M09-0143 – Presentation

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

In Space Propulsion Technology Project
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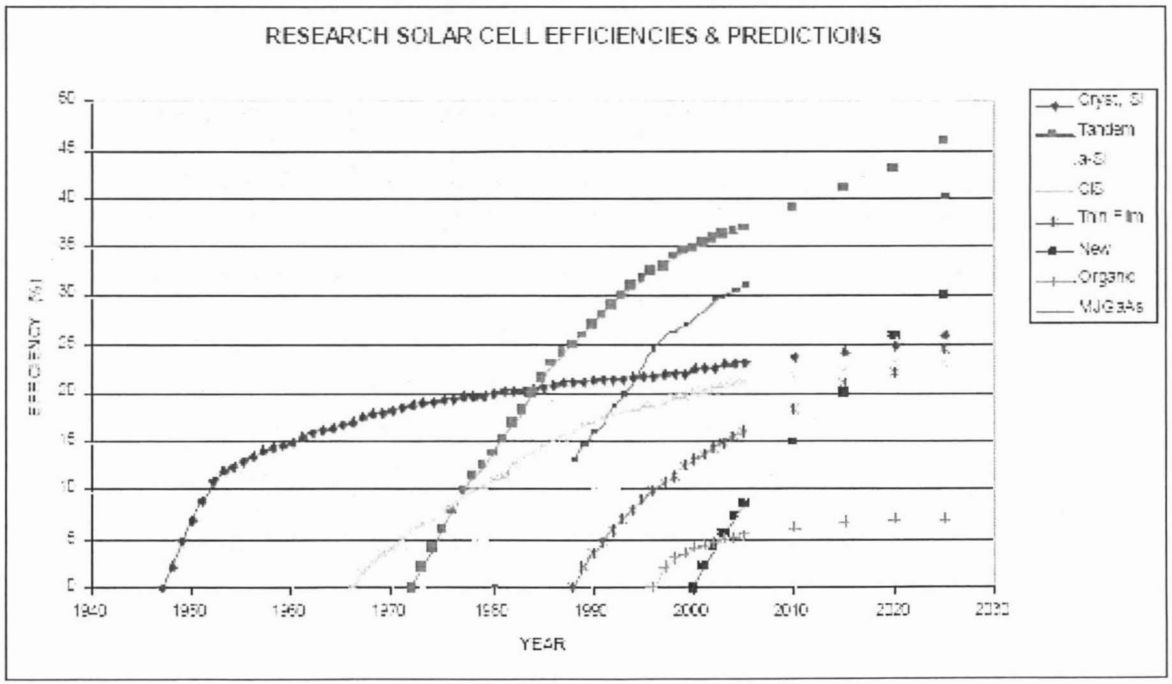
Growing Technology Base for SEP

- Solar-electric propulsion (SEP) is becoming of interest for application to a wide range of missions.
- Benefits of SEP are strongly influenced by system element performance, especially the power system.
- Solar array performance is increasing rapidly and promises to continue to do so for another 10 to 20 years (graph)
- Cost per watt is decreasing.
- Radiation hardness is increasing.

New concepts for designing SEP are emerging. These 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.



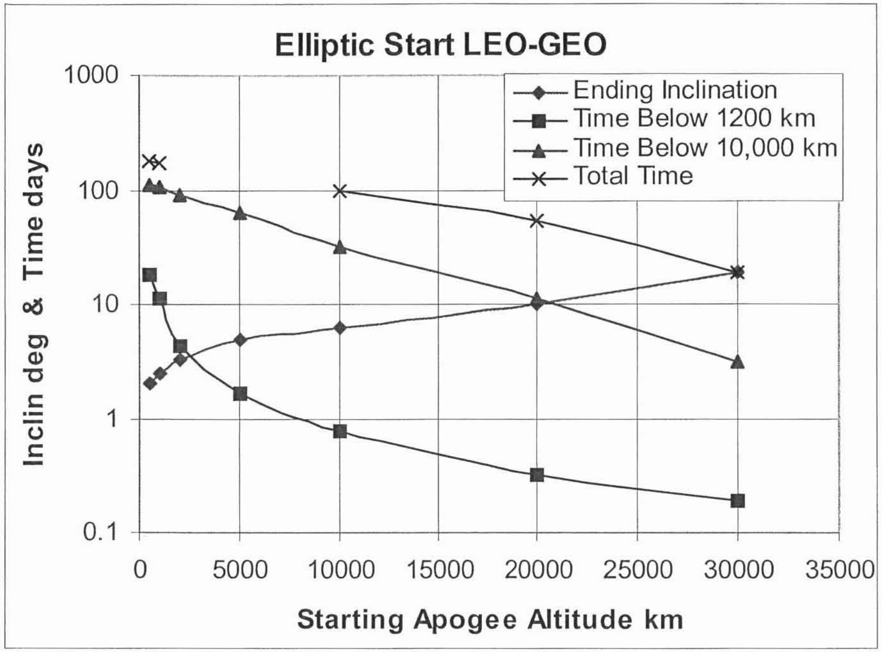
Why Use SEP?

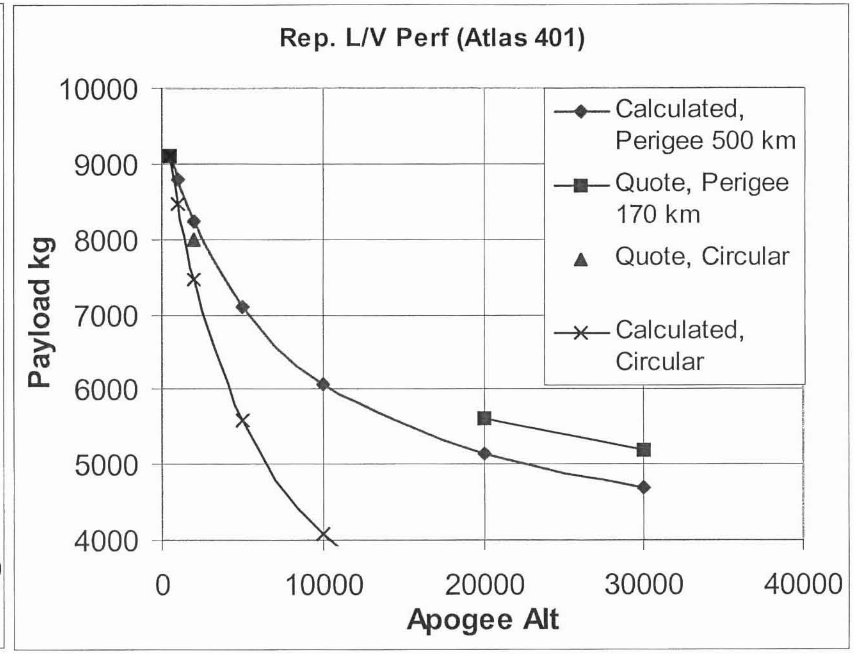
- **Application of SEP technology is favored when:**
  - (1) the mission is compatible with low-thrust propulsion: no need to produce accelerations comparable to planetary gravity fields and adequate time to achieve the mission ideal delta V;
  - (2) the mission needs high total delta V such that chemical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

SEP Mission Regimes

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.

Effects of Elliptic Starting Orbits

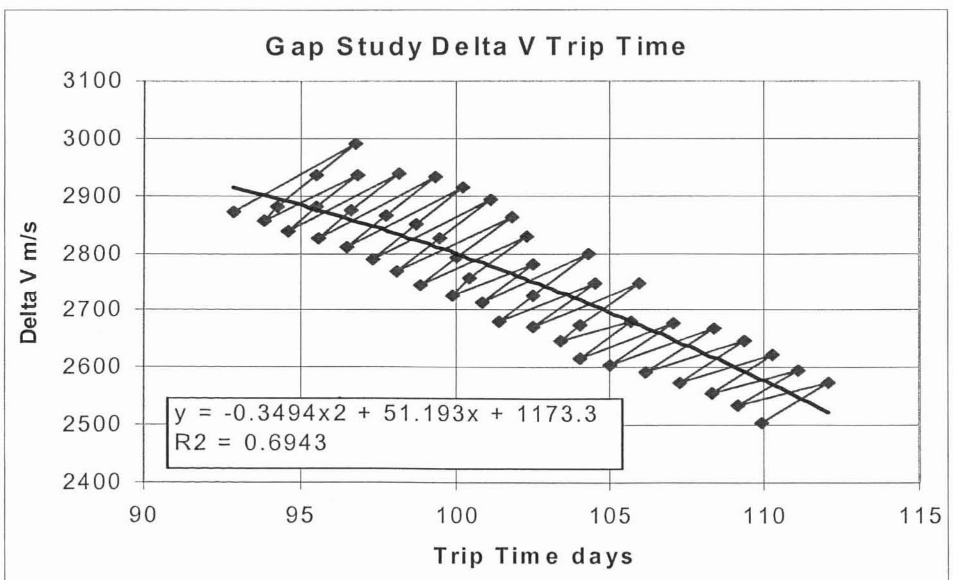


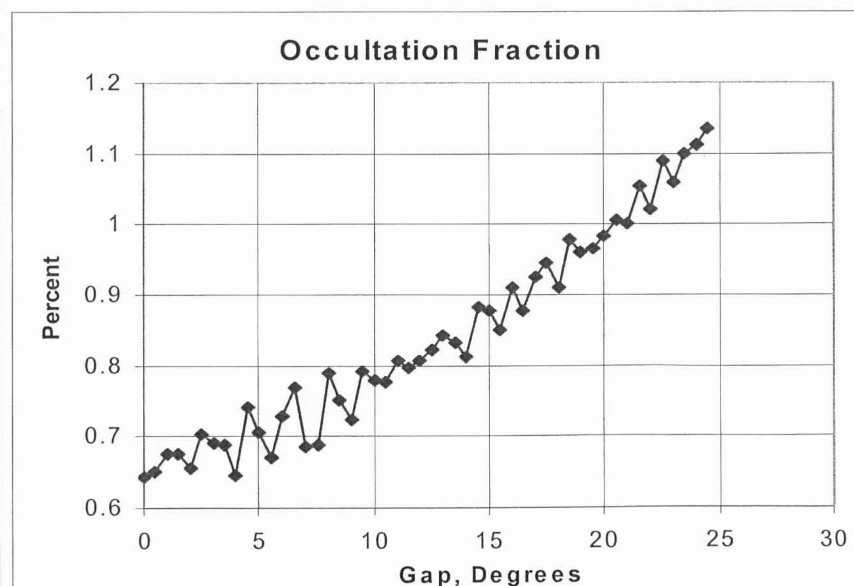


- Low circular starting orbit causes lengthy exposure to van Allen belts and significant periods in the LEO debris environment.
- Launch vehicles pay a high performance price for higher circular orbits
- Elliptic starting orbits pay off in reducing radiation and debris exposure time
- For all-chemical propulsion the GEO/LEO payload ratio is about 20%; even with an elliptic starting orbit, SEP is expected to double that

Orbit Raising Thrusting Strategy

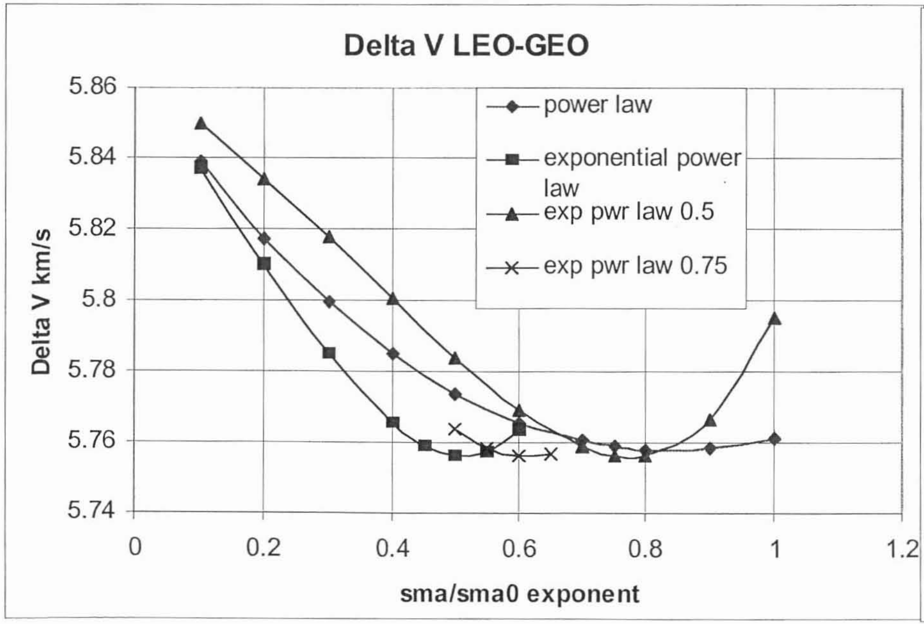
- Two questions: (1) thrust all the time or part of the time, and (2) thrust in the direction of flight or in some other direction?
- Apogee "gap", period of non-thrusting centered on apogee; thrusting at apogee contributes least to orbit raising. (Thrust pointed in direction of flight. About 15 days of added trip time reduced delta V by a few hundred m/s and could increase payload several percent.
- Irregularity of data caused by ellipticity of orbit and difference in contribution to apogee radius between perigee and apogee thrusting.

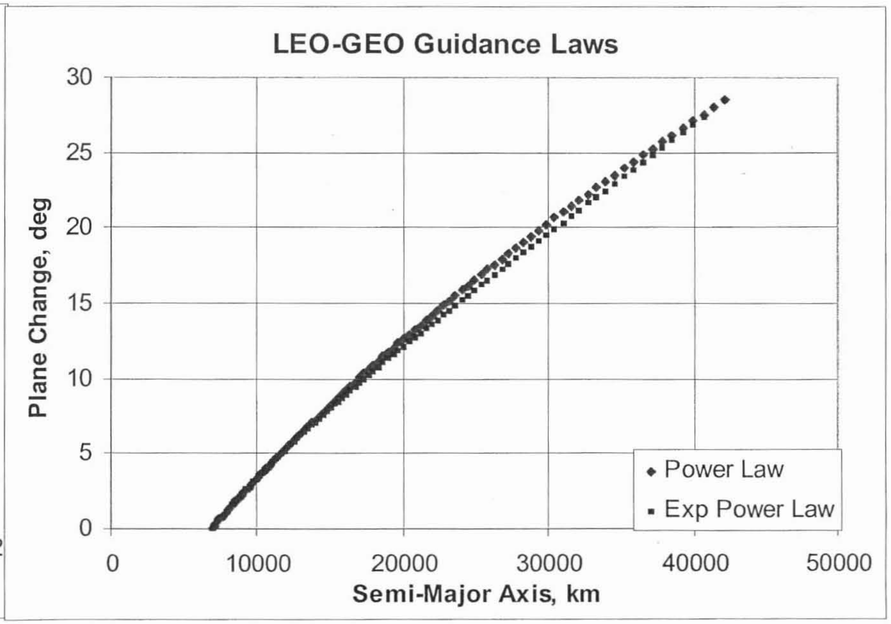




Plane Change During Ascent

- Orbit raising is often from an inclined starting orbit to zero inclination GEO.
- One can optimize how plane change is effected. We seek a control law practical for spacecraft GN&C. We find: $\tan \beta = \lambda_p \cos \theta/(2\lambda_a a)$ where λ_p is a constant and λ_a varies with a (semi-major axis). θ is angle around the orbit from ascending node and β is yaw angle.
- Tried forms $1/(\lambda_a a) = ka^\alpha$ where k and α are adjusted to give minimum ΔV and desired plane change, and $\lambda_a = k_1 \exp(k_2 a^\alpha)$, same general idea.
- Exponential power law is very slightly better than the simple power law





Plane Change with Elliptic Starting Orbit

- For elliptic starting orbit, eccentricity is enough that we need a yaw steering law that includes eccentricity and argument of periapsis.
- Derivation uses coordinate transformations. Best one to formulate the problem is orbit coordinate system where x axis points through line of nodes instead of periapsis.
- Used a momentum vector approach, where instantaneous plane change is given by ratio of (1) orbital momentum vector rotation around the line of nodes due to thrust, to (2) current orbital momentum vector. The result is:

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

New Terms:

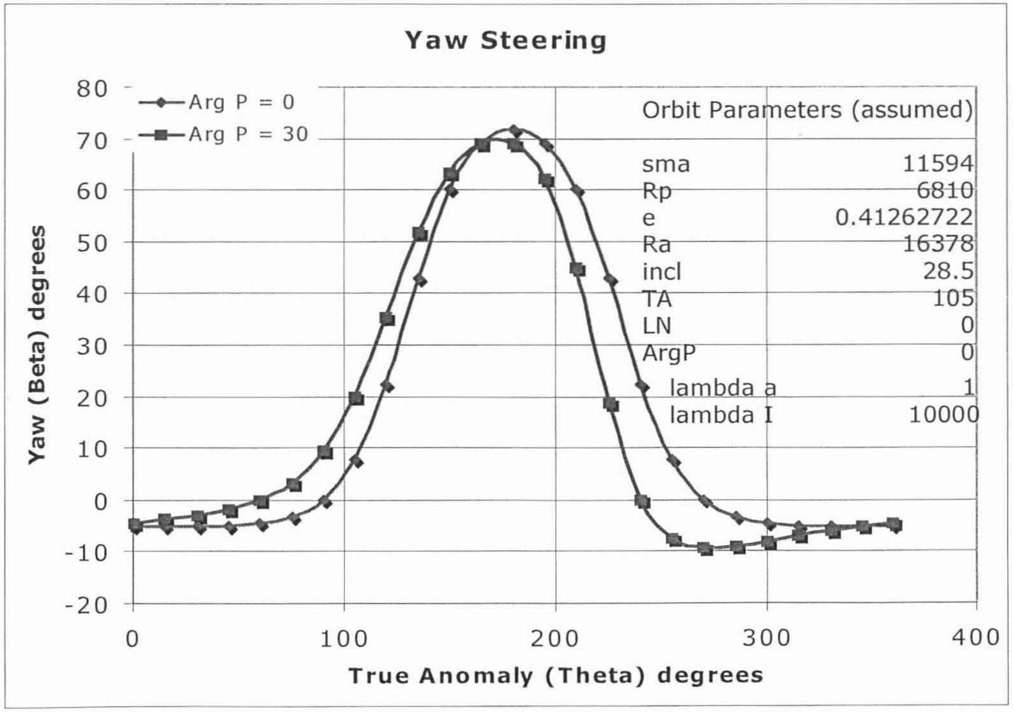
ω argument of periapsis

μ Earth geopotential

V SEP velocity

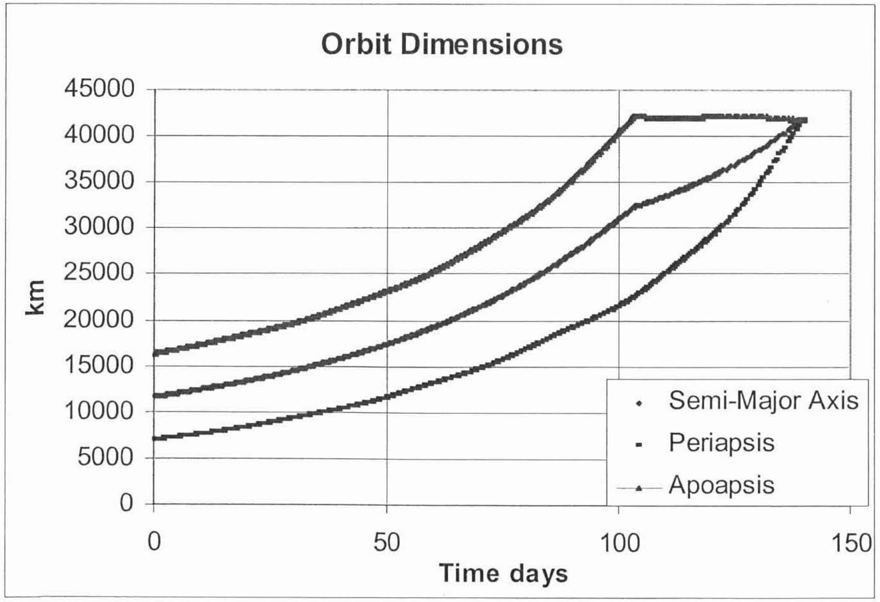
T_1 1 + e cos θ

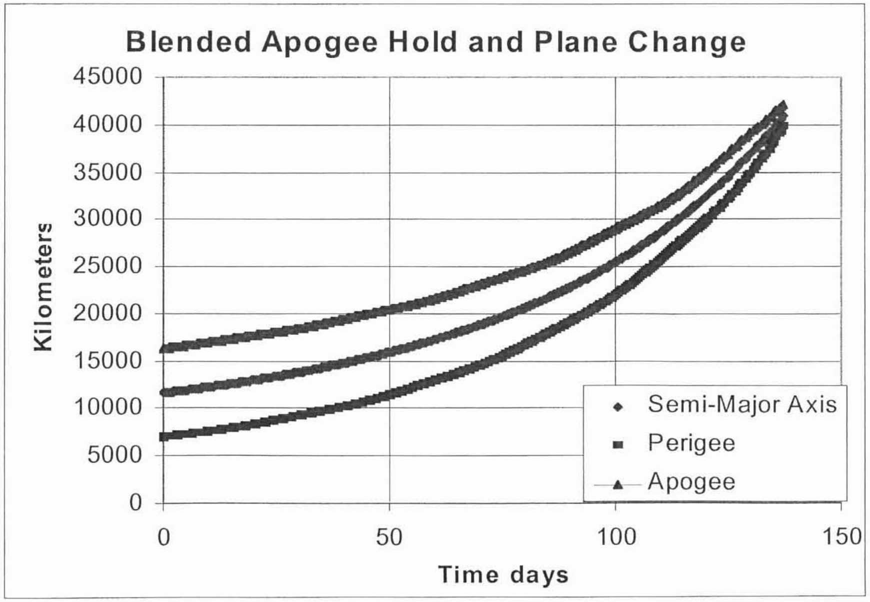
T_2 sqrt(1 + 2e cos θ + e^2)



Further Work Needed:

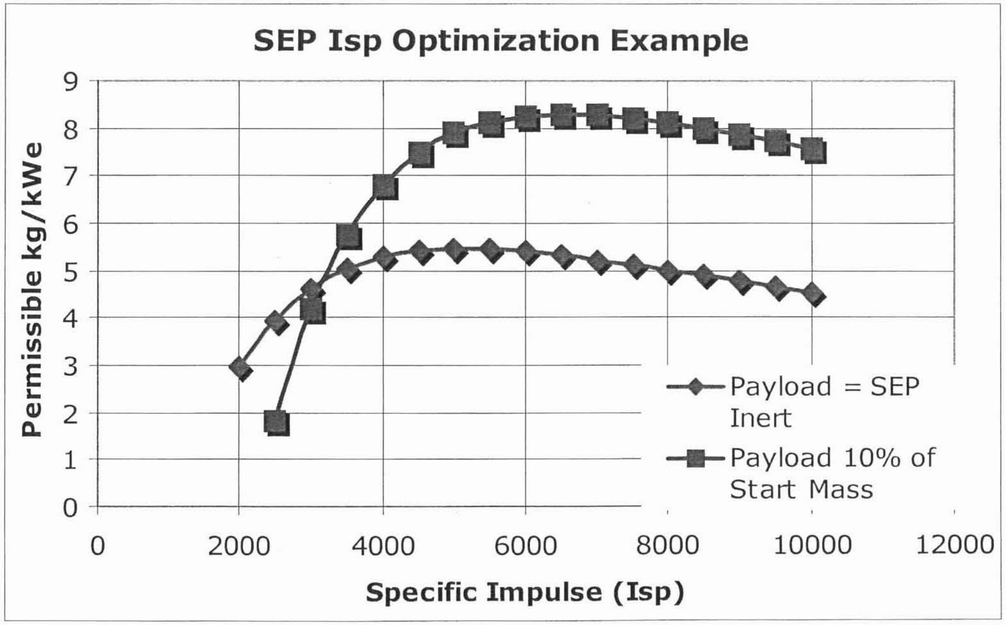
- When apogee reaches GEO, must hold it or otherwise circularize the orbit at GEO altitude.
- Apogee hold pitch modulation works but is inefficient. Blending the apogee hold (uses some fraction of apogee hold pitch) is better.
- Should derive a guidance law where eccentricity is also represented in the Hamiltonian so that circularization can be optimized.





Isp Optimization

- For electric propulsion, with limits on operating time, there is an optimum Isp. Too low Isp, too much propellant consumption. Too high, too much SEP inert mass.
- The graph illustrates with a “simple” escape trajectory not using a planet swingby. A minimum T/W is required to avoid thrust from falling off faster than mass in a spiral away from the Sun. Delta V assigned, 50 km/s, was to occur after Earth escape to produce solar system escape with significant residual velocity.
- The chartillustrates optimum electric propulsion Isp. The required mass factor, 10 kg/kWe or less for an entire SEP, is very difficult.

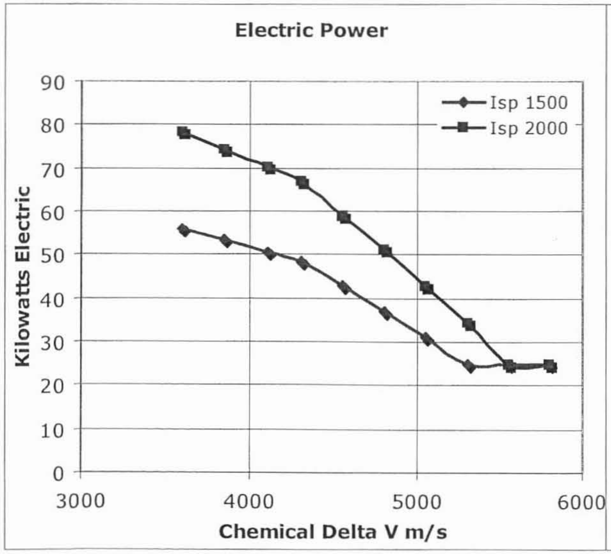


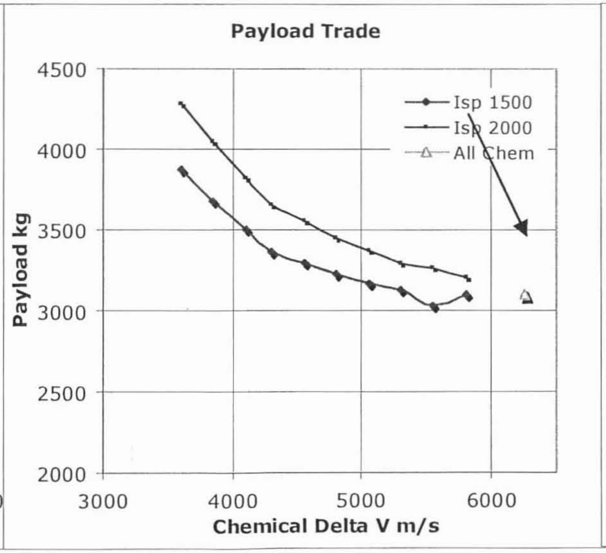
Lunar Science Application

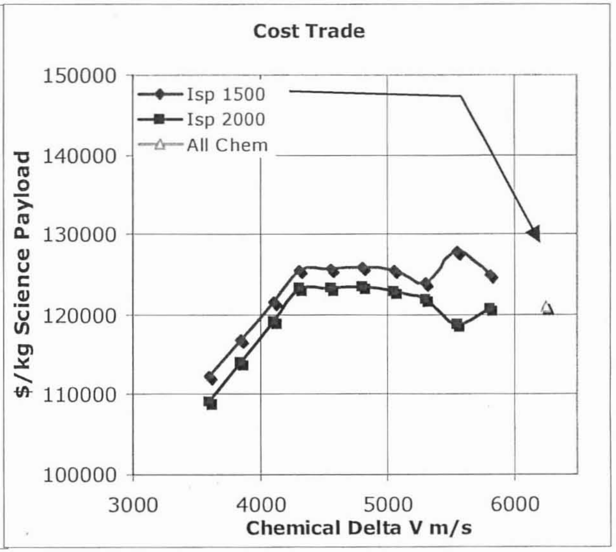
- **Mission profile, useful for ISRU experiments, e.g. oxygen production.**
  - Combined SEP-chemical propulsion system delivered to low Earth orbit.
  - The chemical system propels the vehicle to an elliptic Earth orbit, to reduce trip time and radiation/ debris exposure.
  - SEP takes over and delivers the vehicle to a lunar intercept.
  - SEP or chemical system does lunar capture to a low lunar orbit.
  - The chemical system performs lunar landing.
  - Upon landing, the SEP solar array is re-deployed to power the surface mission.
- **Modest performance augmentation (not analyzed) obtained by jettisoning SEP thrusters and power processors before landing.**
- **Tradeoff looked at how much delta V should be delivered by SEP.**
- **Chemical was assumed to use hydrogen-oxygen with Isp 450.**
- **Delta IV Heavy delivery to the starting elliptic orbit; payload to a typical starting orbit 250 x 10,000 km is about 16,800 kg.**
- **Delta IV could deliver to TLI, but we assumed lunar lander chemical propulsion provided all chemical delta V beyond starting orbit.**
- **EP run time held constant 210 days by adjusting power level.**
- **We assumed power required on the surface is 25 kWe. If 25 kWe yielded trip time less than 210 days we accepted that.**

Lunar Science Application Results

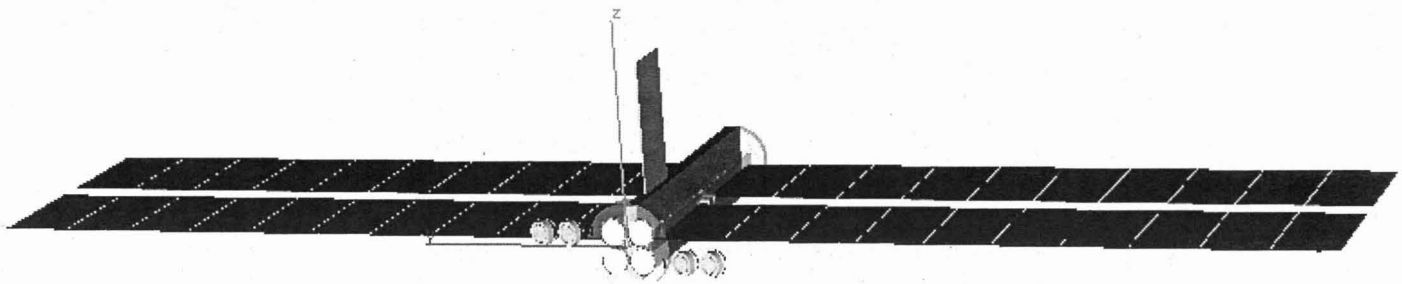
- **SEP Isps 1500 and 2000. Minimum chemical delta V occurs with a 1500 m/s boost to elliptic orbit and 2100 m/s for lunar landing. Maximum occurs with a 3000 m/s boost to an elliptic orbit with apogee near lunar distance, most of the lunar orbit insertion, and the landing. SEP delta V ranges from a minimum of 900 m/s to a maximum of 6010 m/s.**
- **Maximum payload, minimum cost with maximum use of electric propulsion**
- **Three final caveats:**
  - (1) complex trade, present result is a trend analysis only.
  - (2)Assumed equal value for any kg of payload. Great value only if a smaller launch vehicle can be used.
  - Cost estimates were crude; result is sensitive to cost of electric propulsion hardware. If solar electric propulsion is more costly relative to other costs there may be no cost benefit to this particular use of electric propulsion.



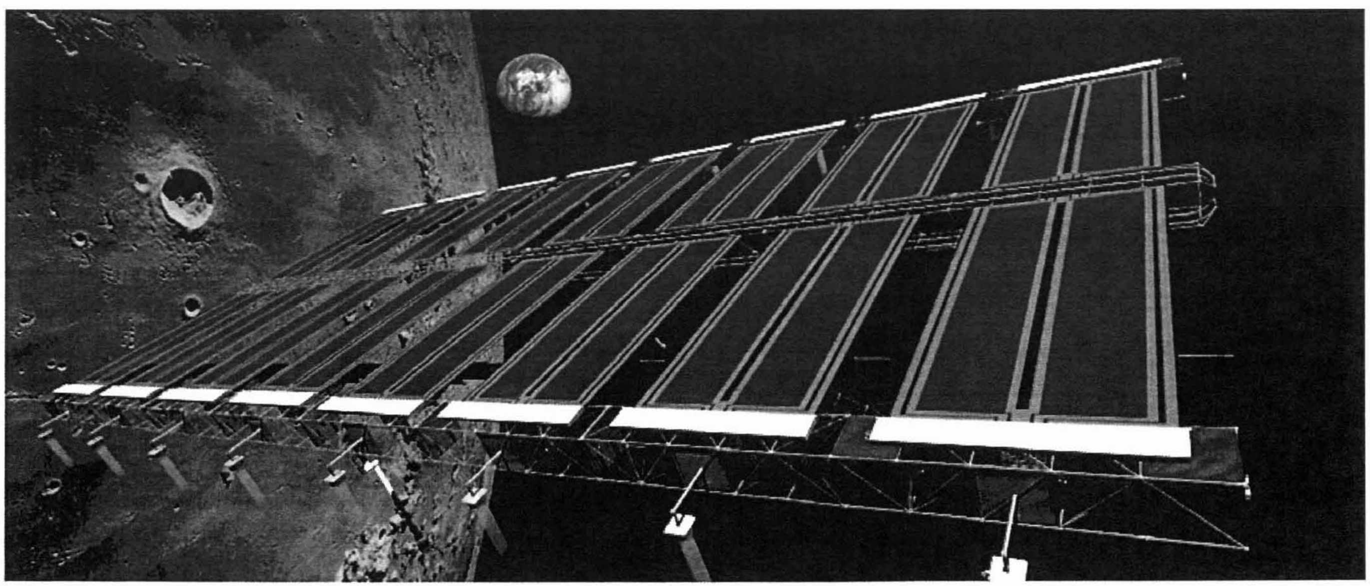




500 kWe SEP Concepts



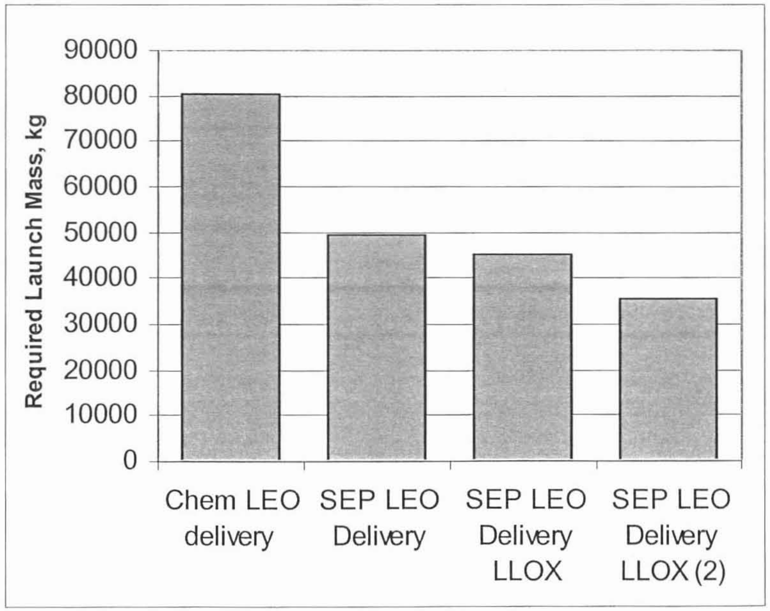
Conventional



Modular, Distributed Thrusters

Lunar Missions Requiring Heavy Cargo

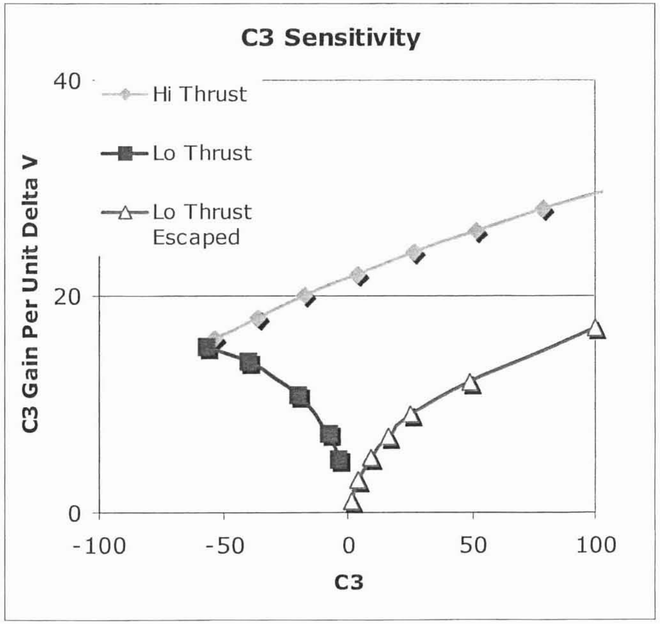
- **SEP can deliver an LSAM-derived cargo lander to lunar orbit and return it to Earth after it unloads the cargo on the surface and returns to lunar orbit. LSAM would use pump-fed LOX-methane or LOX-LH_2 for descent and ascent.**
- **Four options analyzed:**
  - (1) Conventional expendable one-way chemical propulsion delivery.
  - (2) Replaces translunar stage with 500 kWe SEP; SEP and the cargo lander are now re-used; cargo lander, returns empty to lunar orbit.
    - (3) Supply a modest amount of lunar oxygen to help return the lander to lunar orbit.
    - (4) Provide (after the first landing) all of the lander oxygen (after the first landing). Load oxygen on the lunar surface, fuel from Earth in lunar orbit.
    - All cases land 10 – 11 t. of cargo. Launch requirements incrementally reduced with each technology advance.



Planetary Science Missions

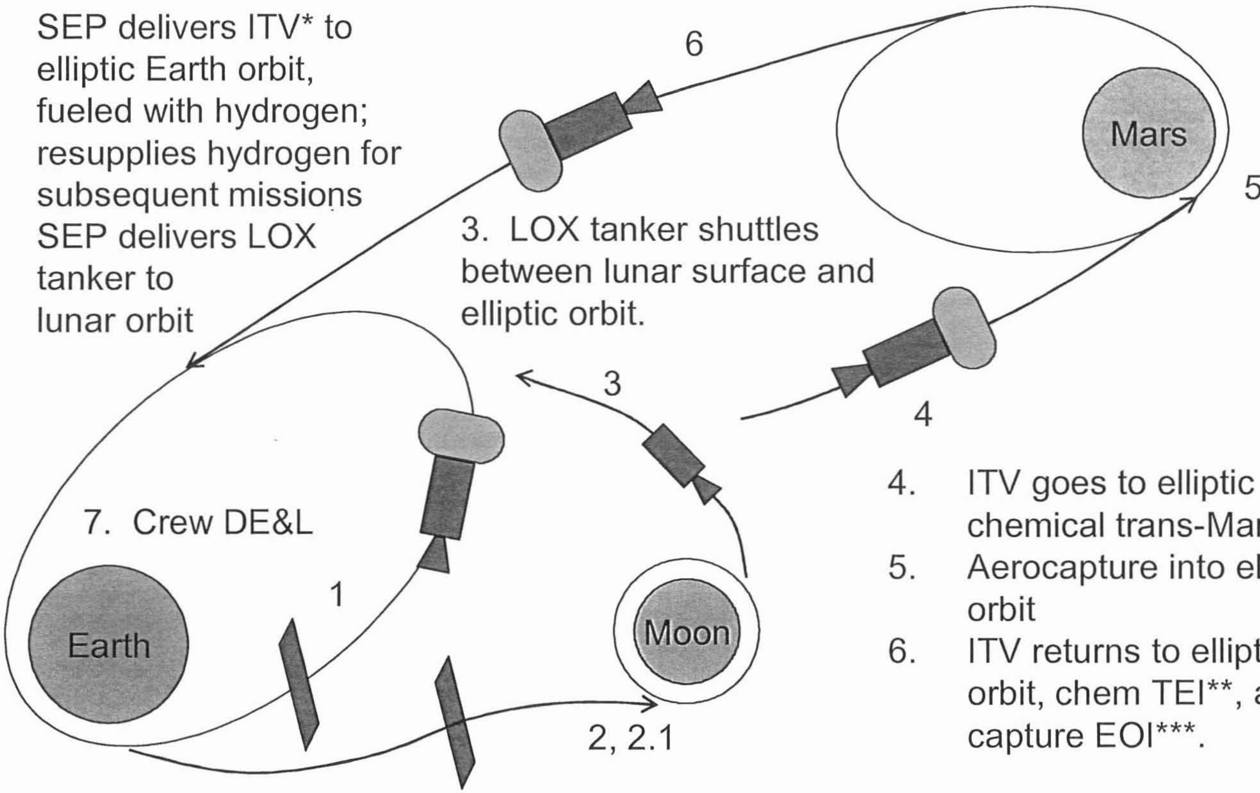
- Launch vehicle high-thrust propulsion has high leverage deep in Earth's gravity well; optimization increases C3 until SEP leverage due to its high Isp exceeds the high-thrust leverage. Graph shows the relative leverage of high thrust and SEP propulsion versus C3. If one starts optimization at C3 0 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.

- A simple test calculation showed that a SEP starting from elliptic orbit with C3 ~ -35 and spiraling to escape could deliver more payload than using the same launch vehicle to launch a smaller SEP to C3 ~ 15. The cost trade depends on SEP cost delta for larger & more power versus launch cost.



Planetary Exploration

- **SEP and other new technologies have high leverage for reducing human Mars mission transportation cost. Best mission profile we have found:**



1. SEP delivers ITV* to elliptic Earth orbit, fueled with hydrogen; resupplies hydrogen for subsequent missions
2. SEP delivers LOX tanker to lunar orbit

3. LOX tanker shuttles between lunar surface and elliptic orbit.

4. ITV goes to elliptic Mars orbit, chemical trans-Mars injection.
5. Aerocapture into elliptic Mars orbit
6. ITV returns to elliptic Earth orbit, chem TEI**, aero-capture EOI***.

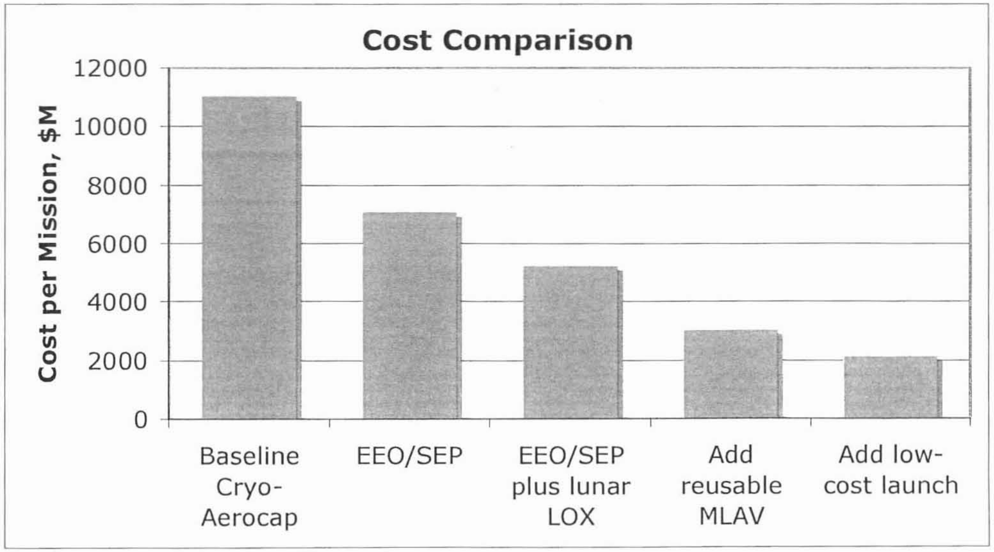
2.1 SEP delivers LH_2 to lunar orbit to replenish LOX tanker

*Interplanetary transport vehicle
**Trans-Earth injection
***Earth orbit insertion

Mars Mission Cost Results, Per Trip

- Elliptic Earth parking orbit enables interplanetary vehicle to return with its aerobrake and transfer habitat for re-use (2nd bar).
- Diagram of previous chart adds lunar oxygen, middle bar below.
- Production of LOX-methane propellant on Mars enables re-usable Mars lander/ascent vehicle (MLAV) based on Mars. Ascent of entire vehicle on Mars propellant; descent using propellant from Earth.
- Low-cost launch represents a partially re-usable launcher, flyback booster with expendable LOX-LH_2 upper stage.

- SEP for delivery from LEO to elliptic orbit is 500 kWe – 1 megawatt. More cost effective than a multi-megawatt SEP going to Mars and back because non-recurring investment is much less.



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