Solar Sailing – Mission Opportunities and Innovative Technology Demonstration

C.R. McInnes
Department of Aerospace Engineering, University of Glasgow, Glasgow, United Kingdom

M. Eiden
Mechanical Systems Division, ESA Directorate of Technical and Operational Support, ESTEC, Noordwijk, The Netherlands

P. Groepper
ESA Directorate of Industrial Matters and Technology Programmes, ESTEC, Noordwijk, The Netherlands

T. Peacock
Science Payloads & Technology Division, ESA Directorate of Scientific Programmes, ESTEC, Noordwijk, The Netherlands

Solar sailing is a unique and elegant form of propulsion that transcends reliance on reaction mass. Rather than carrying propellant, solar sails acquire momentum from photons, the quantum packets of energy from which sunlight is composed. In addition, since solar sails are not limited by reaction mass, they can provide continual acceleration, limited only by the lifetime of the sail film in the space environment. Therefore, solar sails can expand the envelope of possible missions, enabling new high-energy mission concepts that are essentially impossible with conventional reaction propulsion, and enhancing current mission concepts by lowering launch mass and reducing trip times.

Figure 1. The 20 m x 20 m sail deployment test in December 1999 (DLR)

Introduction

Solar-sail technology was developed to some extent by NASA/JPL during the mid-1970s for a proposed rendezvous mission with Comet Halley. Although not attaining flight-readiness, the study sparked international interest in solar sailing for future mission applications. More recently, due to advances in payload miniaturisation and a recognition of the need for high-energy propulsion for demanding future missions, NASA is again aggressively pursuing the development of solar-sail technologies. A strong interest in solar sailing is also emerging in Europe, supported by a successful joint ESA-DLR ground deployment test of a 20 m x 20 m solar sail (Fig. 1) and a series of ESA-funded mission studies at the University of Glasgow (UK).

Since the momentum transported by an individual photon is extremely small, solar sails require a large surface area in order to intercept a large flux of photons. Furthermore, to generate as high an acceleration as possible from the momentum transported by these intercepted photons, solar sails must also be extremely light. For a future solar sail, the mass per unit area of the entire spacecraft, the so-called sail loading, may be of order 20–30 g/m^2. In addition to the sail loading, the sail assembly loading is a key measure of technology. This parameter is defined as the ratio between the mass of the sail structure plus reflective film (excluding the payload and bus), and the sail area. A goal for mid-term solar sails for large planetary or space-physics missions would be an assembly loading of order 10 g/m^2, although near-term missions would be less demanding.

Not only must solar sails have a small mass per unit area, they must also be near-perfect reflectors. Then, the momentum transferred to the sail can be almost double the momentum transported by the incident photons. By adding the forces due to incident and reflected photons, the total force exerted on the sail is directed almost normal to its surface. By controlling the orientation of the sail relative to the Sun-line, the sail can gain or loose orbital angular momentum. In this way the solar sail is able to spiral outwards or indeed inwards along the Sun-line. Achieving a useful characteristic acceleration (acceleration at 1 AU) of order 0.1 – 1 mm/s^2 from solar radiation pressure poses great engineering challenges in terms of innovative technology demonstration of low-mass deployable structures, thin-film sails, and also importantly payload miniatur-0isation.

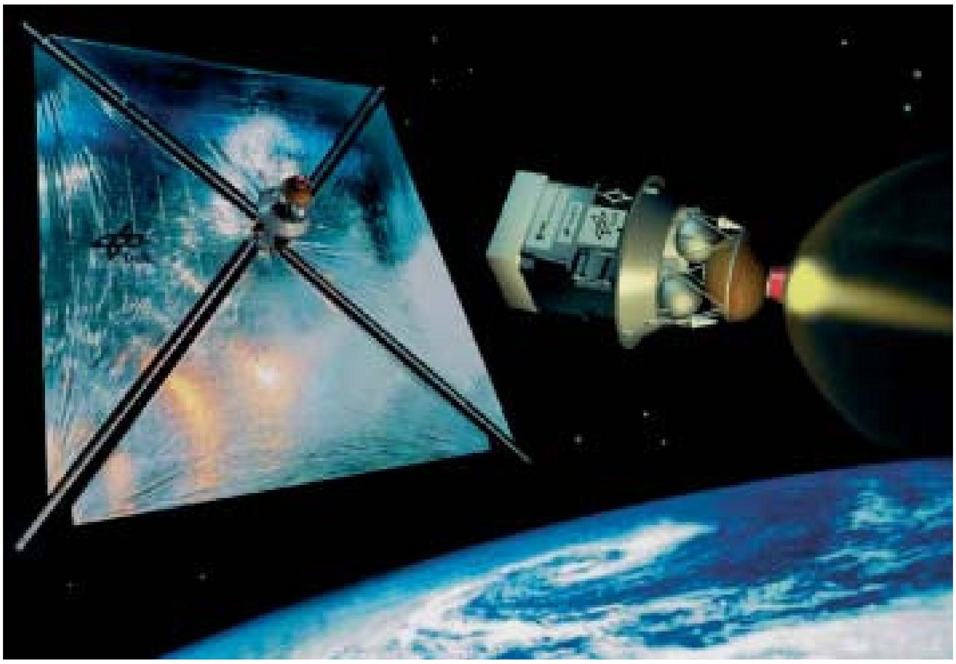


Figure 2. Solar-sail in-orbit deployment demonstration

Innovative technology demonstration: the project challenge

While the major technology challenge of entering the field of solar sailing is to build extremely light, large-area deployable structures and to combine them with micro-spacecraft for carrying out science exploration missions throughout our Solar System and beyond, the programmatic challenge is to cleverly subdivide the big leap in technology needed into technically and programmatically manageable project steps. Mass-efficient space hardware of many kinds is well known, but a vehicle with a mass per area ratio of only a fraction of that of writing paper is something unusual and a new challenge in itself, and one beyond the usual approach taken in spacecraft engineering.

In order to carry out a first and vital European flight-hardware step in the direction of developing solar sails, ESA's Industrial Policy Committee (IPC) at its March 2001 meeting approved a procurement proposal to develop and launch a solar-sail in-orbit deployment demonstrator. This deployment demonstrator, as the first step in solar-sailing validation, is concentrating on the successful functioning of sail deployment, which is compromised on the ground by the 1g environment and where in-orbit anomalies have been experienced with a number of large deployable structures.

The demonstrator has booms sized for the deployment of a 40 m x 40 m sail, but considering the low-cost approach adopted for this demonstration mission, the deployed sails will be only 20 m x 20 m. The design of the deployment module and the envisaged material for the 40 m x 40 m sail target an overall assembly loading in the order of 35 g/m^2. Further reductions appear feasible in view of the advanced sail films under study in the USA. In its stowed launch configuration, the demonstrator will be only 60 x 60 x 80 cm^3. Within this box envelope there is room to accommodate a future miniaturised spacecraft also. Four coiled booms, each about 14 m long, and four triangularly shaped sails will be deployed consecutively and form a 20 m x 20 m flat square sail.

The profile and scope of the mission are fully geared to the demonstration of the in-orbit deployment (Fig. 2). All elements of this technology project, including mission definition, hardware and software development and manufacturing, launch and in-orbit operations will be focused towards this single goal. Given the limited budgetary resources available, the project's implementation from the ground demonstration model to the flight hardware will be challenging. A successful solar-sail deployment demonstration will imply that important challenges associated with designing, building and operating a large, complex, multifunctional lightweight mechanism and the associated sails have been achieved. Mastering this first important technological step in European solar sailing will therefore establish the required confidence in solar sailing as a viable technology for the promising mission scenarios described below.

Potential applications for science missions

When we examine the potential benefits that solar-sailing technology may provide for future science missions, several promising future mission scenarios for the exploration of our Solar System and beyond can already be identified.

A mission to study the Earth's magnetosphere – Geosail

While some planetary exploration missions will require rather large solar sails, a number of science missions closer to home have been identified that require only modest sail areas. This incremental approach therefore builds on the sail deployment demonstration mission described above and allows the technology to develop in a mission-focussed manner. The Geosail example is therefore described here not only as a demonstration of solar-sail technology beyond that of simple in-orbit deployment, but also of the ability of such sailing techniques to enable totally new science mission concepts to be undertaken.

Conventional geomagnetic-tail missions require a spacecraft to be injected into a long elliptical orbit to explore the length of the geomagnetic tail. However, since the orbit is inertially fixed, and the geomagnetic tail points along the Sun–Earth line, the apse line of the orbit is precisely aligned with the geomagnetic tail only once per year. Approximately 4 months of data can be acquired when the spacecraft is in the vicinity of the tail, but only about 1 month of accurate data when on the tail axis itself. Artificially precessing the apse line of the elliptical orbit to keep the spacecraft in the geomagnetic tail during the entire year would simply be prohibitively demanding using chemical propulsion. In a scientifically meaningful elliptical orbit of 10 x 30 Earth radii, for example, a Δv of order 3.2 km/s per year of operation would be required for apse-line rotation.

Although the total Δv for apse-line rotation is large, only a small acceleration continuously directed along the apse line is needed in principle to achieve the necessary orbit change. Calculations and simulations have shown that a continuous acceleration of only 0.14 mm/s^2 is required for such an orbit. Since the precession rate of the orbit's apse line is chosen to match that of the Sun-line, the sail normal can be directed along the latter. This has significant operational advantages, since such a Sun-facing attitude can be achieved passively. The evolution of the mission orbit over 50 days is shown in Figure 3.

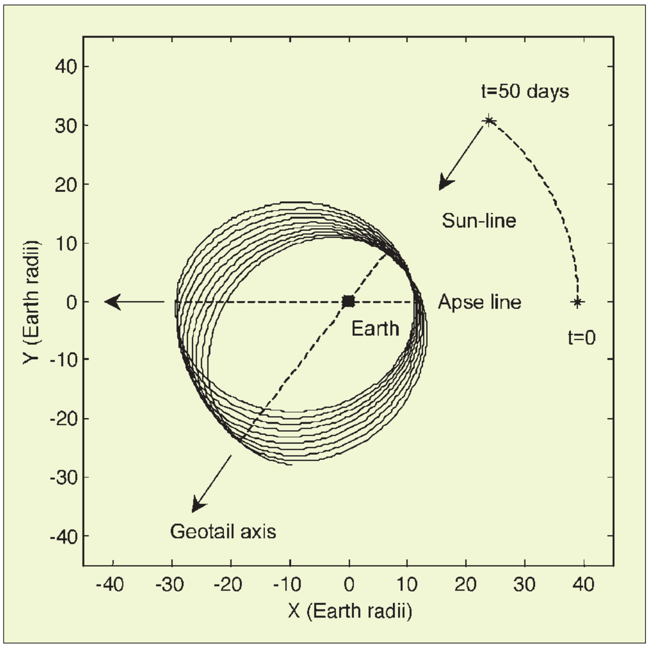


Figure 3. A 10 x 30 Earth radii orbit rotating with the Sun–Earth line

Table 1. Mass budgets for the Geosail mission

SAIL	(kg)
Booms (100 g/m)	11
Sail film (7.5 µm)	15
Coatings (Al + Cr)	0.5
Bonding	2.5
Mechanisms	20
SAIL TOTAL	49
PAYLOAD	
Instruments	5
Bus	25
Adapter	1
PAYLOAD TOTAL	31
LAUNCH MASS	80

Having established the acceleration requirements, it is now possible to size the sail. First we have assumed a micro-satellite of about 30 kg mass for the bus and payload, which could be appropriate for a low-mass space-physics payload with a magnetometer and plasma instruments. The instrument mass is assumed to be of order 5 kg with a 26 kg spacecraft bus (i.e. the instruments are about 20% of the spacecraft dry mass; cf. Table 1). This is far less than the mass of a single ESA Cluster spacecraft, which is around 1200 kg: 72 kg payload, 480 kg spacecraft bus and 650 kg propellant necessary to reach and then modify the orbit during the mission. Thus, in the case of Cluster, the payload-to-spacecraft mass ratio is similar to that of the micro-satellite. One can, however, now identify the necessary parallel development of low-mass technology solutions at spacecraft and instrument level. With this caveat in mind, we have sized the sail at 38 m x 38 m so as to generate a characteristic acceleration of 0.14 mm/s^2. This is therefore a modest evolution over the 20 m x 20 m demonstration sail. The booms are assumed to be of CFRP with a specific mass of 100 g/m, while the sail film is assumed to be commercially available 7.5 micron thick kapton, vapour-coated with aluminium on one side and chromium on the other for thermal control. The total launch mass is about 80 kg, which also falls within the mass budget of an Ariane-5 ASAP auxiliary payload.

The spacecraft can be delivered to the 10 x 30 Earth radii orbit by simply using the solar sail itself to spiral from geostationary transfer orbit (GTO). For a standard Ariane-5 midnight launch, the apse line of the orbit is directed sunward, opposite to the geomagnetic tail. A six-month (or 18-month) orbit-raising phase is therefore required while the solar sail manoeuvres to the required operational orbit, and the apse line aligns with the geomagnetic tail. During this time, the orbit plane must also be rotated so that the final orbit lies in the ecliptic plane. Long-term orbital integration studies of the solar sail in a 10 x 30 Earth radii orbit, which includes a geopotential model as well as the lunar-solar gravitational perturbations, has demonstrated the validity of precessing the operational orbit.

Alternative launch scenarios can be considered which, after its delivery into GTO, would inject the sail and spacecraft-bus package directly into the operational 10 x 30 Earth radii orbit using a dedicated launcher or a piggy-back launch with a chemical kick-stage. With this scenario, if the sail deployment were to fail, the spacecraft bus could be separated to perform a conventional, albeit scientifically degraded geomagnetic-tail mission without apse-line precession, although the launch costs would of course be increased. The risk of using solar-sail technology on such a first science mission would, however, be significantly reduced.

The above mission concept is extremely attractive since it provides a scientifically useful application for a solar-sail technology-demonstration mission.

A mission to study a planet in our Solar System – Mercury-Sail

BepiColombo is an ambitious ESA Cornerstone science mission to explore Mercury comprehensively using a diverse array of instruments. It requires solar-electric propulsion, chemical propulsion and multiple gravity assists to deliver a large payload with a single Ariane-5 launch. The payload consists of a 360 kg planetary orbiter, a 165 kg magnetospheric orbiter and a 44 kg hard lander. The three-axis-stabilised orbiter will perform global imaging of the planetary surface using visible and infrared cameras, while a range of spectrometers will determine surface composition. These remote mapping functions will be complimented by a radio-science payload. The smaller spin-stabilised magnetospheric orbiter will investigate the interaction of Mercury's magnetic field with the solar wind from a long 400 km x 12 000 km elliptical orbit using a three-axis magnetometer and a range of plasma instruments. Finally, the hard lander will deliver a small rover plus imager and science package to explore the physical properties of the planetary surface.

The mission profile required to deliver such a large payload centres on the use of solar-electric propulsion (SEP) and multiple gravity assists to reach Mercury, while orbit capture is performed using a chemical bi-propellant stage. Launch opportunities occur every 1.6 years with a 2.2–2.6 year trip time for the SEP option with a Venus swingby, depending on the particular launch window selected. The total launch mass is some 2500–2800 kg, which could be delivered with a hyperbolic excess speed of order 3 km/s by an Ariane-5 launcher, or with two Soyuz-Fregat launchers, which would launch the MPO (1255 kg) and MMO+MSE (1265 kg) separately.

We can use this BepiColombo mission as a reference to assess the potential benefits of solar sails for future planetary exploration missions. It has therefore been reconfigured for delivery by solar-sail propulsion alone. To allow for a realistic comparison, the solar-sail solution must provide a comparable trip time to the baseline SEP mission without lunar swingby, i.e. 2.5 years. It was found that a characteristic acceleration of 0.25 mm/s^2 is adequate for the Mercury orbiter mission, but a figure of 0.3 mm/s^2 has been selected to provide some margin. A typical 2.4 year trajectory to Mercury is shown in Figure 4. The trajectory begins with a launch energy C_3=0 and therefore does not require any hyperbolic excess to be delivered by the launch vehicle. In addition, the launch window for such a solar sail mission is in principle unconstrained because gravity assists are not required.

Assuming the chosen acceleration of 0.3 mm/s^2 and a total spacecraft mass (bus + payload) of 590 kg (planetary orbiter, magnetospheric

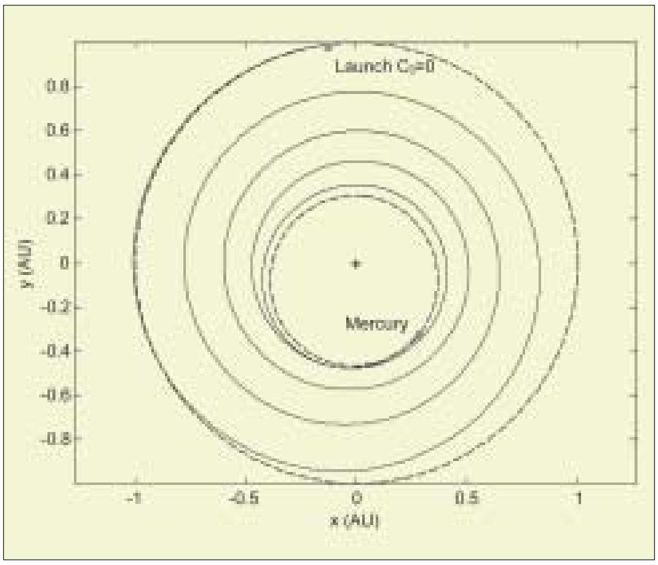


Figure 4. A 2.38 year trajectory to Mercury , with a characteristic acceleration of 0.3 mm/s²

orbiter and lander), the sail size needed and the mission launch mass can be determined. First, the sail assembly loading, which is a function of the level of solar-sail technology available, must be established. A representative value for mid-term missions is likely to be of order 10 g/m². To achieve this, a sail substrate thinner than the commercially available 7.5 micron kapton proposed for the initial demonstration missions is required. A 3 micron substrate, similar to that fabricated by NASA/JPL for a range of near- and mid-term solar-sail applications, will therefore be assumed, with a front aluminium and rear chromium coating and a 10% mass penalty for bonding the sail segments. In addition, CFRP booms with a specific mass of 150 g/m (some 50% heavier than those used for the ESA–DLR ground deployment test) have been assumed. For such large sails, film mass is dominant and the sizing is relatively insensitive to boom properties in that any increase in boom mass can be accommodated via a modest reduction in sail-film thickness. The overall mass of the sail assembly, comprising the coated sail film, the deployable booms and the associated mechanisms turns out to be around 319 kg.

A mass-breakdown comparison for the solar-sail and SEP powered BepiColombo missions is shown in Table 2. The solar-sail option offers significant advantages in that the total mission launch mass is reduced to 872 kg, below the C_3=0 capacity of a Soyuz/Fregat launcher and the mission payload mass fraction is significantly improved, from 0.24 to 0.63.

Table 2. Mass budgets for the Mercury-Sail and Mercury-SEP missions

SAIL	(kg)	SEP	(kg)
Booms (150 g/m)	76	Ion stage	675
Sail film (3 µm)	136	Chemical stage	140
Coatings (Al + Cr)	12	Xenon fuel	433
Bonding	19	Bi-propellant fuel	332
Mechanisms	76	Margin	139
SAIL TOTAL	319	SEP TOTAL	1719
PAYLOAD		PAYLOAD	
Planetary orbiter	463	Planetary orbiter	463
Magnetospheric satellite	59	Magnetospheric satellite	59
Lander	31	Lander	31
PAYLOAD TOTAL	553	PAYLOAD TOTAL	553
LAUNCH MASS (C_3=0)	872	LAUNCH MASS (C_3>0)	2272

While this example really highlights the benefits of the solar-sail approach for such deep-space scientific exploration missions, the extended and flexible mission scenarios that it allows can provide additional scientific return. For example, in the case of the Mercury mission it would allow several additional follow-on ‘end-of-life’ applications. If, for instance, the mission profile were modified such that the planetary orbiter and lander are jettisoned at Mercury, but the magnetospheric orbiter remains attached to the sail, the magnetospheric orbiter can subsequently be used for other secondary exploration missions. After the planetary orbiter and lander are jettisoned, the characteristic acceleration of the solar sail increases significantly to 0.69 mm/s².

From the Mercury magnetospheric orbit, the solar sail and magnetospheric orbiter could spiral in to a close orbit around the Sun and use the onboard suite of field and particle instruments to investigate the plasma environment near the solar co-rotation region. This co-rotation region occurs at an orbital radius of 0.172 AU, where the circular orbit period of 26 days is equal to the solar equatorial rotation period. With the increased solar-sail characteristic acceleration, the trip time to a 0.2 AU circular orbit would be only 80 days. Clearly, the sail would experience significant thermal loads, but such end-of-life mission applications, attempted only after the primary mission is complete, add little risk and are clearly highly cost-effective. A dual mission, in which delivery of a Mercury orbiter and a close solar orbiter are part of the primary mission, would clearly have an impact on the engineering of the sail and payload. One can even consider other secondary mission scenarios in which the spacecraft in its circular 0.2 AU orbit initiates a new manoeuvre. Essentially the sail can crank up its orbital inclination to a solar polar orbit in a further 290 days.

Finally, an alternative end-of-life mission application is to spiral from Mercury and return to Earth orbit in only 1.2 years to demonstrate the round-trip capabilities of solar sailing for future sample-return missions.

A mission to study the Sun's inner heliosphere – Solo-Sail

The Solar Orbiter mission Solo is a ESA F-class (flexible) mission to view the Sun from a close solar orbit (0.2 AU) and from out of the ecliptic plane. To reach a high-energy orbit in order to meet these demanding mission objectives, a combination of solar-electric propulsion and multiple gravity assists is required. The mission requires a single Soyuz-Fregat launch to an Earth-escape trajectory with a hyperbolic excess of 2.4 km/s. The spacecraft then uses SEP to reduce its orbit's semi-major axis and target multiple Venus gravity assists to decrease the perihelion radius and increase the orbital inclination. A total flight time of 1.9 years is required to reach an initial elliptical science orbit, with a perihelion radius of 0.21 AU and an aphelion radius of 0.89 AU. Thereafter, additional gravity assists are used to crank the orbital inclination to 30 deg after an additional 2.9 years, and possibly to 38 deg for an extended mission requiring a further 2.3 years of flight time. The spacecraft will deliver an X-ray and EUV imager, along with a range of field and particle instruments, for a total mission launch mass of 1510 kg. Approximately 634 kg of this can be attributed to the spacecraft bus and science payload.

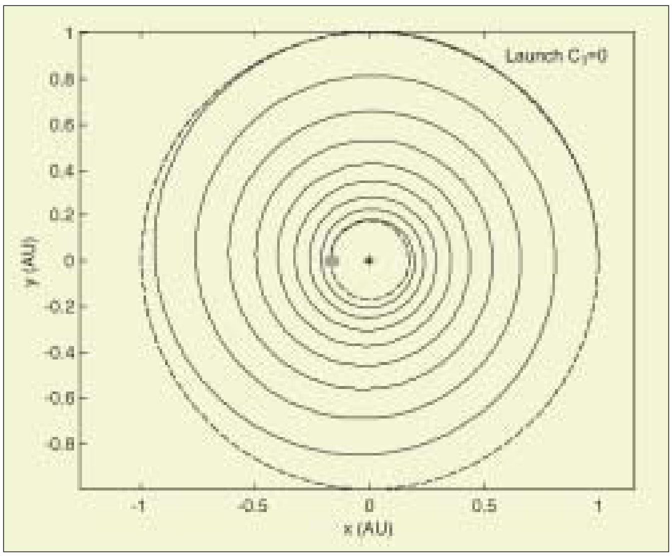
Again as an example of the advantages of solar sailing for future missions, we consider the Solar Orbiter mission profile in the context of using solar-sail propulsion to transport a payload to a close circular orbit, deep within the Sun's gravity well. This is a mission that is essentially impossible with chemical propulsion and would involve an extremely large launch mass with a solar-electric propulsion system. The ideal mission orbit is at a heliocentric distance of 0.172 AU, where the local circular orbit period is 26 days, synchronous with solar equatorial rotation. The primary science goals of this sailing mission are the same as those of the current Solar Orbiter mission to ensure a reasonable comparison. In the case of the solar-sailing mission, the closer circular orbit provides a better vantage point than the elliptical baseline orbit. In particular, the unique 26-day solar synchronous orbit allows continuous observations of particle acceleration above active regions. Although the primary science goals are achieved once on station in the 26-day orbit, useful cruise science can begin from the start of the mission during the inward spiralling phase.

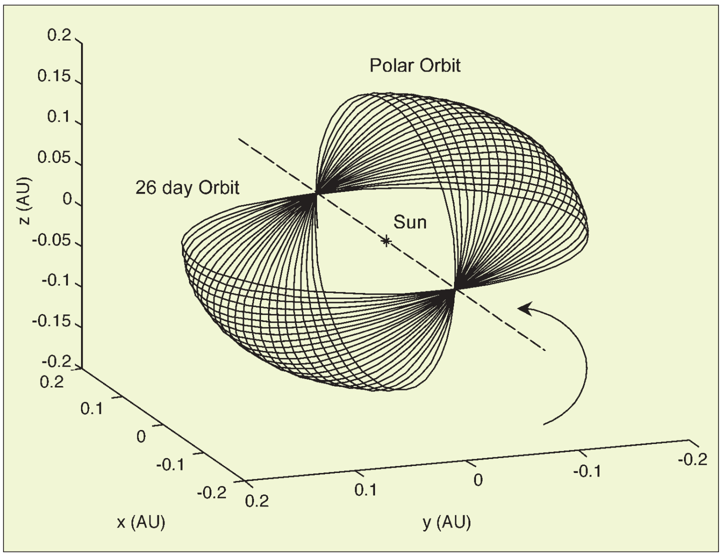
The proposed sailing mission concept assumes that the stowed solar sail is delivered by the launch vehicle to an Earth-escape trajectory and spirals inwards to the mission orbit using an optimal, minimum-time trajectory. To provide a reasonable transfer time to the initial mission orbit, a sail characteristic acceleration of 0.25 mm/s^2 is selected. This provides a 3.1 year transfer to 0.172 AU, as shown in Figure 5. From this initial orbit, the solar sail can rapidly crank the orbital inclination to a solar-polar orbit after an additional 1.7 years, as shown in Figure 6.

In the initial 26-day orbit, the equilibrium sail temperature is in the order of 300°C, assuming a chromium rear coating, which is close to the operating limits of most polyimide films. Therefore, high-emissivity coatings are required to provide adequate thermal control for the sail.

Figure 5. A 3.12 year trajectory to 0.172 AU, with a characteristic acceleration of 0.25 mm/s^2

Figure 6. A 1.7 year cranking to solar polar orbit





The thermal loads on the payload are also extreme, although the sail may be utilised as a sunshade. They can be reduced by increasing the initial orbit radius from 0.172 AU, at the expense of maintaining strict Sun-synchronous orbital conditions. For illustrative purposes, however, we maintain a 26-day orbit.

The baseline 634 kg spacecraft bus and payload are used for sizing the solar sail, with only an appropriate mass reduction to eliminate SEP-specific components. Again, a sail-assembly loading of 10 g/m^2 is assumed. Since the required characteristic acceleration is now only 0.25 mm/s^2, a 167 m x 167 m solar sail is required, with a total launch mass of 911 kg. Table 3 compares the mass breakdowns for the solar-sail mission and the current mission using SEP. It can be seen that the solar-sail approach offers a launch-mass reduction of over 300 kg, allowing a significant increase in payload mass. Again, the launch mass is to a C_3=0 trajectory with an unconstrained launch window due to the absence of gravity-assist manoeuvres. More importantly, the use of a solar sail leads to a significant increase in the quality of the science from the mission by achieving a true solar-polar orbit, and offering the opportunity to reach the solar co-rotation region. As solar sail-technology develops, therefore, future solar missions could benefit significantly from this approach.

A mission to study the Earth – Polar Observer
Geostationary orbit provides a convenient location for communications 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 is also an ideal vantage point for Earth observation, providing coverage of large geographical regions. While the advantages of geostationary orbit for communications 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 the latter is degraded by foreshortening, while the poles are entirely excluded from view. Likewise, communications 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 about 81 deg.

Table 3. Mass budgets for the Solo-Sail and Solo-SEP missions

SAIL	(kg)	SEP	(kg)
Booms (150 g/m)	71	Thrusters	25
Sail film (3 μm)	119	PPU	27
Coatings (Al + Cr)	10	Tanks	34
Bonding	17	Cruise power	155
Mechanisms	61	Xenon fuel	299
SAIL TOTAL	278	SEP TOTAL	540
PAYLOAD		PAYLOAD	
Instruments	145	Instruments	145
Structure	131	Structure	131
Orbiter power	64	Orbiter power	64
Common systems	182	Common systems	182
Mechanisms	46	Mechanisms	67
Harness	10	Harness	34
Pyros	6	Pyros	10
Adapter	50	Adapter	50
PAYLOAD TOTAL	634	PAYLOAD TOTAL	683
LAUNCH MASS (C_3=0)	912	LAUNCH MASS (C_3>0)	1223

It can be shown that solar sails may be used to generate families of artificial equilibrium solutions (Lagrange points) in the Sun–Earth three-body system. Artificial out-of-plane equilibria may be used for continual, low-resolution imaging of the Earth's high-latitude regions. In fact, if the artificial Lagrangian point is located high enough above the ecliptic plane, the solar sail may be stationed directly over the north pole, or indeed the south pole, during the summer solstice. The solar sail can be stationed directly over the north pole at the summer solstice, as shown in Figure 7, but will not remain over the pole during the entire year due to the tilt of the polar axis. From this unique vantage point, a constant daylight view of the north pole is available at the summer solstice, but six months later at the winter solstice the polar regions are in permanent darkness (Fig. 8).

The solar-sail performance needed can be minimised by appropriate selection of the polar altitude. It can be shown that an equilibrium location some 3.8 million km (~600 Earth radii) above the North Pole will minimise the demands on the solar sail's performance. Closer equilibrium locations are possible using larger or higher performance solar sails, or by selecting a less demanding viewing geometry. While this is clearly a long path length, a number of applications can be identified.

In this example, we have assumed that a 100 kg bus and instrument payload will be necessary to support a 35 kg optical imager with 65 kg of associated subsystems in the proposed orbit. To station the payload at this unique polar viewing point requires an 139 m x 139 m square solar sail, with a total launch mass of 299 kg, assuming a sail-assembly loading of 10 g/m^2 (Table 4). The booms are again assumed to be made of CFRP with a specific mass of 100 g/m, and the sail is assumed to be

NASA/JPL-developed 3-micron aluminised film. The total launch mass is suitable for delivery to a C_3=0 escape trajectory from a range of small launchers. The sail is used to spiral from the escape trajectory to the artificial equilibrium point high over the Pole.

Although the solar sail's distance from the Earth is large for imaging purposes, there are potential applications for real-time, low-resolution imaging for continuous viewing of large-scale polar weather systems, along with Arctic ice and cloud coverage, for global climate studies. Although such images can be acquired by assembling a mosaic of instrument swaths from a conventional polar-orbiting satellite, many high-latitude passes are required to form a complete image. High resolution is then possible, but the completed image is not acquired in real time. Similar applications for real-time, low-resolution whole-Earth imaging are being developed for the NASA Triana mission located at the classical L1 Lagrangian point, sunward of the Earth.

For a 30 cm-aperture instrument stationed 3.8 million km from the Earth and operating at optical wavelengths, a minimum ground resolution of order 10 km is possible, assuming near-diffraction-limited optics. In practice though, the actual resolution obtained will be degraded due to such factors as the pointing stability of the camera. Higher resolution is possible if an equilibrium location closer to the pole is selected, at the expense of increased demands on the solar sail's performance. Other applications of these orbits include line-of-sight, low-bandwidth communications to high-latitude users. Applications for continuous data links to Mars polar landers and surface rovers have also been explored for a solar sail stationed high above the poles of Mars.

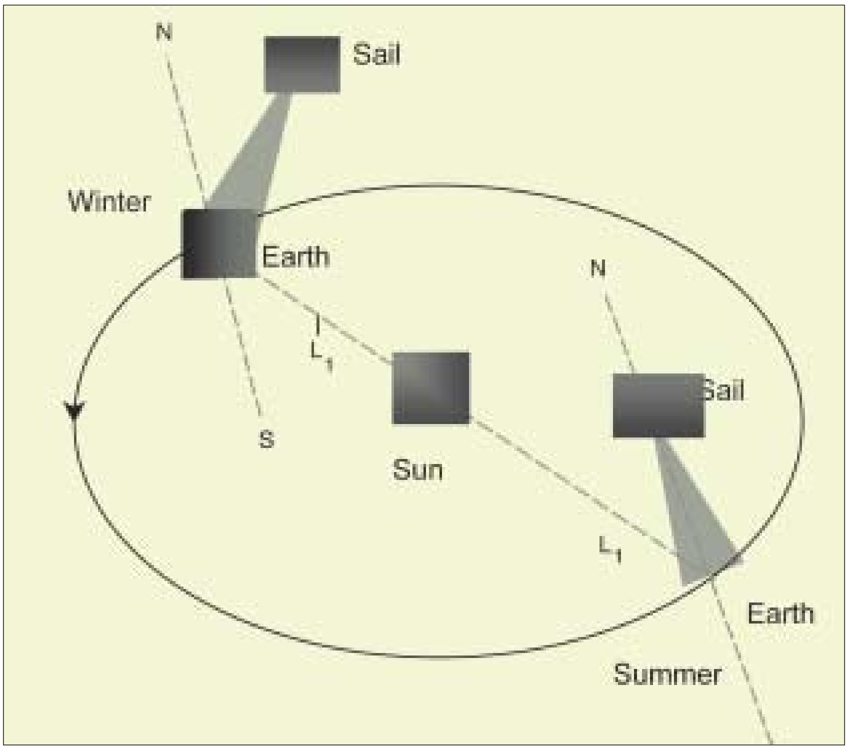


Figure 7. The Polar Observer mission concept

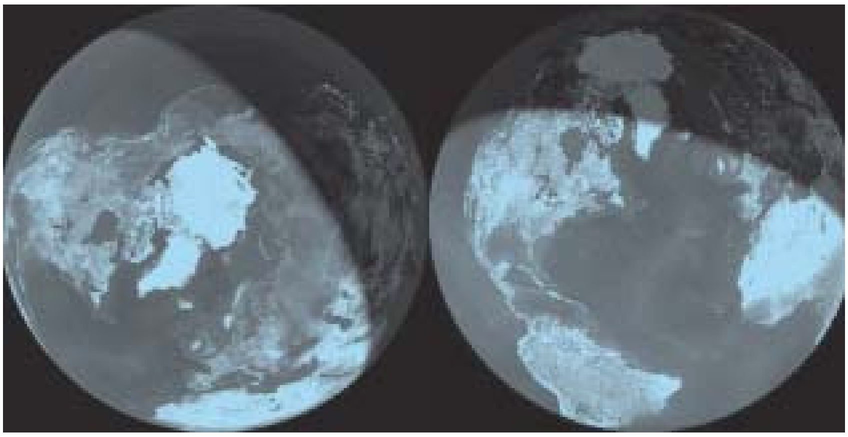


Figure 8. Summer and Winter Solstice views

Table 4. Mass budgets for the Polar Observer mission

SAIL	(kg)
Booms (100 g/m)	39
Sail film (3 µm)	84
Coatings (Al + Cr)	7
Bonding	9
Mechanisms	55
SAIL TOTAL	194
PAYLOAD	
Instruments	35
Bus	65
Adapter	5
PAYLOAD TOTAL	105
LAUNCH MASS	299

Conclusions

By assessing the mission scenarios of a range of future space missions, very promising opportunities for potential mission applications for solar-sailing propulsion have been identified. This low-cost delivery system with basically unlimited Δv capability holds promise for significantly enhancing or even enabling space-exploration missions in the new millennium. Clearly, the readiness of the solar-sailing technology for implementation in an actual space mission will require the technological demonstration of in-orbit deployment, orbit raising and navigation of a solar sailcraft. The first essential step will be achieved through the in-orbit deployment demonstration project now in progress.