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A Comparison of Electric Propulsion Systems for Mars Exploration

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A COMPARISON OF ELECTRIC PROPULSION SYSTEMS FOR MARS EXPLORATION

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ABSTRACT

Earth-Mars trajectories for multiple solar-powered spacecraft configurations were generated using Hall and ion propulsion systems utilizing the Direct Trajectory Optimization Method. Payload and power trades versus trip time were examined. Performance was compared for purely interplanetary flight and interplanetary flight with estimated spiral in to Mars orbit. Evaluating current ion and Hall thruster technologies, similar payload masses were delivered by each at equivalent trip times, but with the Hall thruster operating at a power level 10 kilowatts, on average, less than the ion thruster. The power difference for equivalent payload delivered should result in a significant cost savings.

INTRODUCTION

NASA has identified Mars as an important interplanetary destination due to its proximity and interest to the space science community. Approximately every two years, the synodic period of the Earth-Mars system, the distance between the bodies reaches a minimum. Previous missions have capitalized on this dating back to the 1960s with the Mariner flyby missions and continued with the Viking landers in the 1970s and the Mars Pathfinder lander/rover in the 1990s. Currently, the Mars Global Surveyor and 2001 Mars Odyssey are orbiting the Red Planet. And just recently, the Mars Express and MER-A/B missions were launched. Planning continues for the next synodic period opportunity in 2005 with the Mars Reconnaissance Orbiter¹.

Solar electric propulsion (SEP), first flight-demonstrated in the 1960s^{2,3}, uses power provided by solar arrays to accelerate a working fluid that propels a spacecraft. Many studies have been performed to show the applicability and performance of SEP systems for interplanetary missions. The study of SEP main-belt asteroid^{4,5} and comet⁴ missions has shown benefit with both ion and Hall thruster propulsion. Other missions to planets and their moons have also been studied and shown feasible with SEP, such as Pluto flyby⁵ and Europa orbiter⁴ missions. SEP Mars missions have also been studied. SEP sample return missions to Mars' moon Phobos⁵ as well as orbiter and sample-return

missions directly to Mars^{4,6} have been studied and show the benefit from the application of SEP.

Each of these interplanetary missions analyzed required 20-100 kilowatts of on-board power at launch, which is substantial compared to the 2.5 kilowatts required by the recent Deep Space 1 (DS1) spacecraft⁷. However, DS1 was the first time NASA implemented an interplanetary spacecraft incorporating substantial on-board power specifically for primary propulsion applications. This paradigm shift represents a willingness of NASA to accept the additional costs and complexity of SEP systems to achieve substantial performance benefits relative to traditional chemically propelled spacecraft. In light of this paradigm shift, SEP missions to Mars using currently available technologies were evaluated. For these analyses ion and Hall thruster propulsion technologies, which have both been used operationally, were considered. These two propulsion technologies, which have been evaluated through development projects that are ongoing^{8,9}, have complimentary performance. State-of-the-art (SOA) ion thrusters can provide specific impulses (I_{sp}) ranging from 2500 to 4000 seconds with efficiencies between 50% and 70%. SOA Hall thrusters can provide 1500 to 3500 seconds of I_{sp} with efficiencies between 45% and 65%.

The goal of this study was to assess SOA ion and Hall system performance benefits for a Mars orbiter mission. Performance comparisons were made to assess each system's strengths. The missions analyzed

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all have similar baselines with different arrival schemes. The baseline mission launched on a Delta IV (5,4) launch vehicle to Earth-escape. The SEP interplanetary portion of the mission had a burn-coast-burn profile ("burn" refers to the electric propulsion system producing thrust). Arrival at Mars was analyzed with three different arrival schemes. First, the interplanetary trajectory was examined alone to simulate capture into a high Mars orbit, to study the Mars-Phobos-Deimos system for example, or as a communications relay satellite. This trajectory was similar to an aerobrake or aerocapture trajectory, but aerobraked or aerocaptured missions were beyond the scope of this analysis and the tools used. An SEP spiral to an altitude appropriate for the study of Mars' satellite, Deimos, and to low Mars orbit for planetary studies were also estimated for total trajectory performance comparisons. The impacts of using each different propulsion technology on mission capability are discussed.

SPACECRAFT SYSTEM AND MISSION DEFINITION

Prior to performing the mission analysis, the spacecraft and mission were defined. For these analyses, the spacecraft model was kept generic to assess results based on thruster performance and not spacecraft and system mass considerations. The spacecraft mass model accounted for mass changes resulting from the varying power level, both from the power system (array) mass and the changing number of thrusters required on the spacecraft. It also adjusted the mass of the payload structure and tankage based on the payload mass and propellant mass.

Spacecraft Definition

The spacecraft system model used was an updated version of the JPL Team X spacecraft from the baseline next generation ion design reference mission analyses⁴. The spacecraft was defined as having six components: fixed SEP stage, payload, payload structure, propellant, propellant tank, and power system. The fixed SEP stage accounted for thrusters, power processing units, feed systems, and any associated structure. The payload and propellant were determined by the optimized trajectory. Payload mass was determined by subtracting all other spacecraft components from the launch mass. Propellant was calculated by multiplying mass flow rate by thruster operation time. A fraction of payload and propellant mass defined the payload structure and propellant tank masses, respectively. Finally, the power system mass was calculated using a specific mass, that is, unit mass of power system per unit power required. Table 1 details the parameters

used to define the spacecraft. The total mass of the spacecraft was defined as follows

$$m_{wet} = m_{SEP} + m_{PL} + K_{st}m_{PL} + m_{prop} + K_t m_{prop} + \alpha P_0 \quad (1)$$

where m_{wet} is the spacecraft wet mass,
 m_{SEP} is the fixed SEP stage mass,
 m_{PL} is the payload mass,
 K_{ST} is the payload structure fraction,
 m_{prop} is the propellant mass,
 K_t is the tankage fraction,
 α is the power system specific mass, and
 P_0 is the initial spacecraft power.

Table 1: Spacecraft Defining Parameters

SEP Stage Mass (m_{SEP})	577 kg
Payload Structure Fraction (K_{st})	0.085
Tankage Fraction (K_t)	0.156
Power System Specific Mass (α)	9.2 kg/kW

Propulsion System

Ion Propulsion System

The ion propulsion system utilized a 1.5 to 6 kilowatt thruster that operated at specific impulses (I_{sp}) between approximately 3000 and 4000 seconds and efficiencies between approximately 55% and 70%. The performance of the thruster was tied to available power through a throttle table (see Figures 1 and 2). For the following analyses, only the curve defining the high I_{sp} versus power was used.

The power available also determined the number of thrusters operated at any given time. Because the thruster was power-constrained, thrusters were turned off as the power dropped with distance to the Sun squared and vice-versa as the available power increased. In instances where it was possible to run a higher number of thrusters at lower I_{sp} and efficiency or a lower number of thrusters at higher I_{sp} and efficiency,

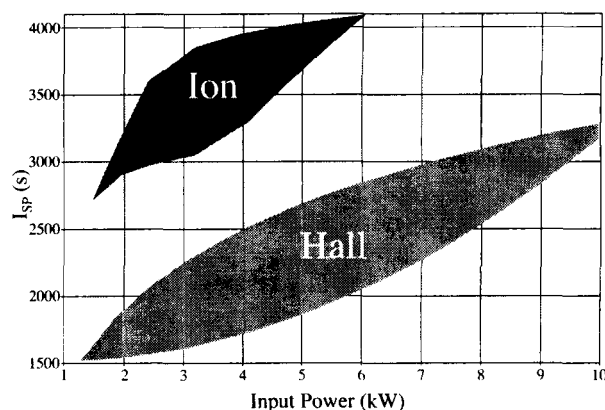


Figure 1: Specific Impulse vs. Power Throttle Table

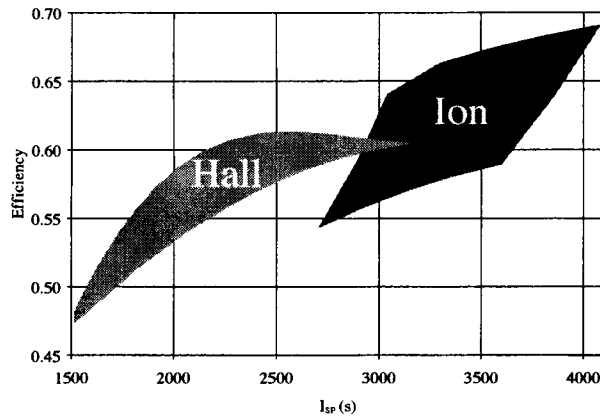


Figure 2: Efficiency vs. Specific Impulse

fewer thrusters were operated because of enhanced performance. A minimum of two thrusters was used during periods of thrusting to maintain roll control.

Hall Propulsion System

The Hall thruster system was modeled similar to the ion thruster. The throttle table ranged from 1.3 kW to 10 kW in power, approximately 1500 seconds to 3400 seconds in I_{sp} , and approximately 45% to 60% in efficiency (see Figures 1 and 2). Once again, only the high I_{sp} curve was used. Because the SOA power levels differ for these technologies, ion and Hall thrusters operating in the same power range would not have represented current capabilities. Thus, the Hall thruster's wider power range with higher maximum power compared to the ion thruster, required fewer thrusters, which reduced spacecraft mass and complexity.

Mission and Trajectory Optimization Method

Direct Trajectory Optimization Method

Trajectory design and optimization was completed using the Direct Trajectory Optimization Method (DTOM) code. As the name suggests, the DTOM is a direct method for obtaining optimal, low-thrust, interplanetary trajectories¹⁰. The DTOM numerically integrates the equations of motion using modified equinoctial orbital elements to accommodate circular orbits ($e = 0$)¹¹. The parameterized continuous-time control, thrust and coast lengths, launch date scaling factor, and Earth-escape parameters define the generic design space. More specialized problems can be defined with planetary gravity assists, loiter periods at the target body (used for sample-return missions), optimization of power level and specific impulse (either single value or parameterized continuous-time profile), and specialized thruster system models.

Thruster performance can be modeled in one of several ways depending on mission requirements. For single-point operation, a constant power, I_{sp} , and

efficiency can be specified. The second option relates specific impulse and efficiency (η) independent of power through the use of two performance constants, bb and dd , using the following relationship

$$\eta = \frac{bb I_{sp}^2}{dd^2 + I_{sp}^2} \quad (2)$$

The third method, and the technique used for these studies, is a throttle table relating thruster power processing unit (PPU) input power to specific impulse and efficiency (see Figures 1 and 2). Because a range of specific impulse exists at discrete power levels, one must choose between operating at the upper or lower boundary of I_{sp} , or at a point between determined by the optimizer. Once these three parameters are determined, the thrust magnitude, used in the equations of motion, was calculated using the following equation

$$T = \frac{2\eta P}{g I_{sp}} \quad (3)$$

where T is the thrust produced,
 P is the PPU input power, and
 g is the acceleration of gravity.

Mission Definition and Analysis Method

The mission used for these analyses was a direct Earth-Mars transfer launching in early 2014 with a thrust-coast-thrust profile. The spacecraft was launched to escape using a Delta IV (5,4) launch vehicle. A duty cycle of 90% was assumed to allow for navigation updates during thrusting. Parameters optimized by the DTOM were the parameterized continuous-time steering profile, thrusting and coasting period lengths, launch energy (C_3), departure angles from low Earth

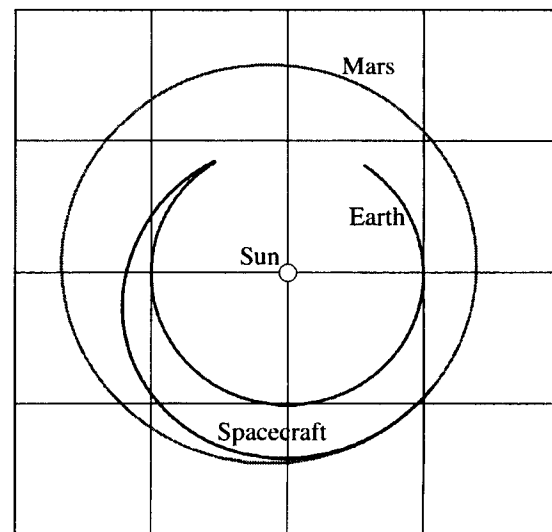


Figure 3: Sample Earth-Mars Trajectory

orbit, and PPU input power at Earth departure. A sample trajectory can be seen in Figure 3.

Fixed trip time trajectories were utilized to complete the analysis. The spacecraft payload was maximized for trip times ranging from 250 to 600 days. Sets of optimal trajectories for each propulsion option were generated and used to draw the conclusions found herein.

These analyses were used for comparison of the interplanetary travel. However, estimated spirals to a near-Deimos altitude of 23,520 km and 400 km altitude were included for additional performance comparisons. This estimation was completed using the method described by Battin in Reference 12. In this analytical formulation, the tangential thrust acceleration was assumed constant, but considering low-thrust spacecraft the decreasing mass must be accounted for in thrust acceleration calculations. Therefore, the average tangential thrust acceleration over the spiral was used in the calculation, and the spiral time (t_s) was approximated by

$$t_s = \frac{\sqrt{\frac{\mu}{r_0}}}{\bar{a}_{T_i}} \left[1 - (20)^{1/4} \left(\frac{r_0^2 \bar{a}_{T_i}}{\mu} \right)^{1/2} \right] \quad (4)$$

where μ is the gravitational parameter of the attracting body,
 r_0 is the initial circular orbit radius for escape or final circular orbit radius for capture, and
 \bar{a}_{T_i} is the average tangential thrust acceleration over the spiral trajectory.

The final mass after spiral was then calculated by multiplying the mass flow rate of the thruster by the spiral time. This final mass was then used to recalculate the final and average thrust accelerations and this process was iterated until convergence.

ELECTRIC PROPULSION SYSTEM PERFORMANCE

Analysis was completed through the generation of maximum payload trajectories over a range of trip times. Payload and initial power curves for each of the three arrival cases are presented in Figures 4 through 6. The ion and Hall thrusters consistently delivered roughly equivalent payload mass to Mars over the majority of the range of trip times. The only advantage was that the Hall thruster delivered more payload mass at the shortest trip times and the ion thruster delivered more mass at the longest trip times. These payload

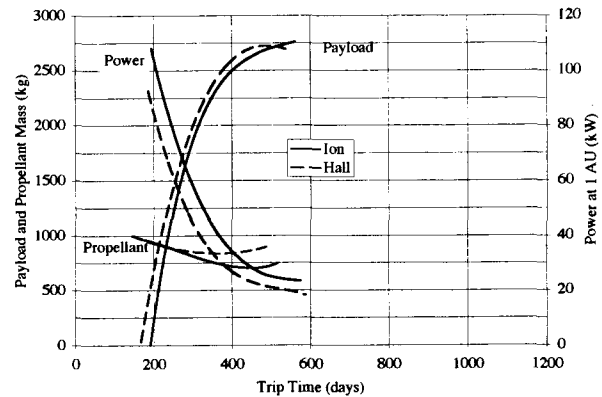


Figure 4: Payload, Power at 1 AU, and Propellant vs. Trip Time, Interplanetary Trajectory Only

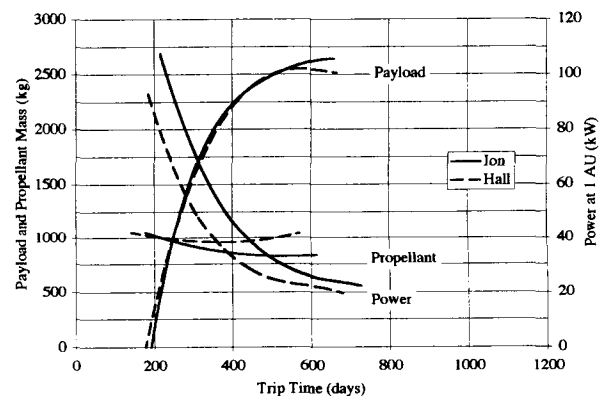


Figure 5: Payload, Power at 1 AU, and Propellant vs. Trip Time, Interplanetary Trajectory with Mars Spiral to Deimos

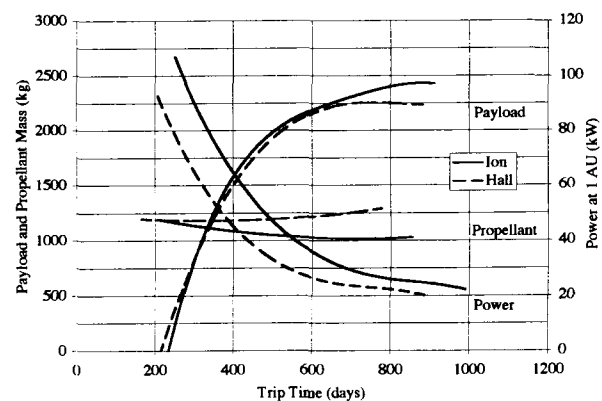


Figure 6: Payload, Power at 1 AU, and Propellant vs. Trip Time, Interplanetary Trajectory with Mars Spiral to 400 km Altitude

advantages were exaggerated as the spiral distance increased.

The relationship of propellant to I_{sp} was also evident. Because mass flow rate was considered inversely proportional to specific impulse squared, according to the following equation,

$$\dot{m} = \frac{T}{gI_{SP}} = \frac{2\eta P}{(gI_{SP})^2} \quad (5)$$

and the Hall thruster operated at a lower I_{SP} than the ion, it used more propellant (see Figures 4-6). This relationship held except for very short trip times where the power level required by the ion thruster at 1 AU (Astronomical Unit, the average distance from the Earth to the Sun) was high.

The factor that separated the performance of the ion and Hall thrusters dramatically was the power level required from the solar arrays at 1 AU (see Figures 4-6). For the strictly interplanetary trip times, the Hall thruster power level at 1 AU was an average of 10 kW less than the ion thruster for the three arrival schemes. The reduced size of the solar array required for the Hall thruster spacecraft could potentially save \$1 million per kilowatt.

The final relationship evident from these performance plots was the effect of the spiral on the performance of these systems. As expected, as the spiral distance increased, the total trip time also increased. Also, as the required power level decreased with increased trip times the spiral trip time increased. This relationship had the effect of stretching the curves toward longer trip times with increasing spiral distances.

Each of these propulsion systems operated in the high power range of their throttle tables (see Figure 7). A minimal amount of throttling was required by these propulsion systems to successfully complete this mission. The propulsion systems began by processing the highest power allowable by their respective throttle tables. As the power dropped with increasing distance to the Sun, thrusters were switched off and the high power end of the throttle table was utilized again.

Figure 7 illustrated the difference between the thrusters' operating points. This difference

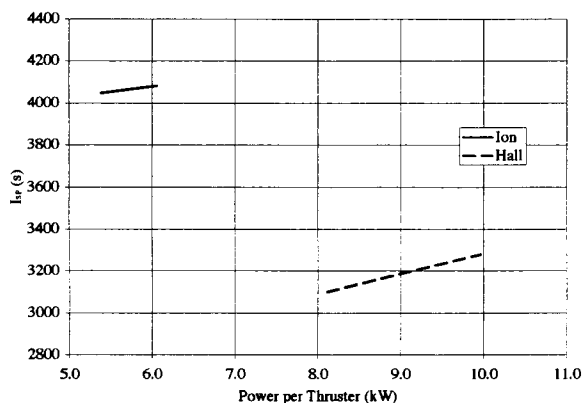


Figure 7: Specific Impulses Utilized by each Propulsion System for the 200-day Interplanetary Trip Time Case

demonstrated one reason why the Hall thruster outperformed the ion, especially at the short trip times. The Hall system's lower I_{SP} , which produced more thrust and reduced the required power and power system mass, and its ability to process more power per thruster than the ion system, which reduced the system mass, combined to allow the Hall system to deliver equivalent payload mass as the ion system but at lower power levels.

Examination of the launch energy (C_3) provided insight into the relative performance of these systems. Figure 8 showed the relationship of C_3 to the interplanetary trip time. At short trip times higher launch energies were used, at the cost of launch mass, to propel the spacecraft to higher velocities. The spacecraft implementing the Hall thruster system used higher C_3 at the shorter trip times to reduce the Δv (velocity change induced by the SEP system) required by the propulsion system. Because payload mass delivered was approximately equal over the range of trip times, the trade then became one between propellant mass and power system mass. The lower power level required by the Hall thruster system reduced the power system mass, which allowed an increase in propellant mass. Conversely, the ion thruster system's higher required power level increased the power system mass. The difference between these masses on the spacecraft was the fact that the power system was brought, as a whole, to the destination while the propellant mass decreased over the length of the trip. That is, a constant fraction of the total power was required to propel the power system to the destination while a decreasing fraction of the total power was required to propel the propellant load to the destination. Because the Hall thruster spacecraft had lower power levels, the ability to process more propellant, and delivered equivalent payload mass as the ion spacecraft, the Hall thruster's I_{SP} was more optimal for this mission than the ion thruster's I_{SP} .

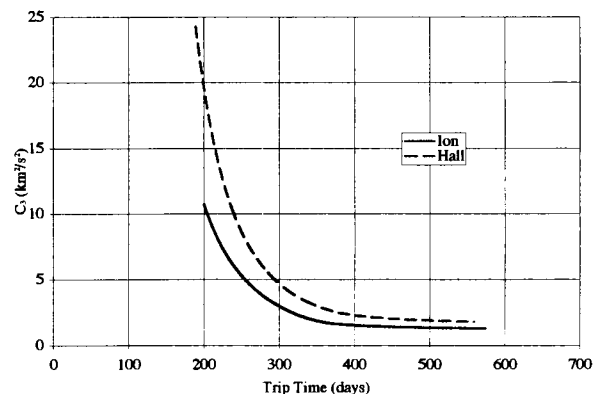


Figure 8: Launch Energies Required over the Range of Interplanetary Trip Times

IMPACTS AND ASSESSMENTS

Hall and ion thruster systems are appropriate propulsion systems for Mars exploration. Each of these systems have heritage through flight demonstrations and have technology development programs in progress. Hall thrusters typically do not have the lifetime of ion thrusters, but the trip times demonstrated for this Mars mission are of same order of magnitude of current Hall thruster lifetimes. Because of the Hall thruster's lower optimal power, it may be more appropriate than the ion for these missions, as long as the increased propellant loading for the Hall thruster is not prohibitive.

The development of higher power thrusters also is beneficial. Fewer thrusters operating at higher power reduces the required system mass such that performance is improved over a system with more thrusters operating at lower power levels. This can be seen in a general sense with this study. The Hall thruster's delivered payload mass is buoyed by the fact that fewer thrusters were required to operate at the higher power levels (shorter trip times) because it operated at up to 10 kW per thruster as opposed to the ion which operated at up to 6 kW per thruster. This power-to-mass relationship also supports the development of higher power thrusters, such as HiPEP (ion) and HIVHAC (Hall) funded through NASA Research Announcements.

The minimum trip time demonstrated by this study was approximately 200 days. However, at these short trip times, minimal payload was delivered to Mars. Chemically propelled spacecraft can complete the journey from Earth to Mars in slightly less than 200 days, when Earth and Mars are at their closest approach, and deliver a useful amount of payload mass. Therefore, a comparison which needs to be made, but was not included in this study, is the performance of a SEP spacecraft versus a comparable chemically propelled spacecraft.

These missions required significantly higher power levels for propulsion than have been demonstrated on DS1. As solar power systems for spacecraft are developed to higher power levels, the Hall thruster spacecraft Mars missions will be enabled before the ion thruster. In fact, for a required payload mass to be delivered to Mars, the Hall thruster will always require lower power levels than the ion thruster in this spacecraft configuration. Therefore, the Hall thruster can be considered a nearer-term technology for this Mars mission.

CONCLUSIONS

Maximum payload Earth-Mars trajectories were generated to assess the performance of SOA ion and Hall thrusters. The ion and Hall thrusters delivered approximately equivalent payload masses over a range of trip times. However, the Hall thruster required less power than the ion thruster. This power savings is substantial in terms of the cost of spacecraft systems. The lower powered Hall system used fewer thrusters, which is also a benefit due to the reduced complexity of the propulsion system, and also make it a nearer-term technology as solar power systems evolve to higher powers. Minimum transfer times for this mission were approximately 200 days and comparisons must be made to similarly configured chemically propelled spacecraft. The relative performance of these systems showed that the I_{sp} of the Hall thruster system was more optimal for this mission than that of the ion propulsion system. Inclusion of the spirals around Mars in the numerical optimization and allowing optimization of the I_{sp} profile within the bounds of the throttle table would improve upon these analyses. While the Hall thruster system may have outperformed the ion thruster for this Mars orbiter mission, by its nature, the ion thruster will outperform the Hall for other missions for which it is more suited. Both ion and Hall thruster systems have a place in the repertoire of interplanetary propulsion systems.

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