ANALYSIS OF HALL-EFFECT THRUSTERS AND ION ENGINES FOR ORBIT TRANSFER MISSIONS

Frank S. Gulczinski III*, Ronald A. Spores**
Propulsion Directorate
OL-AC Phillips Laboratory
Edwards AFB, CA 93524

ABSTRACT

Analytical methods were combined with actual thruster data to create a model used to predict the performance of systems based on two types of electric propulsion thrusters, Hall-effect thrusters and ion engines, for several orbit transfer missions. Two missions were trip time constrained: a LEO-GEO transfer and a LEO constellation transfer. Hall thrusters were able to deliver greater payload due to their higher overall specific power. For the power limited orbit topping mission, the choice of thruster is dependent on the user's need. Ion engines can deliver the greatest payload due to their higher specific impulse, but they do so at the cost of higher trip time. Study of reusable electric orbit transfer vehicle systems indicates that they can offer payload mass gains over chemical systems, but that these gains are less than those offered by other electric propulsion transfer scenarios due to the necessity of carrying propellant for return trips. Additionally, solar array degradation leads to increased trip time for subsequent reusable transfers.

INTRODUCTION:

The US Air Force has recently completed several studies to investigate the potential advantages of advanced space propulsion for several orbit transfer scenarios. The first study investigated advanced propulsion concepts for expendable orbit transfer vehicles[1] and concluded that the potential launch vehicle downsizing that resulted from the use of high specific impulse thrusters provided significant cost savings over base line chemical launch vehicle/upper stage systems. The second study looked at reusable advanced upper stages[2] and preliminary indications are that while there remains the potential for launch vehicle downsizing, it is significantly reduced compared to expendable systems. This difference was largely due to the added propellant required to perform the round trip mission from low-earth orbit to geostationary orbit. Both studies pointed out advantages for advanced electric propulsion systems based on xenon propellant. The objective of this paper is to analyze the trade-offs between Hall-effect thrusters and ion engines as a high power propulsion system for orbit transfer missions.

Both the Hall-effect thruster and the gridded ion engine are classified as electrostatic thrusters and operate on heavy noble gases, primarily xenon. These electric propulsion devices are capable of specific impulses ranging from approximately 1500 to 4000 seconds, compared to chemical systems which typically operate in the range of 300 to 400 seconds.

Electric propulsion is a type of rocket propulsion for space vehicles and satellites which utilizes electric and/or magnetic processes to accelerate a propellant at a much higher specific impulse than attainable using classical chemical propulsion. The concomitant reduction in required propellant mass results in increased payload mass capability.

The method of analysis used in this study is based on the model developed by Messerole.[3] It has been modified to reflect the most current information on thruster development levels and

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* Research Aerospace Engineer, Member AIAA
** Group Leader, USAF Electric Propulsion Lab, Member AIAA

to allow for greater flexibility in system component variation.

There are four missions examined in this study. They are a low earth orbit to geosynchronous earth orbit (LEO-GEO) transfer, a LEO to intermediate orbit transfer for constellations of satellites, a geostationary orbit insertion with partial chemical propulsion, and a reusable orbit transfer vehicle concept. These missions are representative of the range of orbit transfer scenarios that the Air Force presently envisions for an electric propulsion upper stage, and are among those likely to be attempted over the next 10 to 20 years.

ORBIT TRANSFER HARDWARE:

The key components of an electric propulsion orbit transfer vehicle are the thrusters and the power generation sub-system. In this study, two types of thrusters are examined: the Hall thruster and the ion engine. Power is generated by a concentrator solar array.

Hall-effect thrusters are a type of electrostatic thruster in which ions are generated by electron bombardment. The ions are then accelerated by an electron cloud which is held in place by a magnetic field perpendicular to the direction of acceleration. The electron cloud is generated by an applied electric field.

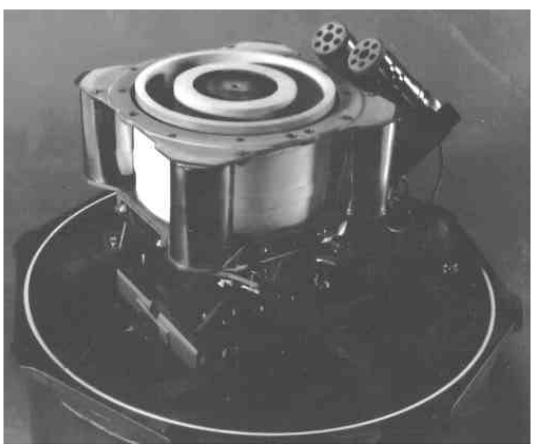

Figure 1: Hall Thruster (SPT-100)

Initial work on Hall-effect thrusters began in the 1960's in the United States and the former Soviet Union. Due to difficulties achieving the same levels of efficiency reached by the gridded ion engine, work ceased in the United States around 1970.[4] In the Soviet Union, research into the ion acceleration mechanism led to improvements in efficiency and further research and development. Two basic types of Hall thrusters were developed: the Stationary Plasma Thruster (see Figure 1) developed under the leadership of A.I. Morozov at the Kurchatov Institute and the Anode Layer Thruster developed under the leadership of A.V. Zharinov at TSNIIMASH.[5] The primary differences between the two types are that the acceleration region of the SPT is within the thruster itself while for the ALT it is in front of the thruster and the lack of an acceleration chamber insulator in the ALT. This study does not distinguish between the various types of Hall thruster concepts with regards to performance.

Over sixty-four SPT-50 and SPT-70 units have flown aboard Russian spacecraft, beginning with the Meteor satellite in 1972[6] (the numerical designation in the name of a SPT is the outer diameter of the discharge chamber in millimeters). The first SPT-100s flew in 1994 on the GALS spacecraft. Larger thrusters, the SPT-140, SPT-200, and SPT-290 have undergone various levels of laboratory development. With the end of the Cold War, this technology became available for evaluation and use in the West. Work in the United States to further quantify SPT performance and flight qualify SPTs for western spacecraft has been done primarily at the NASA Lewis Research Center[7] and the Jet Propulsion Laboratory.[8] Space Systems Loral has developed power processing units for the SPT-100 and is working to develop higher power PPUs.

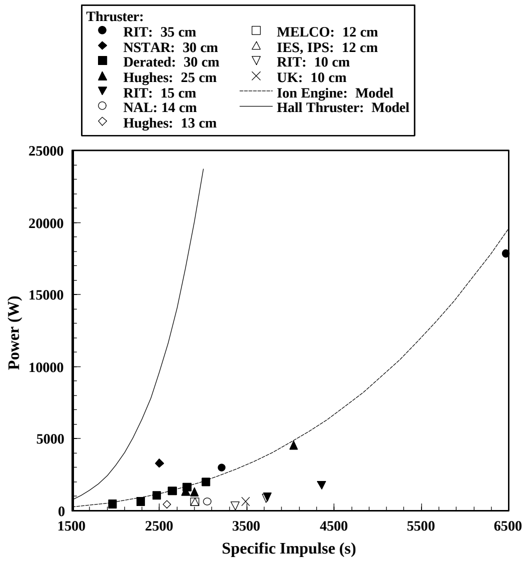

Figure 2: Thruster Power versus Specific Impulse

By examining a range of thruster sizes and operating conditions,[9,10,11] we are able to make modeling predictions. One of the most important parameters for this study is the curve of thruster power in terms of the specific impulse (I_{sp}). As specific impulse increases, the discharge voltage will increase as well. The Hall thruster power is modeled using a polynomial fit. This relation is shown in Figure 2.

The thruster efficiency is modeled using the relationship:

$$\eta = \frac{a}{1 + \frac{b}{\left(g_o I_{sp}\right)^2}} \tag{1}$$

which is based on the ion thruster efficiency equation developed by Brophy.[12] This efficiency model is shown in Figure 3.

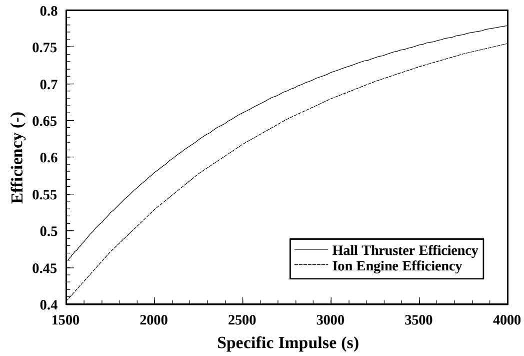

Figure 3: Thruster Efficiency versus Specific Impulse

In the gridded ion engine (see Figure 4), propellant is injected into an ionization chamber and ionized by electron bombardment. The propellant is then electrostatically accelerated through a series of biased grids. Traditionally, these grids have been molybdenum, though recent work has been done to develop carbon-carbon composite grids.

In the United States, ion engines were developed at NASA's Lewis Research Center in the late 1950's under the guidance of Dr. Harold Kaufman.[13] The original models used primarily mercury or cesium for propellants. Thruster development has continued at various levels, using thrusters with diameters ranging from 2.5 to 150 centimeters and power levels ranging from 50 W to 200 kW. Flight experiments have included SNAPSHOT, a US Air Force satellite that flew a cesium ion engine in 1965; SERT-2, a NASA satellite that flew a mercury ion engine in 1970; and ETS-3, a Japanese satellite that also flew a mercury ion engine in 1982.[14] In the 1980's, emphasis shifted to xenon and other noble gases because of concern over spacecraft contamination and environmental issues during ground testing. As part of its New Millennium program, NASA has been developing NASA's Solar Electric Propulsion Technology Application Readiness (NSTAR) engine for use on the first New Millennium mission, which includes an asteroid and comet fly-by. Other nations including the United Kingdom, Germany, and Japan are also currently developing ion engine technologies.[15] Hughes Space and Communication Co. has undertaken an extensive development effort and has baselined ion engines for North-South Station-keeping on it's next generation of communications satellites.[16]

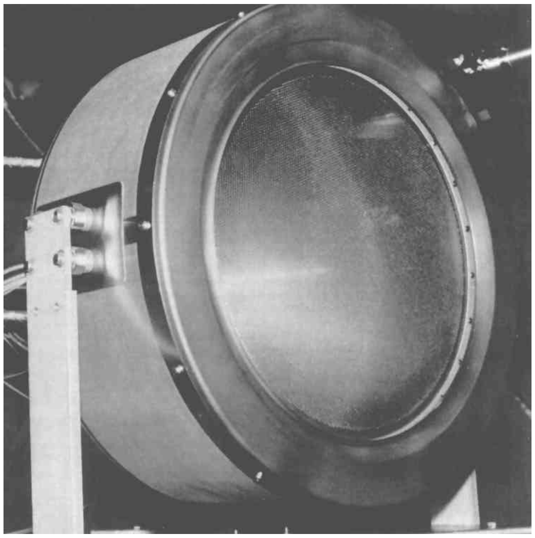

Figure 4: Ion Engine

In determining the thruster power curve, Figure 2, we took data from a wide range of ion engines.[15,16,17,18,19,20,21,22] Figure 2 shows two distinctly different trends for the ion engines depending on the diameter of the thruster. Thrusters larger than 20 cm follow a steeper power versus specific impulse curve than those smaller than 20 cm. These smaller thrusters are intended primarily for station-keeping missions. Therefore, since this study is concerned with orbit transfer missions, data from thrusters smaller than 20 cm is omitted.

Ion engine efficiency was modeled using the same form as the Hall thruster, and is shown in Figure 3.

One of the main concerns about both Hall thruster and ion engines for orbit transfer is lifetime. Lifetimes of up to 7400 hours (308 days) for the Hall thruster (SPT-100)[6] and 4350 hours (181 days) for the xenon ion engine[21] have been demonstrated, with an 8000 hour (333 days) test set to begin on the NSTAR ion engine.[18] The primary concern is thruster erosion (wall erosion for the SPT, guard ring erosion for the ALT, and grid erosion for the ion engine) which will become even more of an issue with higher power thrusters.

Of paramount importance for any solar electric propulsion vehicle is solar panel technology. For orbit transfer missions, concentrator arrays offer several advantages over conventional planar arrays.[14] Concentrator arrays use optics to focus solar radiation onto the solar cells, with concentration ratios up to 100:1. These optics provide inherent radiation shielding that reduces the need for a thick coverglass. This results in both greater radiation resistance and reduced cell area compared to conventional planar arrays. A concern of concentrator arrays is that they demand accurate 2-axis pointing, increasing the complexity of the orientation system.[23] Pointing accuracies of $\pm$ 3° are required as compared to $\pm$ 18° for planar arrays.[24] Projections for concentrator array systems indicate a specific power approaching 100 W/kg and a power to area ratio of well over 200 W/m^2, for technologies including gallium arsenide and multijunction arrays. Similar performance can be achieved with planar arrays, but with much greater radiation degradation.

MODEL:

The centerpiece of the model is the payload mass fraction equation based on the method developed by Messerole.[3] This method is derived by starting with an initial mass breakdown:

$$\begin{aligned} m_o &= m_{pl} + m_{pwr} + m_p + m_{tf} + m_{ps'} \\ &\quad + m_{att} + m_{adap} + m_{ss'} + m_{cont} + m_{rp} \end{aligned} \qquad (2)$$

where the terms are defined in Table 1.

Term	Name	Explanation
m_o	Initial Mass	Total System mass at beginning of E.P. transfer
m_{pl}	Payload Mass	Useful on-station mass
m_{pwr}	Power Mass	Mass of power dependent components (thrusters, power processors, solar arrays, and radiators)
m_p	Propellant Mass	Mass of xenon used for transfer
m_{tf}	Tank and Feedsystem Mass	Mass of fuel tank and associated components
$m_{ps'}$	Primary	Mass of satellite's

	Structure Mass	major structural components
m_{att}	Attitude Control Mass	Mass of attitude control system
m_{adap}	Adapter Mass	Mass of propulsion to satellite adapter
$m_{ss'}$	Secondary Structure Mass	Mass of power related system structures
m_{cont}	Contingency Mass	Contingency mass for power related systems
m_{rp}	Radiation Protection Mass	Mass of shielding to protect payload from radiation damage

Table 1: Mass Breakdown

The masses are divided by the total initial system mass to obtain mass fractions. The propellant tank and feedsystem mass fraction is calculated by dividing by the mass of the fuel and is expressed as the tank and feedsystem fraction, f_{tf}. The primary structure mass, attitude control mass, and adapter mass are combined into one term, the primary structure fraction, f_{ps}. The secondary structure and contingency mass fractions are expressed in terms of the propulsion system dry mass (power mass plus tank and feedsystem mass) and are combined into the secondary structure fraction, f_{ss}. The radiation protection mass is expressed in terms of the payload mass using the radiation protection fraction, f_{rp}. Combining these terms, Equation 2 can be rewritten to express the payload fraction as:

$$\frac{m_{pl}}{m_o} = \frac{\left(1 - f_{ps}\right) - \left(1 + \left(1 + f_{ss}\right)\left(\frac{m_{pwr}}{m_p} + f_{tf}\right)\right)\frac{m_p}{m_o}}{1 + f_{rp}} \quad (3)$$

Other than the fractions, there are two terms in the payload fraction equation. The first term is the propellant mass fraction and the second is the power mass to propellant mass ratio. The propellant mass fraction is calculated from the rocket equation:[25]

$$\frac{m_p}{m_o} = 1 - e^{-\Delta V_{Tot}/g_o I_{sp}} \quad (4)$$

Where $\Delta V_{Tot} = \left(1 + f_{perf}\right)\Delta V$ and f_{perf} is an orbit transfer performance factor designed to account for thrust vector misalignment, off nominal thruster performance, and other contingencies.

The power mass to propellant mass ratio, (m_{pwr}/m_p), is calculated using the propellant mass flow rate to obtain:

$$\frac{m_{pwr}}{m_p} = \frac{m_{pwr}}{t_t \dot{m}_p} \quad (5)$$

Where t_t is the thrusting time, equal to the trip time (t) multiplied by the non-occulated transit percentage. It is taken to be 86%, which is the case for a 180 day LEO to GEO mission, but is not adjusted for changes in trip time or mission. Using the definition of specific impulse and efficiency, we can then rewrite Equation 5 as:

$$\frac{m_{pwr}}{m_p} = \frac{\left(g_o I_{sp}\right)^2}{2 t_t \eta_{ppu} \eta_t f_{ave} \alpha} \quad (6)$$

The term f_{ave} is the mission average power fraction which accounts for solar array degradation due to radiation damage. It is dependent on the solar array technology and trip time. For planar arrays, it is 0.70 for a 180 day transfer, for concentrator arrays, it is 0.97 for the same trip time.[3]

The overall specific power (for the power dependent components), α, is defined as:

$$\alpha = \frac{P_{B.O.L.}}{m_{pwr}} = \frac{P_{B.O.L.}}{m_a + m_{ppu} + m_t + m_{rad}} \quad (7)$$

Where $P_{B.O.L.}$ is the power at the beginning of the orbit transfer vehicle's life. Then we determine the specific power by inversely summing the component specific powers:

$$\alpha = \left(\alpha_a^{-1} + \alpha_{ppu}^{-1} + \alpha_t^{-1} + \alpha_{rad}^{-1}\right)^{-1} \quad (8)$$

Solar array specific power, α_a, is taken from data on concentrator array technologies to be 100 W/kg.[14]

For the thruster specific power, α_t, baseline conditions are taken from the SPT-100 and the

NSTAR ion engine. It is then possible to show that if thruster mass is proportional to mass flow rate and mass flow rate is invariant with respect to input power, the specific power varies according to the relation:[3]

$$\alpha_t = \alpha_{to} \frac{\eta_o}{\eta} \frac{I_{sp}^2}{I_{spo}^2} + \alpha^* \qquad (9)$$

The constants α_o and α^* are determined by curve fitting actual thruster data as shown in Figure 5. For the ion engine, only large (diameter ≥ 20 cm) ion engines are considered relevant. The optimal values found for these are summarized in Table 2.

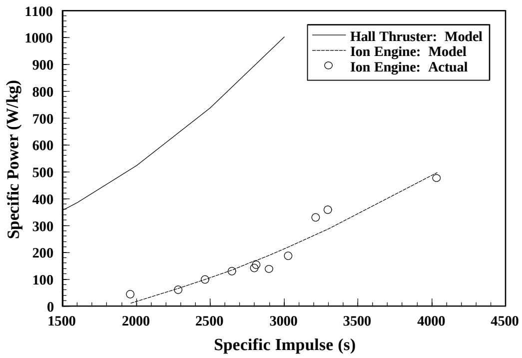

Figure 5: Thruster Specific Power versus Specific Impulse

Thruster	α_o	α^*	I_{spo}	η_o
Hall Thruster	385.7	0.0	1600	0.49
Ion Engine	347.0	-243.0	2500	0.62

Table 2: Thruster Specific Power Constants

For the power processing unit's specific power α_{ppu}, data was gathered from documented Hall thruster and ion engine PPUs.[6,18] Additionally, due to a lack of other high power (>3 kW) space qualified PPUs for xenon propellant systems, data from the US Air Force's 26 kW arcjet Electric Propulsion Space Experiment's (ESEX) PPU is included.[26] The ESEX PPU is a 1991 design and is considered a conservative estimate of future PPU specific power performance. The resulting specific power versus PPU input power (shown in Figure 6), was linearly curve fit with two straight lines. However, since power is an output quantity from this analysis, the recommended power per thruster is used to determine the PPU specific power by assuming one PPU per thruster.

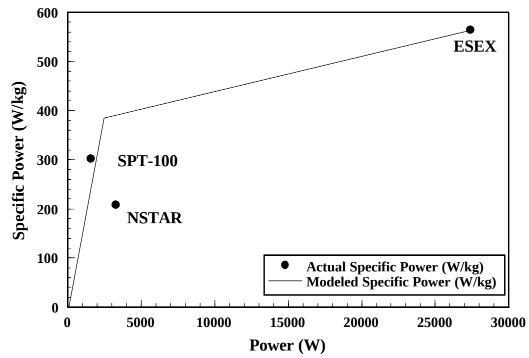

Figure 6: PPU Specific Power versus Input Power

Based on documented PPUs, the efficiency was set to 93% for the Hall thruster[6] and 90% for the ion engine.[17] The lower efficiency of the ion engine PPU is due to its more complex power needs.

Finally, for the radiator specific power, α_{rad}, a value of 32 W/kg is assumed for a system with a PPU of 92% efficiency.[17]

The required beginning of life input power for a given thruster system can be determined from the overall specific power, the power mass to propellant mass ratio, the propellant mass fraction, and the initial mass:

$$P_{B.O.L.} = m_o \frac{\left(g_o I_{sp}\right)^2}{2 t_t \eta_{pcu} \eta_t f_{ave}} \left(1 - e^{-\Delta V_{Tot}/g_o I_{sp}}\right) \qquad (10)$$

MISSIONS:

Four orbit raising missions are considered in this paper:

1. Low earth orbit (LEO) to geosynchronous earth orbit (GEO) all-electric propulsion transfer

2. LEO to LEO (higher) orbit transfer for constellations of LEO satellites
3. Geostationary orbit insertion with partial chemical propulsion (orbit topping)
4. Reusable Orbit Transfer Vehicle (ROTV) for LEO to GEO transfer

These missions represent a cross section of future orbit transfer missions for which electric propulsion is a viable option. High I_{sp} and ΔV missions involving interplanetary transit are not considered since they do not fall with the US Air Force's current mission parameters.

Though not a direct part of the orbit transfer mission, the launch vehicle makes a major impact on mission parameters, especially with regards to launch mass and payload faring size. The launch vehicles considered in this study were the McDonnell Douglas Delta II (7920), the General Dynamics Atlas IIAS, and the Lockheed Martin Titan IV with Hercules Solid Rocket Motor Upgrade (SRMU). Launches to LEO were not to the traditionally used 100 nautical mile (185 km), 28º circular orbit. This was due to the fact that the large solar arrays necessary for electric propulsion OTVs coupled with their low thrust create a thrust to drag ratio dangerously close to one. It was found however, that by raising the launch altitude to 300 km, approximately 98% of the LEO mass could be retained while raising the thrust to drag ratio to almost ten. In all cases, the largest payload faring available was used with the booster to insure that maximum space was provided for large components such as solar arrays. With these considerations, the masses inserted into the 300 km LEO orbits are summarized in Table 3.

Booster	Launch Mass
Delta II	4915 kg
Atlas IIAS	8300 kg
Titan IV	21150 kg

Table 3: LEO Masses[27]

Thruster sizing is a major concern of satellite manufacturers. Due to the existence of a parametric series of Hall thrusters, it is possible to derive a sizing relation with respect to specific impulse based on the thrusters' characteristics. This relation is shown in Figure 7 with existing and planned thrusters identified. However, the physical dimensions of the ion engine are not as sensitive to changes in specific impulse (and thus power) as the Hall thruster. Therefore, all ion thrusters are assumed to be 30 cm in this study. Thirty centimeters is the average size of the large ion engines considered in this study.

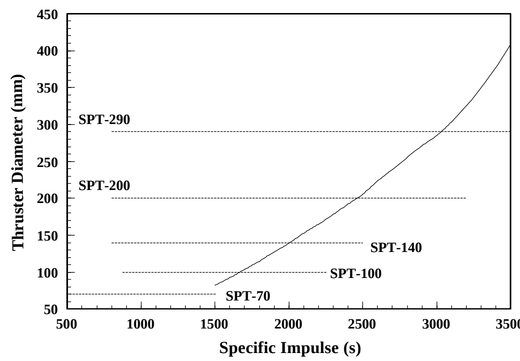

Figure 7: Hall Thruster Diameter versus Specific Impulse

In this analysis, ΔV is calculated using the first order approximation:

$$\Delta V = \sqrt{V_o^2 + V_f^2 - 2V_o V_f \cos\varphi} \quad (11)$$

The effect of gravity on ΔV, due to the longer electric propulsion trip times is ignored in this analysis. In other cases, such as the orbit topping mission, ΔVs had previously been calculated using sophisticated orbit transfer codes like SECKSPOT[28] which include the gravity ΔV. These previously calculated ΔV's were manually entered for analysis. In no cases were the two ΔV methods mixed.

The mission parameters (ΔV and mass fractions) for the four missions examined are shown in Table 4.

LEO-GEO Transfer:

The first mission examined was a basic LEO-GEO orbit transfer. This mission is applicable not only to communications satellites, but also to proposed observation and reconnaissance missions. Transit times of 180, 270, and 360 days were examined for launches on all three launch vehicles. The values used in the payload fraction equations are shown in Table 4.

Key parameters determined are the payload mass fraction, the array output power required, and the number of thrusters necessary. We note that payload mass fraction is independent of initial mass and thus invariant with launch vehicle. However, the power and number of thrusters is strongly dependent on the initial mass launched into LEO.

It is clear that no single parameter determines the best thruster configuration for any given mission.

Parameter	LEO-GEO	LEO Constellation	Orbit Topping	Reusable OTV
ΔV	5233 m/s	276.8 m/s	Varies	5233 m/s
f_{perf}	0.01	0.01	0.01	0.01
f_{ps}	0.06	0.06	0.06	0.20
f_{ss}	0.10	0.10	0.10	0.10
f_{tf}	0.12	0.12	0.12	0.12
f_{rp}	0.03 (t/180)	0.20	0.03 (t/180)	0.03 (t/180)
f_{ave}	1-0.03 (t/180)	0.97	1-0.03 (t/180): $t \geq 180$ 0.97: $t < 180$	$\prod_{i=1}^{n} f_{avei}$ n = transfer number
f_{avei}				1-0.03 (t/180)

Table 4: Mission Parameters

It must be a combination of critical factors including trip time, payload mass fraction, power required, and number of thrusters.

Figures 8 through 10 show that for LEO-GEO transfers Hall thrusters deliver a higher maximum payload fraction than ion engines. This is driven by the fact that ion engine systems have a lower overall specific power compared to Hall thruster systems. From Equation 6, we see that this gives a higher power system mass to propellant mass ratio, resulting in a lower payload fraction from Equation 3. The major factor driving the overall specific power differential is the lower specific power of the ion engine itself. This is clearly seen from Figure 5. A secondary effect is the lower power per thruster for the ion engine (as seen from Figure 2), resulting in ion engine PPUs having lower specific powers, from Figure 6, for similar specific impulses. Comparing Figures 8 through 10, we note that increasing trip time decreases the differential between Hall thrusters and ion engines. The decrease results from Equation 6, where increasing trip time lessens the overall effect of specific power on payload fraction.

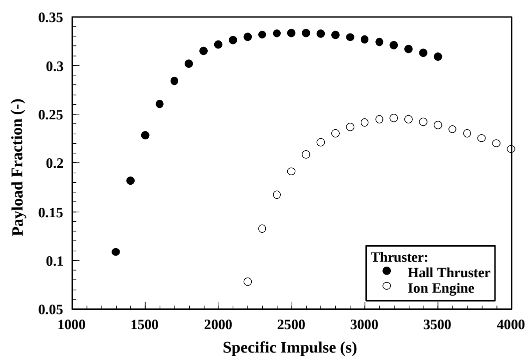

Figure 8: Payload Mass Fraction versus Specific Impulse, Trip Time = **180 Days**

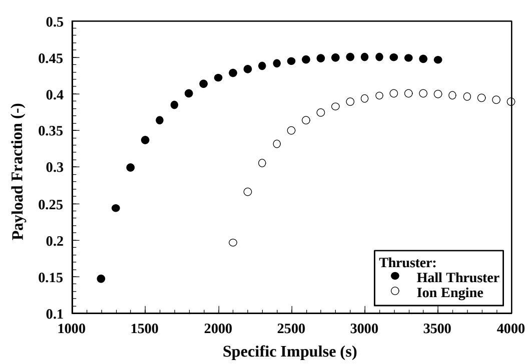

Figure 9: Payload Mass Fraction versus Specific Impulse, Trip Time = **270 Days**

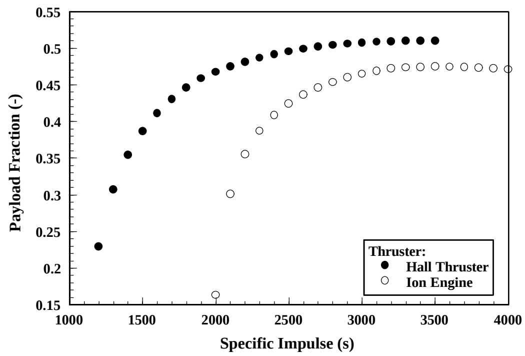

Figure 10: Payload Mass Fraction versus Specific Impulse, Trip Time = **360 Days**

Figure 8 shows a peak in the payload fraction. This peak occurs for all trip times, but as trip time is increased, it occurs at progressively higher specific impulses. The peak can be explained as follows. As specific impulse is increased, we see from Equation 4 that the propellant fraction decreases. Looking at the power mass/propellant mass term in Equation 6, we see that the numerator increases with the square of the specific impulse. The terms in the denominator dependent on the specific impulse, the specific power and the thruster efficiency, also increase with specific impulse, but not as rapidly as the numerator (except at low specific impulses, where the efficiency term as seen in Figure 3 increases rapidly). Thus, the power mass/propellant mass term increases with increasing specific impulse. As specific impulse is increased, the increasing power mass/propellant mass term is countered by the decreasing propellant mass fraction in Equation 3, resulting in the initial rise in payload fraction versus specific impulse that we see in Figure 8. However, we note from Equation 4 that the rate of decrease of the propellant fraction slows with increasing specific impulse. Therefore, eventually the increase in power mass/propellant mass will overcome decreases in propellant fraction, resulting in the eventual decrease in payload fraction that is also observed in Figure 8. To summarize, increases in power system mass fraction with increasing specific impulse begin to dominate the system, eliminating room for payload.

Next, we look at required power versus specific impulse as shown in Figures 11 through 13.

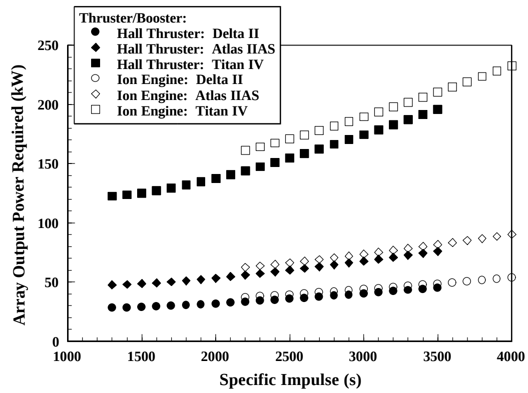

Figure 11: Required Power versus Specific Impulse: Trip Time = **180 Days**

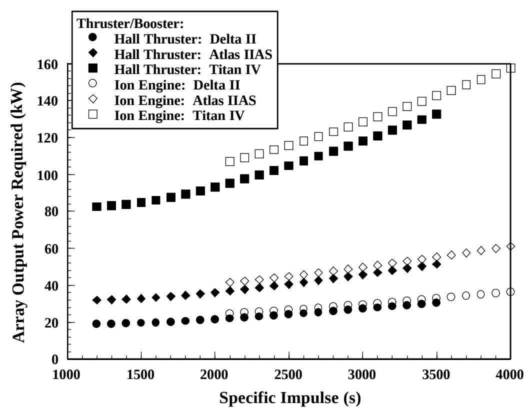

Figure 12: Required Power versus Specific Impulse: Trip Time = **270 Days**

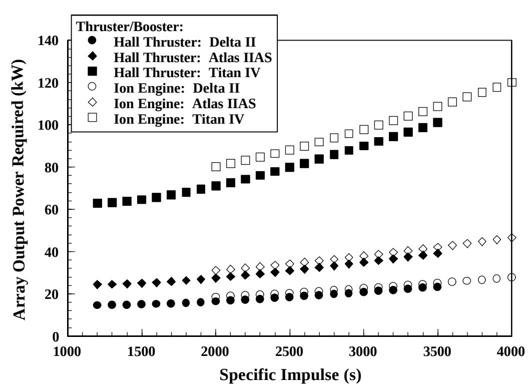

Figure 13: Required Power versus Specific Impulse: Trip Time = **360 Days**

For a given specific impulse, trip time, and payload the power levels are very similar between Hall thruster and ion engine systems, as expected from Equation 10, since the only differences are the thruster and PPU efficiencies. However, when comparing power levels for equal payload fractions, Hall thruster systems require significantly lower power. Given projected trends in GEO satellite power levels, it appears from figures 11 through 13 that transfers of Delta II and Atlas IIAS payloads with trip times in the 270 to 360 day range should be feasible within the next 15 years. However, it appears that transfers of Titan IV class payloads will be impractical for some time to come.

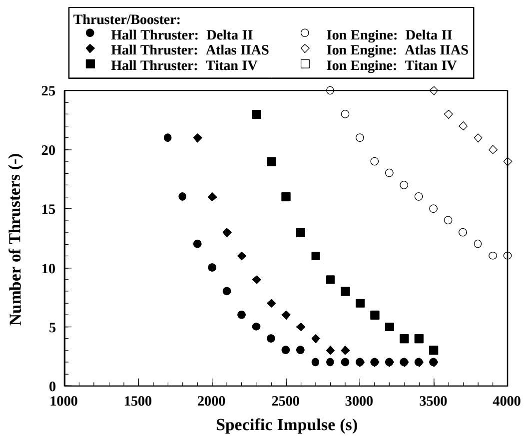

Figure 14: Number of Thrusters versus Specific Impulse: Trip Time = **180 Days**

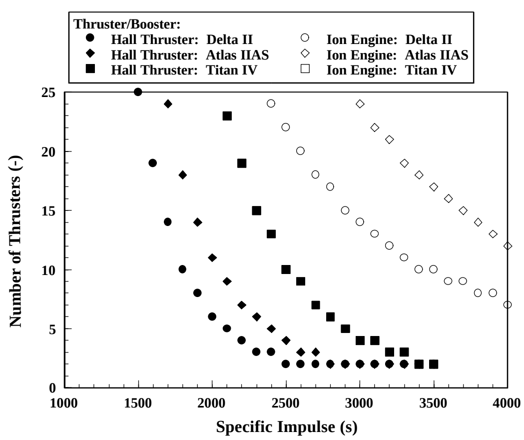

Figure 15: Number of Thrusters versus Specific Impulse: Trip Time = **270 Days**

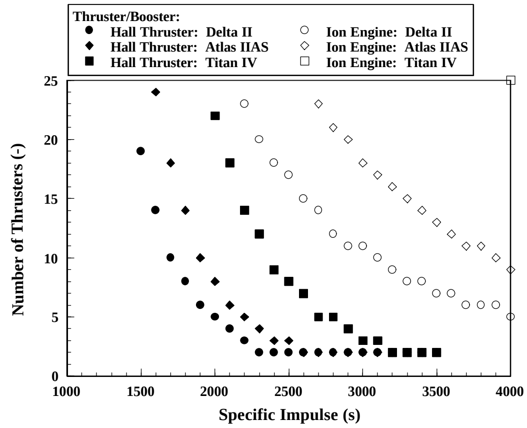

Figure 16: Number of Thrusters versus Specific Impulse: Trip Time = **360 Days**

Next, we look at the number of thrusters, operating at nominal conditions, necessary to accomplish the orbit transfer. From Figures 11 through 13, we see that the total power for a mission goes up with increasing specific impulse. However, examining Figure 2, we see that the power capacity per thruster also increases, and at a higher rate. Therefore, the number of thrusters, determined by dividing the total power by the power per thruster, decreases with increasing specific impulse. For redundancy purposes, we never consider fewer than two thrusters.

From Figures 14 through 16, we note that at lower specific impulses an impractical number of thrusters is required. As specific impulse is increased, the number of Hall thrusters necessary drops to realistic levels (less than 10) in all cases. Ion engines, however, only reach the ten thruster cutoff for the 270 and 360 day Delta II missions and the 360 day Atlas IIAS missions. The Titan IV mission does not appear in Figures 14 and 15 since the minimum number of ion engines is 50 and 32 respectively. Ion engine systems require higher power and, more importantly, ion engines increase in power capacity per thruster more slowly than Hall thrusters, as can be seen from Figure 2.

Faring volumetric constraints of the propulsion system are anticipated to be another major concern of satellite manufacturers. LEO-GEO transfers require a large number of present generation thrusters. For example, a Delta II launch with an 180 day trip time delivers maximum payload using three 210 mm (approximate) Hall thrusters operating at 2500 s specific impulse and 35.6 kW of total power. This results in a total thruster area of approximately 0.26 m^2. By comparison, maximum payload for an ion engine system is obtained from eighteen thrusters (30 cm diameter), operating at 3200 s specific impulse and 45.5 kW total power. This is 3.16 m^2 of thruster area, or over twelve times the area of a Hall thruster system. A more practical system would probably have four larger ion thrusters, of approximately 64 cm each (the equivalent grid area of the eighteen smaller thrusters), however these larger thrusters may have significantly different operating conditions.

It appears in general that Hall thrusters are the best choice for LEO-GEO transfers. Ignoring Titan IV payloads, Hall thrusters in the specific impulse range of 2000 to 3000 s can deliver high payload fractions at power levels that are consistent with future trends for GEO satellites, and can do so with a realistic number of thrusters.

The true benefit of electric propulsion becomes even clearer when it is compared to chemical propulsion systems. Looking at the maximum payload deliverable to GEO and payload fraction for the Hall thruster cases examined and for all chemical systems in Table 5, we see large increases in payload capacity. This increase in payload capacity gives three major benefits. First, and most obvious, it permits increases in useful payload delivered to orbit. This increased payload could result in increased satellite hardware or additional propellant for station-keeping to lengthen the life of the satellite. Second, increases in payload capacity will relax constraints on satellite design, allowing greater flexibility. Third, and perhaps most importantly, it allows for launch vehicle downsizing in some cases, as can be seen for 270 and 360 day transfers for the Delta II compared to the chemical Atlas IIAS. The obvious deterrent to these systems is the trip time required.

System	Delta II	Atlas IIAS	Titan IV
Chemical[27]	900 kg 18.3%	2100 kg 22.6%	4550 kg 21.5%
180 Day EP	1637 kg	2766 kg	7047 kg
33.3%			
270 Day EP	2214 kg	3739 kg	9528 kg
45.0%			
360 Day EP	2506 kg	4232 kg	10784 kg
51.0%			

Table 5: Chemical and Electric Propulsion GEO Masses

LEO Constellations:

The second case is representative of transfers needed for LEO constellations. The particular case studied here is a space based observation system. It is based on the use of a large constellation (1148 satellites) in a Walker orbit at an altitude of 500 miles for theater level reconnaissance as discussed by Fiedler and Preiss.[29] Though this concept was not recommended in their study due to the cost of such a large number of satellites, the operating parameters nevertheless provide an excellent reference point for other potential LEO satellite constellations. For this case, we use a ΔV of 276.8 m/s, which is the ΔV from 162 to 500 miles. This mission ΔV does not include any inclination change or repositioning during the satellite's lifetime. Here, trip times of 180, 90, and 30 days are considered using each of the three launch vehicles.

There are two other parameter changes for this mission. The first is that since these satellites will be operating in a much more radiationally intense orbit than geosynchronous satellites, the radiation protection fraction, f_{rp} (the mass of the radiation shielding necessary to protect the payload, expressed in terms of the mass of the payload), is set to 0.20 as compared to 0.03 to 0.06 for the LEO-GEO case. Second, the average power fraction is maintained at the 180 day level, which is 0.97 for all missions. The parameters used are summarized in Table 4.

In proceeding to the analysis of LEO constellations, one notes that the same quantities are of interest as for the LEO-GEO transfer case.

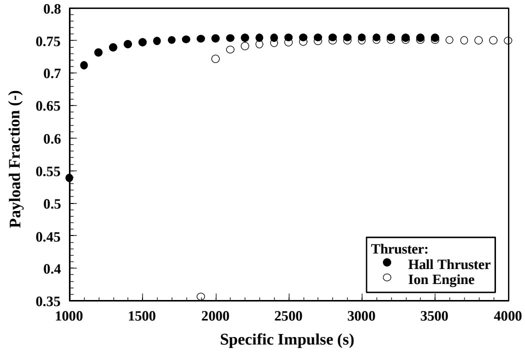

Figure 17: Payload Mass Fraction versus Specific Impulse, Trip Time = **180 Days**

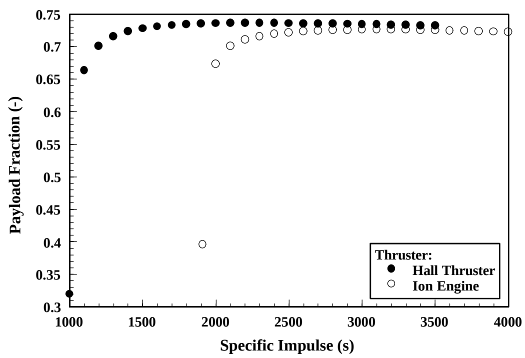

Figure 18: Payload Mass Fraction versus Specific Impulse, Trip Time = **90 Days**

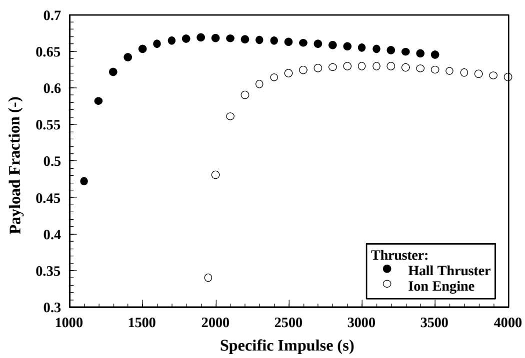

Figure 19: Payload Mass Fraction versus Specific Impulse, Trip Time = **30 Days**

Looking at Figure 17, we see that for 180 day trip times, Hall thrusters and ion engines are nearly equal with regard to payload fraction. As trip time is decreases, we note from Figures 18 and 19 that the Hall thruster delivers a higher payload fraction. The reasons for this behavior is the same as in the LEO to GEO case. The higher payload fractions compared to Figure 8 result from reductions in propellant fraction at the lower ΔV. For Hall thrusters of higher than 1500 s specific impulse, payload fractions greater than 0.65 can be achieved at trip times as short as 30 days, assuming adequate power is available.

Examining the power requirements for LEO constellation transit shown in Figures 20 through 22, we see that they are much lower than those for LEO-GEO transfers with the same initial mass. However, given the lower power levels expected for LEO constellations, Atlas transfers of 30 days and Titan transfers of 90 days appear to be at the upper boundary of what is likely to be possible in the next 10 to 15 years.

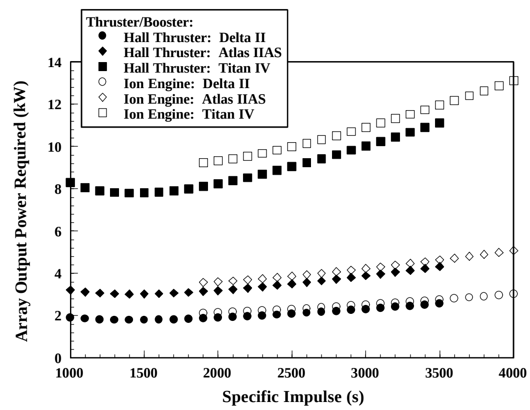

Figure 20: Required Power versus Specific Impulse: Trip Time = **180 Days**

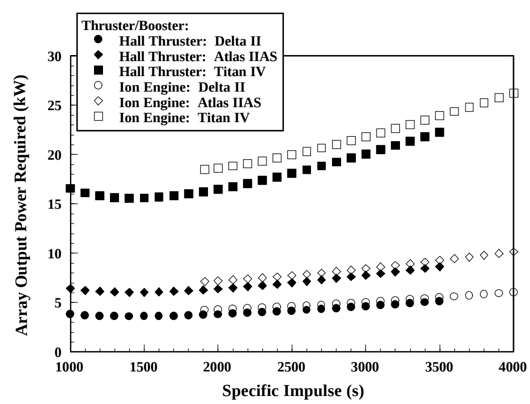

Figure 21: Required Power versus Specific Impulse: Trip Time = **90 Days**

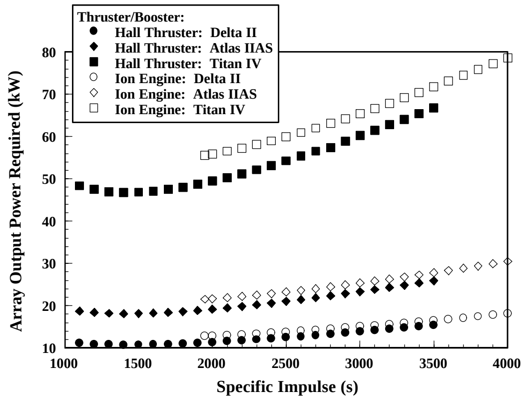

Figure 22: Required Power versus Specific Impulse: Trip Time = **30 Days**

The number of thrusters reduces similarly to the LEO-GEO case, but due to the smaller amount of thrust necessary, there are fewer needed. The only case where a system with fewer than 10 thrusters is not possible is the 30 day transfer of a Titan IV class payload using ion engines.

Orbit Topping:

One of the most intriguing applications of electric propulsion is orbit topping. Proposed by Free[30] and further studied by Oleson, et al.[28] and Spitzer,[31] the concept involves performing an initial portion of a LEO-GEO transfer using chemical propulsion, with the final GEO insertion done using electric propulsion. The primary advantage of this approach is that it allows a significant propellant mass savings over an all-chemical transfer without the long trip times of an all-electric transfer, which translates into a payload increase.

In the work of Oleson, et al., the SECKSPOT orbit transfer code was used to optimize a launch using an Atlas IIAS with a Centaur impulsive chemical stage. A combination of on-board chemical propulsion and electric propulsion is then used to insert the satellite into a geostationary orbit. In their study, the amount of on board chemical ΔV was decreased incrementally while the amount of electric propulsion ΔV was increased (the Centaur stage is the same in all cases). They considered several thrusters (arcjets, Hall thrusters, and ion engines) and two power levels, and found that the ΔV's for the transfer were approximately the same for all cases. The study indicated that the greatest mass gains could be made using an ion engine (2.5 kW NSTAR), with a stationary plasma thruster (1.5 kW SPT-100) coming in close behind. It was noted in Reference 28, however, that a more appropriately powered SPT (2.5 kW) may have increased benefits.

In our study, thrusters of equal power levels were compared against each other. Power levels up to 5 kW per thruster were studied here, compared to 1.5 kW Hall thrusters and 2.5 kW ion engines examined by Oleson, et al. The higher specific power devices investigated in this paper illustrate even greater advantages for electric propulsion orbit topping than those shown by Oleson, et al.

Using the electric propulsion starting conditions from the work of Oleson, et al., side-by-side comparisons of Hall thrusters and ion engines operating at 1.66, 2.5, and 5.0 kW were made for missions with 10 and 15 kW of total power. These total power levels determine the number of thrusters used for each case. Operating conditions were the same as the standard LEO-GEO transfer, with ΔV based on the SECKSPOT calculated orbits. The trip time and payload are calculated by decreasing trip time until the mission power level is reached. Array degradation was determined through the average power fraction, f_{ave} which was maintained at 0.97 for trip times of under 180 days and increased for trip times greater than 180 days using the same form as for LEO-GEO transfers. These fractions are summarized in Table 4.

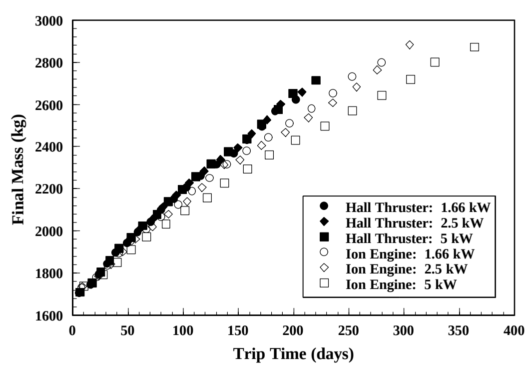

Figure 23: Final Mass versus Trip Time: Power = **10 kW**

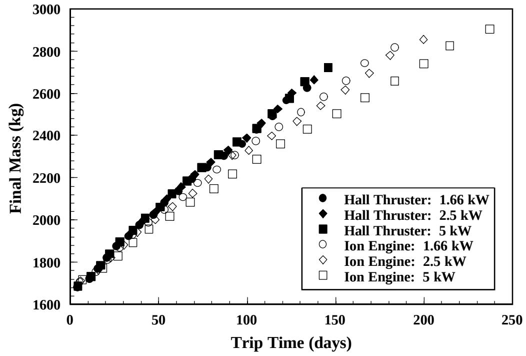

Figure 24: Final Mass versus Trip Time: Power = **15 kW**

Since power and number of thrusters are a given for orbit topping, the only pertinent results are payload and cost. Data is presented in the same way as Oleson, et al., with final mass plotted versus trip time in Figures 23 and 24. Final mass is defined as useful on-orbit mass including payload, power systems, structure, and attitude control. It also includes a mass penalty for array degradation (i.e., 1-f_{ave} multiplied by the mass of the array). Each data point represents an individual ΔV case.

The orbit topping scenario is power limited by fixing the amount of power available for the propulsion system. This differs from the two previous cases which were trip-time limited. By allowing trip time to vary, we see from Figures 23 and 24 that for a given amount of ΔV, ion engines can deliver more payload from the intermediate orbit to GEO, however they require more time to do so. The additional payload and longer trip time results from the fact that at a given input power, the ion engine has a higher specific impulse. From Equation 4 and other basic rocket equations, we see that the higher specific impulse results in lower propellant mass (and thus higher payload mass) and longer trip times. Additionally, in both Figures 23 and 24, there is a much larger spread between the power levels for the various ion engines than for the Hall thrusters. This spread is due to the larger increase in specific impulse in going from a 1.66 kW ion engine to a 5 kW ion engine, compared to the same change in power level for the Hall thrusters. Increasing power from 10 kW to 15 kW does not significantly affect the final mass delivered, but it does reduce trip time by approximately 33%.

We note that for trip times on the order of satellite check-out periods (~30 days) payload gains of over 100 kg can be achieved. For comparison, Oleson, et al. calculated that a system using all chemical propulsion for orbit transfer would have a mass of 1723 kg if it used SPTs for North-South Station Keeping (NSSK) and 1748 kg using ion engines.

The choice of thruster type for this mission will depend on the requirements of the user. For minimum trip time on a given mission (ΔV case), 1.66 kW SPTs or other low specific impulse Hall thrusters would be the best choice. However, if the user wants to maximize payload for a given mission, 5 kW or higher power ion engines would be the best thruster.

Reusable Orbit Transfer Vehicles:

The final mission examined in this study is a reusable orbit transfer vehicle (ROTV). The particular concept for ROTV studied in this paper is a modification of that proposed as part of a recent US Air Force study.[32] In that concept, there were two modules: a reusable power system composed of the solar arrays, bus, and docking module; and an expendable propulsion system, launched with the payload, comprised of thrusters (arcjets, resistojets, or ion engines), propellant tank, and power processing unit. In our study, the thrusters were changed to either Hall thrusters or ion engines, and the PPU was moved to the power system along with the thermal radiator.

For this study, it is assumed that the propulsion system/payload is launched on a Delta II, using the full payload capability. The power system is also launched on a Delta II, but due to the mass breakdowns of the system, the payload capacity of the second Delta II will not necessarily be used by the power system (the unused payload capability can be used in some cases for a second power system module, or for auxiliary payloads). The actual mass launched on the second Delta II is determined as a function of the trip time and the specific impulse of the thruster used.

The thrusters examined are Hall thrusters with specific impulses of 1600, 1900, 2200, and 2500 seconds and corresponding power levels from Figure 2. The corresponding ion engine specific impulses were determined by matching the thruster input powers to those of the Hall thrusters.

Missions where the power system is used for up to five complete transfers from LEO to GEO and back were studied. Initial trip times of 180, 270, and 360 days each way were examined. Subsequent transfers will take more time due to solar array degradation.

The parameters for this mission are summarized in Table 4. There are two differences between the ROTV and standard LEO-GEO transfer. First, due to the requirement of on-orbit coupling and decoupling of the power and propulsion systems, the adapter mass must be increased. There are three docking modules necessary in this concept: one to separate the payload and one each on the power system and propulsion system to dock the two together. It is assumed that each module comprises 5% of the total vehicle mass. This raises the primary structure fraction to 0.20 as compared to 0.06 for the other missions examined. Additionally, since the power system makes multiple trips through the Van Allen belts, the array degradation is cumulative. Therefore, f_{ave} is determined by multiplying the individual transfer power fractions (f_{avei}).

For the reusable orbit transfer scenario, the initial masses were computed by determining the mass breakout of the system between the expendable payload and propulsion system and the reusable power system. The ratio of expendable component mass to reusable component mass was lower for the ion engine than for the Hall thruster. Since it was assumed that the expendable/payload launcher is a full capacity Delta II, this results in a more massive reusable power system for the ion engine and thus a larger overall starting mass in LEO.

The mass transported to GEO includes both the satellite payload and the propellant for the return trip. Satellite payload fractions are shown in Table 6 for the thruster configurations studied, both Hall thrusters and ion engines. These fractions represent the satellite payload mass (not including the propellant for the return trip), divided by the overall starting mass in LEO.

Cases with no payload listed are ones for which the payload capacity was insufficient to transport enough fuel to return the ROTV in the specified trip time even with zero payload.

Thruster	Initial Trip Time (days)	Satellite Mass Percentage
Hall Thrusters		
1600 s, 1.04 kW	180	---
	270	3.56 %
	360	10.5 %
1900 s, 2.45 kW	180	---
	270	12.5 %
	360	18.7 %
2200 s, 5.08 kW	180	2.95 %
	270	16.6 %
	360	22.9 %
2500 s, 9.59 kW	180	5.09 %
	270	19.1 %
	360	25.7 %
Ion Engines		
1930 s, 1.04 kW	180	---
	270	---
	360	1.22 %
3195 s, 2.45 kW	180	---
	270	17.3 %
	360	25.7 %
4100 s, 5.08 kW	180	---
	270	18.2 %
	360	27.5 %
5092 s, 9.59 kW	180	---
	270	15.8 %
	360	26.5 %

Table 6: ROTV Delivered Satellite Masses

The other critical parameter is the power required for these cases. Due to solar array degradation, the available power will be highest for the first transit; subsequent trips will have less power available. Since payload is considered constant, the trip time must therefore increase for subsequent round trips. The initial power required is shown in Table 7 and the trip time increase is shown in Figure 25. (We note that for any given round trip, the downward leg can be made in the same amount of time or less than the upward, since the mass transported has been decreased by the mass of the payload and the fuel used in the upward trip. Therefore, the power required is lower. This analysis assumes that the downward trip time is the same as the upward leg.)

Thruster	Initial Trip Time (days)	Power Required
Hall Thrusters		
1600 s, 1.04 kW	180	---
	270	27.8 kW
	360	19.7 kW
1900 s, 2.45 kW	180	---
	270	29.1 kW
	360	20.7 kW
2200 s, 5.08 kW	180	55.3 kW
	270	31.7 kW
	360	22.4 kW
2500 s, 9.59 kW	180	61.8 kW
	270	34.8 kW
	360	24.5 kW
Ion Engines		
1930 s, 1.04 kW	180	---
	270	---
	360	25.3 kW
3195 s, 2.45 kW	180	---
	270	49.1 kW
	360	33.4 kW
4100 s, 5.08 kW	180	---
	270	64.4 kW
	360	42.6 kW
5092 s, 9.59 kW	180	---
	270	86.9 kW
	360	55.1 kW

Table 7: ROTV First Transit Array Output Power

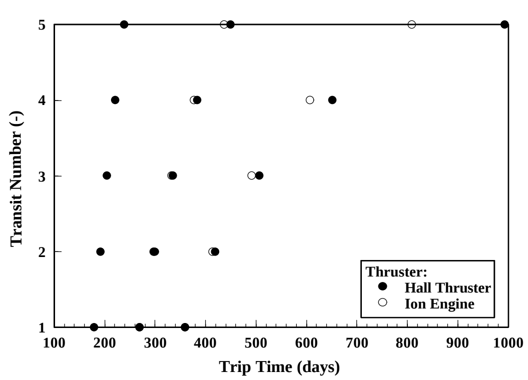

Figure 25: ROTV Trip Times with Transit Number

From Table 6 we see that for equivalent thruster power and trip time, the ion engine delivers a higher satellite payload fraction than the Hall thrusters at high specific impulses. The overall mass fraction delivered to GEO (satellite plus return trip fuel) is lower for the ion engine, but since it operates at higher specific impulses it requires a lower propellant fraction for the return trip, allowing a higher satellite payload fraction. In Table 7, we see that for equivalent cases the power required for the initial round trip is much higher for the ion engine than for the Hall thruster, due to their higher specific impulse and initial payload. Since the power per thruster is fixed, this means that ion engine systems will require more thrusters (assuming present day designs) than Hall thruster systems. Finally, from Figure 25 we see that the trip time increase for subsequent transfers is more substantial for the Hall thruster than for the ion engine.

Comparing the payload fraction data in Table 6 to the standard LEO-GEO transfer as shown in Figure 10, payload capability is less for ROTV's because of the need to carry propellant for the return trip. Figure 25 shows that in spite of the use of highly resilient concentrator arrays, for initial trip times greater than 180 days, array degradation quickly drives the trip time to unacceptable levels for round trip number two and beyond.

CONCLUSIONS:

The analysis in this study shows that if the trip time for a mission is fixed, then Hall thrusters can deliver higher payload fractions, due to their higher specific power. If, however, the power for a mission is fixed and trip time is allowed to vary, ion engines can deliver greater payload since they typically operate at higher specific impulses.

Examining all four missions and taking into account power and trip time requirements, the mission that seems most practical at this time would be orbit topping. Significant payload gains could be made for 10 or 15 kW systems with trip times of the order of 30 days, which is approximately the on-orbit check-out time for most satellites. Also practical in the near term would be small orbit transfers for LEO constellations, since they offer high payload fractions with short trip times.

A full, all electric propulsion, LEO-GEO transfer is practical using today's technology, as long as the user is willing to accept the long transfer times. However, by performing the shorter term orbit topping and LEO constellation type missions, the user can be introduced to the benefits of electric propulsion transfers without immediately suffering the large trip time penalties.

The same comments apply to an even greater extent to reusable orbit transfer vehicles. Though there can be payload benefits compared to chemical systems, the trip time penalties are much more severe than expendable LEO-GEO systems. Other issues to consider for ROTVs include: increased power requirement and logistical concerns (autonomous control is probably necessary for a practical system). These combine to make ROTVs the most long term of the missions examined here.

For all of these missions, there are concerns with regards to thruster size and payload faring volume. This is especially true for ion engine systems since they require a higher number of thrusters and ion engines are typically physically larger than Hall thrusters.

It is important to note that all of the analysis in the paper is based on the current generation of ion engines and Hall thrusters. Future thrusters that operate at higher specific powers and thruster power to specific impulse ratios, could increase system performance. Work has already been done in this area, such as the 50 cm and larger ion engines that have been tested by NASA[33,34] and Hall thruster designs such as the SPT-N series.[11] Advanced thrusters such as these also offer other improvements including higher efficiency and lower beam divergence.

Improvements in other areas can also improve electric propulsion capability. Solar array technology continues to improve, with increases in specific power and material density that will make performance improvements for both Hall thrusters and ion engine systems possible. Improvements in the cryogenic storage of xenon can benefit both types of thruster by reducing the tankage fraction.

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