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NEAR-TERM, LOW COST MISSIONS FOR SOLAR SAILS

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Solar sails have long been considered for a range of future mission applications. Most of these applications have been conventional missions where the solar sail is utilised primarily as a means of efficient, low thrust propulsion for orbit transfer. For this reason solar sailing has historically been neglected in favour of electric propulsion, which has seen considerable development in recent years. However, there are many high energy missions which are either enabled by solar sailing or can be achieved at lower cost than with electric or chemical propulsion. In this paper some of these low cost, high energy missions will be reviewed. In particular, new families of non-Keplerian orbits will be investigated which are quite unique to solar sailing. These orbits have novel mission applications, many of which only require the use of only moderate performance solar sails. Since the orbits and applications are unique to solar sails, they are mission enabling rather than mission enhancing, compelling mission planners to develop and utilise solar sail technology.

1. INTRODUCTION

Solar sails utilise the continuously available flux of momentum transported by photons from the Sun as a source of motive force and therefore do not require reaction mass. As such they provide an ideal form of propulsion for high energy missions, and indeed can enable entirely new classes of mission which are essentially impossible using chemical or electric propulsion. While the early development of solar sailing centred on concepts requiring extremely large sails, recent developments in payload miniaturisation have led to much more modest sail concepts. It has also been realised only recently that solar sails offer the possibility of unique, non-Keplerian orbits with novel mission applications. Historically though solar sailing has been pushed as a technology, rather than being pulled along by mission applications. However, due to developments in payload miniaturisation and the identification of new unique applications, solar sailing is now under serious consideration for a number of near-term missions [1].

In order to compare solar sail designs a standard performance metric is required. The most common metric is the solar sail characteristic acceleration, defined as the solar radiation pressure acceleration experienced by a solar sail facing the Sun at a distance of one astronomical unit (au), the mean distance of the Earth from the Sun. At this distance from the Sun the magnitude of the solar radiation pressure P is 4.56×10^{-6} N m^{-2}. Therefore, multiplying this pressure by the sail area A yields the solar radiation pressure force exerted on the solar sail. Dividing by the sail mass m then yields the solar sail acceleration. A factor of two must also be added to account for the sail reflectivity since reflected photons impart a reaction of equal magnitude to incident photons. However, a finite sail efficiency η must be incorporated to allow for the non-perfect optical properties of the sail coating and billowing of the sail film. From this calculation the solar sail characteristic acceleration a_0 is then defined as $a_0 = 2\eta P/\sigma$ with $\sigma = m/A$ where σ is the solar sail mass per unit area, termed the sail loading. While the actual solar sail acceleration is a function of heliocentric distance, and indeed the sail orientation, the characteristic acceleration allows a comparison of solar sail design concepts on an equal footing. If a thin sail film, say 2 μm thick, is available a typical solar sail characteristic acceleration may be of order 1 mm s^{-2} corresponding to a sail loading of 9 g m^{-2}. While this is a useful canonical value, first generation solar sails are likely to have a somewhat lower performance.

The last major effort to develop solar sailing on behalf on a national space agency was the 1977 NASA/JPL study to design and fabricate a solar sail to rendezvous with comet Halley during its pass through the inner solar system in the mid-1980s [2].

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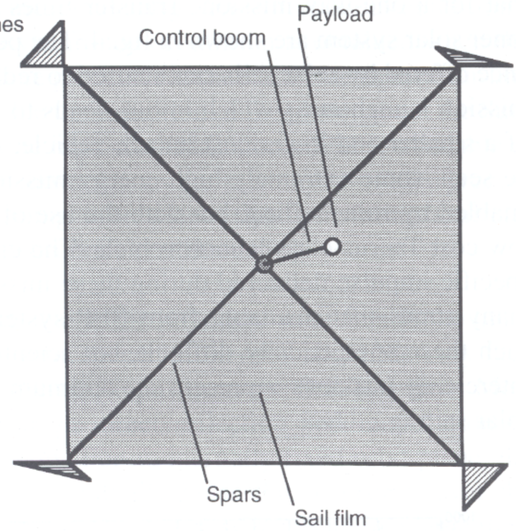


Fig. 1 Schematic square solar sail configuration: attitude control using reflective spar tip vanes and/or centre-of-mass displacements using a control boom.

The NASA/JPL effort led to designs for an 800 x 800 m square sail (fig. 1) and a 14 blade heliogyro (fig. 2). For large sails the rotating heliogyro configuration in principle offers a more reliable deployment scheme by using long blades of reflective sail film, rather than a continuous square sheet of film. The individual sail blades can simply be unrolled from a rotating hub, whereas the square sail requires a deployable structure which is extended to unpack the folded sail film. Due to the large sail area required to rendezvous the 850 kg science payload with comet Halley the heliogyro configuration was ultimately selected at the conclusion of the NASA/JPL study. However, following an assessment of the readiness of solar sailing and the competing solar electric propulsion (SEP) mission concept, solar sail development was terminated. Ultimately though the

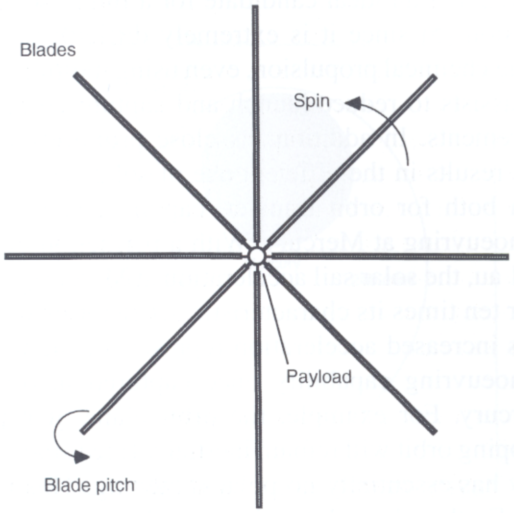


Fig. 2 Schematic heliogyro configuration: attitude control using collective and cyclic blade pitch modes.

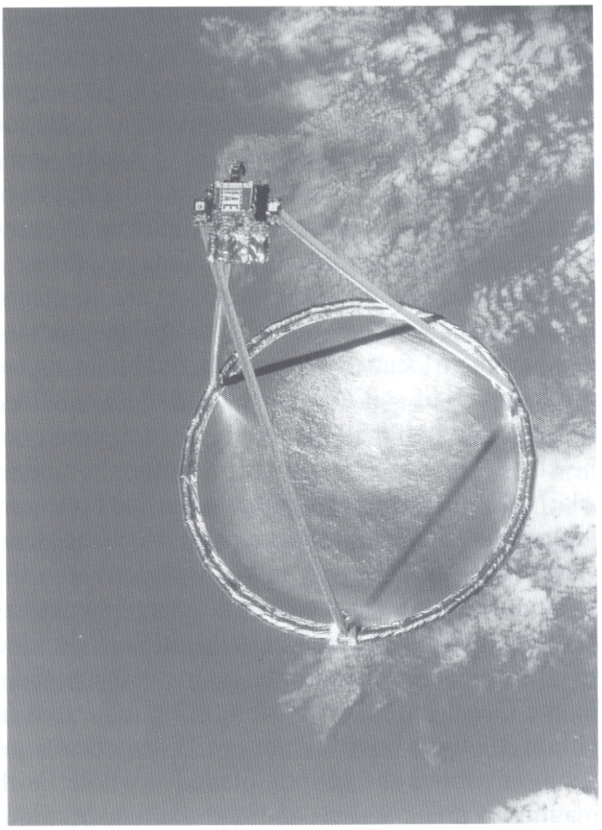


Fig. 3 Inflatable Antenna Experiment (IAE) deployment on STS-77 20 May 1996. (NASA/JPL)

SEP mission concept was dropped and along with it any hope of a comet Halley rendezvous mission [3]. Unfortunately the large sail concepts for the comet Halley mission then came to symbolise solar sailing during the 1970s and 1980s.

More recently though, developments in payload miniaturisation and a new management enthusiasm for enabling technologies has led to a re-evaluation of solar sailing for future mission applications. In particular, several new mission concepts capitalise on the successful demonstration of inflatable structures performed during the Inflatable Antenna Experiment (IAE) conducted during the STS-77 mission in May 1996 (fig. 3). While the IAE structure, with three 28 m long booms, is not large enough for a modest solar sail, it clearly demonstrated the reliable deployment of inflatable booms and thin film membranes. Further advances in low mass structures have been made by DLR using carbon-fibre techniques for coilable booms. These developments form the basis of the proposed ODISSEE technology demonstration mission, which has undergone an extensive pre-phase-A study by DLR and JPL. The sail is designed to be launched as a piggy-back payload on the Ariane-5 ASAP auxiliary payload ring to geostationary transfer orbit from where orbit raising experiments will be conducted [4].

Along with technology development, new classes of solar sail mission have recently been devised. These mission utilise families of non-Keplerian orbits which rely on the continuous solar radiation pressure force exerted on the sail to enable some rather unusual orbits. One particular family of non-Keplerian orbits is obtained in the Sun-Earth system where new artificial equilibrium points may be generated. These new equilibrium points are similar to the classical Lagrange gravitational balance points, however their location can be selected by an appropriate choice of sail loading and orientation. Since the local gravitational acceleration in the vicinity of the classical Lagrange points is small, only modest solar sails are required for these missions. Of particular interest are equilibrium points displaced sunward of the L_1 Lagrange point or displaced above the L_1 point, high above the ecliptic plane. Locations sunward of L_1 provide an ideal location to provide early warning of solar storms, which is the concept for the Geostorm ST-5 mission, to be discussed in section 3.1. Locations high above L_1 provide a vantage point above the ecliptic plane for whole Earth imaging of polar regions and high latitude communications. This is the concept for the Polar Observer mission to be discussed in section 3.2. Both of these missions use the same family of orbits although the sail performance requirements are somewhat more demanding for the Polar Observer mission.

2. PAYLOAD DELIVERY MISSION APPLICATIONS

It can be argued that solar sails must have infinite specific impulse since they do not require reaction mass to generate a motive force. While this is true, for a finite mission duration the solar sail will only deliver a finite total impulse. In particular, if the solar sail is used to deliver a payload to some high energy orbit, the mass of the solar sail itself becomes redundant after the payload is delivered. Therefore, if it happens that the mass of the solar sail is greater than the mass of a chemical propulsion system required to perform the same mission, the solar sail can hardly be said to be more efficient, even although it has in principle infinite specific impulse. Infinite specific impulse is only achieved in practice for an infinite mission duration.

For such payload delivery missions the effective specific impulse delivered by a solar sail can be extremely high, particularly for inner solar system and sample return missions. For missions in the inner solar system the solar sail performance is greatly enhanced by its proximity to the Sun while the Δv for a sample return mission can be double that for a one-way mission. Transfer times for the inner solar system are shown in fig. 4. The performance offered by solar sails can be used to reduce the mission launch mass which in turn leads to the use of a smaller and lower cost launch vehicle. As will be seen, some extremely high energy missions are enabled by solar sailing, but with the use of only a low cost Taurus-class launch vehicle. The effective specific impulse delivered during these missions is many times that available from SEP systems. As such these energetically difficult, but scientifically interesting missions represent an optimum use of solar sailing.

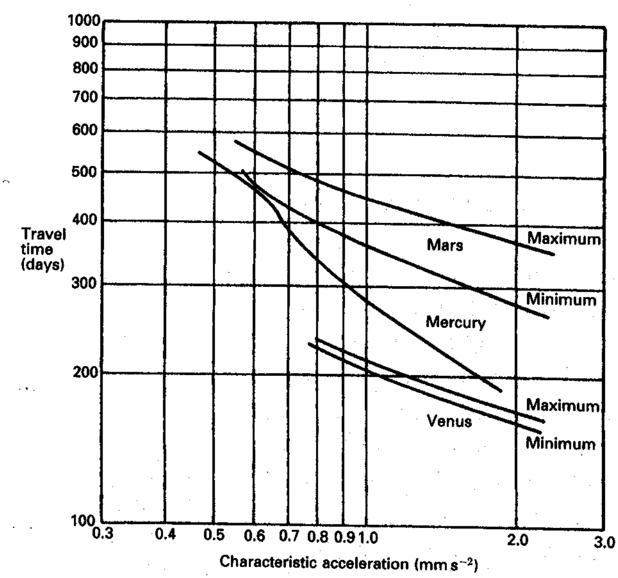


Fig. 4 Optimised solar sail transfer times in the inner solar system. (NASA/JPL)

2.1 Mercury Orbiter Mission

Mercury is an ideal candidate for a future solar sail mission [5] since it is extremely difficult to reach using chemical propulsion, even using multiple gravity assists to reduce launch and capture energy requirements. In addition, its close proximity to the Sun results in the efficient use of solar sail propulsion both for orbit transfer, capture and for orbit manoeuvring at Mercury. With a perihelion of only 0.31 au, the solar sail acceleration at Mercury can be over ten times its characteristic acceleration at 1 au. This increased acceleration represents a significant manoeuvring capability once captured in orbit at Mercury. For example, the orbit plane of a polar mapping orbit will remain inertially fixed since Mercury has essentially no polar flattening. This is unlike Earth polar orbits where the harmonics of the geopotential can be used to naturally precess the orbit plane, generating Sun-synchronous conditions. A Sun-synchronous condition is obtained when the orbit plane precesses at the same rate as the Sun-line rotates due to the motion of the planet about the Sun. Due to the high solar radiation pressure available at Mercury however, a solar sail may be used to precess the mapping orbit plane, thus generating artificial Sun-synchronous conditions [6].

Again due to the increased solar flux in the inner solar system, spacecraft in orbit about Mercury face a large thermal input from the Sun and from Mercury through direct reflection and re-radiation of solar energy. For an inertially fixed orbit plane, the orbit will eventually bring the spacecraft over the sub-solar point on Mercury resulting in a large thermal input on both the front and rear surfaces of the spacecraft. For low mapping orbits it can be shown that conventional thermal control methods are insufficient for such extreme conditions. In addition, an inertially fixed orbit will in general result in regular thermal shocks as the spacecraft transits the terminator through the day-night cycle. In order to avoid these difficulties a Sun-synchronous polar mapping orbit close to the terminator is required, as shown in fig. 5. Such an orbit provides global mapping of the surface of Mercury under constant lighting conditions and limits heating due to reflection and re-radiation from Mercury. As noted, such an orbit can be generated using a solar sail to artificially precess the orbit plane as Mercury orbits the Sun. The Δv for orbit precession corresponds to 3.6 km s^{-1} every 88 days (Mercury year), representing a significant overhead for any mission concept using reaction propulsion.

A Mercury orbiter study by DLR [5] has identified a baseline Sun-synchronous Mercury orbiter

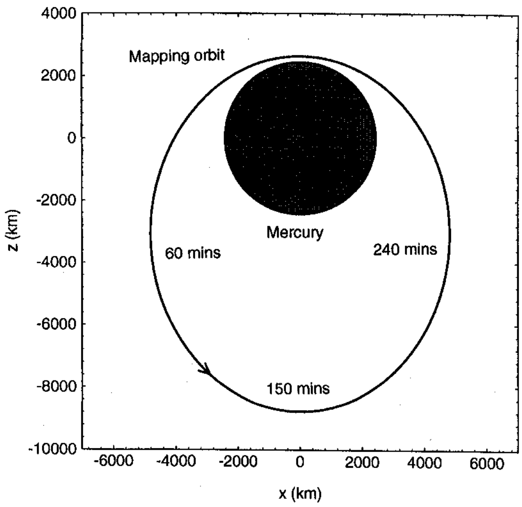


Fig. 5 Sun-synchronous polar mapping orbit at Mercury.

mission suitable for delivery using a low cost launch vehicle, such as the Taurus family with a small kick-stage. The solar sail design utilises low mass carbon-fibre coilable boom technology and can deliver a small science payload with 20 kg of instruments (excluding the spacecraft bus). The baseline mission requires an 86 x 86 m solar sail with a launch mass of 240 kg generating a characteristic acceleration of 0.25 mm s^{-2}. The solar sail is injected directly into an Earth escape trajectory using the Taurus launch vehicle and uses solar sail propulsion for orbit transfer, capture and orbit manoeuvring at Mercury. A cruise duration of order 3.5 years is required to reach Mercury with a further 4 weeks for a spiral orbit capture. The capture phase is relatively short due to the large solar radiation pressure acceleration experience by the solar sail at Mercury. A more advanced option requires a 150 x 150 m solar sail with a characteristic acceleration of 0.55 mm s^{-2} resulting in a transfer duration of only 1.8 years. The larger launch mass of order 350 kg would require a more capable member of the Taurus launch vehicle family.

2.2 Solar Polar Sail Mission

The Solar Polar Sail mission [7] requires a solar sail to transfer a science payload from Earth escape to a polar orbit 90° to the solar equator, corresponding to an inclination of 83° to the ecliptic plane. The polar orbit is achieved by rotating the sail twice per orbit so that the solar radiation pressure force exerted on the sail is directed alternately above and below the orbit plane. This so-called cranking manoeuvre is an efficient means of rapidly increasing orbit inclination [8]. In addition to a polar orbit, the mission requires that the spacecraft remains almost normal to the Sun-Earth line. With this particular orbit geometry coronal mass ejections (CMEs) can be imaged against dark space as they propagate along the Sun-Earth line. As will be discussed in section 3.1.1, CMEs propagating towards Earth are responsible for geomagnetic storms and cannot be adequately imaged from Earth against the bright solar disk. In order to keep the spacecraft normal to the Sun-Earth line the orbit period is chosen to be resonant with that of the Earth. Therefore, the polar orbit period must be $1/N$ years, for some integer N. This resonance condition also avoids configurations where the solar sail is occulted by the Sun.

The 1:1 resonance obtained when N=1 corresponds to a polar orbit at 1 au. This orbit ensures that the solar sail remains almost normal to the Sun-Earth line, providing ideal viewing geometry for CMEs propagating towards Earth, as shown in fig. 6. Although the viewing geometry is good for CMEs,

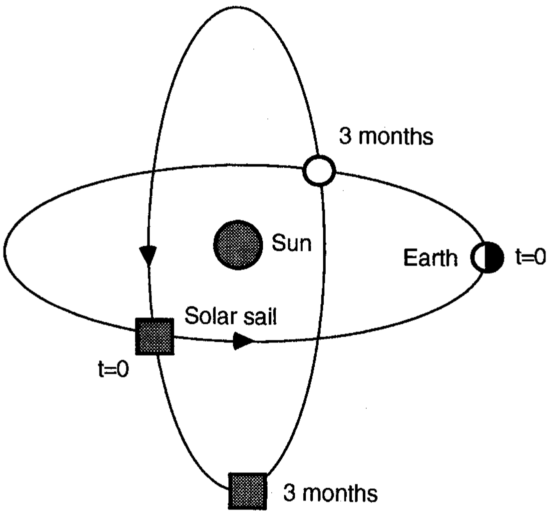


Fig. 6 1:1 resonance solar polar orbit.

the large orbit radius requires large aperture instruments for effective imaging of the solar disk and corona. However, a shorter 3:1 resonance obtained with an orbit radius of 0.48 au provides good solar imaging while keeping the spacecraft close to 90° from the Sun-Earth line for much of the mission. This is the orbit selected for the baseline mission concept.

The primary instrument for the Solar Polar Sail mission is a coronograph to image the solar corona. The coronograph provides wide field images of the Sun at a single wavelength with an occulter placed in the instrument field of view to remove the bright solar disk from the image. Views of the solar corona obtained from high above the ecliptic can be combined with in-plane images obtained from Earth to allow the three-dimensional structure of coronal features to be constructed. The second main instrument is an all sky imager, used primarily for imaging CME propagation. The imager comprises a fish-eye lens with a CCD detector and a system of optical baffles to reduce interference from stray light. The imager can then track the propagation of CMEs against the dark sky background. Again, it is only from a solar polar orbit, normal to the Sun-Earth line that CMEs propagating towards Earth can be adequately viewed against dark sky.

Aside from the fundamental solar physics to be investigated during the mission, the ability to view CMEs propagating along the Sun-Earth line allows useful early warning of geomagnetic activity. While normal science data can be telemetered in regular sessions, warning data is required in near real-time. A solution to this problem is provided by beacon-mode technology, currently under development for other deep space missions. During beacon mode only a simple two-level tone signal is telemetered. A change from the quiet tone to an active tone indicates that a CME has been detected. A high gain antenna from the deep space network (DSN) can then be used to down-link the CME images and supporting data. Since a low bit rate signal is normally telemetered during beacon mode, only small ground stations are required for real-time monitoring. A network of at least three such ground stations is required for full 24 hour coverage.

The baseline concept for the mission is a 150 x 150 m square solar sail with coilable tubular booms based on the DLR concept discussed in section 2.1. In order to minimise the sail mass, a thin 2 μm Kapton sail is required with an emissive Chromium thermal control coating on the rear surface of the sail film. The bus and payload are mounted at the end of a 10 m deployable boom which is articulated to displace the centre of mass of the sail to provide attitude control torques, as shown in fig. 7. Since an Earth escape spiral is not required attitude control demands are modest, although periodic reversals of the solar sail orientation are required during the cranking orbit phase. The bus and payload are located at the centre of the sail in a circular cut-out which prevents additional thermal input due to solar radiation reflected from the sail film.

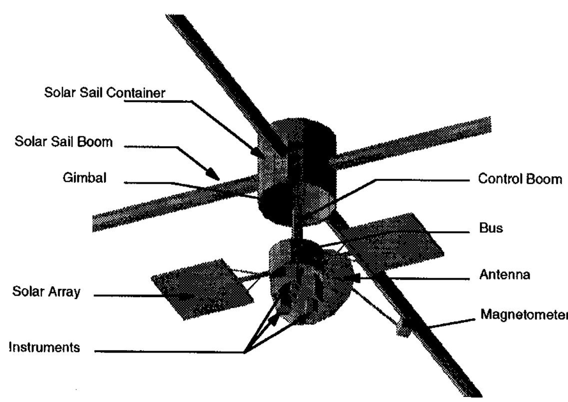


Fig. 7 Solar Polar Sail mission configuration. (NASA/JPL)

On reaching the mission orbit the solar sail is jettisoned so that the bus and payload become an independent free-flying spacecraft. The bus must therefore fully support the payload, providing attitude control functions for the stringent pointing requirements demanded by payload instruments such the coronograph. The bus must almost provide propulsion for a clean separation from the solar sail. Based on a suite of field and particle instruments in addition to the coronograph and all-sky imager, the total payload mass is of order 30 kg with a 135 kg bus. Adding the solar sail mass of 150 kg and appropriate contingencies, the total launch mass is some 380 kg, which is within the capacity of the Taurus launch vehicle with a suitable kick-stage. It is quite remarkable that such a high energy mission, with an effective Δv of order 50 km s^{-1}, can be delivered using a relatively small, low cost launch vehicle. The mission is truly enabled by solar sailing and makes optimum use of the technology by tackling a high energy mission which is essentially impossible for conventional propulsion.

3. NON-KEPLERIAN ORBIT MISSION APPLICATIONS

Due to the continually available solar radiation pressure force, solar sails are capable of exotic non-Keplerian orbits which are essentially impossible for any other type of spacecraft [9]. Although some of these missions require advanced, high performance solar sails others are possible using relatively modest sail concepts. The solar sail performance required for these orbits is a function of the local gravitational acceleration. Therefore, to displace a solar sail high above the ecliptic plane for example, requires a characteristic acceleration of order 6 mm s^{-2}, while to generate an artificial Lagrange point may only require a characteristic acceleration of order 0.25 mm s^{-2}. While these orbits are not 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 vehicle. As such they represent new families of orbits for solar sail with some novel mission applications.

Firstly, using an advanced solar sail it is possible to choose its total mass per unit area so that the solar radiation pressure force exactly balances the 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 balance point is approximately 6 mm s^{-2}, corresponding to a solar sail loading of only 1.5 g m^{-2}. Such a solar sail would enable solar physics missions which 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 Sun-centred orbits high above the ecliptic plane, with the orbit period chosen to be synchronous with the Earth or some other solar system body. When solar sails are available with a characteristic acceleration of order 6 mm s^{-2} large families of exotic new orbits will be available for mission applications [10].

For high performance solar sails operating in the vicinity of the Earth, it has been demonstrated that they may be used to displace communication satellites above and below the geostationary orbit plane. This concept, proposed by physicist Robert Forward, would allow satellites to be stacked above and below the equatorial plane, greatly increasing the number of available slots. Using more modest solar sails the location of the Earth-Sun Lagrange balance points can be artificially displaced. For example, the interior L_1 point 1.5 million 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 orbit the location of the L_1 point can be artificially displaced, closer to the Sun or even above the ecliptic plane. Since the local gravitational acceleration in the vicinity of L_1 is small, only modest solar sails are required. For example, a solar sail with a characteristic acceleration of only 0.25 mm s^{-2} can double the distance of the L_1 point from Earth. As will be seen, this new equilibrium location appears useful for providing early warning of solar plasma storms, before they reach Earth. A solar sail with double the performance can be permanently stationed high above the L_1 point so that it appears above the Arctic regions of the Earth. Even allowing for the long path length involved, such an equilibrium location would enable continuous communications with high latitude regions or continuous, real-time polar imaging. This is in stark comparison to conventional geostationary satellites which appear low or below the horizon in high latitude and polar regions.

3.1 Geostorm ST-5

3.1.1 *Background*

On March 13 1988, 6 million people in Quebec province were left without electrical power due to the temporary loss of 9450 MW of capacity from the Hydro-Quebec power company grid. This massive power black-out was due to a geomagnetic storm inducing currents in power transmission lines which led to switch gear and transformer failures. Other recent geomagnetic storm events have resulted in the partial or total loss of communication satellites, disruption to satellite navigation systems and interference to terrestrial radar networks. As reliance on networked global communication systems grows, disruption due to geomagnetic storms is likely to increase. The primary goal of the Geostorm mission is to provide enhanced warning of such storms to allow preventative action to be taken to protect vulnerable systems.

Geomagnetic storms are principally the result of

CMEs, the violent release of large volumes of plasma from the solar corona. These high speed bubbles of plasma are transported outwards through the solar system, often driving shock waves into the slower solar wind. If the geometry of the CME release is correct, the CME trajectory may result in the plasma impinging on the Earth's magnetosphere. If the magnetic field lines of the CME are anti-parallel to those of the Earth, magnetic reconnection can occur allowing the CME to deposit energy into the magnetosphere. The resulting perturbation to the Earth's magnetic field can then ultimately lead to currents being induced in terrestrial conductors. Most susceptible to these induced currents are long electricity power transmission lines and long pipelines. Areas with igneous rock are particularly susceptible since this rock type has low electrical conductivity. Rather than seeking paths through the ground, the induced currents will flow through man-made conductors.

Historically, the prediction of geomagnetic storms has been an inexact affair due the complexity of the physical process involved. However, due to the increasing civil and military reliance on satellite communications and the economic consequences of major power black-outs, the accurate prediction of 'space weather' is now of considerable importance. Currently, predictions of future activity are made by the National Oceanic and Atmospheric Administration (NOAA) Space Environment Centre in Colorado using terrestrial data, such as images of the solar disk, and more recently using real-time solar wind data obtained from the Advanced Composition Explorer (ACE) spacecraft. The ACE spacecraft is a 785 kg spin-stabilised platform launched on 25th August 1997. The spacecraft is stationed on a halo orbit about the interior L_1 Lagrange balance point some 1.5 million km sunward of the Earth, as shown in fig. 8. From this vantage point the spacecraft has a continuous view of the Sun and its instruments are free from any disturbances by the Earth's magnetic field. A halo orbit is required to ensure that when the spacecraft is viewed from Earth for data returns it is sufficiently far from the solar radio disk to avoid interference from solar radio noise. Importantly, since the spacecraft is located sunward of the Earth, solar wind disturbances sensed by the suite of instruments on-board the ACE spacecraft can be used to provide early warning of impending geomagnetic storms. In particular, the spacecraft magnetometers can detect the polarity of a CME which determines if the CME will deposit energy in the magnetosphere, as discussed earlier. Typically a prediction of order one hour can be made from the Lagrange point orbit, enhancing the quality of forecasts and alerts to user groups.

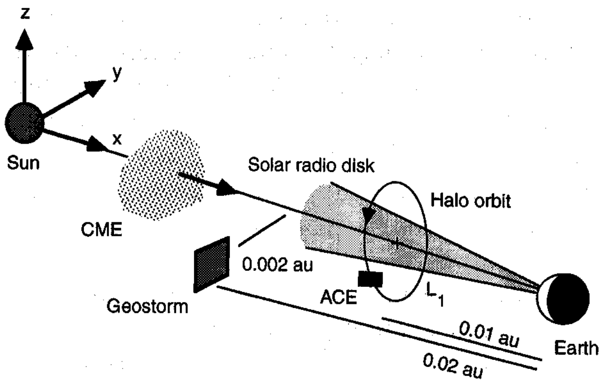


Fig. 8 ACE and Geostorm spacecraft relative to the Sun-Earth L_1 point.

The enhanced storm warning provided by the ACE mission is limited by the need to orbit the L_1 point on the Sun-Earth line. Since L_1 is a natural equilibrium point, only a modest Δv budget is required for station-keeping (due to the inherent instability of L_1 halo orbits). Moving closer to the Sun would enable even greater enhancements of warning times but would require an artificially generated Lagrange point. For a conventional spacecraft, a large Δv budget would be required to artificially orbit sunward of the L_1 point, while remaining close to the Sun-Earth line. For example, to achieve the Geostorm mission orbit described below, a conventional spacecraft would required a Δv budget of order 9 km s^{-1} per year of operation. However, since solar sails have in principle unlimited Δv, such artificial Lagrange points can be sustained indefinitely and in fact require only relatively modest solar sails.

The concept of enhancing warning times by orbiting sunward of L_1 was proposed to NOAA by the NASA Goddard Space Flight Centre in the early 1990s. It was not until June 1996 however that NOAA requested that the NASA Jet Propulsion Laboratory develop a mission concept for an operational storm warning mission [11]. The principal motivation for this request was to explore mission opportunities to ensure continuity of real-time solar wind data following the end of the ACE mission, projected for 2000-2002. In addition, developments in microspacecraft technologies and the successful demonstration of the IAE space inflatable structure in May 1996 (described in section 1) prompted serious consideration of solar sail technology. Ultimately, the goal is to station a solar sail twice as far from the Earth as L_1 while remaining close to the Sun-Earth line, as shown in fig. 8. Since CMEs will be detected earlier than at the natural L_1 point, warning times for operational forecasts and alerts will be at least doubled. As will be seen, this significant enhancement of warning time only requires a modest solar sail with a total loading of order 29 g m^{-2}. This high design loading then allows the use of commercially available 7.6 μm Kapton film for the sail substrate, and does not require the manufacture of specialised thin films.

It is clear then that Geostorm represents an unique mission application which is truly enabled by solar sailing. In addition, this key operational use of solar sail technology only requires a modest solar sail, which is important for a first mission. In earlier development plans such a modest solar sail would otherwise have served only as a flight test for more ambitious future interplanetary missions. However, with missions like Geostorm, investments in solar sail development can now be amortised much earlier in the development of the technology. It is also somewhat ironic to note that although more advanced mission applications for higher performance solar sails have been considered for some time, these unique applications for modest solar sails have been recognised only recently. The Geostorm mission concept is now an inter-agency partnership between NOAA, the US Air Force and the US Department of Energy and has been proposed to the NASA New Millennium programme as a candidate for the fifth programme mission (ST-5). If selected, the mission will be launched in the 2003 time frame and is likely to be the first operational solar sail mission.

3.1.2 Mission Orbit

The baseline mission concept requires that the solar sail is firstly transferred to a conventional halo orbit at L_1. The launch to L_1 will be a piggy-back to GTO or a dedicated launch using a Taurus class vehicle. For a launch to GTO an upper-stage will provide the necessary Δv of order 740 m s^{-1} to transfer to the Lagrange point halo orbit at a heliocentric distance of 0.99 au. At L_1 the sail will be deployed and the solar sail will then spiral inwards to the operating station at a heliocentric distance of 0.98 au. In the event of a failure in the deployment of the sail, the sail package can be jettisoned and the mission will continue at the initial L_1 halo orbit. In this eventuality real-time storm warning data is still provided, although the enhancement in storm warning time is not. The back-up mission at L_1 is of particular importance as it insures the end users of the mission data against the use of an untested technology. To avoid down-link interference from the solar radio disk, the solar sail will also be displaced away from the Sun-Earth line. For example, a displacement of 0.002 au normal to the Sun-Earth line corresponds to the solar sail being stationed on the edge of the 5° solar radio disk, as viewed from Earth.

For a nominal mission orbit 0.98 au from the Sun and 0.002 au from the Sun-Earth line, a three-body analysis can be used to determine the required solar sail performance [12]. It is found that rather than singular equilibrium points, large equilibrium surfaces are available in the vicinity of the classical Lagrange points. Firstly, the performance can be determined for an ideal solar sail to characterise the magnitude of the solar sail loading. It is found that for a perfect reflectivity a total solar sail loading of 29.6 g m^{-2} is required [13]. This is a rather modest performance in comparison to the demands of some interplanetary mission concepts. It is also found that a sail pitch angle of -0.82° is required to provide the necessary side force to displace the equilibrium location away from the Sun-Earth line. As the sail reflectivity is decreased the required solar sail loading increases, as does the required pitch angle. Alternatively, if the solar sail loading is fixed at 29.6 g m^{-2}, the equilibrium point must be displaced closer to the Earth [14] thus degrading the warning time enhancement somewhat, as shown in fig. 9. Although these artificial equilibrium points are relatively easy to achieve, they are in general unstable. However, it can be shown that although unstable, the instability is strictly controllable using feedback to the sail orientation.

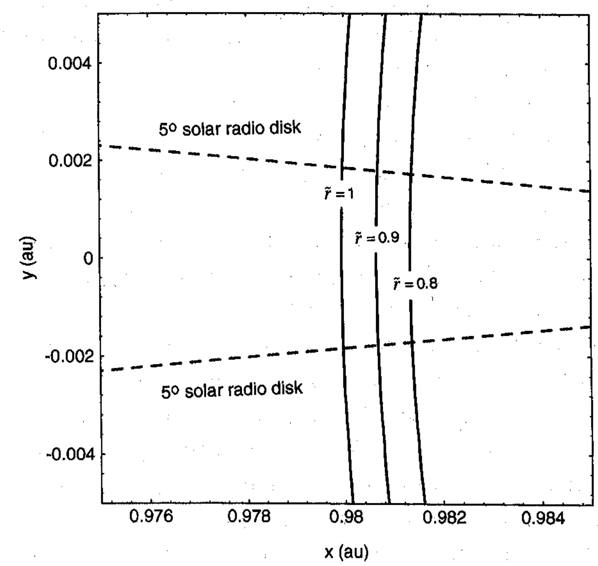


Fig. 9 Artificial equilibrium surfaces in the ecliptic plane for a sail loading of 29.6 g m^{-2} as a function of sail reflectivity.

3.1.3 Solar Sail Design

As noted in section 3.1.1, the Geostorm mission has been designed to capitalise on the successful dem-

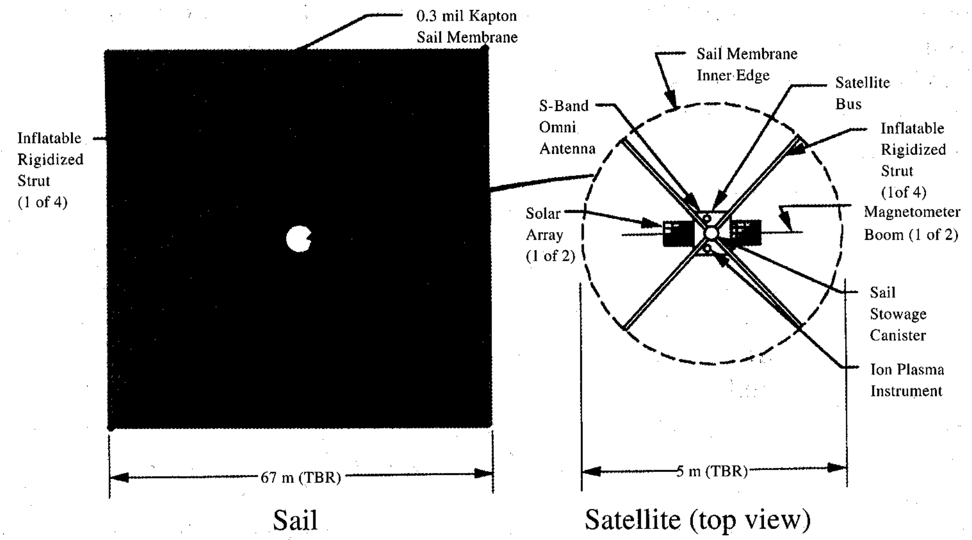


Fig. 10 Geostorm solar sail configuration. (NASA/JPL)

onstration of inflatable structures performed during the IAE mission on STS-77 in May 1996. The baseline solar sail design therefore centres on inflatable technology for both deployment and rigidisation of the sail film. Due to the modest performance demands on the solar sail, the sail film is manufactured from commercially available 7.6 μm Kapton, aluminised on one side. The use of off-the-shelf materials and prior inflatable technology leads to a small, low cost and, due to the back-up mission at L_1, relatively low risk mission for the first operational use of a solar sail.

In order to achieve a total solar sail loading of 29.6 g m^{-2}, a 67 x 67 m square solar sail is required for a total spacecraft mass of order 130 kg, including approximately 70 kg for the sail film and inflatable structure, as shown in fig. 10. The sail structure consists of a set of 47 m long, 0.1 m diameter tubular diagonal spars which inflate for deployment, drawing out the packaged sail film. The spars, along with the sail film, are compactly folded and stored separately in a canister on top of the spacecraft bus for launch and transfer to L_1. Following deployment the spars rigidise in a similar manner to the IAE structure. The bus and payload are located at the centre of the solar sail in a circular cut-out and are attached to the sail spar structure, again shown in fig. 10. Attitude control is provided by conventional hydrazine thrusters mounted on the spacecraft bus. Since solar sail propulsion is not required for either escape from GTO or transfer to L_1, only slow turning rates are required for station-keeping at the artificial Lagrange point.

In addition to returning operational storm warning data, the Geostorm mission is also conceived as a demonstration of solar sail technology, so that imaging of the sail deployment sequence is required. Images will be obtained using two microcameras, one on each face of the sail which are to be deployed at the ends of tension wound wires which unwind automatically on release. The images from the cameras will be stored and slowly returned by embedding compressed image data in the stream of operational storm warning data. Storm warning is provided by two magnetometers and an ion plasma instrument to characterise the solar wind plasma. The magnetometers provide magnetic field strength and direction every few seconds while the ion plasma instrument provides measurements every few minutes. Knowledge of the magnetic field orientation is important to determine the polarity of the CME, as discussed in section 3.1.1. The entire spacecraft has a design life of 3 years with a 5 year goal, providing storm warning data until at least 2006. If successful, the mission will form the first in a series of missions providing enhanced solar storm warning data using solar sail technology.

3.2. Polar Observer

3.2.1 Background

Geostationary orbit provides a convenient location for communication 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, providing coverage of large geographical regions. While the advantages of geostationary orbit for communications and Earth observation are undisputed, there are however some 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 effects 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 ±81°.

High latitude regions are of importance for a number of military, commercial and environmental interests. Firstly, the high Arctic was a strategically important region during the cold war. While times have changed, there is still a need to provide high latitude military communications. In addition, the growing interest in the Arctic and Antarctic regions for mineral and oil extraction may lead to a demand for communication services. The Arctic and Antarctic are also of great environmental importance and there is a requirement for relaying data from remote weather stations or other automated monitoring platforms. Additional environmental requirements for polar services include continuous imaging of polar weather systems, real-time imaging of aurora, which can cause radar clutter and indicate energy deposition in the magnetosphere, and monitoring polar ice coverage for climate studies.

The limitations of geostationary orbit for such applications may be overcome to some extent by using satellites in polar orbit. Imaging of high latitude regions can be accomplished with good viewing geometry and high resolution, although complete mapping relies on the assembly of a mosaic of individual instruments swaths obtained during high latitude passes. The field of view of the satellite can be broadened by raising the orbit altitude. However, this increases the satellite orbit period and so leads to a longer wait between imaging. Similarly, communication services can be provided using high inclination Molniya orbits or through constellations of satellites in low Earth orbit. Molniya orbits are inclined, highly elliptical orbits which are oriented so that the orbit apogee is fixed high above the Arctic or Antarctic. When in view, the satellite therefore appears to move slowly across the sky well above the local horizon. At least three satellites are required to provide continuous coverage, with the communication link switching between satellites as each satellite rises and sets. Similarly, low Earth orbit satellite constellations can provide communication links between high latitude regions and any other point on the surface of the Earth by passing digital data packets between members of the constellation. While such systems are well suited for infrequent mobile communications, they are expensive to use if a continuous, real-time link is required for relaying data to and from remote platforms or other users.

3.2.2 Mission Concept

It has been shown in section 3.1.2 that solar sails may be used to generate artificial Lagrange equilibrium solutions in the Sun-Earth three-body system. While in-plane equilibria have applications for missions such as Geostorm, detailed in section 3.1, out-of-plane equilibria may be utilised for continual, low resolution imaging of the high latitude regions of the Earth. In fact, if the artificial Lagrange point is located high enough above the ecliptic plane, the solar sail may be stationed almost directly over the north pole, or indeed the south pole, during the appropriate time of year. Such orbits have recently been investigated by the University of Glasgow for NOAA to define mission applications for a solar sail displaced high above the L_1 point [15,16]. The solar sail can be stationed directly over the north pole at the summer solstice, as shown in fig. 11, but will not remain over the pole during the entire year due to the tilt of the Earth's spin axis. From this unique vantage point a constant daylight view of the north pole is available during summer, however six months later at the winter solstice the polar regions are in permanent darkness. A schematic illustration of the view obtained at both the summer and winter solstice is shown in fig. 12.

It is found that the required solar sail performance can be minimised by an appropriate selection of polar altitude. It can be shown that an equilibrium

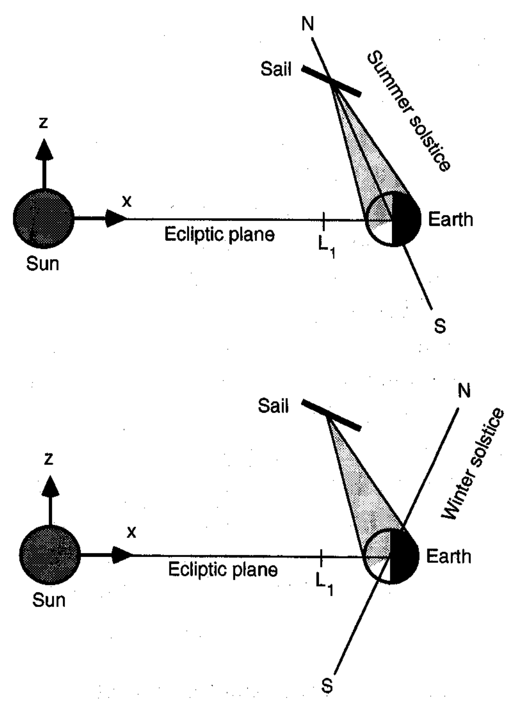
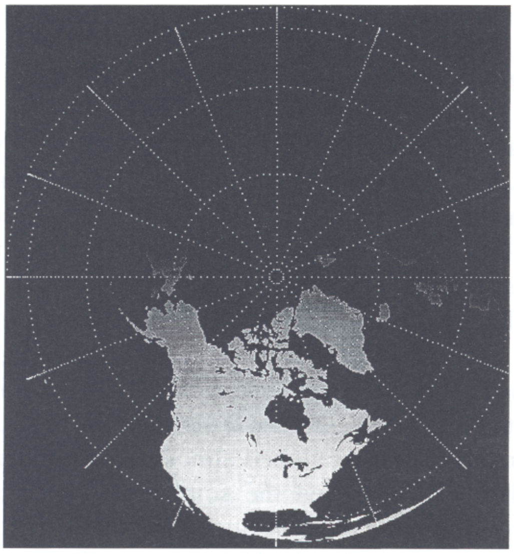


Fig. 11 Polar Observer mission concept illustrating the field of view at the summer and winter solstice.



a

b

Fig. 12 Polar Observer view at **(a)** summer solstice **(b)** winter solstice.

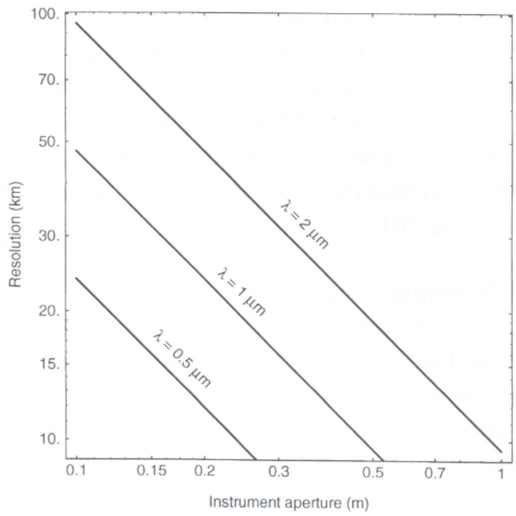


Fig. 13 Minimum ground resolution at the Polar Observer mission orbit.

location some 3.9 million km above the north pole will minimise the demands on the solar sail performance [16]. Closer equilibrium locations are possible using larger, or higher performance solar sails, or indeed selecting a less demanding viewing geometry. At this location 3.9 million km above the Earth, the solar sail is stationed directly over the north pole during the summer solstice. During the winter solstice the solar sail still appears 43° above the horizon at the north pole and can be viewed from latitudes down to 49.5° at this time.

Although the distance of the solar sail from the Earth is large for imaging purposes, there are potential applications of real-time, low resolution images for continuous views of large scale polar weather systems along with Arctic ice and cloud coverage for weather prediction and global climate studies. The minimum ground resolution obtained from such imaging is shown in fig. 13 for a range of instrument apertures. For example, a 30 cm aperture instrument stationed at 3.9 million km from the Earth and operating at optical wavelengths provides a minimum ground resolution of order 9 km, assuming near diffraction limited optics. In practice though, the actual resolution obtained will be degraded due to factors such 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 performance. Space weather applications of these continuous polar orbits are also of interest for real-time imaging of the polar magnetosphere from out-of-the-ecliptic plane and for providing a viewing off-set to image Earth-bound coronal mass ejections away from the solar disk.

In addition to imaging missions, direct communication links with high latitude ground stations are enabled where geostationary satellites appear close to, or indeed below the local horizon. Such missions have been proposed by physicist Robert Forward as part of the related 'Statite' concept [17]. In this concept equilibrium locations about the L_2 point are envisaged, although these locations do not appear attractive for a realistic solar sail with non-perfect reflectivity. Since the path length to a solar sail high above the L_1 point is much greater than that to geostationary orbit, only moderate bandwidth links appear feasible for a small solar sail and ground segment. Such limited data links do however have a number of useful applications. One such application was investigated in some detail by the Canadian Dynacon Corporation for military and civil communication links to northern Canada [18]. A ground station at mid-latitudes would be used to up-link data to a transponder located high above the Arctic using a solar sail. The data would then be down-linked to a ground station in the Arctic for further dissemination to local users. Using a small solar sail delivered using a Pegasus launch vehicle, data rates of up to 1 Mbit s^{-1} could be sustained. The cost of the continuous link was estimated at less than half of the equivalent cost of procuring time on a commercial satellite constellation, or launching a dedicated series of conventional polar orbiters.

A similar communication scheme could be used to obtain continuous, real-time data from automated polar weather or science stations. Indeed, such a scheme may have applications for data returns from landers or surface rovers at the polar regions of Mars. While a polar orbiter relay satellite can also perform the same function, the orbiter is only visible at most once per orbit. A solar sail could be used to deliver the landers or rovers to Mars and then act as a transponder to enable continuous data returns. Lastly, from a vantage point high above the Arctic, a solar sail could also be utilised as a real-time solar physics platform. The solar sail would have continuous visibility of both the Sun and a single high latitude ground station for continuous data returns. This is in contrast to conventional Lagrange point solar physics missions which require a complex and costly network of at least three low latitude ground stations to ensure continual links to the satellite. The solar sail could be used as an instrument platform, or merely as a transponder for a number of conventional Lagrange point satellites [19].

3.2.3 Mission Orbit

The solar sail loading required to achieve the Polar Observer mission orbit can be determined from the three-body analysis used to evaluate the Geostorm mission artificial Lagrange point [14,16]. Both the Geostorm and Polar Observer orbits belong to the same connected family of equilibrium solutions in the vicinity of the L_1 Lagrange point. It will be assumed that the solar sail is non-ideal and has a realistic reflectivity of 0.85. Then, it is found that for a given solar sail loading there is a large surface attached to L_1 on which the solar sail will remain in equilibrium, with the gravitational, light pressure and centripetal forces all in balance. This surface extends sunward, providing the equilibrium location for Geostorm, and above the ecliptic plane, providing the equilibrium location for the Polar Observer mission. A section of the resulting surfaces of solar sail loading is shown in fig. 14. The Polar Observer solutions high above L_1 can be seen along with a second family of solutions near L_2. However, these solutions are not attractive for polar applications due to their restricted viewing geometry at the summer solstice. The surfaces S_1 and S_2 define the volume of space within which equilibrium is possible.

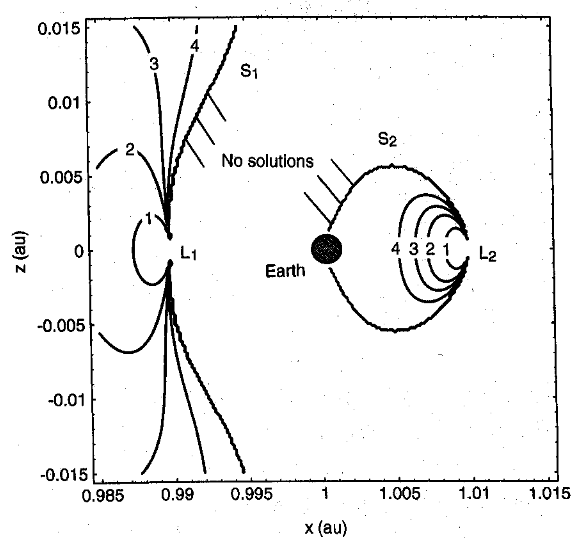
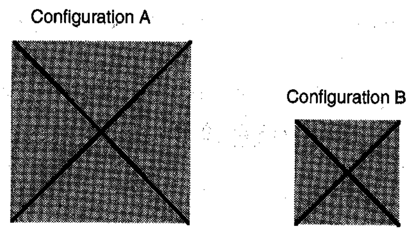


Fig. 14 Required solar sail loading for out-of-ecliptic equilibrium near the Sun-Earth L_1 point. Sail loading contours ($g\ m^{-2}$) (1) 76.5 (2) 38.25 (3) 25.50 (4) 15.30.

If a single instrument is to be delivered to the mission orbit, for example a CCD optical imager, the bus and payload may be limited to a mass of order 150 kg. Then, for a moderate sail assembly loading of 6.0 g m^{-2}, a 126 x 126 m square solar sail is required, as shown in fig. 15. A sail assembly loading (the loading of the sail and structure only) of 6.0 g m^{-2} requires a relatively thin sail film and a low mass boom structure, representing an improvement in technology from the Geostorm in-plane Lagrange point mission. The total spacecraft launch mass of 245 kg is however well within the Earth escape capacity of the Taurus or Athena family of launch vehicles. A larger payload with an array of imaging devices would of course require a larger solar sail and launch vehicle. Similarly, for the same mission orbit a small 50 kg payload would require a significantly smaller sail within the reach of the Pegasus launch vehicle, again shown in fig. 15.

It is appropriate to note here the recent interest in the Triana mission, to provide whole Earth imaging from the natural L_1 point, some 1.5 million km from the Earth. Studies have investigated a small L_1 spacecraft with a total mass of order 220 kg carrying a three colour CCD camera. A 25 cm telescope will provide a ground resolution of order 14 km, providing an Earth image every three minutes and requiring a down-link data rate of 50 kbit s^{-1}. Such mis-



Configuration	A	B
Payload mass (kg)	150	50
Assembly loading (gm^{-2})	6	6
Sail side (m)	126	73
Total mass (kg)	245	82
Launch vehicle	Athena II /Star 37V	Pegasus/Star 24C

Fig. 15 Scaled representation of two solar sail configurations for the Polar Observer mission (without contingency or launch adapter).

sions can exploit some of the advantages of the Polar Observer mission for continuous imaging of large geographical regions, for example weather systems and cloud cover, but from an in-plane location the poles are of course not in continual view.

4. FUTURE PROSPECTS

While solar sailing offers significant advantages for solar system exploration, it is not a general purpose propulsion system which can meet the requirements of every conceivable mission. Neither is it a technology worth developing for its own sake. While the notion of sailing through the solar system using nothing more than light pressure is a truly wonderful thought, solar sailing must be pulled forward by mission applications at the same time as it is pushed by technology development. Solar sailing proponents must therefore be practical, hard-headed romantics. This rule also holds true for initial flight tests of solar sailing. Unless such flight tests provide confidence in the technology and a clear path towards some enabling capability, they will not perform a useful function. The consequences of a deployment failure on a single orbital test may be profound. If the first true solar sail fails to deploy it may bury solar sailing for another twenty years.

Since the JPL studies of the mid-1970s technology has matured considerably. In particular the opportunities provided by micro-spacecraft technologies will be immense, allowing highly capable missions with only small solar sails. However, there are other developments which have yet to exploited. In particular there is much work to be done to devise new ways of fabricating ultra-thin deployable sail films. A conventional solar sail film requires a plastic substrate to allow handling and folding, typically between 1 and 7.5 μm thick. This substrate is then coated with a thin layer of Aluminium, typically 0.1 μm thick, to provide a highly reflective surface. However, for a well designed solar sail structure the stresses imposed on the sail film are low so that the plastic substrate is somewhat redundant after deployment. Since the substrate comprises most of the mass of the sail film, removal after deployment would enable truly high performance solar sails. While concepts exist, such as plastic substrates which will sublimate under the action of solar UV radiation, little experimental work has been undertaken. Recently though, experiments have demonstrated that Diamond like Carbon (DLC), a metastable form of Carbon, can be used as a UV sensitive buffer between the substrate and reflector. This scheme will in principle allow the substrate to detach shortly after deployment. Removable substrates are not the only means of improving solar sail performance. If the sail film is perforated with holes smaller than the mean wavelength of sunlight, the mass of the sail film is greatly reduced, but without significantly degrading its reflectivity. This is analogous to wire mesh reflectors used in radio frequency antennae. Again, the technology to fabricate such films is at hand since the semi-conductor industry uses similar techniques on a daily basis. A deployable, all-metal film with perforations would enable quite remarkable solar sails to be manufactured.

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 relies on the investigation of those niche missions and applications which are unique to solar sails. There are too many long term vested interests to expect solar sailing to be developed for non-optimum missions where existing technologies meet the challenge. The exception being cases where solar sails may lead to significant cost reductions, for example by allowing a smaller, lower cost launch vehicle to be used.

In the near term the main application of interest appears to be artificial Lagrange points displaced sunward of L_1 for solar storm warning. This is the Geostorm mission, described in detail in section 3.1, which is some way towards graduating from a study to a fully funded mission. Geostorm is an ideal first mission for solar sailing as it brings together several key elements into a single compelling concept. Firstly, only a modest solar sail is required with a characteristic acceleration of order 0.25 mm s^{-2}. Then, the mission application has a powerful group of users in the US military and Department of Energy. Advanced warning of solar storms will be used to protect satellite communication links and electricity distribution networks. It is important to note that these users are not at all interested in solar sailing per se, but only the unique mission application which it enables. There are lessons here for future missions. Lastly, the user group are insured against the use of an untested technology. If the solar sail fails to deploy, the mission can continue at the natural L_1 point sunward of the Earth. Because of this insurance Geostorm can in fact be used as a fast track to a first solar sail mission. If deployment fails, then the mission is only degraded, not lost.

Other than artificial Lagrange points, the near term applications which appear particularly attractive are the solar polar and Mercury orbiter missions. Both of these mission are confined to the inner solar system where the solar sail will experience enhanced acceleration. As such, the effective Δv and specific impulse delivered by the solar sail is quite immense. Either of these would be a spectacular demonstration of the mission enabling properties of solar sailing. While a limited Mercury orbiter mission is certainly possible in the near term using chemical propulsion with gravity assists, a close solar polar orbit is not.

5. ACKNOWLEDGEMENTS

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