Electric Propulsion Upper-Stage for Launch Vehicle Capability Enhancement

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The NASA In-Space Propulsion Technology Project Office initiated a preliminary study to evaluate the performance benefits of a solar electric propulsion (SEP) upper-stage with existing and near-term small launch vehicles. The analysis included circular and elliptical Low Earth Orbit (LEO) to Geosynchronous Earth Orbit (GEO) transfers, and LEO to Low Lunar Orbit (LLO) applications. SEP subsystem options included state-of-the-art and near-term solar arrays and electric thrusters. In-depth evaluations of the Aerojet BPT-4000 Hall thruster and NEXT gridded ion engine were conducted to compare performance, cost and revenue potential. Preliminary results indicate that Hall thruster technology is favored for low-cost, low power SEP stages, while gridded-ion engines are favored for higher power SEP systems unfettered by transfer time constraints. A low-cost point design is presented that details one possible stage configuration and outlines system limitations, in particular fairing volume constraints. The results demonstrate mission enhancements to large and medium class launch vehicles, and mission enabling performance when SEP system upper stages are mounted to low-cost launchers such as the Minotaur and Falcon 1. Study results indicate the potential use of SEP upper stages to double GEO payload mass capability and to possibly enable launch on demand capability for GEO assets. Transition from government to commercial applications, with associated cost/benefit analysis, has also been assessed. The sensitivity of system performance to specific impulse, array power, thruster size, and component costs are also discussed.

Nomenclature

g	=	gravitation acceleration
I_{sp}	=	specific impulse
$\dot{m}$	=	propellant flow rate
P	=	power
ΔV	=	change in velocity
η	=	efficiency
dt	=	time step
r	=	rate of return
t	=	time
T	=	total time

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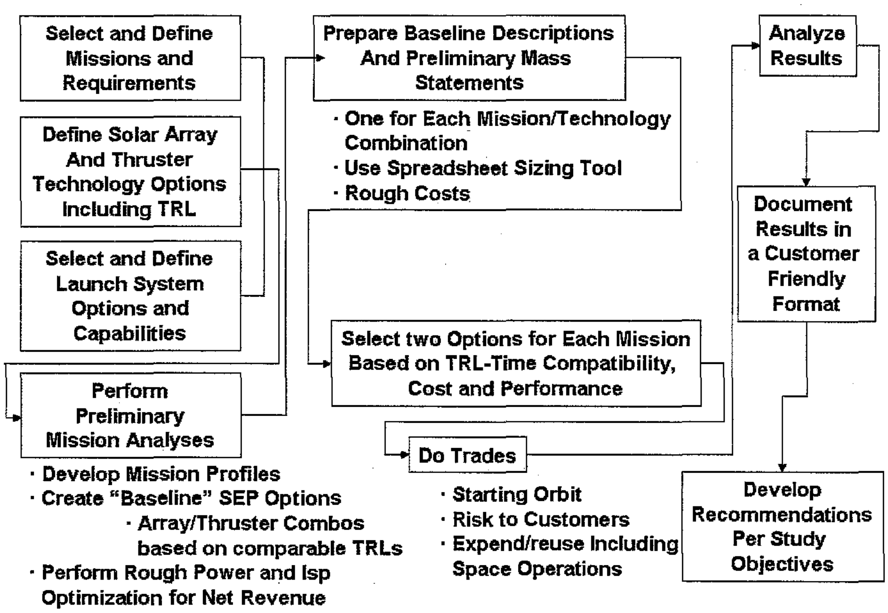
INTRODUCTION

Solar Electric Propulsion (SEP) is becoming a common in-space propulsion option for near-Earth applications. Kilowatt-class arcjet, Hall, and gridded ion propulsion systems are now routinely used for orbit maintenance, while higher power SEP systems are planned for orbit topping and orbit transfers. As the risk of using electric propulsion decreases and the available power increases, the practical use of SEP for more ambitious missions becomes increasingly attractive.

An electric propulsion upper stage study was conducted to assess the increased payload capability or possible decreased costs enabled through the use of an electric propulsion stage with existing and near-term launch vehicles. The objectives of the study included selecting and evaluating representative mission opportunities for NASA, DoD, NOAA, etc., to identify the key SEP attributes and required advancements in both propulsion system components and solar power technologies, to assess benefits of single use versus an orbiting asset tug, and provide a recommended development approach.

The approach for the study is shown in Figure 1. The first step included defining the mission requirements and collected the SOA and near-term technology performance capabilities. The study was specifically directed to optimize the stage performance for cost. For cost comparison, revenue was estimated for a commercial satellite and a net present value calculation was used to compare various stage configurations. The comparisons were then completed for various launch vehicles.



PERFORMANCE REQUIREMENTS

In order to determine the SEP metrics best suited for each mission, the ΔV requirement for the mission must be known. The trajectory optimizer SECKSPOT[1] was used to calculate the velocity change for various starting orbits to the target GEO orbit. Because of the nature of SECKSPOT as a calculus of variations solver, the initial guesses for the co-state variables are critical to obtaining a converged solution. The optimizer was used for a variety of starting altitudes, eccentricities, and inclinations.

PERFORMANCE ASSUMPTIONS

With each case leading to a different start mass due to the launch vehicle performance, only a baseline case was run for the various starting orbits. The same is true for available power and occultation effects. While the start mass and available power can lead to significant changes in interplanetary ΔV requirements, it was assumed that near earth transfer ΔV requirements were independent of start mass and power. Figure 2 illustrates the variation of ΔV on a LEO-to-GEO orbit transfer as the power of the systems is varied. The ΔV requirements from SECKSPOT appear to have an oscillation as the thruster power is varied, however; the ΔV precision is within 2% for all cases. The cause of the oscillating effect has not been verified.

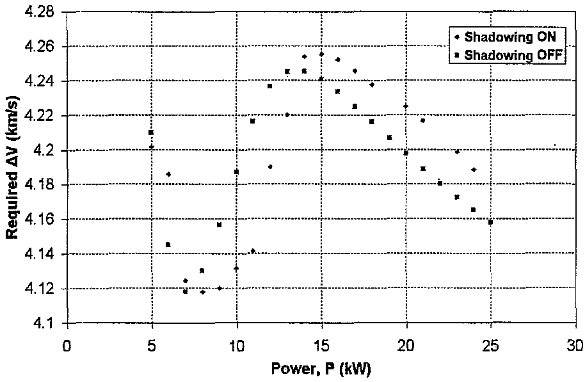


Figure 2: Variation of ΔV as function of power

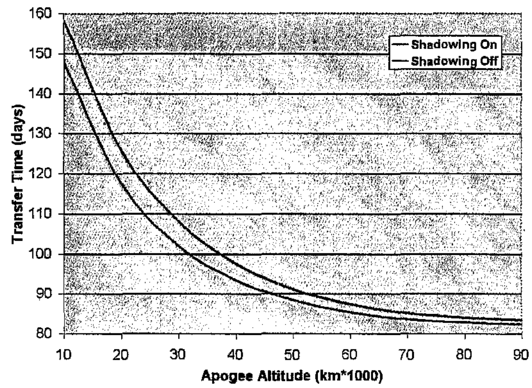


Figure 3: Effects of occultation on transfer time

The SECKSPOT optimizer has the ability to account for occultation effects but again has convergence challenges associated with the function. Because of the numerous starting cases, solutions were converged without shadowing and then a trip time penalty was applied. Also, while occultation effects are more severe with lower starting altitudes and lower accelerations, it was assumed that occultation effects were a maximum penalty of 10%. A single baseline case was completed with occultation effects both on and off to determine the true penalty. Figure 3 shows, as expected; that occultation effects are amplified at lower starting orbits with a maximum delay penalty of 12.7% from the baseline starting conditions. It is important to note that occultation effects are highly dependent on inclination, however; the initial starting orbit inclination was determined by the launch vehicle launch site. Also, rather than converging each case for each power level, efficiency, specific impulse, etc., the thruster burn-time was calculating using Equation 1.

$$t_b = \frac{I_{SP}^{\,2} \cdot g^2 \cdot \dot{m}}{2 \cdot P \cdot \eta}\left(1 - e^{-\frac{\Delta v}{I_{SP} \cdot g}}\right) \tag{1}$$

Additional assumptions include:

i) Solar array degradation effect on performance was neglected; instead a 20% Beginning of Life (B.O.L) power margin was applied.
ii) Throughput capability of the thruster was always sufficient to meet mission needs.
iii) Spacecraft configuration is driven by cost rather than delivered mass performance.
iv) Thruster efficiency is independent of specific impulse.
v) The optimal SEP stage can fit within the launch vehicle fairing.

Solar array degradation will have some effect on the system performance. Various solar array technologies, such as thin film, stretch lens arrays, multi-junction, etc., have widely different radiation tolerances. Also, cover plates can be used to prevent degradation. The radiation influence is also dependent on the starting orbit. If a more detailed study is conducted, the trade of solar array technology to best meet the mission needs is recommended. Optimizing a design for cost, not delivered mass, does produce different results. Mass optimized designs tend to use a higher specific impulse than cost optimized systems. It has also been observed that higher specific impulse systems tend to operate at higher efficiencies. Because the analysis was also conducted using thrusters with known efficiencies, the parametric portion was conducted with an average efficiency. This assumption helped to perform both Hall and gridded ion technology parametrically and concurrently; typically, gridded ion thrusters have higher efficiencies than Hall thrusters. Finally, the stowage volume of large solar arrays and the propulsion system may be of concern and a point design is shown on page twelve.

COST ASSUMPTIONS AND BASELINE CASE

Because the study is to specifically address the cost of the stage, the component costs are critical in determining the overall system configuration. The subsystem component costs are listed in Table 1. The baseline is for a Hall thruster propulsion system; based on heritage systems, higher thruster and power processing unit costs were assumed for gridded ion thrusters. The $30,000 per kg per year revenue potential is based on revenue generated from communication satellite transponders with only the transponder mass representing the revenue generating payload. The study also performed a sensitivity trade to the subsystem costs. Finally, a baseline mission case was established to provide a comparison of the various technology and configuration options. The baseline attributes are listed in Table 2. Unless otherwise noted, results refer to the baseline case.

Table 1. Subsystem cost assumptions

Array Cost ($/kW)	800
Thruster Cost ($/unit)	400,000
PPU Cost ($/unit)	800,000
Feed System Cost ($)	500,000
Tank Cost ($)	450,000
Structure Cost ($/kg)	25,000
Array Mount/Tilt Cost ($/kg)	50,000
Thermal Control Cost ($/kg)	50,000
Battery Cost ($/kg)	25,000
Solar Array Cost ($/kg)	50,000
Avionics Cost ($/kg)	100,000
Propellant Cost ($/kg)	1,200

Table 2. Baseline mission metrics

Launch Vehicle	Delta II 2925H-9.5
Launch Vehicle Cost ($)	125,000,000
Starting Apogee (km)	10,000
Destination	GEO
Inclination (degrees)	28.5
Insurance Percentage (%)	25
Revenue Stream ($/kg·year)	30,000
Return on Capital (%)	15
Duration (years)	15

STARTING ORBIT SELECTION

The performance requirements are highly dependent on the starting orbit. Various launch vehicles have different delivered mass capabilities to a multitude of inclinations and altitudes. Leveraging the launch vehicle performance can decrease the trip times associated with low thruster transfer, the time spent in the radiation belt, and gravity losses.

CIRCULAR VS. ELLIPTICAL

The initial mass to LEO (IMLEO) for a specified launch vehicle has a sharp decline for higher altitude orbits as shown in Figure 4.[2] Also, the launch vehicles tend to exhibit a steeper decline in performance with high circular orbits

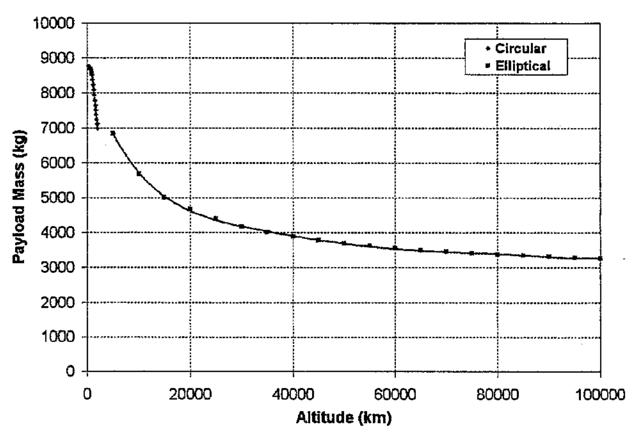


Figure 4: Launch vehicle capability vs. apogee altitude

and require an apogee burn not available on launch vehicles using a solid upper stage. Starting with a higher orbit can reduce the ΔV requirement of the SEP stage and therefore reduce the orbit transfer time.

RADIATION

What may be of more concern than total trip time is the duration spent in the debris and radiation environments at lower altitudes. There is an increased risk of orbital debris collision at altitudes lower than 1,200 km, and there is also a concern of radiation degradation while the vehicle is below 10,000 km. Both of these are risk factors for the payload and may negatively impact the performance or the requirements of the transfer vehicle. Again, this is a case where launching to an elliptical orbit can decrease the time spent in the hazardous environment without limiting the launch vehicle performance as severely. The elliptical orbit provides a method of injecting into a lower energy orbit with minimal time spent in the harshest environments. Figure 5 shows the time spent in the debris and radiation environments as a function of elliptical apogee.[3]

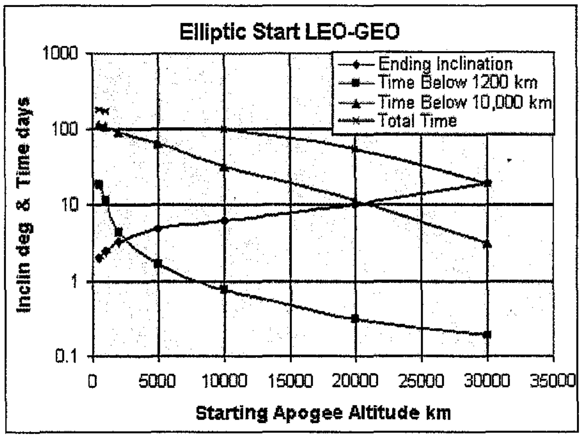
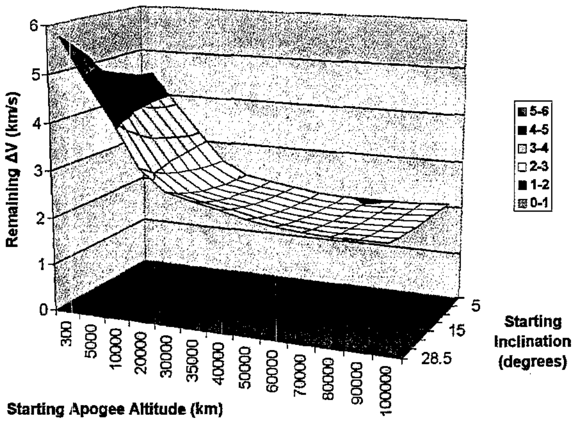
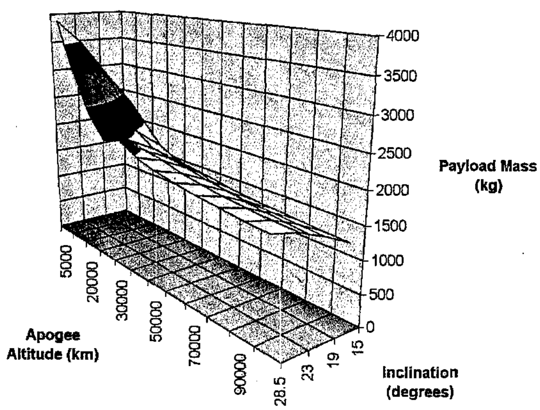


Figure 5: Duration in hazard environment vs. apogee altitude

INCLINATION



(a)



(b)

Figure 6: Effect of starting orbit on remaining ΔV (left) and launch vehicle capability (right)

Launch vehicles can perform plane change if necessary but it is generally a high ΔV maneuver. The launch vehicle performances for launches out of Cape Kennedy at an inclination of 28.5° have their best performance to their starting inclination with a steep falloff in mass capability to an equatorial orbit. Figure 6a illustrates the remaining velocity change that must be imparted to achieve the final orbit. The remaining ΔV is highest when the launch vehicle does the least amount of work; no plane change with a low altitude.

Figure 6b provides clear insight into the advantages of the SEP upper stage. As long as the starting altitude is not too low to cause significant drag or gravity losses, the greatest delivered mass will be obtained when the higher performance SEP stage is used for the largest amount of ΔV. Due to long transfer times; this is not necessarily the most cost effective approach, but it is the best mass performance approach. Also, the inclination change is not a significant ΔV penalty for an optimized SEP stage, but it does have a large mass penalty on the launch vehicle performance.

COSTING METHODOLOGY

Because the study was directed to optimize a SEP stage for cost and return on investment, a net present value calculation was applied to determine overall cost. The concept is that a spacecraft will start to generate revenue after it has been placed in it target orbit, but an electric propulsion stage will take considerably longer to reach the target orbit than a chemical system. Also, the attributes of the SEP stage, such as the power, thruster type, efficiency, etc., will affect the transfer time.

Assuming a continuously compounded rate of return, the net present value is calculated by Equation 2.

$$\int_0^T e^{-rt} = \frac{1}{r(1-e^{-rT})} \tag{2}$$

A typical expected rate of return for commercial capital investments is approximately 15%. As an example, consider whether it is worth investing in an additional thruster capable of reducing the transfer time by 30 days at the additional cost of $1M. Assume that it takes three years from the time it is purchased to begin generating revenue of $50M per year, and that the satellite has a lifetime of 15 years.

The total revenue the spacecraft will generate is:

$$\frac{1}{r(1-e^{-rT})} = \frac{1}{0.15(1-e^{-(0.15)(15)})} = 7.45 * \$50M = \$372.6M \tag{3}$$

The value of the money three years earlier at the time of purchase is:

$$e^{-3r} = e^{-0.45} = 0.638 * \$372.4M = \$237.6M \tag{4}$$

The value of the return if the trip time is reduced by 30 days is:

$$e^{-(3-\frac{30}{365})(0.15)} = e^{-0.433} = 0.649 * 372.6M = 241.7M \tag{5}$$

Because the cost of the additional thruster is lower than the additional $4.1M in added revenue value, it is worth purchasing the thruster for the shorter trip time.

It should be noted here that the government missions will not use the same rate of return. The government does not rely on generated revenue with an expected rate of return to determine spacecraft design, however; they would benefit from both the reduced cost per kilogram and an increased mass capability per launch attributed to a SEP stage.

RESULTS

LAUNCH VEHICLE

The launch vehicle has a significant impact on the SEP stage design. With a larger start mass, a higher thrust stage is necessary to avoid excessively long transfer times. As expected, a larger SEP prefers to operate at a higher power and a higher specific impulse. Smaller launch vehicles tend to optimize to relatively low power levels which is more practical as the smaller launch vehicle fairings may be volume limited. Figure 7 shows the specific impulse that provides the largest total revenue for the baseline mission. The circles represent the maximum revenue and the squares represent the BPT-4000 thruster. The curves are relatively flat, which shows that the specific impulse is not critical, and a slight

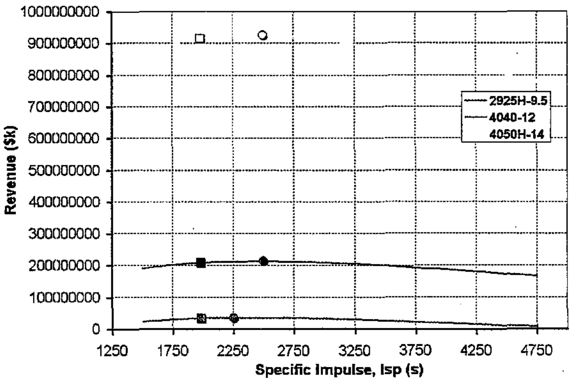


Figure 7: Baseline and optimal specific impulse revenue potential

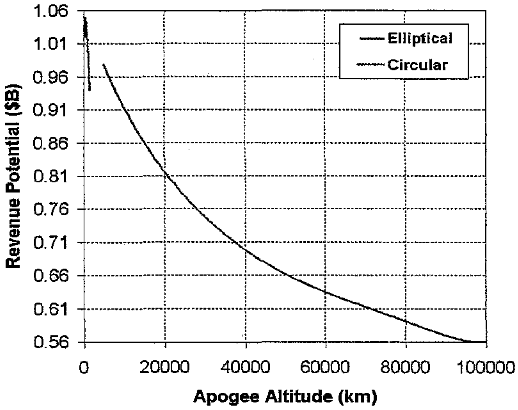


Figure 8: Revenue potential vs. starting orbit.

increase or decrease will have little impact on the revenue of the asset. It should be noted that the various systems are not all at the same power, the larger launch vehicles use a higher power solar array. The optimal SEP power levels for a Delta II, Delta IV-M and Delta IV-H are 9, 18 and 58.5 kW respectively for the Hall thruster and 13.8, 26.7 and 82.8 kW respectively for the gridded ion engine.

STARTING ORBIT

The starting orbit used with the SEP upper stage may not be driven purely by cost. As mentioned previously, there is additional risk with flying satellites at low altitudes due to both the debris and radiation environments. The customer spacecraft may not wish to subject their asset to additional risk for a small increase in performance. At this point, how much risk the customer is willing to accept is subjective and perhaps not fully accounted for within this analysis. In terms of cost, results indicate that the mass delivered to the final orbit is more important than the trip time associated with an EP transfer. The delay of a few weeks for using a higher specific impulse thruster is outweighed by the benefit of the added delivered mass. The results shown in Figure 8 illustrate that the best starting orbit would be a low circular orbit which would allow for the largest amount of final mass. However, there is a point where the poor performance of the launch vehicle to higher circular orbits becomes significant and an elliptical orbit should be used if a faster transfer is preferred for non-revenue based decisions.

SOLAR ARRAY SENSITIVITY

The solar array of the SEP is a large percentage of the SEP stage costs. Figure 9 shows the effect that the cost of the solar array can have on the preferred SEP power. The competing factors are that a larger starting power can allow for a faster transfer and therefore the payload can start generating revenue earlier. However, as the power increases the inert mass increases as well as the cost for the SEP systems reducing the benefits of a faster transfer by decreasing the final delivered payload. Even if the solar array were free, the inert mass becomes quite significant. An apparent observation when choosing the appropriate SEP power level is that if an additional thruster is warranted, then the solar array should be increased to utilize the maximum power capability of the available thrusters.

The mass of the solar array is also a critical factor because the solar array makes up a large portion of the stage total mass. Figure 10 shows how changes in the solar array mass affect the overall revenue generated. It is also worth noting that as the array sizes become very large, as is the case with large launch vehicles; the revenue is very sensitive to power and is therefore a critical factor in the system configuration.

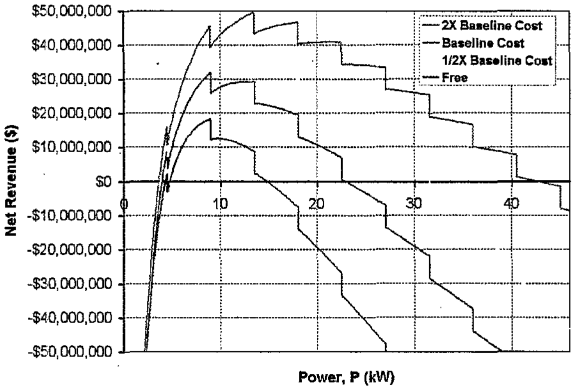


Figure 9: Sensitivity to solar array cost

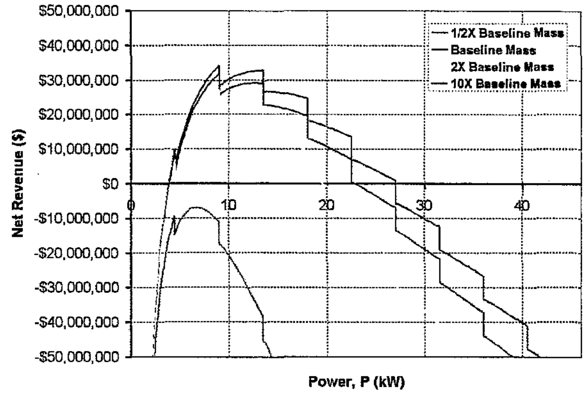


Figure 10: Sensitivity to solar array mass

PROPULSION SUBSYSTEM SENSITIVITY

Like the solar arrays, the propulsion system components make up a large portion of the stage mass and costs. Figure 11 shows the significance of changes in SEP subsystem cost and mass. NASA's In-Space Propulsion Technology Project has been investing in lightweight reliable feed systems and has plans for a reduced mass PPU development in the near-term future. These investments are expected to significantly impact future gridded ion and Hall thrust systems. While the BPT-4000 is already a qualified thruster and is not likely to see appreciable improvements in subsystem mass, there are potential decreases in hardware cost as demand increases for the thruster.

Gridded ion systems also tend to optimize to slightly higher specific impulses than the Hall systems;

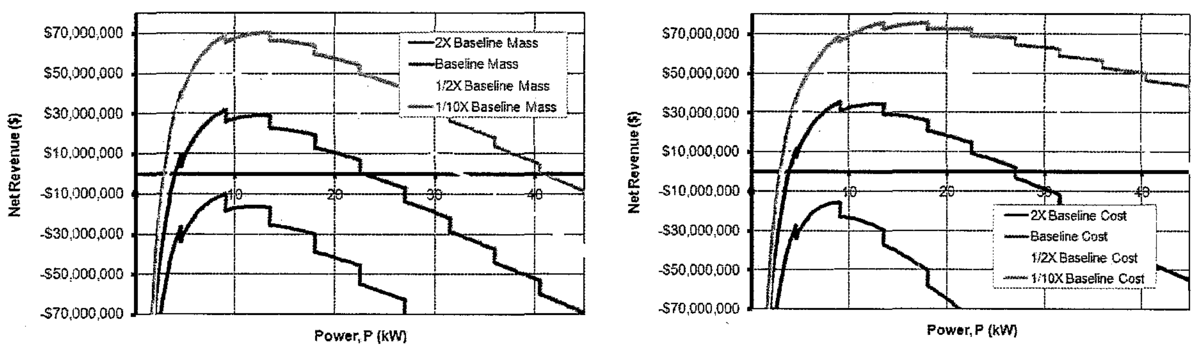


Figure 11: Sensitivity of SEP subsystem mass and cost.

this is illustrated in Figure 12. The higher specific impulse is likely due to the higher efficiency of the thrusters. After there is enough available thrust for reasonable accelerations, additional available power is used to conserve propellant mass.

All of the subsystem components can also be traded for cost or performance. It was observed on the Dawn mission that the cost of a lightweight propellant tank was considerably more expensive than an off-the-shelf standardized tank. A tank costing $50,000 with a 7% mass ratio was compared with composite overwrap tank with a cost of $450,000 and a mass fraction of 4.5%. For the baseline case, the total revenue changed to $31,504,436 from $33,011,095. This shows that subsystem mass is more important than subsystem cost. Therefore an upfront cost increase of 900% for the tank is worth the investment.

The propulsion system configuration can also have a significant impact on cost and performance. While it has not been fully explored, the data trends show that the best upper stage configuration usually consists of a 2+1 system, two operating and one spare engine with the maximum power of the chosen thruster to be equal to that of one half the optimal solar array power level. The 2+1 thruster configuration allows for a limited number of thrusters with a small penalty of carrying a spare engine. A 1+1 system results in substantial thruster cost and mass penalties and more complex systems with several thrusters are often undesirable. Also, using two thrusters gimbaled can allow for attitude control using the SEP stage instead of a chemical ACS or momentum wheels. While this analysis assumed a spare thruster requirement, integrated SEP stages for commercial applications may not impose such a restriction.

REVENUE AND RATE-OF-RETURN SENSITIVITY

The rate-of-return used for this study was base-lined at 15%. It is certainly possible that investors would be willing to try business opportunities that could yield a 10% return. Also, the government may not be expecting a large return, if any. Figure 13 shows the power level sensitivity to the rate-of-return. As the expected rate of return increases, the transfer time becomes of more importance and higher power systems are preferred. If time is only a minor consideration, then only enough power to achieve the desired orbit is necessary.

BPT-4000 VS. NEXT

The results of a direct comparison between the BPT-4000 and the NEXT gridded ion engine for the baseline mission are shown in Figure 14. The best system configuration for both thrusters is a 2+1 system. For lower available power, the BPT-4000 can generate more revenue than the NEXT. At higher power, the NEXT outperforms the Hall thruster. This concurs with previous a observation that after enough power is available to generate the necessary thrust for reasonable transfer rates, additional power should be used for mass performance through higher specific impulses. The highest perform for the BPT-4000 and NEXT are at 9kW and 14kW respectively, the full power rating of two simultaneously operated thrusters.

Figure 15 shows the optimal power per kilogram using the BPT-4000 and NEXT. The power per kilogram for the Hall system is consistently lower than the gridded ion system, the Hall thruster optimized between the values of 2.88 and 3.43 W/kg while the NEXT thruster between 4.42 and 4.85 W/kg for all launch vehicles. The optimal thrust-to-weight increases as the initial mass increases.

The thruster throughput capability is something that cannot be ignored. The throughput requirements for the various missions ranged from 56.1 to 66.6 kg/kW for BPT-4000 and from 19.8 to 21.7 kg/kW for NEXT. Table 3 shows the required propellant throughput capability for various launch vehicles. The throughput capability of the BPT-4000 is currently only 53 kg/kW[4] and that does not include the 150% qualification standard that would further reduce the operational capability to only 35 kg/kW. The NEXT thruster is expected to have a throughput capability exceeding 40 kg/kW[5] including the qualification margin. Therefore only the Hall system appears to be limited in throughput capability. Multiple thrusters could be added for lifetime, but this is not often a desired system configuration and could lead to substantial stage mass and cost increases. Another option is using a novel lifetime extension mechanism as is under development with the NASA 103M HiVHAC thruster.[6] The Hall thruster development goal is an aggressive throughput capability

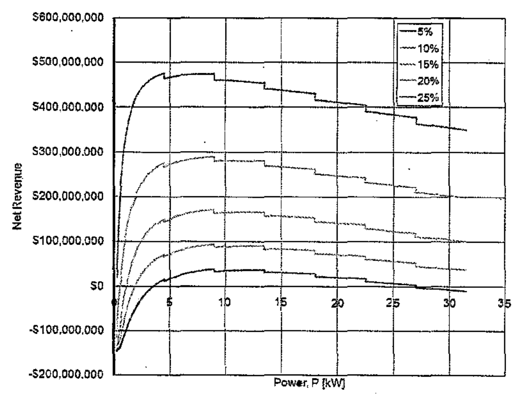


Figure 13: Sensitivity to Rate of Return

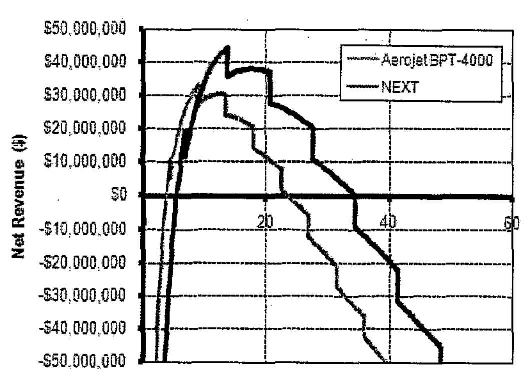
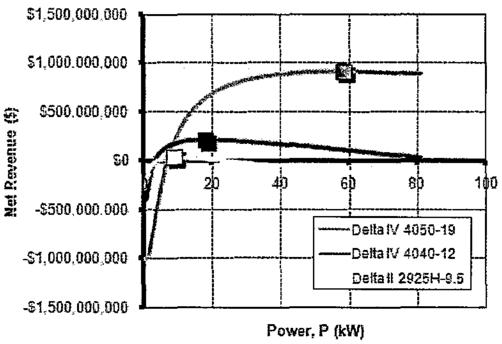


Figure 14: BPT-4000 vs. NEXT

of over 80 kg/kW.



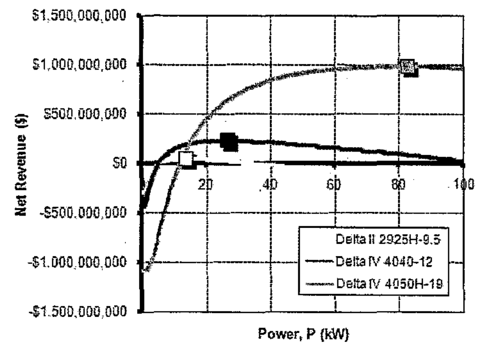


Figure 15: Optimal power for baseline BPT-4000 (left) and NEXT (right) configurations

Table 3. SEP Performance for various launch vehicles and thrusters for LEO to GEP Mission

	SEP Performance for Various Launch Vehicles and Thrusters for LEO to GEO Mission					
	Delta II 2925H-9.5		***Delta IV 4040-12***		***Delta IV 4050H-19***	
	BPT	*NEXT*	*BPT*	*NEXT*	*BPT*	*NEXT*
Propellant (kg)	599.7	299.8	1091	545.8	3282	1641
Throughput (kg/Thruster)	299.9	149.9	273	136.5	252.4	136.7
Optimal Power (kg/kW)	66.6	21.7	60.6	19.8	56.1	19.8

COMMERCIAL OPPORTUNITIES

As the results have shown, there is a considerable potential to generate additional revenue through the use of an electric propulsion upper stage in place of a chemical orbit insertion. Unfortunately, the most profitable orbit for commercial enterprises is likely to continue to be a geostationary orbit. Geostationary orbits require large chemical stages integrated with the spacecraft or a very high cost launch systems. With medium launch vehicles costing over $100 million dollars, there is a very high entry level investment into geostationary satellites. Lower cost launch vehicles, such as the Orbital Sciences Minotaur and potentially the SpaceX Falcon 1, have a maximum payload mass to LEO approaching 600 kg. Using a chemical propulsion stage with these low cost launches is impractical for delivering reasonable payloads into geostationary orbits; however, a SEP upper stage could deliver considerably more payload at a much lower cost per kilogram.

Using the desired 15% rate-of-return on investment, these low cost launchers would need payloads that can generate revenue on the order of $35,000 per kg. This is not far beyond current transponder values. Of course, these spacecraft can still generate substantial profits, albeit at a slightly lower return. By reducing the entry barrier for entrepreneurs by an order of magnitude in mission costs, the number of investors willing try new ideas will rise substantially. Low cost GEO satellites could open the door for startup industries such as internet providers, private communication satellites, foreign countries, etc., before they can build the demand requiring larger asset investments.

CHEMICAL ONLY COMPARISON

All of the examples have shown that SEP upper stages can be used to generate revenue. We also know that substantial revenue is usually generated in the satellite industry using chemical systems. Figures 16 and 17 provide a comparison of the improvement in revenue by using the SEP upper stage in lieu of a complete chemical system. The SEP system can provide for a 25% increase in revenue in this baseline case and can also deliver substantially more payload to GEO than can be with today's chemical systems. Note that it is assumed that the upper stage is delivering the spacecraft to the target orbit. Medium class launch vehicles can and have delivered spacecraft with their own chemical apogee motors for GEO insertion.

DEVELOPMENT STRATEGY

The commercial industry can anticipate insurance costs on the order of 25% on new spacecraft. This is a high burden to pay although the payoff can be high. Unless wealthy industry or private enterprises are willing to pay for the risk, it is likely that the government will be the first user of a SEP upper stage. It has been the role of governments to assume the high risks associated with advanced technology developments that can benefit their nation. Once the technology has been flown, commercial industry could quickly use the flight experience for a variety of profitable mission opportunities.

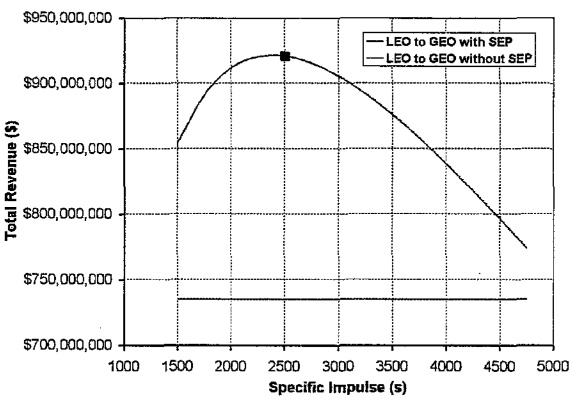


Figure 16: Revenue comparison both with and without a SEP upper stage

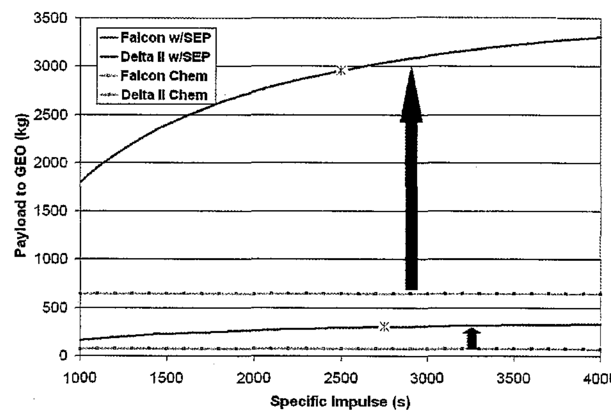


Figure 17: Performance comparison of Chemical and SEP upper stage

As mentioned in the previous section, entrepreneurs may be more likely to invest in new technology if the entry cost is substantially lower. Therefore, one approach for electric propulsion upper stage technology infusion is to begin with a government led program that can verify the performance of the stage suitable for low cost geostationary satellites. Because the ΔV requirement for low lunar orbits is not far greater than geostationary satellites and the recent call for robotic and lunar explorations of the moon, a government led technology demonstrator mission to the moon is a logical starting point. As the science community has already marketed, there is a lot of science that can be gained from low-cost lunar missions. After the government has flight validated the technology, commercial enterprises would be much more likely to use the heritage stage because it will increase revenue, the primary concern of commercial investors. After the

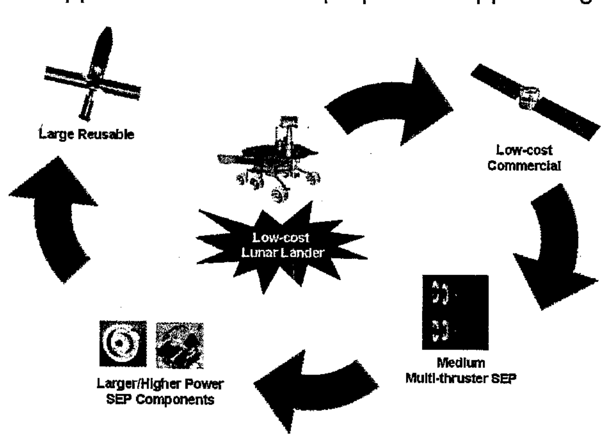
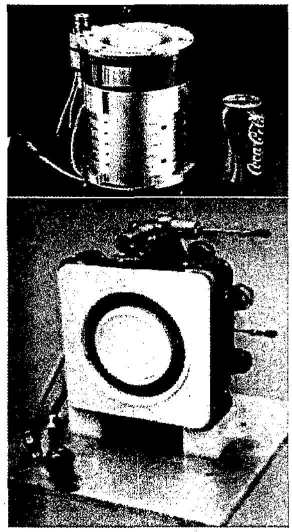


Figure 18: Development strategy

commercial industry has been flying SEP stages, they are likely to evolve first with only engineering modifications for production improvements, integration processes, and enhanced capability by adding similar strings, and then eventually technology improvements for more capable, higher power and larger throughput stages. An example development evolution path for SEP stages is illustrated in Figure 18.

LOW-COST LUNAR MISSIONS

A point design was evaluated in greater detail for the use of a SEP upper stage with a low-cost launch vehicle to perform lunar science. Based on low-cost launch capabilities.[7,8] we started with a rough allocation 200 kg for SEP inert mass, 200 kg for propellant and 200 kg for payload, which was presumed to be a micro-lander. We also considered jettisoning the SEP propulsion system (it is about half the inert mass) and landing the solar array, for example as a power source for an ISRU experiment. Of course, a lunar orbit payload could also be flown.



NASA-103M
- Max power: 3.5 kW
- Thrust: 150 mN
- Specific Impulse: 1250-2750 sec
- Thruster Mass: 4.8 kg
- Estimated PPU mass 10.5 kg (based on 3kg/kW)
- Max PPU input power : 3.7KW (η =0.95 at full power)

Aerojet BPT-4000
- Max power: 4.5 kW
- Thrust: 244 mN (typ)
- Specific Impulse: 1750-2000 sec
- Thruster Mass: 12.3 kg, includes mounting bracket and cabling
- Estimated PPU mass 12.75 kg
- Max PPU input power : 4.7KW (η =0.95 at full power)

Figure 19: Candidate thrusters

This allocation led to an estimate that we could produce between 5 and 10 kWe electric power, and we settled on approximately 7 kWe for electric propulsion and a few hundred watts for spacecraft bus power. Two Hall thrusters were chosen for this analysis; the Aerojet BPT-4000, and a Glenn Research Center thruster, the NASA-103M. The former is flight-qualified for more than the propellant load for this application; the latter is developmental and is scheduled for substantial testing in 2007. These are illustrated in Figure 19. The NASA-103M thruster is lighter and has a slightly higher specific impulse than the BPT-4000. It is rated at somewhat less power, 3.5 kWe, but that is all that is required for this mission. We use two thrusters so that gimbal motion can provide roll control as well as pitch and yaw control.

CONFIGURATION

The small fairing size places constraints on the configuration. It cannot be too big in diameter and must be relatively short to leave room for a payload. A basic arrangement of two propellant tanks, two solar wings and two propulsion power processing units (PPUs) was found to be relatively compact. This arrangement is shown in Figure 20. This layout fits in the Falcon 1 fairing, leaving 1/2 meter of cylinder for payload and there is about a meter of tapered section above that. The PPUs will be able to "see space" and likely contain enough radiating area to cool themselves.

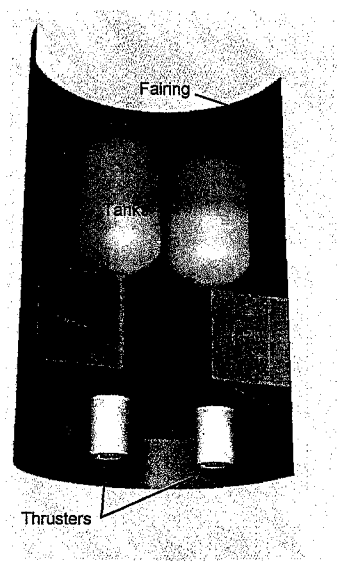
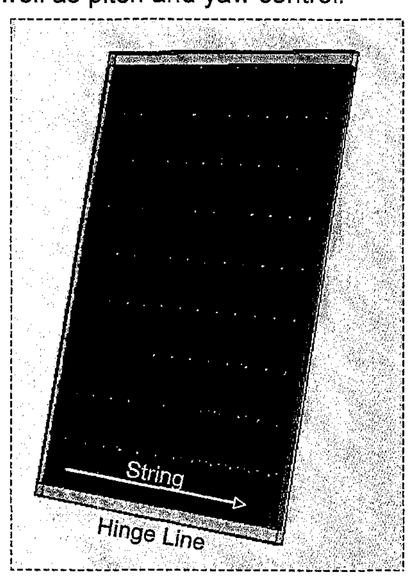


Figure 20: General arrangement



13 strings, each 9 cells long

Figure 21: Single panel cell arrangement

SOLAR ARRAY

The small fairing diameter and need for compact packaging leads to an array configuration with small (0.51 x 0.75 m) individual panels. Modern high-performance cells are relatively large, about 30 sq cm. A string can run either way on the panel with length 13 cells the short dimension or 19 the longer dimension. The shorter string uses the available area slightly better with 117 cells per panel vs. 114. It also is more convenient to hook up. The arrangement is shown in Figure 21. These cells have output voltage about 2.7, so a 13-cell string produces about 35 V. That's a pretty low voltage for 7 kWe, requiring conductors for 200 amps.

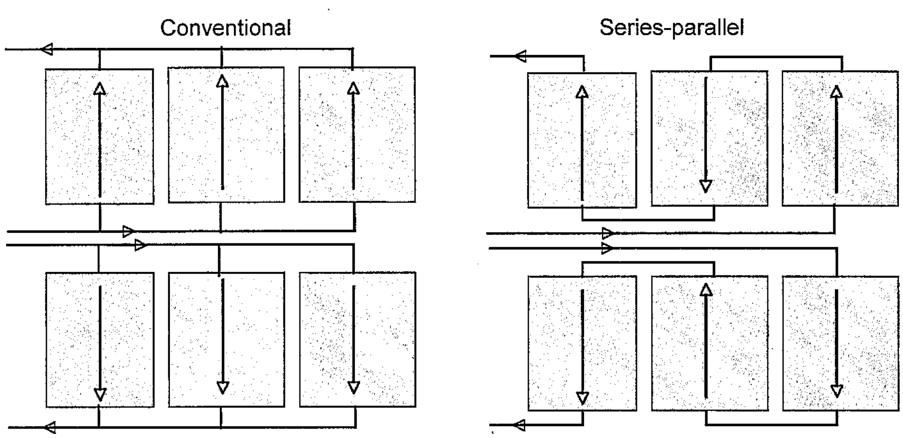


Figure 22: Conventional Array Arrangement Compared with Series-Parallel

We decided to use a series-parallel hookup for the cells as shown in Figure 22. That produces about 100 V which interfaces better with electric propulsion PPUs and requires less conductor mass. This requires that the array wings have a number of panels divisible by 3; we chose 15. There is also a half-panel inboard and outboard for symmetric folding. (Figure 23 shows one wing partly deployed.) The inboard half-panels can produce about 200 W at 35 V and are used as a backup power supply for critical vehicle functions such as communications and computing. The outboard half-panels are only structural frames for folding. Panels are deployed by single redundant-drive extensible masts in the center of the array.

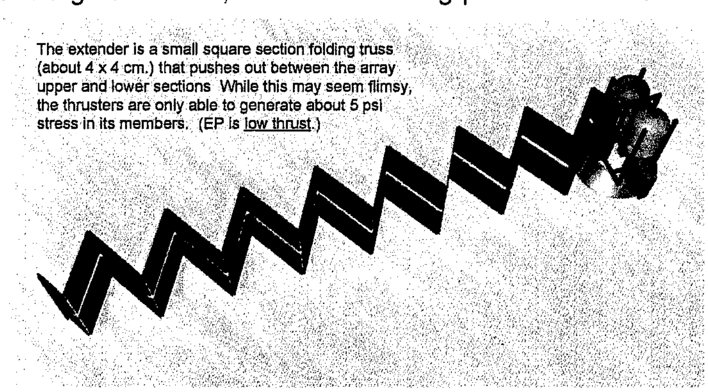


Figure 23: System With One Array Partially Extended

PERFORMANCE AND CONTROL

The usual method of control for a SEP vehicle is to roll the vehicle until the solar array long axis is perpendicular to the Sun vector and then tilt the array around this axis until it exactly faces the Sun. The control system attempts to maintain pitch and yaw zero. Nominal thruster pointing is aft (thrust vector forward). This mode of control involves significant roll motion, hence the need for two thrusters to provide roll control. Using chemical or cold gas thrusters for this is very expensive in propellant usage in and near low Earth orbit. It should be noted that the ESA SMART-1

Table 4: Mass estimate for LEO-LLO spacecraft.

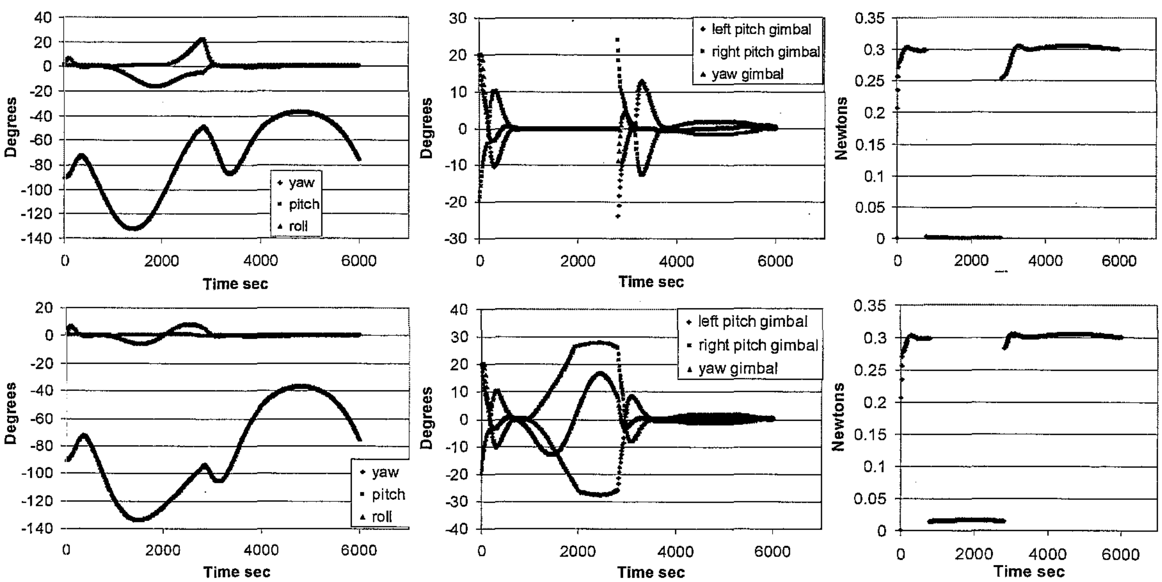
Solar Array Installation	40.74
Propulsion System	72.80
Structure	29.95
Avionics	30.50
Cold Gas ACPS	7.83
Dry Mass	**181.82**
Residual and Reserve	18.80
Inert Mass	200.62
Main Propellant	**188**
Payload	**180**
Total Mass	**568.62**

only had a single thruster, but it started in a geosynchronous transfer orbit.

Moments of inertia were calculated and a 6-DOF SEP flight simulation was used to assess controllability. The small lateral spacing of the thrusters limit roll control authority. Figure 24 shows attitude motions and errors for one orbit at minimum altitude (worst case for control). The upper graphs are with no thrust during occultations and the lower are for 5% power (from a battery) during occultations. We also ran a simulation for about 45 orbits to assess performance losses due to control activity (propellant consumption versus ideal propellant consumption for the same increase in altitude); they were found to be very small, about 2%. These losses will decrease with altitude. Array off-pointing due to lag in attitude versus commanded attitude is slight. Power loss was calculated by a cosine rule and was negligible. The power loss would require further investigation should the use a concentrator array such as the SLA be desired.

MASS PROPERTIES

Mass properties were estimated from known mass of solar cells and propulsion hardware, and calculated mass for array panels and structure. The propellant tanks represent an off-the-shelf design for 4500 psia helium tanks. Actual pressure rating needs to be about 2500 psia, so there is a potential for mass savings here. Adding a reasonable margin to structure and avionics adds about 30 kg to the dry mass. The avionics as given in Table 4 is can likely be reduced.



(a) Vehicle Attitude (b) Gimbal Motion (c) Thrust

Figure 24: Results of 6-DOF flight simulation for one orbit: top – no battery; bottom – 5% battery

CONCLUSIONS

In general there appears to be a large potential market for electric propulsion upper stages with both existing and near-term launch vehicles. Electric propulsion upper stages can trade trip time for mass performance, and commercial applications tend to be more sensitive to delivered mass than the delay of using a low thrust transfer. With a clear revenue generating capability for a SEP upper stage, it is hopeful that with a government led technology validation, SEP upper stages may become as common to launch vehicles as the STAR motors are today.

Critical observations are as follows:

i) For cost and mass, the transfer should leverage the performance of the SEP stage instead of using the launch vehicle to place the payloads into higher altitude orbits.
ii) High circular orbits have very low performance compared to both low circular orbits and high elliptical orbits.
iii) Low circular starting orbits have the potential to generate more revenue because mass delivered is more critical than transfer time.
iv) Cost optimized electric propulsion systems have lower specific impulses than performance optimized systems.
v) Any SEP stage should have enough power to fully utilize the capability of the available thrusters.
vi) SEP stage performance is relatively insensitive to specific impulse.
vii) SEP stage performance is very sensitive to solar array power.

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