

Electric Propulsion Concepts Enabled by High Power Systems for Space Exploration

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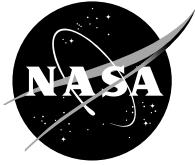
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This paper describes the latest development in electric propulsion systems being planned for the new Space Exploration initiative. Missions to the Moon and Mars will require these new thrusters to deliver the large quantities of supplies that would be needed to support permanent bases on other worlds. The new thrusters are also being used for unmanned exploration missions that will go to the far reaches of the solar system. This paper is intended to give the reader some insight into several electric propulsion concepts – their operating principles and capabilities, as well as an overview of some mission applications that would benefit from these propulsion systems, and their accompanying advanced power systems.

Nomenclature

A_g	ion thruster grid area
a	vehicle acceleration
\mathbf{B}	Magnetic Field strength
b, d	empirical terms related to the losses due to plasma production and acceleration
DC	Direct Current
EP	Electric Propulsion
g_o	gravitational constant
h	hours
I_{sp}	specific impulse
J	parameter characteristic of the mission difficulty
J_b	ion current flux through the grids
J	current through thruster
$\mathbf{J} \times \mathbf{B}$	Lorentz force arising from orthogonal currents and magnetic fields
J_θ	azimuthal thruster current
ΔV	change in velocity
\dot{m}	mass flow rate of propellant
M_i	vehicle initial mass
M_f	propellant mass
M_t	propulsion system mass
M_l	payload mass
MPD	magnetoplasmadynamic
NEP	Nuclear Electric Propulsion
P_e	power input to the thruster
T	thruster operating time
T_{MPD}	value of thrust from an MPD thruster

V_b	voltage which accelerates the ions
α	specific mass
μ	actual vehicle payload fraction, M_l/M_i
η	efficiency

I. Introduction

As space power research provides the technology to build future space vehicles with advanced nuclear systems, this enables electric propulsion devices that require kilowatts to megawatts of power. Larger diameter, higher I_{sp} ion thrusters and Hall thrusters are being built and tested in vacuum chambers jointly with advanced high power system concepts such as Brayton systems. High power electric propulsion devices are being built and demonstrated in the laboratory, such as Magnetoplasmadynamic (MPD) thrusters and Pulsed Inductive Thrusters (PIT).

This paper describes the latest developments in electric propulsion systems being planned for the new Space Exploration initiative. Missions to the Moon and Mars will require these new thrusters to deliver the large quantities of supplies that would be needed to support permanent bases on other worlds. The new thrusters are also being used for unmanned exploration missions that will go to the far reaches of the solar system. This paper is intended to give the reader some insight into several electric propulsion concepts – their operating principles and capabilities, as well as an overview of some mission applications that would benefit from these propulsion systems, and their accompanying advanced power systems.

II. Generic Thruster Figures of Merit

A vehicle propelled by electric propulsion will, of course follow the standard rocket equation:

$$\frac{M_f}{M_i} = e^{-\Delta V / g I_{sp}} \quad (1)$$

This holds true for any propulsion system, and demonstrates the benefit of high I_{sp} . I_{sp} is the ratio of thrust to weight flow rate of propellant at sea level, $I_{sp} = T/(g_0 \dot{m})$, and is proportional to the rocket exhaust velocity. Some refinements due to the nature of a power-limited system such as EP require some different forms of the above equation, which will be offered here without derivation. The first form of the low-thrust rocket equation is

$$\frac{1}{M_f} - \frac{1}{M_i} = \frac{1}{2\eta P_e} \int_0^T a(t)^2 dt = \frac{J}{2\eta P_e} \quad (2)$$

where “a” is the vehicle acceleration, and the “J” parameter is a characteristic of the mission difficulty, much like ΔV is for a high thrust mission. J is a mission invariant for low thrust systems, whereas ΔV often is not¹. Equation 2 further differs from a high thrust relationship in that the M_f term includes the propulsion system mass (M_t), which can be comparable to the payload mass (M_l). The propulsion system, including power and thrusters, is often parameterized by the specific mass, α , expressed in kg/kWe: $M_t = \alpha P_e$.

A third version of the low thrust rocket equation shows the relative role of all the power and propulsion system performance parameters in mission analysis²:

$$e^{-\Delta V / g_0 I_{sp}} = \frac{\mu + \frac{\alpha g_0^2 I_{sp}^2}{2\eta T}}{1 + \frac{\alpha g_0^2 I_{sp}^2}{2\eta T}} \quad (3)$$

Where μ is the actual vehicle payload fraction, M_l/M_i and T is the thruster operating time. Equation 3 can be used to assess the effects of various thruster performance parameters to determine optimal operating conditions. The result of such analysis is to show that there is an optimal specific impulse for a given mission that results in the maximum payload fraction (μ)³. The optimum I_{sp} is usually *not* the highest I_{sp} attainable. A detailed analysis would also include the variation of η with I_{sp} , as is experimentally observed in electric propulsion thrusters. System performance can be characterized in terms of the repeating parameter in Equation 3:

$$\frac{\alpha g_0^2 I_{sp}^2}{2\eta T} \quad (4)$$

While Eq. 3 is a complex, transcendental equation, the general trend is that Eq. 4 should be small compared to unity to increase μ . If the I_{sp} is significantly higher than the mission difficulty (ΔV), the exponential term becomes insensitive to I_{sp} , and the variation of Eq. 4 becomes the dominant effect. This translates to low α , high η , long T , and the minimum I_{sp} ($\gg \Delta V$) to accomplish the mission. Note that the α term is generally dominated by the power system mass, so that the thruster-specific term is then

$$\frac{I_{sp}^2}{2\eta T} \quad (5)$$

For nominal NEP system parameters of 10 kg/kWe, 5000 s I_{sp} , 0.6 η , and an operating time of 10,000 hours, Eq. 4 has a value of 0.12. For a ΔV of 20 km/s, the corresponding μ is ~ 0.63 .

Unspecified in this discussion, and beyond the scope of this discussion, are the power and lifetime requirements of the system and how they affect the mission design. In brief, Eq. 5 is related to a power per unit propellant mass, and the higher its value, the greater the power. Thus, the above relation can also be used to calculate power requirements for various missions.³ Power requirements have been reviewed above. Thruster lifetime plays a role in mission performance through the thruster operation time T . For $T >$ thruster life, multiple thrusters are required to perform a mission. This requires redundant thruster sets, and increases the mass of the thrust subsystem. With the above background into the effects of various thruster parameters on mission performance, the high power thruster options currently available can be described with their corresponding parameter ranges defined.

III. Thruster types

As will be discussed below, most missions requiring high power also tend to require higher I_{sp} , that is, values greater than 2000 seconds. This requirement focuses the available electric thrust mechanisms to either electrostatic or electromagnetic propulsion. Electrostatic thrusters use a DC applied electric field to accelerate ions to the desired velocities. Ion and Hall Effect thrusters use this mechanism. Electromagnetic thrusters use the Lorentz force arising from orthogonal currents and magnetic fields ($\mathbf{J} \times \mathbf{B}$) to accelerate neutral plasma. Magnetoplasmadynamic (MPD) thrusters and the Pulsed Inductive Thruster (PIT) are electromagnetic.

A. Ion

1. Physics

The ion thruster uses conducting grids to apply a kilovolt (kV) potential gradient to neutral plasma and thereby accelerate ions to high exhaust velocities. For a given voltage, the particle velocity depends on its atomic mass, with heavier ions reaching lower speeds. A schematic of the ion thruster is shown in Figure 1. Plasma is generated in the discharge chamber through electron bombardment. The discharge hollow cathode at the rear of the chamber generates the electrons. The discharge plasma is quasi-neutral – it has zero net electric charge. At the exit of the discharge is the first grid, called the screen grid, which is biased slightly negative to the plasma. This allows ions to drift between the grid holes, into the region between the screen grid and the accelerator grid (also called accel grid). The kV potential is applied between the screen and accel grids to drive the ions to high I_{sp} . Upon exiting the accel grid, the ion beam is neutralized external to the thruster by a neutralizer cathode.

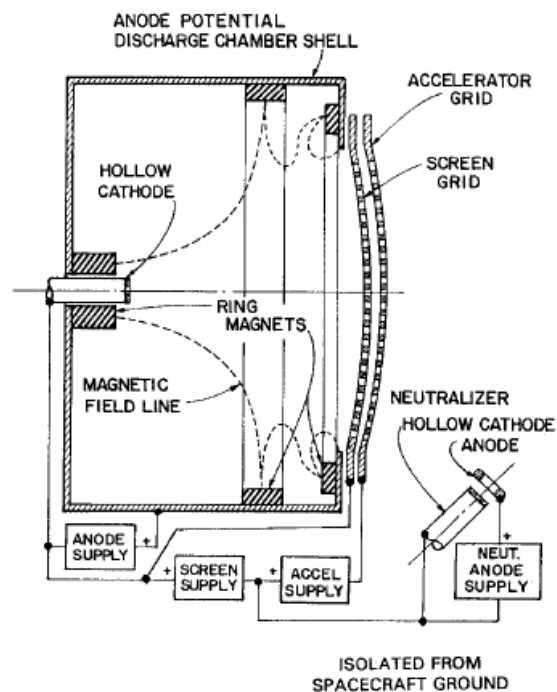


Figure 1.—Ring-Cusp Ion Thruster Schematic.⁴

The ion current through the grids is limited by several factors. First, the space-charge limited current density restricts the amount of current that can flow across a potential gap without shielding out the potential. Second, the grids are exposed to impacts from the high-speed ions, as well as from high-speed neutrals generated by charge-exchange collisions with neutral gas near the grids. Such impacts can sputter material from the grids, eroding them with time. To allow grid lifetimes comparable with mission operation times (~ 10 - $20,000$ hours, typically), the allowable erosion rates translate to a maximum current flux through the grids. In general, grid life, in particular accel grid life tends to be the limiting factor in determining current flux.⁵

2. Figures of Merit

Specific Impulse, I_{sp}

For state-of-the-art, low power missions, I_{sp} 's from 2000 to 4000 s are desirable. To achieve this level, Xenon is used as the propellant of choice because of its high atomic mass. These values are obtained with xenon propellant at accelerating voltage levels of 1 to 2 kV. Recent work for higher power robotic missions to the outer planets has focused on increasing operating voltage, and therefore I_{sp} , up to 6000 – 9000 s.⁶ An alternative approach to reaching higher I_{sp} values is to use krypton or argon as propellants – because they are lighter, the applied voltage for high I_{sp} is lower than for xenon. Depending on the choice of propellant, the voltages required would range from 2 up to 10 kV or more. The appropriate range of operation for the three propellants in terms of I_{sp} and efficiency are shown in Figure 2.

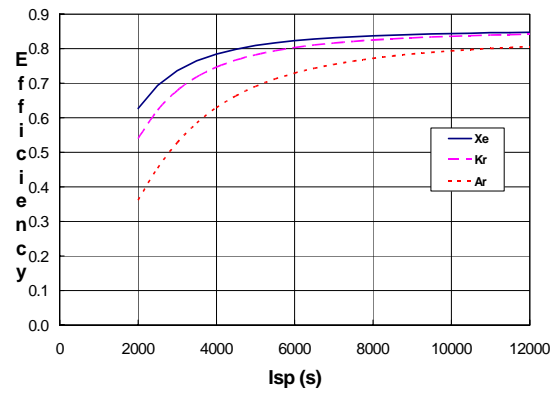


Figure 2.—Representative ion thruster efficiency, η , for xenon (Xe), krypton (Kr), and argon (Ar).

Thruster Efficiency, η

As can be seen from Figure 2, thruster η is quite high over a range of operating conditions. Values greater than 70% are obtainable with ion thrusters operating at high I_{sp} . Typically, the efficiency for ion thrusters can be described mathematically in the form:

$$\eta = \frac{b I_{sp}^2}{d^2 + I_{sp}^2} \quad (6)$$

Where b and d are empirical terms related to the losses due to plasma production and acceleration.⁷ The corresponding b and d parameters for the three propellants discussed are given in Table 1.

Table 1.—Efficiency coefficients for ion thrusters.³

Propellant	b	d (ks)
Xenon	0.86	1.21
Krypton	0.86	1.53
Argon	0.84	2.29

Lifetime

Ion thrusters have demonstrated lifetimes up to 30,000 h in laboratory life tests, and up to 8000 h in flight operation.⁸ As described previously, the thrusters were designed with current densities low enough to keep accel grid wear to acceptable limits over the thruster life. Additional life limiting components included the discharge chamber hollow cathode and neutralizer cathode. The discharge cathode, in particular, is subject to ion sputter damage. In one life test, the cathode cover eroded completely away, although the cathode continued operating. Recent thruster designs at the 20 - 40 kWe level are for 15,000 h.^{6,9}

Scaling to high power

The majority of power input to ion thrusters is deposited in the exhaust beam, so that $P_e \sim j_b A_g V_b$. j_b is the ion current flux through the grids, A_g is the grid area, and V_b is the voltage to which the ions are accelerated. V_b is determined by the ion species (atomic mass) and the I_{sp} . j_b is determined by the grid lifetime, propellant species, and voltage across the grids. To increase thruster power at fixed I_{sp} , therefore, essentially requires a larger area thruster. However, increasing the thruster area while maintaining a uniform small intergrid spacing presents structural challenges to designing a reliable thruster, particularly when vibration due to launch are considered.

An earlier study based on the space-charge limited current density, but neglecting accel grid wear, resulted in a 7800 s, 1.25 MW_e argon ion thruster requiring rectangular grids 1 m by 5 m⁷. More recent calculations based on accel grid lifetime could increase the required grid area for 1 MW_e to 15 square meters or more at comparable I_{sp} levels.¹⁰

B. Hall Effect Thrusters

1. Physics

Hall effect thrusters combine a static radial applied magnetic field with an axial electric field to operate as electrostatic accelerators without grids. A Hall schematic is shown in Figure 3. Concentric cylindrical magnet poles at the exit of the device impose the radial field. An anode at the rear of the device generates the plasma from which the ions are accelerated. An external cathode injects electrons down a voltage gradient between the anode and cathode. The electrons are trapped by the radial magnetic field in a circular Hall current; the magnetic field is designed so that ions are not trapped.

2. Figures of Merit

Specific Impulse, I_{sp}

Hall Thrusters have been primarily operated using Xenon (Xe) propellant in the 1000-3300 s range of I_{sp}. Recently, the NASA-457M thruster was operated on Krypton (Kr) up to 4500 s and at power levels up to 70 kW_e.¹¹ I_{sp}'s as high as 8000 s have been projected for a two-stage Hall thruster operating on bismuth; however this has not been experimentally verified.¹² For operation with Xenon propellant, the above I_{sp}'s were achieved at discharge voltages of 300-600 V. Increased I_{sp} ranges were obtained for voltages of 1 kV.

Thruster Efficiency, η

Hall thrusters operating on Xe have demonstrated discharge efficiencies up to 70% in the I_{sp} range given above. Operation with Kr yielded a maximum discharge efficiency of 64%. A projected efficiency value of 80% has been given for a bismuth propellant Hall thruster, but this has also not been experimentally verified.

Lifetime

Hall thruster lifetime is limited by erosion of the ceramic chamber walls. Lifetimes up to 7400 h have been demonstrated in low power devices¹³, based on present technology, lifetimes from 6000 – 8000 h might be expected in high power thrusters, depending on the propellant and I_{sp}'s required.¹⁴

Scaling to high power

Scaling Hall devices to higher powers will require increased thruster diameters and propellant and current throughput. If I_{sp} must also be increased, higher discharge voltages and applied fields may also be required. A chief engineering concern will be thermal management of the waste heat deposited into the anode, as well as thermal design of the channel insulators to allow long lived, high temperature operation.¹⁴

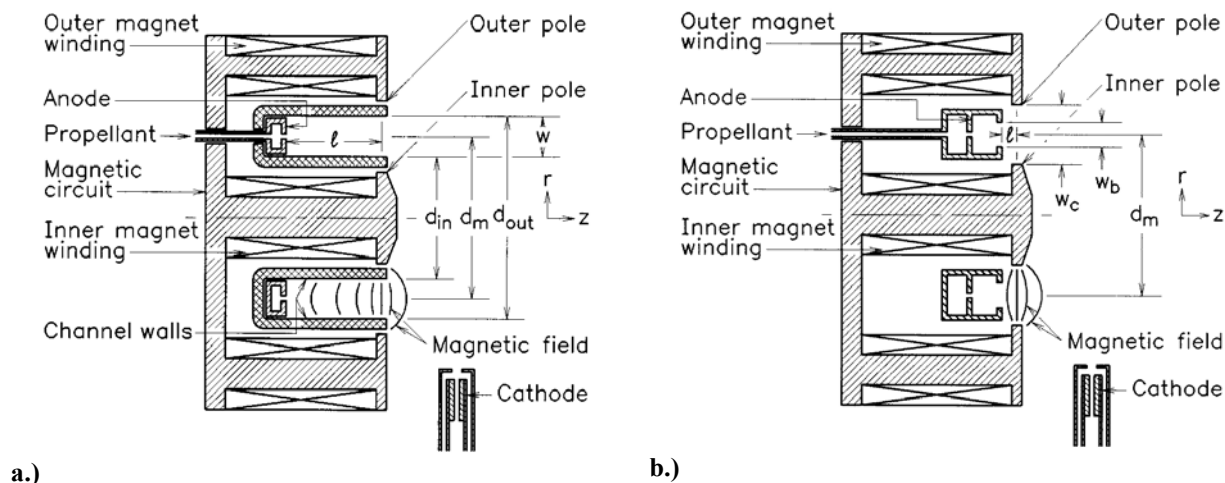


Figure 3.—Two hall thruster designs: a.) Stationary Plasma Thruster (SPT) b.) Anode Layer Thruster (TAL).

C. Magnetoplasmadynamic (MPD) Thrusters

1. Physics

MPD thrusters utilize the Lorentz force resulting from orthogonal currents and magnetic fields to accelerate neutrally charged plasma to high velocities. Because of this, they are sometimes also referred to as Lorentz Force Accelerators. The concepts currently under consideration are cylindrical, with an inner central cathode and an outer coaxial anode (Figure 4). The acceleration processes inherent in these devices can be separated into two classes. In the first, the magnetic fields in the thruster are generated by the flow of current across the inter-electrode gap and along the cathode. This form is called “Self-field” thrust. An external axial magnetic field can also be applied to the thruster. This is referred to as the “applied-field” thruster¹⁵.

In the self-field thruster, the thrust is dependent purely upon the current and thruster geometry, through the relation $T_{MPD} = b J^2$, where b is a geometric parameter and J is the current through the device. The applied-field thruster has been observed to augment the self-field thrust in some cases, although the mechanisms by which additional thrust is generated are not well understood.¹⁶ Because of the electromagnetic nature of these thrusters, they operate at high currents and low voltages. They have also demonstrated their most efficient performance at >MW power levels. At these levels the current is on the order of kAmperes and the voltages are 200-400 V.

Because of the dearth of MW level space power supplies, and the difficulties of steady state testing at MW levels (more accurately, at the g/s flow rates that are required at MW levels), MPD thruster development has been sporadic, with limited opportunities for uninterrupted research and optimization.¹⁷ A majority of the thruster testing has been using argon as a propellant, strictly from an availability basis. The two propellants identified as preferable for high performance operation are hydrogen or lithium.

2. Figures of Merit

Specific Impulse, I_{sp}

MPD thrusters have been operated with a variety of propellants from 1000 to over 10,000 s I_{sp} . A caveat to this range is that a limiting aspect of MPD thruster operation is the “onset” limit, a phenomenon which occurs at high ratios of J^2/\dot{m} . At onset, high amplitude, high frequency voltage oscillations occur, and they are accompanied by a marked increase in anode erosion. This limit can also effectively limit the maximum I_{sp} of the MPD thruster. Extension of this limit is an ongoing avenue of research. I_{sp} 's ranging from 2000-8000 s are possible with lithium¹⁸, hydrogen operation results in values from 5000 to over 10,000 s.¹⁹

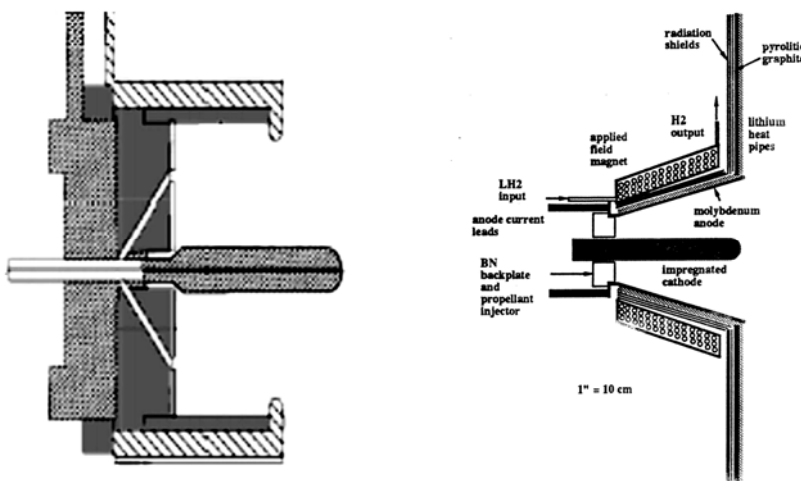


Figure 4.—Self- (left) and Applied-Field (right) MPD Thruster Schematics.

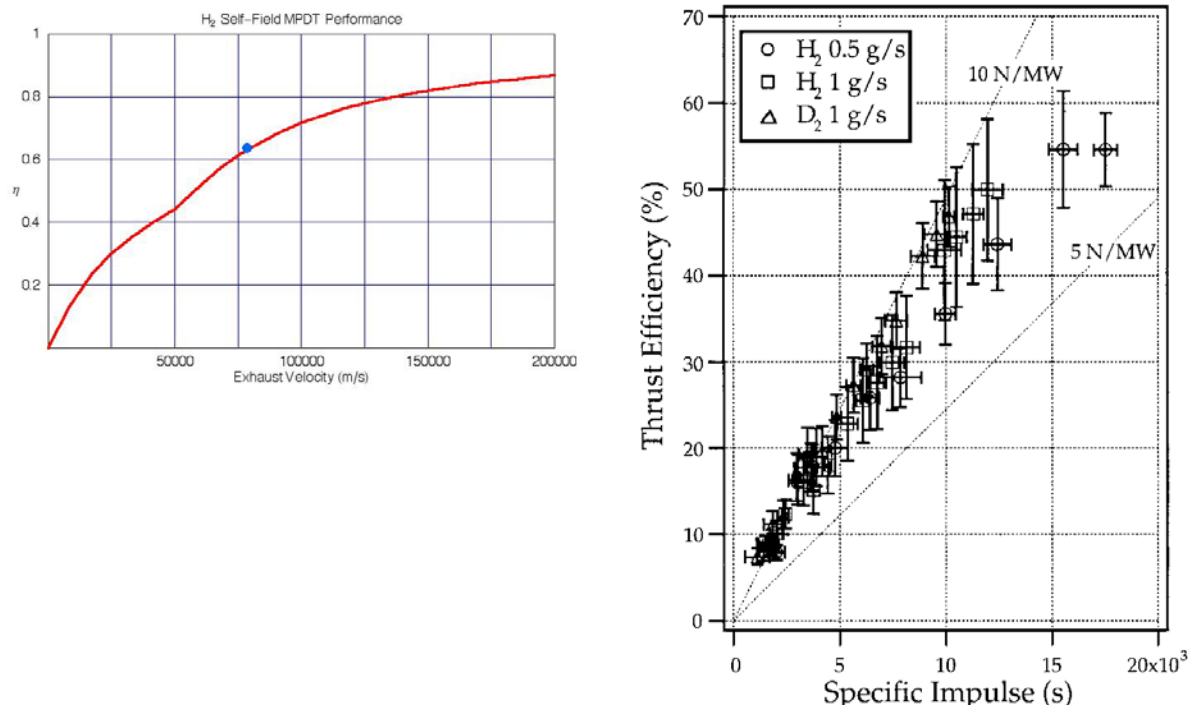


Figure 5.—Projected²⁰ and measured¹⁹ self-field MPD thruster performance using hydrogen.

Thruster Efficiency, η

Projected maximum thruster efficiencies range from 60% at over 8,000 s with hydrogen²⁰, and 65% at 8000s with lithium. Measured efficiencies thus far are 55% at 6000 s with lithium, and 50% at 15000 s with hydrogen.¹⁹ Example self-field projected and measured performances are shown in Figure 5. The projected increase in efficiency is assumed to come from optimized thruster geometry and mass flow distribution, but has not yet been experimentally proven²¹.

Lifetime

Both the extension and measurement of lifetime remain issues in MPD thruster operation. Operating below onset, MPD thruster lifetime is determined by the lifetime of the cathode. The cathode is exposed to high current densities and high velocity plasma. Cathode mass loss due to localized evaporation from small, high-current current attachment points distributed over the surface can be ameliorated by using low work function materials and operating at a temperature high enough to allow thermionic emission of the necessary current⁷. The cathode erosion is then caused by sputtering by the high velocity ions at its surface. Lifetimes of at least 5000 h are needed to make MPD thrusters relevant to missions of interest.

Scaling to high power

While the plasma physics of MPD thrusters favor high power, some engineering concerns remain. In the case of lithium thrusters, propellant management and distribution of a room temperature solid must be addressed, including propellant heaters, pumps, and valves. If applied field coils are used, either a normal conductor coil can be used with accompanying power losses and heat dissipation, or a superconducting coil with its attendant cooling and insulation requirements. For either thruster, a majority of the wasted heat is deposited in the anode, and must be rejected either through radiation or active cooling⁷.

D. PIT

1. Physics

The PIT is also an electromagnetic accelerator using the same Lorentz force as the MPD, but in different topology. A PIT schematic and test device are shown in Figure 6. The PIT consists of a flat multi-turn coil powered by a pulse-forming network. A mass injection system in front of the coil sends a mass pulse of propellant backward to cover the coil surface with a uniform gas layer. The pulse forming network discharges through the coil, producing a μs pulsed radial magnetic field at the surface of the coil. By Faraday's law, the changing magnetic field induces an azimuthal electric field in the gas layer, ionizing the gas to plasma and driving a current through the plasma. The interaction of $\mathbf{J}_\theta \times \mathbf{B}_r$ results in an axial force that ejects the plasma loop at high speeds.

Because the current is induced, the PIT is inherently a pulsed device. Charging voltages at performance levels of interest are in 5 – 15 kV range, and experimental pulse energies are on the order of kJ. Pulse lengths are typically 5-10 μs long, giving instantaneous power readings in the MWe. Experimental results to date have been in a single-shot mode, rather than in repetitive operation; the average power of the PIT in repetitive operation would depend on the duty cycle²². The optimal size of this thruster has been determined to be ~ 1 m. Test data described below are for this optimal diameter coil.

2. Figures of Merit

Specific Impulse, I_{sp}

The PIT has been operated on a wide variety of propellant types: argon, ammonia, hydrazine, and helium. In ammonia, I_{sp} 's from 500 s up to 8000 s have been measured. Ammonia is currently the propellant of choice, having demonstrated the highest performance for reasons that are not yet fully understood.^{23,24}

Thruster Efficiency, η

PIT efficiency in ammonia has been measured up to 55% at 5000 s. The variation in efficiency is relatively low (between 50 and 55%) over a range of I_{sp} from 4000 to 9000s, which is fairly unique in electric propulsion. Results for other propellants are generally below 50% for any I_{sp} . The highest efficiency was measured at the highest charging voltage of 16 kV.

Lifetime

Because the PIT has not been operated in a repetitive mode, lifetime is unknown. However, the fact that the device has no metallic electrodes exposed to high velocity plasma mitigates the life limiting material issue inherent in the steady state devices described previously. Conversely, the pulse forming and switching electronics which accompany this device will have to operate reliably for millions of pulses at instantaneous powers of MW, which is an outstanding challenge to this device. Experimental switching was done with spark gaps; this technology reintroduces electrodes to the system and reduces the lifetime advantage of the electrodeless coil. Solid state switching would be required for an operational device.

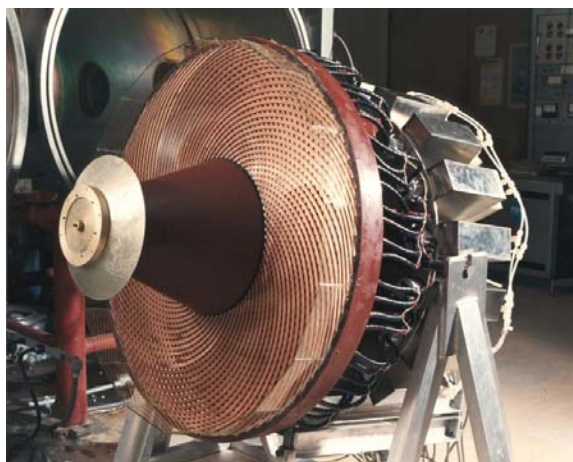
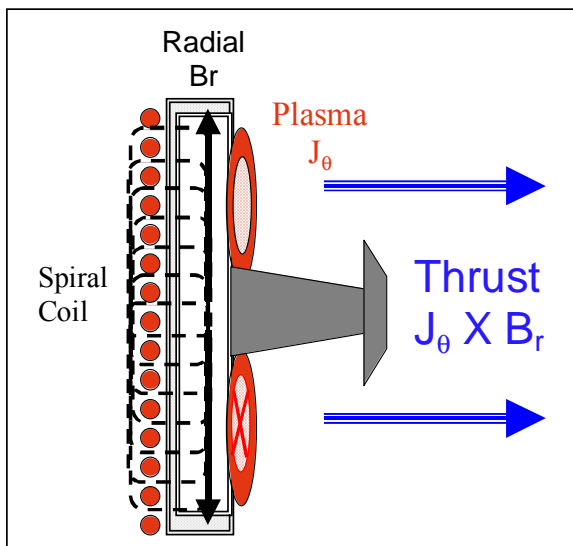


Figure 6.—Pulsed Inductive Thruster (PIT) operation schematic (left) and operational device (right)²². Device diameter is ~ 1 m.

Scaling to high power

In essence, scaling the PIT to MWe levels is a matter of high rep rate operation. The size of the thruster is optimized for single pulse performance, so no change in thruster dimension is required. The engineering issues that arise are the durability of the coil and shielding materials to the flexing and impact of the coil during the pulse, the exposure of the coil and shield to UV light from the energetic plasma, and heat management in the coil and pulse network for high rep-rate operation.

IV. Ancillary Issues for High Power EP

A. Thruster Testing

For any thruster capable of operation at MWe and thousands of seconds I_{sp} , testing at propellant mass flow rates on the order of g/s is required. This provides a stringent vacuum pumping requirement of Ml/s, which is available at only a few facilities in the world. Thruster life testing at MWe levels would require such a facility for thousands of hours to provide life data for flight qualification, at least initially. The facility requirements would, of course, depend on propellant choice. Condensable propellants such as lithium or bismuth could be easily cryopumped, but present other safety issues. Noble gases can and are cryopumped in Hall and Ion thruster tests¹². Requirements for a hydrogen facility are still under definition.

B. Power Processing

Each thruster concept would require power processing to convert the output power from the power system and transmission lines to a format useful for the thruster. Some of the generic requirements are summarized in **Table 2**. Regardless of the specific needs, such processing units at MWe levels will have to deal with thousands of volts or amperes reliably and efficiently. At a Power Processing Unit (PPU) efficiency of 0.92 and 1 MWe, the heat rejected is 80 kWe; more than the total spacecraft power available today. Rejection of this heat at low operating temperatures will drive up the PPU radiator mass and inhibit the acceleration of the vehicle, thus impacting mission performance. It is important that these technologies (which are beyond the scope of this paper) be addressed in concert with the thrusters.

Table 2.—Representative electrical operation requirements for high power thrusters and power processors.

	Ion	Hall	MPD	PIT
Power (kW)	30-1000	1 – 500	1000 –10000	30 - 10000
Voltage	2 – 20 kV	.2-1 kV	.1 - .5	Charging to kV
Current	10-100	100-1000	1000 – 10000	1000 (peak pulse)
AC/DC	DC	DC	DC	AC (pulsed)

V. High Power Electric Propulsion Missions

A. Robotic Missions

Missions require increasing power levels, to maintain reasonable trip times, as mission difficulty and payload delivery requirements increase. Essentially, a higher thrust, high efficiency power and propulsion system is required. Higher thrust can be obtained in one of two ways, higher power or lower I_{sp} . Lower I_{sp} will provide more thrust at the same power level, however more propellant will be required, thus the system is less efficient. Also, the I_{sp} has an optimal value for each mission, and cannot be relied upon to provide the increasing thrust requirement.

Therefore, increased power levels are required for spacecraft to achieve each incremental mission step.

Figure 7.—Mission Performance for Varying Power Level, Δv , and I_{sp} .

Figure 7 shows the relationship between power for propulsion, I_{sp} , Δv , and thrust time. Missions are laid out by means of constant Δv lines. The Δv parameter represents the energy change associated with a particular mission, and is relatively constant for a particular class of spacecraft using similar propulsion systems. Associating the Δv values with missions, one could assume that a Lunar mission would have a Δv of approximately 8 km/s, a 20 km/s Δv mission could be a Mars or comet sample-return mission, a 40 km/s Δv mission could be a Jupiter Grand Tour type mission, and a 60 km/s Δv mission could be a Pluto/Kuiper Belt Explorer or Interstellar Probe mission.

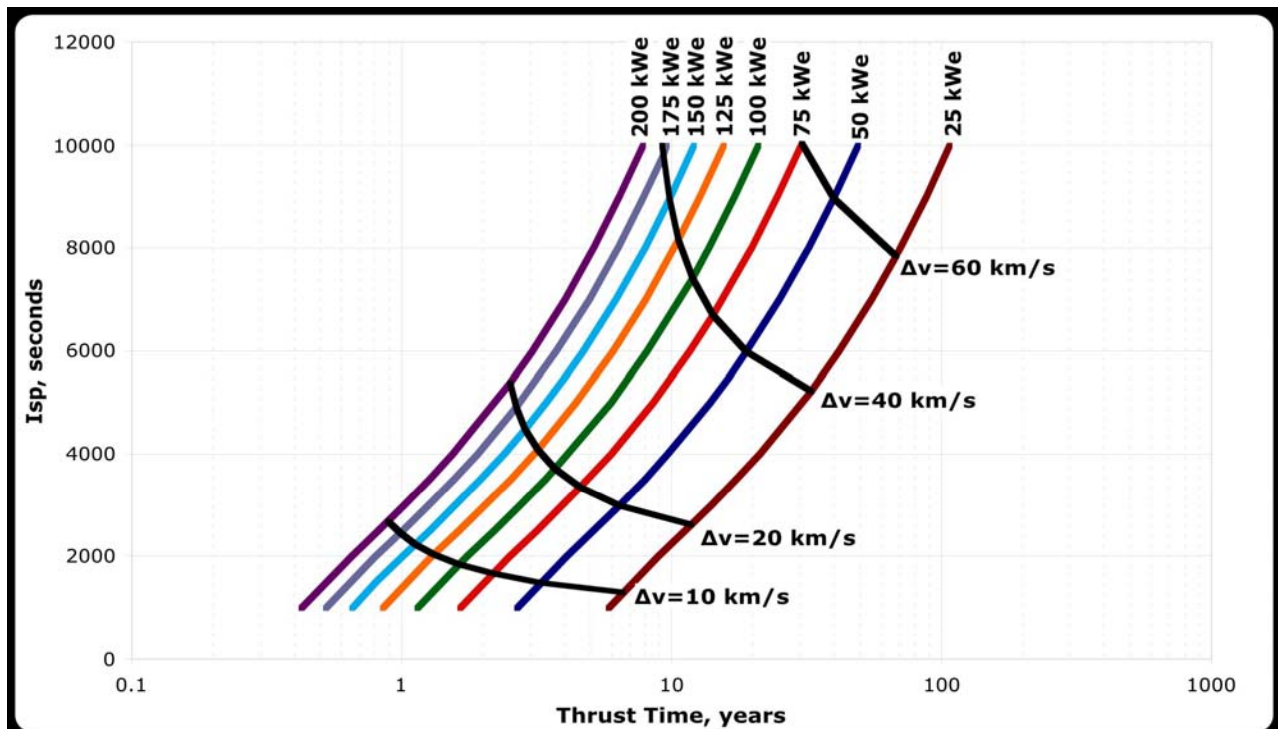


Figure 7.—Mission Performance for Varying Power Level, Δv , and I_{sp} .

Electric propulsion (EP) mission performance is highly dependent on both the specific impulse (I_{sp}) of the EP system as well as the power available for thrust^{25,26}. Figure 7 assumes a Delta IV Heavy launch to a circular orbit with an altitude of 1000 km. The baseline case is the 100 kW case at which the spacecraft has a dry mass of 15,000 kg, which includes a 1500 kg payload. The power system specific mass at 100 kW is assumed to be approximately 55 kg/kW.²⁷ The specific mass is scaled to different power levels via an exponential relationship to take into account the economy of scale, that is, higher power levels obtain higher power system specific mass (~100 kg/kW at 25 kW to ~40 kg/kW at 200 kW).²⁷

The specific impulse values shown, in Figure 7, for a given power level and Δv can be considered to be the thrust time optimal value of I_{sp} . Increasing or decreasing the I_{sp} at the same Δv and power level will lengthen the thrust time required for the mission. As I_{sp} increases, the acceleration of the spacecraft decreases, and the longer thrust time cannot be recovered by the reduction in propellant causing higher acceleration. Conversely, as I_{sp} decreases, the acceleration increases, but the increase in propellant mass still causes a lower acceleration than at the optimal I_{sp} . By examining Figure 7, one can see that for a particular mission a reduction in thrust time can be achieved by increasing the I_{sp} and power. However, the benefits of increasing power are diminished, for these cases, as the power level increases past 100 kW. Note that changing the baseline assumptions, the launch vehicle or dry mass, will vary the relationships shown in Figure 7.

B. Lunar Cargo Missions

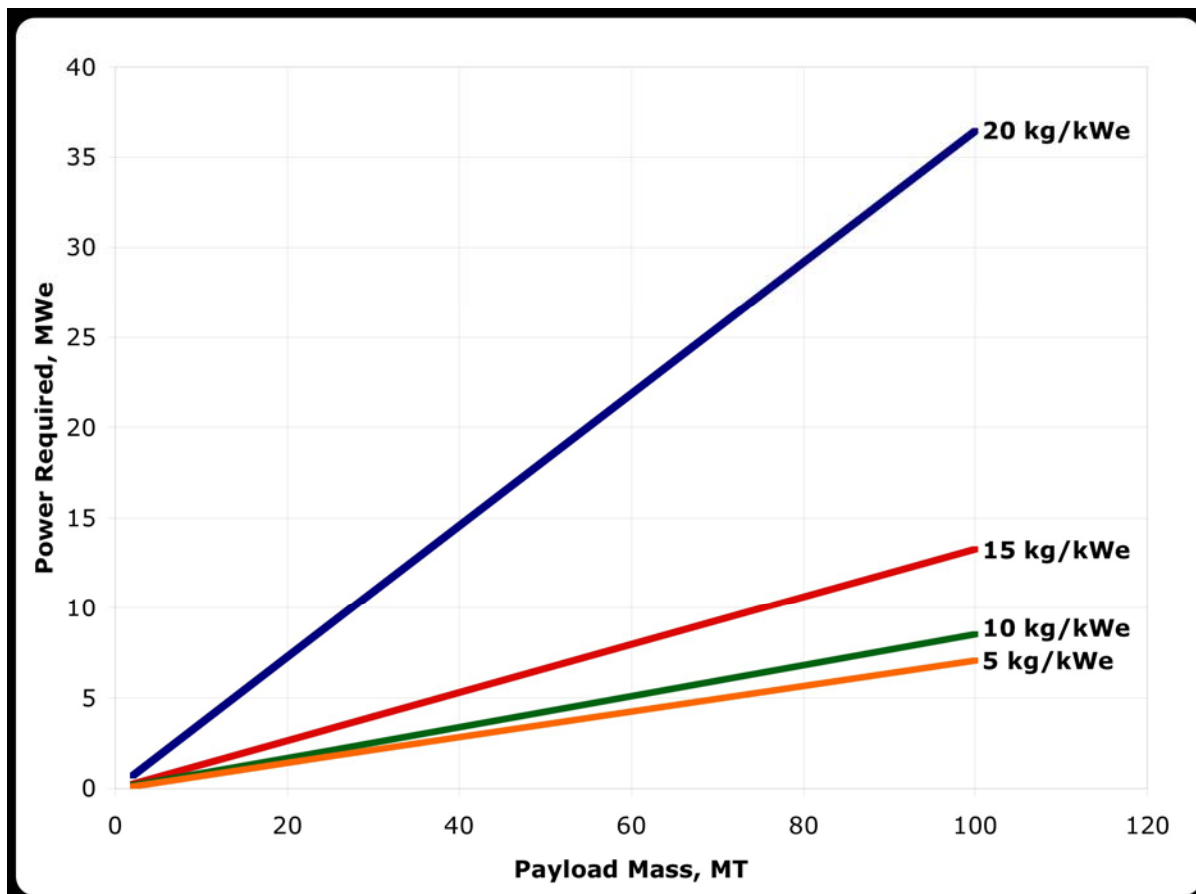


Figure 8.—Optimal Power Levels for Lunar Cargo Missions with Varying Payload Masses at Different Power and Propulsion System Specific Masses, 65 days of Thrusting Time.

Lunar cargo missions will be required in support of the manned lunar missions spelled out in “The Vision for Space Exploration.”²⁸ To efficiently deliver large amounts of cargo, high power EP systems may be used. In order to deliver the most cargo mass per unit of power, these power and propulsion systems will need to have the minimum specific mass possible.

Figure 8 shows the power requirements for different payload masses and power and propulsion system specific masses for a lunar mission requiring 65 days of thrusting to reach low lunar orbit. I_{sp} 's for these specific masses ranged from 4800 seconds at 5 kg/kWe to 2800 seconds at 20 kg/kWe. As expected, higher power levels are required to deliver more payload mass to the Moon. However, this increase in power can be reduced with advanced power and propulsion systems that provide lower specific mass.

C. Mars Piloted Missions

High power EP is also applicable to piloted missions to Mars. The baseline architecture assumption for this piloted Mars analysis is that the payload includes a 33.5 MT crew habitat and a 4.5 MT return capsule. A 2022 launch date is targeted with only the outbound, Earth-Mars, leg analyzed. The Earth-departure assumption is that

the spacecraft is launched to low-Earth orbit where it is assembled (if necessary) and then is flown autonomously towards Earth escape. Approximately 90% of the way to Earth escape, the crew rendezvous occurs with the EP spacecraft, via another launch, where the piloted mission begins.

Two configurations, all-EP and EP with aerocapture, were analyzed at 10 and 20 kg/kWe power and propulsion specific masses (see Fig. 9). For the shortest piloted trip times, which are essential for piloted missions, high power levels are required. Depending on the propulsion system configuration and power and propulsion system specific mass, the power required for propulsion for a 150-day one-way transit ranges between 2 and 7 MWe. The I_{sp} 's for a 150-day one-way transit range between 2000 and 4000 seconds depending on configuration and power and propulsion system specific mass.

VI. Conclusion

As shown in Figure 10, electric propulsion offers many benefits when compared with chemical propulsion for the applications displayed in the chart. A wide range of electric propulsion options is being developed at NASA Glenn for use in all future NASA missions. The selection of a propulsion (and power) system depends on the mission requirements. In order for the propulsion system to take full advantage of the development of higher power capabilities on spacecraft, higher power electric propulsions systems are ideal. Their greater fuel efficiency and compatibility with nuclear and solar-power systems are great assets for mission designers. The Lunar/Mars Exploration Initiative will benefit from the development of many of these electric propulsion concepts.

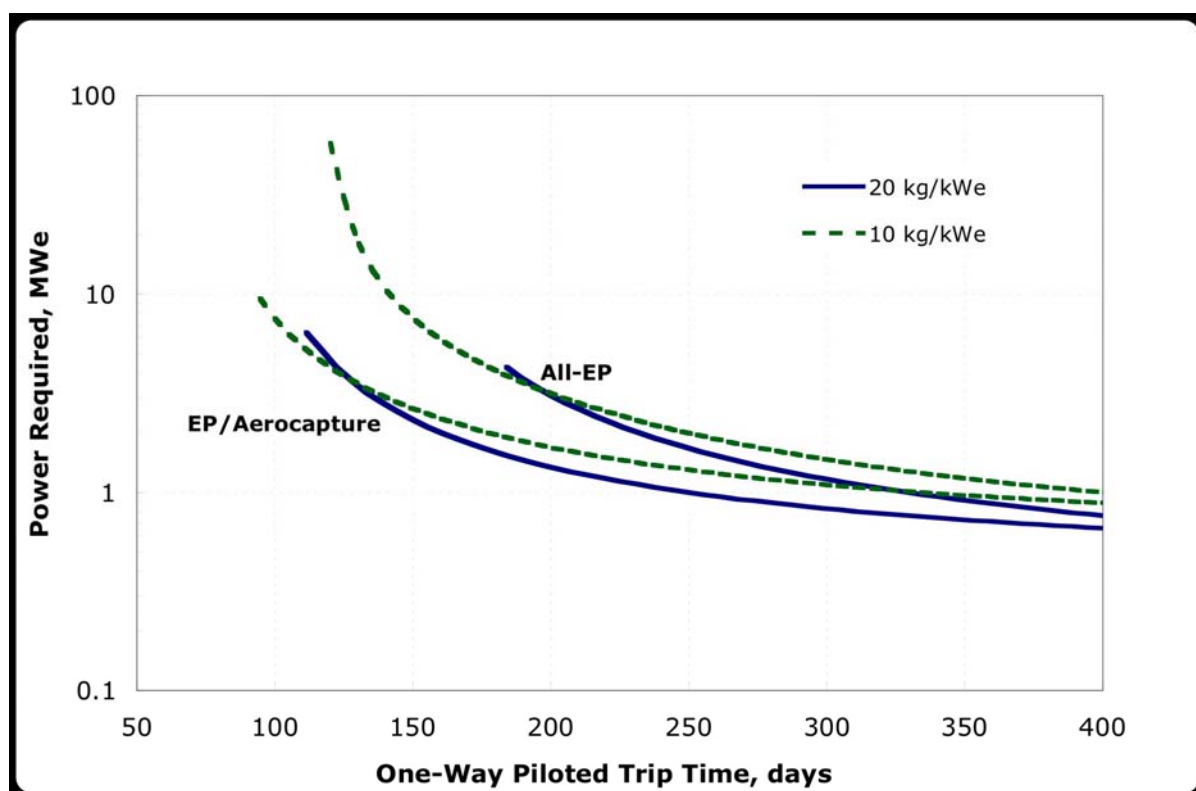


Figure 9.—One-Way Piloted Mars Mission Performance for NEP and NEP/Aerocapture Systems at Different Power and Propulsion System Specific Masses.

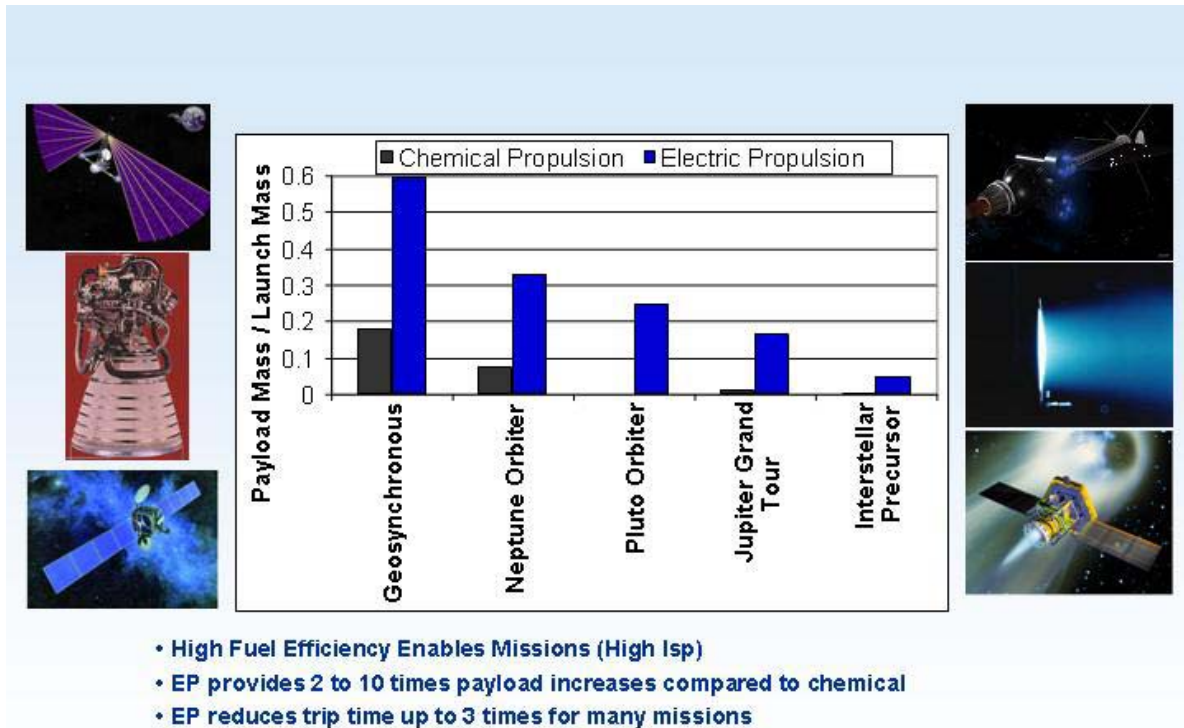


Figure 10.—Electric Propulsion Benefit for Various Missions

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13. ABSTRACT (Maximum 200 words) This paper describes the latest developments in electric propulsion systems being planned for the new Space Exploration initiative. Missions to the Moon and Mars will require these new thrusters to deliver the large quantities of supplies that would be needed to support permanent bases on other worlds. The new thrusters are also being used for unmanned exploration missions that will go to the far reaches of the solar system. This paper is intended to give the reader some insight into several electric propulsion concepts - their operating principles and capabilities, as well as an overview of some mission applications that would benefit from these propulsion systems, and their accompanying advanced power systems.				
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