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Solar sailing: mission applications and engineering challenges

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Solar sailing is emerging as a promising form of advanced spacecraft propulsion, which can enable exciting new space-science mission concepts. By exploiting the momentum transported by solar photons, solar sails can perform high-energy orbit-transfer manoeuvres without the need for reaction mass. Missions such as planetary-sample return, multiple small-body rendezvous and fast missions to the outer Solar System can therefore be enabled with the use of only a modest launch vehicle. In addition, new families of highly non-Keplerian orbits have been identified that are unique to solar sails, and can enable new ways of performing space-science missions. While the opportunities presented by solar sailing are appealing, engineering challenges are still to be solved before the technology finally comes to fruition.

Keywords: solar sailing; space missions; spacecraft engineering

1. Introduction

For all of its short history, practical spacecraft propulsion has been dominated by the unaltering principles of Newton's third law. All forms of propulsion, from simple solid rocket motors to complex solar-electric ion engines, rely on a reaction mass which is accelerated into a high-velocity jet by some exothermal or electromagnetic means. A unique and elegant form of propulsion which transcends this reliance on reaction mass is the solar sail. Since solar sails are not limited by a finite reaction mass, they can provide continual acceleration, limited only by the lifetime of the sail film in the space environment. Of course, solar sails must also obey Newton's third law. However, solar sails gain momentum from an ambient source, namely photons, the quantum packets of energy of which sunlight is composed. It may be surprising that photons with zero rest mass can push matter; however, relativity states that a zero-rest-mass particle with energy E, will also transport momentum $p = E/c$, where c is the speed of light.

The momentum transported by an individual photon is almost vanishingly small. Therefore, in order to intercept large numbers of photons, solar sails must have a large, extended surface, typically a square sail held in tension by four deployable diagonal booms. Furthermore, to generate as high an acceleration as possible from the momentum transported by the intercepted photons, solar sails must also be

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extremely light. For a typical solar sail, the mass per unit area of the entire spacecraft may be an order of magnitude less than that of the paper upon which this text is written. Not only must solar sails have a small areal density, they must also be near perfect reflectors. The momentum transferred to the sail can then be almost double the momentum transported by the incident photons. At best though, only 9 N of force is available for every square kilometre of sail located at the Earth's distance from the Sun, so that the solar sail will experience a small, but continuous acceleration.

Adding the impulse due to incident and reflected photons, it can be shown that the light-pressure-induced force is directed almost normal to the surface of the solar sail. Then, by controlling the orientation of the solar sail relative to the Sun, and thus directing the thrust vector, the solar sail can gain or lose orbital angular momentum. If the solar-sail thrust vector is oriented such that there is a component of thrust in the direction of its instantaneous velocity vector, orbital angular momentum will be gained, while if the solar-sail thrust vector is oriented so that there is a component of thrust opposite to the direction of its velocity vector, orbital angular momentum will be lost. In this way, the solar sail is able to tack, spiralling inwards towards the Sun, or outwards to the farthest edge of the Solar System. Indeed, by orienting the solar sail so that a component of the thrust vector is directed normal to the orbit plane, the orbit inclination can be controlled, allowing three-dimensional transfers to any desired, final orbit. The orientation of the solar sail can, in principle, be controlled by using articulated reflective vanes at the tips of the sail booms to displace the centre of pressure of the solar sail relative to its centre of mass, thus generating attitude-control torques. Alternatively, attitude-control torques can be generated by mounting the payload on a gimballed boom and displacing the centre of mass of the solar sail relative to its centre of pressure.

The picture is clear then. A solar sail is a large shining membrane of thin reflective film, held in tension by some gossamer structure. Using the momentum gained by reflecting ambient sunlight, the solar sail is accelerated slowly, but continuously, to accomplish any number of possible missions. Without the violence of reaction propulsion, the solar sail is tapping a tiny fraction of the energy released through nuclear fusion at the core of the Sun. Solar sailing, with its analogies with terrestrial sailing, may seem a fanciful and romantic notion. However, as will be shown in this paper, the romanticism is overshadowed by the immense practicability and quiet efficiency with which solar sails can be put to use.

2. Historical background

Although solar sailing has been considered as a practical means of spacecraft propulsion only relatively recently, the fundamental ideas are by no means new. While the existence of light pressure was demonstrated in theory by the Scottish physicist James Clerk Maxwell in 1873, it was not measured experimentally until precision laboratory tests were performed by the Russian physicist Peter Lebedew in 1900 (see Lebedew 1902). As early as the 1920s, the Soviet father of astronautics, Konstantin Tsiolkovsky and his co-worker Fridrickh Tsander wrote of using 'tremendous mirrors of very thin sheets' (Tsiolkovsky 1936) and using 'the pressure of sunlight to attain cosmic velocities' (see Tsander (1924) and references therein). Although there is some uncertainty regarding dates, it appears that Tsander was the first to write of practical solar sailing, some time late in the summer of 1924.

The modern concept of solar sailing was reinvented much later by Richard Garwin at the IBM Watson laboratory in New Jersey. Garwin authored the first paper on solar sailing in a western technical publication (the journal *Jet Propulsion*) in 1958, and coined the term 'solar sailing' (Garwin 1958). During the course of NASA-funded work in the early 1970s, Jerome Wright at the Batelle Laboratories in Ohio discovered a trajectory that could allow a solar sail to rendezvous with comet Halley during its pass through the inner Solar System in the mid 1980s. The flight time of only four years would have allowed for a late-1981 or early-1982 launch. Previously, a seven-to-eight year mission had been envisaged by NASA, using solar-electric-ion propulsion, requiring a launch as early as 1977. Solar-electric propulsion uses large deployable photovoltaic arrays to generate kilowatt levels of electrical power, which is then used to accelerate a stream of ions (typically xenon) through an intense electric field, resulting in an efficient high-speed, but low-thrust jet. The design of a comet-Halley-rendezvous mission using solar-sail propulsion was initiated at the NASA Jet Propulsion Laboratory in November 1976 (Friedman 1978).

During the initial design study, an extremely large (800 m $\times$ 800 m) three-axis stabilized square solar sail with four deployable booms was considered, but was dropped in May 1977 due to the perceived risks associated with boom and sail deployment on such a scale. The design work then focused on a spin-stabilized heliogyro configuration. The heliogyro (which was to use 12 blades of film, each 7.5 km long, rather than a single sheet) had been proposed some 10 years earlier. In principle, the heliogyro could be more easily deployed than the square solar sail by simply unrolling the individual blades of the spinning structure. Although the heliogyro was seen as the optimum configuration for the large comet Halley mission, recent solar-sail mission concepts require much smaller sails, due to advances in spacecraft-payload miniaturization. The square-solar-sail configuration is now seen as optimum for these smaller solar sails.

As a result of the strong interest in solar sailing, proponents of solar-electric propulsion re-evaluated their performance estimates and in the end were competing directly with solar sailing. After an evaluation of these two advanced propulsion concepts, NASA selected the solar-electric system in September 1977 upon its merits of being a lesser, but still considerable, risk for a comet Halley rendezvous. A short time later, a rendezvous mission using solar-electric propulsion was also dropped, due to escalating cost estimates (Logsdon 1989).

Although a true solar sail has yet to be flown, the 1990s saw the development and flight testing of some key technologies for future use. The Russian company NPO Energia successfully deployed a spinning 20 m reflector from a Progress supply vehicle in February 1993. Another spectacular demonstration of a large deployable reflector was achieved in May 1996 during the STS-77 space shuttle mission. The 14 m diameter Inflatable Antenna Experiment was designed to test the deployment of a large, space-rigidized inflatable structure, to be used principally as a radio frequency reflector, although space-rigidized-inflatable-boom technology has driven recent concepts for near-term solar sails.

More recently, the European Space Agency (ESA) and the German aerospace agency DLR (Deutschen Zentrum für Luft-und Raumfahrt) co-funded the fabrication of a 20 m $\times$ 20 m solar sail which was successfully ground-tested in December 1999, as shown in figure 1. The solar sail used lenticular carbon-fibre booms, which can be pressed flat and wound onto a roller for packing, and then unwound for deployment

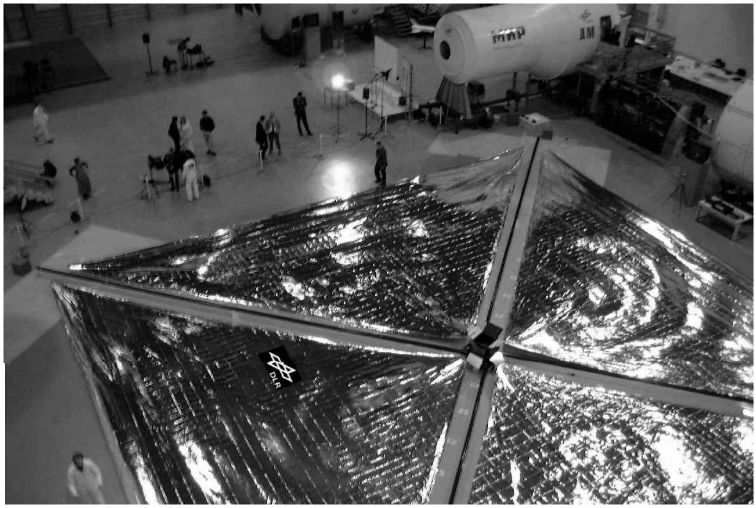

Figure 1. 20 m × 20 m solar-sail deployment test, December 1999. (Reproduced with permission from DLR.)

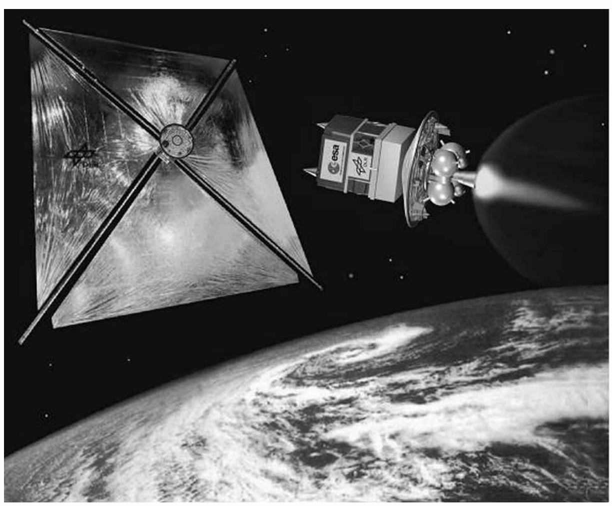

Figure 2. ESA solar-sail deployment test scheduled for 2005. (Reproduced with permission from Babakin Science and Research Space Center.)

with the stored elastic energy in the booms restoring their original lenticular cross-section (Leipold 1998). Following this successful ground test, the agency funded a series of mission studies at the University of Glasgow to investigate the potential of the technology for future space-science-mission applications. It was quickly established that solar sailing could reduce both the trip time and launch mass, and hence cost, of several future ESA missions, such as the COLOMBO Mercury orbiter and the SOLO high-inclination solar physics observatory, both of which will use solar-electric propulsion. Looking to the future, an in-orbit deployment test of a 20 m × 20 m solar sail is currently being funded by ESA, with the launch scheduled for 2005, as shown in figure 2. Aside from these agency activities, the non-profit Planetary Society is coordinating the fabrication of a small solar sail in Russia, funded by a US media corporation, with the launch currently scheduled for late 2003 or early 2004.

3. Solar-sail design

The fundamental measure of performance of a solar sail is its characteristic acceleration, defined as the light-pressure-induced acceleration experienced by the solar sail while oriented normal to the Sun at a heliocentric distance of one astronomical unit (1 AU), the mean distance of the Earth from the Sun (McInnes 1999*b*). The characteristic acceleration is a function of both the efficiency of the solar-sail design and the mass of the payload. At a distance of 1 AU, the magnitude of the solar-light pressure P exerted on a perfectly absorbing surface is $4.56 \times 10^{-6}\ \mathrm{N\,m^{-2}}$. Therefore, allowing for the reflection of photons (factor of two) and the finite efficiency of the sail η, the characteristic acceleration a_0 is defined by

$$a_0 = \frac{2\eta P}{\sigma}, \quad \sigma = \frac{m_\mathrm{T}}{A}, \tag{3.1}$$

where σ is the solar-sail loading, with m_T the total mass of the solar sail and its payload, and A the sail area. The sail efficiency η (typically around 0.85) is a function of both the optical properties of the sail film and the sail shape due to billowing and wrinkling. The total mass of the solar sail will now be partitioned into two components, the sail film and structural mass m_S and the payload mass m_P. Therefore, the characteristic acceleration of the solar sail may now be written as

$$a_0 = \frac{2\eta P}{\sigma_\mathrm{S} + (m_\mathrm{P}/A)}, \quad \sigma_\mathrm{S} = \frac{m_\mathrm{S}}{A}, \tag{3.2}$$

where σ_S is the mass per unit area of the sail assembly. This so-called sail-assembly loading is a key technology parameter and is a measure of the thickness of the sail film and the efficiency of the solar-sail structural and mechanical design.

Current ESA funded solar-sail development work using carbon-fibre booms and commercially available 7.5 μm Kapton film projects a sail-assembly loading of the order of $30\ \mathrm{g\,m^{-2}}$, which is adequate for future technology demonstration missions. Other development work at NASA and elsewhere to fabricate ultra-thin sail films with a thickness of the order of 1 μm, and high stiffness, low-mass booms, could lead to a sail-assembly loading of the order of $5\ \mathrm{g\,m^{-2}}$ or less for future missions (Salama *et al*. 2001). Most of the mass of the sail film is due to the polyimide plastic substrate required to provide sufficient tensile strength for fabrication, packing and in-orbit deployment. Typically, the reflective coating on the front side of the sail would be a vapour-deposited 0.1 μm thick layer of aluminium, while the back of the sail would be coated with a 0.01 μm thick layer of high-emissivity chromium for thermal control. Future concepts also envisage the use of substrates that will evaporate under the action of solar UV radiation, leaving a thin metallic film, or indeed metallic films with submicrometre perforations (to reduce mass while maintaining good optical reflectivity), resulting in significant improvements in solar-sail performance.

Now that the key solar-sail design parameters have been defined, the process of sizing a solar sail will be considered. From equation (3.2) it can be seen that the solar-sail payload mass may be written as

$$m_\mathrm{P} = \left[\frac{2\eta P}{a_0} - \sigma_\mathrm{S}\right] A. \tag{3.3}$$

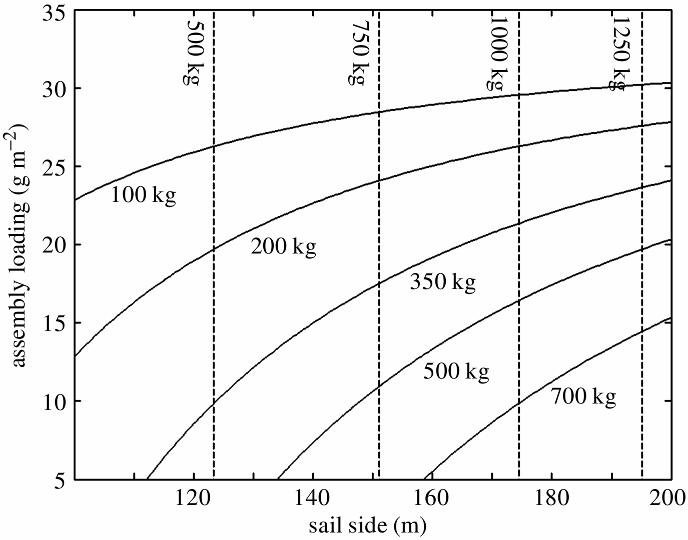

Figure 3. Solar-sail design space. Solid line, payload mass; dashed line, total mass.

Similarly, from equation (3.1) the total mass of the solar sail may be written as

$$m_{\mathrm{T}} = \frac{2\eta P A}{a_0}. \tag{3.4}$$

For a required characteristic acceleration, equations (3.3) and (3.4) may now be used to size a solar sail, while imposing constraints on the total mass of the solar sail to satisfy the capacity of the launch vehicle. A design chart is shown in figure 3 for a characteristic accelerations of 0.25 mm s^{-2}, which is representative of the level of performance required for future planetary missions. Both the payload mass and the total launch mass are shown. It is clear that, for a large payload, typical of previous planetary missions, a large sail is required, with a sail side longer than 100 m. This requirement clearly poses challenges for the reliable mechanical deployment of large, low-mass structures, which will be discussed later. It should be noted, however, that one of the key drivers for solar sailing has been the trend towards dramatic reductions in the payload-mass requirements for deep-space missions. Future advances in payload miniaturization will allow highly capable deep-space missions with modest solar sails of side length 50–100 m, particularly if a sail-assembly loading as low as 5 g m^{-2} can be achieved.

4. Mission applications

Now that the principles of solar sailing have been discussed, a range of potential mission applications will be investigated. Before specific mission concepts are presented, the useful domain of operation of solar sails will be outlined. Traditionally, solar sailing has been seen as an efficient means of delivering science payloads to planetary or small Solar System bodies (McInnes *et al.* 2001*a*). However, as will be discussed later, solar sails can also be used to enable highly non-Keplerian orbits (NKOs). These new families of orbits are extensions to the classical two- and three-body problems of orbital mechanics. By exploiting the continuous low thrust available from a solar sail, exotic orbits can be found, which are displaced high above the plane of the Solar System (McInnes *et al.* 1994; McInnes 1998, 1999*a*) or orbits can be artificially precessed to track the motion of the Earth's magnetic tail (McInnes *et al.* 2001*b*).

(*a*) *Inner- and outer-Solar System missions*

Due to the enhanced light pressure available in the inner Solar System, solar sails can easily deliver payloads to close, polar orbits about the Sun for solar-physics mission applications. Payloads can be delivered by firstly spiralling inwards to a close, circular heliocentric orbit, the radius of which is limited by the thermal tolerance of the sail film (typically of the order of 0.2 AU). The orbit inclination of the solar sail is then cranked by turning the sail and alternately directing a component of the light-pressure force above and below the orbit plane every half orbit. For example, a solar sail with a characteristic acceleration of 1 mm s^{-2} can deliver a payload to a solar polar orbit at 0.5 AU with a mission duration of only 2.5 yr starting from an Earth-escape trajectory (Wright & Warmke 1976).

Other inner-Solar System missions, such the delivery of payloads to Mercury, offer quite spectacular opportunities. A ballistic transfer to Mercury using conventional chemical propulsion requires an extremely large Δv (accumulated change in velocity) of the order of 13 km s^{-1}, although this can be reduced using gravity assists at the expense of increased mission duration (*ca.* 5 yr for the NASA MESSENGER mission). A solar sail with a large payload mass fraction and a characteristic acceleration of 0.25 mm s^{-2} will deliver a payload to Mercury in 3.5 yr, while a solar sail with double the performance will require only 1.5 yr (Leipold *et al.* 1995). These inner-Solar System missions then make optimum use of solar sailing by using solar light pressure to enable extremely high-energy missions.

For payload delivery to Mars, outward spiral times tend to be somewhat longer than those for ballistic transfers. However, solar sailing is not constrained by the waiting period between ballistic launch opportunities. Again, for a characteristic acceleration of 1 mm s^{-2}, the trip time to Mars is of the order of 400 d, with an additional 100 d required for capture to an initial highly elliptical orbit and the subsequent inward spiral to a low planetary orbit. While solar sails can in principle deliver a larger payload mass fraction than that possible by chemical propulsion, one-way Mars missions do not make optimum use of solar sailing, since the required Δv is relatively modest.

Although one-way Mars missions do not appear attractive, two-way sample-return missions do provide opportunities. For a ballistic mission, the mass delivered to Mars must include propellant for the return leg of the trip. For a solar-sail mission, however, propellant is only required for a lander to descend to and ascend from the Martian surface. Therefore, solar sailing can be used to reduce launch mass, and hence mission costs, for such sample-return missions.

Other inner-Solar System missions for the more distant future include the use of large solar sails to reduce the total mass to low Earth orbit, and so total cost, required for the human exploration of Mars. A solar sail with a large payload mass fraction and low characteristic acceleration can deliver logistics supplies which are not time critical to Mars. For example, a large 2 km × 2 km solar sail of mass 19 200 kg can deliver a 32 000 kg payload to Mars orbit in 4.2 yr from an initial Earth parking orbit. The solar sail can then return to Earth in 2 yr to be loaded with another payload for delivery to Mars. This appears to be the largest solar sail which could be reasonably delivered to Earth orbit in a single launch using the Space Shuttle, or a large expendable vehicle such as Titan IV (McInnes *et al.* 2002).

Due to the diminished solar-radiation pressure in the outer Solar System, insertion of payloads into planetary orbit must be achieved using storable chemical propulsion, or aerobraking if appropriate. Payloads can be delivered to Jupiter and Saturn with minimum transfer times of 2.0 and 3.3 yr, respectively, using a solar sail with a characteristic acceleration of $1\ \mathrm{mm\,s^{-2}}$. After launch to an Earth-escape trajectory, the solar sail makes a loop through the inner Solar System to accelerate it onto a quasi-ballistic arc. A solar sail of similar performance can also be used to deliver payloads to Pluto in *ca.* 10 yr (Leipold & Wagner 1998).

Again, due to the potentially unlimited Δv capability of solar sails, multiple small body rendezvous missions are possible, as are small-body-sample returns. For example, a sample return from comet Encke can be achieved with a mission duration of the order of 5 yr, again using a sail with a characteristic acceleration of $1\ \mathrm{mm\,s^{-2}}$. An Encke mission is particularly difficult to achieve, since the comet has a high eccentricity, requiring significant energy for both the rendezvous and return phase. Also of interest is the possibility of a survey of multiple asteroids. This is a particularly attractive and cost-effective concept, since the mission is essentially open ended, allowing repeated science returns using the same suite of instruments. A solar sail with autonomous on-board navigation and planning software offers exciting possibilities for such missions.

For fast Solar System-escape missions, solar sailing offers significant performance gains over competing propulsion systems. For these missions, to the heliosphere (where the solar wind merges with interstellar space) at 100 AU and beyond, retro-propulsion is not required so that trajectories using a close solar pass can be used to accelerate payloads to extremely high cruise speeds. Using a high performance solar sail with a characteristic acceleration of the order of $3\ \mathrm{mm\,s^{-2}}$ and a close solar pass at 0.25 AU, cruise speeds of over 15 AU per year can be achieved resulting in relatively short trip times to the edge of interstellar space.

(*b*) *Mercury sample return*

Although it is a relatively near neighbour of Earth, Mercury is a largely unknown body due to the large energy required to transfer spacecraft to the inner Solar System. Acquiring and then returning a surface sample from Mercury is therefore one of the most demanding, high-energy Solar System missions that can be envisaged. A sample return is of particular interest, since Mercury appears to have frozen volatiles in its permanently shaded polar craters, deposited by cometary impacts early in the history of Solar System. The propulsive requirements for such a sample-return mission makes it an essentially impossible mission for conventional chemical propulsion, and an exceedingly difficult mission for solar-electric propulsion. Previous internal studies of a Mercury sample-return mission by the ESA defined a requirement for an Ariane V launch vehicle and a large solar-electric propulsion cruise stage to deliver a lander, ascent vehicle and Earth-return vehicle to Mercury. A combination of solar-electric propulsion and multiple gravity assists was required for the inward transfer, with chemical propulsion used for capture to a 500 km altitude polar parking orbit. A separate spin-stabilized Earth-return stage was envisaged using chemical propulsion and multiple gravity assists to deliver an entry capsule containing the surface samples to Earth. The launch mass was estimated to be of the order of 6600 kg, comprising 4577 kg for the solar-electric and chemical propulsion stages to transport a 2023 kg lander and spacecraft bus, with a total mission duration of 7.2 yr. However, studies at the University of Glasgow have shown that a Mercury sample-return mission can be enabled using a 190 m $\times$ 190 m solar sail with a launch mass of the order of 1200 kg, assuming a sail-assembly loading of 10 g m^{-2}. Such a mission architecture would allow the use of a significantly smaller, lower cost Souyz-ST launch vehicle and a somewhat shorter mission duration of 5.7 yr (Hughes & McInnes 2002*c*).

The scenario envisaged makes use of the solar sail to deliver a lander and ascent vehicle to an extremely low (125 km) altitude, near polar parking orbit close to the planetary day–night terminator. By targeting a low parking orbit, the propellant mass for the descent and ascent stages can be minimized, leading to significant reductions in launch mass. For a launch date of 1 January 2015, a 1210 d (3.31 yr) minimum-time trajectory is required to reach Mercury's sphere of influence on 25 April 2018, as shown in figure 4*a*. The trajectory optimization process uses a genetic algorithm to provide a starting solution for a sequential quadratic programming algorithm, which then minimizes the transfer time while enforcing the boundary conditions of the problem. The orbit capture and spiral to the 125 km parking orbit then requires 180 d, arriving at the parking orbit on 23 October 2018 (Macdonald & McInnes 2002). The arrival date is just after planetary aphelion (where Mercury is furthest from the Sun) so that the thermal loads on the lander are minimized during the subsequent short surface stay.

After the solar sail manoeuvres to the 125 km altitude near polar parking orbit, the stacked descent and ascent stage separates from the cruise stage and drifts relative to the solar sail. The stack is then slewed and an initial impulse of 104 m s^{-1} is effected to inject the stack onto an elliptical transfer orbit with a pericentre (closest approach) 20 km above the planetary surface. The pericentre is reached after a 1.49 h coast phase and the descent-stage propulsion is re-started to initiate a gravity-turn descent with an effective Δv of 3067 m s^{-1}. An additional Δv of 100 m s^{-1} is included in the descent-stage propellant budget to provide some margin for active manoeuvring and hazard avoidance prior to landing at the end of the main gravity-turn phase of the descent.

Since the parking-orbit plane has been selected to be close to the planetary day–night terminator, a relatively short stay (of 14 d) on the planetary surface is assumed to avoid the severe thermal loads associated with operations on the planetary dayside. During this phase of surface operations, sampling tools are assumed to acquire surface samples (*ca.* 250 g) which are then placed in the ascent stage. The return of the samples to the solar sail waiting in the parking orbit is a similar process to the descent, with a gravity-turn ascent, coast along a transfer ellipse followed by orbit circularization.

Following capture of the sample container at the cruise stage, the solar sail can perform an escape spiral to begin the return trip. After the 14 d period of surface operations and sample acquisition, the escape spiral begins on 6 November 2018 and uses a sail-steering law that maximizes the rate of gain of orbit energy (Macdonald & McInnes 2001). The solar sail reaches escape conditions on 1 December 2018 after a 26 d escape spiral. The escape spiral has a shorter duration than the capture spiral, since the sail mass has now decreased significantly (the lander has been left on the

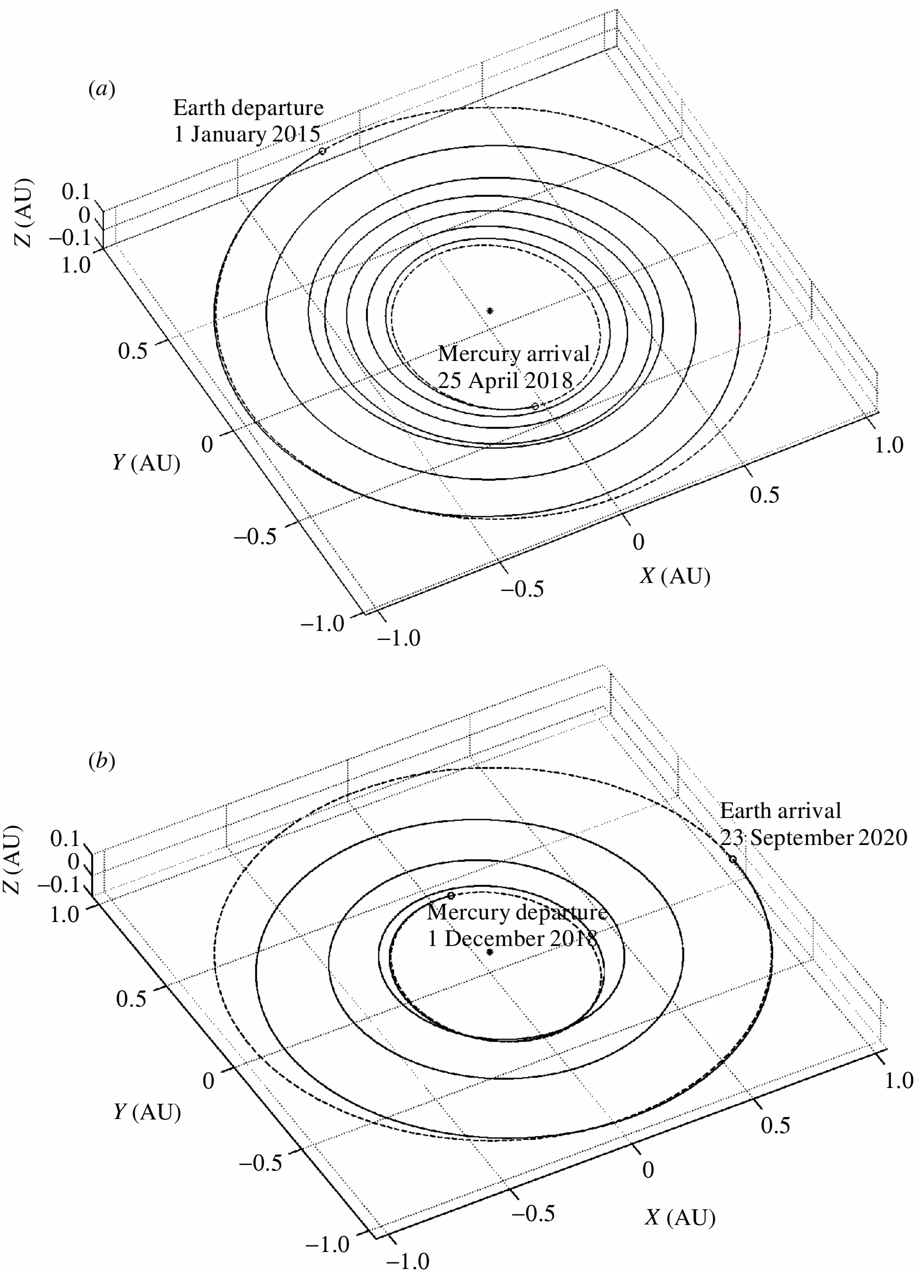

Figure 4. (*a*) Minimum-time Earth–Mercury trajectory.
(*b*) Minimum-time Mercury–Earth trajectory.

planetary surface and the ascent stage has been jettisoned), so the sail characteristic acceleration increases from 0.25 mm s^{-2} for capture to 0.44 mm s^{-2} for escape.

The departure date from Mercury, after the prescribed lander stay-time of 14 d and 26 d escape from the parking orbit is 1 December 2018. The trajectory optimization process is then found to generate a minimum-time trajectory for the return spiral of 662 d (1.81 yr), arriving at the Earth's sphere of influence on 23 September 2020, as shown in figure 4*b*. Again, the return trip is shorter than the initial inward spiral to Mercury, since the sail characteristic acceleration almost doubles after the lander and ascent stages have been discarded.

At the Earth's sphere of influence an atmospheric entry capsule spins-up, separates from the cruise stage and enters the Earth's atmosphere with a relatively modest entry speed of 11.1 km s^{-1} at an altitude of 120 km. The 45 kg capsule aerobrakes from the entry trajectory and soft-lands the samples at a pre-designated recovery site using a conventional parachute decelerator system. The total mission duration will have been 5.73 yr with an initial launch mass of only 1208 kg, corresponding to a spectacular reduction in launch mass of the order of 80% and a reduction in trip time of the order of 20%.

(*c*) *Small-body rendezvous*

In a similar manner to planetary sample-return missions, many asteroid and comet rendezvous missions pose immense energy requirements. For example, missions to rendezvous with more than three target objects make chemical or even solar-electric propulsion unattractive as candidate propulsion methods. Such multiple objective missions are attractive, since comparative science can be performed at a range of target bodies using the same set of calibrated instrumentation. For solar-sail propulsion, it appears that material lifetimes are the only limiting factors dictating the number and range of asteroids a solar sail can encounter throughout its mission lifetime.

Solar-electric propulsion has been adopted as the propulsion system for the NASA DAWN mission. This is a dual asteroid rendezvous mission scheduled for launch on 27 May 2006 aboard a Delta II launch vehicle with a total launch mass of 1110 kg. The objective of the mission is to rendezvous with the inner main-belt asteroids, Vesta and Ceres. The Dawn mission will take 4.2 yr to reach Vesta and, following 11 months in orbit at Vesta, reaches Ceres after a 3.1 yr transfer from Vesta. The solar-electric propulsion system will use three ion engines, processing 400 kg of xenon propellant via a large 7.5 kW solar array. The total mass of the spacecraft, without propulsion, is of the order of 350 kg.

During studies at the University of Glasgow, the mission has been reconfigured to use solar-sail propulsion for the same launch date of 27 May 2006 and with 11-month stay-times at each asteroid (Hughes & McInnes 2002*b*). A relatively high characteristic acceleration of 1.0 mm s^{-2} is required to provide similar trip times as the base-line DAWN mission, since the solar sail is operating beyond the Earth's orbit. Trajectory optimization shows that the Earth–Vesta phase requires 3.2 yr, while the Vesta–Ceres phase requires 3.7 yr, as shown in figure 5*a*. If a solar sail with an assembly loading of 5 g m^{-2} were available, the total launch mass for the mission could be reduced somewhat to *ca.* 980 kg. More importantly though, the mission objectives could be extended from Vesta and Ceres to further asteroids. For example, an extended tour from Ceres to the asteroid Lucina and then to Lutetia is shown in figure 5*b*, to further demonstrate the benefits of using a solar sail in a main-belt asteroid survey. Again, 11-month stay times are assumed at each target body.

In addition to asteroid missions, solar sailing is an attractive candidate for high-energy comet-rendezvous missions. Indeed, it was the discovery of a fast trajectory to

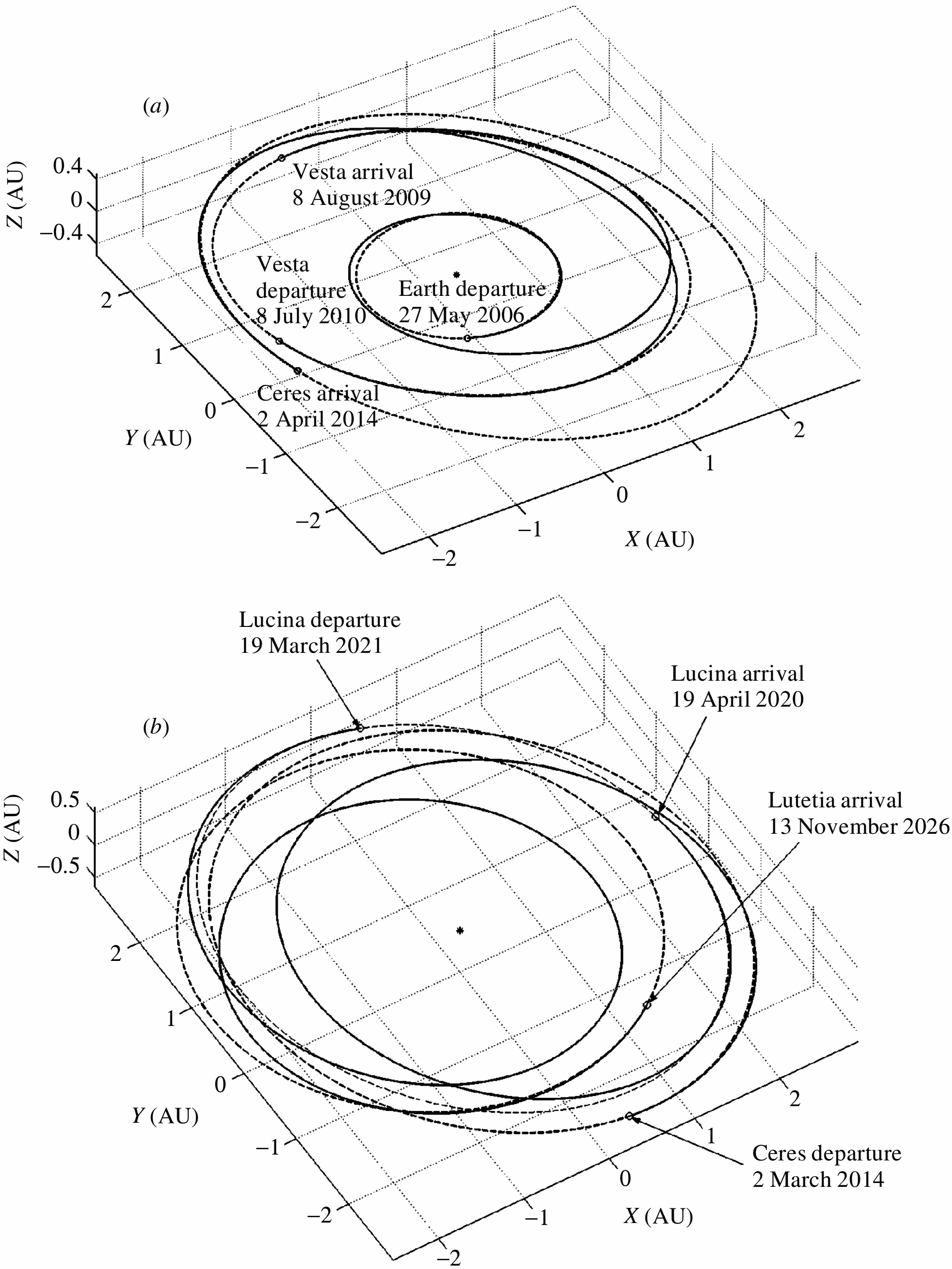

Figure 5. (a) Minimum-time Earth–Vesta–Ceres trajectory.
(b) Minimum-time Ceres–Lucina–Lutetia trajectory.

comet Halley which lead to the major solar-sail development effort at the NASA Jet Propulsion Laboratory in the mid 1970s. The ESA was set to launch the ROSETTA mission to comet 46P/Wirtanen in January 2003, however, the mission has now been postponed due to reliability issues associated with the launch vehicle. An Ariane V

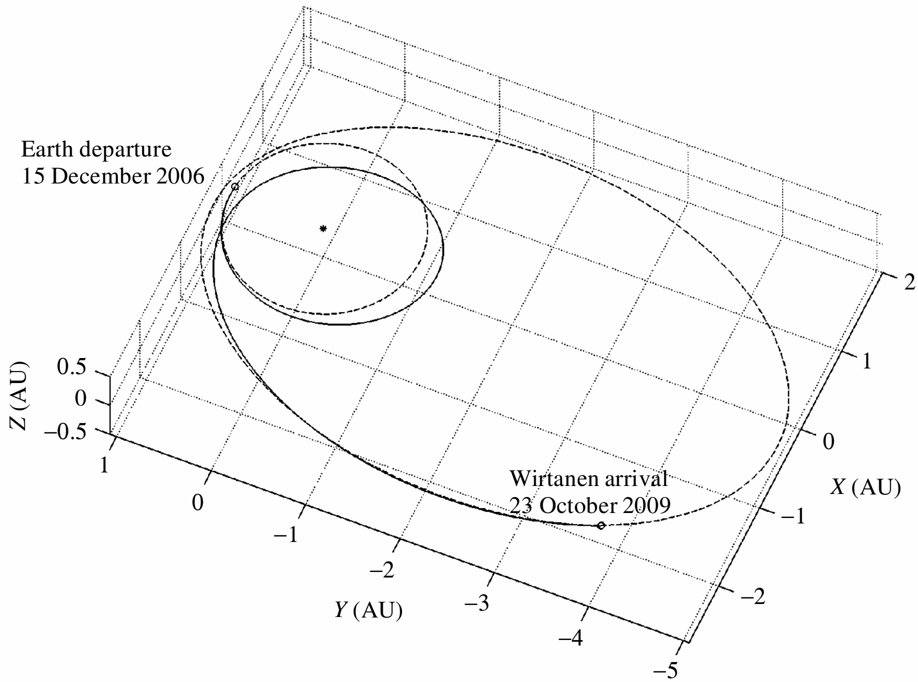

Figure 6. Minimum-time Earth–Wirtanen trajectory.

launch vehicle was to be used to deliver the 2950 kg spacecraft onto a complex 8.9 yr trajectory, which would use one Mars gravity assist and two Earth gravity assists to reach comet Wirtanen on 29 November 2011. The mission will now launch on 26 February 2004 to comet 67P/Churyumov–Gerasimenko.

Again in studies at the University of Glasgow, the mission has been reconfigured to use solar-sail propulsion (Hughes & McInnes 2002*b*). Since rendezvous with a comet on a highly elliptical orbit is a demanding task, a solar sail characteristic acceleration of $1\ \mathrm{mm\,s^{-2}}$ has been selected. The trajectory optimization process selects the optimum launch date as 15 December 2006, and provides a 2.9 yr trip time to comet Wirtanen, arriving on 23 October 2009, corresponding to a reduction in trip time of the order of 70% over the base-line ROSETTA mission, as shown in figure 6. The mass of the ROSETTA spacecraft without propellant and the propulsion subsystem is of the order of 880 kg. This relatively large mass is due to the array of scientific instruments and a surface lander which was to be delivered to comet Wirtanen. Again, if a solar sail with an assembly loading of $5\ \mathrm{g\,m^{-2}}$ were available, the launch mass could be reduced somewhat to *ca.* 2480 kg. More importantly though, the trip time for the mission is greatly reduced, again with the possibility of rendezvous with further targets objects at the end of the primary mission.

(*d*) *NKOs*

Due to the continually available thrust from solar light pressure, solar sails are capable of exotic, highly non-Keplerian orbits, so-called because they do not obey the usual Keplerian rules of orbital dynamics. Although some of these orbits require advanced, high-performance solar sails, others are possible using relatively modest

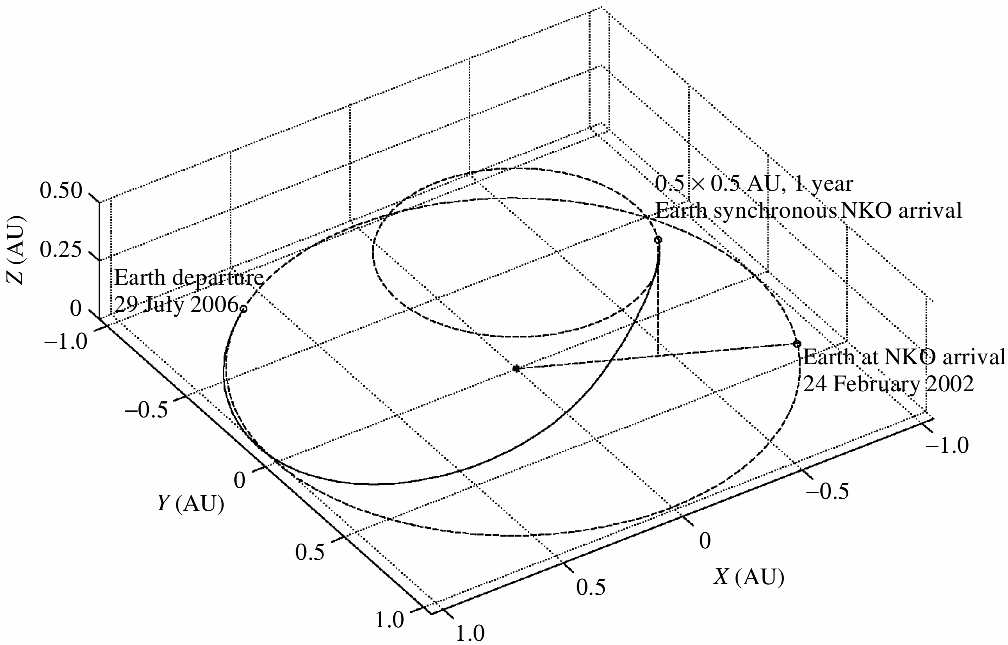

Figure 7. Minimum-time transfer to a displaced NKO.

solar sails. The solar-sail performance required for these orbits is a function of the local gravitational acceleration. Therefore, to displace an orbit high above the plane of the Solar System requires a characteristic acceleration of the order of 6 mm s^{-2} (McInnes 1998), while to generate an artificial Lagrange point near the Earth may only require a characteristic acceleration of the order of 0.25 mm s^{-2} or less (McInnes *et al.* 1994; McInnes 1999*a*). While these highly NKOs are not, in principle, forbidden for other forms of low-thrust propulsion, they can only be achieved for a limited duration, fixed by the propellant mass fraction of the spacecraft.

Firstly, using an advanced solar sail it would be possible to choose its characteristic acceleration so that the light pressure force exactly balances the local solar gravitational force. This is possible since both of these forces have an inverse square variation with heliocentric distance. The required characteristic acceleration for such a force balance is *ca.* 6 mm s^{-2}, corresponding to a mass per unit area of only 1.5 g m^{-2}. Such a high-performance solar sail would enable solar physics missions that could levitate above the solar poles, providing continuous observations, or indeed hovering at any particular location in the Solar System. Such a solar sail could also be used to displace circular heliocentric orbits high above the plane of the Solar System, with the orbit period chosen to be synchronous with the Earth or some other Solar System body (Hughes & McInnes 2002*a*), as shown in figure 7.

Using a more modest solar sail, the location of the Earth–Sun Lagrange balance points can be artificially displaced (McInnes *et al.* 1994; McInnes 1999*a*). The Lagrange points are locations where a conventional spacecraft will remain in equilibrium with respect to the Earth and the Sun. For example, the interior L_1 point, 1.5×10^6 km sunward of the Earth is a favoured location for solar physics missions. Since the solar sail adds an extra force to the dynamics of the problem, the location

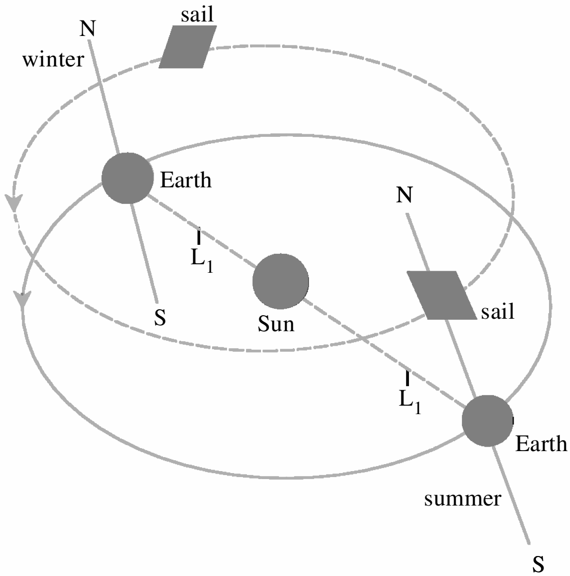

Figure 8. Polar stationary orbit concept.

of the L_1 point can be artificially displaced, closer to the Sun or even above the plane of the Earth's orbit. Since the local gravitational acceleration in the vicinity of L_1 is small (since solar and Earth gravity almost balance), only modest solar sails are required. For example, a solar sail with a characteristic acceleration of $0.25\ \mathrm{mm\ s^{-2}}$ can double the distance of the L_1 point from Earth. Such a new sunward equilibrium location appears useful for providing early warning of disruptive solar plasma storms before they reach Earth, and indeed formed the basis for the NASA/NOAA Geostorm mission concept (West & Derbes 2000). A solar sail with double the performance can be permanently stationed high above the classical L_1 point, so that it appears above the Arctic or Antarctic regions of the Earth, as discussed below.

Geostationary orbit provides a convenient location for telecommunications satellites, providing a fixed line of sight from the satellite to ground terminals. Being located high above a fixed point on the Equator, geostationary orbit also provides an ideal vantage point for Earth observation satellites, providing coverage of large geographical regions. While the advantages of geostationary orbit for telecommunications and Earth observation are clear, there are operational limitations. Due to their location over the Equator, geostationary satellites do not have a good vantage point from which to view high-latitude regions. Imaging of high-latitude regions is degraded by foreshortening, while the poles are entirely excluded from view. Likewise, communication satellites are extremely difficult to view for users at high latitudes due to their close proximity to the horizon, and indeed are below the horizon for latitudes above $\pm 81^\circ$ (Forward 1991).

As discussed above, solar sails may be used to generate artificial equilibrium points close to the Earth. Out-of-plane equilibria, with the solar sail hovering above the Earth's orbit plane, may be used for continuous low-resolution imaging of the high-latitude regions of the Earth. In fact, if the artificial Lagrange point is located high enough above the Earth's orbital plane, the solar sail may be stationed directly over the North Pole, or indeed the South Pole, during the summer solstice, as shown in figure 8. It is found that the required solar-sail performance can be minimized by an appropriate selection of polar altitude and that equilibrium location some 3.8×10^6 km (around 600 Earth radii) above the pole will minimize demands on the solar-sail performance. Closer equilibrium locations are possible using larger, or higher-performance, solar sails, or indeed selecting a less demanding viewing geometry. To station a small 50 kg payload at this unique polar view point requires an 86 m × 86 m sail, assuming a sail-assembly loading of 10 g m^{-2}. Although the distance of the solar sail from the Earth is large for imaging purposes, there are potential applications for real-time, low-resolution images for continuous views of large-scale polar weather systems along with polar ice and cloud coverage for global climate studies. Other applications include line-of-sight, low-data-rate communications to high-latitude military and civilian users. These applications are currently being evaluated by the US National Atmospheric and Oceanic Administration for both Arctic and Antarctic applications.

5. Engineering challenges

While the opportunities presented by solar sailing appear extremely attractive for future space-science missions, challenges are posed in the fields of deployable structures, thin films and active control. In addition to these key technologies, there are secondary issues associated with systems integration which must be identified and solved. Issues which will affect the science payload include occultation of instrumentation by the sail, stray light reflected by the sail into instrument optical paths, vibration of the sail booms and electrical charging of the sail by the solar wind plasma. Perhaps the most important of these integration issues is charging of the sail. The sail film is composed of two electrically conducting surfaces separated by a polyimide plastic substrate and is therefore extremely susceptible to differential charging, and potentially arcing between the sail surfaces. This problem can in principle be addressed by providing suitable conducting paths between the sail surfaces, for example, by perforating the sail substrate at intervals before the coatings are deposited and ensuring electrical contact between bonded elements of the sail film. One of the most likely future applications for solar sails is to deliver field and particle instruments to high-energy orbits for space and solar-physics missions. Therefore, if the sail acquires a significant charge it will disrupt the plasma environment in the vicinity of the sail, invalidating the science data acquired by the science payload.

There is extensive in-orbit experience of deployable structures for applications such as solar arrays, antennae and experiment booms. However, as yet none of these applications has been on a scale suitable for solar sails. A popular deployable structure with potential application to solar sailing is the storable tubular expandable member (STEM) structure. The STEM structure is a metallic or carbon-fibre tube that can be pressed flat and rolled onto a spool for packing. As discussed in § 2, the structure can be deployed using the elastic energy stored in the pre-stressed flattened tube, or using a small drive motor for a more controlled deployment. By using carbon-fibre with layers built up in alternate directions, deployable booms can be manufactured with essentially zero coefficient of thermal expansion. Once deployed, the drive motor and associated hardware can be jettisoned in order to reduce to the total mass of the solar sail. Another popular deployable structure which has been used for many space applications is the continuous-longeron coilable boom (CLCB). The CLCB is a linear truss structure with a triangular cross-section. Repeated triangular elements are joined to three continuous longerons to form the truss, which is given a helical twist and packed into a storage drum. Pre-tensioned diagonal elements store enough elastic energy to allow self-deployment, although a lanyard cable attached to the end of the truss can be used to control the deployment rate. The CLCB typically has a stowed length of less than 1% of its final deployed length and is clearly scalable from current applications to solar sailing.

In addition to deployable mechanical structures, inflatables provide an attractive means of reliable deployment. Inflatable structures have long been considered for solar concentrators, antenna reflectors and truss structures. The main benefit of inflatables is the ease and reliability of deployment with few failure modes. The structure consists of a thin membrane that is deployed solely by internal gas pressure, typically using stored nitrogen- or warm-gas generators (such as sodium azide, used in automobile air bags). Once the structure begins to deploy, the internal gas pressure ensures full deployment and initial rigidization. Additional gas can be provided to account for leakage through seams or micrometeorite penetration, although the requirement for such make-up gas varies as the square of the mission duration. As a lower mass alternative, rigidization can be provided by a space curing resin enclosed between two films in the inflatable membrane. Following deployment, the resin hardens once it is heated by solar radiation. Other rigidization methods rely on fabric membranes impregnated with gelatine. On exposure to vacuum, a water-based solvent is released, thus rigidizing the membrane. Clearly, the time-scales for deployment and rigidization must be carefully characterized in order to avoid rigidization prior to full deployment of the structure.

A novel concept which may see development in the future is to integrate smart materials, such as Nitinol wires, within the booms during manufacture. This would allow the boom profile to be altered by passing a small electric current through these actuators. By warping individual booms, the centre of pressure of the solar sail can then be displaced, generating attitude-control torques. This has major advantages for solar sailing, since high-mass actuators, such as gimballing the payload to displace the sail centre of mass, are not required. Other concepts involve integrating optical fibres within the booms to measure strain and hence boom deflection. The sensing functions would then form a local closed loop with the actuators, so that the sail would become a massively distributed smart structure which can autonomously orient itself relative to the Sun.

The most popular material proposed for the fabrication of sail film is Kapton (DuPont, Wilmington, DE, USA), due to its high tensile strength, thermal stability, good solar-UV resistance and ease of handling. For example, Kapton can be easily coated with aluminium or other metals through vapour deposition, and stored on rollers. Although Kapton is attractive due to these properties, the commercially available film is rather too thick for future solar-sail applications. Kapton is currently available in thicknesses of 7.5 μm, whereas future solar-sail requirements are for 1–2 μm film, although 7.5 μm Kapton is suitable for demonstration missions and some other near-term applications. While 7.5 μm Kapton can be chemically or plasma etched, custom thin films will be required for future solar-sail applications. DuPont has no commercial customer for thinner Kapton and so is unable to re-tool purely for solar-sail applications without a significant cost overhead. As an alternative to Kapton, CP1 film is now available from SRS Technologies (Newport Beach, CA, USA), who have developed the film for solar-sail applications under licence to NASA. The film has similar mechanical properties to Kapton, but is available in a significantly thinner gauge, down to 1 μm.

6. Conclusions

While solar sailing offers significant advantages for Solar System exploration, it is not a generic propulsion system which can meet the requirements of every conceivable mission. Neither is it a technology worth developing for its own sake. While the thought of sailing through the Solar System using nothing more than the pressure of sunlight is a romantic notion, solar sailing must be pulled forward by novel mission applications at the same time as it is pushed by technology development. Proponents of solar sailing must therefore be practical, hard-headed romantics. Since the NASA studies of the mid 1970s, technology has matured considerably. In particular, the opportunities provided by spacecraft miniaturization are immense, allowing highly capable missions with only relatively small solar sails. Similarly, there is now a range of extremely promising concepts for the large deployable structures required for future solar-sail missions and for ultra-thin sail films.

This paper has indicated some missions where solar sailing is used to its optimum advantage. High-energy and/or long-duration missions are the key to solar sailing where it can be used most efficiently. For this reason the possibilities for the long-term development of solar sailing rely on the investigation of those niche missions and applications which are unique to solar sails. By presenting the space-science community with exciting new opportunities, the demand will be created for solar sailing, which will then lead to resources flowing towards the development and integration of the technologies required to bring the concept to fruition. At present, there is a growing awareness of these benefits in the space-science community, particularly in the US, but also in Europe. Where this wave of interest now leads, remains to be seen.

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