N63-15991

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ELECTRIC PROPULSION: A NEW TECHNOLOGY

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Within the past 3 years, electric thrust devices have passed from the first exploratory feasibility experiments to the flight-engineered hardware phase. This rapid surge of development has been punctuated by equally rapid improvements in the performance figures of electrical thrust devices. The most spectacular advances were made in ion engine technology. Total energy expenditure per ion pair has gone from 35,000 ev in 1958 to about 600 ev at the present time leading to power utilization efficiencies of up to 80 per cent, depending upon the specific impulse. The percentage of the particles intercepting the accelerating and focusing electrodes has been reduced from more than 50 per cent in 1958 to less than 0.01 per cent at the present time. Great improvements in ion current per unit area have also been realized, leading to greater thrust "pressures."

The feasibility of arc engines was a widely accepted belief in 1958. At that time, arcs were operated for reentry simulation, and their conversion to propulsion devices was considered easy. Most conversion effort centered upon engineering modifications to render the arcs capable of flight, at the expense of fundamental research leading to improved performance. Only during the past year have the research problems been recognized, and work started on the specific problems of arc propulsion. Due to this late start, progress has been less spectacular than in ion engine research. Nevertheless, the period from 1958 to the present has seen the development of an engine with an overall power conversion efficiency of 56 per cent using hydrogen as a propellant; of an ac operated engine; the first radiation-cooled flight engine; and of an associated technology including propellant feed systems, current control systems, and ignition circuits.

Electromagnetic, or plasma, accelerators remain in an earlier phase of research compared with arc and ion engines. Reliable thrust and efficiency measurements are difficult with plasma accelerators, but efficiencies of 30 to 40 per cent are within the present state of the art. One "pinch plasma engine" is presently being engineered for flight test in the 1962-1963 period.

The three principal types of thrust devices will now be discussed in greater detail.

15.1 Principal Types of Electric Thrust Devices

15.1.1 Ion Engines

Ion engines contain four major components: the feed system, the ion source, the ion accelerator, and the beam neutralization system. To be suitable for an ion engine, an ion source must possess special characteristics. It must be designed so that the average current density drawn from an array of these sources is high enough to keep the total area of the engine to a reasonable size. It should be sufficiently light that its mass is a small fraction of the total vehicle mass. It should convert the electrical power output of the power supply into ion kinetic energy with the highest possible power efficiency. It must have a long and reliable lifetime of the order of a few years. It should convert virtually all of the propellant atoms into ions. High propellant efficiency is desirable not only for flight economy, but also because neutral atoms which escape from the ion source accumulate in the accelerator where they may cause scattering, erosion, and voltage breakdown. Finally, it should produce ions of high atomic weight.

During the past 3 years, several types of ion sources have been developed which meet the above specifications to a greater or lesser degree. These include the surface ionization source, and several modified duoplasmatron and Penning discharge sources. Propellant efficiencies of some of the different ion sources are listed in Table 15.1.

Table 15.1 Propellant and Power Efficiencies of Ion Engines

Ion Source	Propellant Efficiency (%)	Power Efficiency (%) (I_{sp}=10,000 sec)
Electron Bombardment	80	78 [Ref. 1]
Duoplasmatron	90	90 [Ref. 2]
Surface Ionization	98.5	77 [Ref. 3]

The surface ionization source relies upon the principle that, in theory, a high work function material like tungsten can catalytically ionize more than 99 per cent of a propellant such as cesium whose ionization potential is considerably less than the work function of tungsten. Unavoidable radiation from hot ionizing surfaces accounts for the major power loss with surface ionization sources.

The modified duoplasmatron sources utilize a magnetically pinched glow discharge to ionize mercury as a propellant. In the laboratory, these types of sources have achieved the lowest ionization energy expenditures of any sources tested so far, at propellant efficiencies (with mercury) in excess of 90 per cent.

Another promising collision source which has been developed into a complete engine is an electron bombardment source developed by H. Kaufmann at the Lewis Research Center, (NASA). This source offers ionization energies as low as 1100 ev per ion with propellant efficiencies of about 80 per cent.

Once the ions have been produced, acceleration is necessary. The ion accelerators must meet certain specialized requirements. Since ion emission is generally uniform across the surface of the source, the accelerator must provide a uniform electric field across the source to extract ions as rapidly as they are formed. The accelerator must have as low an ion interception as possible since high-energy ion bombardment sputters the electrodes rapidly and results in deterioration of the ion optics.

Two approaches are commonly taken in the difficult job of designing an ion acceleration and focusing system. One consists of specifying electrode shapes and attempting to determine the space-charge flow pattern which results. The electrode configurations are adjusted by trial and error to get a desired flow pattern.

Another approach consists of trying to determine analytically the electrode configurations required to produce any given beam. This is a more difficult problem; solutions do not always exist. For simple rectilinear flow (planar, cylindrical, or spherical geometries), the Pierce principle is used as a first approximation to a more defined design. However, more advanced techniques have been developed for guns employing curvilinear flow.

The ion accelerator should draw off all the ions which the ion source can provide. Space charge effects impose a limitation on ion current density. To increase the maximum current density and thereby enhance the thrust pressure, it is customary to employ an accel-decel system in which the ions are first accelerated to a high velocity, and then decelerated to the desired exhaust velocity.

One of the most controversial questions relating to ion engines is that of the neutralization of the ion beam once it leaves the source. The beam must be neutralized to avoid an accumulation of charge on the space vehicle. It must be neutralized close to the vehicle to avoid the buildup of large potentials in the ion beam which would cause it to spread excessively, and even to reflect some of the ions back to the negatively charged accelerator electrode. All of the present theoretical and experimental results indicate that ion beam neutralization may not be a severe problem [4, 5]. However, the crudeness of the theoretical treatments, and the great difficulty of simulating the unlimited conditions of outer space in laboratory experiments render space tests mandatory to answer the beam neutralization question. Preferably, several neutralization techniques should be tried in these tests, since the stability characteristics of the plasma are quite sensitive to the geometry of its boundaries.

15.1.2 Arc Engines

Arc engines consist of a propellant storage tank, a flow control system, an arc chamber, and a convergent-divergent expansion nozzle. Propellant storage is a significant problem with hydrogen or helium because of low atomic weights and low boiling points, especially when radiation absorption from an onboard nuclear reactor is involved. Other promising propellants are ammonia and a hydrogen-lithium mixture.

Within the arc chamber and the nozzle, the principal problems are erosion of the walls and electrodes and dissociation, or "frozen flow," losses in the nozzle which reduce arc engine power efficiencies. The latter term refers to the energy lost when gases, dissociated and ionized in the arc chamber, fail to recombine in the nozzle to give up their recombination energy and enhance the expansion process.

Careful attention to electrode design and effective cooling techniques can relieve internal erosion. Rotating the arc electrically to vary its point of attachment to the electrodes also helps. Recent experiments with a mixture of hydrogen and lithium as propellants suggest another approach to reducing erosion [6]. The lower ionization potential of this bipropellant means a lower arc voltage, and hence reduced electrode erosion rates. The heat transfer losses can be reduced considerably by careful arrangement of the gas flow pattern, by optimizing the length and shape of the arc, and by employing regenerative cooling. Heat losses to the walls decrease with increasing engine size, simply because the surface-to-volume ratio of engines becomes more favorable as they increase in magnitude.

Much theoretical and experimental effort has been spent recently to learn more about recombination and reassociation in high temperature arc [7]. Helium would be an ideal propellant from the standpoint of dissociation losses--there would be none; however, the storage problem offsets the gain in power efficiency by far. Higher specific impulse means higher chamber temperatures, greater heat losses, and lower efficiency.

Another attractive possibility is the use of ac power instead of dc power. The use of ac permits stable operation with only a negligible ballast resistance and without power rectification losses.

Among the arc engines currently under development are 30 kw dc engines (AF and NASA) and a small 1 kw arc engine (NASA) for flight test late in 1962. Characteristics of these engines are shown in Table 15.2.

Table 15.2 Power Efficiencies of Arc Engines

Specific Impulse (sec)	Propellant	Power (kw)	Efficiency (%)	Company
780	NH_3	1	24	Plasmadyne
900	H_2	3	40	Avco
1000	H_2	30	38	Avco
1000	NH_3	30	40	Avco
1000	H_2	30	56	Plasmadyne
1100	H_2	1	38	Plasmadyne
1100	H_2	30 (ac)	45	GE
1350	H_2	30	41	Avco
2000	H_2	30	20	Plasmadyne

15.1.3 Plasma Accelerators

The majority of plasma propulsion devices currently being explored can be classified into three major categories. These are pulsed plasma accelerators, crossed-field accelerators, and traveling wave accelerators.

The pulsed plasma accelerators depend upon the interaction of a pulsed current discharge in a plasma with either its self-generated magnetic field, or with the magnetic field generated by the external return circuit for the discharge current. The T-tube, rail, and button guns fall in this category, as does the pinch plasma engine presently planned for space flight testing in 1963.

Crossed-field accelerators are steady state devices in which plasma from an arc is accelerated by the $\vec{j} \times \vec{B}$ force which arises when external electric and magnetic fields are applied to the plasma perpendicular to the streaming direction and perpendicular to each other. In the low specific impulse regime (less than 1400 sec), power conversion efficiencies around 30 to 40 per cent have been achieved with various gases as propellants.

Traveling wave accelerators utilize a moving, recurrent magnetic field which accelerates an ionized gas. The field is generated by one of the following methods: (1) a polyphase resonant radio frequency circuit generates a succession of traveling waves in field coils (the induction motor principle); (2) a delay line gives the desired velocity of propagation; or (3) a series of timed, pulsed capacitor discharges energize the field coils.

Each of the above devices has its own specific limitations. The pulsed devices require the development of lightweight reliable condenser banks and of associated equipment to improve power coupling efficiency into the plasma. Nevertheless, plasma accelerator technology is making headway and offers certain advantages over other contenders, provided technical development is successful.

15.2 Future Improvements in Electrical Propulsion Systems

While the first phase of electrical propulsion development is drawing to a close, the next phase has begun to take shape. Very characteristically, this next phase is concerned with improved efficiency and increased lifetime. As a rule, these properties are not in the focus of endeavor during the feasibility demonstration phase of a technical development, but they must receive growing attention as soon as the basic soundness of a concept is assured.

Judging from the present status of ion engine development, it is very likely that future ion engines will have propellant efficiencies between 80 and 98 per cent, and power efficiencies--depending on their specific impulses--between 60 and 90 per cent. Once the efficiencies are in the 90 per cent region, their influence on overall performance is less significant than their influence on secondary effects such as lifetime of ion sources, erosion of electrodes and structural elements, and voltage breakdown across insulators. From this viewpoint, surface ionization sources look promising because of their high propellant efficiency. A very substantial gain in the power efficiency of surface ionization sources would be obtained if the ionizing surface were not heated by precious electric power, but by cheap heat directly from the nuclear reactor. Studies in this direction are underway.

Accumulated operating times of individual electric engines are presently not longer than a few hundred hours. Even though results are promising in general, a number of areas require continued development effort before the desired lifetimes of 20,000 or 30,000 hr can be guaranteed. Among them are the design of electrodes and filaments; the fabrication of homogeneous, large-area porous tungsten ionizers; the development of propellant feed systems; and the design of beam neutralization systems.

Arc engines have been operated up to 50 hr at efficiencies of 40 per cent. The goals for future developments are extension of the specific impulse to 2000 sec, increase of lifetimes, and an improvement of efficiencies. At 1000 sec, specific impulse efficiencies up to 80 per cent seem feasible. Other areas where improvements will be made include high-temperature insulators and materials, electrode geometries, heat transfer analyses, better arc dynamics, and minimization of leakage losses.

Besides the engineering work that must still go into the improvement of thrust-producting engines, a considerable effort will be required for developing nuclear-electric power supplies. One of the first nuclear-electric power generators for space use, the Snap 8, will begin its flight testing phase in 1965. With 30 kw output power and a specific power of about 0.03 kw kg^{-1}, it is marginal for electric propulsion. Power supplies of 100 to 300 kw, 1 to 4 Mw, and about 20 to 40 Mw power, and specific power figures of 0.3 to 1 kw kg^{-1} are desirable for future space missions. Most likely from about 1970 on, power supplies will not be based on turbo-electric generators, but on thermionic or plasma dynamic converters.

The engineering development of electric propulsion should be paralleled continuously by vigorous research programs. Although present technologies will be adequate to make electrically propelled vehicles equivalent, and even superior, to nuclear rockets on planetary flight, further research is urgently needed for improved and refined problem solutions, for more economic performance, and to assure reliability under long-time operation.

15.3 Flight Testing of Electric Engines

Plans and preparations for space flight testing of electric propulsion systems have become firm within the past year and a half. Tests for ion engines are necessary to settle the ion beam neutralization question; they are also necessary as mission-rating tests for all types of electric thrust devices before actual space missions with more expensive power supplies. Present NASA plans call for two test flights in late 1962; four small engines, delivering only a few millipounds of thrust, will be tested. The payloads will incorporate two surface ionization engines, one collision-type ion engine, and one arc engine. The four engines and the flight test capsule are shown in Fig. 15.1.

These engines will be launched by Scout vehicles into nearly vertical trajectories with peak altitudes of about 8000 km and total flight times of well over 1 hr. This will allow each engine about 35 min of operating time under space conditions. Power will be provided by batteries.

Testing on satellites may begin in late 1963, using power supplies like Snap 2 and Sunflower. It seems probable that soon after these tests are completed, arc engine propulsion systems will be harnessed

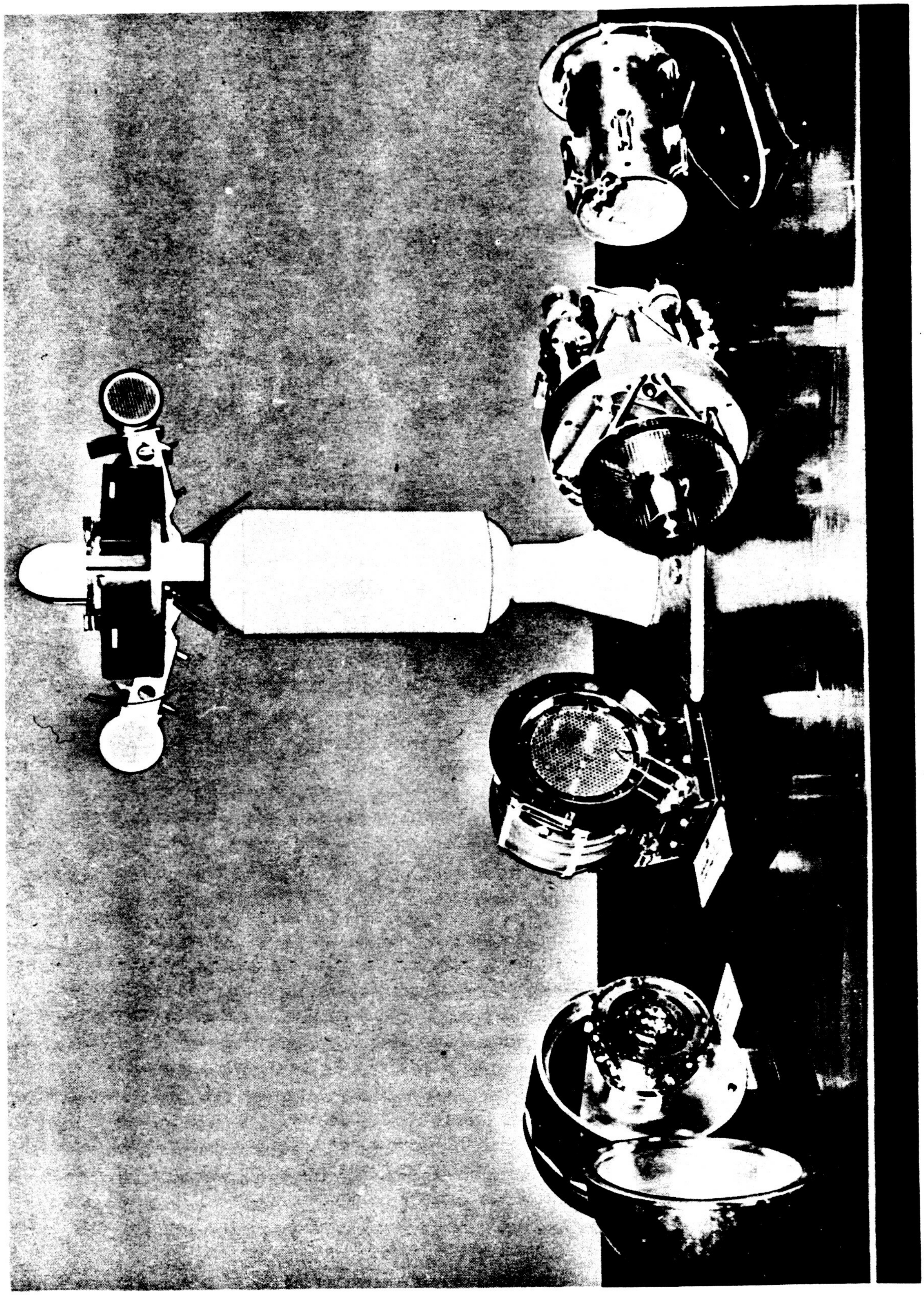

Fig. 15.1 Electric propulsion engines and test capsule for flight testing in late 1962.

for the transfer of satellites from low to high orbits, for the correction of orbital launching errors, and for other applications where fine control of total impulse is required. The next major milestone in the flight test program will be the testing of 30 kw engines in conjunction with the 30 kw Snap 8 power supply in 1965-1966. After that, ion propulsion may be applied to the propulsion of probes to the planets, for out-of-the-ecliptic flights, for deep space missions, and ultimately, in the 1975 to 1985 period, for manned flights to Mars. All of these missions will be discussed below in more detail.

15.4 Power Supplies

Until thermionic and plasma dynamic converters with sufficiently high efficiencies become available, the nuclear turbo-electric power supply system will probably be used. The nuclear fission reactor is the most attractive source of primary energy for the long lifetimes associated with missions employing electric propulsion.

Studies have indicated that the Rankine vapor heat cycle is superior to the Brayton gas cycle in the conversion process. The efficiencies of the Rankine vapor cycle are close to the efficiencies attainable with the ideal Carnot cycle. Figure 15.2 illustrates the cycle efficiencies of the Carnot, ideal Rankine (no system inefficiencies), and the actual Rankine (turbine inefficient) cycles, using potassium as a working fluid.

The fast reactor is best suited for space power applications because of lower core and reflector size and weight. A recent report [8] describes a fast reactor designed by Atomics International for use in a 300 kw (electric) space power plant. It is fueled by uranium monocarbide, cooled by rubidium boiling at 1800°F, produces 2.4 Mw (thermal) energy for more than a year at continuous full-power operation, and weighs approximately 500 lb. Reflectors are made of beryllium oxide and control is provided by moving the reflectors.

The selection of working fluids for a nuclear turbo-electric system is governed by factors such as material temperature limitations, materials compatibility, stability during prolonged nuclear radiation and high-temperature operation, physical and thermal properties of the fluid, and others. Elements, and in particular alkali metals, are most attractive as working fluids. When upper-and lower-temperature and pressure limitations are imposed on the cycle, the most suitable working fluids for nuclear turbo-electric systems are found to be sodium, potassium, cesium, and rubidium.

To reject the thermal energy not converted into electrical energy, a waste heat rejection system must be employed. In a space environment, radiation is the only known method for rejecting heat. Since the

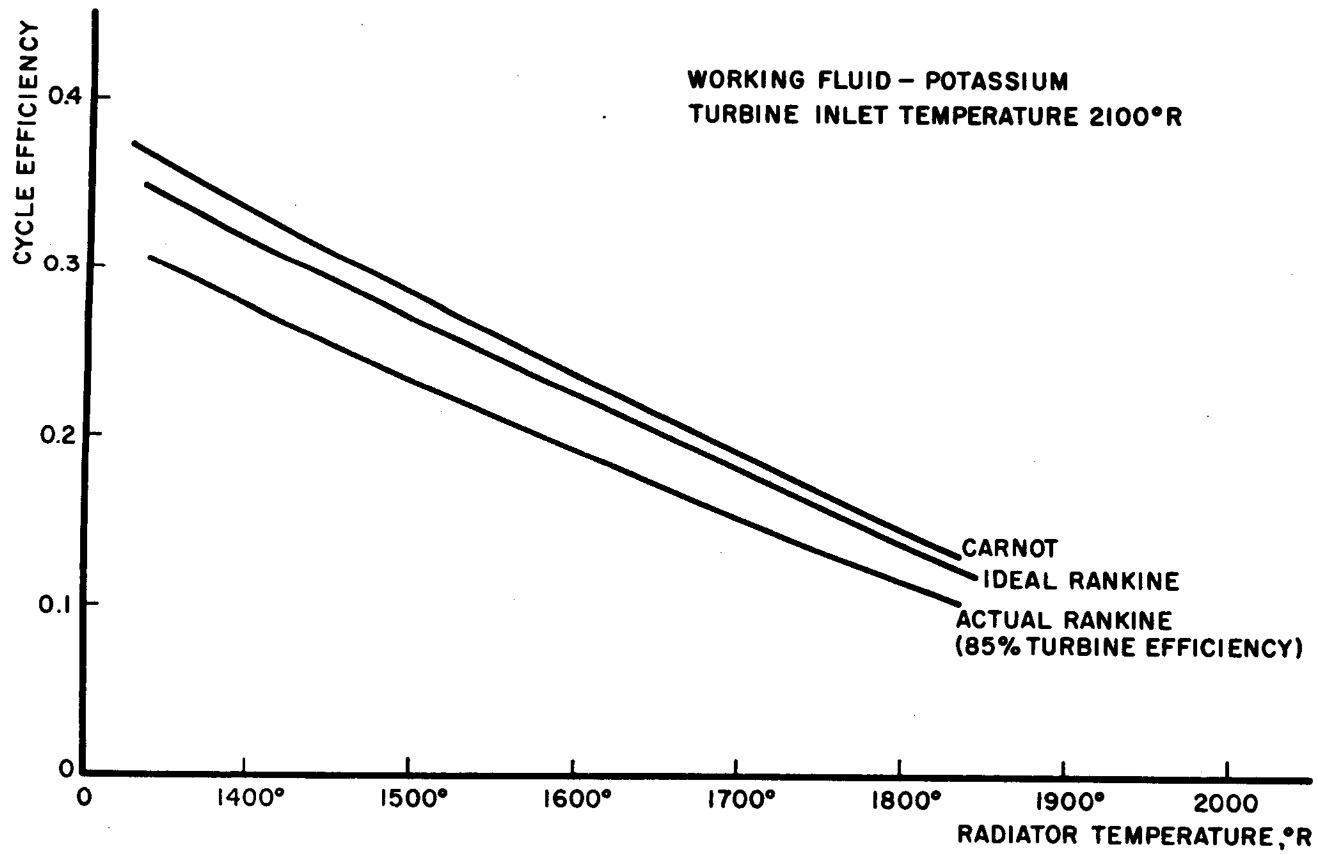

Fig. 15.2 Cycle efficiency versus radiator temperatures for Carnot, ideal Rankine and actual Rankine cycles.

Rankine cycle efficiency is around 20 per cent, the thermal energy to be rejected is significant; the radiator is by necessity very large. In fact, it is the heaviest part of the power supply system.

The heat radiated from a body in uniform surroundings is given by the Stefan-Boltzmann equation

$$Q_R = \sigma\epsilon A_R (T_R^4 - T_S^4) \tag{15.1}$$

where Q_R = heat radiated

σ = Stefan-Boltzmann constant

ϵ = emissivity of radiator

A_R = area of radiator

T_R = absolute temperature of radiator

T_S = absolute temperature of surroundings

Away from a planet, for all practical purposes, T_S is absolute zero, and can be dropped from Eq. (15.1). Since the heat radiated varies as the fourth power of the radiator temperature, a high temperature is desirable. For the actual Rankine cycle, there is a definite relationship between the temperature of the working fluid entering the turbine, T_A, and the temperature of the working fluid in the radiator, T_R, which results in optimum cycle efficiency and minimum radiator area. This relationship is given by

$$\frac{T_R}{T_A} = 1 - \frac{5}{8k} + \frac{1}{8k} (25-16k)^{\frac{1}{2}} \tag{15.2}$$

$$\text{where } k = \frac{\text{Rankine cycle efficiency}}{\text{Carnot cycle efficiency}}$$

The ratio $\frac{T_R}{T_A}$ ranges between 0.75 and 0.80, and it can easily be remembered that the heat rejection temperature, T_R, is always approximately 3/4 of the turbine inlet temperature, T_A, when the radiator area is at a minimum. The relationship in Eq. (15.2) presumes that the radiator is at a uniform overall temperature, and this is essentially the case in Rankine cycle where the working fluid gives up its heat while condensing in the radiator at temperature T_R. The turbine inlet temperature, T_A, is the highest temperature reached by the working fluid, and its upper limit is determined by material temperature limitations in the reactor core. At the present time, around 2500°F (2960°R) is considered the upper limit.

Since the radiator temperature varies almost directly with the turbine inlet temperature in an optimized system Eq. (15.2), a higher turbine inlet temperature will result in a higher radiator temperature and a smaller radiator area. This relationship is illustrated in Fig. 15.3. It is interesting to note that for minimum radiator areas, the radiator temperature is always approximately 3/4 the turbine inlet temperature, as mentioned above. Figure 15.3 is based upon a 1 Mw (electric) power system using potassium for the working fluid, a value of 1.0 for the radiator emissivity, and a uniform radiator temperature.

The materials for radiator construction should be light weight and have good heat transfer properties, and the flow passages must be of a material compatible with the working fluid. To reduce radiator weight, a fin-tube configuration can be used. The fin-tube configuration results in a larger radiator area but the overall weight is reduced because the fins can be selected of a low-density material. Meteoroid puncture protection is required only over the fluid flow passages. Cornog [9] has investigated materials suitable for fin construction. He found that beryllium, graphite, and copper, in that order, have the most attractive thermal conductivity to density ratios at operating temperatures above 1200°F. Beryllium is usually considered to be the best fin material. If an alkali metal is used as the working fluid, the most suitable tube material will be the refractory metals molybdenum, columbium, tantalum, and tungsten, or selected stainless steels and alloys.

As may be seen from Eq. (15.1), the value of ϵ, the emissivity of the radiator, should be high in the temperature range in which it operates. Since the emissivity of beryllium is low, a suitable surface coating is required. Aluminum oxide will be a favorable coating with emissivity values approaching that of a black body in the infrared.

An important design consideration for the power supply is protection from meteoroid impacts. To date, there have not been sufficient measurements in space to accurately determine the meteoroid flux and the mass, velocity, and density distributions. Experiments in the laboratory are now limited by inability to reproduce meteorite velocities. The thickness of meteoroid protection material over tubes in the fin-tube configuration is dependent upon the material used, the exposed area, the mission lifetime, and the desired probability of no punctures. One rather disheartening fact is that as the power levels for electric propulsion systems go higher in the future, the weight per unit radiator area will increase. This happens because as the power level increases, more heat must be rejected, thus greater radiator areas are required, and thus the thickness of the protective material must be increased in order to maintain the same degree of total improbability of puncture.

Meteorite "bumpers" have been proposed as a means of protecting vulnerable areas. The penetration resistance increases substantially if a given thickness of material is divided into more than one sheet

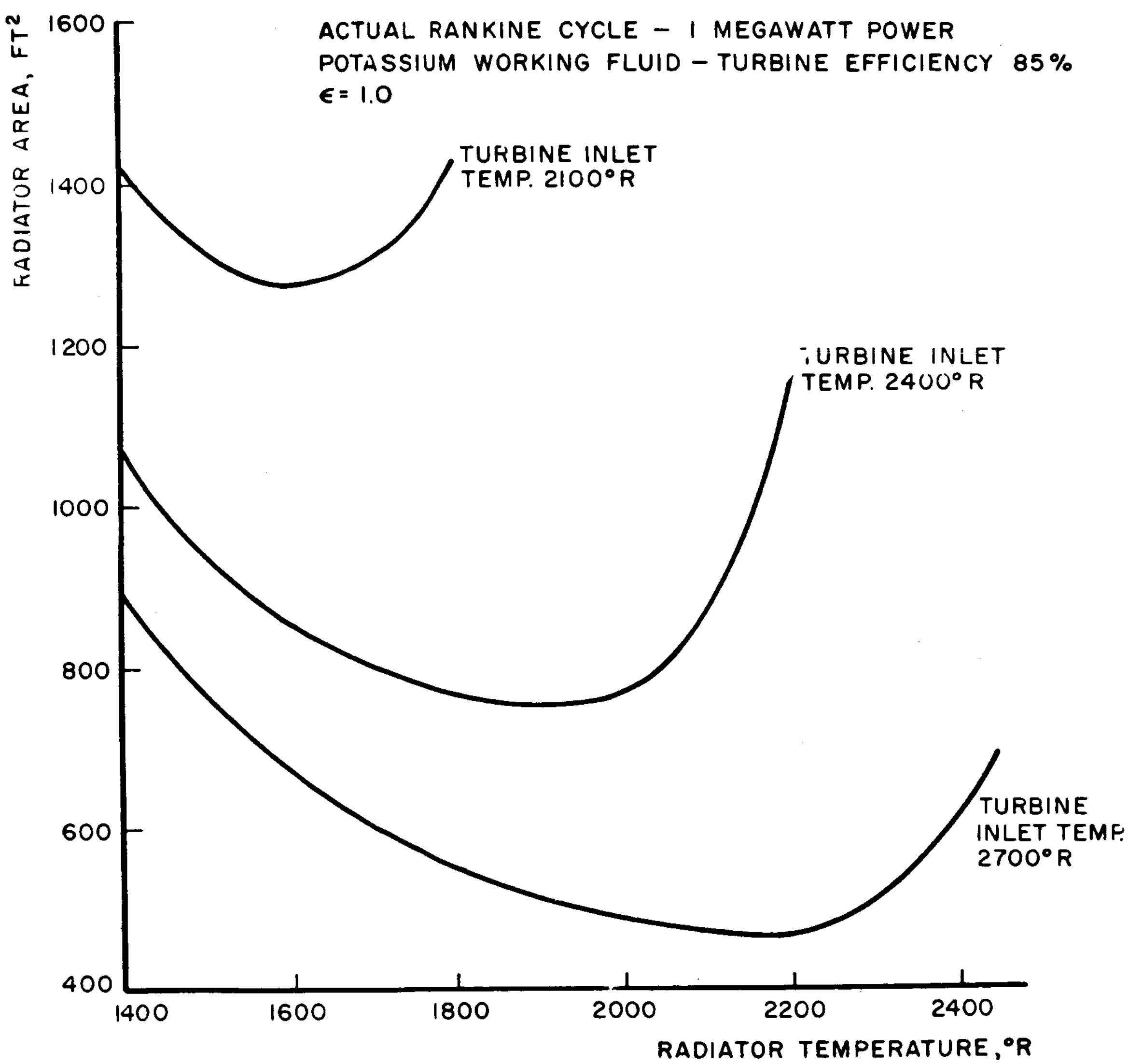

Fig. 15.3 Radiator area versus radiator temperature for varying turbine inlet temperatures.

and separated. Using Pyrex glass spheres, 1/8-in. diameter at velocities up to 11,000 ft/sec , Nysmith and Summers [10] found that the penetration resistance increased by a factor of 1.75 when a single sheet was replaced by two sheets (each 1/2 thickness of the original) with 1/2 in. spacing, and by a factor of 2.2 with 1 in. spacing. Even though "bumpers" may dissipate the energy of impacting meteoroids, their use will result in a definite loss of thermal performance, thus requiring a trade-off until minimum weight is achieved.

When the Snap 8 system is proved operational in space, it will provide valuable information for designing future power supply systems. As thermionic and plasma dynamic converters become competitive with turbo-electric converters, the systems will become lighter, simpler and more reliable.

15.5 First Missions: 1966-1970

Early missions for electric propulsion are expected to begin in 1966, provided electric engines and nuclear-electric power supplies are vigorously developed. After flight testing of arc and ion engines with the 30 kw Snap 8, proven engines will use the Snap 8 for mission flights until about 1970.

Electric propulsion systems depend upon a number of parameters that are unique for these systems [11]. The most important ones are the specific power of the power source, α, the propulsion time, τ, the specific impulse, I_{sp}, and the energy conversion efficiency, η, of the engine. For each set of α, τ, and terminal velocity, μ, an optimum specific impulse is found to make the payload ratio a maximum. Figure 15.4 shows the relationship between the total velocity increment and propulsion time, using specific impulse and payload weight as parameters, based upon specific space ship characteristics taken from Ref. [12]. The velocity increments required for specific missions are functions of propulsion time, launch site, time of flight and flight plan, but these factors are assumed specified in Fig. 15.4. This figure shows that electric propulsion provides a great flexibility for fulfilling specific missions. For any given mission, there is a trade-off between propulsion time and payload weight.

Early missions with arc engines may have propulsion times of 50 to 100 days, as shown in Table 15.3.

Electric engines do not operate with a constant energy conversion efficiency over a full range of specific impulse. Figure 15.5 illustrates the thrust attainable from present-day electric engines as a function of specific impulse. The ideal theoretical electric engine should produce a hyperbolic relationship, but actual engines show lower thrust values at specific impulses between 1200 and 4000 sec. This does not necessarily

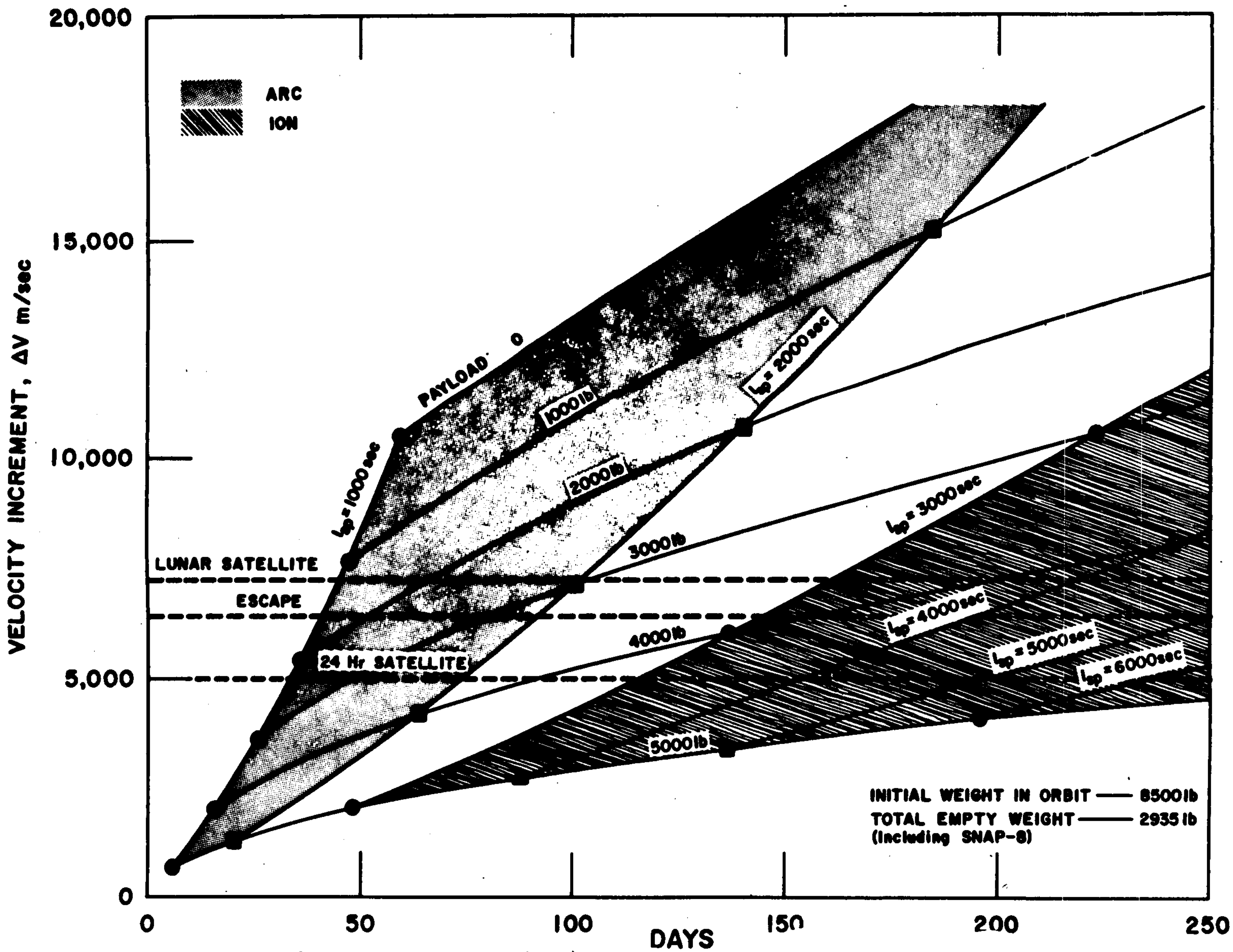

Fig. 15.4 Early mission capabilities of electric propulsion using 30 kw Snap-8 nuclear electric power supply.

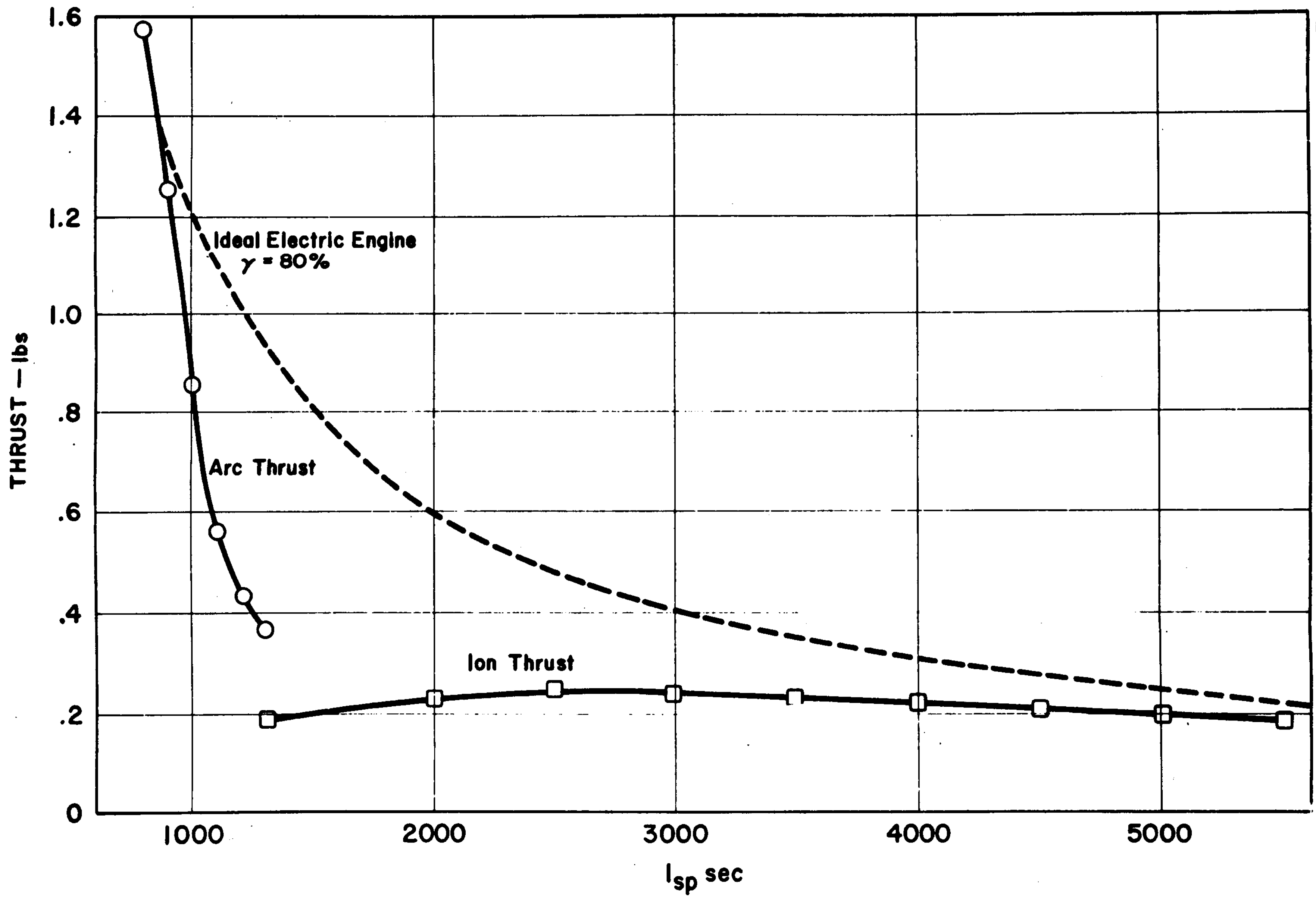

Fig. 15.5 Thrust versus specific impulse for 30 kw electric engines (existing state-of-the-art, December, 1961).

reduce the payload capability, but for a given payload the time of propulsion increases as the characteristics of the actual engines depart from those of the ideal engine. This illustrates the need to increase the energy conversion efficiencies of electric engines. Equally important is the need to increase the specific power of the power source, which would have the same effect on the overall mission.

Table 15.3 Typical Early Arc Engine Missions (Snap 8 Power Supply, Engine Efficiency = 80%, Start From 485 km Orbit With 8500 lb)

Mission	I_{sp} (sec)	Final Weights (lb) Useful Payload	Total	Propulsion Time (days)
Transfer to 24-hr orbit	1500	3140	6075	65
Transfer to 24-hr orbit	2000	3675	6610	75
Transfer to lunar orbit	2000	3000	5935	100

Specific early missions for which some details have been determined and discussed include:

1. Orbital transfer from 485-km nonequatorial orbit to 36,000 km-equatorial orbit (24-hr communications satellite). The power source can be used for operation of satellite equipment after transfer. Possible with 30 kw arc engine with I_{sp} of 1200 to 1600 sec [12 - 18].
2. Attitude control of satellites, using several 1 kw arc engines with I_{sp} of 700 to 1000 sec [15].
3. Mapping of radiation belts, solar corpuscular radiation and magnetic field between Earth and Moon, using 30 kw arc engine [15].
4. Lunar ferry, between Earth orbit and lunar orbit [19]. This mission requries specific power figures higher than are now available. Improved power sources would increase the likelihood of this mission.
5. Planetary missions using the 60 kw version of the Snap 8 and ion engines with specific impulses of 3000 to 5000 sec. These missions include Mars or Venus flyby, Mars or Venus capture, Mars orbiter, 15-deg out-of-the-ecliptic probe Refs. [18, 20 - 24].

The Snap 8 power supply will be the limiting factor in many of the above missions. Its lifetime of 10,000 hr will be an upper limit for propulsion times.

15.6 Advanced Missions: 1970-1980

The potential of electric propulsion will be fully utilized only if nuclear electric power sources are developed in progressive steps such as 100 to 300 kw, 1 to 4 Mw, and about 20 to 40 Mw. Lifetimes must be extended to several years and specific powers should be increased to 0.5 or 1 kw kg^{-1}, and possibly even higher.

Advanced missions for electric propulsion will include uses in Earth orbits, between the Earth and Moon, and to the planets or other targets in the solar system. Some of the proposed missions, in the approximate order of increasing total velocity increment are:

Attitude and position control of space vehicles	Ref. [15, 25]
Orbit correction for low-altitude satellite	Ref. [25]
Orbital transfer	Ref. [15, 25]
Supplies to space station	Ref. [25]
Lunar supply mission	Ref. [19, 25]
Mars or Venus flyby	Ref. [26]
One-way trip to Mars or Venus	Ref. [26 - 27]
15-deg out-of-the-ecliptic probe	Ref. [26, 29]
Manned trips to Mars or Venus	Ref. [25, 28]
Mercury satellite	Ref. [26]
Jupiter flyby	Ref. [20, 26, 27, 28]
Jupiter orbiter	Ref. [25 - 26]
45-deg out-of-the-ecliptic probe	Ref. [22]
Pluto flyby	Ref. [25]
Probe to 100 A.U. distance	Ref. [29]

Reference [13] also discusses these missions. Some of the more ambitious missions can be accomplished only by electric propulsion. It provides a unique propulsion concept for the unmanned and manned exploration of the solar system.

In cislunar space, electric engines can be used for ferries. Many trips will be made between the close Earth orbital supply station and the low lunar orbital station. These missions will require fairly high thrust, so the lower range of the specific impulse spectrum of electric engines is favored. Table 15.4 lists characteristics of electric engines and nuclear power supplies considered feasible for a lunar ferry during the 1970-1980 period.

Table 15.4 Characteristics of a Lunar Ferry Using Electric Propulsion

Earth Orbit Altitude	485 km
Lunar Orbit Altitude	32 km
Total Weight Leaving Earth Orbit (including payload and fuel)	400,000 lb
Total Weight of Propulsion Module (including electric engines, nuclear electric power supply, tanks, structure, etc.)	40,000 lb
Power Supply Output	4 Mw
Specific Power of Power Supply	0.3 kw kg^{-1}
Energy Conversion Efficiency of Engine	50%

The ferry will meet the Earth orbital supply stations and take on supplies and propellant for the electric engines. The electric engines will be started and the ship will spiral out until it reaches escape velocity. Then it will coast without propulsion to the neighborhood of the Moon, where the engines will be started again for the spiral inward until the final lunar circular orbit is achieved. During the engine-off periods the power is consumed by a ballast resistor and radiated to space. The low thrust of the ship will facilitate a rendezvous with the lunar orbital supply station. After delivering the supplies, the ferry reverses the flight plan. Adjustment of inclination to the orbital plane is performed during the propulsion spirals by thrust vectoring. The optimum supply-carrying capability is achieved at a specific impulse around 1300 sec. Since this maximum is very broad, especially on the high-impulse side, electric engines with impulses up to 2000 sec look promising for this mission. The maximum payload per year shifts to higher I_{sp} values with an increase of the power source weight. With a 60,000-lb propulsion module, the optimum I_{sp} is 1540 sec. The payloads for this ferry may consist of scientific instruments, food, tools, propellants, etc. It will be brought down to the surface of the Moon by soft landing vehicles using high-energy chemical propellants.

Payload capabilities and travel times are summarized in Table 15.5.

Table 15.5 Payloads and Travel Times for Lunar Ferry

Time for Single Round Trip	$\frac{V}{g_0}$	Payload for Single Round Trip with Engine Efficiency (η = 0.5)	Payload per Year With Engine Efficiency (η = 0.5)	Payload per Year with Efficiency of Advanced Electric Engine (η = 0.75)
(days)	(sec)	(lb)	(lb)	(lb)
33	1103	129,000	1,430,000	2,145,000
40	1310	159,000	1,460,000	2,190,000
50	1600	189,000	1,395,000	2,090,000
60	1882	212,000	1,295,000	1,940,000
70	2165	230,000	1,200,000	1,800,000
80	2450	243,000	1,110,000	1,665,000
90	2727	253,000	1,025,000	1,540,000
100	3006	263,000	950,000	1,425,000

The weight of propellant consumed decreases with increasing specific impulse, while the payload capability on a yearly basis shows a maximum at 1300 sec impulse.

Reference [19] discusses a flight plan for a lunar ferry with a spiral from the low Earth orbit out to the vicinity of the Moon, a capture maneuver to a lunar orbit and an immediate descent of each detached payload ship to the surface of the Moon.

For space exploration beyond the Moon, ion or magnetofluïddynamic (MFD) engines with specific impulses of 3000 to 30,000 sec will be used. The MFD engines are not as far advanced in their development as ion engines. However, in the period of 1970 to 1980 these engines should reach an advanced stage of development. Selection of the best engine for each mission must be made on the basis of performance and mission requirements. Arc engines are not considered for prime propulsion in interplanetary space. However, they have been discussed in connection with so-called high-powered maneuvers during the approach to a planet and for the initial spiralling out from a low Earth orbit to escape.

15.7 Manned Planetary Vehicles

The most intriguing challenge to the space flight planner is the conceptual design of a manned vehicle for planetary exploration. Electrically propelled vehicles will be particularly suited for this mission because of their liberal payload ration which will allow ample equipment for the crew, and will also allow the necessary shielding against solar protons and Van Allen radiation.

Design studies of electrically powered spaceships for manned planetary flight will probably begin toward the end of this decade. By that time, nuclear-electric power supplies of the 20 to 40 Mw class will have entered the planning stage; our knowledge of meteor fluxes and space radiation will be more realistic than it is now, and many details of the atmosphere and the surface of Venus and Mars will be known from exploratory probes.

The following paragraphs contain, in a very condensed form, the results of a study for a conceptual design of a Mars expedition. More details, including arguments for the selection of certain parameters and schemes, may be found in Ref. [30].

15.7.1 The Master Plan

The objective of the planetary expedition will be manned surface exploration of Mars. A fleet of five electrically propelled vehicles, assembled and checked out in a 320 km orbit around Earth, will travel together toward the planet Mars and establish a satellite orbit at an altitude of 300 km where the atmospheric pressure is 10^{-6} mm Hg. Atmospheric drag will be overcome by continuing low-level operation of the ion engines. Each ship will carry three passengers and their equipment. The basic design of all ships will be identical; however, three ships (type A) will carry landing craft for Mars and a smaller amount of propellant than the other two (type B). The first landing craft to descend will carry only equipment such as surface vehicles, shelters, and tools. The second will transport a small crew, while the third will serve as a backup for the second. Each of the landing craft has the capability of transporting all landing crew members back to the orbiting ships.

The design of the interplanetary vehicle will provide for rotation of the crew compartments at a rate which results in a simulated gravity of about 0.1g. The radius should be large enough to keep Coriolis forces small. It is anticipated that sufficient experience with orbital operations will be accumulated by 1970 or 1975 to confirm, or to improve, the suitability of this assumed gravity.

Each vehicle will contain a radiation shelter to afford protection for the crew members against solar proton outburst and against Van Allen radiation. The shelter will consist of thick layers of shielding material, surrounded by equipment and other supplies such as drinking water, oxygen tanks, propellant for the return flight, samples from Mars, etc. (Fig. 15.7 insert). Solar outbursts will require shelter protection for not more than a few hours; traveling through the Van Allen belts will make it necessary that the crew members stay "in the doghouse" for 22 or 23 hours a day for a period of several days.

It is planned that all five ships will make the complete round trip back to the 320-km Earth orbit. In case of failures, two ships could conveniently accommodate the 15 crew members; in an emergency, one ship could carry the crew back into the Earth orbit.

15.7.2 Flight Plan

The flight plan of a Martian round-trip vehicle depends upon the acceleration of the vehicle, and also upon the relative motions of Earth and Mars. As shown by Moeckel [31], the total round-trip time is shortest when the return leg passes around the Sun inside the Earth's orbit to catch the Earth on the other side (Fig. 15.6). In this way, a propulsion system with a specific power of 0.5 kw kg^{-1} can achieve a complete round trip to Mars, including a waiting time of 40 days in the Martian orbit, in about 560 days. Spiralling around the Earth until escape will take about 48 days, while spiralling down to Mars will require 21 days. The Earth-Mars transfer takes about 140 days, or roughly one half of the Hohmann transfer time. The return transfer time is almost twice as long because of the indirect approach, but a direct transfer after 40 days of Martian exploration would not result in a rendezvous between ship and Earth.

The thrust vector will be tangential to the flight path and constant in magnitude during the spiralling periods. A program of variable thrust may be applied for the planetary transfer periods, optimized for

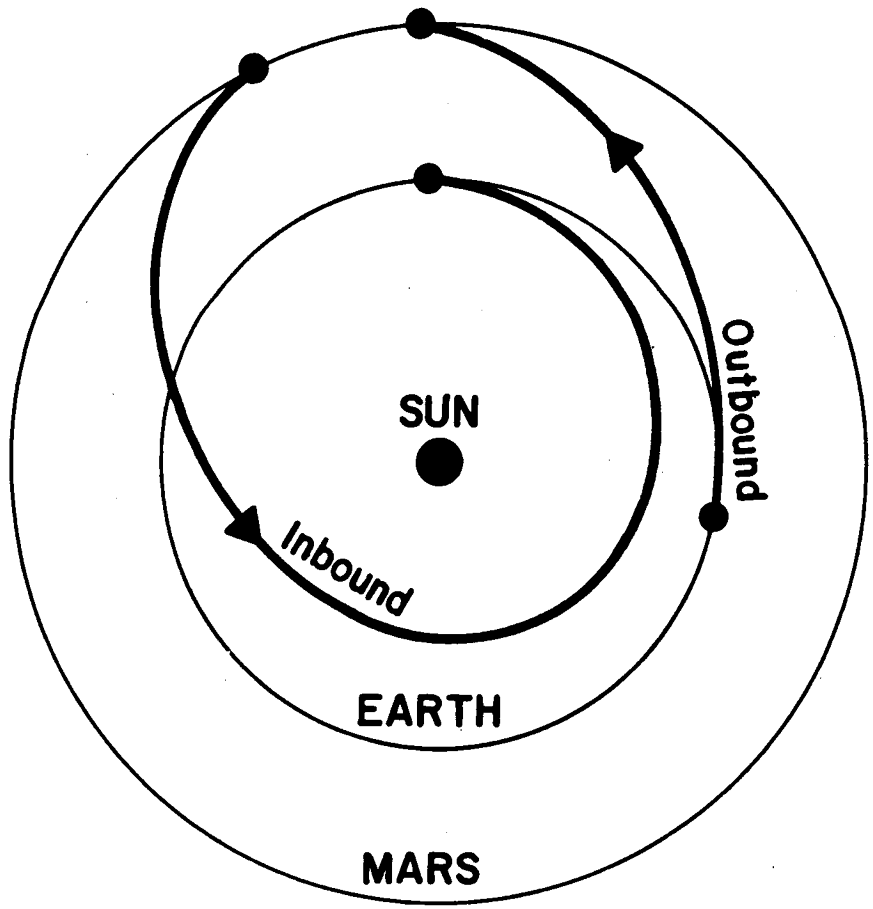

Fig. 15.6 Trajectories of Martian round trip. Direct outbound and indirect inbound legs (Moeckel, Ref. 33).

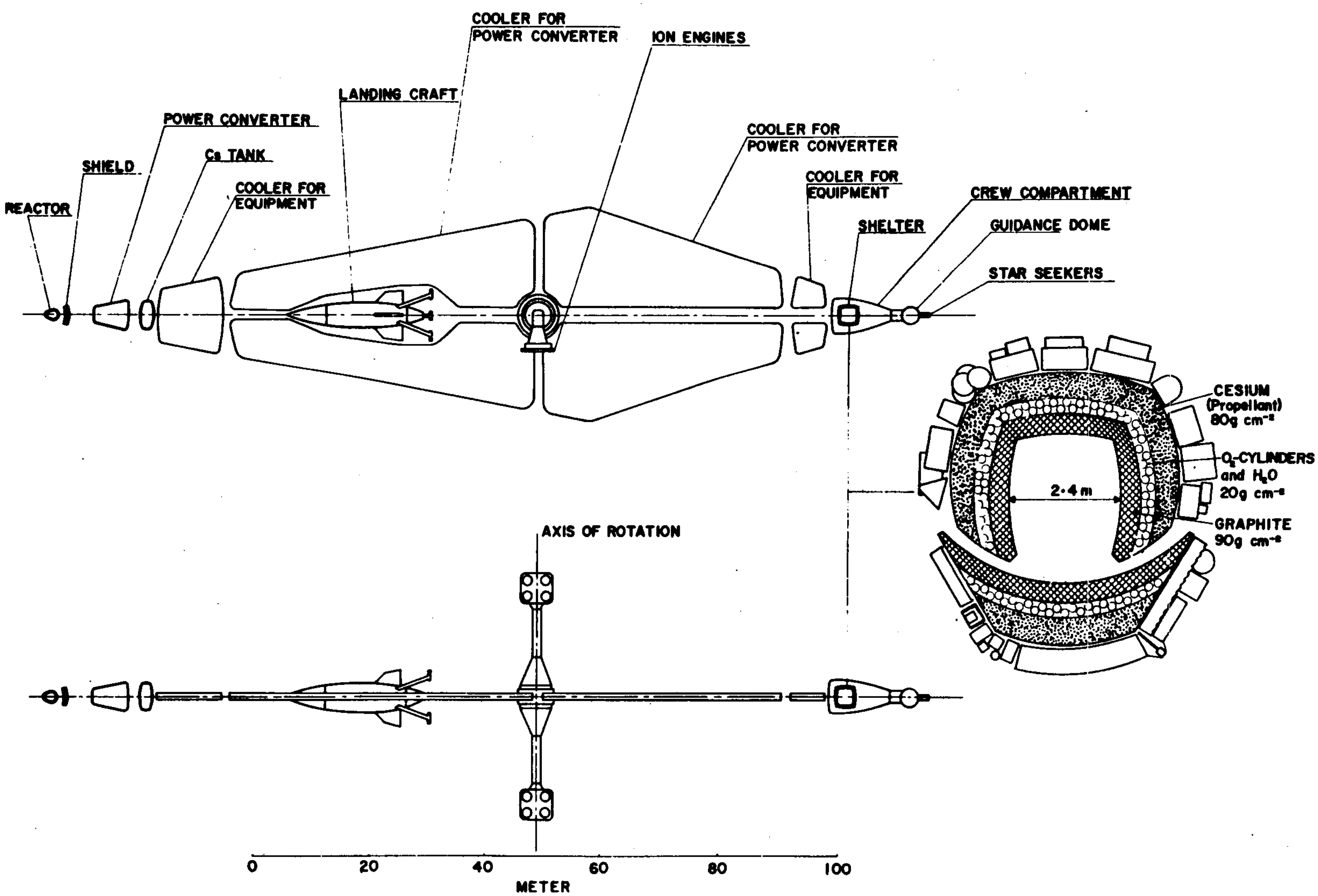

Fig. 15.7 Manned electric spaceship for Martian round trip.

maximum payload on the round trip. It has been shown that a variable thrust program on planetary transfer trajectories results in slightly greater payload capabilities than a constant thrust program [32]; however, the gain is not too great and may not justify the added complexity of a variable thrust system [33]. This Mars project assumes constant thrust throughout the flight, with several brief thrustless periods during the transfer phases. The direction of the thrust vector with respect to the path tangent, however, varies according to a preset program.

Guidance of the spacecraft is achieved by a system of planet and star seekers which constantly read the angles to the planets with reference to a fixed star system. These angles, together with the known positions of the planets as functions of time, permit accurate determination of the ship's location in three-dimensional space. Comparison of this location with the predetermined trajectory will show deviations, if any should develop. Corrective maneuvers will then be initiated immediately by a change of thrust vector direction.

Besides the planet and star seeker system, the ship will carry accelerometers and integrators in three directions to give direct readings of accelerations, velocities, and distances. These readings will enable the crew to monitor constantly the performance of the propulsion system.

Note that the continuous propulsion of an electric planetary vehicle renders the guidance problem relatively easy. Comparing it with the guidance of a chemically propelled planetary vehicle is like comparing the guidance of an airplane flown under visual flight rules over familiar territory with the guidance of a ballistic missile to the same target.

15.7.3 Performance and Design Data

The principle data of each of the five ships are listed in Table 15.6 and Table 15.7.

Table 15.6 Mass Requirements of Proposed Manned Mars Ship

Mass	Type A (tons)	Type B (tons)
Power Plant	80	80
Shelter	55	55
Landing Craft	70	--
Pure Payload	40	40
Propellant	115	185
Total	360	360

Table 15.7 Performance and Design Data of Manned Mars Ship

Parameter	Quantity or Rate
Specific Power	0.5 kw kg^{-1}
Total Power	40 Mw
Initial Acceleration	1.5×10^{-4} G
Exhaust Velocity	140 km sec^{-1}
Characteristic Velocity	90 km sec^{-1}
Voltage	14,500 v
Current	2740 amp
Total Trip Time	560 days
Thrust	56 kg
Propellant Flow Rate	3.75 g sec^{-1}
Active Ion Motor Area (20 ma cm^{-2})	14 m^2
Rotational Rate of Ship	1 rev/40 sec
Artificial Gravity in Crew Compartment	0.1g

Overall efficiency of the propulsion systems has been assumed as 80 per cent. During the escape spiral around the Earth, the propellant for return will be used as additional shielding mass around the shelter. On the return trip, samples from Mars will be stacked around the shelter. The time within the Van Allen belts during return will be only about one half as long as the time on the outbound leg.

The vehicle must be designed and equipped to meet the following requirements: living and working accommodations for three, and, in case of emergency, for up to 15 crew members; separation between reactor and crew compartments of about 150 m; rotation of crew compartments to provide approximately 0.1g acceleration; thrust vector direction through center of mass; Martian landing vehicle to be carried on outbound leg only.

It is assumed that the electric power generator will be of the thermionic or of the plasma dynamic type. Rotation of the vehicle will be initiated at the beginning of the trip by solid spin rockets and will be sustained by an occasional boost from solid rockets. Figure 15.7 shows a sketch of the ship, illustrating the arrangements of the major components. Propellants will flow from the two tanks at such rates that the center of mass of the vehicle remains fixed. Prior to the return trip, propellants from around the shelter will be distributed between the two tanks to preserve the center of mass.

The orientation of the ship will be such that its rotational plane coincides with the plane of the trajectory. In this case, the rotational axis need not change when flight direction changes. Necessary changes of the rotational plane will be effected by varying the thrust vector directions of the two thrust units. A model of the manned Mars ship is shown in Fig. 15.8.

It is anticipated that a manned Martian expedition of this kind will be feasible during the 1980-1985 period. Ten years ago, a mission of this type would have been considered impossible. Within the past few years, however, great strides have been taken in the development of electric engines; the remaining problems are clearly foreseen and appear surmountable; progress is being made in the development of larger power supplies with higher specific powers; missions have been defined and analyzed. Electric propulsion is indeed a new technology and one with vast potential for the exploration of outer space.

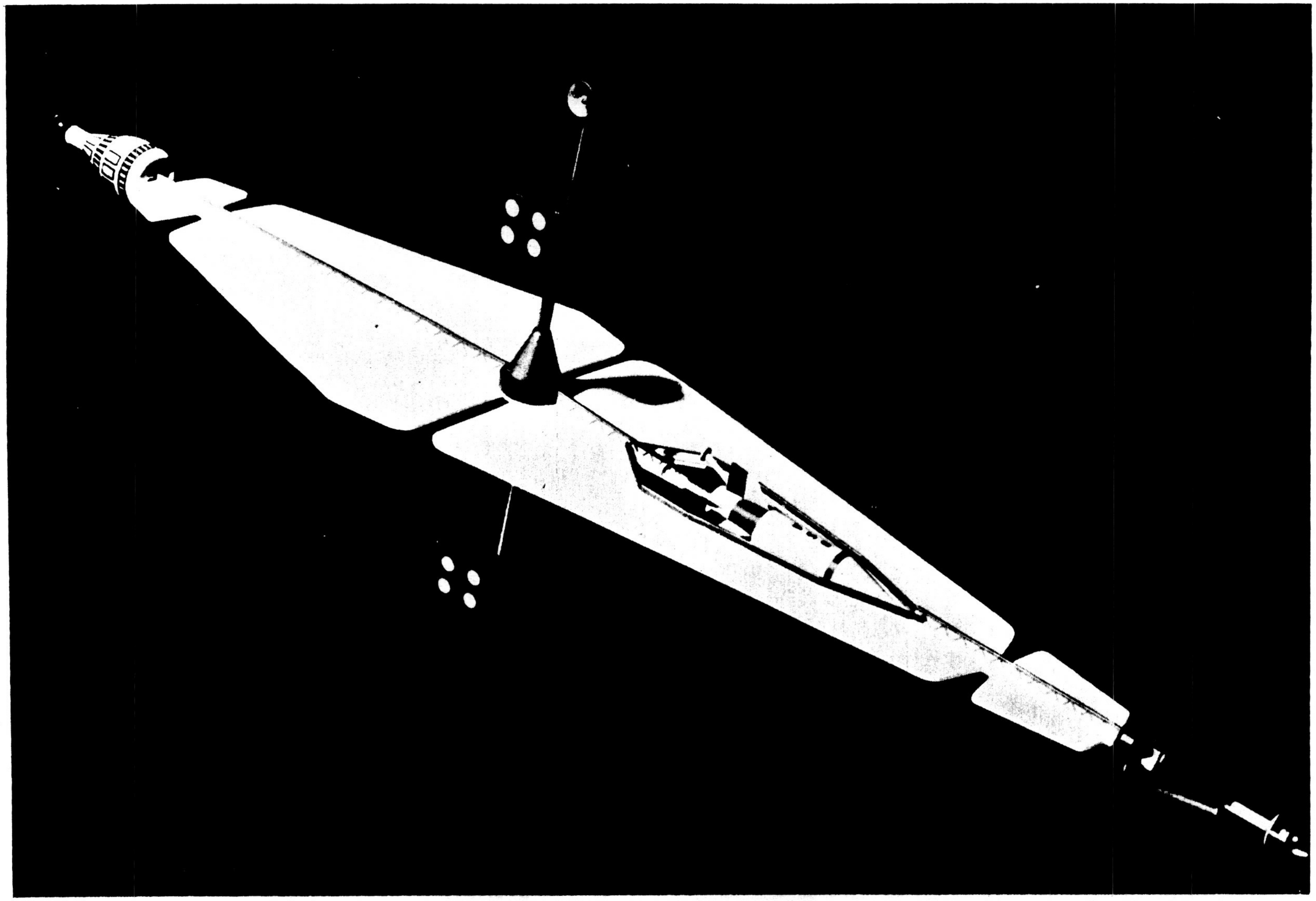

Fig. 15.8 Model of manned Martian electric spaceship.

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