Electric Thruster Performance for Orbit Raising and Maneuvering

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Several electric thrusters are compared to chemical rockets for orbit raising and in-orbit maneuvering applications. For power-intensive payloads and satellites, electric propulsion can show substantial performance advantages over chemical propulsion. The arcjet is promising for near-future applications because of its low power requirements. The electrostatic thruster offers additional performance advantages over the arcjet, but with added power requirements. The performance of the electrostatic thruster can be enhanced considerably if the power processsing requirements can be reached. A performance gap exists between the specific impulse of an arcjet (~1000 s) and the minimum practical specific impulse of an electrostatic thruster (~2000 s). A high-performance thruster (thruster efficiency ≥ 0.5) in this specific impulse range, with minimal power processing requirements, offers time-payload compromises intermediate of those possible with either the arcjet or the electrostatic thruster.

Introduction

THE objective of this paper is to assess the degree to which present electric thruster technology satisfies orbit raising and maneuvering needs. In those areas where needs are not being met, the further objective is to indicate the research directions that appear most promising for satisfying those needs. While economic cost is certainly an important factor in mission decisions, it is beyond the scope of this paper to estimate the dollar cost of the systems postulated for comparison.

Electric propulsion is, at present, not being used for the primary propulsion applications of orbit raising and large-scale maneuvering in the near-Earth environment. To replace chemical propulsion in these applications with a new technology will require the demonstration of major advantages because it is difficult to replace a widely used existing technology with a new technology that only offers marginal improvements. If studies of feasibility show substantial improvement (e.g., 50-100% greater thruster efficiency), then the probability of actual application of the new technology is considerably enhanced.

The mission and system analysis presented herein is simple when compared to the sophisticated studies often presented in the aerospace sciences. The reasons for this simplicity are twofold. First, a sophisticated study is not required to show a major difference if it exists. Second, the very complexity of a sophisticated study often serves to obscure some results when substantially different approaches with different levels of development are compared. More detailed treatments dealing with specific missions and the associated thruster system parameters are available in the literature.[1-3]

Propulsion System Performance

For chemical propulsion, a specific impulse of 500 s is assumed in orbit raising applications. (The jet velocity in m/s can, of course, be obtained from the specific impulse by multiplying the latter by the acceleration of gravity 9.80665 m/s^2.) Such a specific impulse requires cryogenic storage of propellant during the launch up to the low orbit start of an orbit raising mission, a technology that is well within the present state-of-the-art. For maneuvering needs, a reduced specific impulse of 425 s was assumed to be more consistent with long-term storage requirements. A specific impulse of 500 s is also included for maneuvering with mission durations consistent with cryogenic storage.

For electric propulsion, both electrostatic (electron bombardment) and magnetoplasmadynamic (MPD) thrusters are considered. These are the two main types of ion and plasma thrusters being studied in the United States. Both of these are evaluated over a range of specific impulse. An H_2 arcjet is also included for a specific impulse of 1000 s.

Short descriptions of these electric thrusters are included here for those unfamiliar with electric propulsion. Ions are generated in the electrostatic thruster by a low-pressure discharge. The propellant pressure is sufficiently low in this discharge that a magnetic field is required to contain and efficiently utilize the energetic electrons. The ions are electrostatically extracted at one end of the discharge chamber by multiaperture electrodes or ion optics. To provide both space-charge and current neutralization of the overall ion beam as it leaves the thruster, electrons are added from a neutralizer located downstream of the ion optics. Power processing is normally required for the voltages and currents for the various electrostatic thruster functions. This power processing requirement entails both losses and the heat rejection capability for these losses. Despite the apparent complexity of the electrostatic propulsion system, tests of thousands of hours duration have been successfully conducted.

MPD thruster operation utilizes a single discharge between concentric electrodes, with the interaction of the discharge current and a magnetic field providing the thrust. More efficient operation is obtained at high power and thrust densities so that, for most applications, pulsed operation is required to reduce the average power to an achievable value. At high power, the magnetic field involved in thrust generation is usually provided by the discharge current, rather than a separate field coil or permanent magnets. Pulsed operation requires energy storage betweeen the power source and the thruster, as well as some mechanism for generating and controlling pulses of propellant flow. The MPD thruster is at a much lower level of development than the electrostatic thruster. Fewer duration tests have been conducted and no power processing and propellant flow controls exist of appropriate mass for flight.

The arcjet uses electrical energy to heat a propellant, that expands through a nozzle to generate thrust, similar to the

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expansion process in a chemical rocket. The propellant temperature is limited by the temperature limits of the materials involved and the need to limit dissociation and ionization frozen flow losses in the propellant. With consideration of the temperature limits imposed by these considerations, the highest exhaust velocity is obtained with the propellant having the lowest molecular weight. Early studies with hydrogen propellant 10-20 yr ago showed no particular thruster problems, but the cryogenic storage of hydrogen at that time was difficult. As a result, interest in the arcjet thruster waned after the initial studies. Recent advances in cryogenic storage have been the cause of much of the renewed interest in the arcjet thruster.

The electrostatic, MPD, and arcjet thrusters do not by any means complete the list of electric thrusters available. These three, however, provide a performance matrix for discussing most of the advantages and disadvantages of electric propulsion. Other types of electric thrusters are included in the subsequent discussions.

The performance of an electric thruster is measured by its mass, specific impulse, and efficiency for converting electric power to thrust. The electrostatic thruster performance used herein was obtained from a parametric study of performance for both argon (Ar) and xenon (Xe) propellants,[4,5] and is presented in summary form in Table 1. In this table, I_{sp} is the specific impulse in seconds and η_T is the total thruster efficiency. The only modifications made to the results of this study are a reduction of the specific impulse by a factor of 0.975 and a reduction of efficiency by a factor of 0.95, both to account for the "cosine loss" of off-axis thrust components that were not included in the original study. The optimum thruster power is used for the data of Table 1, this power ranged from several kilowatts at low specific impulses to several megawatts at high specific impulses. Larger than optimum thruster power can be accommodated using an array of thrusters, each of optimum power. Smaller than optimum power can be accommodated by a thruster size reduction with only small performance losses as long as the size reduction is moderate. In practice, at least two electrostatic thrusters are normally required to provide full three-axis attitude control during thrusting.

The efficiency of an electrostatic thruster increases with increasing atomic mass of the propellant. This effect accounts for the difference shown for Ar (40 amu) and Xe (131 amu) in Table 1. The electrostatic thruster in the highest state of development[6] uses mercury (Hg) (201 amu) as the propellant. From a propulsion viewpoint, Hg is the preferred propellant because it is easily stored with a very low tankage fraction and yields high thruster efficiency. Environmental considerations, however, will probably prevent its use in large quantities in the near-Earth environment. Xe is also an excellent electrostatic thruster propellant. It has a high enough atomic mass to give high efficiency, and its moderate critical temperature permits noncryogenic storage with tankage fractions‡ of 0.05-0.10. It has a shortcoming of being costly and somewhat limited in supply. For mass requirements of thousands of kilograms per year, though, the cost and limited supply should not be serious limitations. For larger quantities, Ar offers much lower cost with the shortcomings of reduced thruster performance and a need for cryogenic storage to reduce tankage fractions to a reasonable level. Both Xe and Ar are environmentally "clean" in the sense that they are inert and are obtained from the atmosphere, hence are partially returned to their source if used in the near-Earth environment. If used in sufficiently large quantities, however, ionized gases could still have an adverse impact on the upper atmosphere.

Table 1 Electrostatic thruster performance[a]

I_{sp}, s	Ar propellant η_T	Ar propellant kg/kW	Xe propellant η_T	Xe propellant kg/kW
975	—	—	0.324	12.49
1462	0.251	4.34	0.469	7.12
1950	0.328	3.32	0.589	3.36
2438	0.400	2.47	0.672	2.68
2925	0.466	1.88	0.731	2.16
3900	0.570	1.57	0.806	1.50
4875	0.647	1.30	0.848	1.11
5850	0.704	1.11	0.874	0.90
6825	0.746	0.96	0.891	0.73
7800	0.779	0.84	0.902	0.61
9750	0.824	0.67	0.918	0.47

[a] Performance values from Ref. 5.

As indicated, the electrostatic thruster data of Table 1 are optimized performance data. For example, the efficiency of an electrostatic thruster can be increased by increasing the fraction of propellant that is ionized. This increase can be accomplished by using a deeper discharge chamber, resulting in a greater neutral residence time, hence a greater probability of being ionized. Such a chamber shape, however, also results in a smaller fraction of the ions being extracted into the beam. This last effect results in increased discharge energy required per beam ion. For the data in Table 1, the chamber depth is selected to maximize overall thruster efficiency.

The data of Table 1 thus represent a reasonable upper limit for what may be expected from an electrostatic thruster. If significantly better performance is assumed, it should be justified on the basis of completely new design and/or theory.

As mentioned above, the Hg electrostatic thruster has been the subject of considerable development. As a check on the calculation procedures used for the data of Table 1, the performance of the existing 30-cm thruster with Hg at 3000 s closely approximates the calculated performance of the same size of thruster at the same specific impulse with Xe as the propellant. If a Hg thruster were optimized in the manner described for Ar and Xe thrusters, it is expected that the Hg thruster would show performance superior to either Ar or Xe.

The electrostatic thruster requires electric power at a variety of voltages and currents. When electric power is supplied at a somewhat arbitrary bus voltage, there is, as discussed previously, a need for power processing between the electric power source and the electric thruster. Fully developed processing is available for the 30-cm Hg thruster mentioned previously.[6] Using this as a guide, an efficiency of 0.9 and a specific mass of 10 kg/kW are assumed herein. These values are close to the actual values obtainable for the 30-cm Hg power processor.[6] As is shown, the power processing requirement is a critical factor in the evaluation of electrostatic thrusters.

The MPD thruster is not evaluated in the same parametric manner as the electrostatic thruster. The best performance estimates to date appear to be from experimental evaluations.[7] Tabulated efficiencies for two high-performance MPD thrusters are presented in Table 2. The propellant for both of these thrusters is Ar. Although the effect of propellant atomic mass is not as straightforward as for an electrostatic thruster, there appears to be a general shift to higher efficiency with higher atomic mass in an MPD thruster. The efficiencies shown in Table 2 may, therefore, be somewhat limited by the choice of propellant. The terms "benchmark" and "improved" are used by the authors of Ref. 7 to designate specific operating configurations.

As discussed previously, most applications of the MPD thruster asume pulsed operation. Neither the MPD thrusters nor the required power processing for pulsed operation have been studied sufficiently to provide estimates of flight hardware masses. With a view toward selecting optimistic numbers to avoid penalizing the MPD thruster, a power processing

‡Ratio of empty tank mass to full tank mass.

Table 2 MPD total thruster efficiency, η_T[a]

I_{sp}, s	Benchmark thruster	Improved thruster
1000	0.18	—
1200	0.20	—
1500	0.22	0.20
2000	0.25	0.26
2500	—	0.29
3000	—	0.31
3500	—	0.32

[a] Efficiencies from Ref. 7.

Table 3 Mission analysis assumptions

	Vehicle	
Orbit raising	Fixed mass	$0.10m_0$
Maneuvering	Fixed mass	$0.05m_0$
	Propulsion system	
Chemical		None
Electrostatic	Thruster	m and η, Table 1
	Power processor	10 kg/kW, $\eta = 0.9$[a]
	Power source	15 kg/kW[b]
	Other	5 kg/kW
MPD	Thruster	2.25 kg/kW;η, Table 2
	Power processor	2.5 kg/kW,$\eta = 0.9$
	Power source	15 kg/kW[b]
	Other	2.5 kg/kW
Arcjet	Thruster	2.5 kg/kW,$\eta = 0.5$
	Power source	15 kg/kW[b]
	Other	2.5 kg/kW

[a] Power processor mass and losses omitted from some mission calculations for electrostatic thrusters.
[b] A specific mass of 15 kg/kW was used for the power source when it was part of the propellant system. As part of the payload 15 kg/kW was the lower limit.

efficiency of 0.9 is assumed, together with a power processing specific mass of 2.5 kg/kW, based on input electric power from the power source. The specific mass for the MPD thruster is assumed to be 2.5 kg/kW based on the power from the power source (2.25 kg/kW based on power from the power processor).

The arcjet thruster is also included in this electric thruster performance comparison. If H_2 propellant is assumed, an efficiency of about 0.5 is possible at 1000 s.[8] The use of H_2 implies, of course, efficient cryogenic propellant storage. Other propellants result in sharply efficiency or specific impulse.

The arcjet is the subject of considerable previous work but receives no mention in recent surveys.[9,10] To obtain reasonable efficiencies it is necessary to avoid significant dissociation and ionization. For H_2 propellant, this means limiting specific impulse to a maximum of about 1000 s. Because of the higher atomic weights of other propellants their maximum specific impulses are considerably lower. The specific mass of the arcjet is assumed to be 2.5 kg/kW.

Mission Analysis Assumptions

Performance is evaluated in terms of payload and mission time for two missions. The first is an orbit raising mission from a geocentric radius of 7.25×10^6 m (low-Earth orbit) to a radius of 4.23×10^7 m (geosynchronous Earth orbit), with a plane change of 28.5 deg. For high-thrust chemical propulsion, the Δv*for orbit raising is about 5.2 km/s where Δv is the sum of the velocity increments required for a specificed maneuver. For low-thrust electric propulsion, the Δv is about 5.9 km/s. The required Δv for a given electric propulsion mission is, in general, dependent on specific mission parameters. For the range of parameters investigated herein, the value of 5.9 km/s is chosen as a reasonable and representative value. The second mission is in-orbit maneuvering, with the total Δv of the maneuvers equal to geosynchronous orbital velocity, 3.07 km/s.

The calculation procedure can be summarized as using the above Δv in the rocket equation together with the mass-ratio assumptions of Table 3.

A structural/tankage/guidance/reserve propellant mass of $0.10m_0$ is assumed for all orbit raising, with m_0 the initial low-orbit mass. A similar fixed mass of $0.05m_0$ was assumed for in-orbit maneuvering, with m_0 the vehicle mass before maneuvering. These fixed masses are indicated as vehicle assumptions in Table 3.

For the chemical propulsion system, the mass of the rocket engine is assumed to be negligible, hence is included within the fixed mass of either $0.10m_0$ or $0.05m_0$.

All electric propulsion systems require an electric power source. When that source is considered to be part of the propulsion system mass, it has an assumed specific mass of 15 kg/kW exclusive of power processing. When included in the payload, the power source specific mass ranges upward from 15 kg/kW. The value of 15 kg/kW is typical of either solar-cell arrays or nuclear-electric power sources,[11,12] although the solar-cell array value is also supported by hardware tests. For orbit raising, starting from the assumed low orbit, the solar-cell is probably excluded because of Earth shadowing. For higher level orbit raising or in-orbit maneuvering, both power sources are possible alternatives. Within the Van Allen belts, the effects of radiation from those belts on solar-cell efficiency must be considered.

For the electrostatic propulsion system, the thruster mass and efficiency are obtained from Table 1, with the specific mass based on thruster input power (after power processing losses). The power processor specific mass is 10 kg/kW and the efficiency is 0.9, as discussed earlier. An additional mass of 5 kg/kW (based on power source output) is assumed for all cables, support structure, etc., that are peculiar to an electrostatic propulsion system. These assumptions are indicated in Table 3. Although these assumptions are simple compared to more detailed mass studies made previously,[13,14] the predicted total propulsion system mass is in reasonable agreement with the more detailed studies.

The assumptions for the MPD and arcjet propulsion systems are also indicated in Table 3. The "other" mass is reduced by half for these propulsion systems because they are more compact than the electrostatic system. The thruster and power processor masses for the MPD system are both probably optimistically light considering the unresolved problems of pulsed propellant and power flow.

For the arcjet propulsion system, the masses are also uncertain. But the uncertainty is small compared to power-source mass, so that the impact on arcjet mission performance of any errors is also small.

The mission analysis approach used herein is simplified. The intent is not to calculate refined mission performance, but to compare broad characteristics of different propulsion systems. Further, for the comparison to indicate that electric propulsion should supplant chemical propulsion for any of the missions under consideration, large and clearcut performance advantages in terms of mission cost should be demonstrable for electric propulsion.

Propulsion System Thrust-to-Mass Ratio

The relative evaluation of different propulsion systems must rest ultimately on thorough mission analyses. For various electric propulsion systems, however, it is possible to compare performance at the same specific impulse by comparing the thrust-to-mass ratios for the different systems. Using the mass and efficiency assumptions outlined in the previous section, with the power source assumed to be part of the propulsion system, the thrust-to-mass ratios shown in Fig. 1 are obtained. Thrusting time is usually a parameter of interest for electric propulsion, so any comparison should be made for the same thrusting time, hence the same vehicle

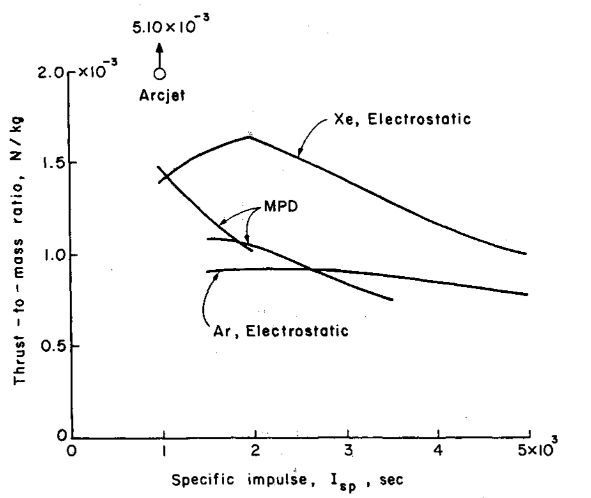

Fig. 1 Comparison of thrust-to-mass ratios for different electric propulsion systems.

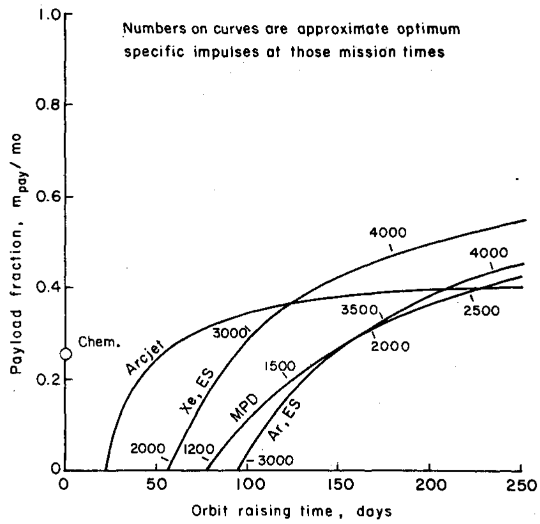

Fig. 2 One-way orbit raising mission with the 15-kg/kW power source included in the propulsion system mass.

acceleration for a given mission. For a given acceleration (F/m_0) the system with the highest thrust-to-mass ratio will have the minimum mass for the propulsion system, resulting in maximum payload mass.

$$m_{\text{prop}}/m_0 = (F/m_0/(F/m_{\text{prop}}) \tag{1}$$

The system mass as shown in Fig. 1 includes a 15-kg/kW power source, but does not include propellant. It should be evident that this mass should be the same for two vehicles with the same initial mass, thrust, acceleration, and thrusting time.

The foregoing applies strictly to a comparison at the same specific impulse. It is possible, though, to make limited comparisons at different specific impulses. Operation of a thruster at a specific impulse below the maximum thrust-to-mass ratio should not be of interest. This is because such operation would result in both increased propellant mass for a given mission and increased propulsion system mass exclusive of propellant.

Returning to Fig. 1, it should be apparent that operation of an electrostatic thruster with Xe is preferred over Ar from a performance viewpoint. Further, operation with Xe below about 2000 s is undesirable because the thrust-to-mass ratio decreases in that range. Next, the performance of the MPD thruster is inferior to that of the Xe electrostatic thruster, except near 1000 s where the arcjet is superior to both. Even at 1000 s, the MPD performance is not equal to Xe electrostatic performance at 2000 s and, because of increased propellant consumption, would be considered definitely inferior. This result is obtained despite some very favorable assumptions for the MPD thruster.

The comparison between the Xe electrostatic and MPD thrusters can be made clearer with numerical values. At a specific impulse of 1950 s (the maximum for Xe in Fig. 1) the specific mass of the Xe electrostatic thruster is 33.7 kg/kW. For the MPD thruster the specific mass is only 22.5 kg/kW. The Xe electrostatic system is clearly much more massive than the MPD system with less than half of its weight consisting of power source.

When the specific masses of the two systems are divided by the efficiency of converting electric power to thrust the relative values are reversed. For the electrostatic thruster, the efficiency product (power conditioning efficiency times thruster efficiency) is 0.530, giving a mass to useful beam kinetic energy ratio of 63.6 kg/kW. For the MPD thruster, the efficiency product is 0.225 (at 2000 s specific impulse), yielding 100.0 kg/kW.

There are, of course, a number of caveats in interpreting Fig. 1. H_2 is required for the arcjet performance shown. If

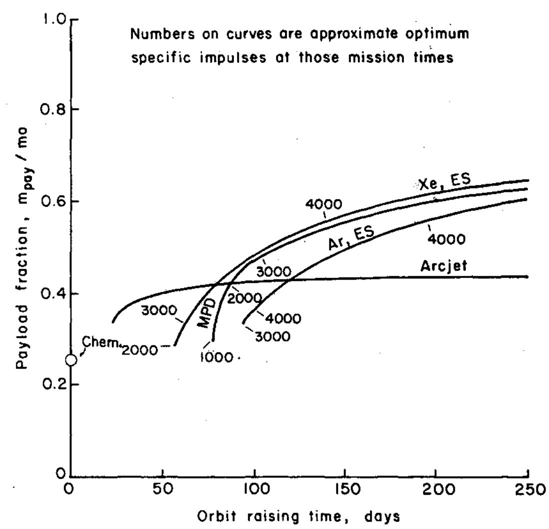

Fig. 3 One-way orbit raising mission with the 15-kg/kW power source included in the payload.

cryogenic storage of H_2 is not possible for the mission lifetime under consideration, the arcjet would have to operate at either a much lower efficiency or a much lower specific impulse. Further, the MPD performance might be much improved by using other propellants. Finally, the electrostatic thruster is more fully developed than most ion and plasma thrusters, and may therefore be closer to its ultimate capability.

Orbit Raising

Three basic types of orbit raising or maneuvering missions are considered. One-way missions with the power source included in the propulsion system constituted one basic type (Fig. 2). One-way missions in which the power source is considered to be part of the payload are shown in Figs. 3 and 4. Finally, 10-trip missions with a reusable tug are shown in Fig. 5.

The one-way missions with the power source included in the propulsion system mass, Fig. 2, reflect strongly the thrust-to-mass ratios of Fig. 1. The minimum mission time is roughly inverse to maximum thrust-to-mass ratio. At longer mission times, the thrust-to-mass ratios at higher specific impulses determine the relative order. A 20% payload advantage over chemical propulsion (at 0.247 payload fraction§) is possible

§Ratio of payload mass to initial total mass.

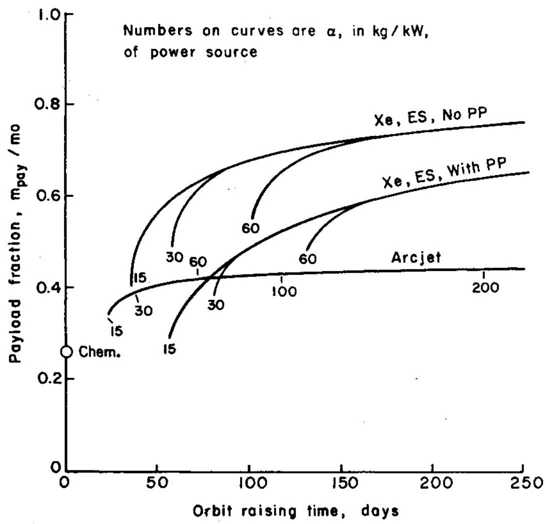

Fig. 4 One-way orbit raising mission with the power source included in the payload.

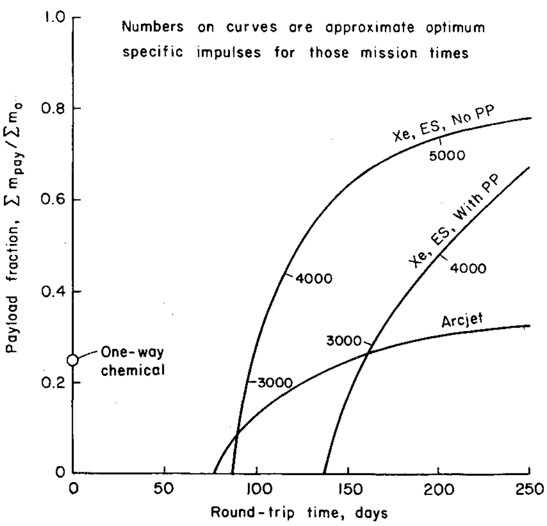

Fig. 5 Ten-trip orbit raising mission with the 15-kg/kW power source included in the propulsion system mass.

with a 65 day mission time for an arcjet, while a 50% payload advantage is possible at a mission time of 126 days. The Xe electrostatic thruster shows a 50% advantage over chemical propulsion at the same 126 day mission time, but cannot match the arcjet at shorter mission times. For a 50% payload advantage over chemical, the Ar electrostatic thruster requires 192 days and the MPD thruster requires 202 days.

The results shown in Fig. 2 indicate that electric propulsion requires mission times on the order of 100-200 days for significant (in this case 50%) payload advantages over chemical propulsion.

Similar calculations with the power source included in the payload might be considered a fortuitous coincidence. However, many activities in space will be power-intensive, resulting in the frequent need to transport power sources. For the 15-kg/kW specific mass assumed for the power source, sharp increases in payload ratios occur over those calculated when the power source is part of the propulsion system mass. These increases result, of course, from the shifting of the power source mass to the payload category. Note that the MPD thruster approaches Xe thruster performance for mission times longer than about 90 days. This is because so much power is assumed to be available from the payload that the low efficiency of the MPD thruster is not much of a penalty. With more limited power availability the MPD thruster would not fare as well.

A significant result from Fig. 3, compared to Fig. 2, is the greatly reduced mission times possible for the same payload advantage over chemical propulsion. For the arcjet thruster a 50% advantage is possible in 30 days, while for the Xe electrostatic thruster a 50% advantage is possible in 68 days.

A payload that generates power with a specific mass of 15 kg/kW is obviously very power-intensive, and probably almost all power source. The more frequent expectation would be that only a fraction of the payload would be power source and the specific mass of the overall payload would be much larger. This situation is indicated in Fig. 4. Because of the added variables included, only curves for the arjet and Xe electrostatic thruster are shown.

Considering first the Xe electrostatic thruster with power processing, the 15-kg/kW curve for this thruster and the arcjet curve are the same as shown in Fig. 3. The arcjet curve is notable in that increasing the specific mass of the payload has no effect except to limit the minimum mission time. This is because the mission time determines the thruster power level, and with specific impulse limited to 1000 s additional power cannot be utilized. For the 50% payload advantage over chemical propulsion cited in connection with Fig. 3, the requirement for the payload is actually about 23 kg/kW or less. For payloads with higher specific masses, the arcjet mission time would simply stretch out beyond 30 days. To a close approximation, the arcjet mission time is proportional to the specific mass for the payload at these higher values.

For the Xe electrostatic thruster there are different payload fraction curves for different payload specific masses. At long mission times, though, the payload fraction curves tend to coalesce. Comparison of the 30-kg/kW Xe curve in Fig. 4 (with power processing) with the 15-kg/kW MPD curve of Fig. 3 shows that they are almost identical. This means that, for minimum mission times, the MPD thruster requires a more power-intensive payload (by about a factor of 2) for the same performance. For longer mission times, the specific mass of the payload is not as important, due to possible tradeoffs between specific impulse and efficiency.

Another Xe electrostatic curve is also shown in Fig. 4 for no power processing. The 10-kg/kW power processing mass, together with the efficiency of 0.9, serves to reduce payload capability substantially from that which would otherwise be possible. Reduced power processing requirements have already been investigated,[15] so that the improvement shown in Fig. 4 is based on a present trend. Furthering this present trend would clearly be advantageous in terms of mission capability.

Using a 60-kg/kW specific mass for the payload as a basis of comparison, the arcjet thruster shows a 68% payload advantage over a chemical propulsion, with a mission time for the arcjet of about 73 days. For the Xe electrostatic thruster with power processing, the payload advantage is about 96% with a 131 day mission time. Without power processing the payload advantage is increased to 150%, while the mission time is reduced to 103 days. The Xe electrostatic thruster, with or without power processing, shows substantial additional payload advantages for longer mission times.

Performance for the 10-trip tug is indicated in Fig. 5. Here, the power source must be a part of the propulsion system in order for the tug to make the return trip to low orbit. A greater number of round trips per vehicle would, of course, show a greater advantage for this variation. However, if a significant advantage exists it should be evident for a 10-trip lifetime.

The chemical rocket is really not practical for a 10-trip tug under the assumptions herein. (Only about 2% of the low-orbit mass can be delivered to a geosynchronous orbit if propellant for the return trip must also be carried along.) The one-way payload capability for a 500 s chemical rocket is still included, however, as a basis of comparison.

Although readily capable of appreciable payload as a 10-trip tug, the arcjet shows only marginal advantages over one-way chemical propulsion. For a 20% advantage a round-trip time of about 170 days is required, while for a 50% advantage about 600 days is required. The 1000 s specific impulse of the arcjet is clearly a limitation for the 10-trip tug mission.

For the Xe electrostatic 10-trip tug with power processing a 20% advantage is obtained with a round-trip time of about 164 days, while a 50% advantage requires about 176 days. In 250 days, the advantage can be about 163%.

Without a power processor, the Xe electrostatic thruster shows a 50% advantage in under 110 days. For a 250-day round-trip time, the advantage would be about 216%.

In the interest of simplicity, considerations such as thruster refurbishing have been ignored. It should be clear, though, that significant payload advantages should still be possible using a multiple-trip tug with a Xe electrostatic thruster. These advantages can be enhanced if power processing requirements can be reduced.

In-Orbit Maneuvering

The propulsion system mass fraction required for maneuvering was calculated for a Δv of 3.07 km/s. If a single large maneuver was required, the Δv requirement for a chemical rocket could be smaller by a few percent. Both storable (425 s specific impulse) and cryogenic (500 s specific impulse) propellants are considered for chemical propulsion.

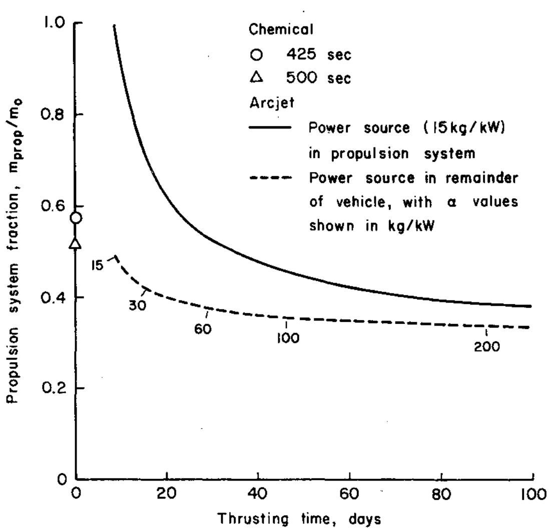

Fig. 6 In-orbit maneuvering capability of arcjet system.

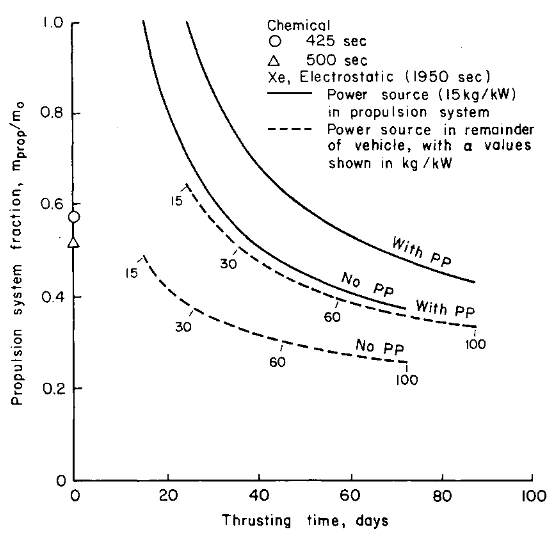

Fig. 7 In-orbit maneuvering capability of Xe electrostatic system.

The chemical propulsion system is compared to arcjet propulsion in Fig. 6. If an electric power source must be carried along for the exclusive use of the arcjet, equal propulsion system mass requires a cumulative thrusting time of about 24 days (at 425 s specific impulse) and 32 days (at 500 s specific impulse). For a 20% advantage, these times stretch out to about 46 and 64 days, respectively.

If power is available from an onboard power source, then the power source does not need to be included in the propulsion system mass. The arcjet performance then becomes dependent on the amount of power available. If the remainder of the satellite, other than the maneuvering propulsion system, has a specific mass of 60 kg/kW, the mass advantage is about 35% at 425 s specific impulse and 28% at 500 s specific impulse for a thrusting time of about 29 days. Longer thrusting times result in only small decreases in arcjet propulsion mass, but do permit less power-intensive satellites to be used effectively. As with orbit raising, the thrusting time varies roughly proportionally with specific mass of the remainder of the satellite.

The in-orbit maneuvering capability of the Xe electrostatic thruster is shown in Fig. 7. Even with the power processing mass and losses omitted, the Xe thruster requires thrusting times in excess of 50 days to match the mass of the arcjet when both systems include the power source mass.

If the power source is assumed to be part of the remainder of the satellite, propulsion system advantages can be shown relative to the arcjet, but only for relatively more power-intensive satellites.

A point should be made concerning specific impulse of the Xe electrostatic thruster. With thrusting time, propulsion system mass, and the remaining satellite mass all variable, a simple optimum in performance is not possible in all cases without additional restrictions. For the lower values of specific mass shown in Fig. 7, a specific impulse of 1950 s actually gives optimum performance in all respects. For the higher specific masses, the shortest thrusting times are obtained with 1950 s, with only small mass penalties compared to higher specific impulses.

If higher thrust-to-mass ratios were possible for the Xe electrostatic thruster below 1950 s, Fig. 1, then clearly that performance is advantageous for some in-orbit maneuvering applications. With the moderate maneuvering Δv assumed herein, the need appears to be for a thruster that can operate efficiently at a specific impulse between the 1000 s of the arcjet and the minimum practical value of 1950 s for the Xe electrostatic thruster.

Conclusions

A number of significant conclusions can be drawn from the study presented herein. For orbit raising missions with power sources included in the payloads, the arcjet could show significant 60-70% payload advantages over chemical propulsion in the next few years. The mission time with an arcjet is primarily dependent on the power available from the payload, but can be well under 100 days for payload specific masses of 60 kg/kW or less.

The Xe electrostatic thruster can show further payload gains over the arcjet, presumably in the more distant future. The degree of advantage over the arcjet depends in a large measure on the power processing required.

Although not calculated in detail, the present 30-cm Hg electrostatic thruster is expected to approximate the optimized Xe electrostatic thruster in mission capability.

The MPD thruster is penalized by having very low efficiency. This problem appears to be more serious than the absence of well-developed propellant controls and power processing for pulsed operation.

Multiple-trip orbit raising tugs appear to be practical only for high-efficiency electrostatic thrusters (Xe and Hg).

Reduction of power processing requirements will greatly increase the practicality of multiple-trip tugs.

There is no clearcut advantage for either the arcjet or the Xe electrostatic thruster in the in-orbit maneuvering application, but the arcjet is generally more promising where short thrusting times are desired and the onboard electric power is limited. It does not appear promising to use either electric thruster type if a power supply must be dedicated solely to maneuvering propulsion.

For future research, it is clear that the arcjet should have a high priority. Actually, this type of thruster should be considered a class rather than a single type. The tradeoff between additional losses and higher specific impulses should be explored above 1000 s. The resistojet is also a promising alternative for many arcjet applications. Other alternatives may also exist.

Despite its well-developed state, the electrostatic thruster still shows promise of further efficiency improvements. Improved discharge efficiency, in particular, offers one possibility.[16] The advantage of higher efficiency is twofold. It permits the substitution of a lighter atomic weight propellant with smaller performance penalties. It also further reduces the power processing needs, in that a larger fraction of the total power would go into the beam, with the possibility of supplying beam energy by direct coupling to the power source. Note that for the Xe electrostatic thruster, the efficiency is not, in itself, a problem. That is, in the usable range above 1950 s, the efficiency exceeds 67%.

Although efficient, the Xe electrostatic thruster is not practical for applications below about 2000 s. The limitation in thrust-to-mass ratio below this specific impulse results primarily from limitations in the electrostatic acceleration process. Some other acceleration process, such as Hall-current acceleration, may permit efficient ion thrusters in the 1500-to 2000-s range. Such alternative electric thrusters appear to be desirable research objectives, to permit time-payload compromises intermediate of those possible with either the arcjet or the Xe electrostatic thruster, or their closely related types.

References

[1]Zafran, S., Gran, M.H., Fosnight, V.V., and Callens, R.A., "Ion Propulsion for Shared Shuttle, Sun-Synchronous Missions," AIAA Paper 81-099, April 1981.

[2]Nagorski, R.P., and Boain, R.J., "An Evaluation of Nuclear Electric Propulsion for Planetary Exploration Missions," AIAA Paper 81-0705, April 1981.

[3]Fearn, D.G., "The Application of Ion Propulsion to the Transportation and Control of Solar Power Satellites," AIAA Paper 81-0760, April 1981.

[4]Kaufman, H.R., "Performance of Large Inert-Gas Thrusters," AIAA Paper 81-0720, April 1981.

[5]Kaufman, H.R. and Robinson, R.S., "Large Inert-Gas Thrusters," AIAA Paper 81-1540, July 1981.

[6]"30-Centimeter Ion Thrust Subsystem Design Manual," NASA TM-79191, June 1979.

[7]Burton, R.L., Clark, K.E., and Jahn, R.G., "Measured Performance of a Multimegawatt MPD Thruster," *Journal of Spacecraft and Rockets*, Vol. 20, May to June 1983, pp. 299-304.

[8]Childs, J.H. and Cybulski, R.J., "SERT and Early Electric Propulsion Systems," *Astronautical Aerospace Engineering,* May 1963, pp. 112-117.

[9]Hudson, W.R., "NASA Electric Propulsion Technology Program," AIAA Paper 79-2118, Oct.-Nov. 1979.

[10]Loeb, H.W. et al., "European Electric Propulsion Activities," AIAA Paper 79-2120, Oct.-Nov. 1979.

[11]Young, L.E., "Solar Array Technology for Solar Electric Propulsion Missions," AIAA Paper 79-2086, Oct.-Nov. 1979.

[12]Buden, D., "100-kW$_e$ Nuclear Space Electric Power Source," AIAA Paper 79-2089, Oct.-Nov. 1979.

[13]Byers, D.C., Terdan, F.F., and Myers, I.T., "Primary Electric Propulsion for Future Space Missions," AIAA Paper 79-0881, May 1979.

[14]Byers, D.C., "Characteristics of Primary Electric Propulsion Systems, AIAA Paper 79-2041, Oct.-Nov. 1979.

[15]Rawlin, V.K., "Reduced Power Processor Requirements for the 30-cm Diameter Hg Ion Thruster, AIAA Paper 79-2081, Oct.-Nov. 1979.

[16]Kaufman, H.R. and Robinson, R.S., "Electric Thruster Research," NASA CR-165603, Dec. 1981.