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PERFORMANCE REQUIREMENTS FOR NEAR-TERM INTERPLANETARY SOLAR SAILCRAFT MISSIONS

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Solar sailcraft provide a wide range of opportunities for high-energy low-cost missions. To date, most mission studies require a rather demanding performance that will not be realized by solar sailcraft of the first generation. However, even with solar sailcraft of moderate performance, scientifically relevant missions are feasible. This is demonstrated with a Near Earth Asteroid sample return mission and various planetary rendezvous missions.

Introduction

Utilizing solely the freely available solar radiation pressure for propulsion, solar sailcraft provide a wide range of opportunities for low-cost interplanetary missions, many of which are difficult or impossible for any other type of conventional spacecraft due to their large Δv-requirement. Many of those high-energy missions are of great scientific relevance, such as missions to Mercury and to Near Earth Objects (asteroids and short period comets) with highly inclined or retrograde orbits[1]. Within the inner solar system (including the main asteroid belt) solar sailcraft are especially suited for multiple rendezvous and sample return missions due to their (at least in principle) unlimited Δv-capability. Even missions to the outer solar system may be enhanced by using solar sailcraft, albeit the solar radiation pressure decreases with the square of the sun–sail distance. For such missions solar sailcraft may gain a large amount of energy when first approaching the sun, thereby performing a so-called 'solar photonic assist' maneuver that turns the trajectory into a hyperbolic one [4][5][13]. Such trajectories allow reasonable transfer times to the outer planets (and to near interstellar space) without the need to perform any gravity assist maneuver. However, without the use of additional propulsive devices and/or an aerocapture maneuver at the target body, only fast fly-bys can be achieved due to the associated large hyperbolic excess velocities.

[1]More than 55% of the NEO population has inclinations larger than 10°, more than 30% has inclinations larger than 20°. Reaching such inclinations with spacecraft requires a very large Δv.

Several mission studies for high-energy interplanetary solar sailcraft missions have been carried out at DLR [4][5][8][9] and elsewhere [13][15]. Most of them require a rather demanding sailcraft performance to keep mission durations short (see Table 1). However, taking the current state-of-the-art in engineering of ultra-lightweight structures into account, solar sailcraft of the first generation will be of relatively moderate performance. For such near-term solar sailcraft few mission examples can be found in the literature.

The aim of this paper is to narrow down this gap and to get a lower bound on solar sailcraft performance for interplanetary missions that are under consideration. It will be shown, that challenging scientific missions are feasible at relatively low cost, even with moderate performance sailcraft of the first generation. This will be demonstrated below by the trajectory analysis of a proposed sample return mission to Near Earth Asteroid 1996FG$_3$ (mission duration approx. 9.4 years).

Solar Sailcraft Orbital Mechanics

The magnitude and direction of the solar radiation pressure (SRP) force acting on a flat solar sail due to the momentum transfer from solar photons is completely characterized by the sun-sail distance and the sail attitude. The latter is generally expressed by the sail normal vector $\mathbf{n}$, whose direction is usually described by the sail clock angle α and the sail cone angle β (Figure 1). Figure 2 gives a picture of the forces exerted on a flat and perfectly reflecting solar sail (ideal sail) of area A by the solar radiation pres-

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Target body	Sailcraft performance		Transfer time	References
	a_c [mm/s^2]	σ [g/m^2]	[yr]	
Mercury	0.5	16.0	1.4	[13]
Pluto (fly-by)	0.7	11.4	10.4	[4][5]
(4) Vesta	0.75	10.7	3.3	[4][8]
2P/Encke	0.85	9.4	3.0	[4]
21P/Giacobini-Zinner	1.0	8.0	6.8	[15]
Venus	1.0	8.0	0.6	[13]
Mars	1.0	8.0	1.0	[13]
(433) Eros	1.0	8.0	1.2	[13]
(1566) Icarus	1.25	6.4	1.2	[15]

Table 1: Fast solar sailcraft missions using advanced sailcraft (rendezvous, if not stated otherwise)

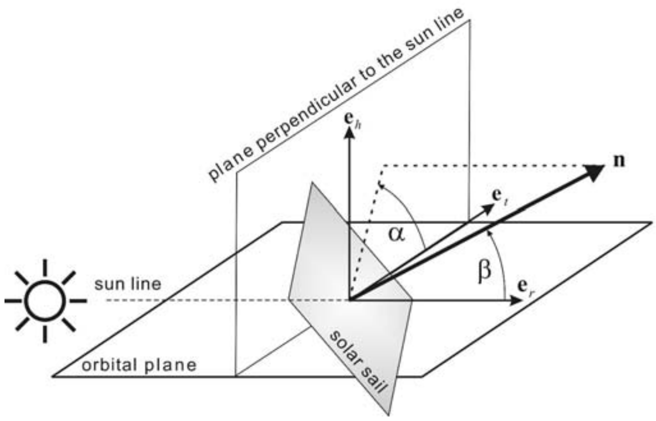

Figure 1: Definition of the sail clock angle α and the sail cone angle β

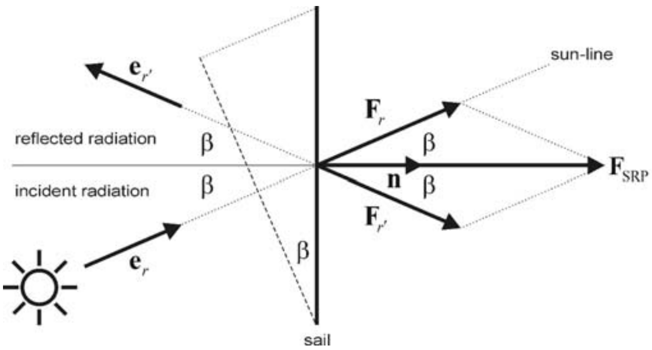

Figure 2: Perfect reflection

sure P acting on the sail's center of surface. From the geometry of Figure 2 the total SRP force $\mathbf{F}_{\mathrm{SRP}}$ can be easily calculated:

$$\begin{aligned}\mathbf{F}_r &= \ \ PA(\mathbf{e}_r \cdot \mathbf{n})\,\mathbf{e}_r \\ \mathbf{F}_{r'} &= -PA(\mathbf{e}_r \cdot \mathbf{n})\,\mathbf{e}_{r'}\end{aligned}$$

and making use of $\mathbf{e}_r - \mathbf{e}_{r'} = 2(\mathbf{e}_r \cdot \mathbf{n})\,\mathbf{n}$:

$$\begin{aligned}\mathbf{F}_{\mathrm{SRP}} &= \mathbf{F}_r + \mathbf{F}_{r'} = \\ &= 2PA(\mathbf{e}_r \cdot \mathbf{n})^2\,\mathbf{n} = \\ &= 2PA\cos^2\beta\,\mathbf{n}\end{aligned}$$

Thus, in the case of perfect reflection, the thrust force is always along the direction of the sail normal vector $\mathbf{n}$. At 1 AU, the solar radiation pressure is $(P_0)_{1\mathrm{AU}} \doteq 4.563 \cdot 10^{-6}\,\mathrm{N/m^2}$. Therefore, the effective pressure (force per unit area) acting on an ideal sail normal to the sun-line is twice the solar radiation pressure, $2(P_0)_{1\mathrm{AU}} \doteq 9.126 \cdot 10^{-6}\,\mathrm{N/m^2}$. However, a real solar sail is *not* a perfect reflector and a thorough trajectory analysis must take into account the optical properties of the real sail. Since in this case a small but significant fraction of the incoming sunlight is absorbed or reflected non-specularly, a tangential force component is acting on a real solar sail, so that $\mathbf{F}_{\mathrm{SRP}}$ is no longer along the direction of $\mathbf{n}$. However, for preliminary mission analysis this tangential force component may be neglected, since the resulting small angular deviation of $\mathbf{F}_{\mathrm{SRP}}$ from the sail normal can be compensated by the sail steering strategy for interplanetary transfer trajectories (where $\beta > 55.5°$ is not required [10]). Nonetheless, an overall sail efficiency parameter η should be used, which takes into account the reduced magnitude of $\mathbf{F}_{\mathrm{SRP}}$ due to the non-perfect reflectivity of the sail including its deflection/warping under load. Assuming a conservative sail efficiency of $\eta \approx 0.85$ (aluminum coated plastic film), we get

$$(P_{\mathrm{eff}})_{1\mathrm{AU}} = 2\eta(P_0)_{1\mathrm{AU}} \doteq 7.757 \cdot 10^{-6}\,\mathrm{N/m^2}$$

for the effective pressure acting at 1 AU on a solar sail that is oriented normal to the sun-line and

$$\mathbf{F}_{\mathrm{SRP}} = (P_{\mathrm{eff}})_{1\mathrm{AU}} \left(\frac{1\,\mathrm{AU}}{r}\right)^2 A\cos^2\beta\,\mathbf{n}$$

for the respective SRP force in a distance r from the sun. Thus, to experience a reasonable acceleration, solar sailcraft must be large and very lightweight.

The orbital dynamics of solar sailcraft is in many respects similar to the orbital dynamics of other spacecraft, where a small continuous thrust is applied to modify the spacecraft's orbit over an extended period of time. However, other continuous thrust spacecraft may orient its thrust vector in any desired direction and vary its thrust level within a wide range, whereas the thrust vector of solar sailcraft is constrained to lie on the surface of a 'bubble' directed away from the sun (see Figure 3). Nevertheless, by controlling the sail orientation relative to the sun, solar sailcraft can *gain* orbital angular momentum (if $\mathbf{F}_{\mathrm{SRP}} \cdot \mathbf{e}_t > 0$) and spiral outwards – away from the sun – or *lose* orbital angular momentum (if $\mathbf{F}_{\mathrm{SRP}} \cdot \mathbf{e}_t < 0$) and spiral inwards – towards the sun.

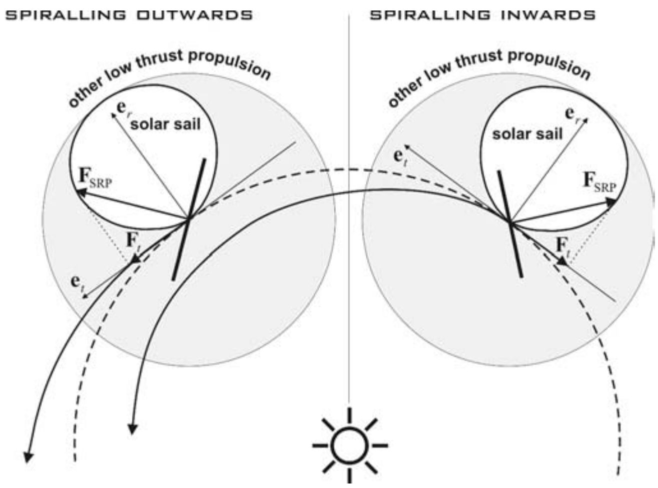

Figure 3: Spiralling inwards and outwards

Solar Sailcraft Performance Parameters

Before talking about performance of near-term solar sailcraft, the most common performance definitions should be given. The performance of solar sailcraft may be expressed by the following parameters:

- the sail assembly loading

$$\sigma_s = \frac{m_s}{A}$$

is defined as the mass of the sail assembly (the sail film and the required structure for storing, deploying and tensioning the sail, index 's') per unit area. Thus, the sail assembly loading is the key parameter for the performance of a solar sail and the efficiency of its structural design.

- the sailcraft loading

$$\sigma = \frac{m}{A} = \frac{m_s + m_p}{A} = \sigma_s + \frac{m_p}{A}$$

is defined accordingly as the specific mass of sailcraft including the payload (index 'p'). It should be noted, that the term payload stands for the total sailcraft except the solar sail assembly (i.e. except the propulsion system).

- the characteristic acceleration a_c is defined as the maximum acceleration at 1 AU solar distance. It can be calculated via

$$(P_{\mathrm{eff}})_{1\mathrm{AU}} A = m a_c = \sigma A a_c = = (\sigma_s + \frac{m_p}{A}) A a_c$$

$$\Rightarrow \quad a_c = \frac{(P_{\mathrm{eff}})_{1\mathrm{AU}}}{\sigma_s + \frac{m_p}{A}}$$

Using the characteristic acceleration, the SRP force acting on the sail can be written as

$$\mathbf{F}_{\mathrm{SRP}} = m a_c \left(\frac{1\,\mathrm{AU}}{r}\right)^2 \cos^2\beta\,\mathbf{n}$$

- the lightness number λ, which is independent from solar distance, is defined as the ratio of the SRP acceleration experienced by a solar sail normal to the sun line and the solar gravitational acceleration ($5.93\,\mathrm{mm/s^2}$ at 1 AU)

$$\lambda = \frac{a_c}{5.93\,\mathrm{mm/s^2}}$$

Using the lightness number, the SRP force acting on the sail can be written as

$$\mathbf{F}_{\mathrm{SRP}} = \lambda \frac{\mu m}{r^2} \cos^2\beta\,\mathbf{n}$$

where $\mu = GM_{\mathrm{sun}}$.

DLR Ground-based Demonstration of Solar Sail Technology

In December 1999 a ground-based demonstration of solar sailcraft technology was performed at the German Aerospace Center (DLR) at Cologne, where a $20\,\mathrm{m} \times 20\,\mathrm{m}$ solar sail was successfully deployed in a simulated zero-g environment and ambient environmental conditions (Figure 4) [6][14].

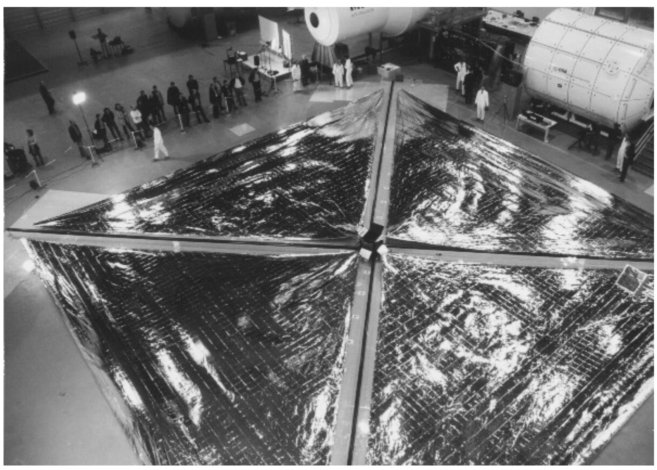

Figure 4: Fully deployed $20\,\mathrm{m}\times 20\,\mathrm{m}$ solar sail at DLR

The square solar sail consisted of four CFRP (Carbon Fiber Reinforced Plastics) booms with a specific mass of 101 g/m and of four triangular sail segments made of aluminum-coated ($0.1\,\mu$m) plastic films with a thickness between 4 and 12 μm. The booms consisted of two CFRP shells that were bonded at the edges to form a tubular shape, so that they can be pressed flat and rolled up (Figure 5).

Figure 5: DLR deployable CFRP boom

The booms were rolled up in a 60 cm × 60 cm × 65 cm-sized deployment module, from where they unfolded automatically. After deployment they returned to their tubular shape with high bending and buckling strength. Subsequently, the four sail segments were deployed by ropes. To assess the handling behavior of different sail materials, the sail segments were made of three different aluminum-coated plastic films, 12 μm polyethylene terephtalate (PET, Mylar®), 7.5 μm polyimide (PI, Kapton®) and 4 μm polyethylene naphthalate (PEN). All segments were reinforced along the three edges of the triangle to prevent rips. The specific mass of the sail film was 18.9 g/m^2 for the Mylar®-segment, 12.4 g/m^2 for the Kapton®-segment and 10.5 g/m^2 for the PEN-segment. The deployment module and the cross section of the booms for this ground-based demonstration were dimensioned for a 40 m × 40 m solar sail, which was too large for an in-door demonstration. For the structural sizing of the booms two load cases were considered, bending – due to the SRP force – and buckling – due to sail deployment and sail tensioning forces. According to FEM (Finite Element Method) calculations, similar booms could be used also for larger sails [7].

DLR Mission Proposal for ENEAS

Near Earth Asteroids (NEAs) are a promising category of target bodies for a first solar sailcraft mission, since they can be accessed relatively easily and since they are of great scientific interest. Therefore, in August 2000, a dedicated mission for the exploration of NEAs with solar sailcraft (ENEAS) was proposed by DLR in cooperation with the Westfälische Wilhelms-Universität at Münster (Germany) as a candidate within the German small satellite program for extraterrestric sciences [3][14]. Based on the successful deployment experiment described above, ENEAS (Figure 6) was intended to feature a deployable 50 m × 50 m solar sail that would be capable to transport a micro-satellite with a mass of 65.5 kg to a NEA within less than five years. Table 2 summarizes the ENEAS parameters.

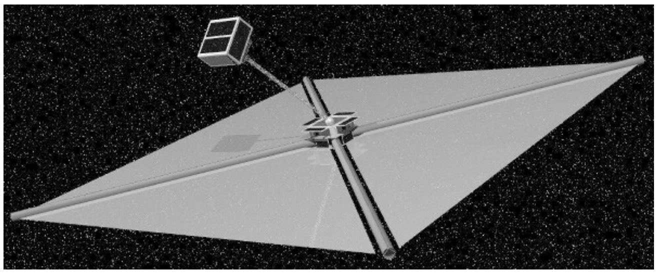

Figure 6: DLR ENEAS solar sailcraft with deployed control mast (artist's view)

Sail area	A	$(50\,\mathrm{m})^2$
Sail assembly loading	σ_s	29.2 g/m^2
Sail assembly mass	m_s	73 kg
Payload mass	m_p	65.5 kg
Total sailcraft mass	m	138.5 kg
Sailcraft loading	σ	55.4 g/m^2
Lightness number	λ	1/42.4
Characteristic acceleration	a_c	0.14 mm/s^2
Characteristic SRP force	$F_{\mathrm{SRP},c}$	19.5 mN

Table 2: Parameters for the ENEAS solar sailcraft

For propulsionless attitude control, the solar sail and the micro-satellite would be separated by a commercially available 10 m collapsible control mast, which is housed inside the deployment module in its stowed configuration. This control mast is attached to the deployment module via a two degree of freedom actuator gimbal, which allows to rotate the mast including the attached micro-satellite with respect to the sail (Figure 6). Thus, by rotating the control mast, the center of mass (CM) can be offset from the center of pressure (CP). The resulting external torque may be used to rotate the sail about any CM-intersecting axis parallel to the sail plane (this attitude control concept was originally proposed by [1]).

1996FG$_3$ was chosen as the target body for the ENEAS mission, since 1996FG$_3$ has orbital elements not too different from that of Earth and since it is an object of exceptional scientific interest. Observations indicate that 1996FG$_3$ is a binary asteroid, consisting of a central body with a rotation period of about 3.60 hours and a satellite with an orbital period of about 16.15 hours. The determined average bulk density is $1.4 \pm 0.3\,\mathrm{g/cm}^3$ which is highly suggestive of a 'rubble pile' structure [12]. ENEAS is intended to determine the physical properties and the evolution of the 1996FG$_3$ system.

Trajectory optimization using the calculus of variations revealed, that the ENEAS sailcraft can reach 1996FG$_3$ in 4.5 years (1640 days), if it is inserted directly into an interplanetary trajectory with a hyperbolic excess energy of $C_3 = 4\,\mathrm{km^2/s^2}$. However, more recently performed trajectory optimization, based on artificial neural networks and evolutionary algorithms produced a better trajectory for the same launch date, which is closer to the (unknown) global optimum (Figure 7) [2].

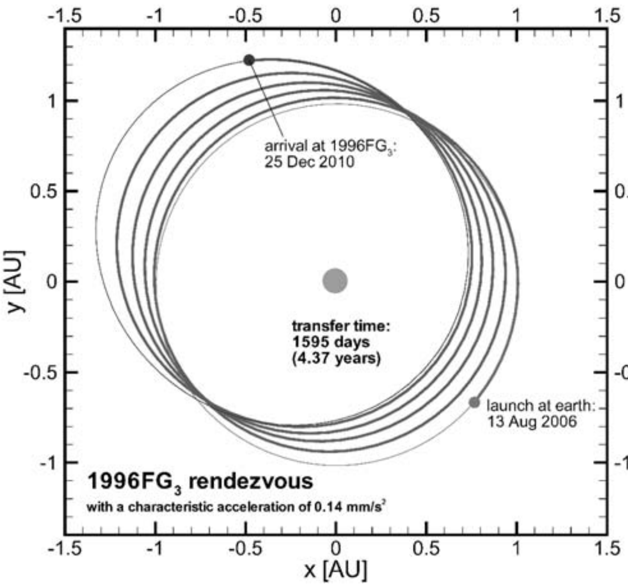

Figure 7: 1996FG$_3$ rendezvous trajectory for $a_c = 0.14\,\mathrm{mm/s^2}$

The flight time could be reduced by 45 days (3%), reducing at the same time the C_3 requirement from $4\,\mathrm{km^2/s^2}$ to $0\,\mathrm{km^2/s^2}$, thus permitting a reduction of launch cost. The accuracy of the trajectory generated by the artificial neural network is $\Delta r < 11\,000\,\mathrm{km}$ for the relative distance to the target body at rendezvous and $\Delta v < 43\,\mathrm{m/s}$ for the relative velocity (even without performing a local fine tuning of the trajectory) [2].

Near-Term Solar Sailcraft Performance

Looking at the equation for the characteristic acceleration of solar sailcraft with a square sail,

$$a_c = \frac{(P_{\mathrm{eff}})_{1\mathrm{AU}}}{\sigma_s + \frac{m_p}{s^2}},$$

one can see that the performance depends on three design parameters, the sail assembly loading σ_s, the payload mass m_p and the side length s (or area s^2) of the solar sail, defining a three-dimensional solar sail design space. Figures 8 and 9 show parametric sections of this design space for a fixed $\sigma_s = 29.2\,\mathrm{g/m^2}$ and a fixed $s = 50\,\mathrm{m}$ respectively (as for the ENEAS sailcraft). As can be seen from the diagram in Figure 8, a characteristic acceleration of up to $0.265\,\mathrm{mm/s^2}$ can be achieved without any payload. For a smaller a_c a positive payload mass can be accommodated, depending on the sail size. To achieve a characteristic acceleration beyond $0.265\,\mathrm{mm/s^2}$, the sail assembly loading has to be further reduced (Figure 9).

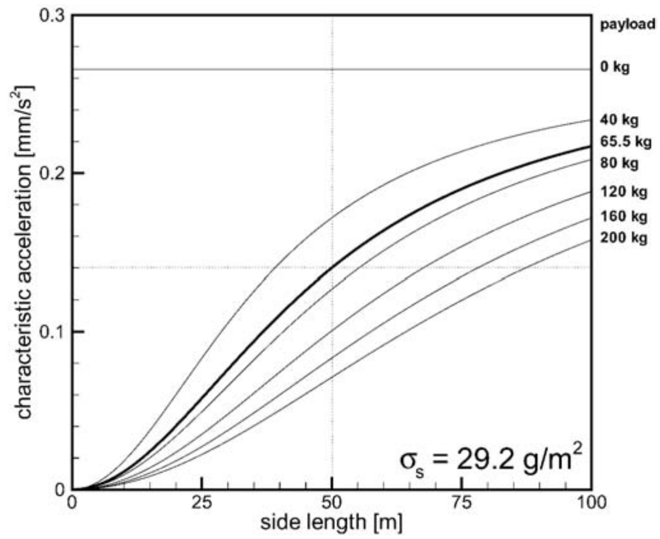

Figure 8: The characteristic acceleration a_c as a function of s and m_p for $\sigma_s = 29.2\,\mathrm{g/m^2}$

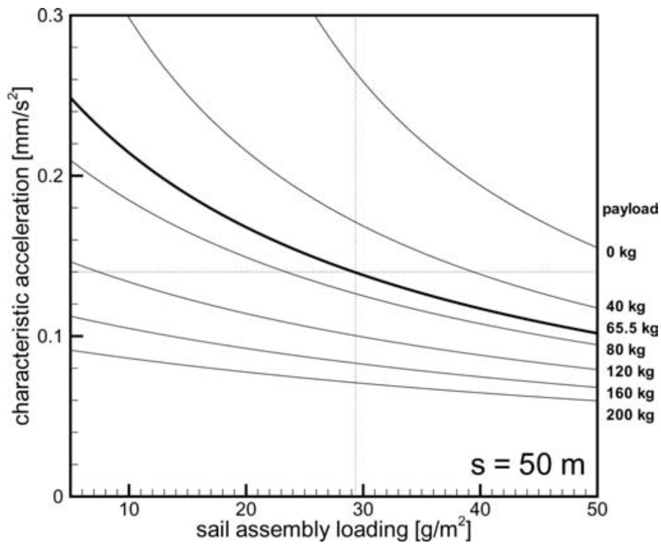

Figure 9: The characteristic acceleration a_c as a function of σ_s and m_p for $s = 50\,\mathrm{m}$

By different combinations of the three design parameters any desired characteristic acceleration can be achieved[2]. An increase in payload mass can, for example, be offset with a proportional increase of s^2 or with a (not inversely proportional) decrease of σ_s.

[2] It should be noted that m_p and s can be chosen independently, whereas $\sigma_s(s)$ is a function of s with $\partial\sigma_s/\partial s < 0$, since the mass of the booms and the deployment module scale less than linearly with the sail area. However, we are on the safe side, when we assume $\partial\sigma_s/\partial s = 0$ to keep calculations simple.

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Those design sensitivities can be determined quantitatively using sensitivity functions, which provide an indication of the relative importance of each design parameter for a given point in the solar sail design space [10]. The sensitivity function for any design parameter $\nu \in \{\sigma_s, m_p, s\}$ may be written as

$$\frac{\Delta a_c}{a_c} = \Lambda_\nu \frac{\Delta \nu}{\nu}$$

with

$$\Lambda_{\sigma_s} = -\frac{1}{1 + m_p/\sigma_s s^2}$$
$$\Lambda_{m_p} = -\frac{1}{1 + \sigma_s s^2/m_p}$$
$$\Lambda_s = \frac{2}{1 + \sigma_s s^2/m_p}$$

For the ENEAS sailcraft, we have $\Lambda_{\sigma_s} = -0.537$, $\Lambda_{m_p} = -0.473$ and $\Lambda_s = +0.946$. As can be seen, the side length of the sail is the most critical parameter with respect to the ENEAS sailcraft performance. Thus, an increase in performance is best done by increasing the size of the solar sail.

If costs can be described by (known or estimated) functions of the three design parameters, then the optimum (cost minimal) sailcraft design for a given performance can be determined.

ENEAS with Sample Return

The ENEAS sailcraft was intended to rendezvous $1996FG_3$ for remote sensing with a minimum scientific payload mass of 5 kg (CCD camera + IR spectrometer + magnetometer). To study the $1996FG_3$ system in more detail, it would be necessary to place a lander on the surface of the asteroid (e.g. for mass spectrometry and/or alpha-proton spectrometry). Some investigations (e.g. micro-structure and isotope analysis) to determine the age and the evolution of $1996FG_3$ could be achieved only by taking samples of the asteroid back to Earth. Due to their unlimited Δv-capability, solar sailcraft are especially capable to perform such sample return missions. However, compared to the ENEAS rendezvous mission, the payload mass has to be increased considerably. The key questions for the ENEAS-SR (sample return) mission design are:

Q1: What is the maximum acceptable mission duration $T_{\max}$?

Q2: What is the minimum characteristic acceleration $a_{c,\min}$ to perform the mission in $T_{\max}$?

Q3: What is the expected sail assembly loading σ_s and sail dimension s for near-term solar sailcraft?

Q4: What is the maximum payload mass to get $a_{c,\min}$ for the specified σ_s and s?

Answer to Q1: At present, the maximum acceptable mission duration seems to be determined by the trip time required with chemical propulsion, including (eventually multiple) gravity assist maneuvers. Due to the relatively large Δv-requirement of about $6 - 10$ km/s for a mission comparable to ENEAS-SR, but with chemical propulsion, such a mission would require either an expensive launch vehicle and heavy spacecraft, resulting in a short trip time of a few years, or several gravity assists, resulting in a long trip time[3]. Since our approach aims at low-cost missions, only the gravity assist option seems to be a reasonable conventional alternative. Thus, for the ENEAS-SR mission we assume a total mission duration of more than ten years as not acceptable.

Answer to Q2: Trajectory calculations show, that an ENEAS-SR mission to $1996FG_3$ can be achieved even with a characteristic acceleration of 0.10 mm/s^2 in 9.40 years, including a rendezvous trajectory of 6.27 years (2290 days, Figure 10), 340 days of operations at the asteroid and an Earth return trajectory of 2.20 years (805 days, Figure 11).

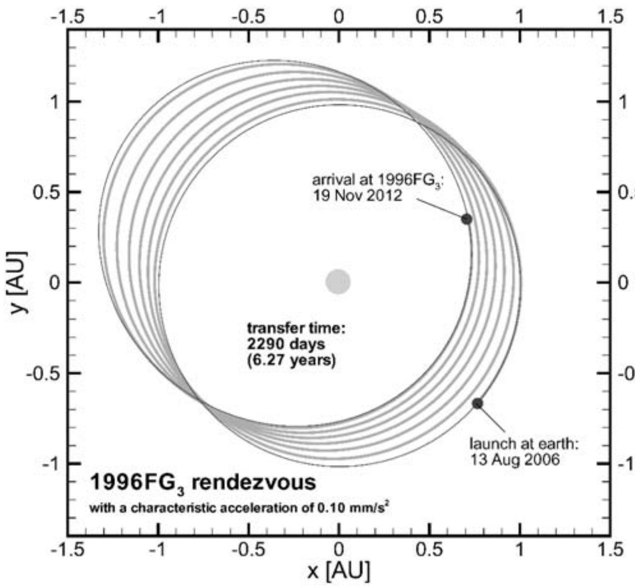

Figure 10: $1996FG_3$ rendezvous trajectory for $a_c = 0.10$ mm/s^2

Answer to Q3: The diagram in Figure 12 shows the required sail size for different sail assembly loadings and payload masses, to obtain a characteristic acceleration of 0.10 mm/s^2. Based on the experiences with the ground-based solar sail technology demonstration described above, we consider a maximum sail size of 70 m × 70 m with a sail assembly loading of 29.2 g/m^2 (sail film + booms + deployment module) as a realistic – however still challenging – baseline for the ENEAS-SR mission.

Answer to Q4: The specified σ_s and s yield a payload mass of 237 kg to get a characteristic acceleration

[3] similar to the Rosetta mission to comet 46P/Wirtanen, which will have three intermediate gravity assist maneuvers (Mars-Earth-Earth) and a trip time of approximately nine years

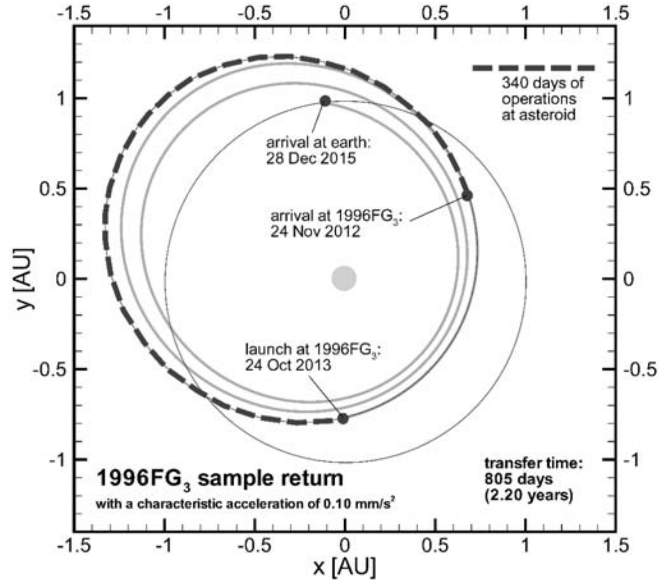

Figure 11: 1996FG$_3$ sample return trajectory for $a_c = 0.10\,\mathrm{mm/s^2}$

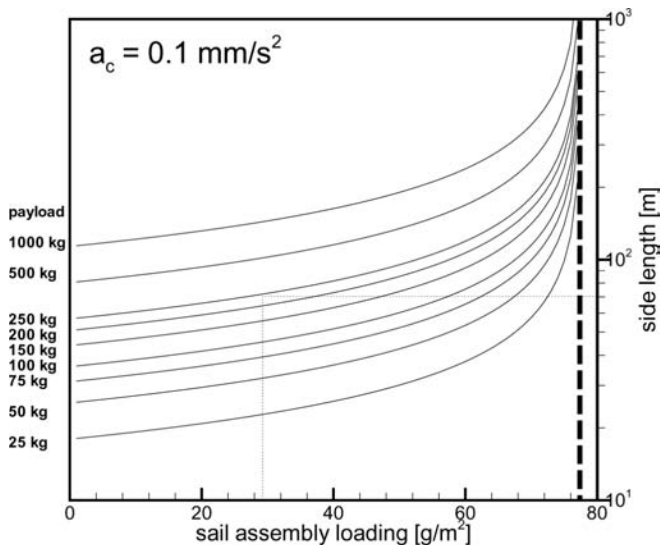

Figure 12: The side length s of the solar sail that is required to achieve a characteristic acceleration of $0.10\,\mathrm{mm/s^2}$ as a function of σ_s and m_p

of $0.10\,\mathrm{mm/s^2}$. Current research at our department indicates that it should be possible to realize such a mission within the specified mass budget, including a lander of about 140 kg and a sample return capsule of about 40 kg. Table 3 summarizes the ENEAS-SR parameters.

ENEAS-SR Mission Scenario

Since for solar sailcraft of moderate performance gaining orbital energy in the Earth's gravitational field is difficult and time consuming, the launcher will insert the ENEAS-SR solar sailcraft directly into an interplanetary trajectory with a hyperbolic excess energy of $C_3 = 0\,\mathrm{km^2/s^2}$. After the injection, the sail and

Sail area	A	$(70\,\mathrm{m})^2$
Sail assembly loading	σ_s	$29.2\,\mathrm{g/m^2}$
Sail assembly mass	m_s	143 kg
Payload mass	m_p	237 kg
Total sailcraft mass	m	380 kg
Sailcraft loading	σ	$77.6\,\mathrm{g/m^2}$
Lightness number	λ	1/59.3
Characteristic acceleration	a_c	$0.10\,\mathrm{mm/s^2}$
Characteristic SRP force	$F_{\mathrm{SRP},c}$	38.0 mN

Table 3: Parameters for the ENEAS-SR solar sailcraft

the attitude control mast are deployed in a 3-axis stabilized mode. Then the sail is oriented to follow a pre-calculated attitude profile, leading to an optimal interplanetary transfer trajectory. During the transfer, the ENEAS-SR solar sailcraft will run almost autonomously, so that ground monitoring will be carried out on a weekly basis only. At the end of the transfer trajectory the solar sailcraft will be making a rendezvous with 1996FG$_3$ within its gravitational sphere of influence (Hill-sphere) of between 70 km radius (at perihelion) and 150 km radius (at aphelion). Even in the near-field of the asteroid, the SRP acceleration of between $0.05\,\mathrm{mm/s^2}$ (at aphelion) and $0.21\,\mathrm{mm/s^2}$ (at perihelion) is larger than the asteroid's gravitational acceleration (0.01 to $0.00005\,\mathrm{mm/s^2}$ in a distance ranging from 5 to 50 km), so that the sailcraft is able to hover on an artificial equilibrium surface in the hemisphere that is opposite to the sun (Figure 13).

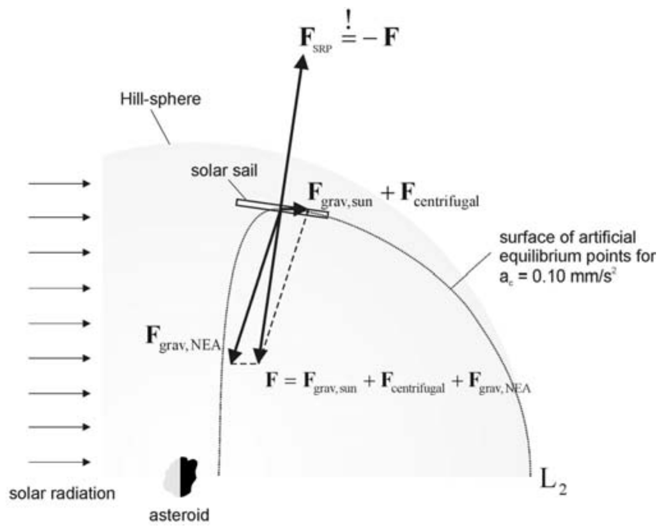

Figure 13: Hovering at the asteroid

Those quasi-stationary hovering positions are unstable but can be stabilized using a feedback control loop to sail attitude alone [11]. Hovering near the asteroid, the (likely complex) gravitational field of the target body is studied, so that a coarse gravitational field model can be determined. Thereafter, the lander with the Earth return capsule is separated from

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the solar sail to go into closer orbit about the asteroid. While measuring the asteroid's gravitational field with increasing accuracy, the orbit of the lander is continuously lowered until a safe landing trajectory can be computed (some or all of those extensive computations may be performed on Earth). Once landed, the sample is fed directly into the Earth return capsule and brought back to the hovering sailcraft. In this mission phase, the sailcraft is waiting edge-on (so that no SRP force is acting on the sail) at the L2 Lagrange point for the lander in order to assist the rendezvous. The lander design, the sample extraction mechanisms and the subsystems required to rendezvous the waiting sailcraft require further studies and are beyond the scope of this paper. Since 1996FG$_3$ is a binary system, it would be interesting to land and extract samples from both bodies to investigate the origin and the collisional evolution of the 1996FG$_3$ system. Since the gravitational acceleration is very low near the asteroid and the required Δv for the lander less than 10 m/s, a cold gas system with a propellant mass of less than 4 kg will suffice to perform all operations. After rendezvous with the hovering sailcraft, the re-docked ENEAS-SR solar sailcraft returns the sample to Earth. The return trajectory is much faster than the transfer trajectory to 1996FG$_3$ since no rendezvous is required at Earth. Thus, the sailcraft may arrive with a relatively large hyperbolic excess velocity of about 8.4 km/s. The gravitational acceleration of Earth adds another 11.2 km/s, so that the Earth reentry velocity may reach about $\sqrt{8.4^2 + 11.2^2}\,\mathrm{km/s} = 14.0\,\mathrm{km/s}$. Finally, just before the arrival of the ENEAS-SR solar sailcraft at Earth, the return capsule is separated from the lander and injected into an Earth reentry trajectory, where it is decelerated by atmospheric friction and breaking parachutes.

Other Promising Missions for Near-Term Solar Sailcraft

We have investigated the performance of near-term solar sailcraft also for rendezvous missions with celestial bodies other than Near Earth Objects. Table 4 gives the minimum rendezvous times for solar sailcraft with a characteristic acceleration of 0.10 to 0.20 mm/s^2 for several target bodies. It shows that even with near-term solar sailcraft planetary rendezvous mission are feasible within the inner solar system, if relatively long trip times can be tolerated. However, a characteristic acceleration of 0.10 mm/s^2 seems to be a lower bound for rendezvous missions within the inner solar system.

Target body	$\Delta v_{\min}$ [km/s]	Transfer time [yr] for a_c [mm/s^2]		
		0.10	0.15	0.20
Mercury	20.1	8.3	5.9	4.2
Venus	6.8	4.6	2.9	2.0
Mars	6.2	9.2	7.5	5.1

Table 4: Minimum transfer times to the inner planets for solar sailcraft with a characteristic acceleration of 0.10, 0.15 and 0.20 mm/s^2 ($\Delta v_{\min}$ denotes the minimum impulsive Δv-values for elliptical non-coplanar Hohmann-like orbit transfers with zero hyperbolic excess velocities at both ends of the trajectory)

Summary

We have investigated the minimum solar sailcraft performance requirements for various interplanetary missions. We were able to show, that the characteristic acceleration must be at least 0.10 mm/s^2 in order to avoid unacceptable long mission durations even for relatively easily accessible inner solar system bodies. A 70 m × 70 m solar sail with a sail assembly loading of 29.2 g/m^2 (sail film + booms + deployment module) was considered to be a realistic – however still challenging – near-term baseline. With this solar sail, a characteristic thrust of 38 mN can be achieved. The characteristic acceleration – defining the mission duration – depends on the actual payload mass m_p and ranges from 0.265 mm/s^2 ($m_p = 0$ kg) to 0.10 mm/s^2 ($m_p = 237$ kg). We have also demonstrated, that a sample return mission to a Near Earth Asteroid with such a solar sail is feasible within a mission duration of approx. 9.4 years. In addition to the scientific value of such a mission, the demonstration of the technical capabilities of solar sail propulsion in deep space would be a central objective.

References

[1] F. Angrilli and S. Bortolami. Attitude and orbital modelling of solar-sail spacecraft. *ESA Journal*, 14:431–446, 1990.

[2] B. Dachwald and W. Seboldt. Optimization of interplanetary rendezvous trajectories for solar sailcraft using a neurocontroller. Monterey, August 2002. AIAA/AAS Astrodynamics Specialist Conference. AIAA-2002-4989, in preparation.

[3] E. K. Jessberger and W. Seboldt et al. ENEAS – exploration of near-earth asteroids with a sailcraft. Proposal for a Small Satellite Mission within the Extraterrestric Sciences Program of DLR, August 2000.

[4] M. Leipold. *Solar Sail Mission Design*. Doctoral thesis, Lehrstuhl für Flugmechanik und Flugregelung; Technische Universität München, 1999. DLR-FB-2000-22.

[5] M. Leipold. To the sun and pluto with solar sails and micro-sciencecraft. *Acta Astronautica*, 45(4-9):549–555, 1999.

[6] M. Leipold, M. Eiden, C. E. Garner, L. Herbeck, D. Kassing, T. Niederstadt, T. Krüger, G. Pagel, M. Rezazad, H. Rozemeijer, W. Seboldt, C. Schöppinger, C. Sickinger, and W. Unkenbold. Solar sail technology development and demonstration. Laurel, 2000. 4th IAA International Conference on Low-Cost Planetary Missions. IAA-L-0707.

[7] M. Leipold, C. E. Garner, R. Freeland, A. Herrmann, M. Noca, G. Pagel, W. Seboldt, G. Sprague, and W. Unckenbold. ODISSEE – a proposal for demonstration of a solar sail in earth orbit. *Acta Astronautica*, 45(4-9):557–566, 1999.

[8] M. Leipold, E. Pfeiffer, P. Groepper, M. Eiden, W. Seboldt, L. Herbeck, and W. Unkenbold. Solar sail technology for advanced space science missions. Toulouse, 2001. 52nd International Astronautical Congress. IAF-01-S.6.10.

[9] M. Leipold, W. Seboldt, S. Lingner, E. Borg, A. Herrmann, A. Pabsch, O. Wagner, and J. Brückner. Mercury sun-synchronous polar orbiter with a solar sail. *Acta Astronautica*, 39(1-4):143–151, 1996.

[10] C. R. McInnes. *Solar Sailing. Technology, Dynamics and Mission Applications.* Springer–Praxis Series in Space Science and Technology. Springer–Praxis, Berlin, Heidelberg, New York, Chicester, 1999.

[11] E. Morrow, D. J. Scheeres, and D. Lubin. Solar sail orbit operations at asteroids. *Journal of Spacecraft and Rockets*, 38:279–286, 2001.

[12] S. Mottola and F. Lahulla. Mutual eclipse events in asteroidal binary system 1996FG$_3$: Observations and a numerical model. *Icarus*, 146:556–567, 2000.

[13] C. G. Sauer. Optimum solar-sail interplanetary trajectories. San Diego, August 2000. AIAA/AAS Astrodynamics Conference. 76-792.

[14] W. Seboldt, M. Leipold, M. Rezazad, L. Herbeck, W. Unkenbold, D. Kassing, and M. Eiden. Ground-based demonstration of solar sail technology. Rio de Janeiro, 2000. 51st International Astronautical Congress. IAF-00-S.6.11.

[15] J. L. Wright. *Space Sailing.* Gordon and Breach Science Publishers, Philadelphia, 1992.