Results of Evaluation of Solar Thermal Propulsion

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ABSTRACT

The solar thermal propulsion evaluation reported here relied on prior research for all information on solar thermal propulsion technology and performance. Sources included personal contacts with experts in the field in addition to published reports and papers. Mission performance models were created based on this information in order to estimate performance and mass characteristics of solar thermal propulsion systems. Mission analysis was performed for a set of reference missions to assess the capabilities and benefits of solar thermal propulsion in comparison with alternative in-space propulsion systems such as chemical and electric propulsion. Mission analysis included estimation of delta V requirements as well as payload capabilities for a range of missions. Launch requirements and costs, and integration into launch vehicles, were also considered.

The mission set included representative robotic scientific missions, and potential future NASA human missions beyond low Earth orbit. Commercial communications satellite delivery missions were also included, because if STP technology were selected for that application, frequent use is implied and this would help amortize costs for technology advancement and systems development. A "C3 Topper" mission was defined, calling for a relatively small STP. The application is to augment the launch energy (C3) available from launch vehicles with their built-in upper stages

Payload masses were obtained from references where available. The communications satellite masses represent the range of payload capabilities for the Delta IV Medium and/or Atlas launch vehicle family.

Results indicated that STP could improve payload capability over current systems, but that this advantage cannot be realized except in a few cases because of payload fairing volume limitations on current launch vehicles. It was also found that acquiring a more capable (existing) launch vehicle, rather than adding an STP stage, is the most economical in most cases.

PURPOSE

The purpose of the evaluation was to assess suitability of solar thermal propulsion for in-space propulsion applications, by examining performance and probable cost to customers on a range of representative missions, compared to current systems.

BACKGROUND

The study was requested by NASA Headquarters, Code S, to evaluate solar thermal propulsion for the In-Space Propulsion Technology program administered by the Marshall Space Flight Center (MSFC). The study was assigned to SAIC under the ISTA contract, which supports In-Space Propulsion at Marshall. The study was administered by Les Alexander and Bonnie James of the MSFC In-Space propulsion organization. The study was initiated in late July 2002, with a completion date of September, 2002.

Solar thermal propulsion has been under technology development for about 30 years. The fact that hydrogen gas, heated to 2500 – 3000K and expanded through a nozzle, could deliver specific impulse (Isp) in excess of 800 seconds, was well-known through demonstrations in the nuclear rocket program. Solar furnaces are known to reach this temperature range. It was seen as likely that a solar thermal propulsion system could reach much higher efficiency in converting energy of sunlight to thrust that is possible with solar electric propulsion. The reason is that concentration of sunlight onto a thruster, thereby heating hydrogen, might have much higher efficiency than converting sunlight to electricity by solar arrays and powering an electric thruster with the electricity. This higher efficiency, it is argued, would compensate for the lower Isp, making solar thermal propulsion potentially competitive with solar electric propulsion and capable of much shorter trip times.

MISSIONS AND REQUIREMENTS

Missions were selected to suit the objectives of the assessment. A list of missions with reasons for selection is given in Table 1. Estimates of performance requirements are given in Table 2.

The current In-Space Propulsion (ISP) technology program is sponsored by Code S; therefore the study focused on representative Code S missions. Commercial communications satellite delivery missions were included because if STP technology were selected for that application, frequent use is implied and this would help amortize costs for technology advancement and systems development. The "C3 Topper" mission is a case where the STP is relatively small. It does not present a problem for payload fairing volume, and its competition is probably solid propellant, with Isp less than 300 seconds. The HEDS gateway was selected because it is a mission application for which new in-space propulsion development is needed regardless of the technology selected, and STP is not at a non-recurring cost disadvantage relative to other systems. The "no-hydrogen" application to an RLV upper stage is a different competitive environment than ELV launchers because the STP does not compete with developed cryogenic upper stages. In this case the STP cannot use hydrogen but is competing with other systems that also cannot. The STP would probably use ammonia as propellant; methane is possible but it is cryogenic and much more flammable than ammonia.

The payload masses were obtained from references where available. The communications satellite masses represent the range of payload capabilities for the Delta IV Medium launch vehicle family. The payloads cited in Table 2 are, in all cases, the mission payload and do not include apogee propulsion. Communications satellite payloads are often cited in terms of geosynchronous transfer orbit (GTO) but the figures here are payload to the mission orbit, geosynchronous equatorial orbit (GEO).

Table 1: Tabulation of Missions Evaluated for STP Application

• NGST	• Representative small science payload (to ESL2)
• Space Interferometry Mission • Terrestrial Planet Finder	• Representative medium science payload (to ETSO) • Representative large science payload (to ETSO)
• Medium GEO Comsat	• High-demand commercial payload
• Large GEO Comsat	• High-demand commercial payload (larger)
• C3 Topper for outer planets missions	• Smaller STP leads to less volume concern
• HEDS L1 Gateway	• Large payload for EML1 which requires in-space propulsion development
• RLV Upper Stage	• If "no hydrogen" safety constraint applied, STP with ammonia may be competitive

Table 2: Top-Level Requirements for Selected Missions

Description	Destination	Payload Mass (kg)	Remarks
NGST	Earth-Sun L2	1400	From ISP Requirements Matrix
Space Interferometry Mission	Earth Trailing Solar Orbit (ETSO)	3900	From ISP Requirements Matrix
Terrestrial Planet Finder	Earth-Sun L2	4800	From ISP Requirements Matrix
Medium GEO Comsat	GEO	1900	L/V capability less apogee motor
Large GEO Comsat	GEO	3000	L/V capability less apogee motor
C3 Topper	Outer Planets	300 – 1500 kg	Generic capability
HEDS L1 Gateway	Earth-Moon L1	24,000	JSC HEDS DRM briefing
RLV Upper Stage (Non-Hydrogen)	GEO or C3=0	5000 kg or more (NEP)	Rationale is safety

An existing ISP requirements matrix was interrogated to obtain destination and payload mass data for the Code S payloads. The GEO comsat masses represent the smallest and largest Delta IV Medium options. Except for fairing volume considerations, STP upper stages would deliver more payload on the same launch vehicle, but these masses were considered representative. The C3 Topper was examined generically. Scientific payloads for outer planet missions, from the ISP requirements matrix, range from about 300 kg for small, simple payloads such as planetary flyby payloads, up to 1500 kg for a Titan orbiter/lander. Even larger payloads may be of interest at a later time. Examples of greater payload requirements, presently not very quantified, are a large Europa lander intended to penetrate Europa's ice to search for the putative ocean below, and a Titan sample return mission. The HEDS L1 Gateway mission payload was obtained from a JSC planning presentation. The Gateway is a small habitable space station. The RLV upper stage mission presumes that these payloads will utilize the launch capability of an RLV. Smaller payloads may also be of interest. One such case is launch of an experimental nuclear electric propulsion (NEP) stage to LEO with an STP stage designed to transfer the NEP stage to C3=0 so that the nuclear propulsion system is not started in Earth orbit. This case requires an estimated payload mass to C3=0 of 5800 kg.

MISSION PERFORMANCE AND COST ANALYSIS

Payload Performance

A performance baseline was created for application to the GEO and Earth escape missions. These missions are similar in that both require expanding an initially circular orbit to a highly elliptic orbit, for GEO with apoapse at 42,164 km and for escape or Earth-Moon L_1 (EML1), essentially at infinity, i.e. $C3 = 0$. For EML1 the C3 is actually about -2 km^2/sec^2 but this is essentially the same from a delta V viewpoint. For the GEO missions, an apoapse delta V about 1800 m/s is required; for Earth escape no apoapse maneuver is needed, and for insertion at L_1 the maneuver is about 650 m/s. High thrust systems can get to L_1 or L_2 via a powered lunar gravity assist for apoapse maneuvers (2 required) totaling about 250 m/s but STP does not have high enough thrust-to-mass ratio to perform the gravity assist thrusting maneuver.

The simplest way to fly from LEO to these destinations with STP is continuous thrusting. The result is a spiral path away from Earth with substantial G losses. STP does not have high enough Isp to accept these losses; its payload performance would be less than that for conventional chemical propulsion and there would be no benefit to using STP. Glenn Research Center provided an example continuous-thrust trajectory. Their mission profile assumptions were as follows:

- Three Phase LEO to GEO Transfer
- Spiral out from 500km altitude to approximate GEO radius
- Circularize using a maximum-eccentricity rate change steering law
- Plane change to zero inclination using a discontinuous-thrust inclination-change control law
- Decreasing the aggressiveness of the inclination change reduces propellant mass at the expense of trip-time (see Figure 1).

While this isn't an optimal transfer, it will not be far from an optimal result.

5% margins were added to trip time and propellant expenditure to account for small deviations in the final semi-major axis (±100km) and inclination (±1□).

The results, illustrated in Figure 1, show delta V about 6.2 km/s versus about 4.2 for a high-thrust system.

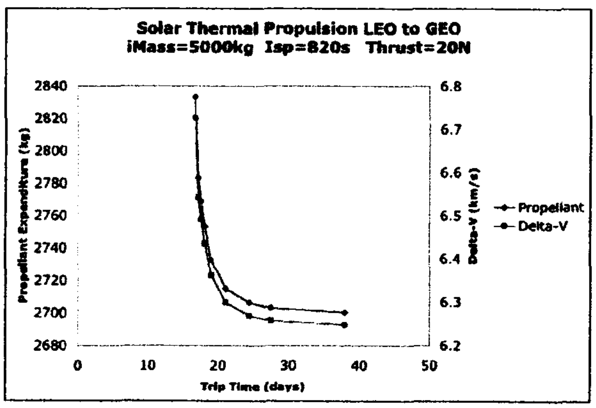


Figure 1: STP Spiral Delta Vs Provided By NASA Glenn Research Center

If the STP operates with intermittent burns near periapse, gravity losses are minimized and the STP can approach the delta V of a high-thrust system. The price for this is increased trip time. The question, clearly, is how much of a trip time increase must be incurred. This, in fact, was the motivation for the energy storage STP concept: one could collect solar energy all around the orbit and deliver it quickly near periapse. Also, if solar energy collection is discontinued during thrusting, simultaneous pointing to the Sun and of the thrust vector is not required, and the STP overall configuration is simplified. However, the very poor demonstrated efficiency of the storage concept (due to heat leak out of the storage system) in early tests led us to doubt its viability.

In the time available for the assessment study, rigorous optimization of intermittent thrusting was not possible. Such an optimization would constrain trip time and minimize delta V within that constraint. We approximated this by adopting a thrusting program that is arguably near-optimal, and evaluating the trip time. The thrusting program uses pitch angle modulation to hold periapsis constant during apoapse raising and to hold apoapsis constant during periapse raising. This relies on the thrusting effects shown in Figure 2. If pitch modulation is not used, the periapsis thrust intervals will raise periapse, resulting in g losses. As thrusting periods are increased, the g losses become greater. The upper limit is continuous thrust as described above. The lower limit is very short periapsis thrust periods and very long trip times. A true optimum is expected to let the apsides increase slightly, reducing pitch angle losses. The pitch modulation decreases thrust effectiveness; for this study an integrated thrust effectiveness of 90% for periapse maneuvers and 95% for apoapse maneuvers was selected. This does not yield optimal time-constrained transfers but was selected for expediency and ability to approximate optimal performance. Integration results for transfer to GEO are shown in Figure 3. Note that this result has a different thrust than assumed for Figure 1; this should be taken into account when comparing trip times. Each plot point in Figure 3 represents one thrusting period.

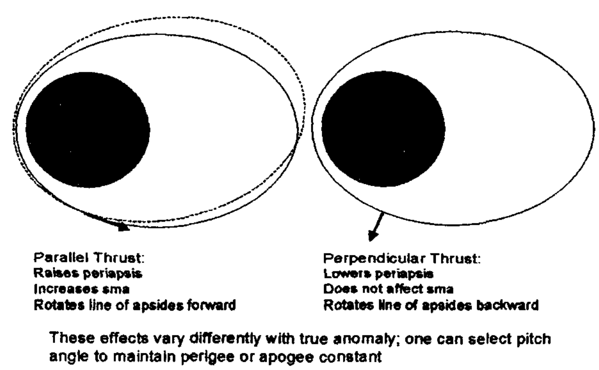
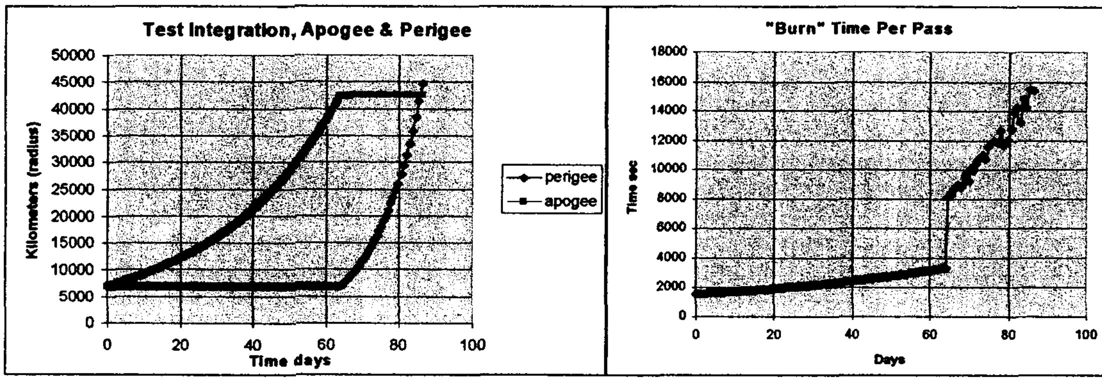


Figure 2: Pitch modulation thrust effects

For escape and libration point missions we assumed the same net effectiveness to obtain delta V for periapse maneuvers, and assumed no gravity losses for maneuvers at the destination.

These delta Vs were used with an STP mass estimating model to develop STP performance capabilities for the missions. Specific impulse for STP thrusters was estimated at 811 seconds, assuming (a) 2800K radiation temperature in the absorber cavity (this presumes collection of the concentrated solar energy in a cavity absorber, and radiative transfer from the absorber to the thruster), (b) 2700K thruster wall



Start mass 5000 kg; Thrust 12 N; Isp 811
Effectiveness criterion set 90% perigee, 95% apogee
Delta V 4350 m/s; does not include plane change; with plane change ~ 4600 m/s
Bottom Line: Trip time and delta V are OK but longer than desirable

Figure 3: Results of integration for LEO-GEO transfer

temperature, (c) 2600K hydrogen temperature, (d) nozzle area ratio 20, (e) 10% energy loss to viscous dissipation in the nozzle, and (f) a 10 degree average flow divergence angle exiting the nozzle.

High-energy missions to the outer planets have generally used multiple gravity assists to attain the trajectory energy needed to reach these destinations in reasonable time. The Pioneer and Voyager spacecraft were launched directly to Jupiter and used Jupiter and other gravity assists to continue on to the outer solar system. More recent missions (Galileo, Cassini) have used multiple inner planet gravity assists to get to Jupiter. It is possible to launch directly to the desired high energy, and this has been discussed as one option for a Pluto flyby. It is also possible to use electric propulsion, probably with a single Venus gravity assist, to perform these missions without requiring the launch vehicle to attain very high launch energy.

A major reason for interest in direct, rather than gravity assisted, trajectories is that Jupiter is not always in a position suitable for gravity assist to the planets of the outer solar system. Jupiter is available for a launch to Pluto in 2004, and offers a slight assist in 2006. It is then out of position for about 10 years.

Existing launch vehicles are tailored for the GTO market. They perform launches to LEO well, and can achieve C3 up to 20 – 40 km^2/sec^2 fairly well. Above this energy range their payload capability declines rapidly and goes to zero before C3 100. The reason is the relatively high inert mass of the upper stage, which starts before orbital velocity is reached.

For high energy, the usual solution is a solid rocket motor (SRM) upper stage as a "C3 topper". Existing-design spacecraft SRMs are suitable. This assessment asked whether STP could fit this application.

Since the total payload in the shroud is much less than the design value, STP's low density is almost certainly not a problem. Its high Isp is a benefit. Unfortunately, STP is at a disadvantage because of its low thrust. The SRM C3 topper delivers its delta V deep in Earth's gravity well and STP cannot, because its burn time is at least many days, while the time to essentially exit Earth's gravity well is less than a day at C3 30 to 40.

Since the gravity well advantage is a function of current and target C3, a high Isp system may have an overall advantage even if it cannot take advantage of the gravity well. This is partially illustrated in Figure 4. The Figure shows the differential advantage as the increment in "hyperbolic excess velocity" per unit delta V. The ratio is one for delta V outside the gravity well. Note that C3 is just the square of the hyperbolic excess velocity. At C3 30 to 40, the gravity well advantage factor is about 2. One may expect that a system with Isp 800 operating outside the gravity well could have an advantage over one with Isp 300 operating in the gravity well.

A spread-sheet analysis was constructed to examine parametrically the performance of an STP C3 topper compared to a solid propellant motor C3 topper. The STP was assumed to operate entirely outside the gravity well and the solid rocket entirely in it, at an altitude of 500 km. Performance was evaluated for a range of launch C3s from 0 to 70 and a range of target

C3 from 100 to 180. (A 14-year Pluto trajectory requires C3 about 160.) Results are shown in Figure 5.

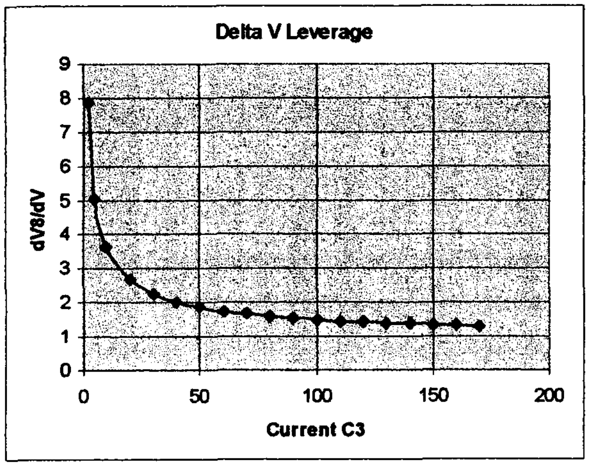


Figure 4: Gravity Well Differential Advantage

An STP propulsion system offers better performance to high C3 than the usual solid rocket. STP is compared here to a solid rocket stage, both as kick stages on a Delta IVM+ 5,4. For either system a launch C3 near 40 is preferred. This is a small STP and it operates only in deep space, so there is little concern about environmental degradation of the concentrator ... (a) the burn is continuous so the time of exposure is less; (b) the severe radiation environment of the van Allen belts is not applicable, nor is concern about atomic oxygen. Payloads are typically 500 – 1500 kg (reference IISTP). Neither system reaches the desired payload at typically desired C3s. For example, the payload for a direct launch to Pluto flyby is quoted as 450 kg and requires a C3 of about 160. A Titan Explorer is quoted at about 1400 kg with C3 for direct launch about 110. A larger launch vehicle such as a Delta IV Heavy would probably enable the desired performance.

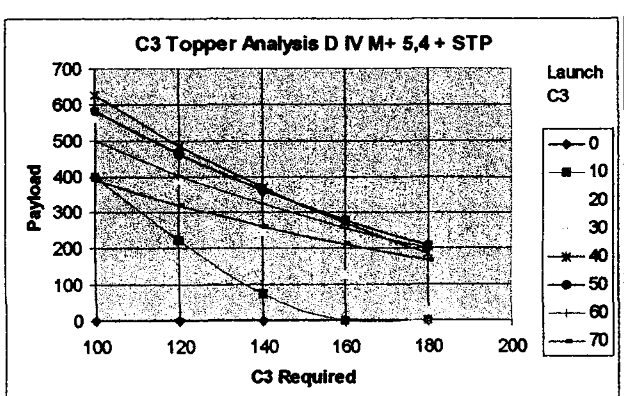
A further consideration is that the Next Generation Ion technology program is presently conducting technology advancement for an electric propulsion system that can do these missions with adequate payload margin. The mission profile for both would employ a single Venus gravity assist. This profile is available every year. Venus gravity assist might also improve the performance of the STP option; this was beyond the scope of the assessment study. It does not improve the performance of the solid rocket option.

The C3 topper may be a useful application for STP but must be evaluated in light of the expected performance capability of solar electric propulsion systems.

Launch Vehicle Compatibility

The issue of low-density hydrogen was mentioned above. This problem arises because the current stable of launch vehicles was not designed for upper stages that operate on only hydrogen, and liquid hydrogen is far less dense than other propellants. The problem is exacerbated because the STP upper stage option gives best performance if launched to LEO, while the design case for these launchers is launch to GTO. Thus in the case of STP, we want not only to reduce the average density of the payload fairing contents but also to increase the mass.

The situation is presented graphically in Figure 6. On the left is a typical planned mission application, as depicted for the Delta IVM+ 5,4. The numerical designation means a five-meter fairing and four strap-on solid propellant boosters. The payload capability to GTO is approximately 6000 kg, which divides roughly evenly as 3000 kg GEO payload and 3000 kg apogee insertion propulsion. The apogee propulsion system for such missions is normally integrated into the spacecraft, but is shown schematically as separate to indicate its relatively high density.



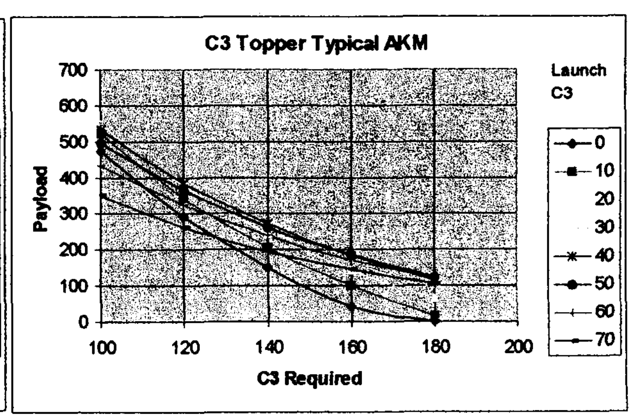


Figure 5: C3 Topper Analysis, STP Versus Solid Propellant Rocket

In terms of payload mass performance capability, the STP would allow stepping down to the Delta IVM+ 5,2 which has only two solid propellant boosters instead of four. Its payload to LEO is about 9000 kg, and this is the mass to be placed in the fairing. Of this, about 2/3 is the STP stage. The same size fairing is used. We replace a high-density 3000 kg apogee insertion stage with a low-density STP of twice the mass that performs both perigee and apogee maneuvers. Quite simply, it doesn't fit because it leaves little or no room for the mission payload.

Cost Factors

We presume that launch customers are not concerned with the technical features of launch systems and care only about price, timely service and risk of failure. A customer views selection of a launch vehicle and upper stage as a single integrated choice and will choose the system that best suits his/her needs. For this analysis we neglected the payload fairing volume issue and considered only payload mass delivery performance and cost to customers. We considered the geosynchronous orbit delivery mission as representative.

A few years ago, the gaps between launch vehicles were large. For smaller payloads, the Delta II family offered configurations with varying numbers of strap-on solids. For median payloads the Atlas II family offered a couple of configurations, as did Ariane IV. Heavy payloads had only the Titan IV-Centaur as a choice. A higher-performance upper stage might offer an opportunity to save large sums in launch cost. Today, assuming all or most of the EELV options in development reach the marketplace, many more options are available. Table 3 summarizes performance of the Delta and Atlas families. In addition, Sea Launch, Ariane IV and V, the Japanese H2A, and various Russian and Chinese vehicles are available. Performance figures for LEO in the table include a column with 10% margin taken out; this is to indicate possible needs for airborne flight equipment to support an STP stage.

A launch customer has a choice of purchasing a launch vehicle which, with an STP upper stage, can meet his/her delivery requirements, or purchasing a larger vehicle that can do so without an STP upper stage. As mentioned before, all the launch vehicles in Table 3 can perform a GEO delivery mission without an STP, by delivering the payload to a geosynchronous transfer orbit (GTO). The payload needs an apogee insertion propulsion system, usually either an apogee kick motor (solid propellant) or a storable propulsion system integrated with the payload. The launch vehicle configuration and cost do not change whether or not an STP is used. Therefore the trade is whether the added performance of an STP stage outweighs its cost when

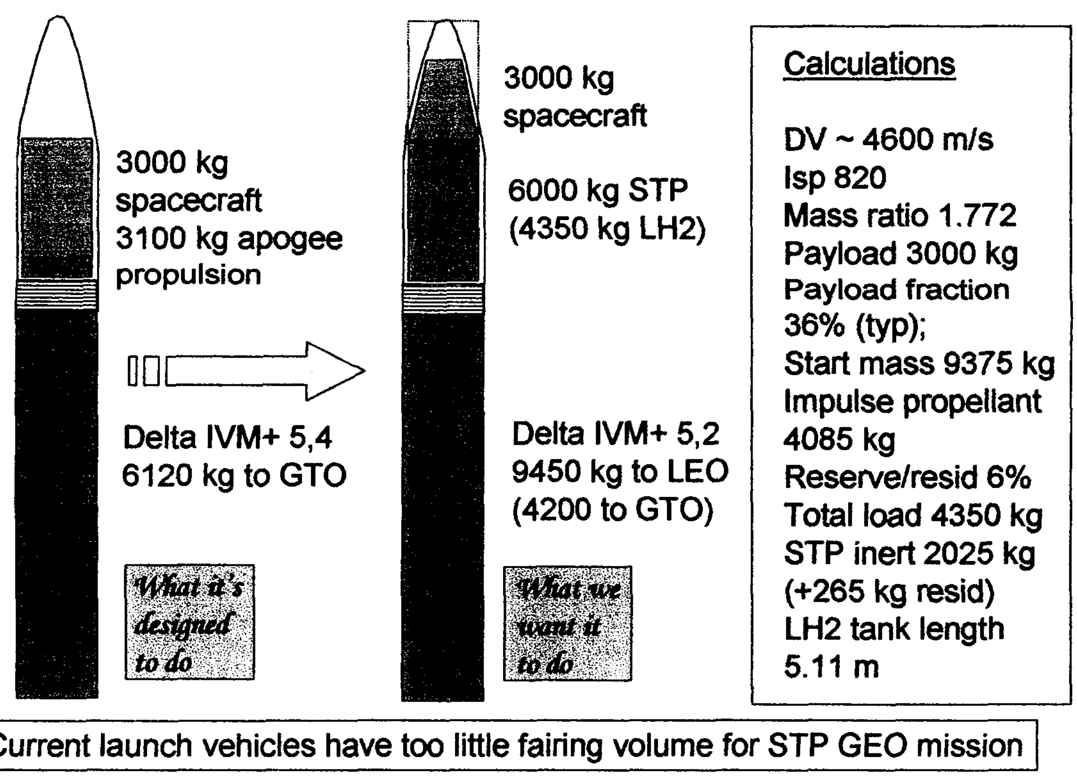


Figure 6: Graphical Illustration of Payload Fairing Volume Problem

considering the alternative of a larger launch vehicle that does not need the STP, and needs only an inexpensive apogee insertion system.

STP costs were estimated as comparable to the costs for a cryogenic upper stage with similar capability.

Table 3: Summary of Launch Vehicle Payload Performance

	LEO	Less 10% for adapters	GTO
Atlas IIA	7316	6584.4	3066
Delta IVM	8500	7650	3900
Atlas IIAS	8618	7756.2	3719
Atlas IIIA	8640	7776	4037
Delta IV M+ 5,2	10500	9450	4200
Atlas IIIB	10718	9646.2	4477
Delta IV M+ 4,2	12000	10800	5200
Atlas V 402	12500	11250	5000
Delta IV M+ 5,4	13700	12330	6120
Atlas V 552	20050	18045	8200
Delta IVH	24500	22050	10500

Figure 7 shows the customer-choice cost comparison developed for this assessment.

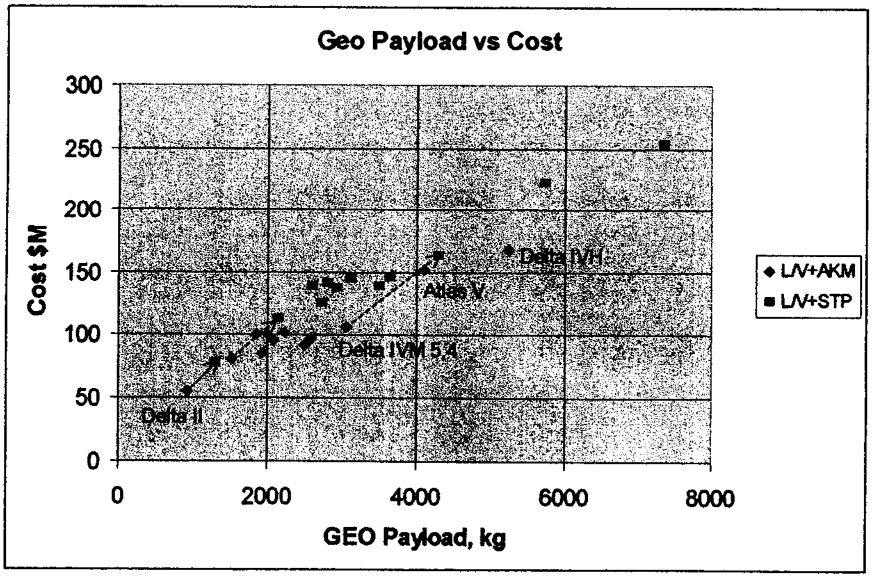
For each launch vehicle, the performance and cost are plotted with and without the STP upper stage. The launch vehicle without STP is plotted as a dark blue diamond, and with STP a magenta square. (Launch vehicle costs were obtained from Isakowitz *Space Launch Systems* Vol. III.) A connecting dotted arrow is shown for a few example cases. For example, at the lower left, the Delta II can deliver about 1000 kg to GEO at a cost of about $55 million. If one adds an STP upper stage, the payload capability increases to about 1400 kg and the cost increases by about $25 million. The alternative is to purchase an Atlas IIA, which can deliver about 1700 kg at a cost about $85 million.

In almost every case, the logical customer choice is clearly to choose the larger launch vehicle. That choice involves less risk and in most cases less cost. The Delta IVM 5,4 with STP shows a slight advantage over the Atlas V but probably not enough to outweigh the risk difference. If a customer were to have a large payload that exceeds the Delta IV Heavy payload mass capability, choosing an STP might be preferred over a two-part delivery with assembly.

It is also important to recognize that a customer who can afford the delivery delay of low-thrust propulsion (a few months), can elect to use payload onboard electric propulsion to complete the GEO delivery. This increases payload performance by about half the gap between conventional and STP-aided performance at very little cost except for the delay.

The conclusion of this part of the assessment is that, even aside from payload fairing volume issues, few customers will choose an STP upper stage instead of a larger launch vehicle.

A caveat on this conclusion is important: If a launch vehicle were designed expressly to use an STP upper stage, and configured to eliminate the cryogenic upper stage, the unit cost/performance tradeoff would



- <u>In general, less costly for customer to upgrade launch vehicle than buy STP</u>
- Delta II is an exception but shroud size is too small
- Delta IVM 5,4 may be an exception, but expect severe volume problems

Figure 7: Launch Customer Cost Trade Summary

probably favor this configuration over a conventional launch system. If this makes business sense (including the non-recurring cost of new development) one could expect one or more commercial launch companies to request that NASA advance STP technology to TRL 6 to reduce the business risk of such a development.

CONCLUSIONS AND RECOMMENDATIONS

a) Solar Thermal Propulsion (STP) offers no unique mission capabilities not available through alternate propulsion technologies. State of the art chemical propulsion can perform all the missions for which STP is a candidate, albeit at a performance disadvantage in many cases. STP could provide better payload mass performance than alternate propulsion technologies in many cases, but as noted next, STPs with this performance don't fit in the fairings.

b) The volume required for STP hydrogen propellant makes most STP missions impractical with current launch vehicles. These launch vehicles are designed to efficiently deliver payloads to a geosynchronous transfer orbit (GTO), using an integral cryogenic upper stage. The cryogenic upper stage is also required for launches to low Earth orbit (LEO). Therefore, if an STP is used as an upper stage, it and its payload must fit in a fairing volume nominally designed for a payload plus dense apogee insertion stage. The STP payload is larger; STP offers a performance improvement; otherwise would not be of interest for this mission. The STP itself is about twice the mass of the apogee insertion stage and has far less density. Thus a severe fairing volume problem is to be expected and in fact exists.

c) Current launch vehicles, as noted, are designed to be efficient for GEO and near-Earth space missions. If the launch vehicle options currently in development all enter the market, several upgrade increments will exist in the payload range of interest. It usually will be cheaper to buy a bigger launcher than to buy an STP upper stage.

d) We found a few applications that could benefit appreciably from STP. In particular, a "C3 topper" mission was found for which STP offers a performance advantage and the payload fairing volume is not a problem. STP was competitive, but not necessarily superior, for a mission of delivery of a "Gateway" payload to the Earth-Moon L1 libration point, and for application as a shuttle upper stage. The shuttle upper stage application did not permit the use of hydrogen, so an STP using ammonia propellant and a conventional bipropellant chemical stage were compared; performance was about equal.