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Low-thrust Propulsion Technologies, Mission Design, and Application

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1. Introduction

Electric propulsion has been widely accepted for station-keeping and final orbit insertion of commercial satellites. NASA, JAXA, and ESA have all used primary electric propulsion systems for science missions. Electric propulsion systems have been recently developed with a significant increase in performance and ability to process large amounts of onboard solar power. While the use of electric propulsion offers significant performance gains, it is not appropriate for all missions, has limitations, and the trajectories have characteristics that may be counterintuitive to those unfamiliar with low-thrust trajectory design. This chapter describes recent U.S. technology investments in electric propulsion thrusters with emphasis on mission application and low-thrust mission design for interplanetary trajectories and geosynchronous transfer using primary electric propulsion.

2. Overview of electric propulsion technologies

Unlike chemical propulsion, which is limited to the energy available through the decomposition or combustion of molecular compounds, electric propulsion makes use of energy from an external source, typically solar power, to electrically accelerate the propellant to higher energies. The efficiency of momentum transfer is often described in terms of specific impulse which is proportional to the average exhaust velocity in the thrust direction.

$$I_{sp} = \frac{v_{Exhaust}}{g} \quad (1)$$

The three basic types of electric propulsion systems are electrothermal, electrostatic, and electromagnetic. The types are categorized by the method of accelerating the propellant. Resistojets, arcjets, pulsed plasma, gridded-ion and Hall thrusters have significant flight experience. Electrothermal thrusters are the most widely used electric propulsion systems to date, but electrostatic systems are the industry's state-of-the-art (SOA) with higher specific impulses. The electrostatic thruster successes are made possible through technology advancements for increased power processing capability and increased thruster life driven by an increase in spacecraft available power.

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2.1 Electrothermal propulsion

In electrothermal propulsion thrusters, electrical energy is applied to heat a working fluid to increase the exhaust velocity. Resistojets are a form of electrothermal propulsion that operate by passing a gaseous propellant though an electric heater and then expanding it through a conventional converging diverging nozzle to create thrust. The typical flight operation is superheating catalytically decomposed hydrazine to leverage the propellant commonality of standard monopropellant chemical propulsion systems. The specific impulse of resistojets is limited by the high molecular mass of hydrazine and the maximum sustainable temperature. Specific impulse values near 350s is achievable; 40 percent higher than the conventional chemical equivalent. Arcjets are another form of electrothermal propulsion that passes propellants through an electric arc that heats the gas before it expands through a nozzle. Arcjet specific impulses are typically in the 500–600s range. Higher specific impulses are achieved because the maximum temperatures are not in contact with engine component walls, though efficiencies are less than that of resistojets.

2.2 Electromagnetic propulsion

Electromagnetic propulsion devices leverage magnetic fields, self-field or applied, to accelerate plasma, typically with a Lorentz force ($\mathbf{J} \times \mathbf{B}$) where the accelerating force is proportional to the cross product of the electric current density and the magnetic field.

The pulsed plasma thruster (PPT) is a form of electromagnetic propulsion that uses a capacitor to store electrical energy and when triggered creates a pulsed arc discharge across the face of a block of propellant, typically polytetrafluoroethylene (e.g. Teflon). This arc ablates and ionizes a small amount of propellant and the self-induced magnetic field acts on the ions to create a Lorentz force accelerating the plasma. The use of PPTs for East-West station keeping began in 1968 on the Lincoln Laboratory LES-6 satellite.

Pulsed inductive thrusters (PIT) and magnetoplasmadynamic (MPD) thrusters are additional forms of electromagnetic propulsion. The majority of these concepts are proposed for high power levels, >100 kWe, and have not gained any flight experience.

2.3 Electrostatic propulsion

While field emission electric propulsion (FEEP) and colloid thrusters fall in the electrostatic category, they are typically perceived as very low thrust devices lending themselves for disturbance force cancellation or precision control. For this reason, they are not candidates for primary propulsion. Large arrays are under investigation for primary propulsion.

Gridded-ion and Hall thrusters are the leading concepts for primary electric propulsion. Ion thrusters can achieve very high exit velocities, and have typical specific impulses in the 3000–4000s range. The first U.S. flight test with an ion engine was in 1964 with the Space Electric Rocket Test (SERT) I. Gridded-ion engines are in routine operation since the 1990s for geostationary north-south station keeping. In 1998, Deep Space 1 (DS1) was the first U.S. demonstration of primary electric propulsion. In September of 2007, NASA launched its first science mission, Dawn, using an ion engine for primary electric propulsion.

Gridded-ion thrusters operate by injecting a neutral gas in a thrust chamber. The gas is then ionized and magnetically contained within the chamber. The positively-charged ions migrate between a set of grids where the ions experience a large voltage potential. The ions are accelerated by a Coulomb force to high exhaust velocity, typically 30,000–40,000 m/s. The electrons inside the thruster chamber are then pumped by the system's power processing unit to a neutralizing cathode to maintain a zero net charge in the plume. An operational schematic of a gridded-ion engine is shown in figure 1.

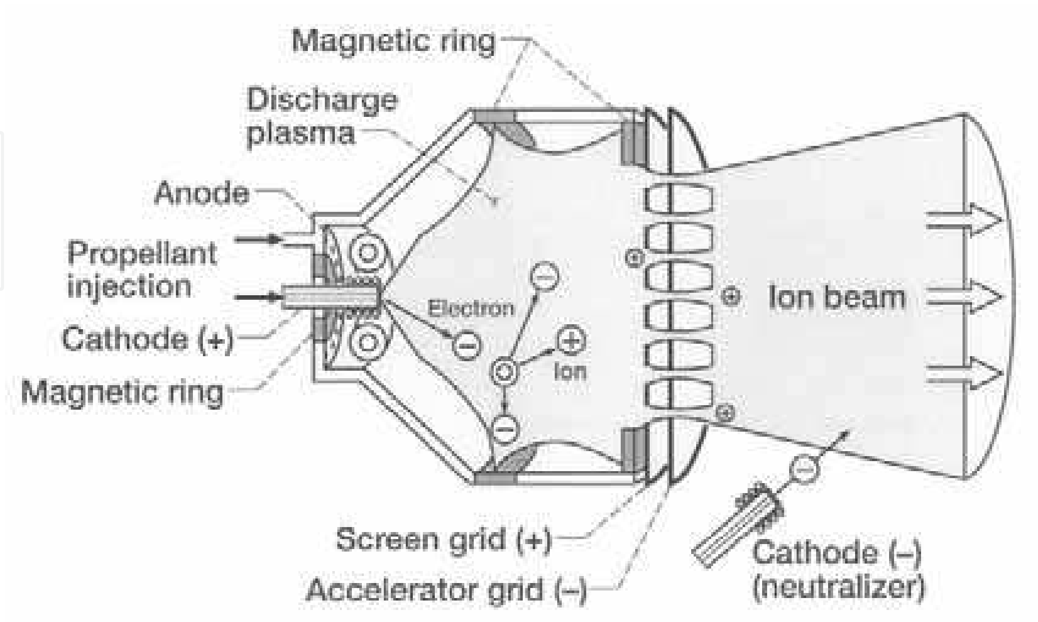


Fig. 1. Operational schematic of a gridded-ion engine.

A Hall thruster is essentially a grid-less ion engine. The thruster operates by employing magnetic fields to deflect low-mass electrons so that they are trapped under the influence of an $\mathbf{E} \times \mathbf{B}$ azimuthal field. The electrons are forced into an orbiting motion by the Hall effect near the exit plane of the thruster. A propellant is injected through the anode where the trapped electrons will collide and ionize the propellant. The ionized propellant will see the potential of the electron plasma and accelerate towards the thruster exit. Hall thruster exhaust velocities are typically 15,000–25,000 m/s. An operational schematic of a Hall thruster is shown in figure 2.

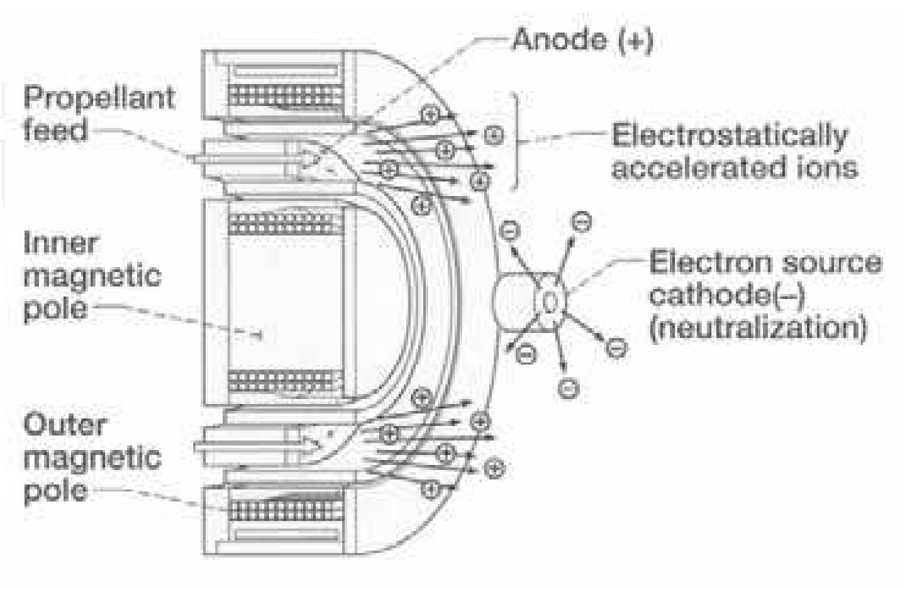
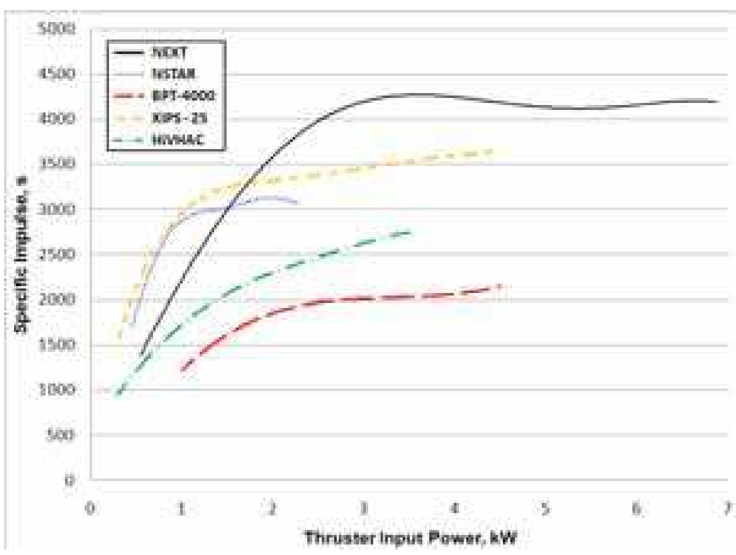


Fig. 2. Operational schematic of a Hall thruster.

3. U.S. advancement in electrostatic propulsion technologies

The state-of-the-art in electric propulsion thrusters suitable for primary electric propulsion is the L-3 25-cm Xenon Ion Propulsion System (XIPS), NASA's Solar Electric Propulsion Technology Application Readiness (NSTAR) thruster, and the Aerojet BPT-4000 Hall thruster. These three systems are all fully qualified with flight experience on the L-3 XIPS and NSTAR thrusters, and Aerojet's BPT-4000 is scheduled to launch in 2010. NASA is also completing the prototype development of NASA's Evolutionary Xenon Thruster (NEXT), and is completing the engineering model development of the High Voltage Hall Accelerator (HiVHAC) with Aerojet. While there are other great advancements in U.S. electrostatic thrusters, such as the BHT-200, it is not considered a candidate for primary electric propulsion and is not discussed. Representative thrust and specific impulse throttle tables of the L-3 25-cm XIPS, Aerojet BPT-4000, NSTAR and NEXT are shown in figure 3. Greater detailed performance data is available in open literature. The thrusters can operate well outside the throttle tables shown with varying thruster efficiencies.



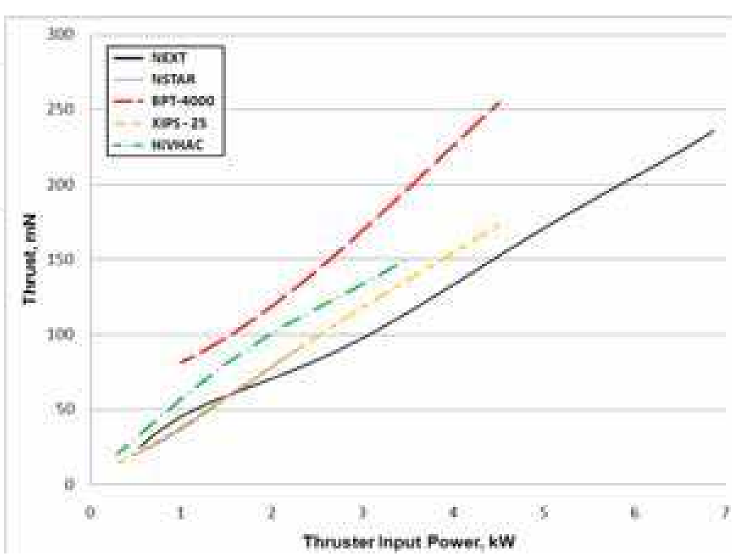


Fig. 3. Representative throttle tables for multiple commercial and NASA EP thrusters.

3.1 Commercial thrusters

Two commercially available thrusters are of high interest for primary electric propulsion; the L-3 XIPS and the Aerojet BPT-4000. Both commercial thrusters have a maximum operating input power near 4.5kW and primarily designed to operate over a few operating points, but have demonstrated large throttleability. The XIPS 25-cm and BPT-4000 thrusters are shown operating in figure 4.

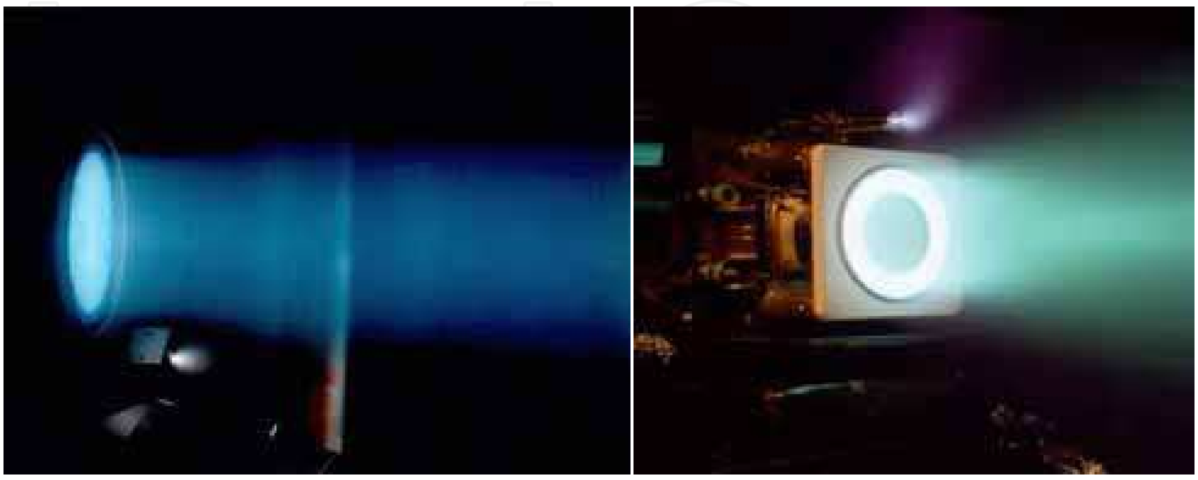


Fig. 4. The XIPS-25 firing (left) and the BPT-4000 thruster firing(right).

The L-3 25-cm XIPS thruster is used on the Boeing 702 communcation satellite for attitude control, north-south and east-west station keeping, momentum dumping, de-orbit, and augmenting orbit transfer (Tighe et al., 2006). The flight system operates in modes, 2.2 kW for attidue control and 4.4 kW for orbit transfer.

The 25-cm XIPS qualification testing demonstrated over 2,500 hours of operation at full power and over 13,000 hours at the lower power setting. While erosion and throughput capability is a function of the operating conditions of the engine, the 25-cm XIPS was projected to have similar life capability as the NSTAR thruster. Additional testing over a large throttle range was demonstrated (Goebel et al., 2006).

Aerojet completed qualification of the BPT-4000 Hall thruster in 2006 (Wilson & Smith, 2006). The BPT-4000 is a 4.5 kW multi-mode Hall thruster developed by Aerojet and Lockheed Martin Space Systems Company as part of a Hall Ion Propulsion System (IPS) for use on geosynchronous satellites. The thruster is designed to operate between 3 kW and 4.5 kW at discharge voltages between 300 volts and 400 volts. The thruster operates at lower voltage for orbit raising maneuvers and higher voltage to provide a higher specific impulse during station keeping.

Qualification life testing processed approximately 272kg for a flight operational throughput capability of 181kg. It is predicted that the thruster will have a mission throughput capability greater than 285kg of propellant that may provide science mission applicability. Multiple sources funded additional life testing of the BPT-4000 for additional erosion data, to demonstrate a larger throughput capability, and to assess performance at low powers (Welander et al., 2006).

3.2 NASA thrusters

NASA leads the U.S. development of primary electric propulsion thrusters. Three particular thrusters, NSTAR, NEXT, and HiVHAC were or are under development for interplanetary science missions under NASA's science mission directorate (SMD), and all were led by the NASA Glenn Research Center (GRC).

NASA's Solar electric propulsion Technology Application Readiness thruster is the state-of-the-art electric propulsion engine for primary propulsion on NASA science missions. The thruster has a nominal full power operation of 2.3kW at 3,100 seconds of specific impulse and 94mN of thrust. The NSTAR engine flew on the (DS1) technology demonstration mission funded through the NASA New Millennium Program in 1998. The DS1 mission successfully demonstrated the capability of the ion propulsion system by processing 81 kg of xenon propellant. A DS1 flight spare thruster was part of an extended life test (ELT) that validated the thruster life up to 235kg of propellant or 157kg of operation life. Probabilistic failure analysis predicts only a one percent failure rate below 178kg (Brophy, 2007). Most recently, NASA launched the Dawn Discovery Class mission to Ceres and Vesta which makes use of multiple NSTAR thrusters to perform the first multi-rendezvous mission, significantly improving the science capability of a single spacecraft.

The NEXT project was competitively selected to develop a nominal 40-cm gridded ion electric propulsion system (Patterson & Benson, 2007) though NASA's In-Space Propulsion Technology (ISPT) project. The objectives of this development were to improve upon the experimental NSTAR system by achieving lower specific mass, higher specific impulse (4,050 seconds), greater propellant throughput (current estimates exceed 700kg of xenon) and increase the power handling capability (6.9kw), thrust (240mN), and throttle range (12:1) to enable use on Flagship Class missions. The ion propulsion system components being developed under the NEXT task include the ion thruster, the power processing unit (PPU), the xenon feed system, and a gimbal mechanism. NEXT multi-thruster testing is shown in figure 5.

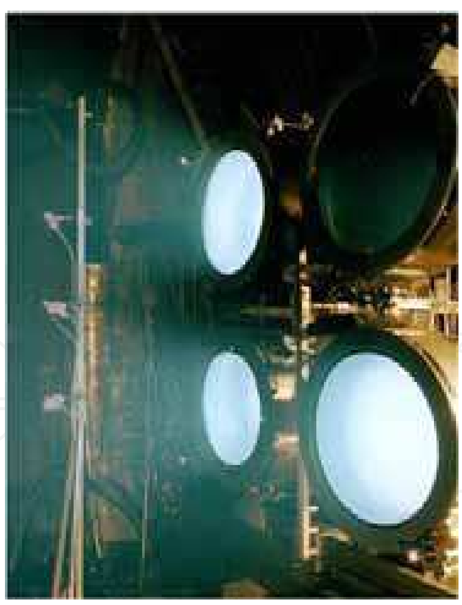


Fig. 5. Multi-thruster testing with NEXT.

The NEXT project is developing prototype-model fidelity thrusters with Aerojet. In addition to the technical goals, the project also has the goal of transitioning thruster manufacturing capability with predictable yields to an industrial source. Recent accomplishments include the production of a prototype-model NEXT thruster which has successfully completed qualification level environmental testing. As of July 1, 2009 a NEXT thruster has demonstrated more than 425kg of xenon throughput and has exceeded 23,000 hours of operation at various throttle conditions. The NEXT wear test has demonstrated the largest total impulse ever achieved by a gridded ion thruster. Wear rates predict first failure beyond 700kg of throughout.

The ISPT project is also investing in the HiVHAC thruster (Manzella, 2007). HiVHAC, shown in figure 6, is the first NASA electric propulsion thruster specifically designed to provide a low-cost electric propulsion option optimized for Discovery and New Frontiers missions. The HiVHAC thruster does not provide as high a maximum specific impulse as NEXT, but a higher thrust-to-power ratio and lower power requirements are well suited for the anticipated demands of these smaller class science missions. The HiVHAC has a nominal full power operation of 3.6kW and a specific impulse of 2,800 seconds at 55% efficiency.

Fig. 6. HiVHAC laboratory model thruster.

Significant advancements in the HiVHAC thruster include a very large (12:1) throttle range, high discharge voltage for an increased specific impulse over conventional Hall thrusters, and a very long-life capability to allow for greater total impulse with fewer thrusters. The Hall thruster has fewer parts and low complexity with significant cost benefits over gridded-ion alternatives. A laboratory model HiVHAC thruster underwent wear testing and successfully completed its test demonstrating over 100kg of propellant throughput and over 4,750 hours of operation at a discharge voltage of 700V. The ISPT project procured the design and development of two engineering model HiVHAC thrusters with Aerojet. The engineering model thruster should be delivered and available to begin performance and life testing starting in the fall of 2009. The engineering model thrusters are projected to have thruster lifetime capability on the order of 30,000 hours of operation or having a total througput in excess of 600kg of propellant (Kamhawi et al., 2009).

4. Interplanetary application of electric propulsion

There is a misconception outside the low-thrust community that high specific impulse always translates directly to higher performance. This belief is founded from Tsiolkovsky's rocket equation.

$$\frac{m_{final}}{m_{initial}} = e^{\frac{-\Delta V}{g I_{sp}}} \tag{2}$$

However, its general application for high thrust, chemical propulsion, systems assumes that the mission ΔV remains relatively constant. If the ΔV remains constant, slight increases in specific impulse can have significant mass benefits to the mission. If thrust is decreased in exchange for higher specific impulse, the efficiency of the maneuver may decrease and the total ΔV requirement could rise, decreasing or negating any gain due to the increased exhaust velocity. One example is a launch vehicle whose specific impulse is increased, but its thrust-to-weight ratio is below one. The vehicle will consume all of its propellant without ever leaving the launch pad.

For electric propulsion thrusters, the thrust is inversely proportional to the specific impulse given a constant power.

$$P = \frac{g(FI_{sp})}{2\eta} \tag{3}$$

The most efficient propulsive maneuvers are impulses, or infinite thrust; though impossible to achieve. Chemical propulsion maneuvers are often treated as impulse maneuvers, but the low-thrust ΔV penalty of long finite burns can be quite severe. One example would be a simple plane change in an elliptical orbit. The ΔV of a plane change is a function of the spacecraft velocity.

$$\Delta V = 2V \sin\left(\frac{\emptyset}{2}\right) \tag{4}$$

The spacecraft velocity is slowest at apoapsis, and therefore, an impulsive maneuver at apoapsis will have a lower ΔV requirement than if the maneuver must be performed over a large arc.

The entire mission trajectory will have a decreased ΔV if the thrust arcs are smaller and centered on the most efficient locations. This will give a clear advantage to engines that can provide higher thrust. There is a trade between specific impulse and thrust. Figure 7 illustrates a Nereus sample return mission trajectory for the NSTAR thruster and the BPT-4000 (Hofer et al., 2006). The BPT-4000 operates at higher thrust, and therefore, has more efficient maneuvers to produce a lower total ΔV requirement for the mission.

Figure 7 illustrates that the higher thrust maneuvers are shorter, and the total ΔV savings is 1.3km/s. In this example mission, the NSTAR thruster requires approximately 190kg of propellant to deliver a final mass of 673kg while the BPT-4000 consumes 240kg of propellant and delivers 850kg back to Earth. It is worth noting that the higher thrust systems typically optimize to a lower launch energy; though the lower specific impulse BPT-4000 requires more propellant for a smaller ΔV, it delivers more final mass because the launch vehicle can deliver more start mass at the lower launch energy. Overall, the BPT-4000 can deliver more mass because of its higher thrust and ability to decrease the launch energy requirement of the launch vehicle. This is primarily due to the higher power processing capability of the thruster.

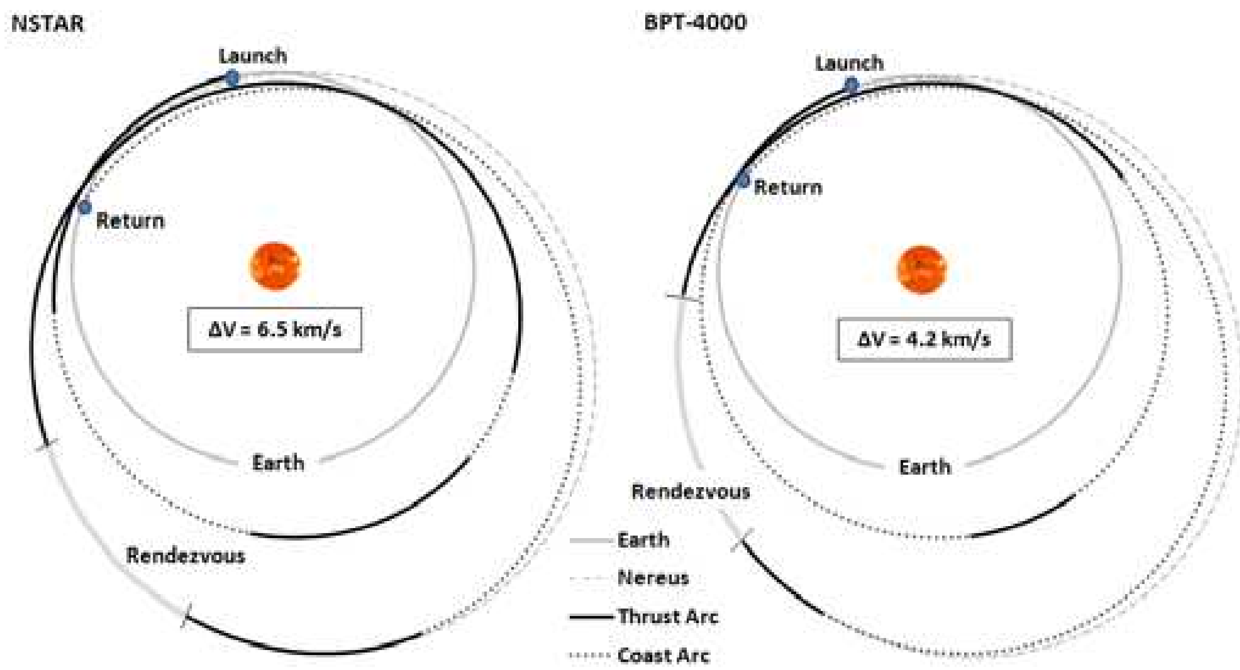


Fig. 7. NSTAR (left) and BPT-4000 (right) trajectories for a Nereus sample return mission.

The NEXT thruster performing the same mission, but de-rated to the maximum power level of the BPT-4000, can deliver 911kg consuming just 188kg of propellant. NEXT still requires a ΔV of 5.3km/s, greater than the Hall thruster, but the higher Isp results in a greater net delivered mass. It is not always obvious which thruster will have the highest performance. It is dependent on the trajectory profile, available power, mission duration, etc.

Another consideration of mission design is the ability to tolerate missed thrust periods. An advantage of higher thrust systems and the decreased thrust arcs is also the robust design of the trajectory. While the NEXT thruster delivers more mass than the BPT-4000, it is required to operate for 513 days of the 1,150 day mission. The BPT-4000 only operates for 256 days for the same mission duration. A missed thruster period, either for operations or an unplanned thruster outage, can have a negative impact on the mission. Higher thrust systems are typically more robust to missed thrust periods with their ability to makeup lost impulse in a short time period. Recalling equation 3, a higher power system can have both a higher thrust and higher specific impulse.

When power is limited, an optimal low-thrust mission will use the available power for higher thrust when small changes in thrust will create large savings in ΔV. When large changes in thrust have a small effect in ΔV, the thruster would use the remaining available power for an increased specific impulse. The trajectory is optimizing delivered mass with the ΔV term of Tsiolkovsky's equation having a strong dependency on thrust. Figure 8 is an example of optimal specific impulse for a rendezvous mission with the comet Kopff. The mission optimized to specific impulses of 2920s, 3175s, and 3420s, at power levels of 6kW, 7.5kW, and 9kW respectively.

A remaining consideration for designing low-thrust mission trajectories is the proper methodology of margin. The trajectory must account for planned and unplanned thruster outages, power margin, thrust margin, propellant margins due to trajectory errors, residuals that cannot be expelled from the tank, or flow control accuracy, ΔV margins, etc. Though the margins are interdependent, the electric propulsion system can offer advantages with an ability to compensate for one area with additional margin in another (Oh et al., 2008).

In general, interplanetary missions with the greatest benefit of using electric propulsion are missions that do not capture into large gravity wells, and have very large total ΔV mission requirements. High ΔV missions include missions to multiple targets, large inclination changes, and deep space rendezvous with trip time limitations. Trajectory analyses were performed in Copernicus and MIDAS for chemical comparison and using SEPTOP, SEPSPOT, and MALTO for the low thrust solutions.

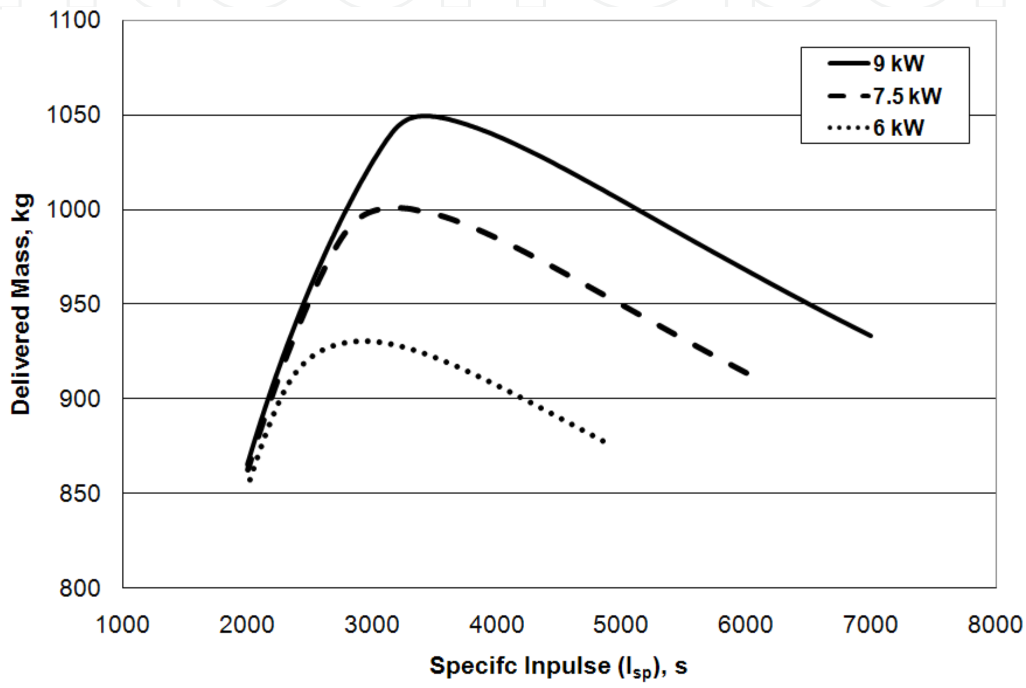


Fig. 8. Optimal specific impulse comparison for a comet rendezvous mission.

4.1 Multiple targets

Multi-target missions are a method to achieve considerably higher science return for a single spacecraft. Multi-target missions can range from two targets in similar orbits, several targets requiring large maneuvers, and to some extent, sample return missions.

The Dawn mission illustrates the mission enhancing capabilities of electric propulsion for just such a mission. It is the first NASA science mission to use electric propulsion. For a mission to be competitively selected and to justify new technology, the science return must be remarkably high. The Dawn mission utilizes a single spacecraft that carries an instrument suite to multiple targets, Ceres and Vesta. By traveling to multiple targets with a single spacecraft there are savings in spacecraft development, instrument development, and launch costs. The mission provides a unique opportunity to compare data from an identical sets of instruments. The Dawn mission was determined to only be viable through the use of electric propulsion. The use of chemical propulsion required significantly higher launch mass and could only feasibly reach a single target. Figure 9 illustrates the Dawn EP multi-rendezvous trajectory.

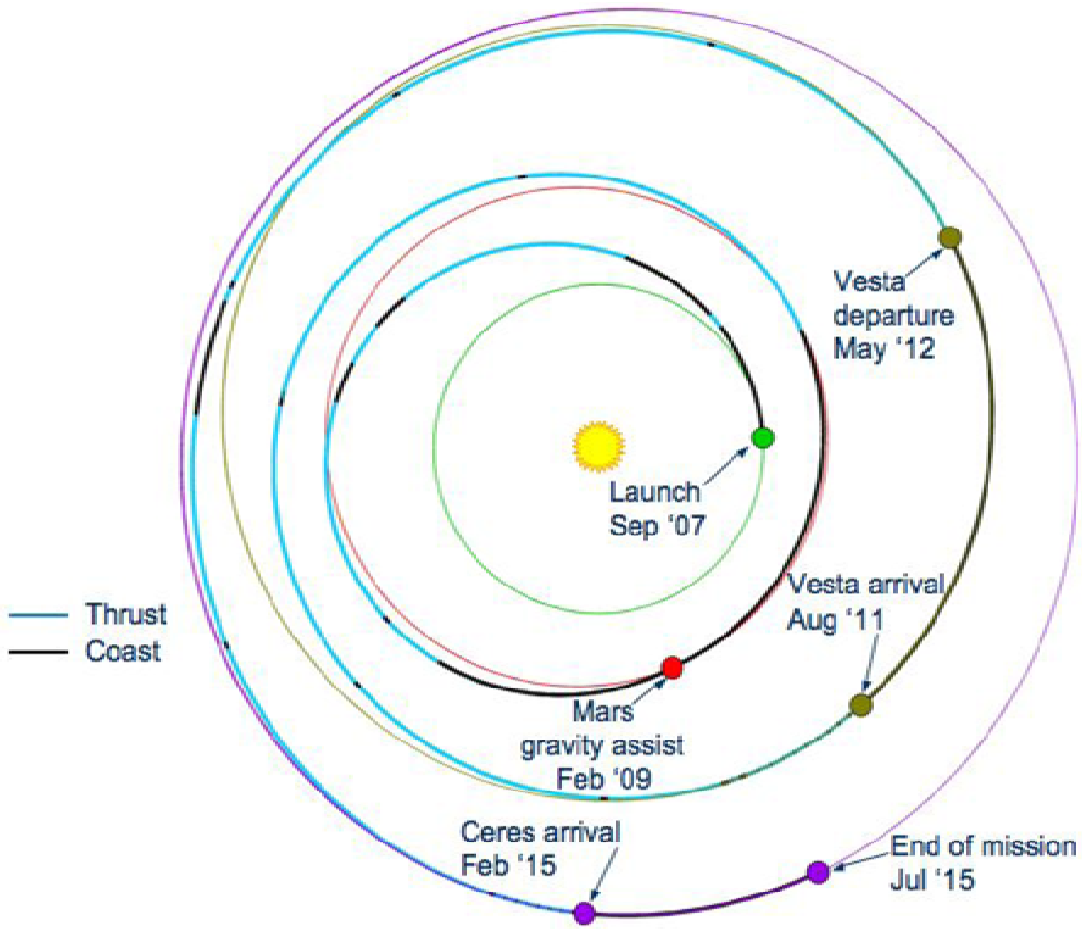


Fig. 9. Trajectory of the Dawn mission to Ceres and Vesta (Brophy et al., 2008).

For a Dawn-like mission, the use of either the NEXT or HiVHAC thruster has significant mission advantages over NSTAR. The throughput capability of a single NSTAR thruster is insufficient to complete the mission. The use of the NEXT thruster can not only deliver greater payload, but can do so using a single thruster. The use of a single thruster has system complexity and spacecraft integration advantages. The HiVHAC thruster, capable of performing the Dawn mission with a single operating thruster, is expected to have numerous advantages including greater payload and reduced cost. The Dawn mission power requirements are driven by the need to operate the thruster throughout the mission, including significant operation at relatively high AU. Figure 10 illustrates the Dawn power profile.

Using a thruster that can throttle to very low power can reduce the spacecraft power requirements, reducing the overall spacecraft cost. Because many small body rendezvous missions can benefit from higher thrust, lower specific impulse, thrusters, a low power Hall thruster can deliver greater payload than the NSTAR thruster. Finally, a Hall thruster system is expected to have lower thruster and power processing unit costs. The HiVHAC system is expected to provide significant reductions in IPS costs over SOA. Figure 11 illustrates performance and cost benefits of ISPT project technologies over SOA. Overall, the NEXT thruster can outperform the NSTAR thruster with reduced system complexity. The HiVHAC thruster, specifically designed for Discovery Class missions, has the potential for increased performance and reduced system complexity and cost (Oh, 2005).

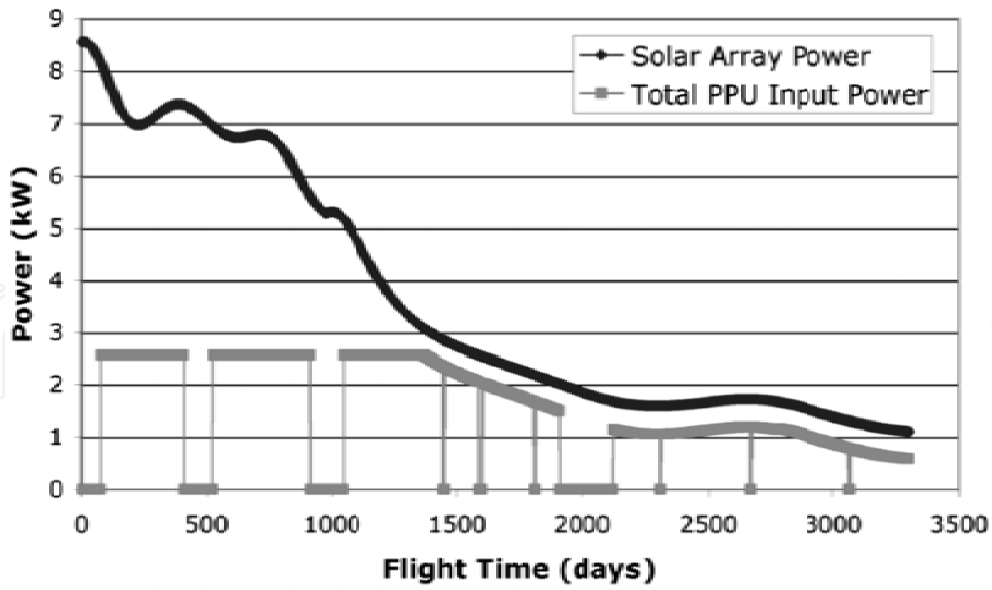


Fig. 10. Power profile for the Dawn mission (Oh, 2007).

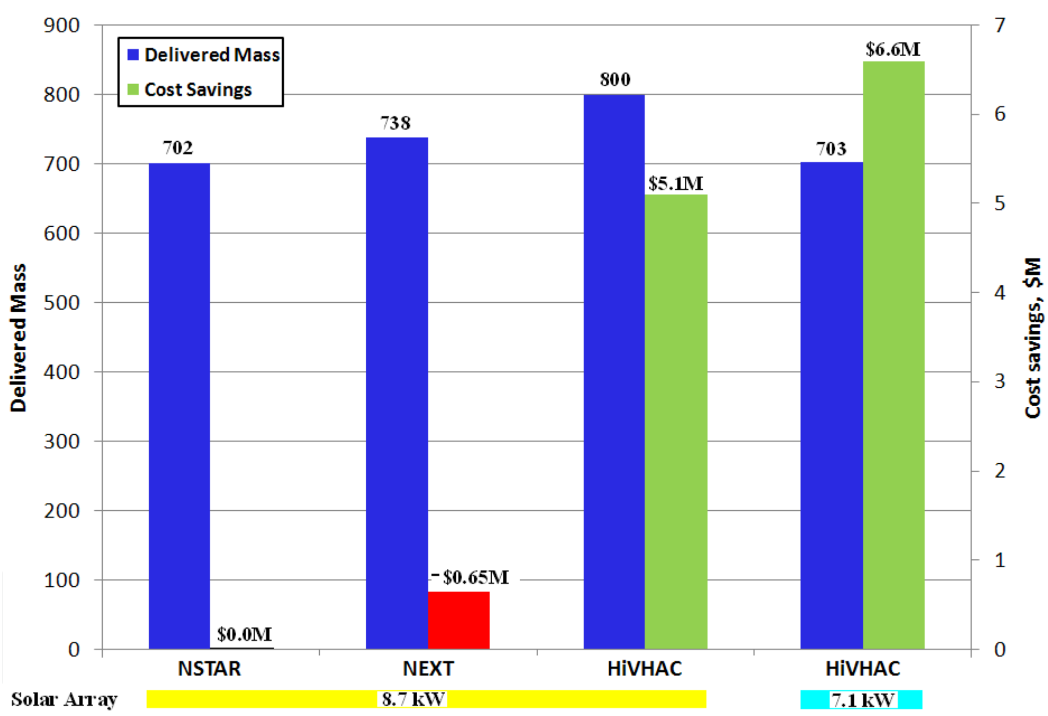


Fig. 11. Mass and cost comparison for the Dawn mission (Reference).

The extreme example of a multiple target mission would be a Super-Dawn. A "Super-Dawn" class mission is the concept of traveling to and stopping at several high interest targets with a single launch. The throughput potential of both NEXT and HiVHAC greatly opens up the trade space of achievable multi-rendezvous options. Unfortunately, most high interest targets are not necessarily co-located to allow for short transfers due to variances in inclination, eccentricity, period, etc. Sufficient throughput capability and creative mission planning could allow missions to visit multiple high interest targets and gain information from several secondary near-by targets.

The targets for a hypothetical “Super-Dawn” mission were chosen from a list of high interest targets formulated by the scientific community. Based on preliminary analysis of throughput requirements and delivered mass, a single spacecraft, with only a 5-kW array, could be used to rendezvous with four high interest near-Earth targets shown in table 1. The final delivered mass is comparable to the Dawn spacecraft. The “Super-Dawn” mission illustrates the tremendous potential of electric propulsion for these types of missions. Studies have looked at using a single spacecraft for tours of near-Earth objects, main-belt asteroids, and even Jupiter Trojans.

Sample return missions are multi-body missions because they need to return to Earth. Sample return missions are often considered high priority because of the higher fidelity science that can be performed terrestrially. Mars sample return was under investigation for many years, but the large costs of such a mission has deterred its implementation. Regolith from Phobos and Deimos are of high scientific value. The mission options offer significantly lower cost with minimal technology development required.

Segment	Target	Start Mass, kg	Propellant Required, kg	End Mass, kg
1	Nereus	1650	309	1341
2	1993 BD3	1341	52	1289
3	Belenus	1289	44	1245
4	1996 FG3	1245	456	789

Table 1. Table of ΔV for a “Super-Dawn” type mission.

Two concepts for a Phobos and Deimos sample return mission were evaluated using solar electric propulsion: a single spacecraft to both moons or twin spacecraft capable of returning samples from either moon. The small bodies of Phobos and Deimos, with small gravity fields (especially Deimos), make electric propulsion rendezvous and sample return missions attractive. Electric propulsion systems can be used for the transfer to Mars, and then to spiral into an orbit around the moons. Chemical systems cannot easily leverage the Oberth effect for the sample return mission from Mars’ moons because of the higher altitude orbit requirement. So while the mission can be completed, it comes at a large mass penalty. Figure 12 illustrates the benefits of using electric propulsion for a Phobos and Deimos sample return mission.

Results show significant savings for using electric propulsion for Phobos and Deimos sample return missions. The baseline case uses a NEXT thruster with one operating thruster, and a spare system for redundancy (1+1). A Delta II class launch vehicle is capable of delivering enough mass for a sample return from both targets. For electric propulsion, the transfer between Phobos and Deimos has minimal mass implications. The mass and technology requirements could potentially fit within the Mars Scout cost cap.

Using an Evolved Expendable Launch Vehicle (EELV), twin electric propulsion vehicles can be sent for a low-risk approach of collecting samples from Phobos and Deimos independently. However, the use of an EELV enables a chemical solution for a sample return mission. Going to a single moon chemically remains a significant challenge and results in a spacecraft that is greater than 70 percent propellant; a mass fraction more typical of a launch vehicle stage. Launching a single chemically propelled spacecraft to retrieve samples from both moons requires staging events adding risk and complexity.

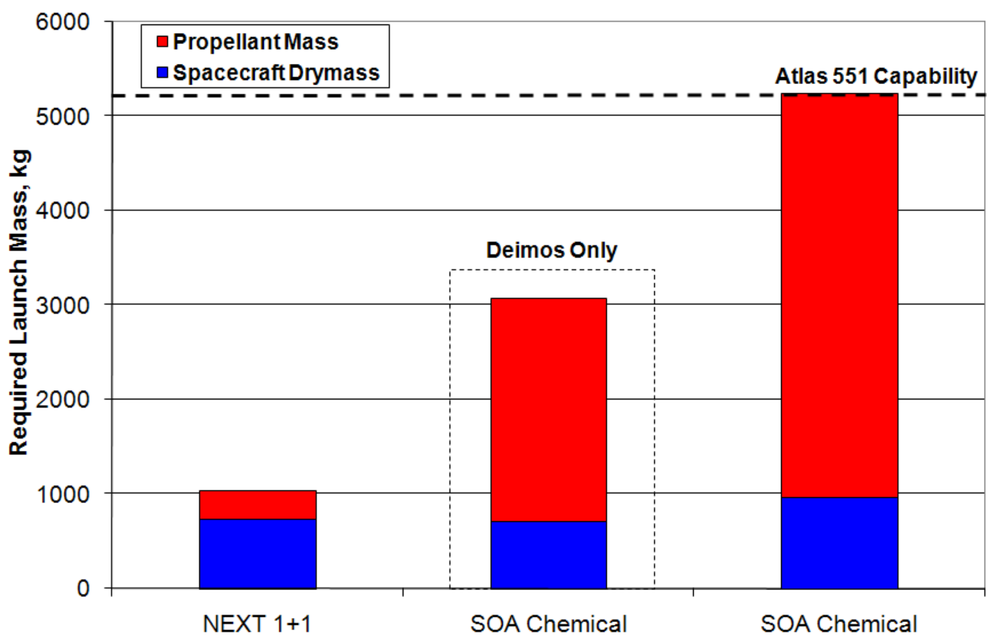


Fig. 12. Comparison of required launch mass for chemical and EP Mars' moons missions.

The use of electric propulsion was studied for various comet surface sample return (CSSR) missions. The results are highly dependant on the targets of interest. Electric propulsion compares favorably with chemical alternatives resulting in either higher performance or reduced trip times. Studies for Temple 1 (Woo et al., 2006) determined the SOA NSTAR thruster to be inadequate due to its propellant throughput capability. The mission required the use of a NEXT thruster. Studies for the comet Wirtanen (Witzberger, 2006) were conducted and determined that the NSTAR could not deliver positive payload while both the NEXT and HiVHAC thrusters can complete the mission with sufficient margin. The largest benefit is that electric propulsion enables a wide range of targets that cannot be reached using chemical propulsion systems.

In 2008, NASA GRC completed a mission design study for a multiple near-Earth asteroid sample return mission (Oleson et al., 2009). The results indicated that it is feasible to use electric propulsion to collect multiple samples from two distinct targets in very different orbits. An Earth fly-by was performed after leaving the primary target and before arriving at the second to releae the sample return capsule for a lower risk mission and mass savings to the secondary target. This mission was not feasible using chemical propulsion. The conceptual spacecraft for the multi-asteroid sample return mission is shown in figure 13.

4.2 Inclined targets

Other missions enabled by electric propulsion are missions to highly inclined targets. There are several Earth crossing targets that are thought to be old and inactive comets. These asteroids typically have inclined orbits. The ΔV requirement for a plane change is a function of the spacecraft velocity and angle of the plane change as shown in equation 1. With the Earth's heliocentric orbital speed near 30 km/s, a simple plane change of even 30 degrees will require a ΔV of at least 15 km/s to perform a fly-by, following equation 4.

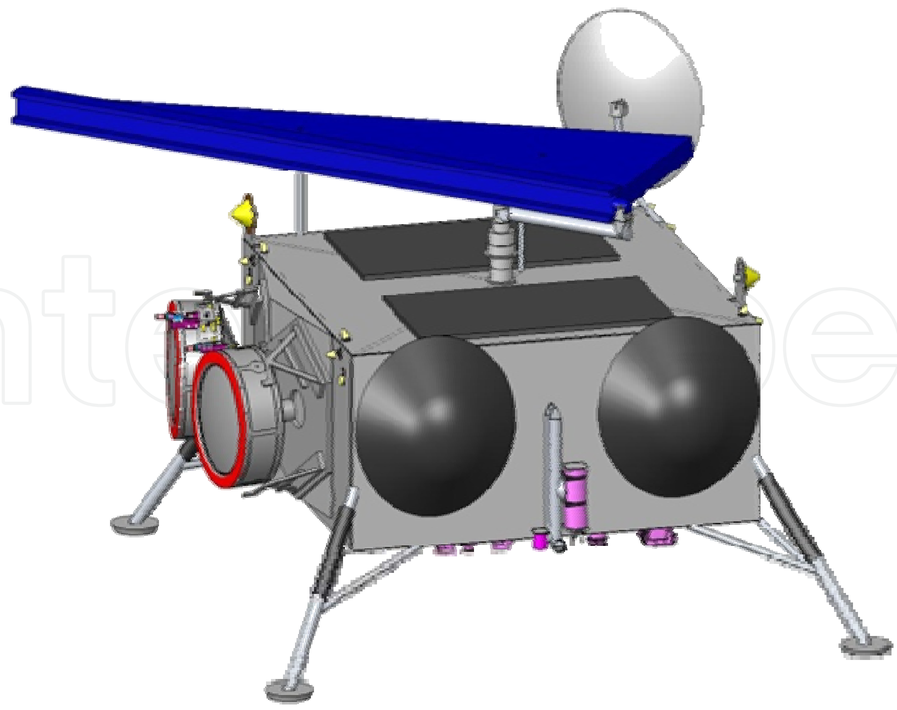


Fig. 13. NEARER spacecraft photo.

An example mission to an inclined target would be to the asteroid Tantalus. Tantalus is an Earth-crossing body with a semi-major axis of 1.29 AU, an eccentricity of 0.3, and an inclination of 64°. Because of the inclination change, the optimal chemical transfer requires a high launch velocity so that the spacecraft can perform the plane change at high AU with a lower velocity. Figure 14 illustrates the chemical transfer which requires a post-launch ΔV of 14 km/s. The Tsiolkovsky's mass fraction is only on the order of one percent dry mass, so the mission is completely infeasible with any launch vehicle using chemical propulsion.

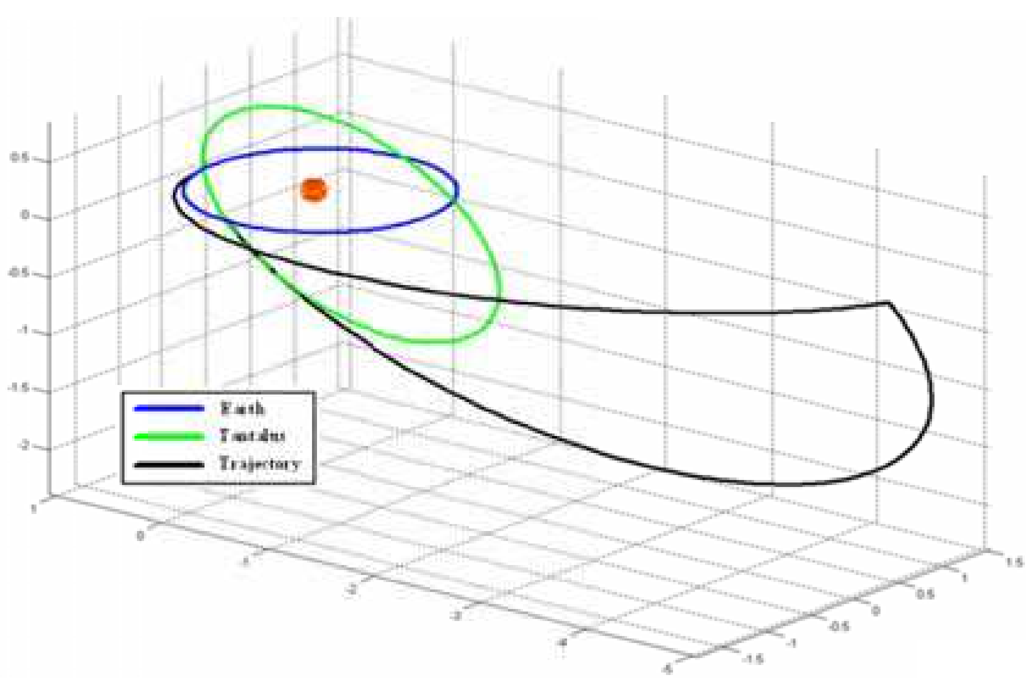


Fig. 14. Optimal chemical trajectory to Tantalus.

The electric propulsion transfer to Tantalus is also a challenging mission. The low-thrust transfer is over 30 km/s over 4.5 years, but can still deliver over 800 kg of dry mass on a rendezvous mission using an Atlas V. The mission would require two NEXT thrusters, and would not be viable with the NSTAR or Hall thruster based propulsion system. Rather than going to high AU to perform the plane change, the low-thrust transfer gradually performs the plan change through several revolutions. Figure 15 illustrates the low-thrust transfer to Tantalus. Because of the advantages of electric propulsion, efficient use of propellant and low-thrust trajectory options, scientists can plan missions to high interest targets previously unattainable.

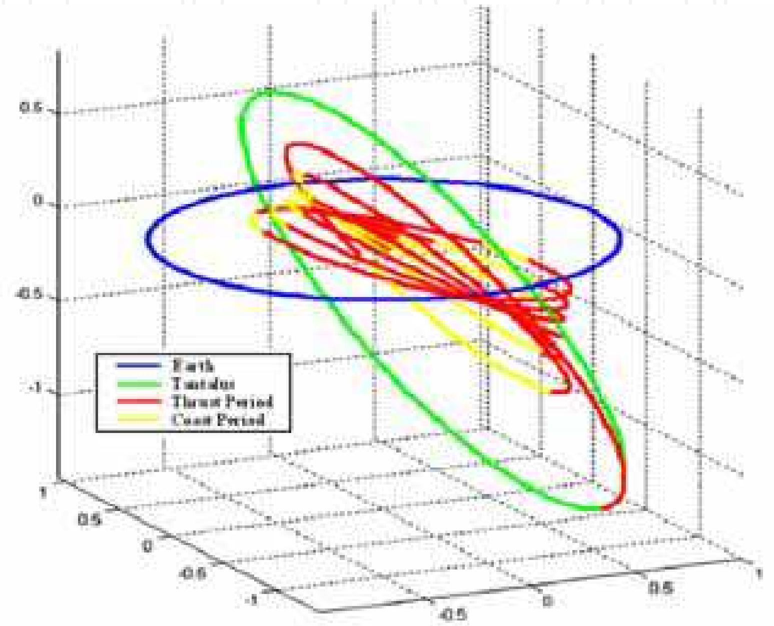


Fig. 15. Optimal low-thrust trajectory to Tantalus.

4.3 Radioisotope electric propulsion

Another area of interest pushing the limits of propulsion technology is the use of a radioisotope power source with an electric propulsion thruster. This achieves high post launch ΔV on deep-space missions with limited solar power. Radioisotope electric propulsion systems (REPS) have significant potential for deep-space rendezvous that is not possible using conventional propulsion options.

One example of mission that can benifit from REPS is a Centaur orbiter. The Centaurs are of significant scientific interest, and recommended by the Decadal Survey Primitive Bodies Panel as a New Frontiers mission for reconnaissance of the Trojans and Centaurs. The original recommendation was for a flyby of a Jupiter Trojan and Centaur. While a flyby mission can use imaging, imaging spectroscopy, and radio science for a glimpse at these objects, a REP mission provides an opportunity to orbit and potentially land on a Centaur. This greatly increases the science return. An exhaustive search of Centair obiter missions concluded that a wide range of Trojan flybys with Centaur Rendezvous missions are pracitical with near-term electric propulsion technology and a Stirling radioisotope generator (Dankanich & Oleson, 2008). With near-term technology, flyby missions may no longer be scientifically acceptable. Investigations are continuing using the enabling combination of electric propuslion and radioisotope power systems. On-going and recent studies include multi-Trojan landers, Kuiper-belt object rendezvous, Titan-to-Enceldaus transfer, and Neptune Orbiters. A conceptual spacecraft design of a Centuar orbiter is shown in figure 16.

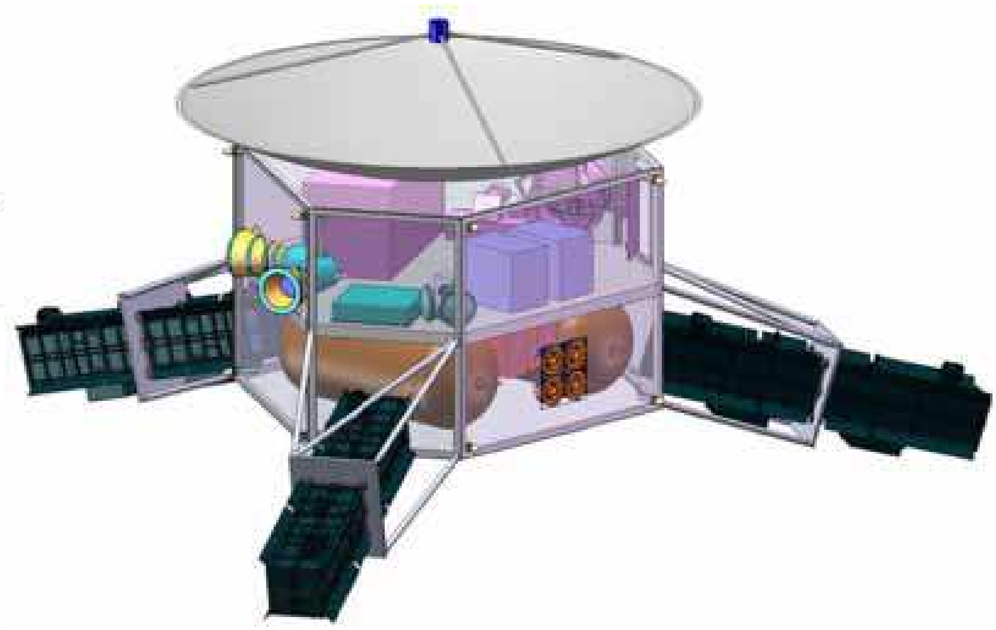


Fig. 16. Centaur orbiter spacecraft design.

5. Near-Earth application of electric propulsion

There are currently over 200 satellites using electric propulsion and the majority of those satellites have multiple thrusters. Only four spacecraft flew beyond geosynchronous orbit altitude, two from NASA, one from JAXA to a near-Earth object, and one from ESA to the moon. The vast majority of mission pull of electric propulsion is for application in the region between low-Earth orbit (LEO) and geosynchronous Earth orbit (GEO,) specifically commercial satellites with GEO operational orbits.

A conjunction of developments greatly increased the practicality and expectations for the use of electric propulsion for Earth-orbit transportation. Key among these advancements are trends of satellite power-to-mass ratios, required spacecraft mass, advancements in space power technology, and the broad use of electric propulsion systems. An illustration of the trends of both increasing power-to-mass ratios and spacecraft mass for commercial communication satellites are shown in figure 17 (Byers & Dankanich, 2008).

Several studies were performed to evaluate the use of electric propulsion for transfer starting at LEO. A strong deterrent is the very long transfer times. One near-term option is to leverage the launch vehicle to an eccentric orbit. The primary design capability of commercial launch vehicles is to a geosynchronous transfer orbit (GTO) with perigee and apogee altitudes of 185 km and 35,786 km respectively. Launching directly to GTO will significantly reduce the orbit transfer time while also reducing risk of orbital debris and exposure to the radiation environment within the Van Allen belts. Analyses for transfer times and ΔV requirements from equatorial launch sites (Baikonur, Kourou, and the Kennedy Space Center (KSC)) were evaluated and characterized as a function of spacecraft power-to-mass ratios (Dankanich & Woodcock, 2006). Results for KSC launches are shown in figures 18 and 19 respectively. Above 3 W/kg, the acceleration is high enough to minimize gravity losses of the transfer. The low-thrust ΔVs are approximately 800 m/s more than a chemical GEO insertion.

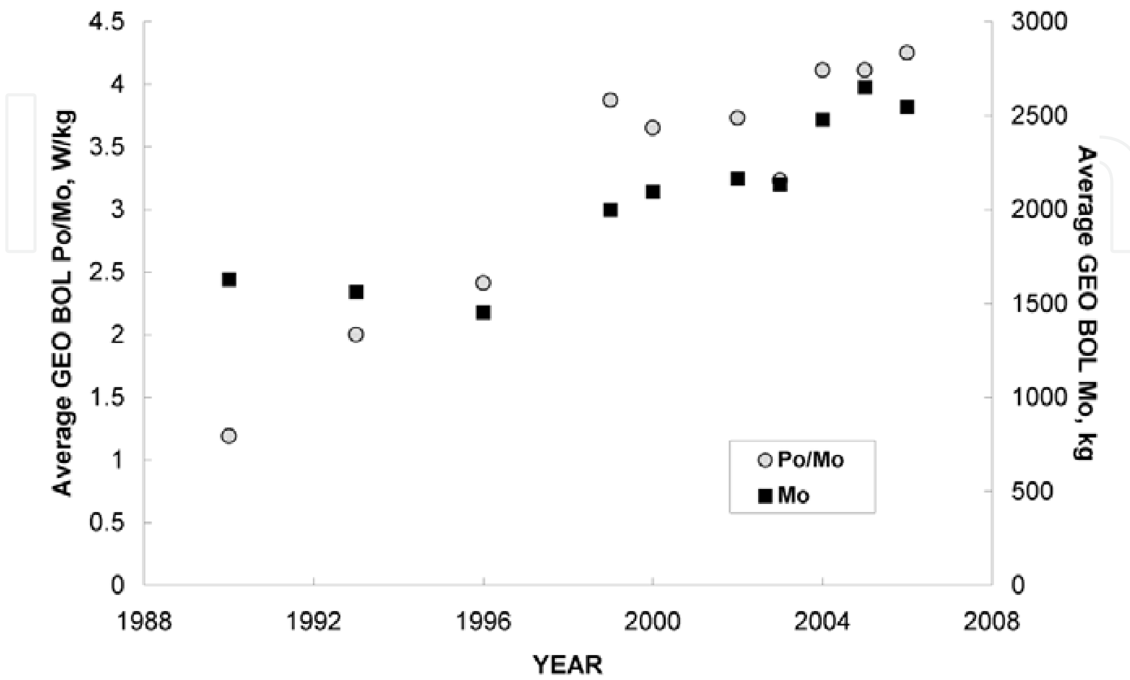


Fig. 17. Trends of commercial satellite beginning-of-life (BOL) P/M ratio and average mass.

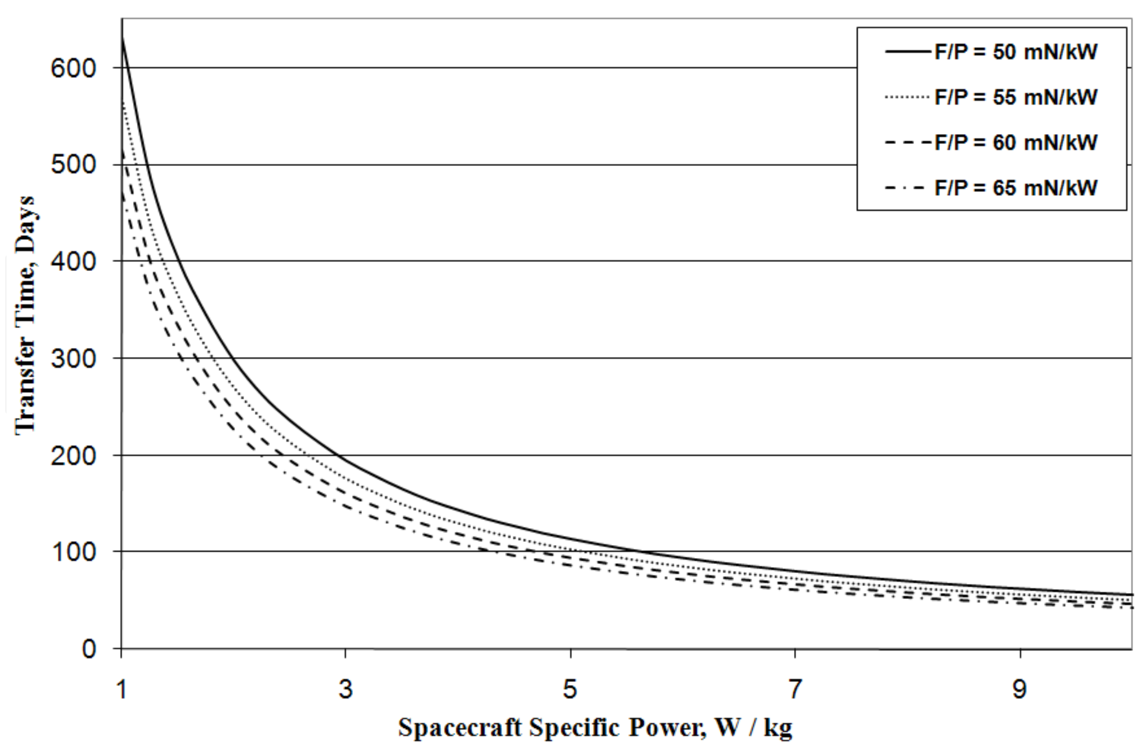


Fig. 18. GTO-to-GEO transfer times as a function of spacecraft specific power.

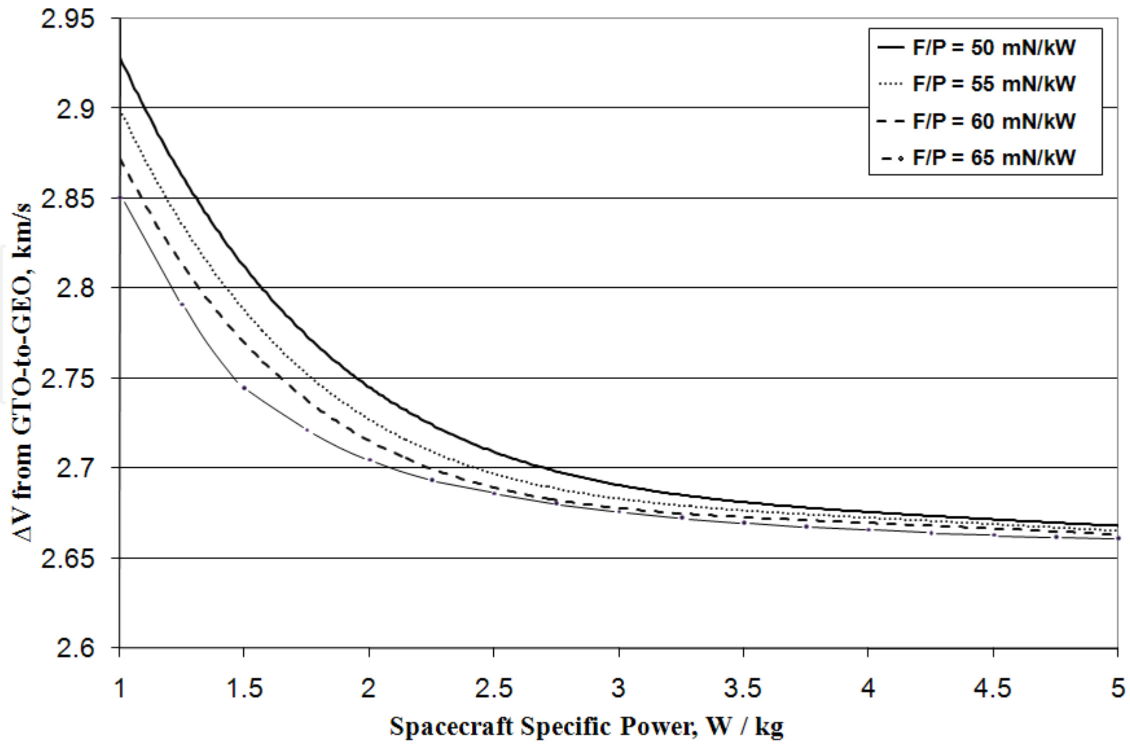


Fig. 19. Required ΔV from GTO-to-GEO as a function of spacecraft specific power.

The GTO-to-GEO transfer time and ΔV is dependant on the launch site, or initial starting inclination. Figure 20 illustrates the penalty of launch at inclined launch sites and the benefit of near-equatorial launches.

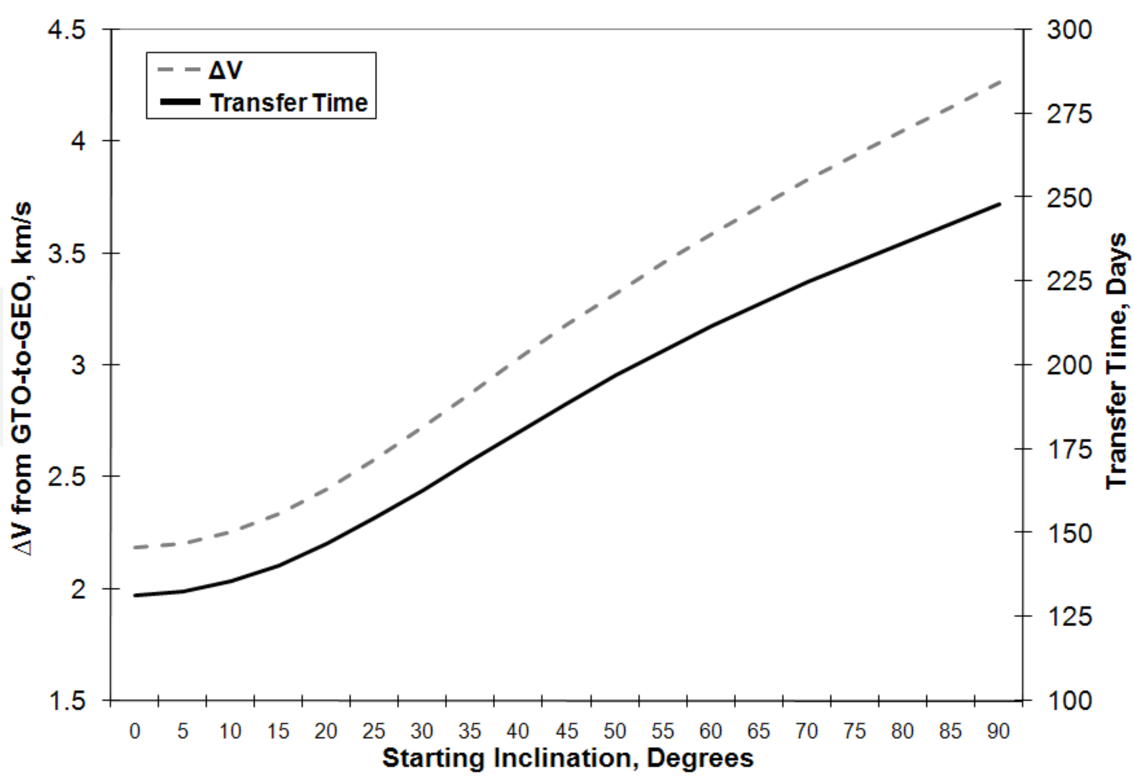


Fig. 20. Effect of starting inclination on transfer time and ΔV from GTO-to-GEO.

There were 32 commercial communication satellites launched in 2005 and 2006 as provided by the Union of Concerned Scientists database. These specific satellites were evaluated for potential to use an integrated electric propulsion system with a specific impulse of 1000 seconds, 1500 seconds, and 2100 seconds. Integrated electric propulsion systems assume the use of 95% of the onboard solar array power of the spacecraft as launched.
Using electric propulsion for the GEO insertion has significant mass benefits. Typically this is evaluated as a method to leverage the launch vehicle performance to deliver the greatest possible mass. Another perspective is to evaluate the potential for existing launch vehicles to meet the demands of the COMSAT market. Figure 21 illustrates that currently launch vehicles with GTO drop mass capabilities in excess of 7,500kg are required for a complete market capture. However, using electric propulsion, a launch vehicle with a drop mass capability of 5,500kg can have complete market capture. A low cost launcher with a capability to deliver 3,500kg to 5,500kg can create a paradigm shift in the commercial launch market. This assumes the commercial entity is willing to endure the long transfer time, ranging from 66–238 days, depending on the spacecraft power-to-mass ratio and EP thruster selected.

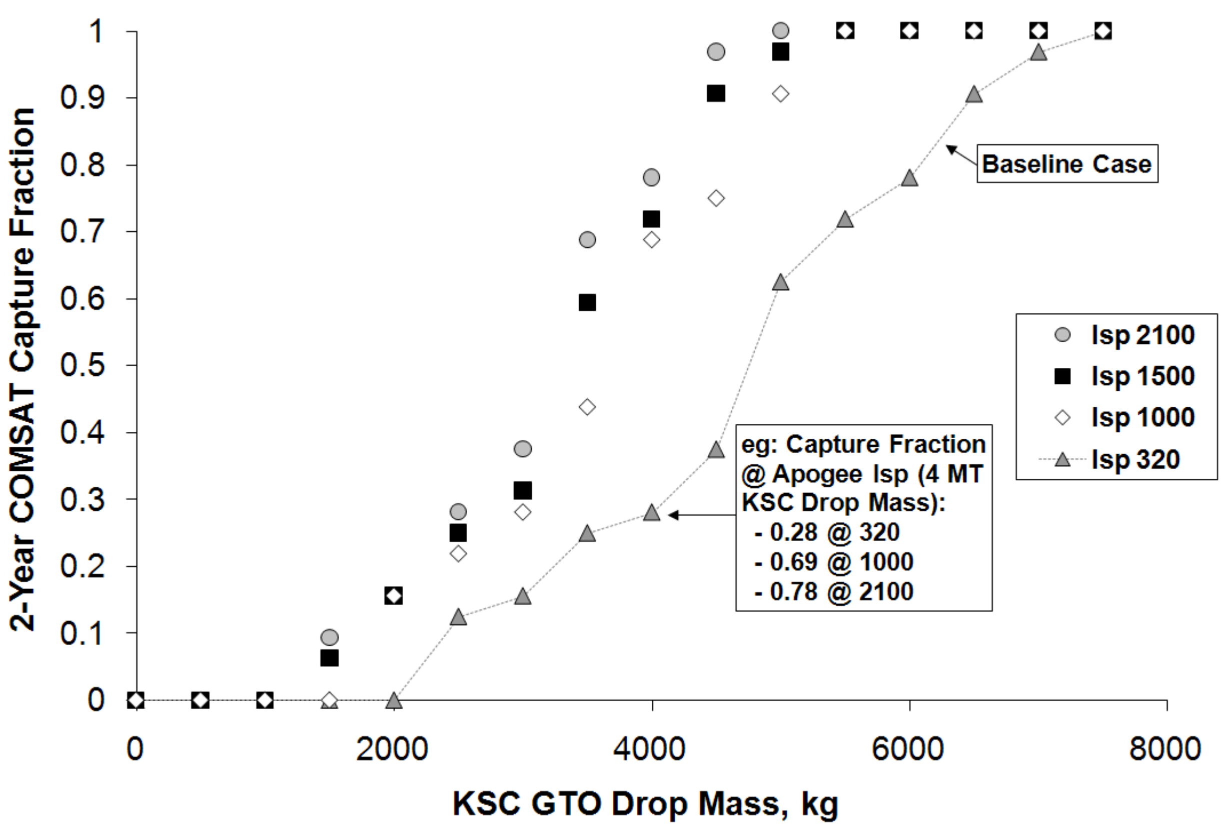


Fig. 21. Capture fraction as a function of GTO drop mass for various propulsion options.

6. Conclusion

Electric propulsion technology is widely used today, and multiple thrusters exist for primary electric propulsion application. NASA and the U.S. commercial market developed several thrusters suitable for primary electric propulsion on full scale spacecraft. The technology drivers for new electric propulsion thrusters include: ability to use available power (i.e. high maximum power with large throttle range), increased total throughput capability, and lower cost systems and integration. The optimal specific impulse is limited by thrust required to minimize propulsive inefficiencies and available power. Due to power constraints, the optimal specific impulse is typically less than 5,000s and closer to 2,000s for near-Earth application. Electric propulsion is an enabling technology for a large suite of interplanetary missions. Several targets are infeasible with advanced chemical propulsion technologies, while practical with today's electric propulsion options. Electric propulsion is well suited for missions with very high post-launch ΔVs including multi-target missions, sample return missions, deep-space rendezvous, and highly inclined targets. Electric propulsion has tremendous capability to impact the commercial launch market by leveraging on-board available power. Today's commercial satellites have mass-to-power ratios for practical GTO-to-GEO low-thrust transfer. As available power and performance demand continues to rise, electric propulsion technologies will continue to supplant chemical alternatives for a wide range of missions. The technology will continue to focus on developing lower cost propulsion systems with higher power and longer lifetime capabilities.

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