Performance and Lifetime Assessment of Magnetoplasmadynamic Arc Thruster Technology

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A summary of performance and lifetime characteristics of pulsed and steady-state magnetoplasmadynamic (MPD) thrusters is presented. The technical focus is on cargo vehicle propulsion for exploration-class missions to the moon and Mars. Relatively high MPD thruster efficiencies of 0.43 and 0.69 have been reported at about 5000-s specific impulse using hydrogen and lithium, respectively. Efficiencies of 0.10 to 0.35 in the 1000- to 4500-s specific impulse range have been obtained with other propellants (e.g., Ar, NH_3, N_2). Electrode power losses of less than 20% at megawatt power levels using pulsed thrusters indicate the potential of high MPD thruster performance. Extended tests of pulsed and steady-state MPD thrusters yield total impulses at least two to three orders of magnitude below that necessary for cargo vehicle propulsion. Performance tests and diagnostics for life-limiting mechanisms of megawatt-class thrusters will require high-fidelity test stands, which handle in excess of 10 kA, and a vacuum facility whose operational pressure is less than 4×10^{-2} Pa.

Nomenclature

e	= electronic charge, 1.6×10^{-19} C
g	= gravity acceleration, 9.81 m/s^2
h_0	= unheated propellant enthalpy, J/kg
I_{sp}	= specific impulses, s
J	= arc current, A
$(J^2/\dot{m})_c$	= onset parameter, A^2-s/kg
k	= Boltzman's constant, 1.38×10^{-23} J/K
$\dot{m}$	= mass flow rate, kg/s
P	= input power, W
P_a	= power loss to anode, W
P_e	= input electric power, W
P_L	= power to water cooling system, W
T	= thrust, N
T_e	= electron temperature, K
V_a	= anode voltage drop, V
ΔV	= velocity increment required for a mission, m/s
v	= average exhaust velocity, m/s
η	= thrust efficiency
η_{th}	= $(P - P_L)/P$, thermal efficiency
ϕ	= work function of anode material, V

Introduction

THE magnetoplasmadynamic (MPD) thruster system is an attractive candidate for lunar and Mars cargo vehicle propulsion as well as orbit raising applications. The high specific impulse (1000–5000 s) provided by the MPD thruster minimizes propellant requirements and improves the mass transfer capability to low Earth orbit (LEO). For example, Mars cargo vehicles using high-performance chemical propulsion require propellant-to-total vehicle mass fractions of 0.7 to 0.8 as shown in Fig. 1.[1] Although aerobraking techniques might be employed at Mars and for Earth return to minimize total vehicle initial mass in LEO, the data of Fig. 1 indicate that the propellant is still approximately 70% of the initial mass.

In a different situation, shown in Fig. 2, a 180-metric-ton payload can be delivered to Mars using a 5000-s specific impulse, 4-MW cargo vehicle employing electric propulsion with the required mass in LEO reduced by a factor of 1.9 compared to cryogenic chemical propulsion.[2] In addition to the benefit of high specific impulse operation, the MPD thruster system may also yield lower total mission duration than encountered using chemical propulsion simply because of the time needed to accumulate additional propellant in LEO through multiple launches.[3]

For the lunar and Mars cargo vehicle applications, the MPD thruster technology goals will need to include demonstrations of 0.1- to 10-MW steady-state thrusters at thrust efficiencies in excess of 0.6 and lifetimes in the 1000- to 10,000-h range. Candidate propellants will probably include hydrogen as well as easily stored propellants such as ammonia, hydrazine, and argon. Scaling relations, performance limits, and lifetime projections of multimegawatt-class MPD thrusters will have to be developed using quasisteady-state devices. Given these preliminary technology goals, an assess-

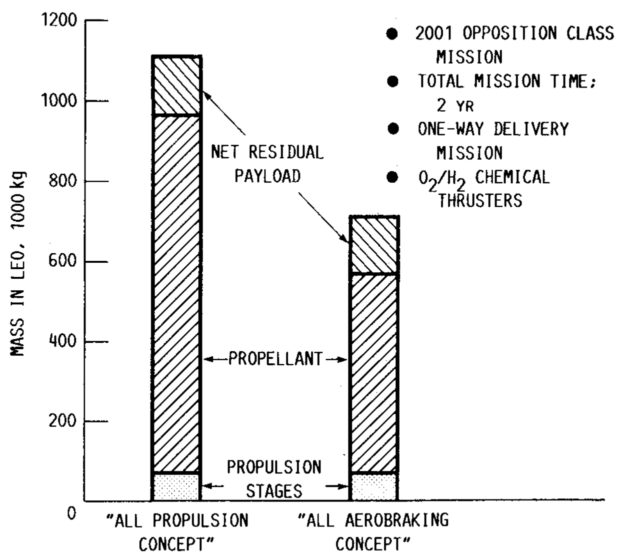

Fig. 1 Mars cargo vehicle mass in LEO using chemical propulsion.

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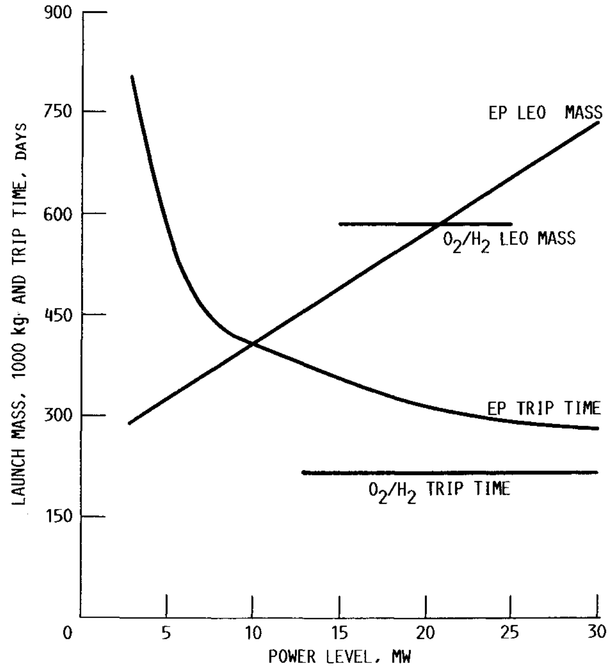

Fig. 2 Mars cargo vehicle trip time and launch mass; payload mass: 180,000 kg; electric propulsion specific mass: 10 kg/kW; specific impulse: 5000 s; O_2/H_2 specific impulse: 480 s.

ment of the current state of MPD thruster technology related to performance and lifetime was undertaken.

This paper presents a summary of performance and lifetime generated by present generation MPD thrusters. Technological activities over the last 25 years will be briefly synopsized. Some of the basic thruster configurations will be described, and the effects of vacuum facility pressure on performance measurements will be discussed. Summary graphics describing operating modes, specific impulse, efficiency, thruster loss mechanisms, and demonstrated life will be presented. Prospects for improvements in specific impulse, efficiency, and life will also be discussed.

Chronology of Magnetoplasmadynamic Thruster Technology

Research and technology efforts using self-field and applied-field MPD arc thrusters have been conducted over the past three decades. During the 1960s and early 1970s, many investigators from industry and government developed steady-state MPD thrusters in the 10- to 200-kW range. A comprehensive review of this work was reported by Nerheim and Kelly in 1967/1968.[4,5] Radiation-cooled thrusters were designed to operate up to 40 kW, and water-cooled devices had power capabilities in excess of 100 kW.[6–9] Applied-field thrusters employed electromagnets, permanent magnets, and superconducting magnets.[10] Self-field MPD arcjets were evaluated at megawatt power levels in the quasisteady-state mode.[11] In this pulsed mode, steady, high-power operation has typically been achieved within a period of a few hundred microseconds. Significant efforts were also undertaken to understand the effects of vacuum facility pressure on performance measurements.[12] The longest life demonstration was a 500-h test using a 30-kW ammonia device, which provided 9×10^5 Ns total impulse.[7] Higher total impulse capability, especially at higher power levels, has been limited by cathode erosion.[13] During the early 1970s, funding for high-power, steady-state MPD thrusters was dramatically reduced mainly because of projections of a limited space electric power capability in the 1970s and 1980s.

Research efforts continued in the United States, Europe, and Japan using primarily self-field, pulsed MPD thrusters, which operated in a single-shot, nonrepetitive mode.[14–16] Pulsed or quasisteady-state MPD thrusters have been operated at multimegawatt levels for purposes of understanding the characteristics of the plasma, erosion mechanisms, stability phenomena, thruster performance, scaling relations, and electromagnetic compatibility with other systems. Space plasma experiments using an MPD arcjet, pulsed at a power of about 2 MW, were conducted aboard Space Transportation System Shuttle Orbiter-9 in 1983.[17]

Much of the recent MPD thruster work in the United States has focused on quasisteady-state thruster modeling, scaling, erosion mechanisms, and lower-power steady-state thruster experiments.[18–20] Technology programs related to both pulsed and steady-state MPD thrusters are being carried out at a variety of institutions in Japan.[16,21,22] A self-field thruster has been cyclically life tested in Japan for one million pulses at a 1.2-MW instantaneous power level yielding a total impulse of about 2×10^4 Ns.[23] Life-limiting phenomena were observed to be electrode wear and localized insulator erosion.

Self-field, steady-state thrusters have been investigated by researchers in Germany.[24,25] Thrusters were operated with argon and nitrogen in the 100- to 300-kW range, producing specific impulse values up to 1200 s at thrust efficiencies between 15 and 20%. During the course of electrode erosion studies, a 200-kW thruster was tested for about 1 h, producing an estimated total impulse of 3×10^4 Ns.

There is a significant interest in intensifying the development of high-power MPD thruster technology because of the emergence of the Space Exploration Initiative, which will provide the critical technology for manned excursions to the moon and Mars.[26] NASA initiated technology programs to gain insight into the performance limits and lifetime of steady-state and MPD thrusters operating in the 0.1- to 1-MW range. These programs will ultimately have to develop high throughput vacuum facilities and performance diagnostics capable of accommodating megawatt-class thrusters so that meaningful performance and life data can be obtained for steady-state operations.

Estimates of Cargo Vehicle Propulsion Requirements

Multimegawatt cargo vehicle missions from LEO to Mars and LEO to the moon using low-thrust electric propulsion will require mission velocity increments in excess of 16 and 7 km/s, respectively.[27,28,30] Assuming a propulsion system of four MPD thrusters operating at a specific impulse of 5000 s, an LEO to Mars mission would require a propellant throughput of 10,000–30,000 kg and a total impulse requirement as high as 1×10^9 Ns for each thruster. The projected thruster lifetime requirements will be about 7000 h for a 10-MW system operating at an efficiency of 50%. A round-trip mission from LEO to the moon using the same propulsion system with a 4-MW power output will require a thruster mass throughput of approximately 7000 kg, a total impulse per thruster of 4×10^8 Ns, and a lifetime of about 4000 h.[2] Using separate thrusters for outboard and return legs would result in only 2000-h operation per thruster. Trades involving higher propulsion power and improved efficiency will result in lower required thruster lifetimes.

Continuing mission scenario development and cargo vehicle trade studies will clarify performance, lifetime, and power requirements for electric propulsion. Based on current information, it is expected that thruster power levels will be in the 0.5- to 3-MW range for systems powers of 2–10 MW. Total impulse per thruster will probably be about 1×10^8 to 1×10^9 Ns for specific impulse values of 2000–5000 s. Based on first-order calculations of lunar and Mars cargo vehicle propulsion requirements, the MPD thruster technology goal for life should be targeted in the 2000- to 8000-h range per thruster.

Thruster Configurations

Early MPD thrusters, such as those shown in Fig. 3, generally resembled thermal arc devices with the throat diameter made larger to permit operation in the 1- to 6-kPa range.[6,31] High-power thrusters operating in the 50- to 200-kW range incorporated water-cooled anode and cathode assemblies. Usually, thoriated tungsten cathodes were employed. Some lower-power thrusters, in the 10- to 40-kW range, were radiation cooled and used tungsten electrodes. Figure 4 shows a schematic of the electrodes of a 33-kW, radiation-cooled MPD thruster that was tested for 500 h.[7] Low-power thrusters usually used solenoid magnets to produce a "magnetic nozzle" or diverging field configuration, which generally enhanced performance since, without the applied fields, the self-fields generated were rather low.[5] Most devices incorporated low-area ratio nozzles to reduce viscous losses, though some early researchers used simple coaxial electrode configurations.[32] Most of the thruster characterization was done with hydrogen, ammonia, argon, and lithium.[4]

Recent high-power, steady-state MPD thruster technology work has been conducted using a self-field thruster with a low area ratio nozzle shown in Fig. 5.[25] The water-cooled cathode and anode assemblies are separated by electrically isolated, water-cooled segments. Thruster characteristics were evaluated at currents up to 4500 A and 270 kW using argon and nitrogen. Propellant flow rates were 0.2–1.5 g/s.

Over the last 20 years, MPD thruster operation at power levels in excess of 1 MW has been performed using capacitor banks pulsed for periods of about 1 ms.[1,14,15,33] These thrusters have been shown to reach quasisteady conditions after about 10–50 μs, after which the discharge characteristics are usually stable for the remainder of the 1-ms pulse.[18] Shown in Fig. 6 is a cutaway diagram of a flared anode thruster.[18,19] The thrusters have usually operated at peak power levels in excess of 2 MW in a single-shot, nonrepetitive mode. This configuration has cathode and anode diameters of 1 and 5 cm, respectively. Argon flow rates used were in the 1- to 3-g/s range at arc currents up to 22 kA and voltages in the 50- to 100-V range.

Figure 7 shows a quasisteady-state thruster with magnetic field coils, which are connected in series with the arc current.[16]

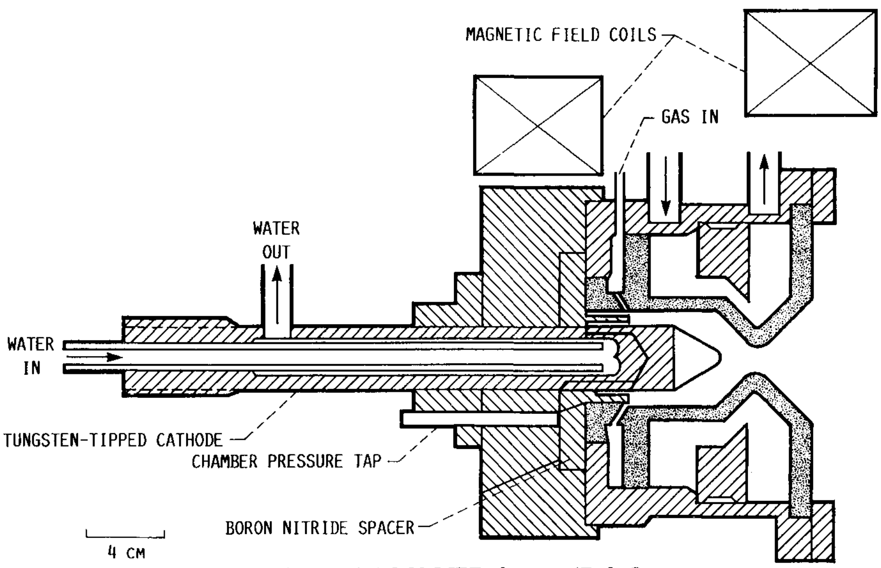

Fig. 3 Applied-field MPD thruster (Ref. 6).

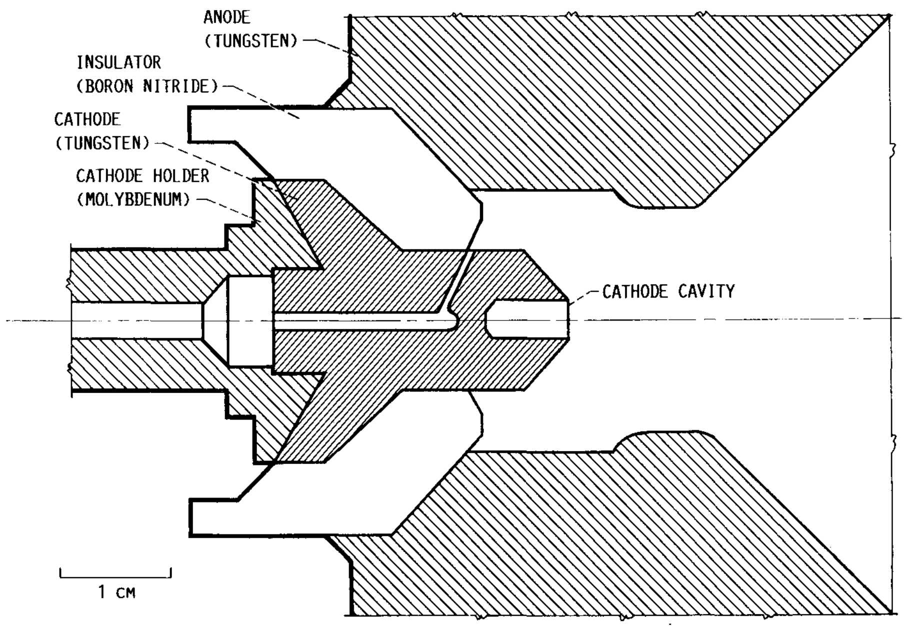

Fig. 4 Radiation-cooled thruster electrode assembly (Ref. 7).

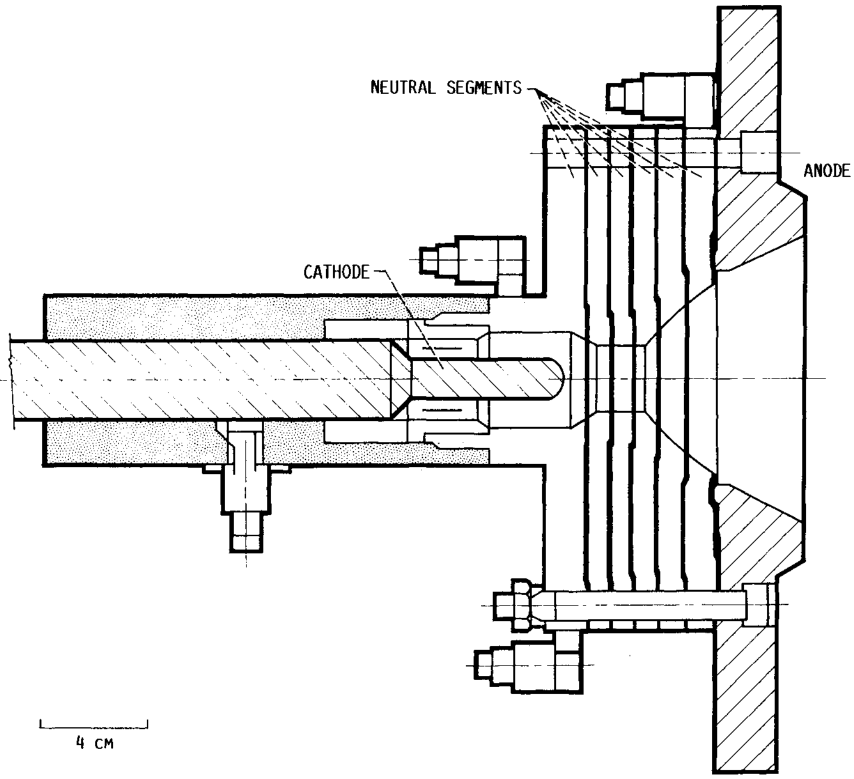

Fig. 5 Steady-state, self-field thruster (Ref. 25).

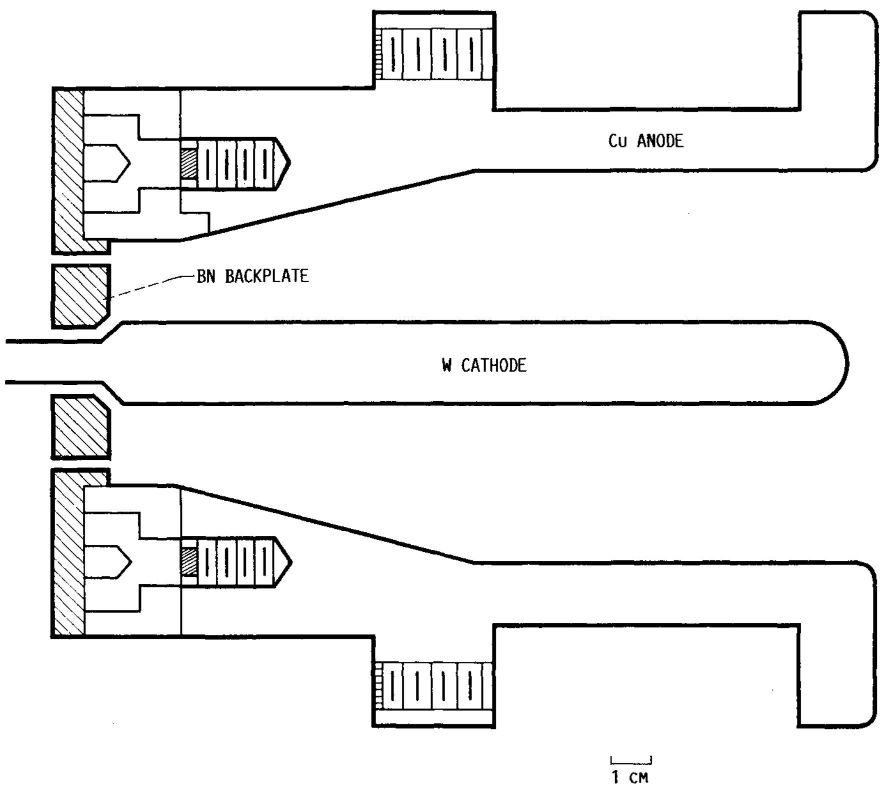

Fig. 6 Self-field, quasisteady-state thruster (Ref. 18).

The minimum anode diameter is about 2.5 cm. Hydrogen and ammonia flow rates were 2.7–4.5 g/s, respectively, at arc power levels of 0.5–4 MW.

In many high-power thrusters, propellant injection in the vicinity of the anode has been found to be effective in allowing operation at higher values of $J^2/\dot{m}$, which is a quantity found to be proportional to specific impulse.[14,15] Large values of the $J^2/\dot{m}$ parameter are limited by the "onset phenomena," which results in an arc voltage increase, arc voltage fluctuations, and, ultimately, anode deterioration.[34]

Generally, MPD thrusters operating in the 0.1- to 4-MW range have throat or minimum anode diameters in the 1- to

10-cm range with cathode diameters of approximately 1–2 cm. Mass flow rates range from 0.02 to 5 g/s. The performance of low-power MPD thrusters can be enhanced by using applied magnetic fields.[10] More detailed information is needed to determine if solenoidal magnets are required for megawatt-class thrusters to enhance performance and/or life.

Performance Measurements

Evaluation of performance is critically dependent on accurate measurements of thrust for steady-state devices and impulse for quasisteady-state thrusters. The quality of these measurements can be comprised primarily by thrust-stand thermal drift, vibration, acceleration of eroded electrode or insulator materials, and ambient gas entrainment into the discharge.[12,14,19,24,25,35] Clearly, the use of megawatt-class MPD thrusters will place significant demands on the design of vacuum facility pumping, thermal control systems, and performance diagnostics.

Figure 8 shows inconsistencies in the argon exhaust velocity diagnostics of a quasisteady-state, self-field thruster where impulse balance and Doppler shift methods were used.[14] The specific impulse values determined by an impulse measurement increase sharply with $J^2/\dot{m}$ to nearly 6000 s, whereas substantially lower values are indicated from Doppler shift diagnostics. The specific impulse calculated from the impulse measurement is extremely high for the heavy argon propellant. Although there is usually a significant uncertainty in

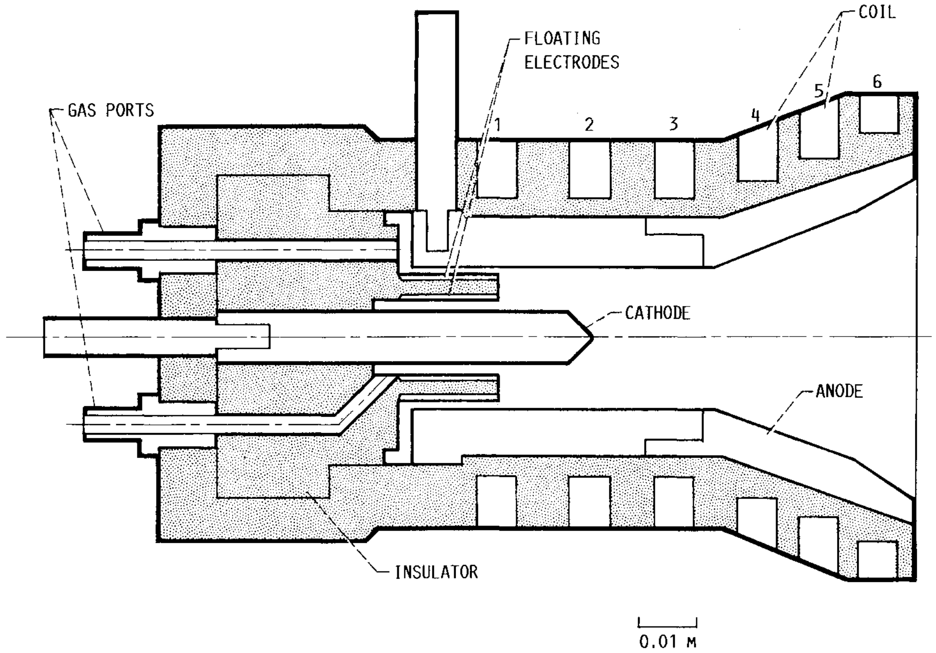

Fig. 7 Applied-field, quasisteady-state thruster (Ref. 16).

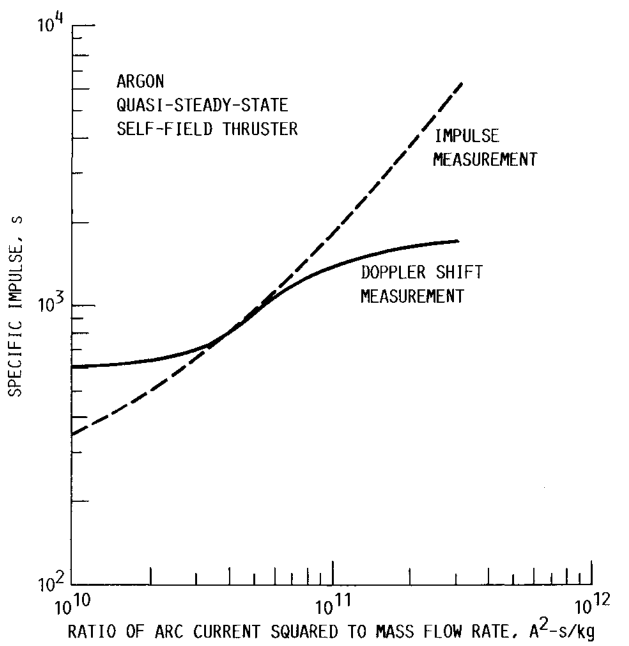

Fig. 8 Comparison of performance by impulse balance and Doppler shift measurement (Ref. 14).

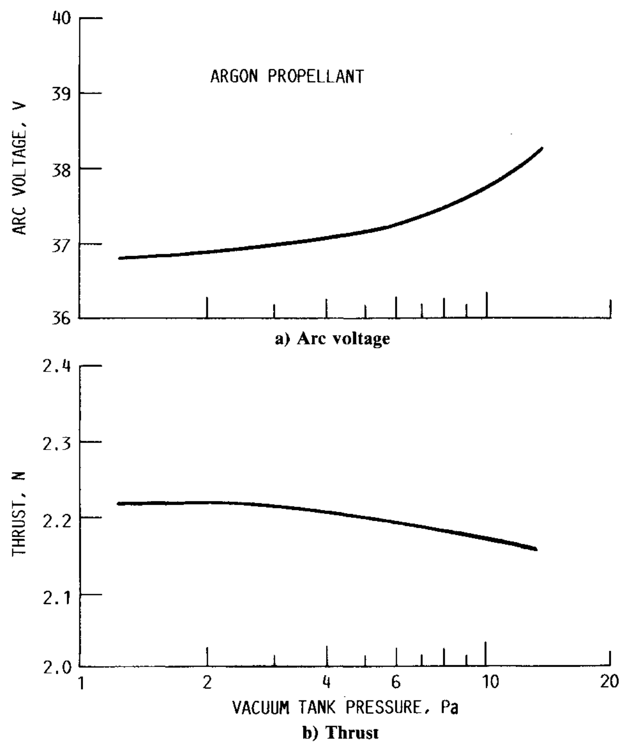

Fig. 9 Effect of vacuum tank pressure on performance of a steady-state, self-field thruster ($J = 2000$ A, $\dot{m} = 0.3$ g/s) (Ref. 24).

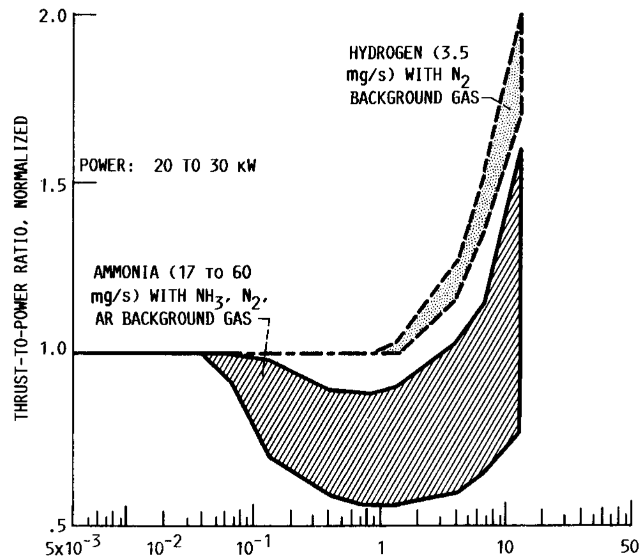

Fig. 10 Effect of vacuum tank pressure on performance of applied-field thrusters (Refs. 31, 36).

specific impulse inferred from Doppler shift measurements, it would be expected that the velocities of ionic species would be higher than those obtained by impulse measurement.[31] These data demonstrate that secondary and validating measurements of thruster performance are very important.

The effects of ambient pressure on the thrust and the arc operating voltage of an argon self-field thruster operating continuously at about 70 kW have been investigated.[24] The ambient pressure was shown to have had no effect on performance in the 1.2- to 2.7-Pa range. At ambient pressures of 13 Pa, the arc voltage increased by about 1 V, and the thrust decreased by a few percent, as shown in Fig. 9. No data were reported at pressures lower than 1.2 Pa; however, there is no indication in any of the literature that self-field MPD thruster performance is affected by ambient gases at pressures lower than 1 Pa.[24]

The effect of vacuum facility background pressure on low-power, applied magnetic-field MPD thrusters was studied from 1965 to 1970.[31,36] These MPD thrusters employed solenoidal magnets, permanent magnets, or a superconducting magnet.[10] The major vacuum facility effect was found to be the entrainment of background gases into the arc discharge at pressures as low as 0.1 Pa. In Fig. 10, normalized thrust-to-power ratio (T/P) is plotted vs the vacuum facility background pressure in order to examine the validity of thruster performance measurements. Thrust to power was chosen as the parameter of interest because there are usually changes in arc voltage as the background pressure is increased. Thrust to power is normalized to a baseline value of T/P at 7×10^{-3} Pa where the background pressure has not been found to influence the thrust or arc voltage. The spread in the T/P data is the result of variations in propellant flow rate and the type of background facility gas employed in the tests. Figure 10 shows that deceptively high performance using hydrogen thruster can be measured in the 1.3- to 13-Pa range with nitrogen as the major ambient gas. In fact, at the high ambient pressures, the thrust could be in error by as much as a factor of two. The ammonia thrust data show a thrust degradation (compared to baseline data at 7×10^{-3} Pa) of 10–45% at 1.3 Pa depending on ammonia flow rate and type of ambient gas. At 13 Pa, the measured thrust can be increased from the baseline value by as much as 60%, particularly at low ammonia flow rates ($\approx$0.02 g/s) with nitrogen and argon ambient gases. Higher ammonia flow rates, in the 0.06-g/s range, generally produced thrust-to-power ratios

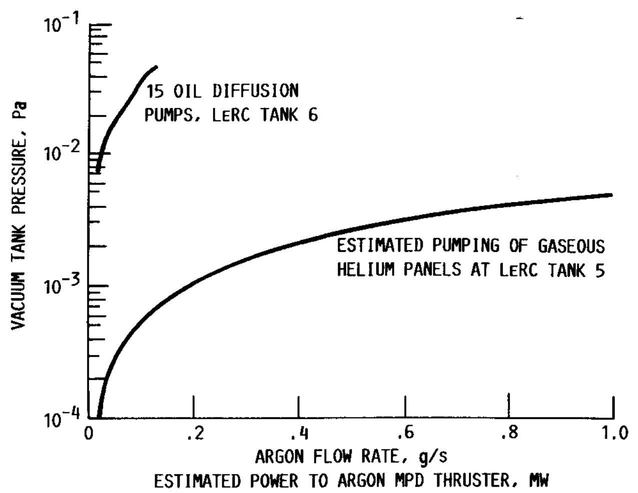

Fig. 11 NASA Lewis vacuum facility pumping capability for an MPD-thruster test stand.

lower than baseline values obtained at 7×10^{-3} Pa. Two competing effects were found. The background pressure could degrade thrust measurement by interfering with the basic acceleration mechanisms, and the background gas could be entrained in the discharge and accelerated to enhance thrust. Thrust measurement validity was found to be dependent on propellant flow rate, vacuum facility pressure, and ambient gas species. These results pertain primarily to applied-field MPD thrusters where Doppler shift velocity measurements indicate acceleration of ionic species takes place as far as 10 cm from the anode exit plane.[31] Investigators using lithium applied-field thrusters have found current streamlines follow magnetic flux lines, and up to 28% of the arc current was collected through a 7.6-cm-diam Rogowski coil located 90 cm from the anode exit plane.[37] It is not surprising that the characteristics of such plasmas can be very sensitive to ambient gas pressure.

Figure 11 shows the vacuum facility pressure sensitivity to argon flow rate in large diffusion-pumped and cryopumped facilities.[38,39] In order to maintain the low vacuum facility pressures needed to minimize ambient gas ingestion in the MPD thrusters, a diffusion pumped, sublimation pumped, or cryopumped environment is necessary. The maximum allowable pressure for the oil diffusion pump operation was approximately 0.07 Pa, because the vapor jets collapsed at higher pressures. For example, a 7.6-m-diam facility, using 15 diffusion pumps, pumped up to 0.1–0.2 g/s of argon. At these flow rates, maximum argon-thruster input power levels would be in the 0.1- to 0.2-MW range. As indicated in Fig. 11, large helium panel cryopumps with a frontal area of about 37 m^2 are projected to provide ambient pressures less than 0.01 Pa at argon flow rates in excess of 1 g/s.[39] If appropriate thruster exhaust heat exchangers are provided, the cryopump should be adequate to evaluate argon MPD thrusters operating at 1 MW. Propellants such as hydrogen or hydrogen containing molecules are poorly cryopumped by gaseous helium systems at about 26 K. Somewhat dependent on propellant type, liquid helium and/or sublimation pumped systems may be required to evaluate adequately such megawatt-class thrusters.

Operating Modes

The electromagnetic acceleration process is a strong function of arc current, and currents in the 5- to 40-kA range will probably be required for megawatt-class operation. For a given thruster geometry and mass flow rate, the current and specific impulse are limited by the "onset phenomena" where the voltage rapidly increases with current, strong arc fluctuations are observed, and ultimately anode deterioration commences.[24,34,40,41] This onset of plasma instabilities is generally described by a critical value of the parameter $(J^2/\dot{m})_c$. High currents and thus high values of $J^2/\dot{m}$ are required for self-field thrusters to produce the specific impulse levels of interest for energetic missions. The onset condition has been defined as that value of $J^2/\dot{m}$ when high-frequency voltage fluctuations (peak to peak) exceed 10% of the arc voltage.[24] One of the explanations for the onset condition is that at the high currents, the plasma is pinched, thus decreasing the pressure near the anode. This condition causes a decrease in plasma density, which leads to the development of a strong electric field in order to support the arc current.[40] Early experiments, using a variety of propellants, found the critical value of $J^2/\dot{m}$ to be approximately proportional to the inverse square root of the propellant molecular weight.[14] More sophisticated formulations of this stability criteria are summarized in Ref. 34.

Figure 12 shows the onset parameter obtained for many different thruster geometries, propellants, and flow rates; $(J^2/\dot{m})_c$ varies from 1×10^{10} to $3.5 \times 10^{11} A^2$-s/kg. Data from Refs. 41–49 as well as previous citations are shown in Fig. 12. Most of the onset data in the 1×10^{10} to $5 \times 10^{10} A^2$-s/kg range were obtained with steady-state devices whose geometries resembled electrothermal arcjets with discharge chamber (or anode) throat-to-cathode diameter ratios less than two (see Figs. 4–6). The quasisteady-state thrusters, which provide onset data from 5×10^{10} to $3.5 \times 10^{11} A^2$-s/kg, had large throat-to-cathode diameter ratios of 3.4 to 5.3. In this regard, a steady-state, low-power MPD arcjet had a $(J^2/\dot{m})_c$ value of $1.1 \times 10^{11} A^2$-s/kg using argon.[20] This thruster had a throat-to-cathode diameter ratio of five. Onset data from the 15-kW steady-state thruster are very similar to that of megawatt-class quasisteady-state thrusters using argon propellant. The arc current capability can be further extended by using propellants lighter than argon,[14,45] and by injecting propellant near the anode wall. By using a throat-to-cathode diameter ratio of 3.4 and anode propellant injection, argon onset conditions in excess of $3 \times 10^{11} A^2$-s/kg have been obtained.[15]

The electromagnetic thrust component of a self-field MPD thruster is a function of J^2.[18] The specific impulse for purely electromagnetic acceleration should then be directly related to $J^2/\dot{m}$ for self-field thrusters.[49–54] Figure 13 shows specific impulse data as a function of $J^2/\dot{m}$ for various propellants over a power range from 10 kW to 5.6 MW. The light propellants (H_2, Li) have the same specific impulse for a given $J^2/\dot{m}$. As shown in Fig. 13, some of the highest values of $J^2/\dot{m}$ ($2.1 \times 10^{11} A^2$-s/kg) and specific impulse (3000 s) for an argon self-field thruster were obtained from a thruster with a 10-cm anode diameter, with anode propellant injection and a flared anode configuration.[15]

The $J^2/\dot{m}$ parameter is not a good indicator of specific impulse capability of applied-magnetic-field thrusters since acceleration mechanisms of such thrusters are more complex than those produced in self-field devices. Relatively high specific impulse measurements have been obtained from low-power thrusters using applied fields in the 0.1- to 0.2-T range. A specific impulse of 2800 s was obtained with argon at very low values of $J^2/\dot{m}$ of 1 to $2 \times 10^{10} A^2$-s/kg.[10,12] The low-power, applied-field thrusters have produced about the same specific impulse as megawatt-class thrusters that operate at values of $J^2/\dot{m}$, which are an order of magnitude higher. It has also been shown that in some MPD-thruster configurations, applied magnetic fields significantly enhance performance at the megawatt power level. At $J^2/\dot{m}$ of about $7.5 \times 10^{10} A^2$-s/kg, the specific impulse of a quasisteady-state hydrogen thruster was increased by 59% to 3500 s by changing from purely self-field to applied-field operation.[16] In this case, the peak pulsed power was 3–4 MW. It is clear that the applied magnetic field can be a strong optimization tool for MPD thrusters particularly at modes values of $J^2/\dot{m}$. The detailed roles of applied magnetic fields in the optimization of megawatt-class thruster performance and lifetime limits need to be carefully examined.

Thruster Performance

Increases in thruster-specific impulse and efficiency can provide major benefits in reduced cargo vehicle mass in LEO and in improved payload capability. Important MPD-thruster, experimentally determined, performance parameters are reviewed. Usually the highest values of specific impulse/efficiency for each propellant are displayed for a given reference. Data were taken from direct thrust measurements of steady state and pulsed MPD thrusters. No calculated performance predictions are included in the report primarily because of the complexities involved in developing models to reliably determine electromagnetic and gas dynamic thrust contributions.

Thrust data are not cited if the average mass flow of electrode and insulator erosion products was reported to be more than 10% of the propellant flow rate. Because the vacuum facility pressure plays an important role in the thrust determination, only those data taken at background pressures less than 0.1 Pa are usually cited. Some thrust data taken at higher pressures are reported here if evidence was presented indicating that facility effects were minimal.

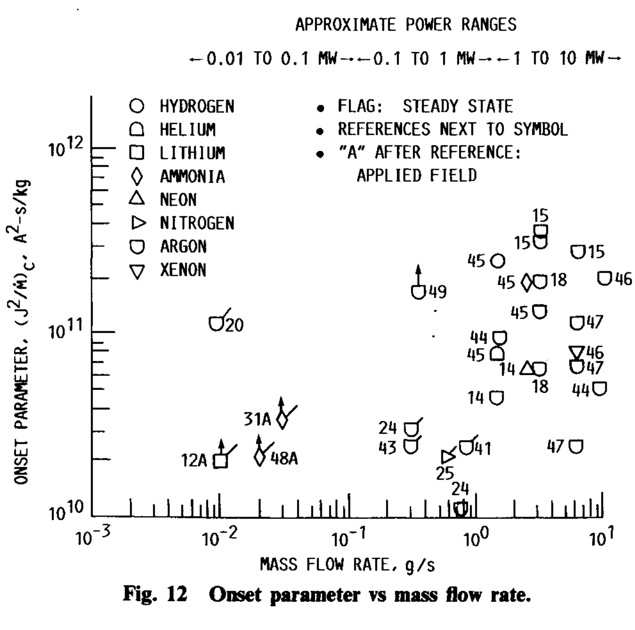

Fig. 12 Onset parameter vs mass flow rate.

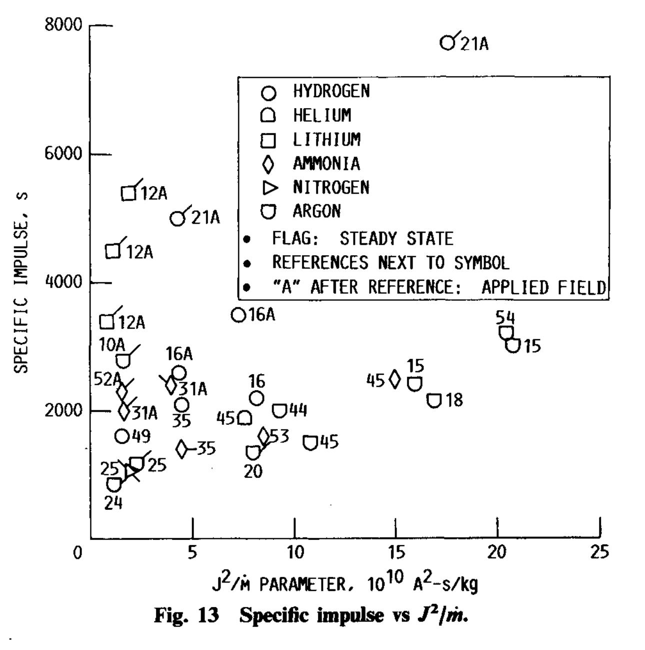

Fig. 13 Specific impulse vs $J^2/\dot{m}$.

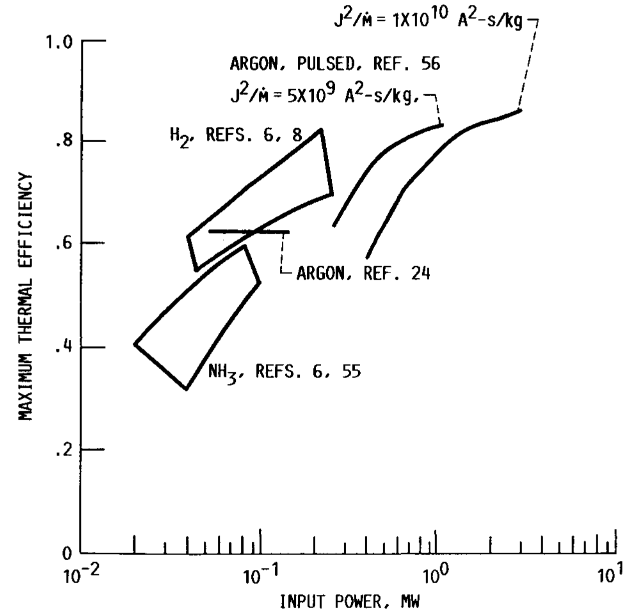

Fig. 14 Variation of thermal efficiency as a function of input power.

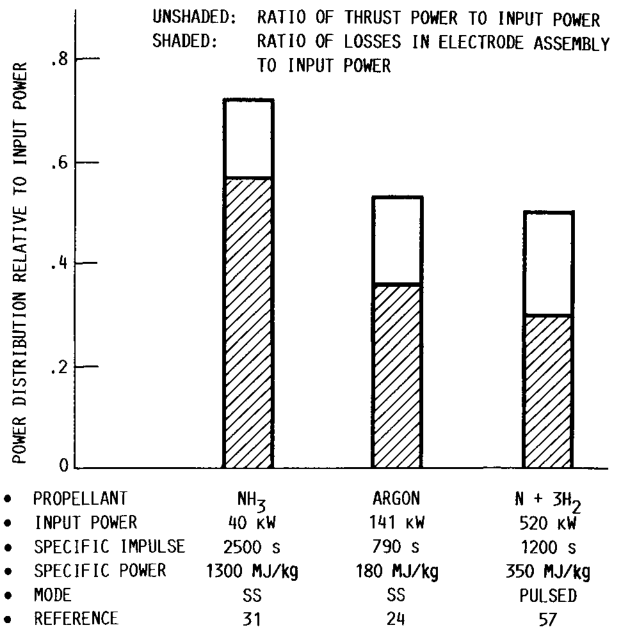

Fig. 15 MPD-thruster power distribution.

The thrust efficiency is:

$$\eta = gTI_{sp}/2P \tag{1}$$

where

$$P = P + \dot{m}h_0 \tag{2}$$

Magnet power is not considered in any of the efficiency calculations. The room temperature gas power $\dot{m}h_0$ is usually small enough to be negligible. The highest contribution of $\dot{m}h_0$ would probably be provided by hydrogen. As an example, a typical 3.6-MW hydrogen thruster flow rate might be 2.75 g/s.[16] In this case, $\dot{m}h_0$ would be only 0.3% of the electric power.

An upper bound on the thrust efficiency is the thermal efficiency, which is a measure of the maximum amount of power that can be transferred to the propellant. The thermal efficiency for actively cooled devices is given by

$$\eta_{th} = P - P_L/P \tag{3}$$

where P_L is the power lost to the arcjet water cooling system. In many experiments, P_L is simply approximated by the power lost to the anode since this loss mechanism dominates.

Figure 14 shows thermal efficiencies of thrusters operating at power levels from 20 kW to 3 MW. Data are cited from Refs. 6, 8, 24, 55, and 56. Thermal efficiencies of argon and ammonia thrusters operating at less than 130 kW are less than 65%. Hydrogen thrusters have demonstrated about 80% conversion of electric power to power available for propulsion. Most of these data are upper bounds because relatively small losses like those associated with the cathode are sometimes not included in the calculation. Thermal efficiencies were also derived from anode heat flux calculations based on temperature data from a megawatt-class, quasisteady-state thruster.[56] Thermal efficiency estimates based on anode losses vary from 78 to 87% in the 1- to 3-MW range using argon as the propellant. Overall, these data imply there is potential for high performance at these high powers since the power input to the plasma increases with power level. Figure 14 also shows there is a reduction in thermal efficiency as $J^2/\dot{m}$ increases in a quasisteady-state thruster. Since increases in $J^2/\dot{m}$ are associated with increases in specific impulse, data of megawatt-class, steady-state thrusters are needed to further examine performance sensitivities to $J^2/\dot{m}$.

The power deposited in the anode of the quasisteady-state thruster can be approximated by the sum of 1) the power associated with the conduction electron anode voltage drop, 2) power invested in heating electrons, and 3) power associated with absorbing the electrons at the anode.[56] The anode power loss P_a is

$$P_a = J\left(V_a + \frac{5}{2}\frac{kT_e}{e} + \phi\right) \tag{4}$$

At 1.3 MW, it was estimated that the power losses associated with the anode drop, heating conduction electrons, and anode work function were 14, 3, and 4% of the input electric power, respectively. The major anode loss was associated with depositing the kinetic energy of the conduction electrons into the anode.

Figure 15 shows the extent of typical arc electrode losses and the thrust power relative to the input electric power.[24,31,57] At the low input power level, the specific power was 1300 MJ/kg and the specific impulse was 2500 s. The resultant electrode losses were 57%. At the lower specific powers of 180–350 MJ/kg, the electrode losses were reduced. Thrust efficiencies in these three cases were about the same and varied from 0.17 to 0.20. The remainder of the losses are associated with dissociation, ionization, excitation, exhaust divergence, and viscous flow effects.

At present, there are no thermal efficiency or performance limit data for megawatt-class MPD thrusters operating in the steady-state mode. Anode losses and impacts of operating at high values of $J^2/\dot{m}$ and high specific impulse near onset should be identified and quantified. Early in technology programs and prior to development of high-fidelity thrust stands, thermal efficiencies for steady-state, megawatt-class MPD thrusters should be obtained so that upper bounds on thrust efficiency can be determined.

The best MPD-thruster performance data obtained to date are shown in Figs. 16–19. References are cited on each figure. Some data in the literature are not reported because of evidence of high electrode erosion. Other data are not included if the vacuum facility pressure was greater than 0.1 Pa since this condition might influence the thrust measurements due to gas entrainment in the discharge. Hydrogen and lithium data are shown in Fig. 16. The best performance using hydrogen has been obtained with quasisteady-state devices. The thrust efficiency of about 0.43 was obtained at specific impulse values of 3500 and 4900 s and input power levels of 3.6 and 1.5 MW, respectively. Both self-field and applied-field devices provided high thrust efficiencies (0.43–0.45).[16,58] High values of thrust efficiency (0.48–0.68) were obtained with lithium at powers less than 20 kW.

At these low power levels, applied magnetic fields were used to provide operations at 3400–5400-s specific impulse. High-performance hydrogen and lithium thrusters, operating at a specific impulse of about 5000 s, have thrust efficiencies that are at least 20% (H_2) and 90% (Li) higher than the best efficiencies obtained with the propellants NH_3, N_2H_4, Ar, or N_2. A common property shared by hydrogen and lithium propellants is that there are virtually no losses associated with the production of multiply charged species since dissociated hydrogen can only be singly ionized, and lithium has a second ionization potential of 75 eV. Lithium may not, however, be an attractive propellant because of the potential for spacecraft contamination due to propellant condensation. Hydrogen is a very attractive propellant for lunar and Mars cargo vehicles because of its high performance and the fact that it is likely that transportation scenarios will involve long-term storage of hydrogen/oxygen for high-thrust propulsion of manned and cargo vehicles.

Figure 17 shows the performance of inert gas MPD thrusters. Specific impulse values in the 2000- to 4000-s range have been obtained using pulsed, self-field thrusters and also low-power, applied-field thrusters using argon propellant. Reported efficiencies for argon have not exceeded 0.35. The efficiencies of high-power, argon, pulsed MPD thrusters in the 2000- to 3500-s specific impulse range were 0.2–0.3. Significant amounts of power can be invested in the production of multiply charged argon ions, as the first three ionization potentials for argon are only 15.8, 27.6, and 40.9 eV.

Data from thrusters using nitrogen containing propellants such as N_2, NH_3, and N_2H_4 (Ref. 59) are displayed in Fig. 18. Values of specific impulse in the 2000- to 3500-s range have been obtained with 30 kW and megawatt-class, pulsed thrusters; the thrust efficiencies are in the 0.2 to 0.3 range. Molecular propellants containing hydrogen have been found to produce relatively high performance compared to argon because, in addition to the electromagnetic thrust component, there is also significant electrothermal acceleration.[42] Ammonia and hydrazine propellants are also attractive from the system standpoint because they are space storable. On the negative side, significant amounts of power can be dissipated in the dissociation and ionization of ammonia and its products. It has been estimated that the energy cost for the ammonia molecule to fully dissociate and ionize is at least 70 eV.[31] Using this estimate, a fully ionized 1-MW thruster with an ammonia flow rate of 1 g/s would have about 40% of its power in dissociation and ionization. A significant effort is needed to assess performance limits of these space storable propellants at megawatt power levels in both the pulsed and steady-state mode.

Figure 19 shows the thrust efficiency for various MPD thrusters operating with a wide variety of propellants over a large range of input electric power. Low power (20–30 kW) efficiency data for hydrogen, ammonia, and argon have been reported as high as 0.30–0.35. Lithium efficiencies are as high as 0.69. In all cases, the low-power devices required applied magnetic fields to attain such performance levels. Efficiency data in the 1- to 6-MW range were obtained exclusively from

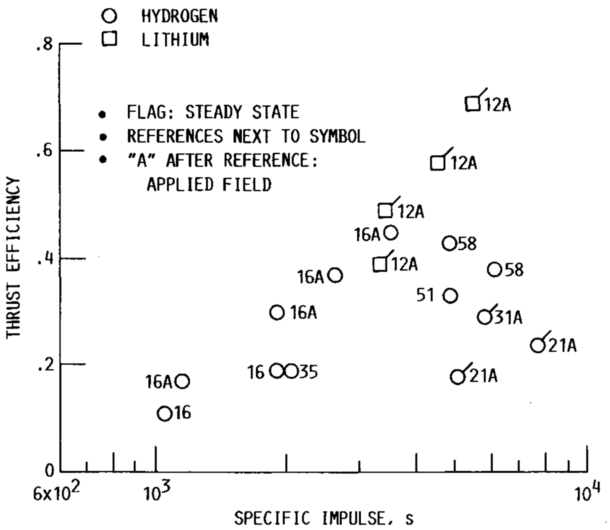

Fig. 16 Performance of hydrogen and lithium thrusters.

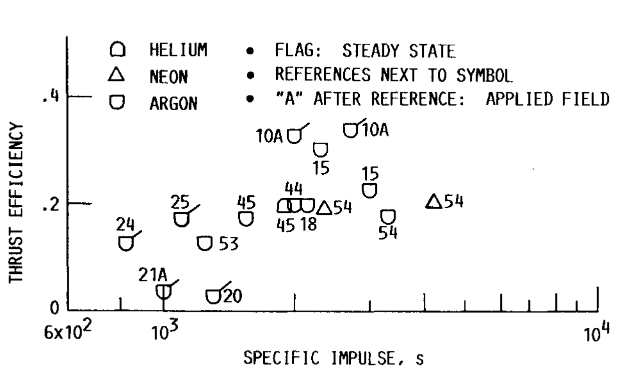

Fig. 17 Performance of inert gas thrusters.

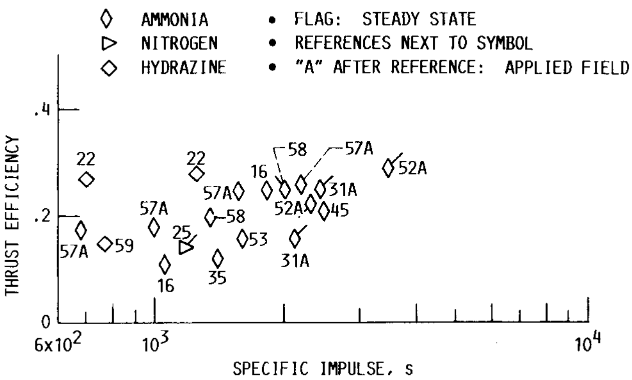

Fig. 18 Performance of propellants containing nitrogen.

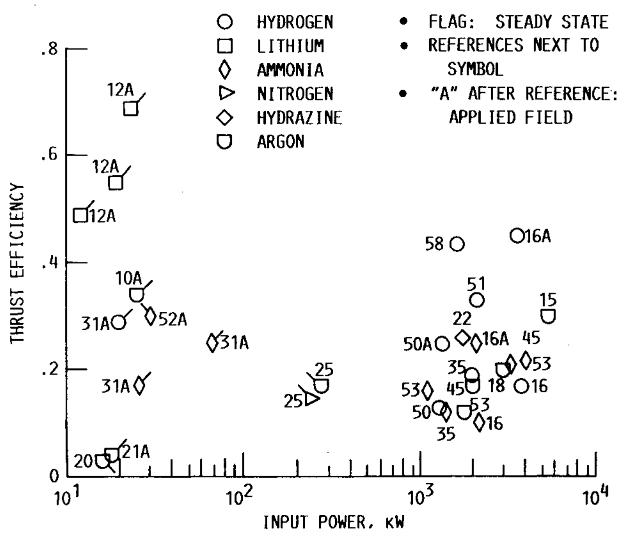

Fig. 19 Thrust efficiency as a function of power and propellant.

Table 1 Demonstrated life[a]

	Self-field		Applied field	
	Quasisteady state	Steady state	Steady state	Steady state
Demonstrated total impulse, Ns	2×10^4	3×10^4	1×10^6	5×10^4
Operating time, h	0.2	1	500	50
Cathode erosion rate,				
g/kA/kh	3600	100	9	0.14
μg/C	1	3×10^{-2}	3×10^{-3}	4×10^{-5}
Gas	NH_3	Ar	NH_3	H_2
Power, kW	1200	273	33	122
Specific impulse, s	2000	1100	1900	5900
Thrust efficiency	0.2	0.17	0.16	0.34
Reference	23	24,25	7	8,60

[a]Uncertainty in performance may exist due to the facility effects.

Table 2 Cathode life estimate

Assumption

Steady-state operation
1-cm diam by 8-cm long thoriated tungsten cathode
0.5-MW thruster, 5000 A, 100 V
Constant erosion rate
10% cathode mass loss is end of life

Result

Reference	23	24,25	7	8,60
Cathode erosion rate, μg/C	1	3×10^{-2}	3×10^{-3}	4×10^{-5}
End of life, h (estimate)	0.6	21	240	15,000

quasisteady-state thrusters. The highest efficiency observed (0.45) was obtained with hydrogen, and data from all other propellants are grouped in the 0.10 to 0.35 range.

If MPD-thruster efficiency requirements fall in the 0.5 to 0.7 range, a significant effort will be required to determine if propellants other than hydrogen are candidates for systems required to perform energetic missions whose velocity increment ΔV is in excess of 10 km/s. In addition to overall performance limit evaluations, it is also necessary to detail the partitioning of power in the MPD thrusters in order to clearly quantify the losses at electrodes as well as the power invested in dissociation, ionization, and excitation. Nonintrusive diagnostics methods to locally determine propellant species, charge-state, densities, and velocities will be important in the characterization of the most promising thruster configurations. Performance limit and power-loss diagnostics of megawatt-class devices will have to be performed on quasisteady-state thrusters and also steady-state thrusters, which require sophisticated test environments and test stands. Final assessments must be made with steady-state thrusters since pulsed devices do not necessarily have the same cathode plasma characteristics, plasma stability relations, or degree of plasma heating for the same input specific power.[25]

Thruster Lifetime

First-order calculations of MPD propulsion requirements for lunar and Mars cargo vehicles indicate that the total impulse and life per thruster should be 1×10^8 to 1×10^9 Ns and 2000 to 8000 h, respectively. Specific impulse was assumed to be in the 2000- to 5000-s range with power per thruster of 0.5–3 MW. With these thruster endurance targets in mind, the present state of the art will be briefly characterized. Table 1 shows the best total impulses and extended test

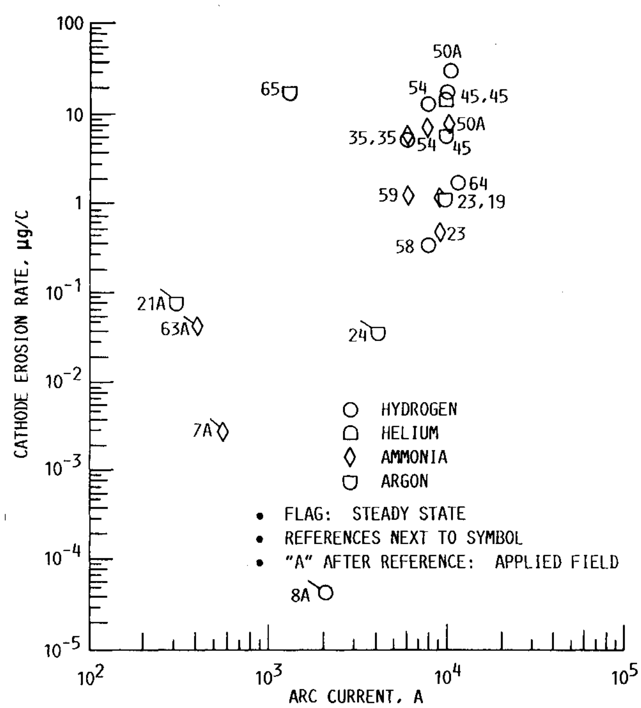

Fig. 20 Cathode erosion rates.

demonstration of MPD thrusters. A quasisteady-state device was operated 1×10^6 cycles with ammonia at 1.2 MW to produce 2×10^4 Ns or a total operating time of about 0.2 h.[23] Steady-state thrusters in the 0.1- to 0.3-MW range have been operated for periods of 1–50 h to produce about 5×10^4 Ns total impulse.[24,25] In 1969, a 500-h test was conducted with a low-power (33 kW) MPD thruster, which provided about 1×10^6 Ns.[7]

Technology demonstrations of MPD thrusters to date have produced total impulse values that are two to three orders of magnitude below estimated requirements for cargo vehicle propulsion. Because extended testing of an MPD thruster is very expensive, short-term tests are usually conducted to determine the mass loss of the cathode. The cathode is the most likely component to fail if operation is not conducted above the "onset" condition. Cathode mass loss has been directly measured, or in situ measurements have been made using radioactive tracers.[19] Cathode erosion rate data are also shown in Table 1. A quasisteady-state thruster[23] demonstrated a million cold starts. Because of the cold starts, a higher erosion rate of 1 μg/C was obtained. For example, an average erosion rate of 1 μg/C results in the loss of 3.6 g of cathode material per kA of arc current for each hour of operation. Reported cathode erosion rates for steady-state thrusters vary from 0.03 $\mu g/C$ in a 100-kW class[24] thruster to 4×10^{-5} $\mu g/C$, which is the remarkable value obtained using an applied-field thruster.[8] The applied-field thruster was tested for 50 h at about 2.1 kA. The very low rate was attributed to operation at very low chamber pressure, which allowed the arc to spread over a large surface of the cathode.[60] Application of the low-power electromagnet may also have been a factor in the successful experiment. It was reported that an applied field of about 0.2 T improved anode life by preventing arc attachment on the anode in the form of localized spots. The effects of applied magnetic fields on the life of both electrodes was found to be dependent on thruster geometry, arc current, and the type of propellant in the limited number of tests that were conducted.[61]

To provide some understanding of the cathode erosion rate impact on lifetime, cathode erosion data are shown in Table 2, and end-of-life estimates are made. It was assumed that the erosion rate is linear with respect to current and operating time. It was further assumed that a 10% cathode mass loss (in the discharge region) represents an end-of-life condition; this criteria is consistent with the lamp industry and resistojet heater end-of-life predictions.[62] Assumptions concerning cathode diameter and discharge conditions ($J = 5000$ A) are shown in Table 2. The projected lifetimes of the quasisteady-state, 0.27-MW, self-field, low-power applied field, and 0.12-MW applied-field thrusters using these assumptions are 0.6, 21, 240, and 15,000 h, respectively. From these estimates, the projected life of steady-state MPD thrusters may possibly be improved by a factor of 700 by appropriate selection of design and operating conditions.

Figure 20 shows the cathode erosion rate vs arc current including a large data base. Data from Refs. 63–65 as well as previous citations are shown in Fig. 20. The highest erosion rates are associated with cold-start, pulsed devices that used thoriated tungsten rod cathodes. Generally, about an order of magnitude lower rates were obtained with pulsed thrusters using porous tungsten cathodes impregnated with barium compounds. Heavy erosion generally takes place on a cathode cold start and results in ejection of glowing particles.[24] Because pulsed devices are usually started cold, the cathode erosion rates are extremely high and range from 3–30 $\mu g/C$. Cathodes with such erosion rates cannot be expected to operate reliably for more than a few hours.

In the steady-state, hot-cathode operation, cathode erosion takes place at a much lower rate.[24] Most of the steady-state cathode erosion rate data fall in the range of 0.003 to 0.07 $\mu g/C$ as shown in Fig. 21. Such erosion rates would result in very high cathode mass losses, which are inconsistent with MPD-thruster lifetimes of a few thousand hours at arc currents in excess of 10 kA.

The lowest cathode erosion rate was 4×10^{-5} $\mu g/C$. The steady-state MPD thruster used to obtain this data was operated with hydrogen at an arc current of 2.1 kA.[8] An MPD thruster with such a cathode erosion rate would very likely be capable of operating at megawatt power levels for thousands of hours. This experiment[8] should be re-evaluated in order to understand better electrode wear mechanisms and to determine potential impacts on performance.

The MPD-thruster extended tests indicate that the total impulse demonstrated with state-of-the-art MPD thrusters needs to be increased by at least two to three orders of magnitude to satisfy cargo vehicle propulsion requirements. Cathode erosion rates for most current thrusters are in excess of 0.003 $\mu g/C$, which is unacceptable for 5–10 kA operation for a few thousand hours. Tests conducted over two decades ago offer encouragement that under certain conditions, cathode erosion rates may be reduced by nearly two orders of magnitude to values near 4×10^{-5} $\mu g/C$.[8]

Concluding Remarks

Space transportation scenarios and development of requirements for exploration class missions are ongoing. Thruster power for low-thrust, cargo-vehicle propulsion is estimated to be in the 0.5- to 3-MW range with total impulses of 1×10^8 to 1×10^9 Ns at specific impulse levels of 2000–5000 s. MPD-thruster lifetimes of 2000–8000 h will probably be required.

MPD-thruster technology has been pursued for about 25 years, although major industrial programs in the United States for the development of high-power thruster technology ceased circa 1970. Plasma research related to MPD thrusters continued from 1970 to the present primarily at universities in the United States, Japan, and Europe. Most of the research centered on the quasisteady-state devices that minimize test facilities requirements.

Over the last three decades, a significant effort has been expended to determine the effect of vacuum facility background pressure on performance. There is no evidence in the literature that self-field thrusters are affected by background gases at pressures less than 0.1 Pa. With the applied-field thruster, two competing effects involving background pressure were found. The background pressure could degrade thrust by interfering with the acceleration mechanisms, and the background gas could be entrained in the discharge to enhance thrust. In order to minimize vacuum facility effects on applied field thruster performance, diagnostic measurements should be carried out in a vacuum environment whose pressure is less than 0.04 Pa. Existing vacuum chambers are adequate to handle the flow of argon or nitrogen MPD thrusters operating up to 1-MW input power. Development work will be required for megawatt-class test facilities. Testing of megawatt-class thrusters using hydrogen-containing propellants will require modifications to vacuum facilities and thermal control systems. In addition to high throughput vacuum systems, high-fidelity test stands are required to adequately characterize MPD-thruster performance. A detailed definition of the appropriate test environment for a megawatt-class MPD thruster has not been defined.

Because high specific impulse is related to high-current capability at relatively low flow rates, definition of operating modes and stability limits become important. Flared anode MPD thrusters operating in the quasisteady-state mode have demonstrated arc currents up to 31 kA (at 3 g/s) by injecting argon propellant near the anode wall. Onset parameters ($J^2/\dot{m}$) were in 3.2×10^{11} A^2-s/kg range. Argon specific impulse values as high as 3000 s have been reported at a $J^2/\dot{m}$ of 2×10^{11} A^2-s/kg. Significant increases in onset parameter, current capability, and specific impulse have been demonstrated by using propellants lighter than argon.

Thermal efficiencies which provide an upperbound on thruster efficiencies have been reported to be about 0.60 for steady-state argon and ammonia thrusters at powers of approximately 0.1 MW. Hydrogen thrusters (0.2 MW) had thermal efficiencies as high as 0.80, and argon pulsed devices had thermal losses estimated at only 13% at 3 MW. These data imply there is potential for high performance at the high power levels.

Relatively high MPD-thruster efficiencies of 0.43 and 0.69 were obtained at about 5000-s specific impulse using hydrogen and lithium, respectively. The hydrogen thruster was a quasi-steady-state device, and the lithium data were obtained with a 20-kW, steady-state thruster. Steady-state and pulsed argon thrusters have demonstrated efficiencies of 0.30 – 0.34 at 2400 – 2800-s specific impulse. All other propellants (NH_3, N_2H_4, N_2, He, and Ne) produced efficiencies of 0.10 – 0.30 in the 1000- to 4500-s specific impulse range. Propellants that are attractive from the systems storage standpoint (NH_3, N_2H_4, and Ar) suffer anode losses that are probably 15–40% of the input power as well as large losses associated with dissociation, excitation, and multiple ionization of propellant species. Generally, the efficiency and specific impulse limits have been increased by using a throat-to-cathode diameter of at least 3.4, anode propellant injection, a flared-anode geometry, and light propellants. In some thruster geometries, applied magnetic fields play an important role. By changing from purely self-field to applied magnetic field, the specific impulse of a pulsed hydrogen thruster was increased by nearly 60%. The maximum thrust efficiencies of applied-field devices operating at less than 50 kW with ammonia or argon are about the same as the best data from pulsed, megawatt-class thrusters using these propellants. Effects of applied magnetic fields in the optimization of megawatt-class thrusters should be further examined. There is also a need for performance-limit evaluations and determination of losses associated with electrodes, dissociation, ionization, and excitation. Nonintrusive diagnostic methods need to be developed to determine propellant species, charge state, densities, and velocities so that accurate theoretical treatments of the MPD plasma acceleration and energy transfer processes can be further developed.

Extended tests of pulsed and steady-state MPD thrusters have demonstrated total impulse capabilities of about 2×10^4 and 1×10^6 Ns, respectively. The longest operating time was 500 h for a 33-kW ammonia MPD thruster with 1900-s specific impulse. The demonstrated total impulse values are at least two to three orders of magnitude below that required for cargo vehicle propulsion.

Cathode erosion rates have been used as a thruster lifetime figure of merit because the cathode is considered the most likely component to fail if operation is conducted above the onset condition. Enormous cathode erosion rates in the 0.3- to 30-μg/C range have been observed for pulsed situations with many cold starts. Steady-state evaluations yield rates over a wide range, 4×10^{-5}–0.07 μg/C. Acceptably low rates (4×10^{-5} μg/C) were obtained with a low pressure arc chamber and an applied electromagnetic field, which apparently uniformly distributed the arc current over a large cathode area. An experimental and analytical effort is needed to thoroughly examine the effects of electrode design, applied fields, and operating conditions on electrode lifetime.

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