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Thrust-Unit Requirements for Electrically Propelled Spacecraft

David G. Elliott

Jerome H. Molitor

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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John J. Paulson, Chief
Advanced Propulsion
Engineering Section

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ABSTRACT

This report presents a preliminary study of the requirements that must be met by electric thrust devices in order to be used with interplanetary spacecraft. Two missions, a Mars orbiter and a Jupiter capture, chosen as representative of the time periods following 1965 and 1970, respectively, are analyzed to determine the thrust and specific-impulse requirements of an electric propulsion system. The state-of-the-art of electric thrust devices is discussed, and it is concluded that with expected advances ion motors can meet all of the requirements of interplanetary missions, with magnetohydrodynamic motors a promising backup.

I. INTRODUCTION

Many interplanetary and deep-space scientific missions are impossible to accomplish with foreseeable chemical or nuclear vehicles. Electric propulsion will be mandatory for such missions and will be preferable for many others. This report presents a preliminary study of the requirements placed on the electric thrust device by the needs of such scientific missions and the prospects for meeting these requirements.

Missions that will be feasible for first-generation electrically propelled spacecraft during the present decade are Mars and Venus probes and orbiters; solar, Mercury, and Jupiter probes; and out-of-the-ecliptic probes. Additional missions possible for second-generation electric spacecraft during the 1970's are Mars and Venus split-capsule orbiter-landers; Mercury, Jupiter, and Saturn orbiters; Neptune, Uranus, and Pluto probes; high-angle out-of-the-ecliptic probes; and solar-system escape. Electric propulsion is particularly attractive for orbiters and split-capsule landers because the slow capture spiral trajectory permits accurate survey of planetary radiation belts and upper atmosphere. In addition, the electric power available permits wideband data transmission from the orbiter, including data relayed from a landing capsule.

The first electric spacecraft flights are scheduled to begin in 1965 using *Atlas–Centaur* as the booster and SNAP-8 as the spacecraft powerplant. After one or two test flights, the scientific missions that will possibly be flown are a Mars or Venus probe or orbiter, a lunar orbiter, or an out-of-the-ecliptic probe. Larger electric powerplants will be developed by 1970 for more advanced electric spacecraft launched by both *Centaur* and *Saturn*.

II. THRUST AND SPECIFIC-IMPULSE REQUIREMENTS

A. Centaur–SNAP-8 Spacecraft

One of the most interesting and desirable missions for a SNAP-8 powered spacecraft is an orbit of Mars. The spacecraft would be launched into an initial Earth orbit by an *Atlas–Centaur* booster. The electric propulsion system would cause the spacecraft to spiral out from Earth until the spacecraft escaped at about the distance of the Moon. The propulsion system would continue to operate throughout transfer to a Mars orbit, except for a mid-course coast period. After matching Mars' velocity, the spacecraft would spiral down to an orbit around Mars. The thrust unit would then be turned off, freeing the electric powerplant for wideband data transmission during the remaining powerplant life.

Figure 1 shows the terminal mass of the spacecraft in a 500-nm Mars orbit as a function of the total flight time. An initial spacecraft mass of 7500 lb in a 1000-nm Earth orbit is assumed. The terminal masses shown are preliminary estimates valid to within about 10%.

Terminal mass is shown as a function of flight time for thrusts of 0.35, 0.40, and 0.45 lb with specific impulse as a parameter; thrust and specific impulse would be held constant during the flight except during the coast period when the thrust unit would be turned off. Two facts are immediately apparent from Fig. 1: (1) flight time depends mainly on thrust, independent of specific impulse, and (2) terminal mass depends mainly on specific impulse, independent of thrust.

The nominal lifetime of the SNAP-8 powerplant will be 10,000 hours (417 days) and it will not be restartable; 10,000 hours will thus be the upper limit for a mission. Figure 1 shows that this necessitates a thrust of at least 0.40 lb.

Table 1. Preliminary thrust-unit requirements for *Centaur*–SNAP-8 interplanetary spacecraft

Input power	60 kw
Minimum thrust	0.40 lb
Minimum specific impulse	4000 sec
Lifetime	8000 hr
Maximum weight	300 lb
Maximum diameter	10 ft

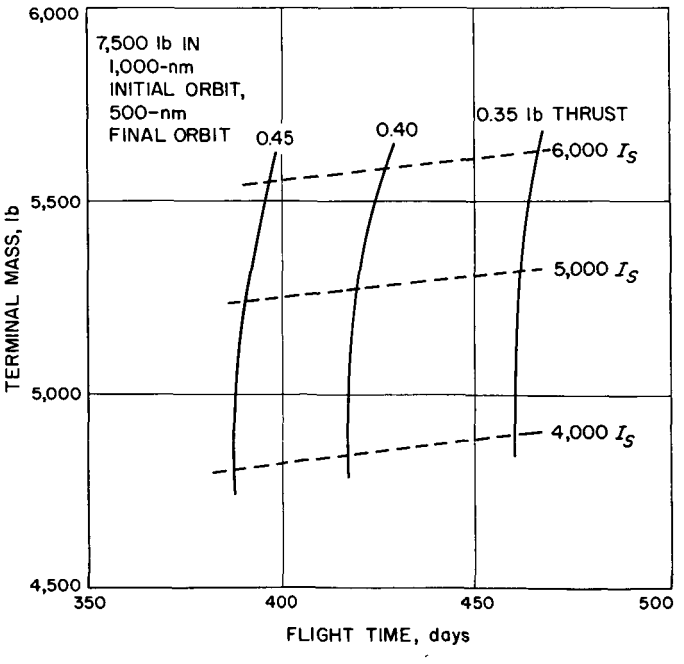

Fig. 1. *Centaur*–electric Mars orbiter capability

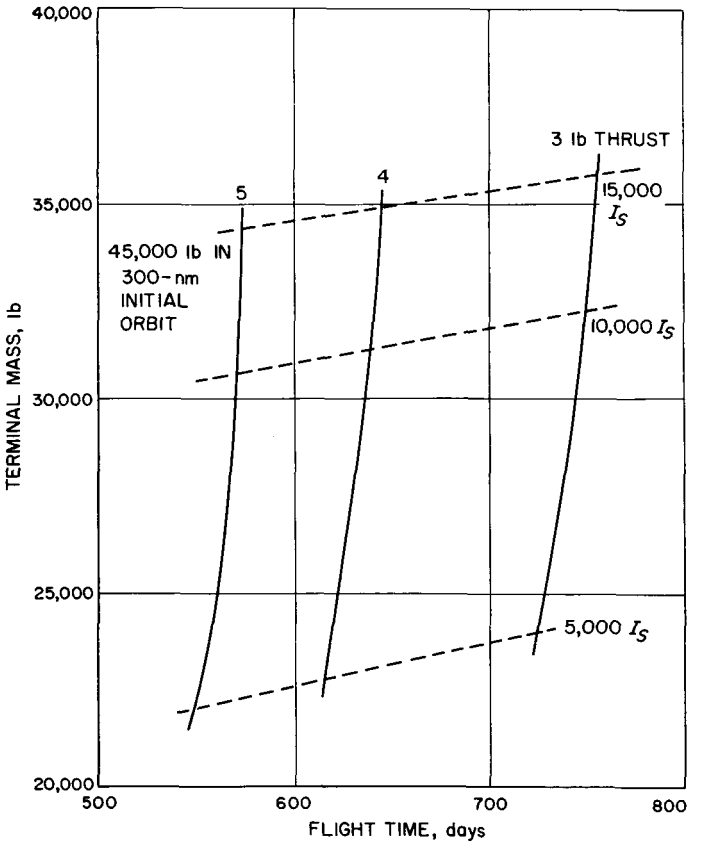

Fig. 2. *Saturn*–electric Jupiter capture capability

The weight of the powerplant and shielding, thrust unit, propellant tank, guidance and control equipment, telemetry, and structure will be at least 4500 lb. Hence, for a useful scientific payload the terminal mass must be at least 5000 lb. Figure 1 shows that a specific impulse I_s above 4000 sec must be provided. Since, for these conditions, the coast will be approximately 2000 hours, the thrust-unit lifetime must be 8000 hours.

To provide sufficient power for the above conditions, the 60-kw version of SNAP-8 must be employed. Accordingly, the preliminary requirements shown in Table 1 can be given for the thrust unit for interplanetary missions with the *Centaur*–SNAP-8 system.

The initial flights in 1965, however, will employ a prototype SNAP-8 of only 90-day nominal lifetime and, perhaps, of only 30-kw power. For these flights, therefore, the lifetime of the thrust unit need only be about 2500 hours and the thrust 0.2 lb at 30 kw.

B. Saturn–*Electric Spacecraft*

Terminal mass versus flight time for a *Saturn*-launched electric spacecraft is shown in Fig. 2. The mission is a Jupiter capture; the spacecraft matches Jupiter's velocity, permitting establishment of an elongated orbit about Jupiter with minimal additional propellant expenditure. Chemical or direct-nuclear flight time for this mission is about 800 days and the terminal mass is 10,000 lb at most. It is seen that with the electric system a 640-day flight can be achieved using 4.0 lb thrust, and a 31,000-lb terminal mass delivered using 10,000 sec specific impulse. This combination requires a powerplant of about 1.0 Mw. Weight estimates for such a powerplant range from 10,000 lb to 20,000 lb; corresponding gross payloads are 21,000 lb and 11,000 lb, respectively. The propulsion time is approximately 400 days. The approximate thrust-unit requirements to be anticipated for *Saturn*-boosted electric spacecraft are shown in Table 2.

Table 2. Anticipated thrust-unit requirements for *Saturn*–electric interplanetary spacecraft

Input power	1.0 Mw
Thrust	4.0 lb
Specific impulse	10,000 sec
Lifetime	10,000 hr

III. GUIDANCE AND CONTROL REQUIREMENTS

The long propulsion durations of electric spacecraft require that any torques for correcting thrust misalignments or for producing steady precession of a rotating powerplant (as in escape and capture spirals) be provided by electric propulsion, since the total impulses involved are too great for any cold-gas or chemical attitude-control system. Gimballing of the main thrust units seems the most likely possibility.

The largest external disturbing force will be meteoroid impacts. Studies show that the largest angular momentum likely to be imparted to an electric spacecraft (with a probability of 3/1000) is 70 ft·lbf·sec. However, the motion from even this extreme disturbance can be removed in 12 minutes, for example, by a thrust of 0.02 lb acting on a 5-ft moment arm. Thus, there appears to be no need for high-thrust devices on an electric spacecraft.

IV. PRESENT STATE-OF-THE-ART OF ELECTRIC THRUST DEVICES

Three basic types of electric thrust devices are presently known: the electrothermal, the electromagnetic (or magnetohydrodynamic), and the electrostatic.

For low specific impulses, electrothermal thrust units are well developed. A particular type is the arc-jet motor shown in Fig. 3, which is a photograph of a Plasmadyne arc-jet motor in operation. The propellant, hydrogen, is heated by an electric arc and exhausted through a converging-diverging nozzle. The efficiency of arc-jet motors is high at specific impulses up to 1000. Lifetime is a problem because of the high operating temperatures, but arc-jet missions have durations of 100 days or less and such lifetimes appear attainable.

For medium and high specific impulses, magnetohydrodynamic, or MHD, propulsion is very promising. A particular type is the steady-flow crossed-field accelerator, such as the Northrop MHD motor shown in Fig. 4. A conducting plasma from an arc-jet motor enters the accelerator and passes through crossed electric and magnetic fields. The plasma leaves the accelerator at increased velocity. MHD motors are still in the laboratory stage and neither high efficiency nor long lifetime has yet been conclusively demonstrated at specific impulses between 2000 sec and 10,000 sec.

For the highest specific impulses, electrostatic propulsion currently performs best. One particular type is the cesium ion motor seen in Fig. 5, which shows a small Electro-Optical Systems ion motor in operation. Cesium is ionized by a heated porous tungsten emitter and the ions are then accelerated by voltages supplied to the electrode assembly.

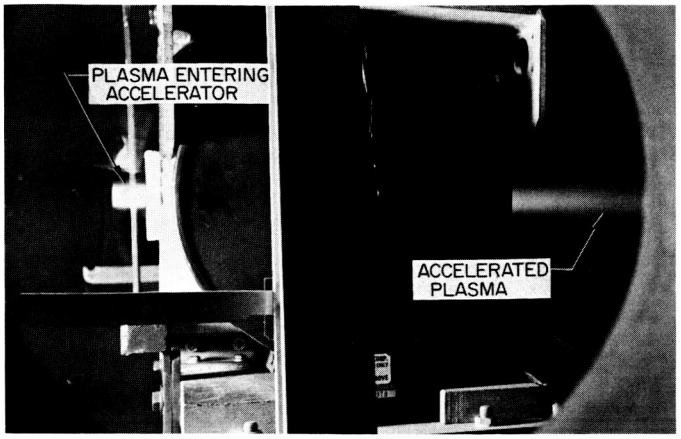

Fig. 4. Cross-field accelerator (photograph courtesy of Space Propulsion and Power Laboratory, Norair Division of Northrop Corporation)

The efficiency of ion motors is compared with that of arc-jet and MHD motors in Fig. 6. Efficiency is defined as

$$\eta = \frac{F I_s}{45.85\, P} \tag{1}$$

and specific impulse is defined as

$$I_s = \frac{F}{\dot{m}} \tag{2}$$

where F is the thrust in lbf, P is the total input power in kw, and $\dot{m}$ is the propellant flow rate in lbm/sec.

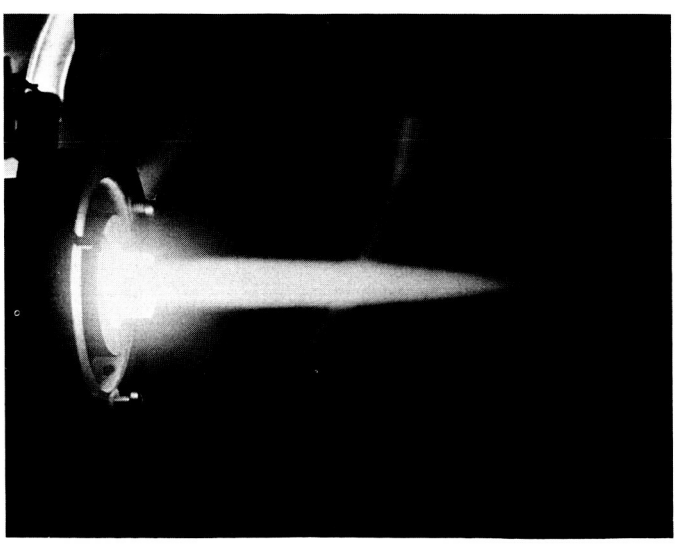

Fig. 3. Arc-jet motor (photograph courtesy of Plasmadyne Corporation)

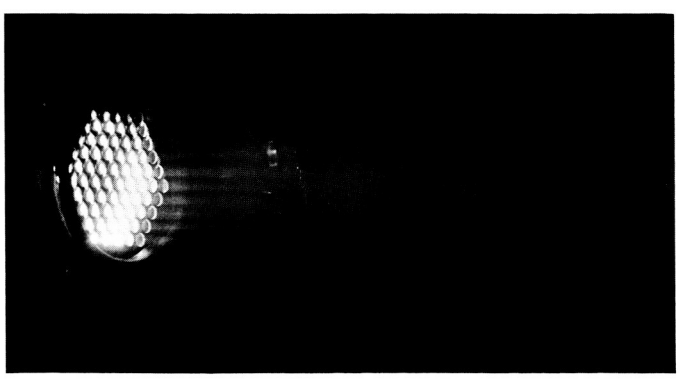

Fig. 5. Cesium ion motor (photograph courtesy of Electro-Optical Systems, Inc.)

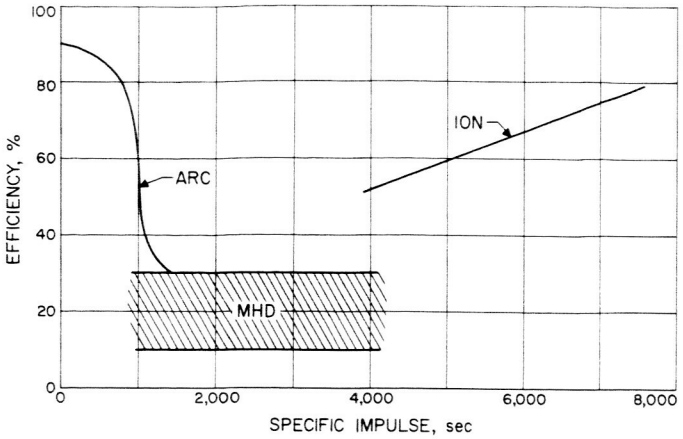

Fig. 6. Currently attainable efficiencies of electric thrust devices

The efficiencies shown in Fig. 6 are ones which have already been achieved or which are targets for current development programs.

Arc-jet motors have good efficiencies up to specific impulses of around 1000, but efficiency drops off rapidly above that level. MHD motors are promising at somewhat higher specific impulses, but have yet to demonstrate efficiencies much above 30%. An ion motor developed by Lewis Research Center, employing an electron bombardment source, has demonstrated the efficiency shown by the solid curve in Fig. 6. Cesium ion motors currently under development will achieve comparable efficiencies.

The lifetime of an ion motor depends on obtaining good focusing of the ion beam so that it does not impinge on and erode away the accelerating electrode structure. If beam interception can be held to 10^{-2} or 10^{-3}%, as appears possible, the necessary lifetime for large ion motors should be achieved.

The ion motor does have one unique problem: production of a neutral exhaust. The ions and electrons are separated before acceleration and must be recombined downstream of the thrust unit. If there is insufficient electron density in the exhaust, a net positive space charge exists and, as a result, the ion beam diverges, the ions stall, and little or no thrust is obtained.

Progress is being made in laboratory testing and analysis of space charge neutralization, but it will be desirable to perform actual space tests to demonstrate conclusively that the ion beam can be neutralized in space. A possible payload configuration for the space testing of ion motors is shown in Fig. 7. The payload is about 3 ft long, weighs 140 lb, and can be launched by a *Scout* rocket on a high-altitude trajectory above the atmosphere for about 1.4 hours. The vehicle carries a 65-lb battery in the lower box, and the telemetry equipment and an inverter for supplying the necessary high voltages in the upper box. Ion motors having up to 10 millipounds thrust could be carried and operated for the duration of the flight. The ion motor would be mounted off-center and its thrust measured by observing the rate of change of the spin rate ω_3. The precession angle θ and precession rate $\dot{\phi}$ would complicate signal-strength determination of ω_3 but facilitate internal accelerometer measurements. With the payload initially spinning at 160 rpm, a 10-mlb ion motor would de-spin it to 0 rpm in 45 minutes and spin it up to 75 rpm in the opposite direction by the end of the flight.

Scout flights are scheduled beginning in 1962, initially to demonstrate beam neutralization, and later to test the final ion-motor design to be used on the first electric spacecraft missions. The mission motor can be space tested on such a small scale because it will probably be a cluster of small-thrust modules, each of perhaps 10-mlb

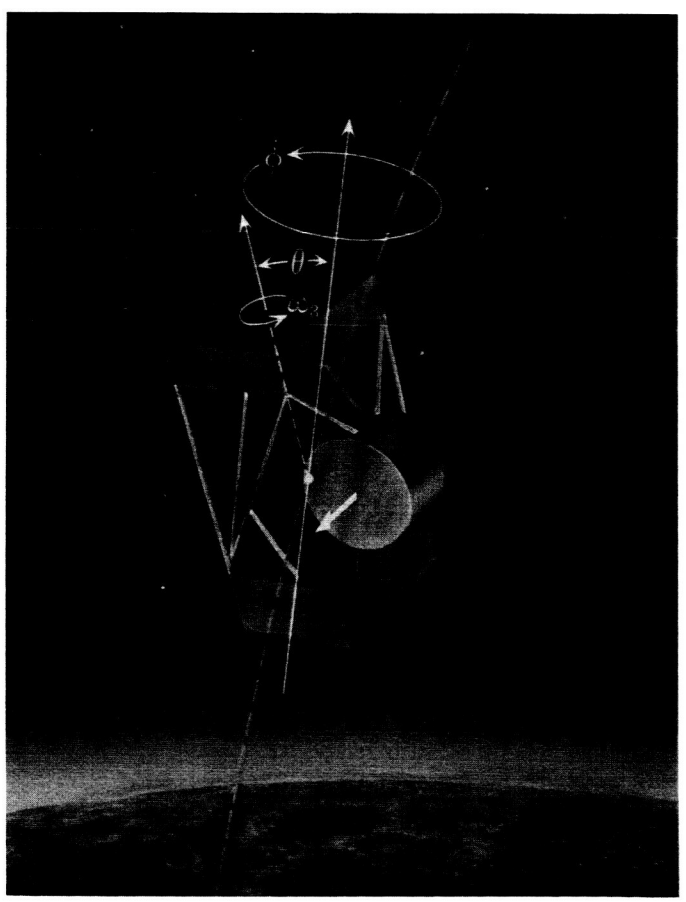

Fig. 7. Possible ion-motor test vehicle configuration

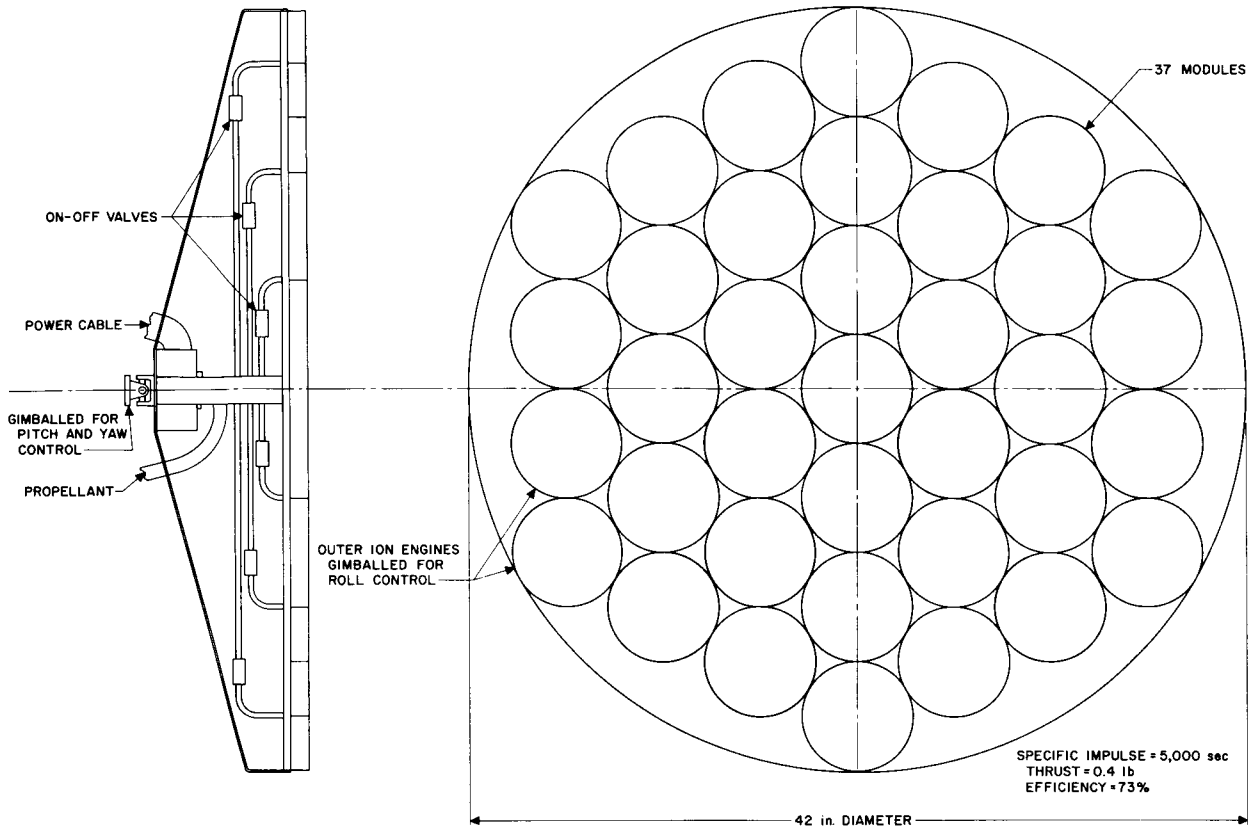

Fig. 8. Sixty-kilowatt ion motor

thrust, and a prototype of these modules can be tested on a *Scout* payload.

To illustrate, Fig. 8 shows a possible 60-kw ion-motor cluster designed for a specific impulse of 5000 sec. The cluster consists of thirty-seven ion motors, each 6 in. in diameter and providing about 10-mlb thrust. The over-all diameter of the cluster is 42 in.

Considering next the state-of-the-art of electric thrust devices as related to guidance and control, the primary criterion for that application is the ratio of total impulse to gross weight of the attitude-control system. Figure 9 shows estimated impulse-to-gross-weight ratio as a function of total impulse for arc-jet and ion motors having thrusts of 0.001, 0.01, and 0.10 lb—the range of interest for a 60-kw spacecraft. For these thrusts arc-jet motors are assumed to weigh 5.0, 7.5, and 10.0 lb, respectively, and ion motors 10, 20, and 60 lb, respectively. The specific impulse is assumed to be 600 for the arc-jet (ammonia as the propellant) and 5000 for the ion motor. Propellant tank weight is assumed equal to propellant weight for the small total impulses considered in Fig. 9. It is seen that arc-jet and ion motors are superior to a cold-gas system at impulses above 200 lbf · sec, with arc-jet motors

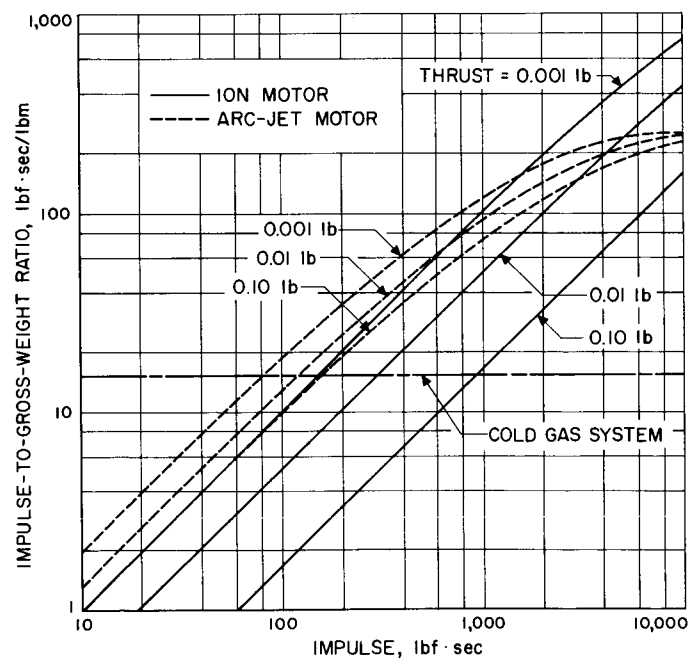

Fig. 9. Estimated impulse-to-gross-weight ratio for small electric attitude-control systems

providing greater thrust than ion motors for a given system weight (and given power). At impulses above 1000 lbf · sec the impulse-to-gross-weight ratio for the arc-jet system approaches a maximum of 300 lbf · sec/lbm for the assumed specific impulse and tank weight; but at these higher impulses the use of liquid hydrogen becomes feasible, raising the peak impulse-to-gross-weight ratio to perhaps 800 lbf · sec/lbm. For impulses much above 10,000 lbf · sec, gimballing or thrust modulation of the main thrust unit on an electrically propelled spacecraft would be preferable to an auxiliary attitude-control system.

V. SELECTION OF THRUST UNITS FOR INTERPLANETARY MISSIONS

It is apparent from a comparison of the thrust-unit requirements listed in Tables 1 and 2 with the thrust-unit performance capabilities shown in Fig. 6 that only ion motors are presently in contention for interplanetary missions. However, progress is being made in MHD motors, and they may ultimately prove competitive or superior, especially for missions requiring relatively low specific impulses.

Arc-jet motors appear inherently limited to specific impulses below those required for interplanetary missions. They are, however, advantageous for missions of lower energy and shorter duration than those considered here, and may achieve use in satellite and lunar missions.

Among ion motors themselves there are several types which appear capable of meeting the requirements listed in Tables 1 and 2. These include various types of electron-bombardment motors and the cesium contact-ionization motor. The choice among these for electric spacecraft will be based chiefly on demonstration of the necessary high efficiency and long lifetime utilizing, preferably, a storable propellant.

For guidance and control of an electrically propelled interplanetary spacecraft there appear to be no thrust requirements exceeding the capability of arc-jet or ion motors. The choice between the two depends on the total impulse and thrust required and on operational convenience.

VI. ACHIEVEMENT OF REQUIRED ION-MOTOR CAPABILITY

It is seen from the first two items in Table 1 that the primary requirement on the thrust unit for the SNAP-8 spacecraft is an input power of no more than 150 kw per pound of thrust. The corresponding required thrust-unit efficiency is presented as a function of specific impulse in Fig. 10 and compared, for illustrative purposes, with the efficiencies anticipated, at the current rate of development, for cesium ion motors in 1964.

The predicted 1964 ion-motor efficiency curve is based on advanced but realistic values for acceleration–deceleration ratios, current densities, and porous tungsten emitter widths. The maximum acceleration–deceleration ratio

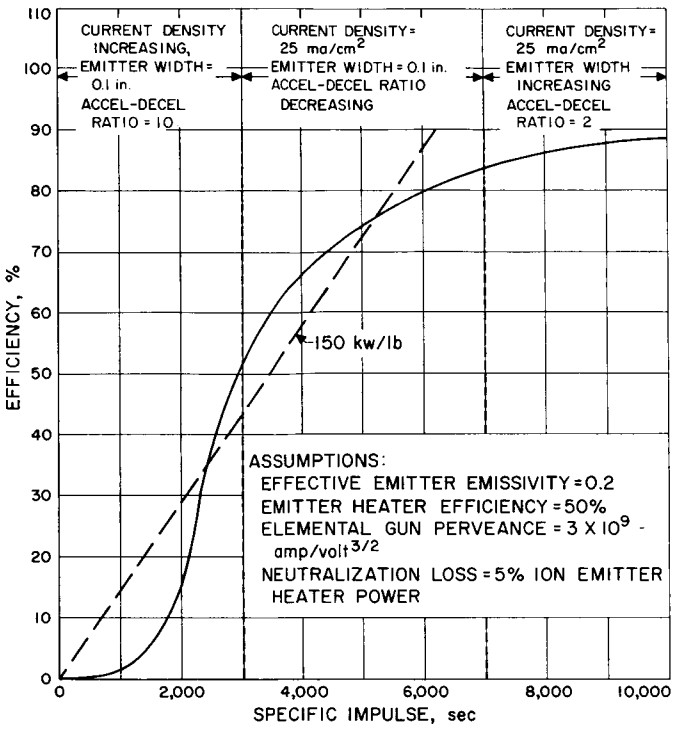

Fig. 10. Comparison of required and predicted 1964 ion-motor efficiency

considered is 10; the minimum is 2. However, for specific impulses greater than 4000 sec, i.e., for interplanetary missions, the range of interest of acceleration–deceleration ratios falls between 6 and 2. These latter values approximate those being used in present ion-motor development programs and, therefore, should be available.

The maximum current density assumed is 25 ma/cm^2. Current densities of the order of 12 ma/cm^2 have been achieved with approximately 1% neutral efflux and there seems to be no fundamental limitation preventing much higher current densities. However, a practical limit may be set by the maximum allowable neutral efflux or the difficulty of focusing high-density beams.

The minimum emitter width considered is 0.1 in. At the higher specific impulses this size requirement is relaxed, the emitter width increasing to 0.17 in. at 10,000 sec. This latter value approaches the present state-of-the-art of porous tungsten fabrication.

The size and weight of future ion motors does not appear to be a serious problem. The 60-kw cluster shown in Fig. 8 should weigh no more than about 250 lb, and by a fortunate combination of circumstances the sizes and weights of motors for future higher-powered spacecraft will not increase proportionally with power. The

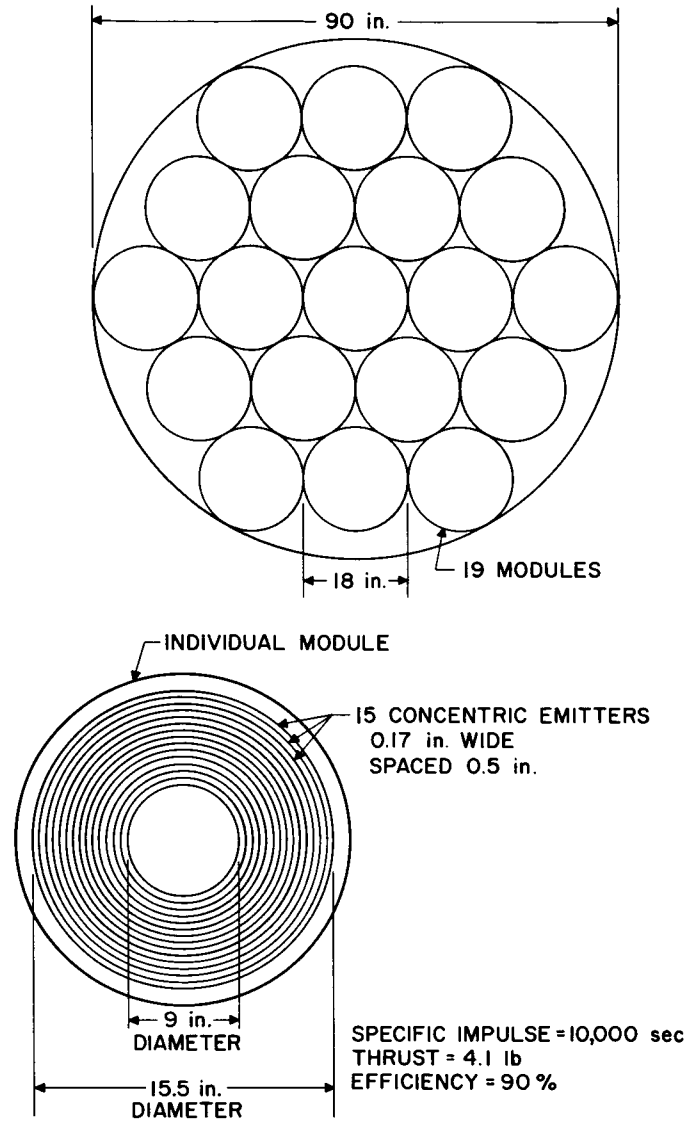

Fig. 11. One-megawatt ion motor

circumstances permitting this are (1) that larger powerplants will have lower specific weights, permitting reasonable spacecraft accelerations at higher specific impulses and (2) that the ion-accelerating voltage increases with the square of the specific impulse, with a corresponding decrease in ion current for a given power. A 1.0-Mw cesium ion motor, for example, operating at 10,000 sec specific impulse and 90% efficiency would have an over-all accelerating voltage of 6600 volts and an ion current of 137 amp. Assuming a current density of 25 ma/cm^2, the required emitter area is 5.9 ft^2. If the emitter is divided into annular rings 0.17 in. wide spaced 0.5-in. apart between centers, and if nineteen thrust modules are used, the 1.0-Mw ion motor would have approximately the dimensions shown in Fig. 11.

VII. CONCLUSIONS

1. For the first electric spacecraft flights beginning in 1965, the thrust unit must deliver 0.4 lb thrust at 60 kw input power (or 0.2 lb thrust at 30 kw) at a specific impulse no lower than 4000 sec. It must have a lifetime of about 2500 hours. For subsequent mission flights the lifetime must be increased to about 8000 hours.

2. For second-generation electric spacecraft the anticipated thrust-unit requirements are, very roughly, the following: 4-lb thrust, 1.0-Mw input power, 10,000-sec specific impulse, 10,000-hour lifetime.

3. Currently, only ion motors appear capable of meetin these requirements. However, MHD motors offer promise, and could ultimately be superior. Arc-jet motors appear inherently unsuitable for interplanetary missions because of their low specific impulse.

4. Arc-jet, ion, or MHD motors appear adequate for all guidance and control requirements.

5. Several types of ion motors appear capable of development to the efficiencies and lifetimes required for both the initial and the subsequent electric spacecraft flights. Sizes and weights appear low enough to constitiute no serious problem, especially for future high-specific-impulse spacecraft.