SMALL SATELLITE MISSIONS TO DEEP SPACE ENABLED BY SOLAR THERMAL ROCKETS

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Summary and Abstract

Solar thermal rockets (STR) were examined to determine how well they could integrate with small satellite systems to propel very small payloads to the outer reaches of the Solar System. The spacecraft would use an STR to achieve hyperbolic velocity while in the vicinity of Earth. A set of calculations estimated the distance from the Sun and flight time of a flyby mission launched from small launch vehicles such as the Pegasus, Taurus and Delta 6920.

Rendezvous missions were modeled using solar photovoltaic or radio isotope thermoelectric generators with electric propulsion to make the spacecraft velocity match that of the target object.

The Pegasus would send 10 to 25 kg sensor payloads to fly by objects near the ecliptic out as far as Saturn. It could also send a 10 kg payload to rendezvous with most of the known Near Earth Objects (NEOs). The Taurus could send these payloads to fly by Pluto. It could also achieve rather fast, 1.5 year fly-by of Jupiter, and less than 3 years to Saturn. The Delta could rendezvous with Pluto on a 20 year mission. It could rendezvous in less than about 15 years by staging.

The STR provides a 16 square meter mirror that can be used as a form of telescope, giving it dual-use as a sensor element.

The general conclusion was that high performance STR enables small satellite systems to achieve high delta V, 15,000 m/s missions not practical with chemical systems. These include fast, 1 year direct missions to fly by Jupiter and most of the objects closer than Jupiter, which include most of the known NEOs. The STR also enables small satellite rendezvous with most of the NEOs and fly by with Pluto. With the help of RTG class electric power generators, small satellite systems can achieve rendezvous with Pluto, and hence everything closer.

Introduction

Objects in the Solar System are becoming more and more interesting as we learn more about their resources and the relation of planetary evolution to our own climate on Earth. Satellite probe missions to these objects have traditionally been very expensive, mostly due to the cost of launch systems. A typical mission to Jupiter or deep space (e.g. Gallileo) costs of order a Billion dollars, making it attractive to include many experiments on the spacecraft, and further increasing cost. The difficulty of reaching distant planets such as Pluto forced the use of large and expensive launch vehicles (e.g. Titan III or larger), and these still resulted in small spacecraft masses (e.g. Voyager).

Alternative much lower cost options have emerged as satellite systems became smaller and as small, low unit cost launch systems became available. The small satellite option would cost of order ten times less than traditional missions and would permit many launches to many solar system objects. But the payload is too small to permit more than a few experiments or sensors per launch. And if we can only use conventional chemical propulsion then the payloads to Pluto become very small on a small launch system, if they are possible at all.

Solar Thermal Rockets (STR) offer a new option for the small launch systems and make some missions possible that were not possible using conventional propulsion. The estimated specific impulse performance of an STR (600 to 960 sec) is about twice that of chemical systems. It would be more like that of a Nuclear Thermal Rocket (NTR). And, unlike a nuclear system which has a minimum mass of order several hundred kilograms, the STR can be scaled down in size and mass to fit the smallest, least expensive launch system. The STR performance permits about a factor of two reduction in the amount of propellant needed to achieve a typical mission. Of order

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50 percent more payload can be added to replace propellant not used.

The more difficult missions such as those to rendezvous with near-Earth Objects (NEOs) or to flyby Pluto, require the rocket to achieve such a high velocity (delta V) that they can not be practically achieved with chemical rockets. The STR enables these missions because it can achieve the required delta V with much reduced launch vehicle and propellant.

Can this increase in rocket propulsion performance enable small launch systems to deliver significant payloads to deep space? The analysis that follows addresses sending small, tens of kilogram sensor payloads to flyby or rendezvous with Pluto, the planets and near-Earth objects. The results suggest the STR might enable a new class of missions: low cost probes to anywhere in the Solar System.

These calculations are estimates to approximate the STR performance and are not precise mission or space craft designs.

Missions and Hardware

Analysis evaluated flyby and rendezvous missions to three destinations: 1. Pluto, 2. the planets, and 3. near-Earth Objects. Three representative low cost launch systems were used as platforms: Pegasus, Taurus and Delta 6920(older vehicle). We used values for the payload to LEO that have since significantly improved. We used 454, 925 and 3900 kg to Low Earth Orbit (LEO), respectively. The current Taurus payload is 1272 kg and the current Delta 7920 payload is 5045 kg. The sensor or experiment payloads (not including power, attitude controls systems etc.) were as meager as can be imagined, ranging from 10 to 50 kg.

The mission profile uses a low cost launch system to place an STR propulsion system, an appropriate bus, guidance, navigation and control system, a liquid hydrogen fuel tank system, and a deep space power supply, bus and payload, all into Low Earth Orbit (LEO). The STR propels this spacecraft (S/C) to an orbit that crosses the path of the target in deep space. Another, higher performing propulsion system, such as solar or RTG electric propulsion is used to match the S/C velocity with that of the target for rendezvous. Almost all the propulsion is performed in the vicinity of Earth.

Starting at LEO, the STR uses a series of perigee pulsing maneuvers to raise the apogee of the orbit to 200 earth radii, placing the S/C in a very high elliptic earth orbit (HEEO). This is shown in Figure 1. The HEEO orbit is almost an earth escape orbit. The STR has just enough thrust to permit the maneuver to occur in the time span of months. (Ref Shoji at Rocketdyne and Laug at Edwards AFB). This maneuver is the most efficient way to raise the orbit.

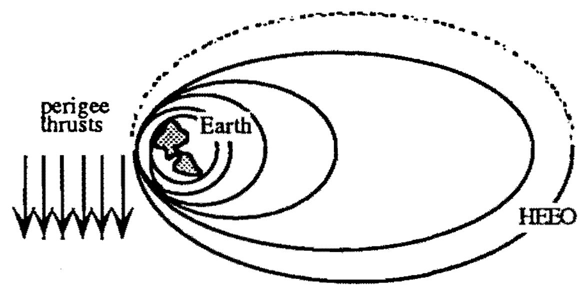

Figure 1. STR uses multiple perigee thrusts to raise orbit to near Earth escape.

Starting from HEEO, the STR accelerates the S/C to the desired deep space orbit with a velocity hyperbolic with respect to Earth., as shown in Figure 2. The vehicle achieves a velocity greater than escape during this final thrust and continues

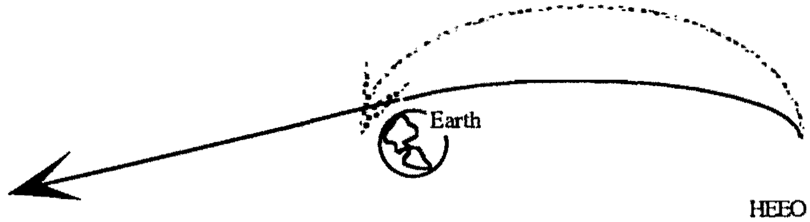

Figure 2. STR accelerates spacecraft to hyperbolic velocity.

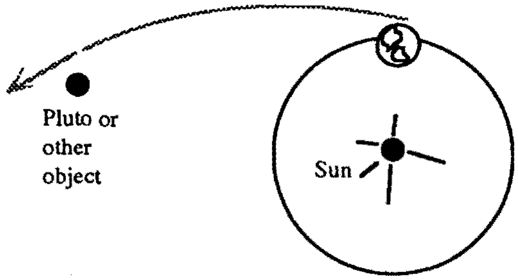

Figure 3. Spacecraft coasts to fly by Solar System objects.

a heliocentric thrust until tt achieves the desired velocity.

This analysis does not separate the tanks. This is not the optimum maneuver, but it is a conservative starting point. The entire S/C that started at LEO proceeds to deep space.

The vehicle coasts to the target. as shown in Figure 3. In practice, small midcourse maneuvers are performed. Also in practice, gravity assists can be used to add S/C velocity. This analysis does not use take advantage of these, and is therefore conservative.

When a rendezvous is required, a hydrogen

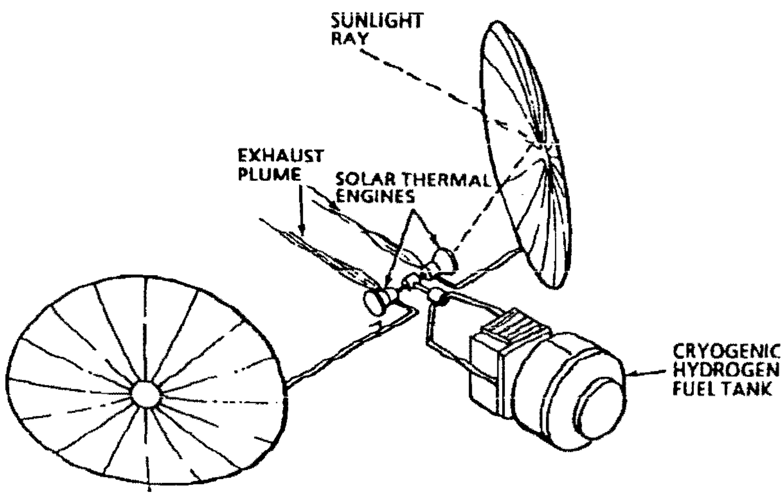

Figure 5. A concept for a Solar Thermal Rocket configuration uses mirrors to focus solar energy on solar thermal rocket engines, heating hydrogenand producing thrust.

Figure 4. Electric propulsion provides rendezvous at the distant object.

arc-jet (1400 sec Isp) or an inert gas ion engine (4000 sec Isp) decelerate the S/C so that it matches the velocity of the target. Electric power is supplied either by solar array or by some form of radioisotope thermal generator (RTG). An advanced form of RTG is the Multi-Hundred Watt Dynamic Isotope Power Supply (MHW DIPS). The latter uses less isotope per electric watt delivered than the RTG. The STR is used in one deep space example to focus the sun on a photovoltaic array. This only provides useful power for distances less than about 3 A.U.

The STR mirror is presumed to be used to augment the sensor system, for example, as a form of telescope.

The mass budget differentiates the baseline bus and S/C vehicle needed for earth escape from the deep space buss and special hardware, which may include special telemetry, antennae and computational capability.

Solar Thermal Rocket

The solar thermal rocket is maturing after about two decades of research. One key advance is the use of very low mass mirrors. The STR uses mirrors to focus the sun onto hydrogen and heats it to a temperature nearing 3060 Kelvin. The hot hydrogen expands in a rocket nozzle, producing thrust. Hydrogen is used because, with its low molecular mass, it provides the

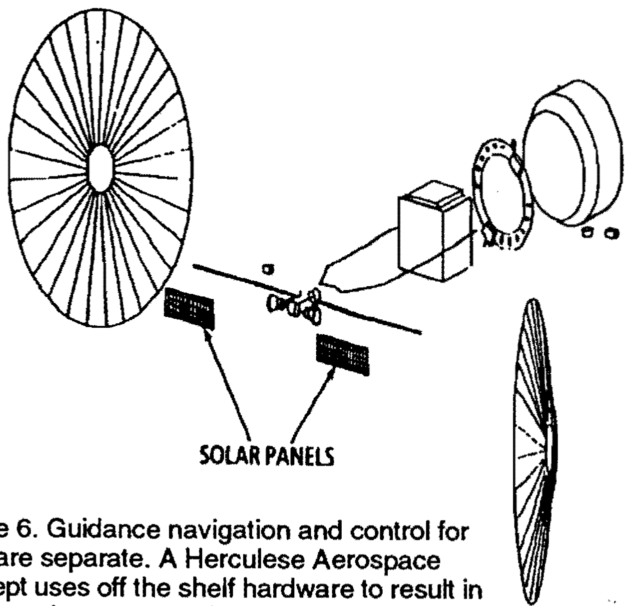

Figure 6. Guidance navigation and control for STR are separate. A Herculese Aerospace concept uses off the shelf hardware to result in 28 Kg package, compatible with small satellites.

highest specific impulse performance at a given temperature of all known propellants. The claimed ultimate performance achievable with a ceramic nozzle heater is 960 sec, which we use in our calcuations. The current capability with a metal nozzle heater today, at 2760 Kelvin, is about 860 sec. Compare this to 440 sec Isp of a liquid hydrogen/oxygen rocket and to about 900 seconds for a nuclear thermal rocket. The STR may have higher specific impulse than a NTR because it has fewer material constraints. For example, the very high temperature materials such as hafnium carbide used in the hydrogen heater and nozzle are not good nuclear reactor materials

and the uranium or plutonium fuels rapidly drop the melt point. Figure 5 sketches a typical STR configuration. Figure 6 summarizes a guidance, navigation and control design. The Guidance, Navigation and Control system for the STR is taken to be 28 kg, as per calculation by Hercules Aerospace, Utah.

Very low mass mirrors permit the STR to deliver about 1.5 KW thermal energy per kilogram of collector. The reference STR uses a 14 kg, 16 square meter mirror to deliver about 21 KW solar energy. Nearly all of this energy is delivered to the hydrogen propellant. This should be compared to electric propulsion, which deliver about 10 times less of the solar energy to the propellant.

The power delivered to the propellant translates directly to the thrust and hence shortens the time to perform the maneuver. This in turn permits the maneuver to have a much lower gravity loss than the very slow electric propulsion systems.

The liquid hydrogen tanks are assumed to hold 8 kg of liquid hydrogen per 1 kg of tank. This is a tankage fraction of about 12%, and is somewhat conservative. Estimates suggest that it may be possible to design tank fractions as low as 6% for this size STR. The fraction is taken at the high end because of the need to insulate the relatively small tank (holding between 100 and 2500 kg of liquid hydrogen) or to refrigerate its contents, both of which consume mass. The tank must retain most of the hydrogen for times as long as months, and may need to hold some hydrogen for the entire mission duration.

The orbital maneuver delta V is calculated for the entire thrusting duration using a simple, two dimensional integrating code. The calculation is necessary because the thrust of the STR is just low enough that simple, point thrust estimates are not sufficiently accurate. The orbit of all the planets are assumed to be in the plane of the ecliptic. The mission times and delta V's resulting from this approximation are close enough for the purpose of this exercise, which is to determine whether or not the missions are within the range of feasibility. The Pluto orbit is inclined at about 17 degrees, so this calculation only provides a good estimate for flyby maneuvers. The rendezvous will require slightly more delta V than this assumption, and a small excess rendezvous delta V was used to assure a good ballpark estimate. The delta V to access NEOs is most conveniently given by the total mission delta V achievable, as per Shoemaker.

Results & Discussion

The summary Table 1 shows the results of sample calculations. Table 2 shows the masses and power used. Pegasus can send a 25 kg payload to flyby Jupiter in an orbit approximately like a Hohman transfer orbit to Jupiter. Lowering the payload to 10 kg permits the Pegasus to send 10 kg to Jupiter in 1.5 years or to Saturn in 3.5 years. Attempts to include small ion or arc-jet thrusters to achieve a rendezvous failed because of the added mass of the thruster hardware. The most frugal payload (10 kg sensors and 10 kg deep space bus) is still too heavy to permit enough mission velocity to reach Pluto.

Table 2, Case 3 shows that the Pegasus can achieve enough delta V to rendezvous with about half of the NEOs in the Solar System. Both Davis and Shoemaker (see references) show that about 16500 m/s total mission delta V is sufficient to dock with about 70% of the NEOs. The Pegasus would achieve this with a 10 kg payload.

The rendezvous would have to occur at less than 3 AU because the arc-jet thruster used to rendezvous obtains its electric power from a solar photovoltaic cell. The STR mirror would serve double duty and focus the sun to provide about 230 watts at 3 AU. These perihelion rendezvous missions typically require more delta V than aphelion rendezvous, but are quite feasible.

To reconfigure the STR mirror to focus the sun on the solar cell instead of the rocket in deep space will probably be a nontrivial exercise given the tight mass budget of the Pegasus.

The Taurus provided enough propellant to for the vehicle to achieve hyperbolic, solar escape velocity. This gives rather rapid trips to Jupiter and Saturn (1.2 and 2.5 years, respectively). Achieving solar escape means that Taurus can deliver a payload to flyby any object in the Solar System.

sensor payload kg	destination	how	trip time years	launch vehicle	case ID
25	Jupiter	flyby/visit	2.6	Pegasus	1
10	Jupiter	flyby	1.5	Pegasus	2
10	Saturn	flyby	3.5	Pegasus	2
10	50% of NEOs	rendezvous	approx 2	Pegasus	3
25	Pluto	flyby	15.5	Taurus	4
25	Jupiter	flyby	1.2	Taurus	4
25	Saturn	flyby	2.5	Taurus	4
25	Uranus	flyby	6.5	Taurus	4
25	Neptune	flyby	12.3	Taurus	4
25	Pluto	rendezvous	19.4	Delta	5
50	Pluto	flyby	11	Delta	6
25	Jupiter	flyby	1	Delta	7
25	Satrun	flyby	<2	Delta	7
25	Uranus	flyby	4.2	Delta	7
25	Neptune	flyby	<7	Delta	7

Table 1. Summary of Results

The Delta has enough mass to deliver a 1 kilowatt RTG-class electric power supply to deep space. This permits rendezvous. However, even 1 kW is not sufficient to reduce the thrusting time to achieve the rendezvous. At this low power the ion thruster takes 9 years to provide the delta V. The existence of thrusters weighing only 3 kg that operate for 9 years at 1 KW may also be questionable.

Tossing the STR can resolve the thrust time dilemma. By eliminating the 468 kg of tanks, which are not needed after the HEEO thrust, changes the deep space vehicle mass enough to reduce the thrust

Case ID		1	2	3	4	5	6	7
Launch Vehicle								
name		Pegasus	Pegasus	Pegasus	Taurus	Delta 6920	Delta 6920	Delta 6920
IMHEEO	kg	321	321	321	655	2807	2807	2807
total ΔV	m/s	10175	12219	16529	14226	17761	13712	17761
deep space								
payload	kg	25	10	10	25	25	50	15
bus	kg	25	10	9	25	20	60	15
power	watts	35.6	35.6	233	35.6	1000	150	36.5
type		RTG	RTG	solar PV	RTG	MHW DIPS	MHW DIPS	RTG
mass	kg	13.5	13.5	1	13.5	175	38.5	13.5
thruster								
type		none	none	arc jet	none	ion	none	none
Isp	sec			1400		4000		
thrust time	years			1.3		5.1		
mass	kg			1		3		
fuel mass	kg			40		50		
fuel type				hydrogen		inert gas		
tank mass	kg			(main tank)		12.5		
STR								
power	KW-th	21	21	21	21	42	42	21
mirror mass	kg	14	14	14	14	28	28	14
GN&C & ST	kg	28	28	28	28	56	56	28
fuel	kg	167	197	170	451	1970	2106	2254
tank	kg	48.2	48.2	48.2	98.4	468	468	468
Isp	sec	960	960	960	960	960	960	960

Table 2. Mass and Power Budget Details.

time by an order of magnitude. This in turn would make the rendezvous practical. Jettisoning tanks is a non-trivial exercise, but is a routine strategy.

The Delta can provide throw weight to send significant, 25 kg sensor payloads to fly by Jupiter in 1 year, Saturn in less than 2 years and Uranus in less than 4.5 years.

The results are quite sensitive to specific impulse, but test cases using various specific impulses above 730 seconds strongly suggest that slight increases in propellant available at LEO completely compensate. This is a direct result of the fact that the ratio of initial mass at LEO to the final mass at hyperbolic velocity is of order 3. Lowering specific impulse raises the ratio to of order 4, ant this is well within current capability. That is, improvements in launched mass characteristic of normal launch vehicle evolution compensate for Isp > 730 s.

The upgraded payload of a Taurus will permit either faster direct flyby of Pluto (as low as 9 years) or use of less developed, metal rocket nozzle heater elements (Isp ~860 to 900 s).

The upgraded payload of a Delta 7920 will permit upgrading every option, including fast fly-by, lower Isp and rendezvous.

Conclusions

The mission delta V achievable with the configurations examined shows that the Pegasus can send sensors to fly by nearly every object closer than Saturn. The Pegasus can rendezvous with about 70% of the NEOs using only solar power systems.

The Taurus can deliver a payload to flyby any object in the Solar System. The Taurus appears to be the smallest vehicle one can use to deliver a flyby payload to Pluto. The flight time would exceed 15 years.

The Delta can deliver significant, 50 kg sensor payloads to fly by Pluto in about 11 years. This maneuver is so fast that it reaches Jupiter in 1 year, Saturn in less than 2 years, Uranus in about 4 years and Neptune in less than 7.

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