As mentioned above, the arcjet geometry of radiation-cooled thruster was the same as that of a previously investigated water-cooled thruster. A comparison of the results shows that the thrust efficiency with the radiation-cooled thruster is about 20% higher, and the specific impulse is about 320-s higher at the same power level.[7] This can be explained by a regenerative cooling effect. The incoming gas is heated to about 900°C in the radiation-cooled device and a greater part of the heat is regained than in the water-cooled thruster. Although the operating characteristics of water-cooled and radiation-cooled devices are quite different, data obtained from water-cooled thrusters are useful for determining general operating conditions and performance trends, and they give a theoretical insight into the physics of the system.[6]

Conclusions

In this Note it is shown that with hydrogen as propellant, specific impulses of more than 1100 s with efficiencies of 35–40% can be achieved. This high specific impulse makes hydrogen arcjets very interesting for orbit transfer applications of larger satellites. Although the highest specific impulses can be achieved with hydrogen, new storage systems have to be developed if this thruster is to be used for orbit transfer vehicles.

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Applied-Field MPD Thruster Performance with Hydrogen and Argon Propellants

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Introduction

MAGNETOPLASMADYNAMIC (MPD) thrusters have demonstrated performance levels and power handling capabilities approaching those required for Earth orbit, robotic planetary, cargo, and piloted missions.[1,2] These thrusters employ electromagnetic body forces generated by the interaction of a discharge current passing through the propellant and either the self-induced or externally applied magnetic field to accelerate a propellant to velocities between 10–50 km/s.[2] Self-field MPD thrusters using argon or nitrogen propellant have been operated continuously at power levels over 500 kW at 20% power-to-thrust conversion efficiency and 1200-s specific impulse, or pulsed at power levels of several megawatts with efficiencies of up to 40% and 5000-s specific impulse.[2] However, the performance of self-field thrusters decreases rapidly as the thruster power is decreased, and the highest power spacecraft currently planned is the SP-100 space nuclear reactor, which will deliver 100 kWe.[1] This power limitation has forced a more detailed evaluation of applied magnetic fields, thruster geometries, and propellants, in the expectation that a high performance thruster design can be established at power levels of interest.

Previous reports have presented results for 12 applied-field MPD thruster geometries operating on argon propellant at power levels between 20–150 kW.[3,4] Thruster performance increased monotonically with the applied magnetic field strength, reaching 23% efficiency at 2300-s specific impulse. The specific impulse increased with decreasing flow rate, but the efficiency decreased. While the applied-field clearly improved thruster performance, the best performance with argon propellant was not high enough to satisfy mission requirements. Work in the 1960s in the U.S., and more recently in Japan, indicated that using hydrogen propellant substantially increased both thruster specific impulse and efficiency.[5,6] However, both those efforts were conducted in facilities with ambient pressures that have been shown to affect thruster performance,[3,7] and neither one provided a quantitative comparison of performance with argon and hydrogen. This report presents results of a study in which a single thruster geometry was tested with argon and hydrogen propellants across the same range of operating conditions. Following a brief description of the thruster design and test facility, the performance results are presented and their implications for power deposition and thruster loss mechanisms are discussed. Finally, a summary of the study conclusions is given.

Experimental Apparatus

Thruster Design

The thruster, shown in Fig. 1, consisted of a 5.0-cm-i.d. copper anode with a coaxial, 1.2-cm-diam, 2% thoriated tungsten cathode. Both electrodes were 7.6-cm long. The thruster was water-cooled, the anode through passages in its wall, the cathode by conduction through a water-cooled cathode clamp. A boron nitride backplate was used to inject half the propellant flow through an annulus around the cathode base and half through 24 holes at a radius of 1.6 cm. The MPD thruster was mounted inside a 15.3-cm-i.d., 15.3-cm-long coil consisting of 36 turns of 1.3-cm-diam copper tubing. The thruster was mounted such that the anode exit plane was flush with the end of the solenoid. Calibrations of the axial field strength at the centerline of the magnet exit plane as a function of coil current yielded strengths of 1.66×10^{-4} T/A.

Test Facility and Diagnostics

The thruster test stand was mounted in a 3-m-diam test chamber separated from the main 7.6-m-diam, 21-m-long tank by a 3-m-diam gate valve. The main tank was pumped by 20 0.9-m oil diffusion pumps backed by three roots blowers and two roughing pumps. The facility pressure was below 0.07 Pa (5×10^{-4} Torr) during all tests.[3]

The thruster power supply consisted of a series-parallel network of six 65-kW arc welding supplies. This network, which was electrically isolated from ground, could supply up to 3000 A at 130 V to the thruster. The applied-field magnet

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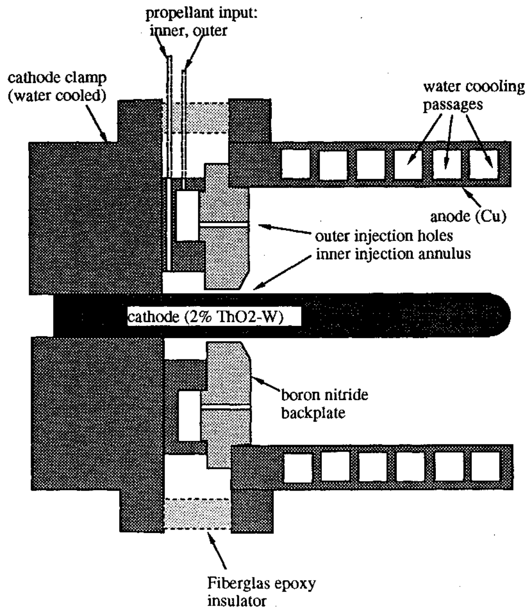

Fig. 1 Applied-field MPD thruster used for these tests.

was powered using a single welding supply which provided up to 1500 A to the coil. A 2-kW, 1400-V dc supply was used to ignite the arc by momentarily connecting it across the thruster electrodes.

Both the thruster and applied-field magnet were water-cooled using closed-loop heat exchangers. The cooling water flow rate was measured using a turbine flow meter calibrated to within 2%. The water flow rate was used with measurements of its inlet and outlet temperatures to calculate the heat transfer to the thruster electrodes.

Propellant was delivered to the thruster using two independent flow paths, each of which was fitted with a thermal conductivity type flow controller with 2% precision. A constant volume calibration system was used to check the accuracy of the flow controllers. The results showed that the flow rates were accurate to within 3%.

The thrust stand consisted of an inverted pendulum with an oscillation damping circuit, remote leveling mechanism, and an in situ calibration mechanism.[8] Extensive tests were conducted to eliminate magnetic, thermal, and facility induced tares on the thrust measurements.[3] This effort reduced the experimental error in the thrust measurement to less than 3%.

Performance Measurements

Tests were conducted using argon and hydrogen propellants at a flow rate of 0.025 g/s and a discharge current of 750 A. The applied-field strength was varied from a minimum of 0.034 T to a maximum of 0.068 T with argon propellant, and 0.084 T with hydrogen propellant. It was not possible to obtain stable operation with zero applied-field, and increasing the field strength above the maxima listed resulted in rapid erosion as evidenced by particulate emission. One effect of using hydrogen propellant was to increase the maximum stable applied-field strength. Weight loss measurements of the cathode showed that electrode erosion contributed less than 0.05% of the flow rate through the chamber, indicating that its effect on thruster performance was negligible.

The discharge voltage increased linearly with applied-field strength for both argon and hydrogen, though the rate of voltage increase with applied-field strength was slightly higher with argon than hydrogen. As shown in Fig. 2, the magnitude of the voltage with hydrogen was 20 V higher than that for argon at a field strength of 0.034 T, though this difference decreased to approximately 10 V at 0.068 T. Maximum voltages for the two propellants were 82 and 110 V for argon and hydrogen, respectively, yielding maximum power levels of 61.5 and 82.5 kW.

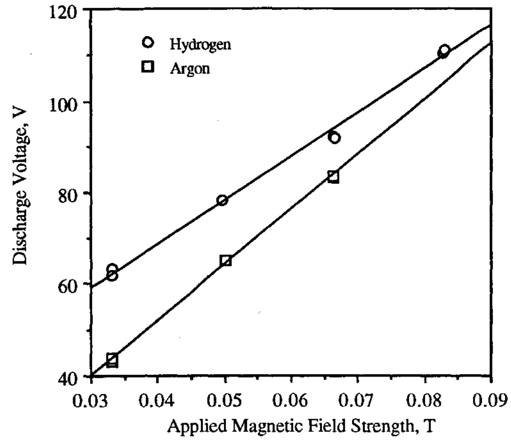

Fig. 2 Discharge voltage vs applied magnetic field strength for argon and hydrogen propellants.

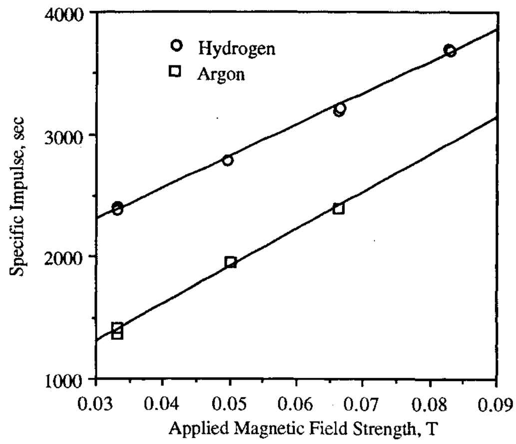

Fig. 3 Specific impulse vs applied magnetic field strength for argon and hydrogen propellants.

As shown in Fig. 3, similar behavior was observed with the specific impulse. Hydrogen yielded approximately 1000 s higher specific impulse at a given applied-field strength, and the rate of increase of specific impulse with applied-field strength was nearly the same for the two propellants. However, the increased stable operating range obtained with hydrogen propellant resulted in a maximum measured I_{sp} of 3700 s, as compared with 2400 s with argon. Specific impulses for self-field accelerators using argon and hydrogen propellants have not exceeded 3000 s across all power levels tested,[2,9] indicating that there are substantial performance benefits to applied-field thrusters.

The dependence of thruster efficiency on the applied-field strength for the two propellants is shown in Fig. 4, and again, the rate of increase was similar for the two propellants. While the values for both propellants are lower than desired, there is a clear advantage to using hydrogen. Efficiencies with hydrogen are nearly a factor of 2 higher than those obtained with argon propellant, reaching 20% efficiency at 3700-s specific impulse. The highest performance measured with argon propellant was 11% efficiency at 2400-s specific impulse. The similar behaviors of discharge voltage, specific impulse, and efficiency for the two propellants suggests that the empirical scaling laws obtained using argon propellant, such as those established in Refs. 3 and 4, may be transportable to hydrogen thrusters.

Power Balance

The calorimetrically measured anode power was divided by the total thruster input power to obtain the anode power

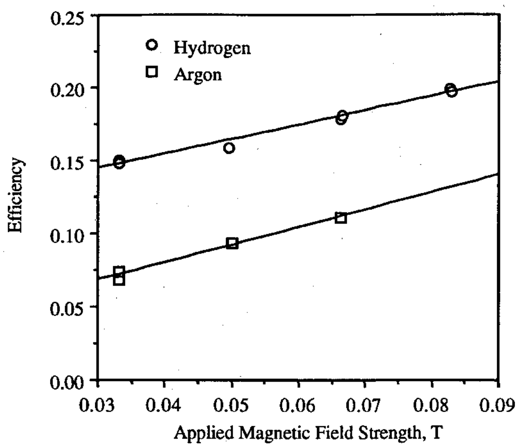

Fig. 4 Efficiency vs applied magnetic field strength for argon and hydrogen propellants.

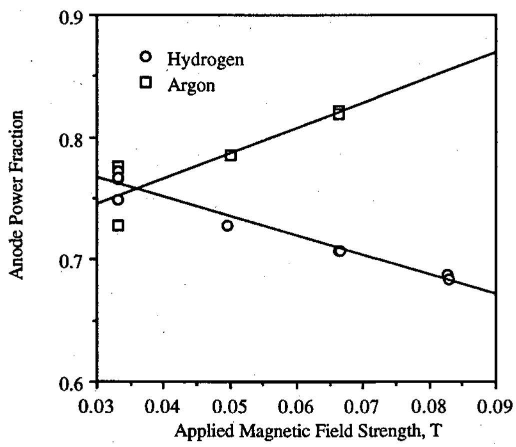

Fig. 5 Anode power fraction vs applied magnetic field strength for hydrogen and argon propellants.

fraction plots shown in Fig. 5. These results show for the first time a fundamental shift in thruster behavior for the two propellants. At low applied-field strengths the anode power fraction was approximately 75% for both propellants, but it increased to 83% with argon and decreased to 65% with hydrogen as the applied-field strength was increased. The dependence of anode power fraction on applied-field strength observed with hydrogen propellant is similar to that observed with argon at mass flow rates of 0.05 g/s and higher,[3,10] indicating that the behavior may be more a function of propellant number density than mass flow rate. This is consistent with studies correlating the anode power loss to the electron Hall parameter near the anode.[11] Because the thruster efficiency increased at approximately the same rate for both argon and hydrogen, the difference in anode loss behavior indicates that the conversion efficiency of power deposited into the plasma into thrust power (flow efficiency) was also different for the two propellants. The flow efficiency was calculated by dividing the measured total efficiency by the thermal efficiency η_{th} where the latter was calculated from

$$\eta_{th} = 1 - (P_{el}/P_{tot})$$

where P_{el}, the power deposited into the electrodes, was obtained from the measured anode and cathode power losses, and the total power P_{tot} was found by multiplying the discharge voltage and current.[4] At the lowest applied-field strength, the flow efficiency was 60% with hydrogen and only 28% with argon. However, at the highest field strengths these values were 67 and 64%, for the hydrogen and argon, respectively, showing that the flow efficiency increased much more rapidly with argon than it did with hydrogen as the applied-field strength was increased.

In addition to the differences in power conversion efficiencies, the results indicate a substantial difference in plasma state for the two propellants. Subtracting the thrust, anode, and cathode powers from the thruster input power yields the maximum power level available for propellant dissociation and ionization. This method overestimates the available power by neglecting losses in thermal energy, cathode radiation, plume divergence, etc. The maximum possible ionization fraction of the plasma leaving the acceleration region was then estimated by dividing this available power by the power required to fully ionize the propellant flow rate. For hydrogen the propellant was first assumed to be fully dissociated and then the remaining power was applied to ionization. Results for argon clearly showed that the propellant could easily be fully ionized, with the power available for ionization always being more than three times that required to fully ionize the flow. This result is consistent with spectroscopic studies indicating that argon was fully ionized in these thrusters.[12] However, for hydrogen there was never enough power available for full ionization, and at low applied-field strengths there was not even enough power to fully dissociate the flow. In fact, the maximum possible ionization fraction increased linearly with applied-field strength, reaching a maximum of 10% at an applied-field strength of 0.084 T. These results show that assuming the propellant is fully ionized, as is done in most current theoretical studies,[2] is incorrect for hydrogen.

Conclusions

An applied-field MPD thruster was operated on argon and hydrogen propellants at the same mass flow rates and operating conditions to establish the effects of propellant choice on thruster performance and power deposition. It was found that operating on hydrogen extended the stable operating range to higher values of applied magnetic field strength. Thruster discharge voltage, specific impulse, and efficiency were 10–20 V, 1000–1300 s, and 5–10% higher, respectively, with hydrogen than argon. The extended operating range with hydrogen resulted in a factor of 2 increase in peak efficiency with hydrogen, with a peak measured performance of 3700-s specific impulse at 20% efficiency. The behavior of thruster performance parameters as functions of applied-field strength were similar for the two propellants, suggesting that geometric scaling rules developed with argon may be applicable to hydrogen thrusters. However, the behavior of the anode power loss was different for the two propellants, and estimates of the maximum possible propellant ionization fraction indicated that while the argon propellant was fully ionized, the hydrogen was less than 10% ionized at the highest applied-field strength, and may not have been fully dissociated at the lowest applied-field strengths.

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