JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 43, No. 4, July–August 2006

Sample Return from Mercury and Other Terrestrial Planets Using Solar Sail Propulsion

Gareth W. Hughes* and Malcolm Macdonald*
University of Glasgow, Glasgow, Scotland G12 8QQ, United Kingdom
Colin R. McInnes†
University of Strathclyde, Glasgow, Scotland G1 1XJ, United Kingdom
and
Alessandro Atzei‡ and Peter Falkner§
ESA, 2200 AG Noordwijk, The Netherlands

A conventional Mercury sample return mission requires significant launch mass due to the large ΔV required for the outbound and return trips and the large mass of a planetary lander and ascent vehicle. It is shown that solar sail spacecraft can be used to reduce lander mass allocation by delivering the lander to a low, thermally safe orbit close to the planetary terminator. In addition, the ascending node of the solar sail spacecraft parking orbit plane can be artificially forced to avoid out-of-plane maneuvers during ascent from the planetary surface. Propellant mass is not an issue for spacecraft with solar sails, and so a sample can be returned relatively easily without resorting to lengthy, multiple gravity assists. A 275-m^2 solar sail with a sail assembly loading of 5.9 g/m^2 is used to deliver a lander, cruise stage, and science payload to a forced sun-synchronous orbit at Mercury in 2.85 years. The lander acquires samples and conducts limited surface exploration. An ascent vehicle delivers a small cold-gas rendezvous vehicle containing the samples for transfer to the solar sail spacecraft. The solar sail spacecraft then spirals back to Earth in 1 year. The total mission launch mass is 2353 kg, launched using a Japanese H2 class launch vehicle, $C_3 = 0$. Extensive launch date scans have revealed an optimal launch date in April 2014 with sample return to Earth 4.4 years later. Solar sailing reduces launch mass by 60% and trip time by 40%, relative to conventional mission concepts. In comparison, mission analysis has demonstrated that solar-sail-powered Mars and Venus sample returns appear to have only modest benefits in terms of reduced launch mass, at the expense of longer mission durations, than do conventional propulsion systems.

Introduction

MISSION and trajectory analysis has been performed for sample return missions from the three terrestrial planets using solar sail propulsion. Because of the inverse-square increase in solar radiation pressure as the solar sail approaches the sun, the most attractive target for a solar sail spacecraft to acquire a sample from is Mercury, deep within the solar gravity well. Mercury sample return missions have been studied at the preassessment stage by the ESA in internal studies. The return of a sample from Mercury by conventional propulsion has also been studied at the Jet Propulsion Laboratory (JPL), California Institute of Technology, where it was noted that, the solar sail option could not be analyzed due to limitations in designing solar sail spacecraft operations in Mercury orbit.[1] Other authors have generated roundtrip trajectories to Mercury with solar sails, but this paper also details the science objectives, systems analysis, sail sizing, and trajectory analysis for a solar-sail-powered Mercury sample return mission. Launch date scans for the outward and return journeys have been generated, along with detailed simulations of the solar sail spacecraft orbital maneuvers at Mercury. To complement this detailed analysis, summaries of Mars and Venus sample return mission concepts using solar sails are presented. All three solar sail missions will be compared with conventional mission concepts. It will be shown that, although solar sails offer modest launch mass benefits for Mars and Venus, it is for Mercury sample return that solar sailing truly excels.

Mercury Sample Return

After the first orbiter and lander missions, a Mercury sample return mission will be required. It is important to ascertain the surface age of Mercury to understand its geologic history, accomplished by accurate rock dating of Mercury surface samples, only possible on Earth. For a sample return mission, the entire descent to Mercury must be via chemical propulsion due to the tenuous atmosphere. A high-latitude landing site is selected due to thermal constraints, and prior imaging of this site from the orbiter at high resolution is necessary. Even at high latitudes, landing in direct sunlight, or, indeed, in permanent shadow, would be undesirable. A landing site within a suitable crater, in partial shade but with some light reflected from the crater walls, is preferable, with a sample drilled from a rock outcrop within the crater.[1] However, recent craters may be contaminated with material from their impactor and should be avoided. In this paper, guided descent is employed for all but the last few meters of the descent to avoid thruster plume contamination of the surface regolith to be sampled. The stroke of the landing legs is used to absorb the remaining kinetic energy of surface impact.

Therefore, the baseline science objectives for a Mercury sample return mission are to acquire a surface sample though a precision landing at a carefully selected high-latitude landing site in partial shadow, within a suitably aged crater, with high-resolution imaging for documentation during terminal descent. Sample preselection and preanalysis will be conducted in situ during landing site characterization using a robotic arm and small mobility device (20-m range). The primary science goal is to acquire 350 g of surface regolith.

Received 1 February 2005; revision received 30 August 2005; accepted for publication 31 August 2005.

*Research Assistant, Department of Aerospace Engineering, James Watt (South) Building.

†Professor, Department of Mechanical Engineering, James Weir Building, 75 Montrose Street.

‡Engineer, Science Payloads and Advanced Concepts Office, European space Research and Technology Centre, Postbus 299. Member AIAA.

§Engineer, Science Payloads and Advanced Concepts Office, European space Research and Technology Centre, Postbus 299.

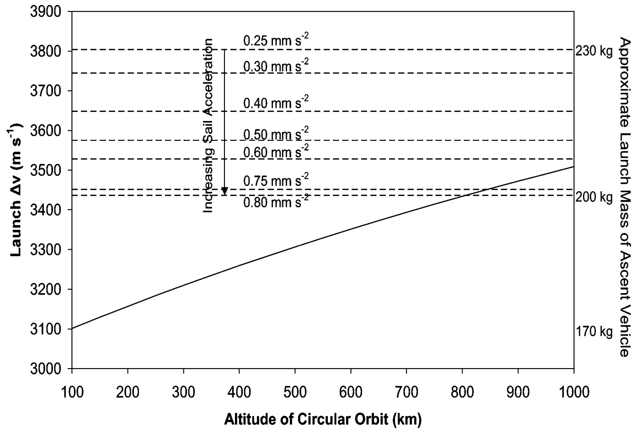

Fig. 1 MAV rendezvous orbit tradeoff: ——, ascent to circular orbit and - - - -, ascent to elliptical sun-synchronous orbit.

Mercury is not thought to be of direct interest to exobiology in the solar system, and so planetary protection measures will be simpler than for Mars missions and more similar to lunar missions.

The extremely high Δv required for Mercury sample return can be met relatively easily by solar sails because propellant mass is not an issue, thus significantly reducing launch mass. Lengthy multiple gravity assists are not required, and, in principle, the launch window is always open. Thermally safe orbit precession at Mercury is possible using continuous thrust. In this paper, the solar sail performance is defined by the characteristic acceleration, the solar radiation pressure induced acceleration at 1 astronomical unit (AU) with the sail normal oriented along the sun line.[2]

Mercury Sample Return Payload Model

A full and detailed solar sail payload has been defined and customized,[3] based loosely on an internal ESA assessment study, with some aspects drawn from a JPL study.[1] A tradeoff of the optimum solar sail spacecraft parking orbit at Mercury was conducted to minimize the Mercury ascent vehicle (MAV) Δv requirements. The use of an artificial sun-synchronous polar orbit at Mercury close to the planetary terminator[4] can be effected to reduce the thermal loads on the orbiter through a constant precession of the line of nodes, enabling a longer orbiter stay time and a much lower parking orbit. The characteristic acceleration of the sail in the parking orbit is defined by the parameters of the sun-synchronous orbit, and so as the acceleration is increased, the sun-synchronous orbit can be increasingly circularized. Figure 1 shows the effect of rendezvous orbit altitude on MAV launch mass. Note that ascent direct to the sun-synchronous orbit requires much more ΔV than ascent to a circular orbit. A circular 100-km orbit was selected to minimize MAV Δv requirements, with the sail used to deliver the lander onto the 100-km orbit, after an initial 44-day science and landing site selection phase on a 100 × 7500 km forced sun-synchronous orbit, 10 deg ahead of the solar terminator. During sample acquisition, until after coplanar MAV launch, the sail rotates the circular 100-km orbit plane to rendezvous with the MAV orbit before spiraling to escape.

The solar sail payload stack comprises a small cold-gas sail rendezvous vehicle (SRV), to conduct proximity maneuvers when transferring the sample from the MAV to the ballistic Earth return vehicle (ERV) attached to the sail cruise stage (SCS). The bipropellant MAV and cold-gas SRV are mounted on the bipropellant Mercury descent vehicle (MDV). The MDV has a large science platform and 0.4-m^2 gallium arsenide (GaAs) solar arrays. Figure 2 shows the lander deployed with its landing legs extended. Tables 1–4 show the mass breakdown of the SRV, MAV, MDV, and SCS, respectively. An analysis of the spacecraft subsystems shows a total solar sail payload mass of 1905 kg to support acquisition of 350 g of surface samples.

The SRV has a 2-kg sample container that holds the surface samples, with 50 m/s of propellant allocated for the rendezvous maneuver. The MAV uses a single stage DaimlerChrysler Aerospace (now part of EADS) S3K class bipropellant engine (specific impulse of 352 s), burning mono-methyl hydrazine (MMH) propellant. However, volume reductions and an increase in thrust to 4 kN would be necessary, and so conservative margins are added. The MDV uses five bipropellant MMH engines, delivering 6 kN each with a specific impulse of 320 s. The SCS allows for on-orbit power generation via 6.25-m^2 GaAs solar arrays. The SCS telecommunications system comprises low- and medium-gain X-band systems, a high-gain X/Ka band system, and a uhf link with the lander. The telecommunications systems have been sized during systems analysis to ensure adequate data return for the mission.

Table 1 SRV mass breakdown

SRV component	Mass, kg	Contingency, %	Total mass, kg
Sample container	2.0	——	2.0
Attitude control	3.1	10	3.4
Command and data	0.5	10	0.6
Power	2.0	10	2.2
Mechanisms	0.1	10	0.1
Telecommunications	1.1	10	1.2
Thermal	1.0	10	1.1
Structure	2.0	10	2.2
Thrusters	0.2	15	0.2
Valves, pipes	0.1	15	0.1
Propellant tank	0.1	15	0.1
System contingency	——	1	0.1
Total SRV dry mass			13.4
Propellant for rendezvous	1.0	15	1.1
Total SRV wet mass			14.5

Table 2 MAV mass breakdown

MAV component	Mass, kg	Contingency, %	Total mass, kg
MAV payload (SRV)	14.5	——	14.5
Attitude control	4.5	10	4.9
Command and data	2.5	10	2.7
Power	2.3	10	2.5
Mechanisms	0.5	10	0.6
Telecommunications	0.0	10	0.0
Thermal	2.0	10	2.2
Structure	5.2	10	5.7
Thruster	15.0	30	17.3
Valves, pipes	2.9	15	3.3
Propellant tank	9.5	15	10.9
System contingency	——	1	0.7
Total MAV dry mass			65.3
Propellant for Δv_1	0.5	15	0.6
Propellant for Δv_2	94.8	15	109.0
Total MAV wet mass			174.9

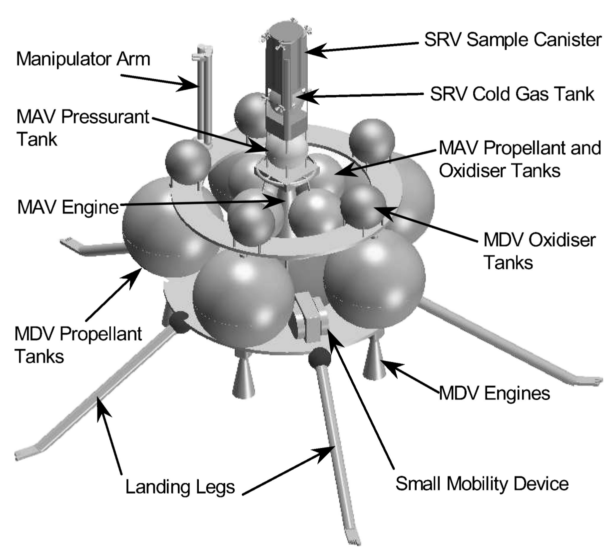

Fig. 2 Mercury sample return lander.

Table 3 MDV mass breakdown

MDV component	Mass, kg	Contingency, %	Total mass, kg
MAV and SRV	174.9	——	174.9
Surface instruments	2.9	——	2.9
Attitude control	15.0	10	16.5
Command and data	4.0	10	4.4
Power	8.8	10	9.7
Mechanisms	22.0	10	24.2
Telecommunications	0.0	10	0.0
Thermal	3.0	10	3.3
Structure	83.0	10	91.3
Thrusters (5 of 6 kN)	50.0	15	57.5
Valves, pipes	8.3	15	9.5
Propellant tanks	83.0	15	95.5
System contingency	——	1	4.9
Total MDV dry mass			494.6
Propellant for Δv_1(deorbit)	4.0	15	4.6
Propellant for Δv_2(descent)	830.8	15	955.4
Total MDV wet mass			1454.6

Table 4 SCS mass breakdown

SCS component	Mass, kg	Contingency, %	Total mass, kg
Lander (SRV/MAV/MDV)	1454.6	——	1454.6
Science payload	31.6	——	31.6
ERV	16.5	5	17.3
Attitude control	14.1	10	15.5
Command and data	10.0	10	11.0
Power	40.2	10	44.2
Mechanisms	161.0	10	177.1
Telecommunications	24.6	10	27.1
Thermal	50.0	10	55.0
Structure	65.4	10	71.9
SCS bus mass with payload	365.3	10	401.8
Total sail payload mass			1905.3

A 28-V, three domain, regulated power system is used. The SCS requires 332 W in sunlight and 310 W during eclipse, met by 365-W 6.25-m^2 GaAs solar arrays and 349-W · h lithium-ion batteries. The MDV requires 71 W, met through a 78-W 0.4-m^2 GaAs solar array. The 56-W MAV power requirement is attained through 53-W · h lithium-ion batteries. The SRV requires 24 W, provided by a 221-W · h lithium-ion battery over the SRV operational lifetime. The ballistic (ERV) uses a 41-W · h primary lithium battery to provide 1.7 W of power.

The on-orbit SCS science payload includes a high-resolution stereo camera (10 W, supporting a data rate of 10–100 bps), laser altimeter (10 W, <1 bps), infrared radiometer (5 W, 100–5000 bps), x-ray florescence spectrometer (10 W, 100–2000 bps), radio science instruments (5 W, 10–100 bps), and associated high-capacity memory (5 W, 2–5 GB capacity). There is also an 8-kg allocation for a payload of opportunity (10 W, 5 kbps). The lander has science instruments and manipulator hardware mounted on the MDV, which include a sampling device, robotic arm, and a small rover vehicle. The total data rate of these instruments corresponds to 92 Mbit every 10 h, with a total power consumption of 11.8 W.

Mercury Sample Return Solar Sail Sizing

A square solar sail is envisaged, using tip vanes for attitude control, sized to provide adequate slew rates for the planet-centered mission phases. The spacecraft (sail payload) is mounted centrally, within the plane of the solar sail, so that both faces of the core structure are free to be used as attachment points for the lander and ERV.

Figure 3 shows the approximate trip times from Earth to Mercury, generated using methods described later in the Trajectory Analysis section. An outbound trip time of 2–3 years is desirable to be competitive with solar electric propulsion (SEP) and chemical propulsion trip times. This is enabled by a characteristic acceleration of 0.25 mm/s^2. The chosen sail conceptual design used in this paper is based on the ATK (formerly Able Engineering) scalable solar sail subsystem (S4) because it can be extrapolated to large sail

Table 5 Solar sail design point data set

Component	Mass, kg
Total sail payload mass	1905
2-μm CP1 film, 2.86 g/m^2	216
0.1-μm A1 coating, 0.54 g/m^2	41
Bonding, 10% coated mass	26
Sail booms, ABLE 0.94-m booms at 70 g/m	54
Mechanical systems, 40% contingency	111
Total sail assembly mass	448
Total mission launch mass	2353
H2A202-4S capacity to $C_3 = 0$	2600
Launch mass margin	247 (9.5 %)

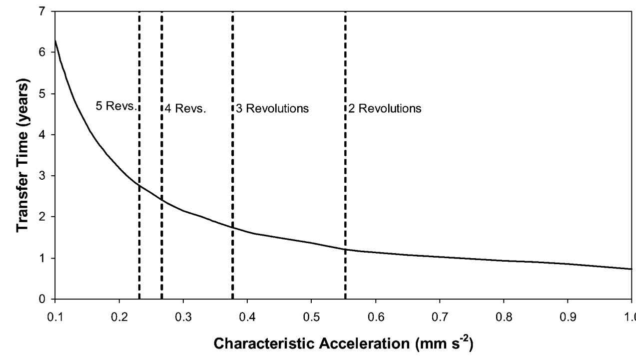

Fig. 3 Approximate Earth–Mercury transfer time.

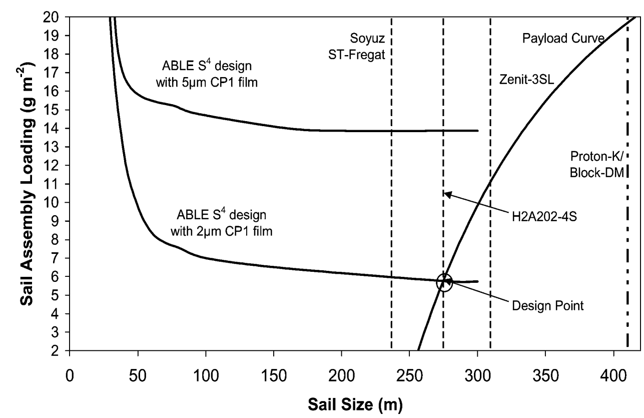

Fig. 4 Solar sail design space, 0.25 mm/s^2 (sail payload contours represent increasing parking orbit radius, with baseline 100-km orbit leftmost).

dimensions.[5] This design is based on ATK/ABLE coilable booms, and the boom linear density as a function of length can be combined with SRS technologies 2- or 5-μm colorless polyimide (CP1) film to obtain the sail assembly loading as a function of sail side length, shown in Fig. 4. It is assumed that conventional coatings are used, with aluminum (85% reflectivity) on the frontside and chromium (64% emissivity) on the backside. Figure 4 also shows the necessary sail assembly loading as a function of sail side length for delivery of a 1905-kg spacecraft to Mercury with a characteristic acceleration of 0.25 mm/s^2.

Note that the intersection of the 2-μm CP1 ABLE S4 sail design curve with the 0.25-mm/s^2, 100-km orbit payload curve yields the sail design point, with an assembly loading of 5.9 g/m^2 (including a 40% margin) and sail dimensions of 275 × 275 m. The total sail area is 75,625 m^2 supported by four booms, each of length 194 m. Note that the characteristic acceleration increases from 0.25 to 0.78 mm/s^2 after sample acquisition because the lander and docking mechanisms are jettisoned at Mercury. This results in a much faster return journey.

The solar sail has a mass of 448 kg with a mass budget as shown in Table 5. A linear boom density of 70 g/m is required with a 0.94-m diameter to maintain a factor of safety against buckling. The total

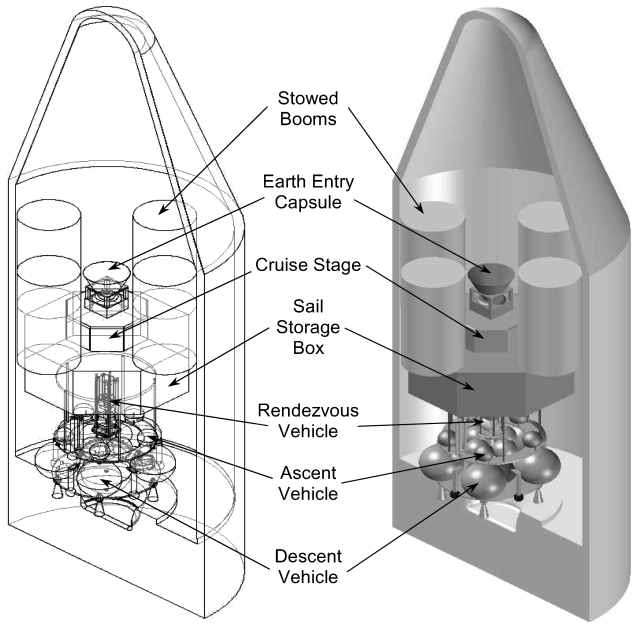

Fig. 5 Payload stack in H2A 202-4S fairing.

launch mass is, therefore, 2353 kg, which enables the use of an H2A202-4S class launch vehicle to Earth escape velocity, $C_3 = 0$. The spacecraft stack with stowed sail fits into the H2A fairing, as shown in Fig. 5.

Mercury Sample Return Cost Analysis

The spacecraft has been costed using parametric cost estimating relationships.[6] Conservative margins have been added, and the cost of specialist components, such as bipropellant engines, have been taken from JPL estimates.[1] Project management and integration and support costs are also estimated using Ref. 6. The most difficult system to cost is that of the solar sail because a sail let alone one of 275-m dimension, has yet to fly, but note that the cost of the sail would be small in comparison with the spacecraft itself. This assumes that the required technologies are available and that the research and development costs are not included. In addition, the reduction in launch cost compared with conventional concepts should make up for the sail cost. Note that, although the H2A launch cost is fairly low at ~\$95 million, the predominant cost component will be the lander, which is mostly independent of the primary propulsion method used. Traditionally, solar sailing is seen to be superior to chemical propulsion or SEP if it can reduce launch mass and cost, but for a sample return mission the sail must significantly reduce launch mass for there to be any appreciable reduction in overall mission cost. The launch cost has been reduced for a solar-sail-powered Mercury sample return mission.

Mercury Sample Return Trajectory Analysis

The required Δv for direct ballistic transfer to a low Mercury parking orbit is of the order of 13 km/s. Chemical propulsion and SEP both require a prolonged sequence of gravity assists to reduce launch mass. Mercury sample return from deep within the solar gravity well is, therefore, an extremely energetically demanding mission concept. However, propellant mass is not an issue for solar sail spacecraft, and so the spacecraft can spiral directly to Mercury, making the best use of the inverse square increase in solar radiation pressure at lower heliocentric distances. Many authors have recognized the benefit of solar sailing to reach Mercury, but this paper provides new data sets by considering both launch windows and return trajectories.

Heliocentric trajectories have been optimized using the constrained parameter optimization algorithm NPSOL based on sequential quadratic programming.[7,8] Homotopic methods were used to obtain initial solutions for optimization. Planet-centered maneuvers are modeled using a set of blended analytical control laws.[9] Mercury capture and escape trajectories have been generated using a control law that maximizes the rate of change of orbit energy. Several control laws are then blended for Mercury-centered transfer maneuvers.

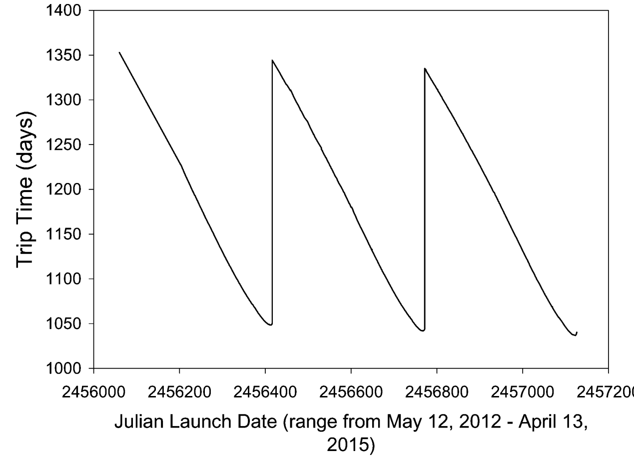

Fig. 6 Earth–Mercury departure date scan.

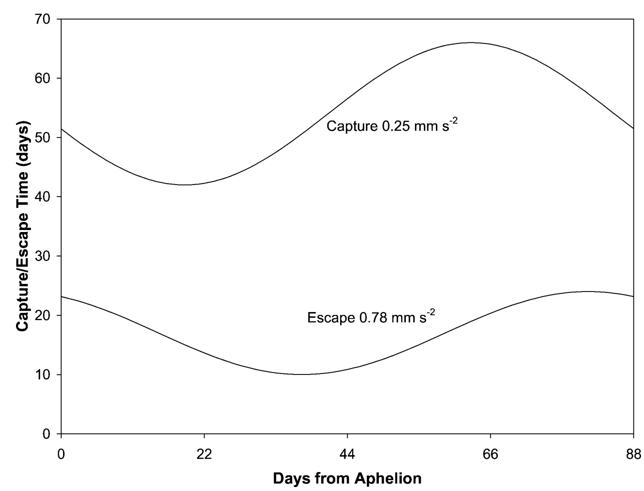

Fig. 7 Mercury capture/escape time variation.

Launch Windows

Figure 6 shows the Earth departure date scan for the selected characteristic acceleration of 0.25 mm/s^2 over a 3-year period. Each point on the curve represents an optimization at that launch date. Note that the minimum-time launch opportunities occur once every year. Solar sail spacecraft are not restricted to launch windows, but it is clear that a saving of 300 days can be achieved depending on launch date. The discontinuities pose problems when incrementing the launch date to find initial solutions for other launch dates. These discontinuities are due to the spacecraft just missing the target and having to execute another revolution of the sun to reach Mercury.

To determine the optimal launch date, consideration must also be given to the variation of the capture and escape times along Mercury's orbit and the return Mercury–Earth phase. Because Mercury has an eccentricity of 0.2056, the available solar radiation pressure will vary significantly over a Mercury year.[10] Approximate capture and escape times are shown in Fig. 7 for the different accelerations outlined earlier.

With an orbiter stay time of the order of 40 days, Figs. 6 and 7 can be used to ascertain that the return scan was only required across a two-year range. The four curves were then mapped together to determine the overall mission duration as a function of Earth departure date. The total mission duration departure date scan was remarkably similar to Fig. 6 because the long-duration outbound

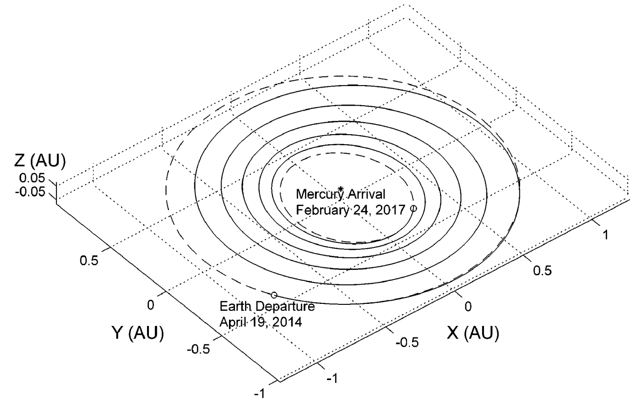

Fig. 8 Earth–Mercury trajectory.

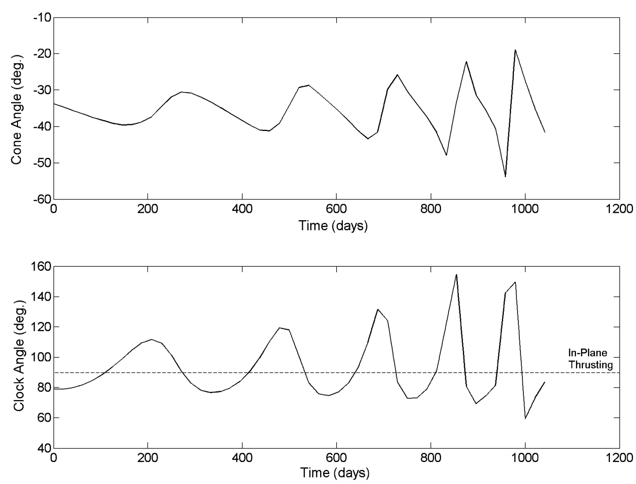

Fig. 9 Earth–Mercury control angle profile.

spiral dominates the total mission duration. The launch opportunity selected was that on 19 April 2014.

Earth–Mercury Phase

The outbound trajectory is shown in Fig. 8, departing Earth with a $C3$ of zero on 19 April 2014. Mercury arrival is on 24 February 2017, 2.85 years later, after $5\frac{1}{4}$ revolutions. The optimal cone and clock control angles are shown in Fig. 9. Even at a relatively coarse control resolution of 50 linear interpolation segments, the profiles are smooth and oscillatory.

The equilibrium sail film temperature was modeled using a blackbody approximation, assuming temperature changes take place instantaneously because the micrometer-scale thickness of the sail film ensures that the thermal capacity is effectively zero. Aluminum/chromium coatings were assumed as was discussed earlier. The temperature is a function of both the radius and the sail attitude, with a maximum value of 444 K at 980 days (0.37 AU). Even face on to the sun at Mercury perihelion, the worst-case temperature would be 495 K, still less than the predicted 520 K upper limit of polyimide films.

Mercury-Centered Maneuvers

It has been assumed that the sail arrives at Mercury with zero hyperbolic excess velocity. The transition from heliocentric to planet-centered motion has not been patched. However, it is assumed that the sail can be used to correct for approach dispersion and can target the correct B plane for capture. As has been prescribed, capture is into a 100×7500 km sun-synchronous polar orbit, 10 deg ahead of the terminator, before subsequent maneuvering into the 100-km parking orbit. This capture spiral takes 28 days, arriving on orbit on 24 March 2017.

A duration of 131 days will be available for orbital science operations, surface observation, and final maneuvering to the lander descent orbit. This orbiter stay time is also a requirement due to the thermal environment on the surface. The thermally benign sun-synchronous orbit (10 deg ahead of terminator) is forced for 44 days until the orbit is in the correct orientation for the landing site. The sail then waits in this orbit for 37 days. Next, a 50-day maneuver transfers the spacecraft to the 100-km polar orbit, where the lander begins its descent on 3 August 2017. Once on the surface, the lander carries out four days of sample acquisition and landing site documentation operations, limited by surface and orbiter thermal constraints. The solar sail spacecraft is used to rotate the orbit plane to account for Mercury landing site rotation, and so that the MAV ascends in a coplanar maneuver. The orbit plane cannot be rotated as fast as Mercury spins, and so the MAV will need to wait in the 100-km orbit (thermally safe) until solar sail spacecraft rendezvous with the MAV. Final proximity maneuvering is accomplished with the SRV, thereby relaxing MAV launch accuracy. After sample transfer to the ERV attached to the sail, the solar sail spacecraft spirals to escape. A control method that maximizes the rate of change of orbit energy while maintaining a positive altitude of periapsis is used. The escape spiral is initiated on 18 August 2017, with escape reached in 16 days.

Mercury–Earth Phase

Return heliocentric spiraling commences after Mercury escape on 3 September 2017. The trip time is 369 days, with arrival back at the Earth with zero hyperbolic excess on 8 September 2018. Figure 10 shows the two revolution trajectory, which is faster because the sail characteristic acceleration has increased to 0.78 mm/s^2. The cone and clock angle control profile is shown in Fig. 11. Finally, the ERV spins up and is separated from the SCS to perform a ballistic entry for sample delivery to Earth. The sail performs a postseparation Earth avoidance maneuver and does not enter a bound orbit, with

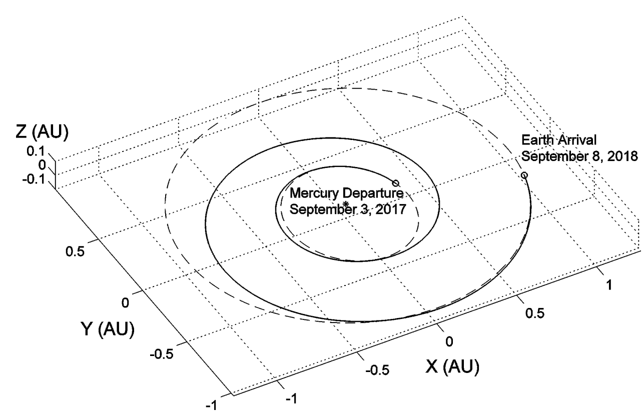

Fig. 10 Mercury–Earth trajectory.

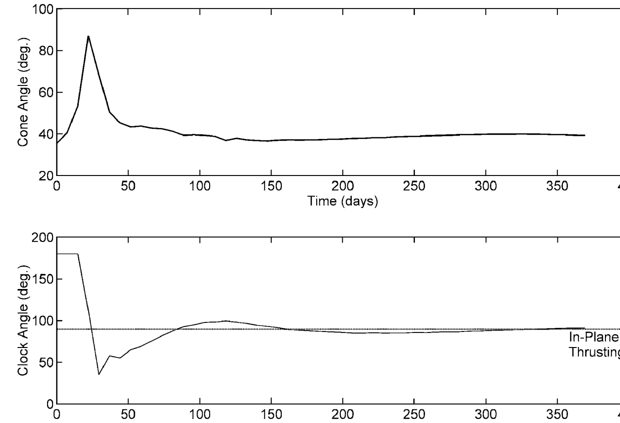

Fig. 11 Mercury–Earth control angle profile.

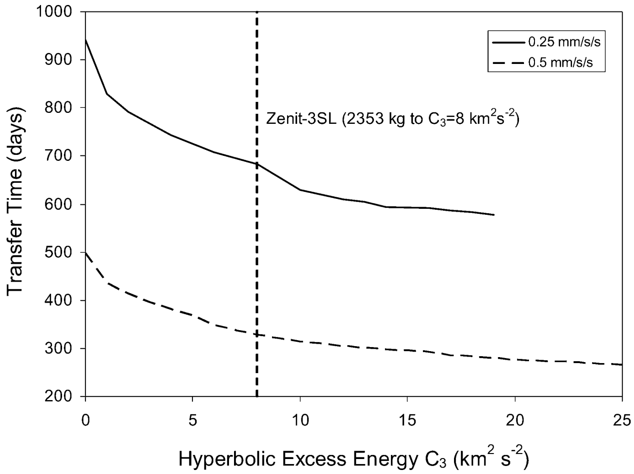

Fig. 12 Effect of hyperbolic excess energy at launch.

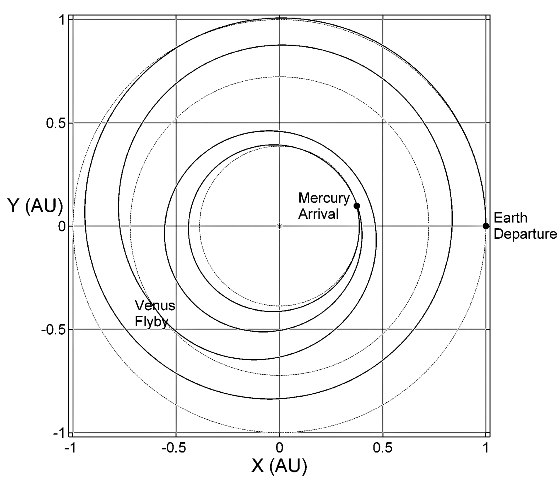

Fig. 13 Venus gravity assist.

the sail steered away from the Earth's sphere of influence. The total mission duration is 4.39 years.

Alternative Trajectory Options

Use of a positive launch C_3 would be highly advantageous for reaching close solar orbits such as that of Mercury. The initial eccentricity for the inward spiral can be easily circularized by the increased solar radiation pressure closer to the sun. Figure 12 shows the effect of using excess launch energy to reduce the trip time to Mercury orbit. Note that the effect is greater for lower accelerations because the trip time is longer and there are more revolutions for $C_3 = 0$. The use of a Zenit 3-SL over an H2A would allow for a $C_3 = 8$ km^2/s^2, which would reduce the outbound trip time by 260 days for the same launcher cost. Figure 13 shows that the inclusion of a Venus gravity assist could reduce the outbound trip time by 140 days (Ref. 7), but gravity assists are not essential for solar sail spacecraft because propellant mass is not an issue.

Mars Sample Return

In addition to Mercury sample return, an assessment of solar-sail-powered Mars sample return mission concepts has been made and will be summarized here. The top-level science goal is to return 250 g of documented surface samples.[11] Several mission architectures have been explored that exploit the benefits of solar sailing by utilizing the solar sail to reduce mission launch mass. A baseline mission concept has been proposed to deliver a lander and ascent vehicle to a 300-km low Mars orbit. To provide a reasonable mission duration, a requirement for a solar sail characteristic acceleration of 0.5 mm/s^2 has been identified. In addition, to provide a useful

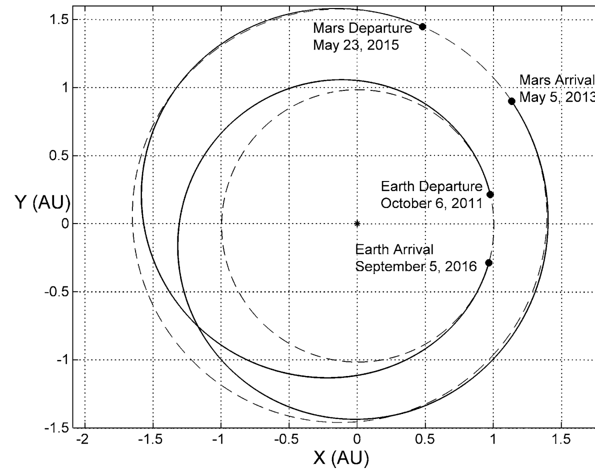

Fig. 14 Earth–Mars sample return trajectory.

reduction in launch mass over conventional Mars sample return mission concepts, a sail assembly loading of order 5 g/m^2 appears necessary. Analysis has shown that, when an internal EADS Astrium 182-kg lander concept is used, a 179×179 m solar sail is required to acquire and return a surface sample in 6.0 years, with a total launch mass of 495 kg (with margin). The forthcoming Vega launcher is selected to deliver the stowed solar sail spacecraft to a 2500-km Earth orbit from where it then spirals to Earth escape (although the prolonged radiation exposure through the Van Allen belts may cause some degradation of the sail and is an important issue). A cost analysis was performed that included ground segment costs and sample handling on return to Earth. A launch opportunity has been identified on 13 August 2010, corresponding to a departure from the Earth's sphere of influence on 6 October 2011, with sample return to Earth on 5 September 2016. Starting the mission from Earth escape would allow the mission duration to be shortened by 1.14 years, at the expense of requiring a larger launch vehicle, such as a Soyuz–Fregat or Delta II 7925. Figure 14 shows the outbound and return interplanetary trajectories.

When availability of 5-g/m^2 sail technology is assumed, it appears that solar sailing can provide some benefits for a Mars sample return mission by reducing launch mass and launch energy requirements, but at the expense of a significantly longer mission duration relative to conventional propulsion. In addition, arrival speeds at Mars and Earth are much more reduced (parabolic approach) than they are for conventional mission concepts, thus reducing thermal loads on the Mars lander and Earth return capsule and improving the Mars landing ellipse. However, specific issues have been identified that are unique to solar-sail-powered Mars sample return mission concepts. In particular, the Mars ascent stage must be the active vehicle during the final rendezvous, and docking in low Mars orbit and systems integration of the cruise stage will require care to provide adequate sensor and antenna visibility.

Venus Sample Return

To complete our assessment of terrestrial planet sample return using solar sails, a solar-sail-powered Venus sample return mission concept has been analyzed with a top-level science goal of returning 200 g of documented surface samples and atmospheric samples.[12] Atmospheric samples are an important aspect, not only for understanding the complex chemistry, but because it is thought that the upper atmosphere may harbor life-critical organic compounds. After the investigation of several mission architectures, the most promising baseline mission concept proposed uses a ballistic Earth–Venus transfer combined with aerobraking to deliver a lander to a low Venusian parking orbit. After sample acquisition, a relatively small solar sail is then used for the Venus–Earth return phase of the mission only. This architecture reduces launch mass while providing a relatively fast trip time and avoids the extremely large sail required for an all-solar-sail mission due to the large descent and ascent stages

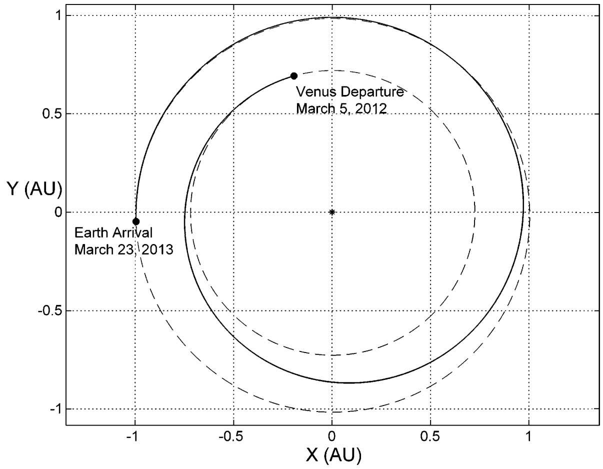

Fig. 15 Venus–Earth trajectory.

necessary for surface sample acquisition as a result of the dense atmosphere and relatively strong gravity of Venus.

Analysis has shown that, using an existing 2053-kg lander concept,[13] the optimum mission architecture is to use a Zenit-3SL launch vehicle to deliver the lander and solar sail return vehicle onto a ballistic Earth–Venus transfer with a C_3 of 8 km^2/s^2. After a short 122-day ballistic transfer to Venus, 250 days of aerobraking are required to transfer to a 900-km Venusian parking orbit. The lander then descends, and an ascent stage transfers the samples to an Earth return capsule attached to a stowed solar sail. The 92×92 m solar sail, with a sail assembly loading of 6.2 g/m^2, then deploys and initiates the Earth return phase of the mission. The solar sail characteristic acceleration is 0.38 mm/s^2, with a total mission launch mass of 2643 kg (with margin). A cost analysis was also performed for this mission. A launch opportunity has been identified on 3 August 2010 with sample return to Earth return on 23 March 2013, resulting in a total mission duration of 2.6 years. Figure 15 shows the return leg, enabled by a small solar sail, deployed once sample acquisition is complete.

It appears that solar sailing can provide benefits for a Venus sample return mission by reducing launch mass and trip times relative to other mission concepts. when a solar sail is used for sample return only, an extremely large sail is avoided, as would be required to deliver the large 2053-kg lander to Venus using solar sail propulsion alone. In addition, by deploying the solar sail after sample transfer has been completed, issues associated with docking to a deployed solar sail, such as thruster plume impingement and mechanical loads, are avoided.

Mission Evaluation

The optimal solar sail mission designs were compared with existing concepts utilizing chemical and/or SEP. The chemical Mars sample return mission was taken from an internal EADS Astrium concept. No comparison was made with a solar electric Mars sample return concept because, as has been found for solar sailing, a low-thrust SEP trajectory is also of long duration. Chemical propulsion can be used to reach Mars relatively easily, and so SEP Mars sample return studies rarely appear in the literature. Note from Table 6 that the solar sail performs surprisingly well in terms of launch mass. Note here that that although launch costs can be reduced with solar sails, the predominant cost for any sample return mission is likely to be in the lander, not the launch vehicle. The Δv for a Mars sample return is not high enough that solar sails would provide significant benefit because the solar sail exhibits decreased thrust at Mars's distance from the sun. The major disadvantage is that the total mission duration is three times longer for the solar sail, making the use of solar sailing unattractive for this scenario.

Venus sample return studies selected for comparison include an all-chemical option[13] and a SEP option.[14] Note from Table 6 that the chemical outbound–solar sail return option can return the sample

Table 6 Comparison with existing terrestrial body sample return concepts

Parameter	Solar sail	Chemical	SEP
	Mars		
Launch vehicle	Vega	Soyuz–Fregat	——
Launch mass, kg	495	1240	——
C_3, km^2/s^2	<0	10.2	——
Mission duration, years	6.0	2.0	——
Launch cost, M$	∼20	∼40	——
	Venus		
Launch vehicle	Zenit-3SL[a]	Delta V	Atlas IIIB
Launch mass, kg	2643[a]	3326	4300
C_3, km^2/s^2	8[a]	8	−16
Mission Duration, years	2.7[a]	2.1	5.1
Launch cost, M$	∼95[a]	∼180	∼115
	Mercury		
Launch vehicle	H2A 202-4S	Ariane 5E[b]	Atlas V 551
Launch mass, kg	2353	6500[b]	5775
C_3, km^2/s^2	0	11.6[b]	7.8
Mission duration, years	4.4	7.2[b]	6.9
Launch cost, M$	∼95	TBD[b]	∼136

[a]Chemical outbound, sail return only. [b]Chemical/SEP.

from Venus in a comparable time to the all-chemical option, but with an appreciable reduction in launch mass. The solar sail option is substantially faster than the SEP concept, with a reduction in launch mass. Of the three propulsion methods, it has been found that the total mission cost is approximately the same for each because by far the largest cost fraction is contained in the lander/ascent vehicle due to the difficulty in retrieving a sample from the surface of Venus.

For the Mercury sample return scenario, the outlook is considerably better when using solar sailing. This is because with chemical propulsion, and even SEP, we have to resort to gravity assists to reach Mercury in an attempt to conserve propellant. Indeed, it is not feasible for an all-chemical spacecraft to return a sample from Mercury, and so the existing concept adopted was a chemical/SEP combination from an internal ESA study. This concept has a 6500-kg launch mass on an Ariane 5E, with a mission duration of 7.2 years. An all SEP concept proposed by JPL has a 5775-kg launch mass, which requires an Atlas V 551 launcher, for a 6.9-year mission.[1] Note from Table 6 that the launch mass is considerably reduced (on the order of 60% less) when using solar sail propulsion, enabling the use of a much smaller and cheaper launch vehicle. The total mission duration is also much reduced (by around 40%) because of the elimination of gravity assists. This significant reduction in launch mass corresponds to an appreciable reduction in launch cost that could have a noticeable impact on the overall cost of a Mercury sample return mission.

Conclusions

Systems and trajectory analysis has been conducted into the use of solar sail propulsion to return samples from the three terrestrial planets, with the Mercury sample return mission analysis described here in more detail. The science objectives for a Mercury sample return mission were outlined, and the necessary spacecraft and lander subsystems were customized and adapted for use with a solar sail. The solar sail payload comprises four modules with a total mass of 1905 kg. Power and thermal subsystems were sized accordingly for Mercury operations with specific science instruments. Solar sail sizing was then performed in a parametric process with a realistic characteristic acceleration of 0.25 mm/s^2. Extrapolation of existing boom and film technologies in a graphical method enabled a square sail side length of 275 m to be identified. The total launch mass obtained required the use of an H2A class launch vehicle. Detailed trajectory analysis, including identification of optimal departure windows and Mercury-centered maneuvering, revealed that the total mission duration was just 4.4 years. Although described here in less detail, similar mission analyses were conducted for solar-sail-powered Mars and Venus sample return.

Comparison was made between solar sail propulsion and existing chemical and SEP sample return concepts. It appears that solar sailing offers only reduced launch mass benefits to returning a sample from Mars or Venus, but the launch cost is probably small compared to the lander cost. Solar sailing is not attractive for Mars sample return due to the threefold increase in mission duration. Venus sample return just about breaks even with a solar sail for return only because the launch mass is much reduced. The dominant cost for this mission is likely to be the lander vehicle, and so there is again little benefit from using solar sailing. However, it has been clearly demonstrated that solar sail propulsion can provide significant benefits for Mercury sample return, which has been discussed in greater detail in this paper. It has been shown that the solar sail can reduce launch mass by approximately 60% while reducing mission duration by 40%. The launch cost reduction will considerably reduce overall mission cost. Therefore, is it recommended that future studies focus on sample return missions where solar sails are truly enabling: that of sample return from Mercury, or, indeed, high Δv missions within the inner solar system in general. It has traditionally been assumed that any appreciable reduction in launch mass and launch cost makes the solar sail the best propulsion candidate. However, the solar sail must significantly reduce launch mass and cost for there to be any significant reduction in overall sample return mission cost because these missions have complex landers that are likely to dominate the total mission cost. Finally, note that these missions require significant solar sail hardware development in addition to generic lander issues, particularly of large sail designs (>100 m) and bulk manufacture of 2-μm film.

Acknowledgment

This work was performed under ESA/European Space Research and Technology Centre Contract 16534/02/NL/NR: Technical Assistance in the Study of Science Payloads Transported through Solar Sailing.

References

[1]Oberto, R., "Mercury Sample Return 2002-01," Advanced Projects Design Team (Team X) Rept. [online], Jet Propulsion Lab., California Inst. of Technology, Pasadena, CA, Jan. 2002, URL: http://www.lsmd.caltech.edu/surf/documents/mercury_sample_return.doc [cited 1 June 2005].

[2]McInnes, C. R., *Solar Sailing: Technology, Dynamics and Mission Applications*, 1st ed., Springer-Verlag, Chichester, England, U.K., 1999, Sec. 1.3.2.

[3]McInnes, C. R., Hughes, G. W., and Macdonald, M., "Technical Note 3—Mercury Sample Return," ESA/European Space Research and Technology Centre CR 16534/02/NL/NR, Dept. of Aerospace Engineering, Univ. of Glasgow, Glasgow, Scotland, U.K., May 2003.

[4]Leipold, M. E., and Wagner, O., "Mercury Sun-Synchronous Polar Orbits Using Solar Sail Propulsion," *Journal of Guidance, Control, and Dynamics*, Vol. 19, No. 6, 1996, pp. 1337–1341.

[5]Murphy, D. M., Murphey, T. W., and Gierow, P. A., "Scalable Solar-Sail Subsystem Design Concept," *Journal of Spacecraft and Rockets*, Vol. 40, No. 4, 2003, pp. 539–547.

[6]Wertz, J. R., and Larson, W. J., *Space Mission Analysis and Design*, Kluwer Academic, Dordrecht, The Netherlands, and Microcosm, Torrance, CA, 1999, pp. 795–802.

[7]Hughes, G. W., and McInnes, C. R., "Mercury Sample Return Missions Using Solar Sail Propulsion," International Astronautical Congress, Paper IAC-02-W-2.08, Oct. 2002.

[8]Hughes, G. W., and McInnes, C. R., "Small-Body Encounters Using Solar Sail Propulsion," *Journal of Spacecraft and Rockets*, Vol. 41, No.1, 2004, pp. 140–150.

[9]Macdonald, M., and McInnes, C. R., "Analytic Control Laws for Near-Optimal Geocentric Solar Sail Transfers," *Advances in the Astronautical Sciences*, Vol. 109, No. 3, 2001, pp. 2393–2413.

[10]Macdonald, M., and McInnes, C. R., "Seasonal Efficiencies of Solar Sailing in Planetary Orbit," International Astronautical Congress, Paper IAC-02-S.6.01, Oct. 2002.

[11]McInnes, C. R., Hughes, G. W., and Macdonald, M., "Technical Note 1—Mars Sample Return," ESA/European Space Research and Technology Centre, CR 16534/02/NL/NR, Dept. of Aerospace Engineering, Univ. of Glasgow, Glasgow, Scotland, U.K., Feb. 2003.

[12]McInnes, C. R., Hughes, G. W., and Macdonald, M., " Technical Note 2—Venus Sample Return," ESA/European Space Research and Technology Centre, CR 16534/02/NL/NR, Dept. of Aerospace Engineering, Univ. of Glasgow, Glasgow, Scotland, U.K., March 2003.

[13]Oberto, R., "Venus Surface Sample Return 1-99," Advanced Projects Design Team (Team X) Rept. [online], Jet Propulsion Lab., California Inst. of Technology, Pasadena, CA, Feb. 1999, URL: http://www.lsmd.caltech.edu/surf/documents/venus_sample_return.doc [cited 1 June 2005].

[14]Sercel, J., "PowerSail for Venus Sample Return," Surf 2002 Final Presentation [online], Lab. for Spacecraft and Mission Design, California Inst. of Technology, Pasadena, CA, Aug. 2002, URL: http://www.lsmd.caltech.edu/projects/powersail/powersail_vsr_presentation.pdf [cited 1 June 2005].

D. Spencer
Associate Editor