Hybrids of Solar Sail, Solar Electric, and Solar Thermal Propulsion for Solar-System Exploration

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Abstract— **Solar sails have long been known to be an attractive method of propulsion in the inner solar system if the areal density of the overall spacecraft (S/C) could be reduced to ~10 g/m^2. It has also long been recognized that the figure (precise shape) of useful solar sails needs to be reasonably good, so that the reflected light goes mostly in the desired direction.**

If one could make large reflective surfaces with reasonable figure at an areal density of ~10 g/m^2, then several other attractive options emerge. One is to use such sails as solar concentrators for solar-electric propulsion. Current flight solar arrays have a specific output of ~100W/kg at 1 Astronomical Unit (AU) from the sun, and near-term advances promise to significantly increase this figure. A S/C with an areal density of 10 g/m^2 could accelerate up to 29 km/s per year as a solar sail at 1 AU. Using the same sail as a concentrator at 30 AU, the same spacecraft could have up to ~45 W of electric power per kg of total S/C mass available for electric propulsion (EP). With an EP system that is 50% power-efficient, exhausting 10% of the initial S/C mass per year as propellant, the exhaust velocity is ~119 km/s and the acceleration is ~12 km/s per year. This hybrid thus opens attractive options for missions to the outer solar system, including sample-return missions.

If solar-thermal propulsion were perfected, it would offer an attractive intermediate between solar sailing in the inner solar system and solar electric propulsion for the outer solar system. In the example above, both the solar sail and solar electric systems don't have a specific impulse that is near-optimal for the mission. Solar thermal propulsion, with an exhaust velocity of the order of 10 km/s, is better matched to many solar system exploration missions. This paper derives the basic relationships between these three propulsion options and gives examples of missions that might be enabled by such hybrids.

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1. INTRODUCTION

Solar sails have been researched for many decades; generally focusing on sail systems that might achieve an effective areal density for the S/C of ten or tens of grams per square meter of sail area. It is reasonable to guess that this might be achievable, since good reflectors such as aluminum can reflect most sunlight using a foil thickness well under one wavelength of light. The areal density of an aluminum layer 1/4 wavelength (say) of the peak of the solar spectrum is 0.37 grams per square meter. So even with a backing membrane, rip-stop fibers, and support structure it is not unreasonable to believe that ten or a few tens of grams per square meters might be achievable. There are many sources that have elaborated on the feasibility of making solar sails in this range of areal density [1].

The reason that this range of areal densities is of interest is that solar sails of this sort are capable of accelerations that are of interest for solar system exploration on human time scales. For example, a solar sailing S/C with an overall areal density of 10 g/m^2 can produce a tangential acceleration of 29 km/s per year at 1 AU from the sun - this is almost equal to the Earth's orbital velocity and so within a fraction of a year it is possible to achieve solar escape velocity (a 41% increase from Earth's orbital velocity). Tangential (as opposed to radial) acceleration is the quantity of interest since acceleration along the direction of motion is much more influential in affecting the orbital energy and angular momentum of an object than acceleration at right angles.

It has long been known that a good strategy for the use of solar sails in the inner solar system is to "dive into the sun"; that is, to orient the sail so as to thrust opposite to the orbital direction (presumably starting from Earth) so as to reduce the orbital velocity around the sun and to drop toward the sun [2]. This strategy generally calls for the sailing S/C to come as close to the sun as thermal and radiation limits allow, and then to re-orient so as to thrust forward along the flight path. This accelerates the vehicle rapidly (if the overall areal density of the sailing S/C is low) so that, by the time it emerges from the inner solar system it's velocity approaches or exceeds solar escape velocity. This strategy is attractive because the increased thrust from being close to the sun offsets the shorter durations of thrust that are

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available as one transits close to the sun. This nearly-impulsive maneuver near the sun allows Hohmann-or-faster flight times to the outer solar system.

A major problem with solar sails for application to solar system exploration missions is that there is no good way to stop when arriving at a target of interest in the outer solar system. This arises from the fact that the thrust of a solar sail is always along the surface normal of the sail (assuming near-perfect reflectivity) since the angle of incidence equals the angle of reflection, so non-normal components of the momentum cancel and the normal component is doubled. As a result, the thrust vector is always in the half-sphere containing the sun, and becomes vanishingly small as one approaches the boundary of that half-sphere due to the small geometric projection of the sail area as seen by the sun. Of course, in the outer solar system, this effect is aggravated by the low intensity of sunlight.

Hence it is desirable to conceive of a S/C that could both take advantage of the consumable-free propulsion of solar sails near the sun but could also effectively maneuver in the outer solar system, presumably using a propulsion system that consumes propellant mass. Because solar sails incorporate large areas of highly-reflective membrane, it is reasonable to wonder if the sail might be used as a solar concentrator in the outer solar system. The figure of a solar sail needs to be reasonably well-controlled so as to produce the greatest possible thrust in the desired direction. This means that any sail system might also have the ability to concentrate sunlight to a focus.

Since the most effective use of a solar sail generally is to produce thrust aligned with the current trajectory, and those trajectories are often nearly circular around the sun, it would be common to angle the sail at $\pi/4$ (45°) with respect to the sun-line. If the figure of the sail varies by some angle $\pm\Delta\theta$ from this nominal value, then the reflected light will not emerge along the trajectory, and the thrust will not be optimal. The angle that the reflected light will emerge will vary from the optimal by $2\Delta\theta$, and the thrust will be fractionally reduced by about $2(\Delta\theta)^2$. To keep this loss acceptable (say 5%), $\Delta\theta$ must be kept below ~0.16 radians. This represents an estimate of the figure requirement on solar sails, which we will use later to judge the utility and difficulty of using these same sails as concentrators in the outer solar system.

2. Optimal Exhaust Velocities for Solar-System Exploration

Solar system exploration missions generally involve total changes in S/C velocity (ΔVs) between a few and tens of km/s. At the high end of this range, consider a sample-return mission to the outer solar system, requiring almost solar escape velocity (an extra 12 km/s above Earth's orbital velocity), plus presumably the ΔV required to rendezvous with an outer-solar-system object, send a probe down to the surface to collect samples and return them to the mother ship, and then the additional ΔV for the mother ship to return to Earth. If the vehicle has an "impulsive" (relatively high thrust) rocket, then one could take advantage of the Oberth effect [3], where changes in velocity deep in the gravity well of Earth or another body pay handsome dividends (often a factor of 2 or more in reduced ΔV requirements) in increasing the final $V_{infinity}$, due to the cross-term in the expansion of $(V+\Delta V)^2$ for the kinetic energy. By choosing a location where V is maximized, the magnitude of this cross-term is also maximized. When low-thrust systems (solar sails or electric propulsion) are used, it is difficult to take advantage of this effect since the duration of the thrusting event is so long compared to the close-approach interval upon swing-by of a massive object.

If one has a fixed amount of energy available to put into propulsion, such as the output for a fixed duration of a solar concentrator (or a nuclear fission reactor), and one wishes to maximize the fraction of this energy that is converted into the kinetic energy of the useful part of the rocket (payload and residual structure, etc.) then one can calculate the results for the rocket equation with different amounts of propellant to see what mass ratio ($m_{initial}/m_{final}$) yields the greatest energy efficiency. This is plotted in Figure 1. The optimal energy efficiency of 64.76% occurs at an initial/final mass ratio of 4.922 and a propellant mass fraction of 79.7%. The final velocity at this maximum-efficiency point is 1.593 times the exhaust velocity. The exhaust velocity for high-performance chemical propulsion (e.g. LH_2/LO_2) is ~4500 m/s, while the exhaust velocity for high-performance thermal propulsion (e.g. Solar Thermal or Nuclear Thermal) is ~8800 m/s (limited by material properties for the thrust chamber). Thus an optimal (in the sense of Figure 1) thermal propulsion system could produce about 14 km/s in total ΔV, assuming sufficient energy is available. For a given energy to mass ratio of E/m, where E is the total kinetic energy in the exhaust and m is the total initial wet mass of the S/C, the ΔV at optimal efficiency works out to $\Delta V=2.524(E/m)^{1/2}$.

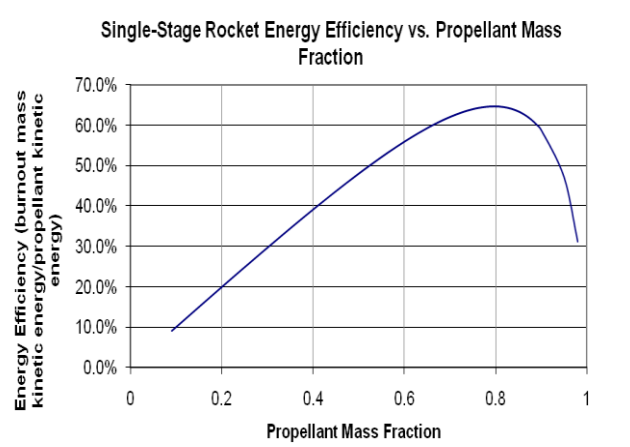

Figure 1: Energy efficiency of single-stage rocket in converting kinetic energy of exhaust into kinetic energy of burnout mass, from rocket equation.

By comparison, a solar sail delivers momentum change E/c, where E is the energy of sunlight that falls on the sail, and c is the speed of light. Thus the ΔV is E/mc, where m is the mass of the spacecraft. The good news for solar sails is that the achieved ΔV is proportional to E/m, rather than the square root of E/m as for normal rockets. The bad news is that the expression for ΔV has c in the denominator, where the speed-of-light c is a very large number. So solar sails must intercept a tremendous amount of energy in order to deliver appreciable ΔV. One can think about solar sails as delivering their exhaust at the speed-of-light, which is not well matched to the ΔVs required for solar-system exploration missions. However, the 14 km/s of total ΔV that can be delivered by an optimally-efficient solar thermal rocket is well matched to the ΔVs needed for solar-system exploration missions, particularly if that ΔV is restricted only to the part of the mission that occurs in the outer solar system.

Optimal solar-electric propulsion is limited not so much by the considerations of Figure 1, but rather by how long one is willing to wait while the solar array collects power. If one is willing to wait arbitrarily long, and the S/C stays in close enough to the sun for the solar array to be effective, then an arbitrarily-large amount of energy can be collected by the solar array. The longer one is willing to accumulate power, then the lower the propellant mass fraction is required to achieve a given total mission ΔV, at ever-increasing exhaust velocities. For mission durations of human interest, and using current-technology solar arrays and electric propulsion systems (e.g. 100 W/kg array output at 1 AU and ~55% efficiency in converting that electric power to kinetic energy in the exhaust [4]), useful exhaust velocities for solar system exploration missions lie in the range ten or tens of km/s (e.g. specific impulses of 1000-5000 seconds).

3. Comparisons of Solar Sails and Concentrators used for Solar Electric and Solar Thermal Propulsion

Figure 2 compares solar sails and equivalent solar concentrators used for solar electric or solar thermal propulsion in various regimes of the solar system. Along the x-axis is given the distance from the sun, ranging from 0.1 AU to 100 AU. For reference, Mercury orbits at about 0.39 AU, Venus at 0.72 AU, Mars at 1.52 AU, Jupiter at 5.2 AU, Saturn at 9.5 AU, Uranus at 19.2 AU, Neptune at 30 AU, and the Kuiper Belt (including Pluto) extends from 30 to about 55 AU [5]. (The Oort cloud is thought to extend out tens of thousands of AU.)

Along the y-axis of Figure 2 is a logarithmic scale ranging from 0.001 to 1000, in units that differ for the different curves. Starting in the upper left corner and working down, we plot the tangential (not radial) component of acceleration for solar sailing S/C having three differing overall areal densities - 10, 20, and 50 g/m^2, expressed in km/s per year. Next we plot the orbital velocity of an object in circular orbit around the sun at that distance. Note that solar sails accelerate at rates up to an order of magnitude greater than the circular orbital velocity per year when they are very near the sun. This is why the "dive into the sun" maneuver described earlier is so effective - the solar sail can accelerate the S/C to solar escape (41% above circular orbit velocity) in a very short period of time, so that it emerges from the inner solar system on a parabolic or hyperbolic trajectory that can be much faster than a minimum-energy elliptical

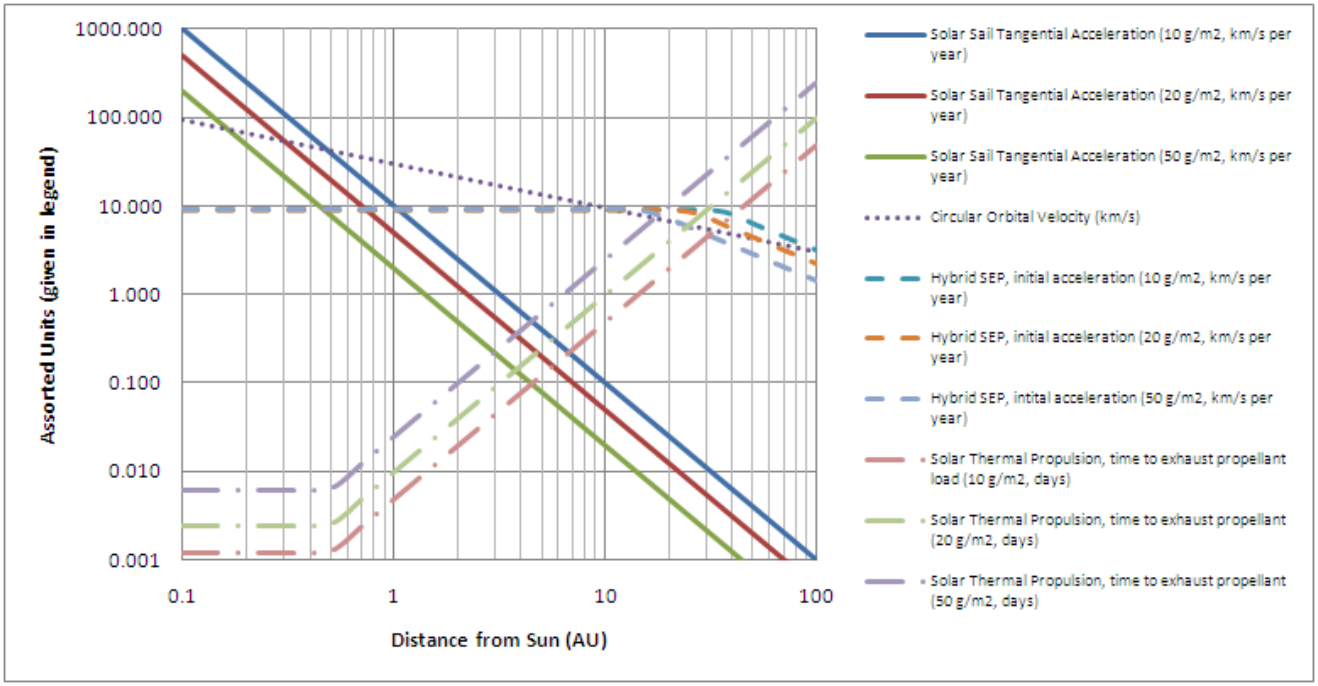

Figure 2: Comparisons of solar sail propulsion and use of solar sail as concentrator for solar electric and solar thermal

Hohmann transfer to the outer solar system. Next we plot the acceleration for hybrid Solar-Electric Propulsion (SEP) vehicles, again for effective areal densities of 10, 20, and 50 g/m^2, and again plotted in units of km/s per year. The acceleration is limited by the maximum power output of the photovoltaic arrays and power distribution system, which in this case we assume is limited to the current-technology values mentioned earlier that might be reasonably achievable for SEP S/C in the near future: a specific output power of 100 W/kg, expressed as the input power to the SEP thrusters as a fraction of the mass of the entire SEP subsystem (solar arrays, power conversion, and thrusters, but not including solar sail concentrator). We assume SEP subsystem is 30% of the initial wet S/C mass, the solar sail is 20%, the SEP propellant is 40%, that the thruster energy efficiency is 55% in converting bus power into the kinetic energy of the exhaust (current technology), and that the propellant is exhausted at a uniform 5-year rate. Note that, using the solar sail as a concentrator, the SEP subsystem can produce full acceleration beyond 10 AU, and only then begins to drop. Table 1 has all the assumptions underlying Figure 2.

Concentrating the sunlight on the solar array has an additional advantage. Current-technology solar arrays exhibit significant Low-Intensity, Low-Temperature (LILT) degradation in their observed conversion efficiency of incident sunlight input to electrical power output. With concentration of the sunlight to levels equal to or greater than the nominal power density at 1 AU, this LILT degradation is avoided.

4. Solar Thermal Propulsion

Solar Thermal Propulsion (STP) has been studied extensively [6] but not used in actual missions. With STP, concentrated solar power is focused on a thrust chamber where reaction mass is heated and ejected out a nozzle. Hydrogen gives the highest exhaust velocity at a given chamber temperature because of its low molecular weight. Specific impulses of 900 seconds or more (exhaust velocities in excess of 8800 m/s) have been achieved in test systems [7].

Continuing with Figure 2, the next plots are for STP systems, again for overall S/C areal densities of 10, 20, and 50 g/m^2. In this case we plot the time required to exhaust all the propellant, assuming the optimally energy-efficient rocket described in section 2: a propellant mass fraction of 79.7% with an exhaust velocity of 8800 m/s, giving a total ΔV of 14 km/s. We assume a power limitation on the thrust chamber closer than 0.5 AU to the sun.

Note that the STP system is able to exhaust all its propellant, and therefore to achieve the entire 14 km/s ΔV, in a relatively short period of time out to about 10 AU. This means that Oberth-type maneuvers for capture into orbit around Jupiter or possibly Saturn is possible, so long as the trajectory doesn't require that the maneuver be done in shadow. Capture at Uranus or Neptune could not be performed with an Oberth-type maneuver, since the thrust times would be too long, but the orbital velocities of these planets is only a few km/s, so it is less important to be able to perform an Oberth maneuver if one has a total of 14 km/s at one's disposal.

SEP propellant Mass Fraction (PMF)	40%	
SEP system mass fraction of total wet S/C	30%	
STP propellant Mass Fraction	79.68%	
STP system mass fraction of total wet S/C	5%	
Reflector mass fraction of total S/C, SEP	20%	
Reflector mass fraction of total S/C, STP	7%	
EP thruster efficiency	55%	
solar panel efficiency	30%	
solar thermal efficiency	60%	
solar sail reflectivity	90%	
Isp of solar thermal propulsion	900	seconds
time to exhaust propellant (SEP)	5	years
specific power (p)	100	W/kg
minimum distance for solar thermal	0.5	AU

Table 1: Assumptions used for plots in Figure 2.

Table 2 gives planetary data which allows a more formal analysis of the value of an Oberth maneuver for the different planets [5]. Listed for each of the planets is its mean orbital velocity around the sun, its escape velocity at the minimum safe approach distance (2% larger than radius for airless bodies; 5% larger for bodies with an atmosphere, especially because the low areal density of the sail/concentrator makes close approach to an atmosphere very dangerous), and a characteristic time during which an Oberth maneuver must be performed - the ratio of the approach radius to the escape velocity. In practice, useful Oberth maneuvers might take as much as twice as long as this characteristic time and still have substantial value. Also listed is the "gain" one achieves by a change of 1 km/s at minimum approach over escape velocity, in terms of the change in $V_{infinity}$. Note that this gain ranges from more than 3 to almost 11 for these planetary bodies, which is a huge, often enabling, benefit for mass- and cost-constrained planetary missions.

	Mean orbital velocity around sun (km/s)	Escape Velocity at minimum safe approach (km/s)	Minimum safe approach radius divided by escape velocity (seconds)	Increment in $V_{infinity}$ per unit velocity change at close approach
Mercury	47.9	4.21	591.4	3.07
Venus	35.0	10.11	628.5	4.61
Earth	29.8	10.91	614.1	4.78
Mars	24.1	4.90	727.9	3.29
Jupiter	13.1	58.14	1289.4	10.83
Saturn	9.6	34.71	1815.0	8.39
Uranus	6.8	20.80	1290.6	6.53
Neptune	5.4	22.99	1131.1	6.85

Table 2: Planetary data allowing assessment of relative value of Oberth Maneuvers

The concentration ratios needed for useful SEP in the outer solar system are roughly those needed to illuminate the solar arrays equivalent to the power density received at 1 AU. At 30 AU this means that the diameter of the solar image needs to be 30-times smaller than the concentrator. Recall from the earlier discussion that solar sails need to achieve a surface normal within ~0.16 radians of the desired shape to maintain good performance. To get a solar image 30-times smaller than the concentrator requires that the surface normal of the reflecting surface be somewhat better than 0.03 radians of the optimal shape. (Or an oversized concentrator could be used to reduce the figure constraints.) This ~5X improvement over the requirement for a solar sail might be achieved by stretching the reflective membrane over a hoop in compression, with the photon pressure forming the membrane into the concave shape needed for a concentrator. Alternatively, very fine tethers (designed to survive micrometeorites [8]) emanating from the focus assembly could attach over the concentrator area to maintain any desired shape.

The concentration ratios needed for useful STP are those required to achieve the needed temperatures. Our presumed 8800 m/s exhaust velocity corresponds to a thrust chamber temperature of ~2800K [6]. If the chamber is modeled as a black body, then concentrated sunlight at about 3.5MW/m^2 is required to maintain this temperature, with no heat going into the propellant. If the thrust chamber has an outer coating made of selective absorber (e.g. thin deposited layers of silicon over gold, with the silicon absorbing in the visible but transparent in the thermal-infrared) it may be possible to deliver a useful fraction of the concentrated sunlight into the propellant at such power densities. This power density requires a concentration of ~2600 at 1 AU, or a focus diameter 51-times smaller than the collector. This is possible since the sun only subtends about 0.086 radians, and so image-quality optics can produce an image of the solar disk that is 116-times smaller than the focal length. As the S/C gets farther from the sun than 1 AU, the angle subtended by the solar disk gets smaller, so it is possible to achieve higher and higher concentration ratios in the outer solar system. However, this would require a very high accuracy figure for the concentrator, which may be difficult to achieve. For missions to Jupiter or perhaps Saturn, where the relatively high thrust of STP would allow Oberth maneuvers to be used with great effect, it may be worth the mass and cost to achieve these precise figures.

One possible advantage of STP over SEP is the relatively high cost of space-rated solar arrays. Such arrays currently cost ~$1M/kW (electrical power output at 1 AU), and this high cost is a major impediment to the expanded use of SEP S/C for solar system exploration [9]. It is possible that STP could be implemented for a much lower cost per useful Watt converted (although it is also possible that the cost of space-rated solar arrays will drop significantly).

On 21 May 2010 the Japanese Space Agency JAXA launched IKAROS (Interplanetary Kite-craft Accelerated by Radiation Of the Sun) as a secondary payload accompanying the Venus Climate orbiter AKATSUKI. IKAROS has become the first interplanetary solar sailing S/C, and it also incorporates low-mass photovoltaic cells into part of the sail assembly to provide power. Their idea is to use such solar cells on subsequent vehicles to power an electric propulsion system, although the IKAROS S/C does not also demonstrate SEP (which was, however, previously demonstrated on the Hayabusa mission by JAXA). IKAROS manager Junichiro Kawaguchi describes this as a hybrid solar-sail/solar-electric propulsion system [10], although it does not use the sail as a concentrator and so is not useful at great distances from the sun as envisioned here. Also, the photovoltaic cells embedded in the sail fabric are highly absorptive, and not reflective, and so reduce the overall solar sail performance and aggravate the thermal design.

5. MISSION IMPLICATIONS

One of the implications of this proposed hybrid propulsion technology is that it could enable development of "mass-produced" sample return missions from all of the large airless bodies of the solar system

The solar system has a considerable number of large, airless bodies - Mercury, the Earth's moon, and all the large moons of the outer planets, except Titan. All these bodies share some important characteristics - they have escape velocities typically in the range of a few km/s, their rotation and orbital periods are in a resonance (usually 1:1), they have large areas that are poorly illuminated and at cryogenic temperatures for long periods of time, and they have convenient Lagrange stationary points where a "mother ship" could park to oversee the activities of an exploration vehicle on the surface.

These common characteristics may make it possible to build a single, standardized type of spacecraft which could be built in quantities of a dozen or so to return samples from all these bodies more or less simultaneously (originally suggested by this author in [11]). Obviously, the Earth's moon is a special case where a first prototype could be conveniently tested. Once a prototype or engineering-model of such a system had performed there as planned, manufacture of the "short production run" for all the other large airless bodies could commence.

The concept is to have a "mother-ship" and a "surface-explorer." The assumed properties of the mother ship are as follows:

1. It would use solar-sailing for propulsion in the inner solar system.
2. It would use SEP or STP in the outer solar system, using the solar sail as a concentrator onto a solar array or solar-thermal target.
3. It would perform all-propulsive capture and rendezvous with the target body, using the high-specific-impulse solar-

electric propulsion (SEP) system, or an Oberth maneuver on close approach to the parent body.
4. It would park at a Lagrange point throughout most of the surface exploration phase so that nearly-continuous line-of-sight communication can be maintained between both the surface-explorer and the Earth.
5. During the surface exploration phase, it would use the abundant power available by using the solar sail to concentrate sunlight onto a solar array to enable for high-bandwidth communications relay between the surface-explorer and Earth.
6. It would have sufficient delta-V to depart Earth, reach the target body in reasonable time, deploy the surface-explorer, oversee the surface exploration phase, rendezvous with a sample canister launched by the surface-explorer, and return that sample canister to Earth.

The assumed properties of the "surface-explorer" are as follows:

1. It would be able to land safely on the surface of the target body following release by the mother-ship.
2. It would be able to survive and operate in a cryogenic, low-illumination, low-gravity, hard-vacuum environment for a duration sufficient to collect samples of scientific interest.
3. It would carry active illumination and a standardized set of sensors and instruments sufficient to ensure that scientifically-interesting samples can be obtained.
4. It would be able to package these samples into a canister, and to launch the canister on a trajectory that could be reached by the mother-ship.

The Surface-Explorer Concept

The surface-explorer concept is a small planetary lander/rover that would carry a sample-return ascent vehicle. One attractive possibility is to do a "stop and drop" landing with a relatively large solid rocket performing the "stop" maneuver, while a storable liquid propulsion system on the rover would manage all the fine maneuvers, attitude control, and terminal descent from the time of separation from the mother-ship to the time of landing. The rover would be powered either by primary batteries (for a relatively short mission duration) or a radio-isotope power system. It would collect samples with a robot arm and put them into the sample canister on the top of a single-stage solid rocket. The rocket would be launched almost horizontally from the rover on the airless trajectory, arriving nominally at rendezvous with the mother-ship. The very high ΔV and power available at the mother-ship would allow it to employ a search radar and to make whatever maneuvers would be necessary to rendezvous, despite the fact that the sample return rocket may be completely unguided (e.g. spin stabilized only). The rover mobility system would be commanded, based on star-sensing by the rover, to accurately point the rocket in the correct direction and to launch the spinning rocket at the correct time. From that point, the rocket could be unguided. Prior efforts have tested the concept of placing a small "apogee kick" rocket motor "backwards" (e.g. with the nozzle opposite to the direction of the main stage) in the spinning assembly to lift the periapsis and hence to accomplish orbit insertion, by firing the apogee-kick stage on a simple timer after half an orbital period. The spinning assembly maintains the attitude of the assembly in space, so after half an orbit the "backwards" nozzle becomes pointed "forwards" to correctly accelerate the sample canister along the trajectory, all without any guidance system [12].

Because the escape velocity of all the large airless target bodies under consideration is a few km/s, a single-stage solid rocket would be suitable for reaching the mother-ship. This is because the exhaust velocity of typical solid rockets is almost 3 km/s, about the same as the escape velocity. Thus the mass ratio needed, by the rocket equation, is only about "e", the base of natural logarithms. This is a reasonable ratio for the mass of propellant to the final spent mass of the sample return rocket.

Because all the target bodies have no atmosphere, multi-layer thermal insulation (MLI) would be effective to protect the surface systems from the cryogenic environment, and to protect that environment from sublimating in the presence of the warm interior of the surface-explorer. MLI leaks heat at the rate of about 1 W/m^2 [13], where one side operates at room temperature and the other side is cryogenic. It is clear that either the waste-heat of a radio-isotope power system, or the internal dissipation of a battery-powered mission would be such that the interior could be kept warm if the leak out through the surface is only 1 W/m^2.

Sample return is, or should be, the "holy grail" of planetary science, because the tools available for analysis of samples here on Earth continue to increase their tremendous advantage over flight-weight instruments, even as those flight instruments continue to become more and more capable. Also, the number and diversity of scientists and students that could participate in a program of sample return would be vastly greater than the number that could participate in creating the limited number of flight instruments that could be afforded.

It is reasonable to believe that a standardized suite of instruments could be agreed-upon that would be sufficient to allow selection of samples from the assorted target bodies considered for this surface system. That would presumably not be true if this were not a sample return mission, since there is sufficient heterogeneity between those targets that purely in-situ analysis would drive custom instrument suites tailored for each of the targets. Since all the target bodies are presumed to have at least some cryogenic ices as part of the surface regolith (including the shadowed polar craters of Mercury and the Earth's moon), the instruments and the sample return capsule would need to be configured appropriately. (The mother-ship may need to perform a non-Keplerian "hover" maneuver to maintain line-of-sight of the surface-explorer in the case of the lunar and Mercury polar craters - fortunately it would have sufficient

performance using the solar sail to maintain this fixed offset above the Lagrange point.)

Industry has long recognized that there is a "learning curve" that results in lower costs when multiple copies of an item are created together as part of a single build-and-test flow. In the automotive industry, it is common for each doubling of production volume to reduce the unit cost by 25% [14]. There are those that don't believe such advantages apply to NASA spacecraft. Yet there seem to be no fundamental reasons to believe that proper attention to manufacturability during the design, development, test, and evaluation process cannot allow a government-industry partnership to produce a dozen or so of these spacecraft for only a few times the price of a single unit.

6. SUMMARY AND CONCLUSIONS

This paper describes the underlying physics and engineering considerations related to the possible use of low-mass solar sails as concentrators for use with Solar-Electric Propulsion or Solar Thermal Propulsion systems in the outer solar system. Such hybrids may enable attractive missions, such as sample return missions, that would otherwise be impossible with solar sails alone. A particular example of a spacecraft system that could be replicated to perform sample return from all the large, airless bodies of the solar system is examined in detail.

This paper documents and expands on a proposal by the author for hybrid solar-sail and solar-electric propulsion for planetary missions to the JPL program office (Mike Devirian and Ross Jones) made on 22 Feb 2000 [15]

7. ACKNOWLEDGEMENTS

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BIOGRAPHY

Brian H. Wilcox *is the Manager of Space Robotics Technology and a principal member of the technical staff at JPL. He was the supervisor of the JPL Robotic Vehicles Group for over 20 years, during which the group was responsible for planetary rover development leading up to the Sojourner and Mars Exploration Rovers. The group was responsible for development of the electronics, control, flight software, ground software, and mission operations of the Sojourner rover that explored part of Mars in 1997. Brian was personally responsible for the imaging and hazard detection sensors and the hazard avoidance algorithms on Sojourner. In 1999, he was selected as a Principal Investigator in the NASA Gossamer program to create a prototype solar sail, which was successfully tested in the NASA KC-135 microgravity simulation aircraft (aka the "vomit comet"). He is now the Principal Investigator for ATHLETE – the All-Terrain, Hex-Limbed, Extra-Terrestrial Explorer (see http://www robotics.jpl.nasa.gov/tasks/ index.cfm). He has a B.S. in Physics and a B.A. in Mathematics from the University of California at Santa Barbara, and an M.S. in Electrical Engineering from the University of Southern California*

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