to the thermionic emitters. A significant improvement in the outlook for out-of-core systems resulted when the heat-pipe concept was applied to transfer the heat from reactor core to the emitters almost isothermally (ref. 7). This concept, if it proves feasible for the high temperatures involved, almost eliminates the specific mass advantage of in-core thermionic systems, and also eliminates some of the major problems.

Studies of such systems at Lawrence Radiation Laboratory (ref. 8) have yielded specific masses of 4 kilograms per kilowatt (unshielded) for a reactor temperature of 1800 K at a power level of 300 kilowatt electric. With a 10° half-angle shield for manned missions (but no peripheral shielding), this specific mass increased to 7.9 kilograms per kilowatt. (The more compact reactor design in this study could presumably also be used for turbo-alternator conversion, if it is attainable.) With the peripheral-shadow shield considered for figure 7, this system should follow the curve of total mass quite well. The diode efficiency assumed for this system (ref. 8) was about 16 percent. When the power conditioning and thruster inefficiencies are included, the overall efficiency of 10 percent assumed for figure 7 is about what could be expected.

Early mass estimates for thermionic systems may be expected to increase somewhat as component and system development proceeds and more actual performance data become available. But it appears that the final specific mass should be considerably lower than for a turbo-alternator system, both because of the higher temperatures and the replacement of the turbo-alternator with stationary converters. The major questions concern the successful development of a high-temperature lithium-cooled reactor and the durability of the required heat pipes and thermionic converters at the required performance levels. Some long-duration tests of heat pipes and fuel elements suitable for this purpose have been made, with generally encouraging results.

The last of the three conversion methods, MHD power generation, has been studied less for space power than have the other two, despite the fact that it has potentially several interesting advantages, particularly at the multimegawatt power level. The system that seems most promising (ref. 9) is a Brayton cycle with gas-cooled, high-temperature reactor, using a noble gas (argon) as the working fluid, with either cesium or xenon as the seed gas to increase electrical conductivity.

The use of a noble-gas-cooled reactor should permit attainment of higher temperatures than the lithium-cooled reactors being studied for turbo-alternator and thermionic conversion. In fact, a gas-cooled reactor that operates at a temperature of 2500 K is already being developed. This is the NERVA reactor. Of course, it operates with hydrogen propellant, and is limited to thrust periods of a few hours rather than a few years. But the lifetime will be much less of a problem with noble gases than with hydrogen. Studies have been initiated by George Seikel at NASA Lewis to evaluate the potential performance and specific mass of an MHD system based on NERVA reactor technology. It is possible that a suitable 2500 K noble-gas-cooled reactor is technologically competitive with an 1800 K-to-2000 K lithium-cooled reactor needed for thermionic systems.

A gas-cooled reactor need not be much larger for a given thermal power than a liquid metal-cooled reactor. Some MHD cycle studies by Seikel and Nichols show that the overall efficiency may be quite high - of the order of 30 or 40 percent - as compared with the typical value of 15 percent for thermionic systems. This means that the reactor thermal power required may be only half as large for the MHD systems as for the thermionic.

With regard to the curves in figure 7, if the overall efficiency were increased to, say, 20 percent, the specific masses of the reactor and shielding would be reduced to half of the values shown, provided that the thermal power density remained the same. Such a reduction could compensate for considerable increase in mass of conversion system. More detailed studies are needed to estimate specific masses that might actually be expected.

Another potential advantage of such an MHD system is that the output power can theoretically be in the high-voltage form needed for ion thrusters, thus simplifying the power conditioning problem.

A slight difficulty at present is that there is no evidence that an MHD channel will generate the anticipated power at the temperatures and with the gases assumed. Several experimental programs are underway, however, to explore this question. For example, at NASA-Lewis, an MHD duct of a size suitable for generating several hundred kilowatts of electric power is in operation with inlet temperatures of 2000 K. At this temperature, the gas is not sufficiently conducting, even with cesium seeding, to produce electric power, unless the electron temperature is raised above the gas temperature. This requires non-equilibrium ionization, which results from preferential heating of electrons by the current flowing in the conducting gas. So far, non-equilibrium ionization has been successfully observed in shock-tube simulation, but not in a steady-state duct. In the Lewis facility, modifications are now being made to the electrodes, heater, and cesium feed system in an attempt to achieve the desired power output. The problems of generating MHD power would be much reduced if a gas temperature of 2500 K were available. This temperature is high enough to produce adequate conductivity without depending on non-equilibrium ionization if cesium seeding is used. But if an inert gas, such an xenon, is required for compatibility with NERVA-type fuel elements, some nonequilibrium ionization is still required, even at 2500 K.

In summarizing the status of nuclear power for electric propulsion, one may say that the previously used phrase "in purgatory" is quite appropriate. With regard to power conversion systems, the most advanced technology is the Rankine cycle turbo-alternator system, which is being directed toward the 300 kilowatt level. This system would require a lithium-cooled reactor at about 1500 K, with 2000 kilowatts of thermal power. No development program is under way to produce such a reactor. Furthermore, the overall specific mass of the complete system would be at the upper end of the range of interest for propulsion of large unmanned planetary exploration vehicles.

With regard to thermionic conversion, out-of-pile systems show promise of yielding interesting specific mass for propulsion of both manned and unmanned planetary vehicles (100 - 10,000 kilowatts). Considerable test time has been accumulated on thermionic elements that should be suitable for such a system. Again, a lithium-cooled reactor (heat extraction by lithium heat pipes) would be needed, at temperatures of 1800 K to 2000 K, and no such reactor is being planned. However, much encouraging work has been done on long-life fuel elements for such a reactor, and on the heat pipes to transfer the heat to thermionic emitters. Consequently, the technology needed to initiate such a reactor development seems to be close at hand.

For the MHD conversion system, the situation is somewhat different, in that a reactor in the required temperature range (2000 K to 2500 K) is in operation (NERVA). But the required lifetime and power level have not been demonstrated, and the feasibility of the entire system has not been adequately studied. The conversion system has also not yet been demonstrated, and its performance is only theoretically known. Potential advantages are high efficiency (relative to thermionic systems), high voltage, and rugged converters suitable for long lifetime.

From the standpoint of potential nuclear electric propulsion applications, the most expeditious approach appears to be to start development soon of a lithium-cooled nuclear reactor suitable for out-of-core thermionic systems in the 300 kilowatt level. This reactor could be derated to operate with a turbogenerator system if the thermionic system encountered major development problems. In the meantime, further study of MHD converter systems should be conducted, and the applicability of NERVA reactor technology should be investigated in detail. This system or a larger version of the thermionic system would then be candidates for the multi-megawatt power levels that would be useful for manned planetary vehicles during the remainder of this century and perhaps beyond.

Thermonuclear Fusion Rockets

The thermonuclear rocket appears to be the most advanced of the high-impulse systems that we can now visualize with sufficient clarity to make preliminary specific-mass estimates. Such estimates require consideration of rather prosaic matters such as materials limitations, inefficiencies of power conversion into thrust, and management of heat fluxes and mechanical stresses in all major components. Some mission and systems analysts have ignored these matters or passed them over as "engineering details," with the consequence that the predicted capabilities of their propulsion concepts have no basis in reality. They neglect just those problems that produce most of the mass of a complete propulsion system. Thermonuclear fusion rockets have occasionally been analyzed in this carefree way, but some attempts have also been made to treat them more realistically.

This is possible, however, only in a very preliminary way because the basic energy source has been demonstrated only for very large power levels and sizes (like stars) or for very brief time periods (like bombs). Consequently, the basic assumption must be made that the world-side research effort to produce fusion power for reasonable durations and at reasonable sizes and power levels will be successful.

The average plasma containment time achievable with magnetic fields determines the minimum size and power level at which a fusion power source will be operable. This containment time can be used as a parameter in the analysis, and the value needed to achieve a size suitable for propulsion can be evaluated, as was done, for example, in reference 1. A propulsion system analysis can then be based on the assumption that this containment time is achievable. Several such studies have been made, based on several proposed configurations. The most recent is described in reference 1, wherein a toroidal magnetic field geometry, similar to those used in several promising fusion experiments, was considered (fig. 8). Among the major technological problems associated with achieving low specific mass for such systems are the generation of strong magnetic fields with superconducting material capable of carrying very high current density and the shielding and cooling of that material from the high-energy radiation and particles emanating from the fusion reactor. The major mass components, other than shielding for manned use and possibly the restarting apparatus, are illustrated in figure 9, together with the circulation of the coolants and the hydrogen propellant. With some extrapolation of the performance of superconducting materials and refrigeration systems, and with some additional assumptions on successful restart techniques and momentum transfer to the propellant, attainment of specific masses in the range of one kilogram per kilowatt of jet power is considered possible.

Naturally, not much confidence can be attached to these estimates at this stage of the fusion program, but they represent an attempt to identify and analyze most of the major mass components of a fusion rocket system. Such studies also serve to point out those aspects of technology that are critical in determining the performance of the system. They show where research emphasis should be placed to obtain maximum benefit. For the thermonuclear fusion rocket, these areas are (1) reduction of plasma diffusion across magnetic fields, (2) increase in stable current density of superconducting material in high magnetic fields and at higher temperatures than are now possible, (3) reduction in mass of refrigeration system for helium coolant, (4) improvement in method of transferring energy from the fusion products to the hydrogen propellant, and (5) the ever-present need for stronger materials capable of withstanding high temperatures as well as extremely low temperatures.

These studies have pointed out other significant differences between fusion reactors for space propulsion and for ground power stations, in addition to the need for lightweight components. These include the need to use the deuterium-helium 3 fusion reaction, which produces primarily protons and alpha particles, rather than the deuterium-tritium reaction which is being studied most for ground stations. The latter produces about 80 percent of its energy in the form of neutrons which must be absorbed in a heavy surrounding blanket and converted to useful power through a thermal cycle. Such a system can operate at lower plasma temperatures than the deuterium-helium 3 system, but would be much too heavy for space applications.

From the studies conducted so far, one can conclude that nothing has yet turned up that rules out the eventual achievement of controlled fusion rocket systems with specific masses considerably lower than those achievable with nuclear fusion electric rockets. Values of the order of one kilogram per kilowatt seem attainable if the assumed plasma containment times are achieved. Recent improvements in plasma containment in the U.S.S. R. with a toroidal apparatus called Tokomak have encouraged fusion scientists and engineers in their belief that the quest will be successful. These results are compared in figure 10 with those obtained with some other devices. Ion temperature is plotted as a function of the Lawson parameter, which is the product of particle density and particle containment time. The 2× and Stellarator are devices under study at Lawrence Radiation Laboratory and Princeton Plasma Physics Laboratory, respectively. The 2× is a multistage magnetic compression experiment with plasma injection. Stellarator is a toroidal apparatus with several types of plasma heating systems. Scylla IV is an open-ended pulsed magnetic compression device under study at Los Alamos Scientific Laboratory. The open circles represent projected performance of devices now under construction.

Because of the encouraging results with Tokomak, the C-Stellarator is being converted into a modified Tokomak device, and several other devices of this type are being built in the United States.

MISSION IMPLICATIONS

The above specific-mass estimates for high-impulse (Type II) propulsion systems permit evaluation of mission capabilities and comparison with high-acceleration (Type I) nuclear systems. As pointed out before, nuclear-electric systems with specific mass in the range of 10 kilograms per kilowatt should be possible in the power range of several hundred kilowatts for unmanned use. These systems could serve as the primary propulsion systems for unmanned exploration of our solar system - particularly for the larger vehicles capable of carrying a landing craft and of transmitting high-resolution television and mapping data. Several such missions were analyzed in reference 10, using chemical and nuclear rockets in combination with electric rockets, and substantial gains in payload with electric propulsion are evident. The desirability of high power for rapid data transmission is another factor favoring use of nuclear-electric rockets for these missions.

For the lower power levels (5 - 50 kilowatts) nuclear-electric systems with interesting specific mass do not appear to be achievable. This a power range where solar cell arrays are the best bet for electric propulsion, even out to Jupiter and possibly Saturn. For the more distant missions, an additional electric power supply would be required to transmit data from the destination. Radioisotope thermoelectric generators are being considered for this purpose.

For manned exploration of the solar system, the results reported in reference 1 give a survey of the relative capabilities of the major propulsion systems of both types. These results were presented in somewhat different form in reference 3, and are shown in figures 11 through 14. Figures 11 and 12 show the trip-time capabilities of Type I and Type II systems, respectively, for a round-trip to Mars and figures 13 and 14 show the capabilities for round-trips to the more distant planets. All trip times are from Earth orbit to the destination planet's orbit and back, with no stopover time included. The payload and stages indicated are for a vehicle that has been launched to escape velocity from Earth by a booster system. The shaded bands in figures 11 and 12 show ranges of specific impulse or specific mass that may be attainable. In figures 13 and 14, the shading indicates a range of payload ratio from 10^{-1} to 10^{-2}. Here a specific impulse of 850 seconds was assumed for the solid-core fission rocket and 2500 seconds for the gas-core fission rocket. A specific mass of 7 kilograms per kilowatt was assumed for the nuclear-fission electric rocket and 1 kilogram per kilowatt for the thermonuclear fusion rocket. Also shown are the trip times that would be achievable if even better Type I or Type II systems could be conceived.

Figures 11 and 12 show that the nuclear-fission electric rocket and the solid-core nuclear rocket can both produce round trip times of about a year to Mars. Other mission studies (e.g., ref. 10) have shown the advantage of using both of these systems - a solid-core fission rocket for the initial boost beyond escape from Earth, and the nuclear electric system for the remainder of the trip. Figures 11 and 12 also show that substantial gains in round trip times to Mars (by factors of two or three) can result from successful development of gas-core fission or controlled fusion rockets.

Figures 13 and 14 show that both the solid-core fission rocket and the nuclear-fission electric rocket require excessively long trip times for round trips beyond Jupiter (greater than 3 years). Of the more advanced systems, the thermonuclear rocket produces the shorter trip times to all outer planets. Possibly with gravitational assist from Jupiter or Saturn, all outer planets could be visited by man within a three-year trip time if controlled fusion rockets can be successfully developed.

CONCLUSIONS

Some conclusions regarding research and development on high-impulse space propulsion systems are as follows:

1. Electric thruster systems have reached a sufficiently advanced stage of development that they can be considered ready for use for primary as well as auxiliary propulsion. Solar-cell arrays now being developed should be ready to provide the needed power source for primary propulsion within a year or two. Space vehicles with masses up to 5000 kilograms could be propelled by these solar-electric rockets. Applications include raising satellites from low orbit to the Earth-synchronous orbit and propulsion of space probes to all regions of interest in the solar system as far out as Jupiter and possibly Saturn.

2. To expedite use of nuclear-electric rockets for space exploration the most pressing need at present is to initiate a program to develop a compact nuclear reactor with outlet temperature of 1800 K or above, and with thermal power level of about two megawatts. Such a reactor would permit development of an out-of-core thermionic power system in the neighborhood of 300 electric kilowatts, which would be suitable for propulsion of large unmanned interplanetary exploration vehicles during the next several decades. It would also be suitable for use with a turbo-

alternator conversion system in the event that the high-temperature thermionic system encountered unexpected difficulties. The technology needed to develop such a system seems to be almost at hand.

3. Experimental and analytical studies should continue on the performance of closed-loop MHD conversion systems operating in the temperature range 2000 to 2500 K, with particular emphasis on multi-megawatt power output. The possibility of using NERVA reactor technology or more advanced reactor concepts to serve as energy source for such a conversion system should be thoroughly evaluated. Such a system could be competitive with thermionic systems in this power range.

4. Studies of the application of controlled thermonuclear fusion power to space propulsion should continue, with emphasis on those aspects and system components which are significantly different for propulsion than for ground power stations.

These conclusions fit quite well with the apparent technological status of the systems, and also with the time periods in which they might be useful. Thus, the relatively small vehicles propelled by solar-electric rockets could be used as soon as they are ready. They could enhance our ability during the next decade or two to explore the inner planets and their satellites, the asteroid belt, the comets, regions close to the Sun, and phenomena out of the ecliptic plane.

The nuclear-electric propulsion system at the power level of several hundred kilowatts could be advantageously used within ten years or less for more intensive unmanned exploration of both inner and outer planets. Since the time required to develop such a system may also be about ten years, there is some urgency in getting started. Development of such a system is especially important if, as seen possible, our exploration of the planets is limited to unmanned flights for several decades. There is no other propulsion system in sight which can do such missions nearly as well as the nuclear-electric rocket. The combination of high payload, high power level and high specific impulse is ideal for this purpose.

On the other hand, the multimegawatt electric systems suitable for manned planetary exploration have no real urgency at present. The time period in which they would be useful is vaguely established as sometime before the end of this century, with the late 1980's as the earliest plausible estimate. Consequently, we should continue to explore alternative systems until more evidence is available to decide which system is likely to yield the best performance when it is needed.

REFERENCES

1. Moeckel, W. E., "Propulsion Systems for Manned Exploration of the Solar System," Astronautics and Aeronautics, Vol. 7, No. 8, Aug. 1969, pp. 66-77.

2. Lazar, J., "Current Status of Electric Propulsion Technology," Paper 69-1107, Oct. 1969, AIAA, New York, N.Y.

3. Anon., "Plasmas and Magnetic Fields in Propulsion and Power Research," SP-226, 1970, NASA, Washington, D. C.

4. Heller, J. A., Moss, T. A., and Barna, G. J., "Study of a 300-Kilowatt Rankine-Cycle Advanced Nuclear-Electric Space-Power System," TM X-1919, 1969, NASA, Cleveland, Ohio

5. Zipkin, M. A., "Alkali-Metal, Rankine-Cycle Power Systems for Electric Propulsion," Journal of Spacecraft and Rockets, Vol. 4, No. 7, July 1967, pp. 852-858.

6. Becker, R. A., "Thermionic Space Power Systems Review," Journal of Spacecraft and Rockets, Vol. 4, No. 7, July 1967, pp. 847-851.

7. Schreiber, R. E., and Salmi, E. W., "A Look at Nuclear Thermionic Systems," Astronautics and Aeronautics, Vol. 6, No. 1, Jan. 1968, pp. 50-56.

8. Loewe, W. E., "Analysis of an Out-of-Core Thermionic Space Power System," IEEE Transactions on Aerospace and Electric Systems, Vol. AES-5, No. 1, Jan. 1969, pp. 58-65.

9. Nichols, L. D., "Comparison of Brayton and Rankine Cycle Magnetogasdynamic Space-Power Generation Systems," TN D-5085, 1969, NASA, Cleveland, Ohio.

10. Mascy, A. C., Dugan, D. W., and Pitts, S. W., "Application of Combined Electric, High-Thrust Propulsion Systems," Journal of Spacecraft and Rockets, Vol. 5, No. 7, July 1968, pp. 785-791.

E-5646

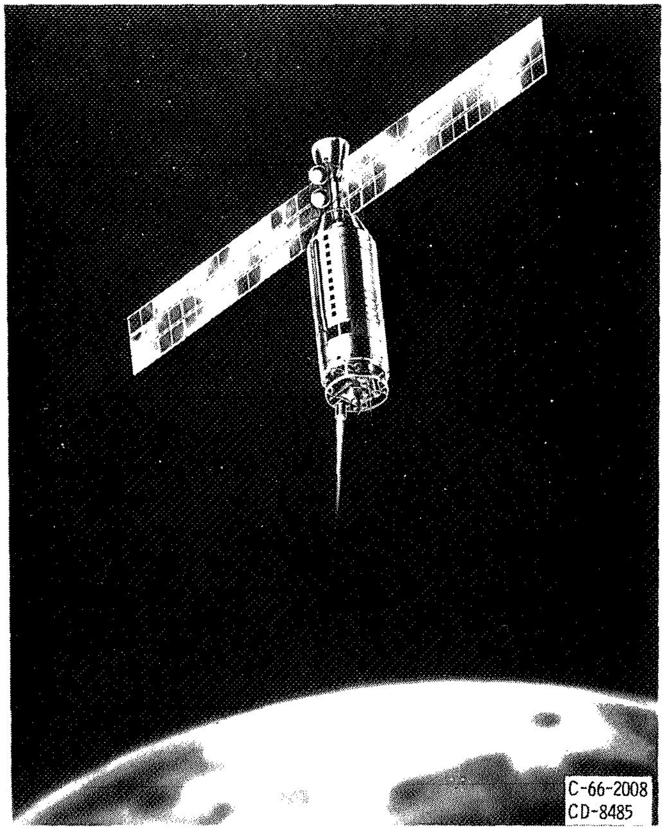

Figure 1. - Artist's drawing of SERT II spacecraft and solar-cell array in orbit.

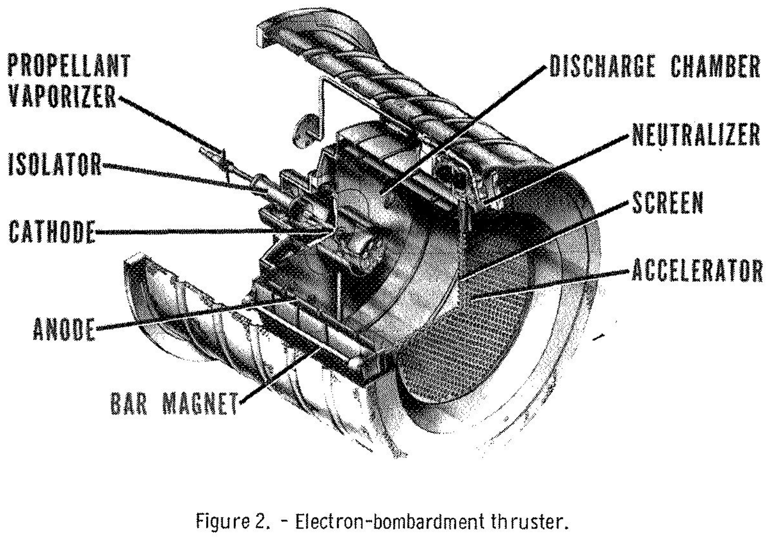

Figure 2. - Electron-bombardment thruster.

E-5646

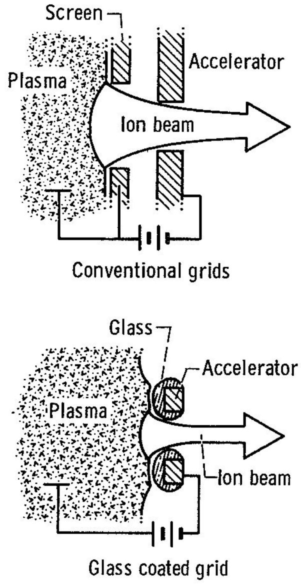

Figure 3. - Comparison of conventional and single glass-coated accelerator grid concepts.

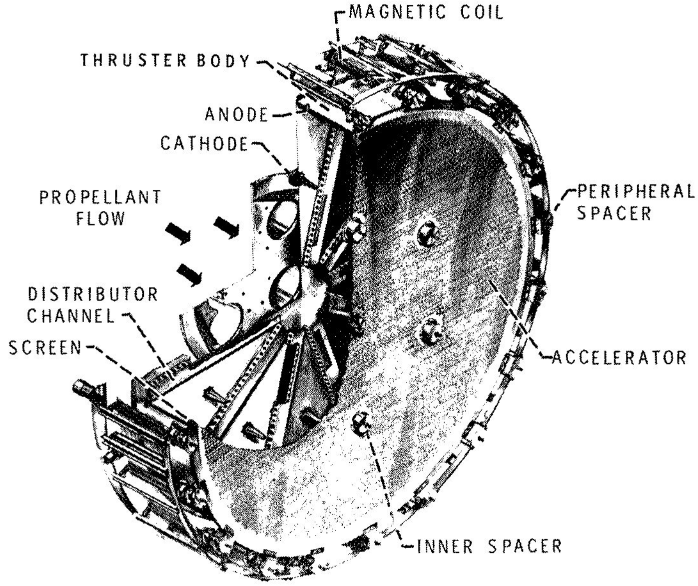

Figure 4. - Cutaway view of 1.5 meter diameter Kaufman thruster.

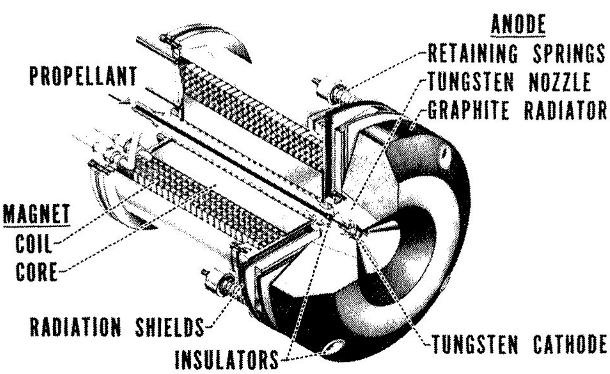

Figure 5. - 30 kW M P D Arc thruster.

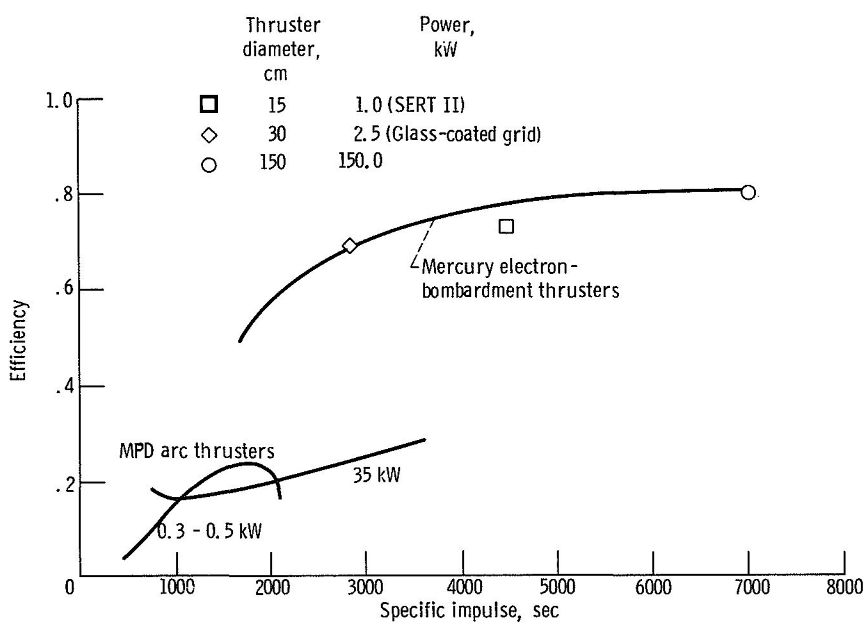

Figure 6. - Efficiencies for several electric thrusters.

E-5646

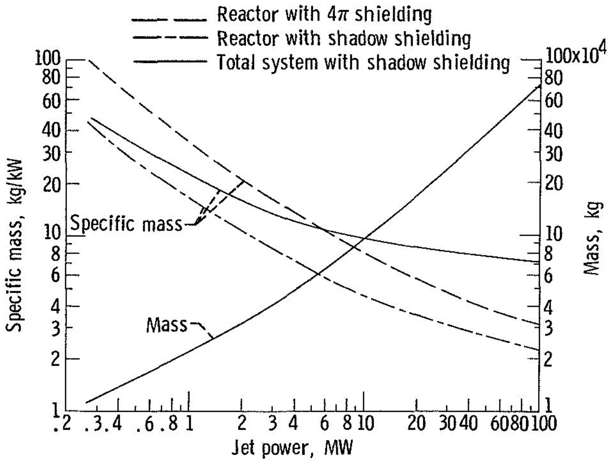

Figure 7. - Variation of mass with power for nuclear-electric propulsion systems. Reactor thermal power density, 100 watts per cubic centimeter; overall efficiency, 10 percent.

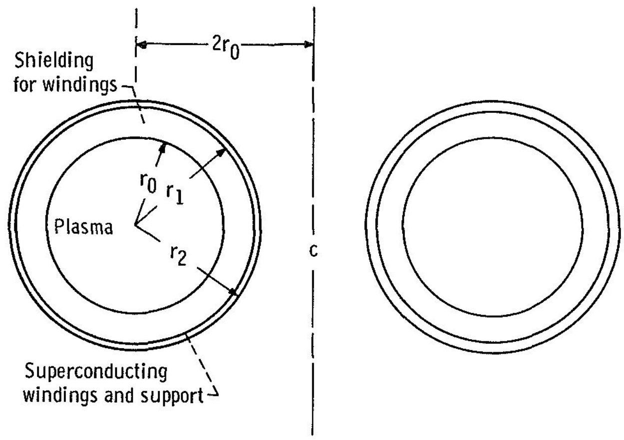

Figure 8. - Toroidal fusion reactor geometry.

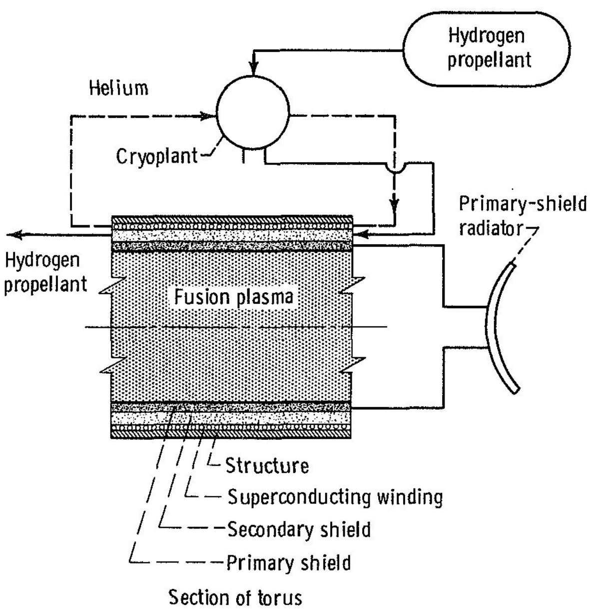

Figure 9. - Schematic diagram of toroidal fusion rocket propulsion system (Englert).

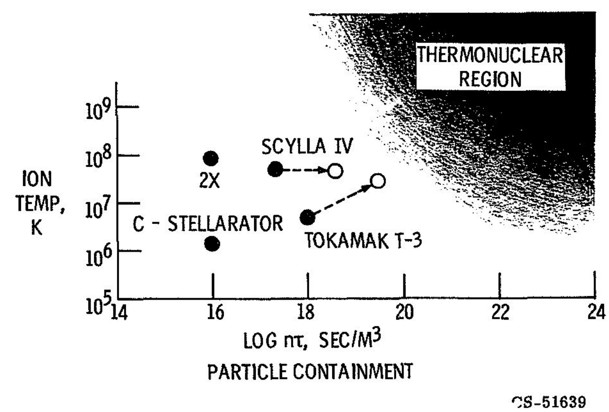

Figure 10. - Lawson diagram.

E-5646

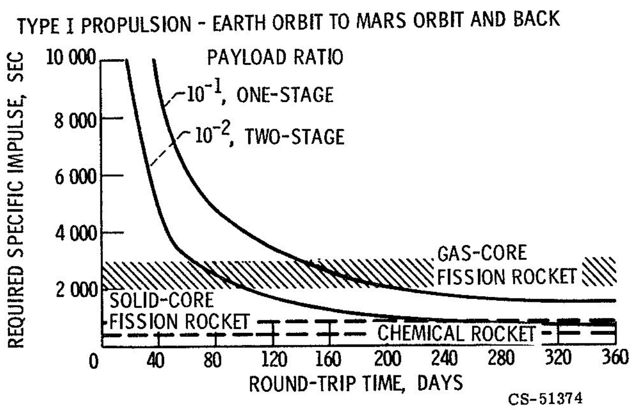

Figure 11. - Mars "Quick Trip" requirements.

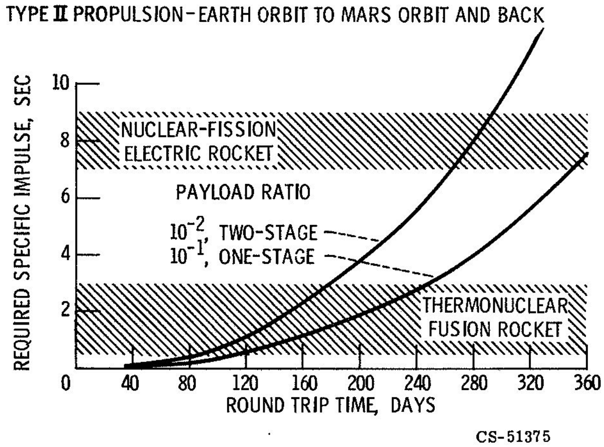

Figure 12. - Mars "Quick Trip" requirements.

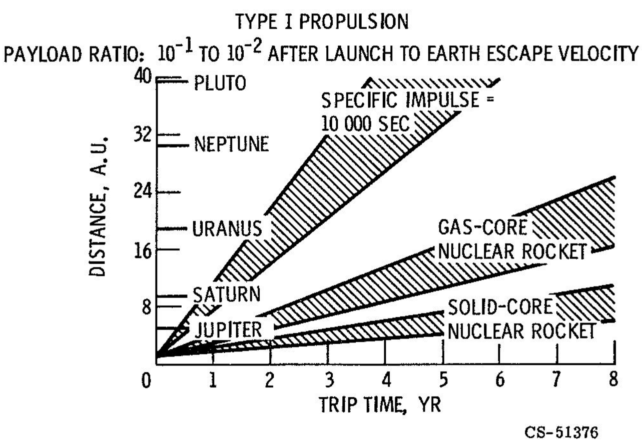

Figure 13. - Round trip time to planets.

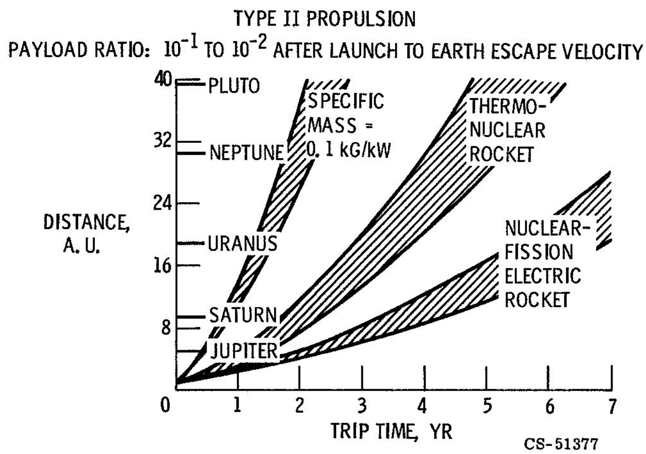

Figure 14. - Round trip time to planets.