Application of Solar-Electric Propulsion to Robotic and Human Missions in Near-Earth Space

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Interest in applications of solar electric propulsion (SEP) is increasing. Application of SEP technology is favored when: (1) the mission is compatible with low-thrust propulsion, (2) the mission needs high total delta V such that chemical propulsion is disadvantaged; and (3) performance enhancement is needed. If all such opportunities for future missions are considered, many uses of SEP are likely. Representative missions are surveyed and several SEP applications selected for analysis, including orbit raising, lunar science, lunar exploration, lunar exploitation, planetary science, and planetary exploration. These missions span SEP power range from 10s of kWe to several MWe. Modes of use and benefits are described, and potential SEP evolution is discussed.

Nomenclature and Acronyms

a	= semi-major axis (sma)
e	= eccentricity
F	Thrust component
k, k_1, k_2	undetermined constants
T_1	$= 1 + e \cos\theta$
T_2	$= \mathrm{sqrt}(1 + 2e\cos\theta + e^2)$
T_3	$= 1 - e^2$
T_4	$= e + \cos\theta$
α	= pitch angle; unknown exponent
β	= yaw angle
γ	= path angle re local horizontal
ΔV	= delta V
θ	= true anomaly (from periapsis)
λ_a	= LaGrange multiplier for sma
λ_p	= LaGrange multiplier for plane
μ	= Earth geopotential
ω	= argument of periapsis

ArgP argument of periapsis
C3 measure of trajectory energy
CEV Crew Exploration Vehicle
EELV Evolved Expendable Launch Vehicle*
EEO Elliptic Earth orbit
EP electric propulsion
ESA European Space Agency
ESAS Exploration Systems Architecture Study
GEO Geosynchronous Earth Orbit
Incl orbit Inclination
Isp specific impulse
kWe kilowatts electric
LN longitude of Node
LOI lunar orbit insertion
LOX liquid oxygen
LSAM Lunar Surface Access Module
MLAV Mars lander/ascent vehicle
NEXT NASA's Evolutionary Xenon Thruster
NSTAR Gridded ion thruster used on Deep Space 1
SEP solar electric propulsion
SEPTOP trajectory optimizing code
TA true anomaly
T/W thrust-to-weight ratio
VGA Venus gravity assist

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I. Introduction

Solar-electric propulsion (SEP) has been widely applied to commercial communications satellites for stationkeeping in geosynchronous orbit and has recently seen use on scientific missions including NASA's Deep Space 1 and ESA's SMART-1 missions.

Solar-electric propulsion (SEP) is becoming of interest for application to a wide range of missions. The benefits of SEP are strongly influenced by system element performance, especially that of the power system. Solar array performance is increasing rapidly and promises to continue to do so for another 10 to 20 years (Fig. 1). At the same time, cost per watt is decreasing. Radiation hardness is increasing. New concepts for how to design a SEP are emerging. These improvements lead to changes in the best ways to apply SEP technology to missions, and broadening of the practical uses of SEP technology compared to competing technologies.

It is timely to discuss some of the emerging uses of SEP, and reasonable steps to advancing the technology to make it a sound choice for these applications.

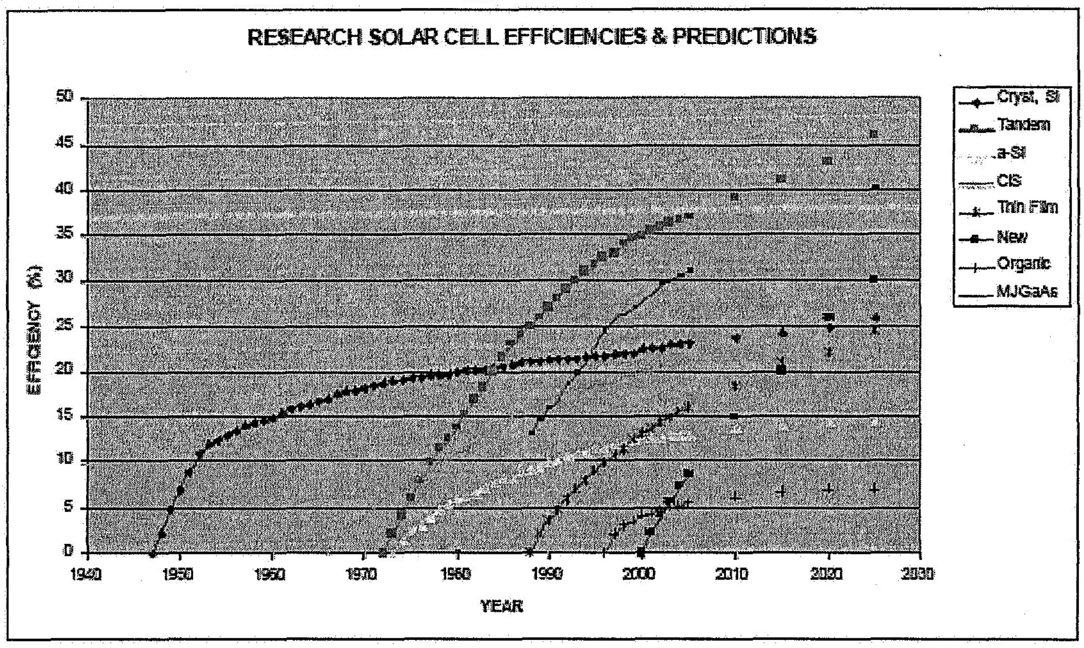


Figure 1. Historical and Projected Solar Cell Performance

II. Mission Applications

Application of SEP technology is generally favored when: (1) the mission is compatible with low-thrust propulsion, i.e., no need to produce accelerations comparable to planetary gravity fields and adequate time to achieve the mission ideal delta V; (2) the mission needs high total delta V such that chemical propulsion is disadvantaged; and (3) performance enhancement is needed to make the mission compatible with existing launch capabilities, or to provide or cost reduction.

Table 1 presents a list of missions by category with potential SEP applications. The categories are borrowed from a survey of mission needs for advanced in-space propulsion, conducted by NASA's In-Space Propulsion Technology organization in 2004. Missions discussed in this paper are underlined.

III. Orbit Raising

A. Why orbit raising?

Orbit raising may fit the criteria set forth in II, depending on trip time considerations. In the near future we may expect to see communications satellites delivered to geosynchronous orbit (GEO) using some of their stationkeeping SEP to assist the orbit raising process by completing the GEO insertion.

More ambitious use of SEP is possible, but requires higher electric propulsion power than installed on current comsat designs. For example, a 2-ton satellite with 25 kWe installed power and 5 kWe electric propulsion thruster power would need about 150 - 180 days to do GEO insertion from a geosynchronous transfer orbit, depending on thrusting strategy. If the installed thrust were designed to utilize 80% of the available electric power (20 kWe) the insertion time would be reduced by a factor of 4, to 40 – 45 days.

Table 1. SEP Applications by Mission Regime

Mission Regime	Representative SEP Applications
Regime 1 - Human Earth Orbit, near-Earth missions (includes ISS)	Orbit adjust and makeup, including application to ISS
Regime 2 - Robotic LEO to near planets; Earth & space observation, planetary science and sample return	Robotic LEO to higher Earth orbits; Mars sample return, missions to NEOs, Venus
Regime 3 - Human HEO to Lunar; space transfer humans/cargo with landing and bases	Cargo mission support for human lunar and other Earth vicinity missions.
Regime 4 - Robotic near Sun; includes Mercury and solar probes/polar missions	Near-sun or out of ecliptic missions requiring high delta V; trade vs solar sails
Regime 5 - Human inner planets; Mars exploration & landing, asteroids, space transfer	Humans and cargo to Mars and NEO and main belt asteroids.
Regime 6 - Robotic outer planets; orbiters, probes, landers and sample return missions	All of these except landers and ascent vehicle themselves.
Regime 7 - Human outer planets; Jupiter and Saturn and their moons, landing and return	Perhaps some portions of trajectories but these missions need thrust far from Sun.
Regime 8 - Robotic beyond planetary system; Kuiper Belt, Oort Cloud, interstellar missions	Very high performance SEP possibly can inject these by producing enough delta V near Sun.

A dedicated SEP vehicle could operate at still higher power and return to low Earth orbit for re-use. The payload fraction (net useful GEO payload/total mass delivered to LEO) could be roughly doubled. This involves 25 to 50 kWe SEP power for typical current comsats, and improvements in SEP radiation hardness so that at least four total uses (round trips) can be obtained. SEP costs are not well established but an estimator of $2000/Watt electric is a reasonable working number. Then a 50 kWe SEP will cost $100 million plus its launch costs. To recoup such an investment through reduction in subsequent comsat launch costs, it must be spread over at least three LEO-GEO round trips, preferably four or more. Also required is a reliable, simple, economic scheme for handing off a launched payload in LEO to the returning SEP vehicle. A suitable technology remains to be demonstrated, but it would appear to be not more difficult than the automated rendezvous and docking used by Russian space station missions for many years.

Issues of radiation and debris exposure, plane change and suitable starting orbits are discussed in the next few paragraphs.

B. Starting Orbit

SEP ascent from a low circular orbit experiences lengthy periods in the van Allen belts and significant periods in the LEO debris environment. Conventional launch vehicles pay a high performance price for higher circular orbits, including decreased payload and a restart of the upper stage. The best solution to this problem is an elliptic starting orbit. Launch vehicle performance to an elliptic starting orbit with low perigee is much better than to higher circular orbits, and does not require an upper stage restart.

Elliptic starting orbits pay off in reducing radiation and debris exposure time, as shown in Fig. 2. Also shown is a typical launch vehicle performance comparison with calculated and quoted[2] data. These are approximate overall effects of elliptic starting orbits. At the price of reduced launch capability, major reductions in total time as well as time in hazardous environments can be obtained. The circular orbit yaw steering formula discussed below was used to generate these general trends. An elliptic orbit formula is also described below and is better for performance but these trends will be vary similar. The SEP can return to the starting orbit in about a month. For all-chemical propulsion the GEO/LEO payload ratio is about 20%; even with an elliptic starting orbit, SEP is expected to double that. A point calculation using Atlas 401 performance found that a 35 kWe SEP starting from a 400 x 10,000 orbit could deliver 3600 kg to GEO in about 130 days, compared to about half that for all-chemical propulsion.

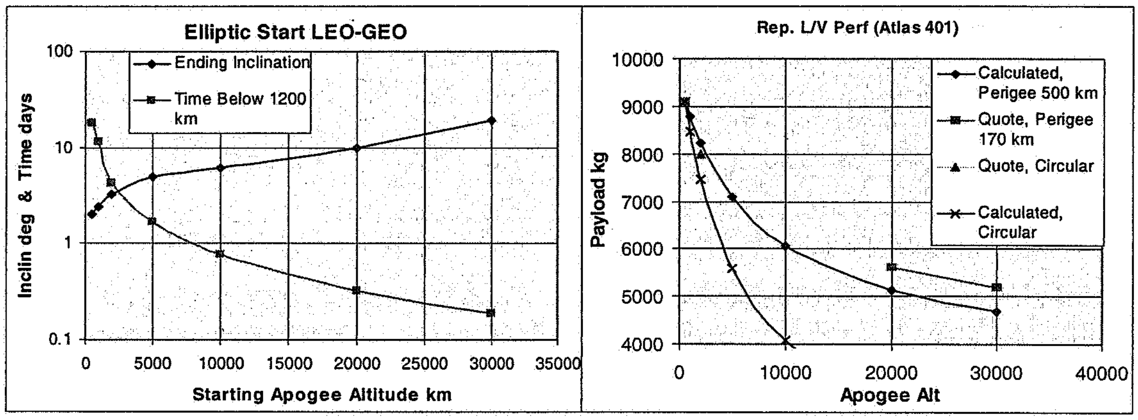


Figure 2. Trends for Elliptic Starting Orbits, as a Function of Starting Apogee

C. Thrusting Strategy

Two questions are posed for orbit transfers: (1) should we thrust all the time or part of the time, and (2) should we thrust in the direction of flight or in some other direction?

The answer to the first question depends on whether we wish to minimize transfer time, or are we willing to trade transfer time for delta V? Numerical experiments were tried with a perigee hold approach (an in-plane pitch program is used to maintain perigee altitude while raising apogee). The pitch equation for perigee hold is

$$\tan\alpha = \frac{2(1-e)T_2 - T_3T_4/T_2}{T_3^2 \sin\theta/(T_1T_2)} \tag{1}$$

Perigee hold cost more delta V and more time to reach the desired orbit. There is no tradeoff of transfer time versus delta V. The next experiment tried was an apogee "gap", a period of non-thrusting centered on apogee. The idea is that thrusting at apogee contributes least to orbit raising. In this experiment, thrust was pointed in the direction of flight. A tradeoff did exist, as shown in Fig. 3. About 15 days of added trip time reduced delta V by a few hundred m/s and could increase payload several percent. Cases may exist where this tradeoff would allow use of a smaller launch vehicle, or enable a marginal payload to be delivered with adequate margin.

The next experiment involved maximizing the increase in apogee. One can write a differential equation for dRa/dp where Ra is apogee radius and p is pitch angle, and set it equal zero to obtain the maximum equation,

$$\tan\alpha = \frac{T_3^2 \sin\theta/(T_1T_2)}{2[(1+e)T_3T_4/T_2]} \tag{2}$$

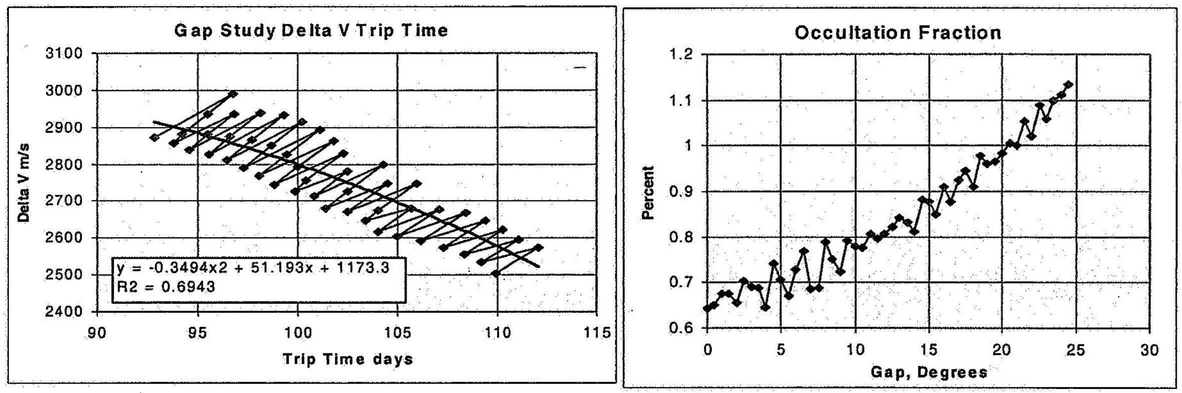


Figure 3: Delta V vs Trip Time and Occultation Effects for Apogee Gap Thrusting

The effects of this scheme are shown in Fig. 4. Over a small increase in semi-major axis, apogee increases faster than in the case of thrusting in the flight direction ($\alpha = 0$). However, the expression for $\partial r_a/\partial m$ includes sma (semimajor axis) as a multiplier. After a while, slower growth in sma negates the advantage and it becomes a loss because smaller sma decreases $\partial r_a/\partial m$. Further, slower growth of perigee leads to more occultation, which also leads to longer trip time. Therefore, this is not a good strategy.

The conclusion is that thrusting all the time in the flight direction provides minimum transfer time, but shutting off thrust over an arc centered on apogee offers a tradeoff of more payload for longer trip time. However, this "apogee gap" slows increase in perigee, therefore increasing radiation and debris exposure.

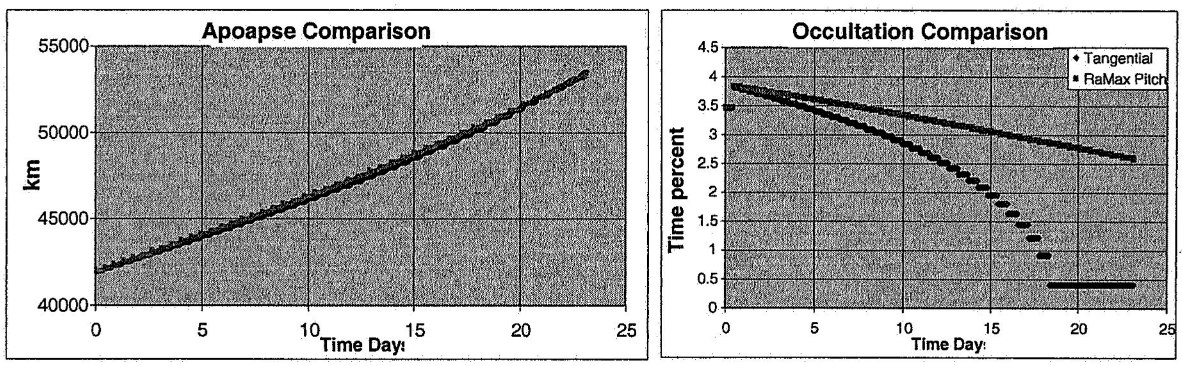


Figure 4: Results of Apogee Maximization Simulation

D. Plane Change During Orbit Raising

Orbit raising frequently requires plane change, as in ascent from LEO at 28.5 degrees inclination to GEO at zero inclination. The delta V added for plane change is significant and should be minimized. This can be done using a general-purpose low-thrust trajectory optimization routine[3], but the approach used for this paper was to derive simple GN&C laws for this specific application.

Using a low-thrust approximation (the orbit remains nearly circular) we can operate on orbit parameters to find the optimum transfer, noting that instantaneous orbit altitude increase is

$da/d\Delta V = 2a^{3/2}/\text{sqrt}(\mu) \cos \beta$, and instantaneous plane change is

$dP/d\Delta V = \text{sqrt}(a/\mu) \cos \theta \sin \beta$, where β is out-of-plane yaw angle.

Writing a Hamiltonian, $H = \lambda_a (2a^{3/2})\cos \beta/\text{sqrt}(\mu) + \lambda_p a^{1/2} \cos \theta \sin \beta/\text{sqrt}(\mu) - d\Delta V$.

We find $\tan \beta = \lambda_p \cos \theta/(2\lambda_a a)$. This tells us that yaw steering oscillates back and forth along the orbit path, with magnitude of the yaw to be determined, and expected to be a function of λ_p, λ_a, and a. λ_p turns out to be constant, and λ_a is not analytically integrable. It could be integrated numerically, but from the form of the differential equation, a solution was presumed to look like $\lambda_a = k_1 \exp(k_2 a^{\alpha})$. One might also experiment with a form $\lambda_a = k_1 a^{\alpha 1} \exp(k_2 a^{\alpha 2})$ but so far we have not done so. Further, a known very good guidance law is $1/(\lambda_a a) = ka^{\alpha}$ (power law), so we compared new guidance laws with that.

As shown in Fig 5, the exponential power law is better than the simple power law but by only 2 m/s. The best exponential multiplier was about 0.75 and the best exponent 0.6. The best exponent for the power law is about 0.85. The two laws are compared at the lower left; they are almost the same. The rate of plane change with delta V increases at higher altitude, as one would expect; velocity is less, so less delta V is needed to obtain a given amount of plane change.

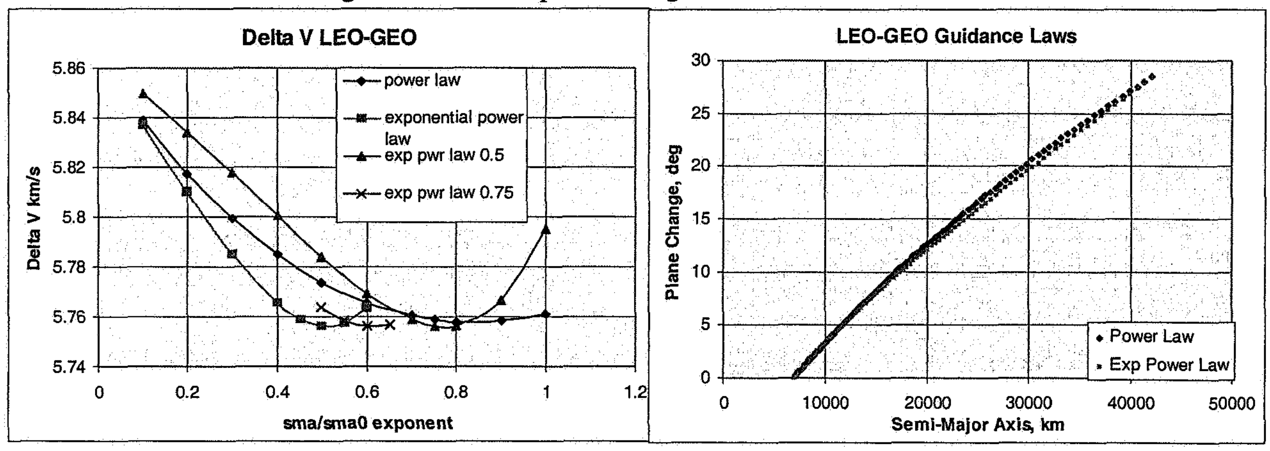


Figure 5. Results of Plane Change GN&C Law Investigation

A low-thrust orbit becomes elliptic during raising. When one nears GEO, it is necessary to circularize. One way, not quite optimal, is to apply an apogee hold algorithm that modulates pitch when apogee reaches GEO altitude. Yaw steering continues. The equation for pitch angle α for apogee hold is: $2(1 + e)\, T_2 + 2\, T_3 T_4/T_2 = -T_3^2 \sin\theta/(T_1\, T_2) \tan\alpha$. The apogee hold pitch angle variation is such that thrusting efficiency is good only near apoapsis, so this algorithm should be used with caution.

Fig. 6 shows a detailed simulation (about 12 million integration steps) demonstrating application of the apoapsis hold algorithm at the end of the ascent (arrow). To account for both apoapsis-hold pitch and plane-change yaw, $F_1 = [1+\tan^2\beta+\tan^2\alpha]^{-1/2}$; $F_2 = F_1 \tan\beta$; $F_3 = F_1 \tan\alpha$. The result is delta V =5751 m/s to end of trajectory. Loss of thrust due to solar occultations are included.

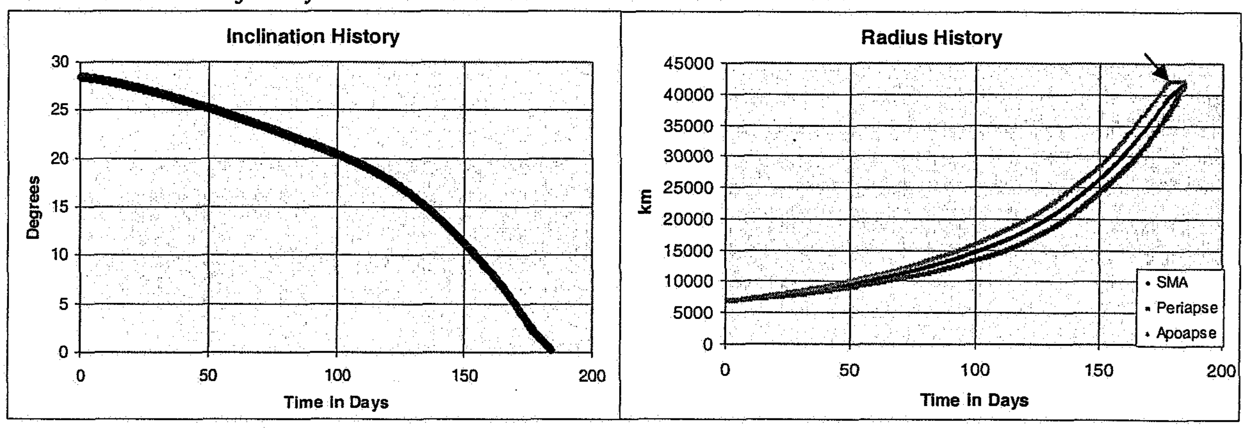


Figure 6. Plane Change Numerical Simulation Results

If one begins with an elliptic orbit, the eccentricity is enough that one should use a yaw steering law that includes eccentricity and argument of periapsis. This involves coordinate transformations. The best coordinate system in which to formulate the problem is an orbit coordinate system in which the x axis points through the line of nodes instead of through periapsis. The formulation used a momentum vector approach, where instantaneous plane change is given by the ratio of (1) orbital momentum vector rotation around the line of nodes due to thrust, to (2) the current orbital momentum vector. The result is:

$$\tan\beta = \frac{\lambda_p \cos(\theta+\omega)\mu}{-2\lambda_a a^2 V^2 T_2 \cos^2 \gamma T_1 T_2} \tag{3}$$

Fig. 7 shows a representative yaw steering case for this equation, with argument of periapse zero and 30 degrees. One would normally start such an orbit raising with ω = 0, but secular perturbations will cause the line of apsides to advance, so it is important to include the argument of periapsis in the formula.

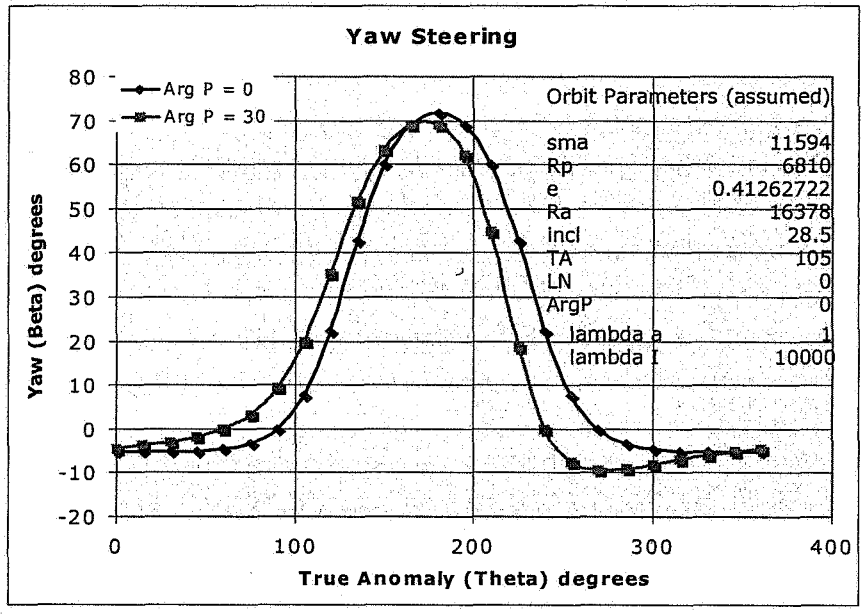
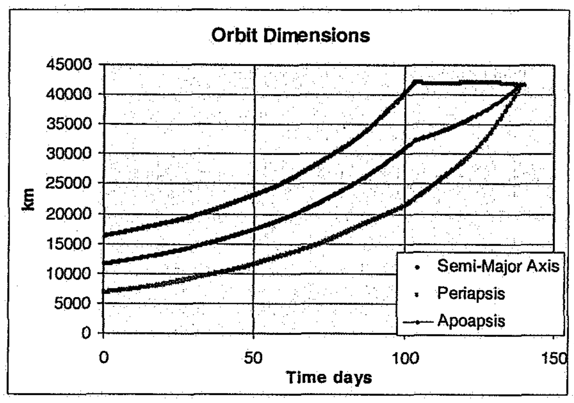


Figure 7. Typical Yaw Steering Commands Generated By Eq. 3

Fig. 8 shows results of a numerical integration with apogee hold applied when apogee reaches GEO. The orbit remains significantly elliptic until the apogee hold begins. As noted, the apogee hold algorithm is not highly efficient, as it involves large pitch angles up to 90° when far from apogee. One can compute the apogee hold pitch angle and apply a fraction of it throughout the trajectory, along with yaw steering. If the right amount is applied, the eccentricity goes to zero just as apogee reaches GEO. This is illustrated in Fig. 9. The lower edge of the velocity envelope for this trajectory, i.e. the velocities at apogee, is almost constant at about 3 km/s. The maximum yaw angle (near apogee) is also almost constant at about 60 degrees. Because this trajectory starts with more energy than a LEO-GEO near-circular trajectory, it must accomplish the plane change more quickly. The integrated delta V is about 4200 m/s compared with 5750 m/s for the case starting at LEO



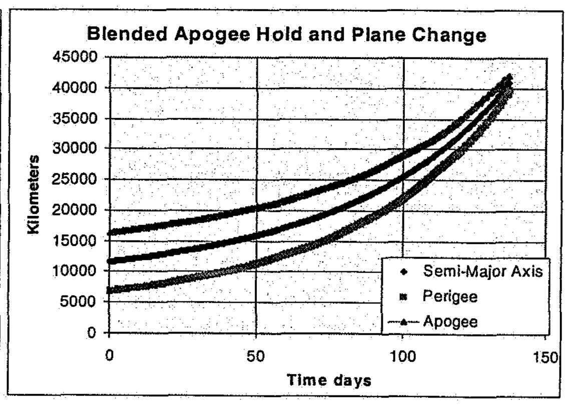


Figure 8. Elliptic Orbit Numerical Integration **Figure 9. Elliptic Orbit, Spread Apogee Hold**

E. Selecting Specific Impulse

The usual finding for chemical propulsion is that more specific impulse is better. That is somewhat affected by propellant density, so that in some applications the low density of hydrogen offsets (or more than offsets) its high Isp. For electric propulsion, if any limitations are placed on operating time, there is always an optimum Isp. Too low an Isp results in too much propellant consumption. Too high an Isp results in too much SEP inert mass, because the trip time limitation forces a certain thrust level, and as Isp increases, power increases causing SEP inert mass to increase. Fig. 10 illustrates this with a "simple" escape trajectory not using a planet swingby. A certain minimum T/W is required to avoid thrust from falling off faster than mass in a spiral. Delta V assigned was assumed to occur after Earth escape and was 50 km/s to produce a solar system escape with significant residual velocity. Obviously, with a Jupiter swingby, solar system escape would be much easier to obtain.

This figure also illustrates existence of an optimum electric propulsion Isp. The required mass factor, 10 kg/kWe or less for an entire SEP, is a very difficult challenge. Two optima are shown, one for payload equal SEP inert mass, which means payload increases as Isp increases because the SEP inert mass increases. The other fixes payload at 10% of the starting mass, and yields a much higher optimum Isp. Obviously, optimization ground rules matter.

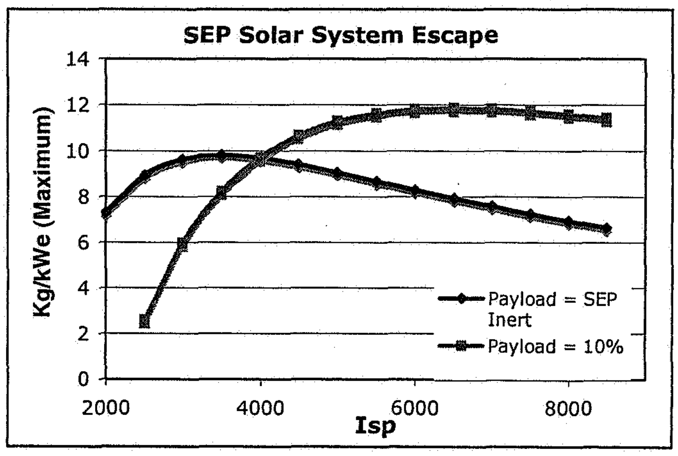


Figure 10. Optimization of SEP Isp

IV. Lunar Science Applications

A SEP can fly from an Earth orbit to the lunar vicinity and become captured into a lunar orbit, then spiral down to a low lunar orbit, as the ESA SMART-1 did. From low lunar orbit, a lunar landing can be accomplished. Some missions need significant amounts of electric power on the lunar surface; examples are missions where the lander needs to supply power to a rover, or includes an experiment to test or demonstrate in-situ resources utilization (ISRU) processing. If significant power is included, it is tempting to use it for electric propulsion on the way to the Moon.

A mission profile was analyzed in which a combined SEP-chemical propulsion system is delivered to low Earth orbit. The chemical system propels the vehicle to an elliptic Earth orbit, to reduce trip time as well as radiation/debris exposure. The SEP takes over and delivers the vehicle to a lunar intercept. Either the SEP or the chemical system accomplishes lunar capture and reaches a low lunar orbit. The chemical system performs the lunar landing. Upon landing, the SEP solar array is re-deployed to produce power for the surface mission. This may be particularly applicable to resource utilization experiments, such as oxygen production. A modest performance augmentation could be obtained by jettisoning the SEP thrusters and power processors before the lunar landing.

A tradeoff was run to determine how much of the total delta V should be delivered by the SEP, the remainder being delivered by the chemical system. The latter was assumed to use hydrogen-oxygen with Isp 450. Delta IV Heavy was assumed for delivery to the starting elliptic orbit; payload to a typical starting orbit 250 x 10,000 km is about 16,800 kg. While Delta IV could deliver the spacecraft to TLI, we assumed the lunar landing chemical propulsion provided all chemical delta V beyond the starting orbit. EP run time was held at a constant 210 days by adjusting the electric power level, except for low SEP deltaV cases. We arbitrarily assumed the power required on the surface is 25 kWe, so if 25 kWe yielded a trip time less than 210 days we accepted the shorter trip.

Fig. 11 shows results. This is a very complex trade, and we sought only general trends. SEP Isps were 1500 and 2000. The minimum chemical delta V occurs with a 1500 m/s boost to elliptic orbit and

2100 m/s for lunar landing. The maximum occurs with a 3000 m/s boost (to an elliptic orbit with apogee near lunar distance), most of the lunar orbit insertion, and the landing. SEP delta V ranges from a minimum of 900 m/s to a maximum of 6010 m/s. It was assumed that portions of the mission profile on chemical propulsion would be short duration.

Maximum payload and minimum cost were found with maximum use of electric propulsion. Use of SEP imposes an obvious trip time penalty, for which we did not set a cost penalty. The sequence in which we decreased SEP delta V was first to do less of the lunar orbit insertion with SEP, and then to increase the starting orbit apogee, which increased the launch delta V and reduced the SEP delta V. SEP delta V required to reach lunar orbit insertion from elliptic starting orbit was determined by numerical integration; a unit of launch vehicle delta V was "worth" 70% more than a unit of SEP delta V. SEP delta V to spiral down to a given lunar orbit altitude was estimated as 40% more than the chemical lunar orbit insertion. This difference is attributed to the difference in depth of gravity well, Earth re Moon. The result of this can be seen in the figure. Adding launch delta V decreases SEP power more rapidly and decreases net payload more slowly than adding chemical lunar orbit insertion delta V. The combination of these two effects causes a reverse in the unit cost curve. A better result would probably have been obtained by reversing the order of delta V exchange, first increasing starting orbit apogee and then decreasing SEP contribution to lunar orbit insertion. Also note that when we have electric power we are not using (because the surface power requirement sets the power level) the result is disadvantageous.

Three final caveats are appropriate. The first was mentioned above, that this is a complex trade and the present result is a trend analysis only. The second caveat is that we assumed equal value for a kg of payload gained (or launch mass reduced). In fact, there is great value in being able to use a smaller launch vehicle and comparatively much less value in saving mass if the same launch vehicle must be used. The third is that cost estimates were crude, and the result is sensitive to the cost of electric propulsion hardware relative to other costs. We used $1500/watt for the aggregate of solar arrays, power processing and thrusters, and a mission cost for everything else of $350 million. If solar electric propulsion is considerably more costly relative to other costs than we assumed, there may be no cost benefit to use of electric propulsion in this manner.

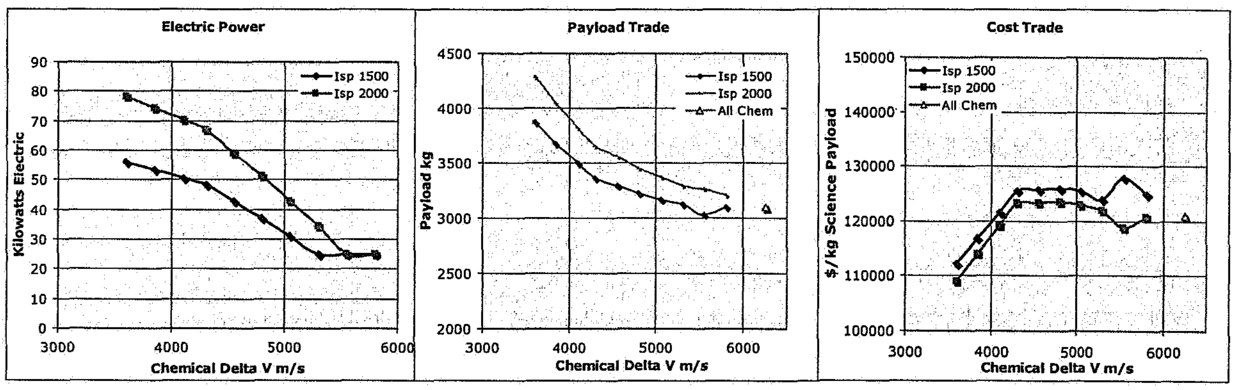


Figure 11. Results of Mixed Chemical-SEP Lunar Lander Trade

V. Lunar Exploration Applications

The applications described above represent near-term potential uses for SEP; the SEP total power levels are similar to, or moderately more than, current comsat practice at 25 to 60 kWe. Thruster powers are in the 10 kWe range, similar to test hardware for which there is extensive operating experience. However, lunar exploration and exploitation applications need hundreds of kilowatts and are therefore technologically more challenging.

SEP is not useful for directly transporting crews to and from the Moon because trip times are several months, while a high-thrust transfer to the Moon is a few days. Elements of crew missions can be transported; for example, an LSAM loaded with propellant, or a propellant tanker to replenish an LSAM, could be transported from LEO to lunar orbit. The crew would travel separately by CEV and rendezvous with the LSAM in lunar orbit. Lunar surface infrastructure with a cargo lander could be transported to lunar orbit by SEP. A rough rule of thumb is that use of SEP saves about 30% launch mass on a crew mission and about half on a cargo mission[8, 10, 11]. This varies with mission profile details, chemical propellants used, abort requirements and other factors. The launch vehicle payload requirement can be much less than for an all-chemical mission, since the SEP-supported mission will normally be divided into two or more Earth launches. The SEP must be re-usable for 4 or 5 trips to be economic. A SEP for this application needs to have 500 – 600 kWe power for trip time to lunar orbit 180 days or less.

A SEP-delivered LSAM would be different than the ESAS baseline[4] in two ways: (1) because the trip time from LEO to lunar orbit is several months, the LSAM descent stage would be designed with pump-fed LOX-methane descent propulsion, and (2) it would not be required to perform the crew mission LOI maneuver because it is not present on that phase of the crew mission. Methane is about 20 times easier, in terms of refrigeration power, to refrigerate than hydrogen, as is oxygen. Ascent propulsion could use pressure-fed storable propellants. If lunar oxygen production is implemented in the future, the storable ascent stage can be removed, the descent methane tanks somewhat increased in size to include the ascent requirement, and the descent stage, reloaded with lunar oxygen, would also perform ascent.

Fig. 12 shows a representative 500 kWe SEP configuration. Thrusters are at the rear, on deployable panels to increase roll moment authority. The payload, not shown, is forward and solar panels extend to the side. The concept is arranged to be compatible with EELV launch. Sun-tracking in low Earth orbit requires a lot of roll activity, but six-degree-of-freedom simulations[5] have shown the vehicle to be controllable. Fig. 13 shows an alternative modular design approach with thrusters distributed around the solar array periphery. Payload is installed behind the array at its center. Thrusters are two-axis gimbaled and individually pointed for thrust vector direction and attitude control. Controllability was addressed in ref. 6 and an updated report is planned for Space 2006.

VI. Lunar Exploitation Applications

Lunar exploitation means lunar missions with a practical product such as lunar oxygen delivered to a lunar orbit, lunar libration point, or highly elliptic Earth orbit. These missions are likely to demand significant delivery of infrastructure to the lunar surface. SEP can operate similarly to the case mentioned above, delivering an LSAM-derived cargo lander to lunar orbit and returning it to Earth after it unloads the cargo on the surface and returns to lunar orbit. As above, the LSAM would use pump-fed LOX-methane for descent and ascent. Fig. 14 shows performance for four options. The first is conventional expendable one-way chemical propulsion cargo delivery. The second replaces the translunar stage with SEP (again about 500 kWe); the SEP and the cargo lander are now re-used; the cargo lander, with a modest additional amount of propellant, can return empty to lunar orbit to rendezvous with the SEP

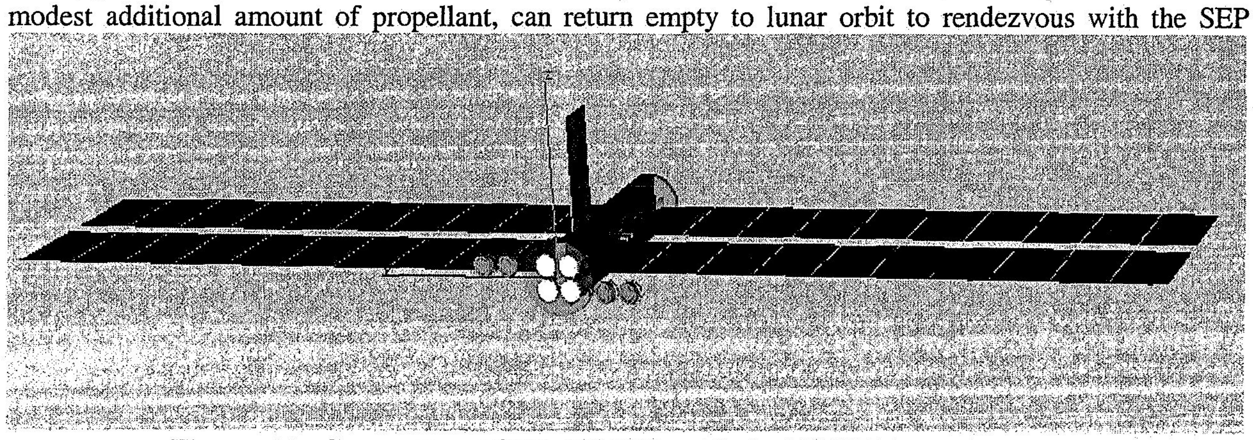


Figure. 12. Conventional SEP, 500 kWe with 4 - 125 kWe Hall Thrusters

and return to Earth orbit transported by the SEP. The third option supplies a modest amount of lunar oxygen to help return the lander to lunar orbit. The fourth provides (after the first landing) all of the lander oxygen. It loads oxygen on the lunar surface and methane brought from Earth in lunar orbit. All of these cases land 10 – 11 t. of cargo. Earth launch requirements are incrementally reduced with each technology advance.

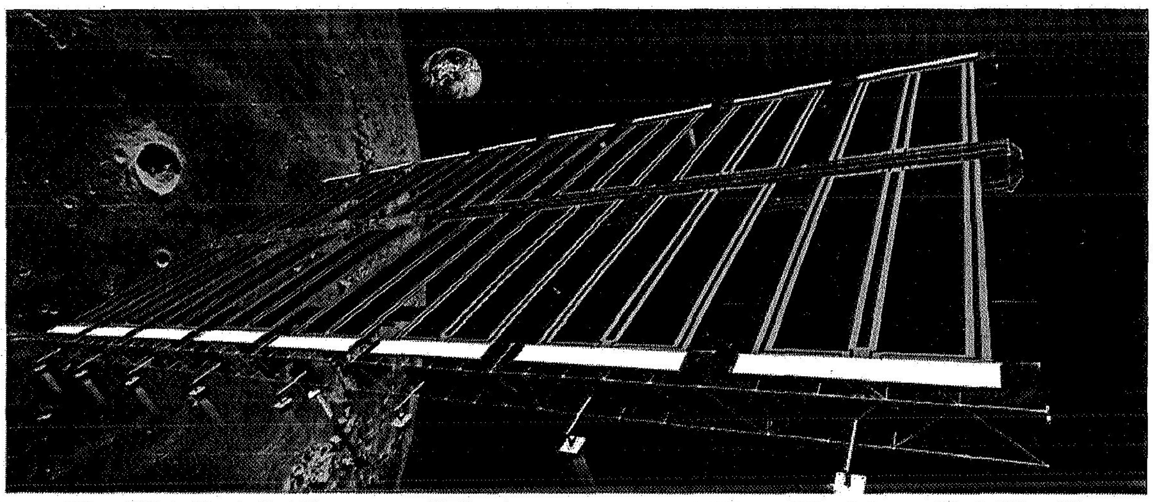


Figure 13. Distributed Thruster SEP Concept, 500 kWe with 16 – 50 kWe Thrusters (Redundancy Required to Ensure Ability to Use All Power), Payload Underslung (not shown).

VII. Planetary Science Applications

In this section, we describe a near-Earth SEP trajectory option that may improve performance for planetary science missions.

Deep Space I was something of a small template for SEP planetary missions. It had a single NSTAR thruster rated at about 2.5 kWe, and visited the comet Borrelly. Numerous studies of larger SEP spacecraft for outer planet missions have identified payoffs[7]. These concepts have typically been rated at about 25 kwe and used 3 to 5 NEXT[7] thrusters rated at about 6 kWe.

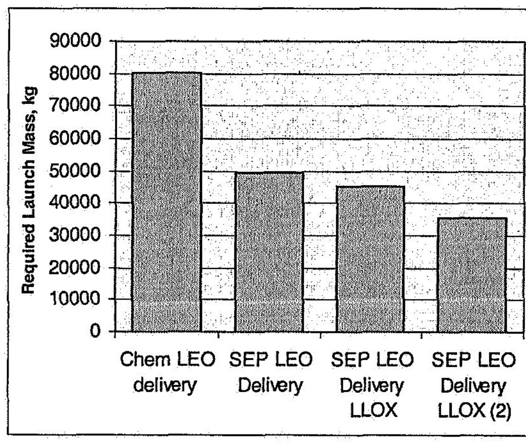


Figure 14. Lunar Transport Enhancement

Outer planet trajectories from these studies have similar characteristics. The SEP is launched to a C3 about 10 to 15. It attains a trajectory with aphelion somewhere beyond Mars, and adjusts its perihelion to about the radius of Venus. A coast period ensues, followed by a thrusting period through the perihelion arc, usually including a Venus gravity assist (VGA). (This type of trajectory works without VGA but works better with it.) Thrusting continues to about the distance of Mars, by which time the SEP has achieved enough energy to reach whichever outer planet is selected. The SEP then ceases propulsion and can be jettisoned. The scientific spacecraft bus carries aerocapture and/or chemical propulsion, or possibly radioisotope electric propulsion for maneuvers in the outer solar system since the SEP is too far from the Sun to provide useful thrust.

These trajectories, including the launch condition, are optimized by SEPTOP or a similar code. Because launch vehicle high-thrust propulsion has high leverage deep in Earth's gravity well, the optimizer increases C3 until the SEP leverage due to its high Isp exceeds the high-thrust leverage. Fig 15 graphs the relative leverage of high thrust and SEP propulsion versus C3. If one starts the optimization at C3 0 or greater, the result will be as noted, C3 10 – 15. However, there is a region at C3<0 where the SEP also has good leverage.

This was investigated with a simple spread-sheet analysis to test the hypothesis that a SEP start at C3<0 might have payoff. At first, a two-stage SEP was postulated, starting in LEO with the booster delivering to C3 about zero (delta V about 7 km/s) and a second stage delivering the delta V to go from C3 = 0 to outer planet injection; this delta V is about 18 to 20 km/s. The SEP booster could be re-usable. The Excel solver was used to do a simple constrained optimization (this was not a trajectory analysis; representative ideal delta Vs were used). It was found that the booster and mission stages had similar power level, and that Isp optimized about 2200 for the booster and 3000 for the mission stage. It was then realized that one could use Hall thrusters with two-step voltage supply to do the Isp switch and eliminate the booster, thus eliminating one set of solar arrays, power processors and thrusters. Crude cost estimating indicated about 25% to 30% cost and mass savings compared to the usual solution with launch to C3 ~ 10 to 15. The power level was ~ 50 kWe. This result needs to be checked with actual trajectories and more detailed systems analysis, but it appears on first look to have promise and might offer opportunity for a smaller launch vehicle in some cases. It would also be useful in a case where the launch vehicle does not have a high-performance upper stage.

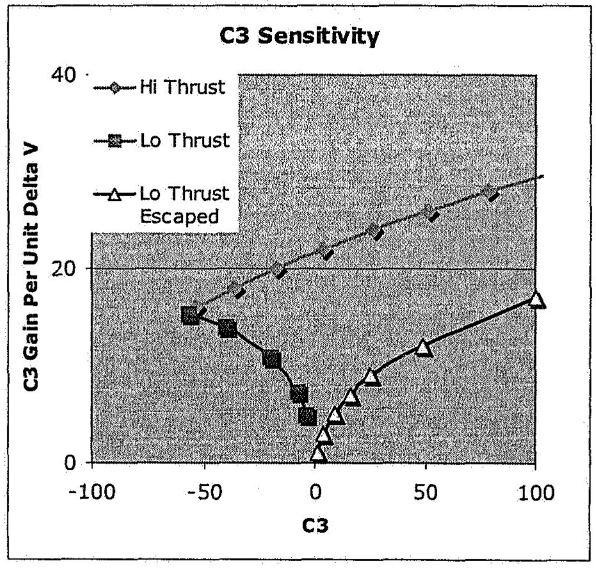


Figure 15. SEP Leverage vs. High-Thrust

VIII. Planetary Exploration Applications

Mars mission concepts have been a subject of study since the 1950s. A wide variety of propulsion concepts have been examined in various studies. Human Mars missions are a difficult challenge because of distance, time required, and cost. An archetypical Mars mission, with expendable heavy lift launch, expendable in-space transportation with chemical propulsion and aerocapture at Mars, may exceed 10 billion dollars recurring cost per mission just for the flight hardware and launches[10]. Nuclear propulsion may not be less cost, because the savings in launch cost will be consumed by the high cost of developing and implementing nuclear propulsion. The crew habitat for the interplanetary trip is literally a space station, and is a major contributor to the high cost of expending a Mars in-space transportation system.

SEP can reduce cost in two ways: (1) reduce launch mass, and (2) provide re-use of in-space transportation. Ref. 8 describes a 10-megawatt-class SEP interplanetary vehicle for transporting crews from a high Earth orbit to Mars and return. This system offered cost savings over the archetypical mission, mainly by making the interplanetary vehicle, including the crew habitat, re-usable. It was burdened by the high cost of developing and building the very large SEP. Subsequently, a better option was found; this option uses SEP operations only in near-Earth space. Several years ago, the Glenn Research Center investigated a range of SEP options for human Mars missions, including one wherein a SEP delivers a cryogenic-aerocapture Mars in-space transportation system to highly elliptic Earth orbit (EEO)[9]. SEP power required (to keep delivery time to the elliptic Earth orbit less than a year) is about 1 megawatt with lunar oxygen and about 1.5 megawatts without it. Delivery to EEO could be divided between two SEP vehicles, allowing use of a lunar exploration-size SEP at about 500 kWe.

Ref. 10 improved this architecture by retaining the aeroshell for the return trip, enabling re-capture into a highly elliptic Earth orbit, for re-use on the next Mars mission. It was improved further for this paper by adding lunar-produced oxygen; the in-space Mars transportation system receives its oxygen from the Moon. Fig. 16 is a mission diagram. Major operational steps are described in the Figure. In brief, the SEP delivers the interplanetary vehicle to an elliptic orbit with low perigee, near Earth escape energy. This greatly reduces the chemical stage delta V needed for trans-Mars injection. The improvements of ref. 10 and this paper further reduce the Earth launch requirements to resupply the mission.

Final improvement steps are to develop a re-usable Mars Lander/Ascent Vehicle (MLAV) operating on Mars-manufactured propellant, and move to a partially re-usable launch system (expected launch rates do not justify developing fully re-usable launch). Fig. 17 shows an estimate of per-mission cost reduction as these improvements are implemented. The cost reduction potential is dramatic. These estimates are based on references 10 and 12.

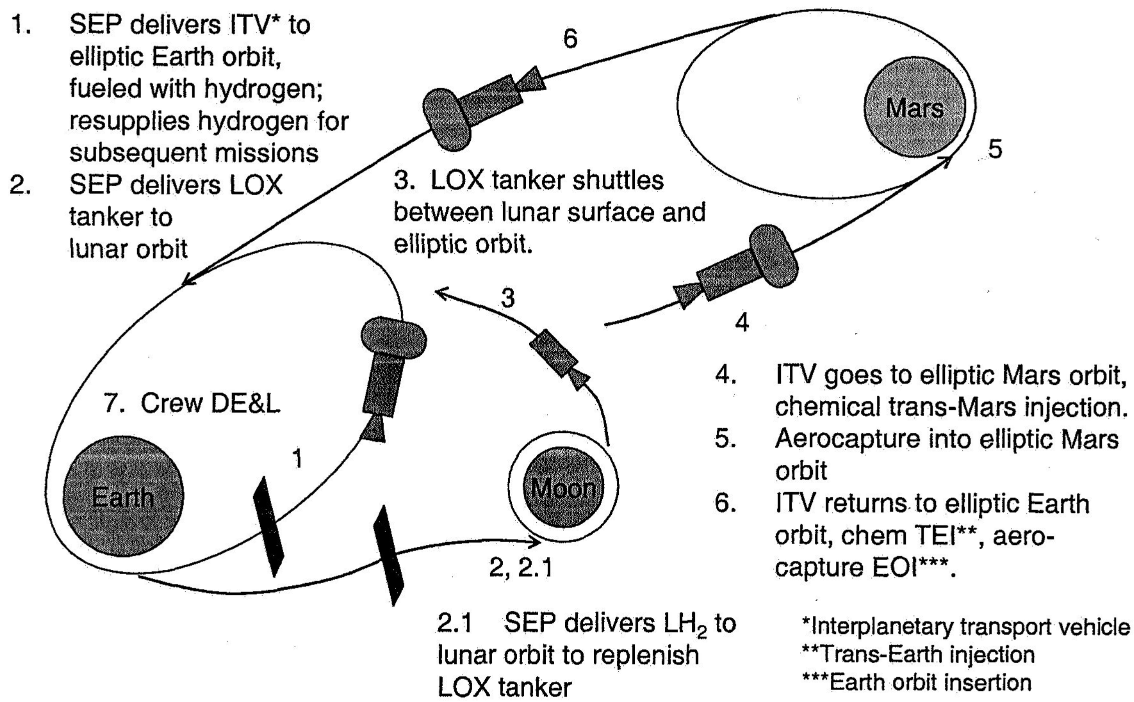


Figure 16. Leveraged Mars Mission Functional Diagram

IX. Conclusions and Recommendations

SEP systems offer much promise for improving mission performance and decreasing costs. A wide range of mission applications show benefits. There is a progression in power level from current missions at about 5 kWe, through lunar and planetary science applications at 20 – 80 kWe and orbit raising for communications satellites at 30 – 100 kWe, to support roles for human exploration and exploitation missions, at hundreds of kWe to a megawatt or more. Some of these applications admit to incremental mission architecture improvements, with accompanying reductions in cost. Solar array performance continues to increase, and concentrator systems promise to improve radiation resistance for orbit raising, so that the requisite re-use of orbit raising SEP "tugs" (needed for cost reduction) can be achieved.

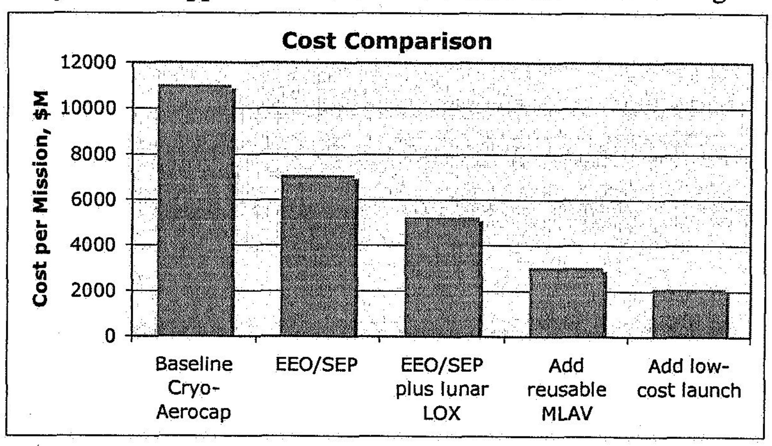


Figure 17. Cost per Mission Decreases with Technology

These mission applications include space science, space exploration, and present and future commercial uses of space. A coordinated technology advancement and mission technology infusion effort would benefit all users. Commercial customers are likely to have less risk exposure in terms of lost time, and may be first users of some advancements, as they have in the past. However, commercial customers are unlikely to lead the way in technology development.

X. Acknowledgement

The work described in this paper was funded by the In-Space Propulsion Technology Program, managed by NASA's Science Mission Directorate in Washington, D.C., and implemented by the In-Space Propulsion Technology Project at Marshall Space Flight Center in Huntsville, Alabama. The program objective is to develop in-space propulsion technologies that can enable or benefit near and mid-term NASA space science missions by significantly reducing cost, mass or travel times.

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