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MULTIPLE RENDEZVOUS AND SAMPLE RETURN MISSIONS TO NEAR-EARTH OBJECTS USING SOLAR SAILCRAFT

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ABSTRACT

Solar sailcraft are due to their high ΔV-capability especially capable to perform multiple rendezvous and sample return missions to near-Earth objects. Even with moderate-performance solar sails of the first generation, challenging scientific missions are feasible at relatively low cost. Within this paper, it will be shown that a 300 kg-spacecraft (including a lander and a sample return capsule), propelled by a $(70\,\mathrm{m})^2$ solar sail with an additional mass of about 110 kg (specific weight $23\,\mathrm{g/m^2}$), is capable to return a sample from a near-Earth asteroid to Earth within 10 years from launch. In another scenario, a solar sail of the same size and weight is able to propel a 75 kg-spacecraft to rendezvous three near-Earth asteroids subsequently within 7.6 years from launch (with about 200 days of operations at each asteroid). A larger solar sail of about $(140\,\mathrm{m})^2$ would even be capable to transport a spacecraft that returns samples from the three near-Earth asteroids to Earth within about 10 years from launch.

1. INTRODUCTION

Being propelled solely by the freely available solar radiation pressure, solar sailcraft provide a wide range of opportunities for low-cost missions, many of which are – due to their large ΔV-requirement – difficult or impossible for any other type of conventional spacecraft. Many of those high-energy missions are of great scientific relevance, such as missions to near-Earth objects (NEOs, asteroids and short period comets) with highly inclined or retrograde orbits.[1] Such missions can be performed with solar sails, but a demanding sail performance (low specific weight) is required to keep mission durations within moderate limits. Taking, however, 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, if assessed w.r.t. flight duration alone, even inferior to already state-of-the-art electric propulsion systems (although solar sailcraft might have an advantage in launch mass). Nevertheless, on the way to more advanced solar sailcraft, they are an indispensable first stepping stone. Near-Earth asteroids (NEAs) with a moderate inclination are a promising category of target bodies for near-term solar sailcraft, since they can be accessed relatively easily. 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 space sciences. Using a $(50\,\mathrm{m})^2$ square sail with a specific weight of about $30\,\mathrm{g/m^2}$, ENEAS was intended to rendezvous a NEA ($1996FG_3$) with a total payload of about 75 kg (incl. spacecraft bus and 5 kg scientific payload) within less than 5 years [1, 2]. Based on our experience gained during the successful ground deployment of a $(20\,\mathrm{m})^2$ solar sail structure [2, 3, 4], we consider a $(70\,\mathrm{m})^2$ solar sail with a specific weight of about $20\,\mathrm{g/m^2}$ to be a realistic, however still ambitious, near-term development goal.

Within this paper, it will be demonstrated that even with moderate-performance solar sails of the first generation, challenging scientific missions, like a sample return mission (ENEAS-SR) or a multiple rendezvous mission (ENEAS+) to near-Earth asteroids, are feasible at relatively low cost. It will be shown that, using a $(70\,\mathrm{m})^2$ solar sail with a specific weight of $23\,\mathrm{g/m^2}$, ENEAS-SR would be able to return a sample from the C type NEA $1996FG_3$ to Earth within 10 years from launch. The payload (the total system except the sail assembly) for this mission is about 300 kg, including a lander and a sample return capsule. With a 75 kg-payload, a solar sail of the same size and weight would also be able to rendezvous three NEAs ($2000AG_6$, 1989UQ, and $1999AO_{10}$) subsequently within 7.6 years from launch (with about 200 days of operations at each NEA). A larger solar sail of about $(140\,\mathrm{m})^2$ would even be capable to transport the ENEAS-SR payload to the three ENEAS+ target bodies and sample all of them. The samples could be returned to Earth within about 10 years from launch.

[1] more than 55% of the NEO population has inclinations larger than 10°, more than 30% has inclinations larger than 20°

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2. SOLAR SAILCRAFT

For the optical characteristics of a solar sail, different assumptions can be made, which result in different models for the magnitude and the direction of the solar radiation pressure (SRP) force acting on the sail. Simple models assume an ideally reflecting sail surface or model its non-ideal reflectivity by introducing an overall sail efficiency factor. It is shown in [5] that those SRP force models should *not* be used except for *very* preliminary mission feasibility analyses, since they misrepresent the direction of the real SRP force. Within this paper, a more realistic SRP force model is used for trajectory simulation and optimization that considers the optical characteristics of the real sail film, which is aluminum-coated on the front side[2] and chromium-coated on the back side[3]. Using the optical parameters for an Al|Cr-coated sail, it can be shown (cf. [6, 7]) that, as Figure 1 illustrates, the SRP force acting on the sail has a component perpendicular to the sail surface

$$F_n = PAQ_n(\beta)\cos\beta \tag{1}$$

and a component parallel to the sail surface

$$F_t = PAQ_t(\beta)\cos\beta \tag{2}$$

where

$$Q_n(\beta) = 1.8272\cos\beta - 0.010888 \tag{3}$$
$$Q_t(\beta) = 0.1728\sin\beta \tag{4}$$

and where P is the solar radiation pressure, A is the sail area, and β is the light incidence angle. Introducing a unit vector $\mathbf{f}$ that is always along the total SRP force direction, the SRP force acting on the sail may be written as

$$\mathbf{F}_{\text{SRP}} = \sqrt{F_n^2 + F_t^2}\,\mathbf{f} \tag{5}$$

and, by defining

$$Q(\beta) = \frac{1}{2}\sqrt{Q_n^2(\beta) + Q_t^2(\beta)}\cos\beta \tag{6}$$

as

$$\mathbf{F}_{\text{SRP}} = 2PAQ(\beta)\,\mathbf{f} \tag{7}$$

where $Q(\beta)$ depends only on the light incidence angle β and the optical parameters of the sail film. The angle between the Sun-line and $\mathbf{f}$ is

$$\delta = \beta - \arctan\frac{Q_t(\beta)}{Q_n(\beta)} \tag{8}$$

The orbital dynamics of solar sailcraft is in many respects similar to the orbital dynamics of other low-thrust spacecraft. However, as Figure 2 shows, other low-thrust spacecraft may orient its thrust vector into any desired direction, whereas the thrust vector of solar sailcraft is constrained to lie on the surface

[2] for high reflectivity
[3] for high emissivity

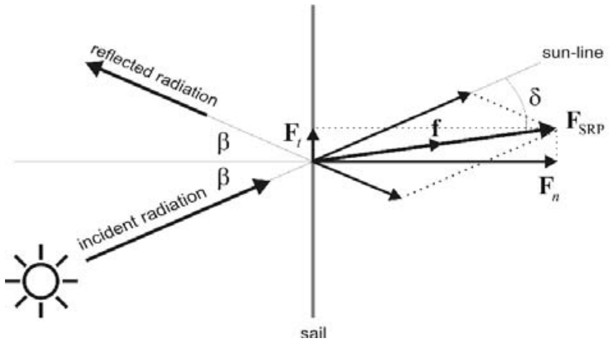

Figure 1: SRP force on a non-perfectly reflecting solar sail

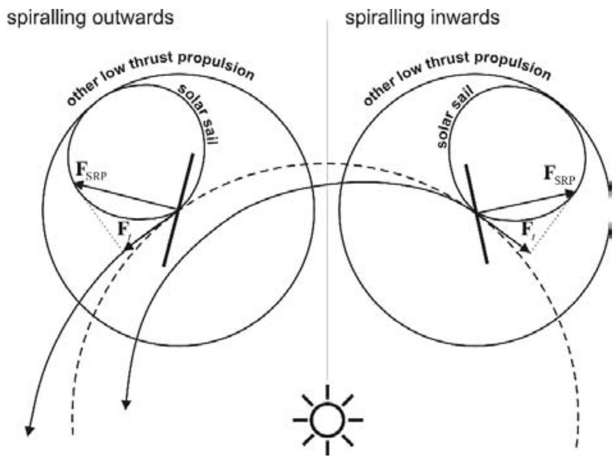

Figure 2: Spiralling towards and away from the Sun

of a "bubble" that is always directed away from the Sun. Nevertheless, by controlling the sail orientation relative to the Sun, solar sailcraft can lose orbital angular momentum and spiral inwards – towards the Sun – or gain orbital angular momentum and spiral outwards – away from the Sun.

Now, the commonly used performance parameters for solar sailcraft will be introduced:

The **sail assembly loading**

$$\sigma_{\text{SA}} = \frac{m_{\text{SA}}}{A} \tag{9}$$

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

The **sailcraft loading**

$$\sigma = \frac{m}{A} = \frac{m_{\text{SA}} + m_{\text{PL}}}{A} = \sigma_{\text{SA}} + \frac{m_{\text{PL}}}{A} \tag{10}$$

is defined as the specific mass of the sailcraft including the payload (index "PL"), where the term payload stands for the total sailcraft except the solar sail assembly (i.e. except the propulsion system).

The SRP force acting on the sail may also be expressed in terms of the **characteristic acceleration** a_c, which is defined as the SRP acceleration acting on a solar sail that is oriented perpendicular to the Sun-line at Sun–Earth distance r_0, where $P_{\mathrm{eff},0} \doteq 1.8163 \cdot P(r_0) \doteq 8.288\,\mu\mathrm{N/m}^2$, and therefore

$$a_c = \frac{P_{\mathrm{eff},0} A}{m} = \frac{P_{\mathrm{eff},0}}{\sigma} = \frac{P_{\mathrm{eff},0}}{\sigma_{\mathrm{SA}} + \frac{m_{\mathrm{PL}}}{A}} \tag{11}$$

The SRP force may then be written as

$$\mathbf{F}_{\mathrm{SRP}} = m a_c \left(\frac{r_0}{r}\right)^2 \frac{2\,Q(\beta)}{1.8163}\,\mathbf{f} \tag{12}$$

At DLR, a $(20\,\mathrm{m})^2$ solar sail was successfully deployed in December 1999 on-ground in a simulated gravity-free environment and ambient environmental conditions (Figure 3) [2, 3]. The DLR solar sail baseline design is a square sail that consists of four CFRP (Carbon Fiber Reinforced Plastics) booms and of four triangular sail segments. The booms are made of two CFRP shells that are bonded at the edges to form a tubular shape, so that they can be pressed flat and rolled up in a central deployment module. During deployment, they unfold automatically and return to their tubular shape with high bending and buckling strength. Subsequently, the four sail segments are deployed by ropes.

Figure 3: Deployed $(20\,\mathrm{m})^2$ solar sail and CFRP boom at DLR

3. NEA RENDEZVOUS MISSION (ENEAS)

ENEAS was intended to transport a scientific payload of 5 kg (CCD camera + IR spectrometer + magnetometer) to a NEA within less than five years (Table 1 summarizes the ENEAS parameters). $1996\mathrm{FG}_3$ had been chosen as the target body for the ENEAS mission, since it has orbital elements not too different from that of Earth and since it is an object of exceptional scientific relevance. Observations [8, 9] indicate that $1996\mathrm{FG}_3$ is a binary C type asteroid, consisting of a primary body with a rotation period of about 3.6 hours and a secondary body with an orbital period of about 16 hours. Based on the typical C type albedo of 0.06, the primary body has an estimated diameter of about 1.4 km and the secondary body an estimated diameter of about 0.4 km. The separation of the two bodies is about 1.7 times the diameter of the primary body. The determined average bulk density is $1.4 \pm 0.3\,\mathrm{g/cm}^3$, which is highly suggestive of a "rubble pile" structure. ENEAS was intended to determine the morphological properties and the compositional properties of the $1996\mathrm{FG}_3$ system via remote sensing [1]. Trajectory optimization using InTrance, an evolutionary neurocontrol based global trajectory optimization method recently developed at DLR [10, 11, 12], yields an optimal transfer time of 4.18 years for $C_3 = 0\,\mathrm{km}^2/\mathrm{s}^2$ (Figure 4) and 2.74 years for $C_3 = 4\,\mathrm{km}^2/\mathrm{s}^2$ (Figure 5). Thus, it would be very beneficial, if the launcher could provide some hyperbolic excess energy for interplanetary injection, but this is not absolutely necessary to accomplish the mission.

Sail area	A	$(50\,\mathrm{m})^2$
Sail assembly mass	m_{SA}	73 kg
Sail assembly loading	σ_{SA}	$29.2\,\mathrm{g/m}^2$
Payload mass (incl. spacecraft bus)	m_{PL}	75 kg
Total sailcraft mass	m	148 kg
Sailcraft loading	σ	$59.2\,\mathrm{g/m}^2$
Characteristic acceleration	a_c	$0.140\,\mathrm{mm/s}^2$
Characteristic SRP force	F_c	20.7 mN

Table 1: ENEAS

4. NEA SAMPLE RETURN MISSION (ENEAS-SR)

To study the $1996\mathrm{FG}_3$ system in 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 $1996\mathrm{FG}_3$ could probably only be accomplished by taking samples of the asteroid back to Earth. Solar sailcraft are – due to their high ΔV-capability – supposed to be especially capable to perform such a mission. However, compared to the ENEAS rendezvous mission, the payload has to be extended considerably, including a lander and a sample return capsule.

To derive the required *minimum* solar sailcraft performance for this mission, the maximal mission duration was set to 10 years, since we consider a longer mission duration to be not acceptable.[4] Trajectory optimization using InTrance shows that the ENEAS-SR mission to $1996\mathrm{FG}_3$ can be achieved even with a

[4]Looking at the Rosetta-mission, about 10 years seem to be the scarcely acceptable maximum duration for a mission with an exceptional scientific return.

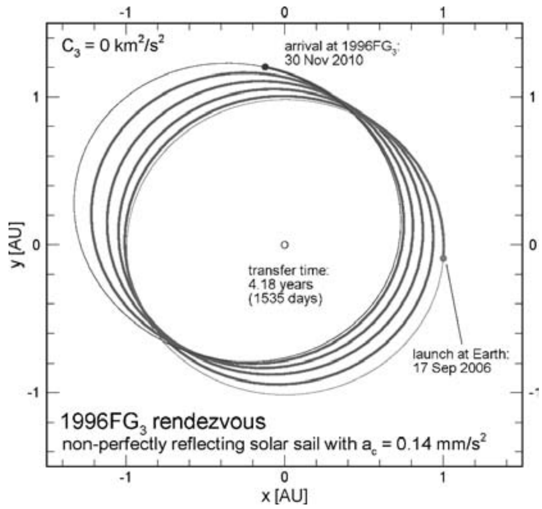

Figure 4: ENEAS trajectory option for $C_3 = 0\,\mathrm{km}^2/\mathrm{s}^2$

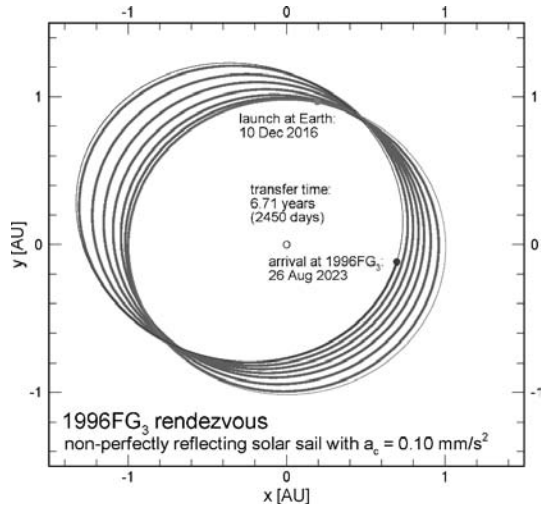

Figure 6: ENEAS-SR outward trajectory option

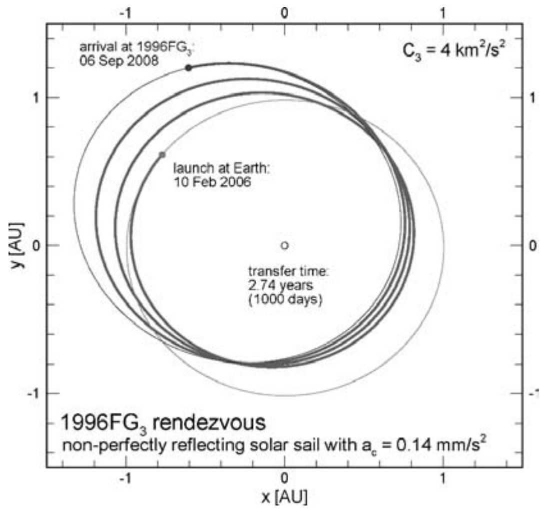

Figure 5: ENEAS trajectory option for $C_3 = 4\,\mathrm{km}^2/\mathrm{s}^2$

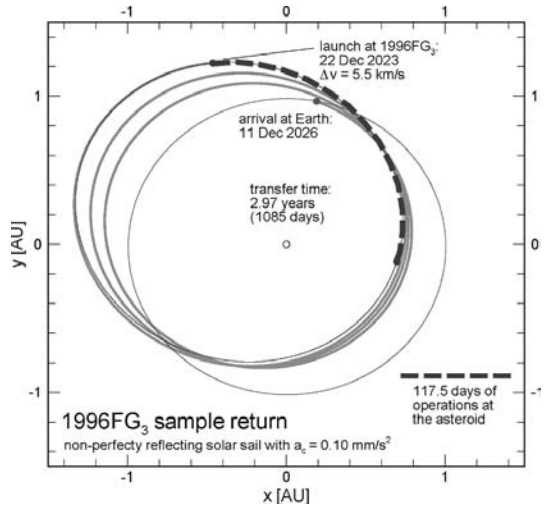

Figure 7: ENEAS-SR return trajectory option

low characteristic acceleration of $0.10\,\mathrm{mm/s}^2$ in exactly 10 years, including a rendezvous trajectory of 6.7 years (Figure 6), 117.5 days of operations at the asteroid, and an Earth return trajectory of 3.0 years (Figure 7). The Earth return leg is much shorter than the outward leg, since no rendezvous is required at Earth.

Looking at equation (11), one can see that the solar sailcraft performance depends on three design parameters, the sail assembly loading σ_{SA}, the payload mass m_{PL}, and the side length s (or area $A = s^2$) of the solar sail, defining a three-dimensional solar sailcraft design space. Figure 8 shows the required sail size for different sail assembly loadings and payload masses, to obtain a characteristic acceleration of $0.10\,\mathrm{mm/s}^2$. Based on our experiences with the ground-based solar sail technology demonstration, a maximum sail size of $(70\,\mathrm{m})^2$ with a sail assembly mass of 111 kg ($\sigma_{\mathrm{SA}} = 22.7\,\mathrm{g/m}^2$, sail film + booms + deployment module) is considered as a realistic – however still challenging – baseline for the ENEAS-SR mission [13]. This yields a payload mass of 295 kg. The realization of a spacecraft within the specified mass budget, including a lander of about 150 kg and a sample return capsule of about 50 kg, appears to be feasible but requires further investigation. Table 2 summarizes the ENEAS-SR parameters.

Sail area	A	$(70\,\mathrm{m})^2$
Sail assembly mass	m_{SA}	111 kg
Sail assembly loading	σ_{SA}	$22.7\,\mathrm{g/m}^2$
Payload mass (incl. spacecraft bus)	m_{PL}	295 kg
Total sailcraft mass	m	406 kg
Sailcraft loading	σ	$82.9\,\mathrm{g/m}^2$
Characteristic acceleration	a_c	$0.100\,\mathrm{mm/s}^2$
Characteristic SRP force	F_c	40.6 mN

Table 2: ENEAS-SR

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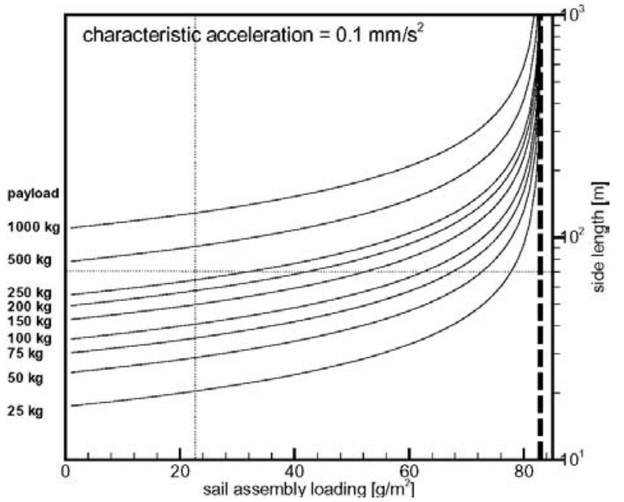

Figure 8: The side length s of the solar sail that is required to achieve a characteristic acceleration of $0.10\,\mathrm{mm/s^2}$ as a function of σ_{SA} and m_{PL}

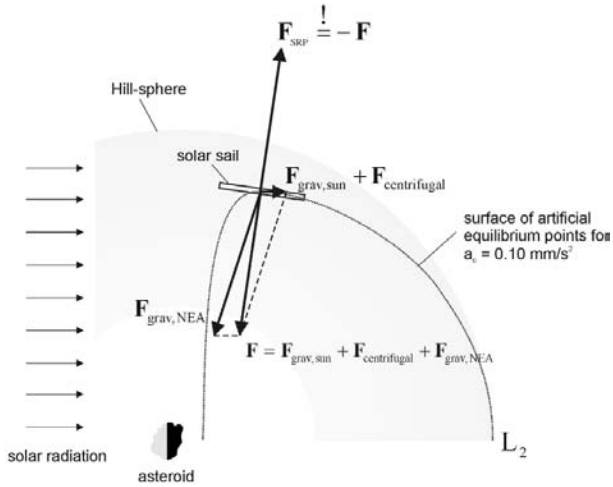

Figure 9: Hovering at the asteroid

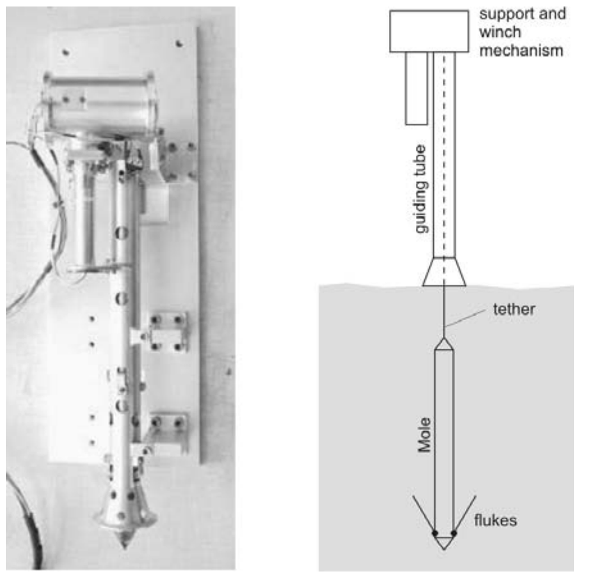

Figure 10: PLUTO system: qualification model (left side) and conceptual drawing of asteroid sampling operations (right side)

Since for solar sailcraft of moderate performance it is difficult and time consuming to gain orbital energy in the Earth's gravitational field, the launcher inserts the ENEAS-SR sailcraft directly into an interplanetary trajectory. After injection and deployment, the sail is oriented to follow the pre-calculated attitude profile that leads to an optimal interplanetary transfer trajectory. During the transfer, the ENEAS-SR sailcraft runs almost autonomously, so that ground monitoring is carried out on a weekly basis only [2]. At the end of the transfer trajectory, the solar sailcraft makes a rendezvous with $1996\mathrm{FG}_3$ within its gravitational sphere of influence (Hill-sphere) of between 70 and 150 km radius (at $1996\mathrm{FG}_3$'s perihelion and aphelion respectively).

Even in the near-field of the asteroid, the SRP acceleration of between 0.05 and $0.21\,\mathrm{mm/s^2}$ (at aphelion and perihelion respectively) 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 artificial equilibrium surfaces in the hemisphere that is opposite to the Sun (Figure 9). Those quasi-stationary hovering positions are unstable but can be stabilized using a feedback control loop [14]. Hovering near the asteroid, the (likely complex) gravitational field is studied, so that a coarse gravitational field model can be determined. Thereafter, the lander with the integrated Earth return capsule is separated from the solar sail to go into closer orbit around 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) and landing can be performed.

For acquiring asteroid samples, an evolution of the "Mole" subsurface soil sampler (PLUTO – Planetary Underground Tool) developed for the Beagle 2 Mars lander of ESA's 2003 Mars Express mission [15] is foreseen. This device, a slender cylinder of about 30 cm in length with a pointed tip, achieves penetration into soil-like materials with an internal hammering mechanism driven by a single motor/gear actuator. With each shock, the medium surrounding the Mole is partially displaced and compressed to allow an additional forward motion of up to a few millimeters per shock, while a tether connecting the Mole to the lander is trailed behind for power supply of the mechanism, data transmission from internal Mole sensors, and to retract the device to the lander. Soil penetration depths of several meters are feasible and have been demonstrated in ground testing prior to the Beagle 2 application of the system on Mars. Since the upper meters of even small asteroids are generally believed to be composed of a regolith produced from billions of years of impact events (including a continual flux of micrometeorites, cf. e.g. [16]), a soil sampling system – as opposed to a rock sampler – is a valid assumption for ENEAS-SR. The Mole's operating principle is attractive because

1. acquisition of samples from below the immediate surface is possible,
2. the overall system including a tether mechanism and supporting elements weights less than 900 g, and
3. no reactive forces have to be accepted by the lander because of the tether-linking of the two elements.

However, for the self-penetration to occur, recoil from the forward hammering must be reacted by the surrounding soil through friction along the Mole's outer surface. In a nearly weightless environment as on $1996FG_3$, skin friction between the Mole and the soil is negligible due to its dependency on gravitational acceleration. Therefore, a modification to the Mole design is foreseen such that during each shock, flukes that greatly increase friction with the soil to react shock recoil are deployed passively. Operationally, the Mole system offers the possibility to obtain several subsurface samples from different depths, either from multiple Mole deployments if only one sample is taken at a time (as for PLUTO on Beagle 2) or from a single deployment if a design with multiple sample chambers is used. The size of a given sample is in any case several tens of cubic-millimeters, which is fully sufficient for analysis, even for rather crude in-situ instruments as presently used in Mars landing missions.

The options presently considered for the Mole release are: ejection from the lander shortly before surface contact and/or release after landing and impression into the surface. In both cases, due to the negligible gravity, the resulting reactive force on the lander must be controlled by the on-board chemical propulsion system. The lander would release the Mole from its holster-like guiding tube which is accommodated vertically between the lander underside and the surface. After sampling, the tether mechanism reels the Mole back into the holster and the sample chamber is detached to be transferred into the Earth return capsule.

After all sampling operations are finished, the lander brings the samples 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 to assist the rendezvous between the lander and the solar sail. 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 is sufficient to perform all lander operations.

After rendezvous with the hovering sailcraft, the redocked ENEAS-SR solar sailcraft returns the sample to Earth. Finally, some hours before the arrival at Earth, the lander with the return capsule is separated from the sail, the capsule is spun-up to maintain the required entry attitude, and injected into an Earth reentry trajectory, where it is decelerated by atmospheric friction and breaking parachutes. 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 5.5 km/s. The gravitational acceleration of Earth adds another 11.2 km/s, so that the Earth entry velocity may reach about $\sqrt{5.5^2 + 11.2^2}\,\mathrm{km/s} = 12.5\,\mathrm{km/s}$. This is slightly less than the entry velocity of NASA's Stardust capsule, which has the highest entry velocity of any Earth-returning mission up to date (12.9 km/s) [17, 18].

To assess the capability of solar sail propulsion for the ENEAS-SR mission, comparative mission analyses have been performed for two SEP systems, using NASA's NSTAR ion thruster and using a cluster of three NSTAR thrusters respectively [10]. With a single thruster, the mission could be performed within a total mission duration of 3.5 years. About 180 kg of Xenon would be required as propellant. The launch mass would, however, be 50% larger than for the solar sail option, requiring eventually a more expensive launch. Using a cluster of three thrusters, the total mission duration would be further reduced to 1.9 years (this would not only require a larger thrust but also about 20% more propellant). In this case, the launch mass would be about twice that of the solar sail option. Therefore, if only the flight duration is considered, the solar sail option is clearly outperformed by the SEP option, whereas, if only the launch costs are considered, the solar sail might be the favorable option. Thus, if the longer arising ground operation costs are lower than the savings in launch costs, and if the mission duration plays a subordinate role with respect to cost, the solar sail might be the better propulsion option for this mission.

5. MULTIPLE NEA RENDEZVOUS AND SAMPLE RETURN MISSIONS (ENEAS+, ENEAS+SR)

For ENEAS-SR, a $(70\,\mathrm{m})^2$ solar sail with a mass of 111 kg was presumed to accomplish a characteristic acceleration of $0.10\,\mathrm{mm/s^2}$ with a 295 kg payload. Within this section, it is investigated, whether – using the same solar sail – a multiple NEA rendezvous mission can be performed with the small ENEAS payload of 75 kg. This multiple NEA rendezvous mission is termed ENEAS+. Table 3 summarizes the ENEAS+ parameters.

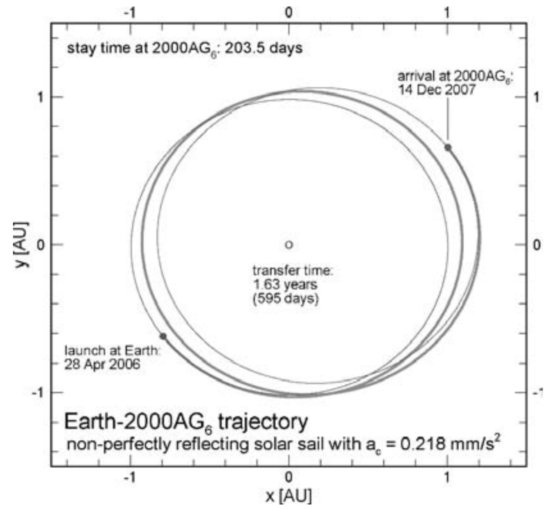

Figure 11: ENEAS+ trajectory option for Earth-2000AG$_6$-leg

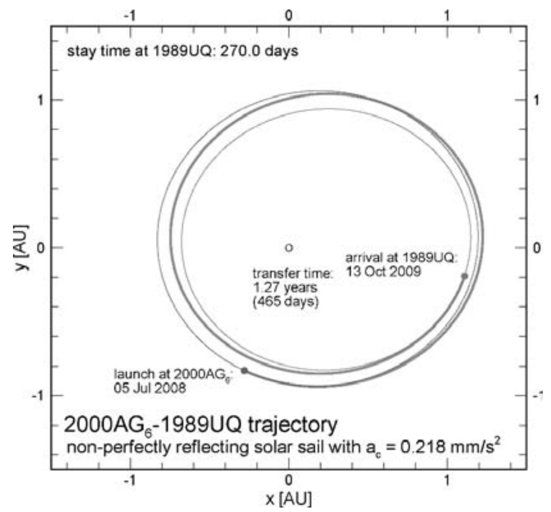

Figure 12: ENEAS trajectory option for 2000AG$_6$-1989UQ-leg

Sail area	A	$(70\,\mathrm{m})^2$
Sail assembly mass	m_{SA}	$111\,\mathrm{kg}$
Sail assembly loading	σ_{SA}	$22.7\,\mathrm{g/m^2}$
Payload mass (incl. spacecraft bus)	m_{PL}	$75\,\mathrm{kg}$
Total sailcraft mass	m	$186\,\mathrm{kg}$
Sailcraft loading	σ	$38.0\,\mathrm{g/m^2}$
Characteristic acceleration	a_c	$0.218\,\mathrm{mm/s^2}$
Characteristic SRP force	F_c	$40.6\,\mathrm{mN}$

Table 3: ENEAS+

For the given solar sail and the given payload, the characteristic acceleration of the ENEAS+ solar sailcraft is $0.218\,\mathrm{mm/s^2}$. To compare the solar sail option with the SEP option, the target objects of the Hera-mission, a proposed multiple NEA sample re-

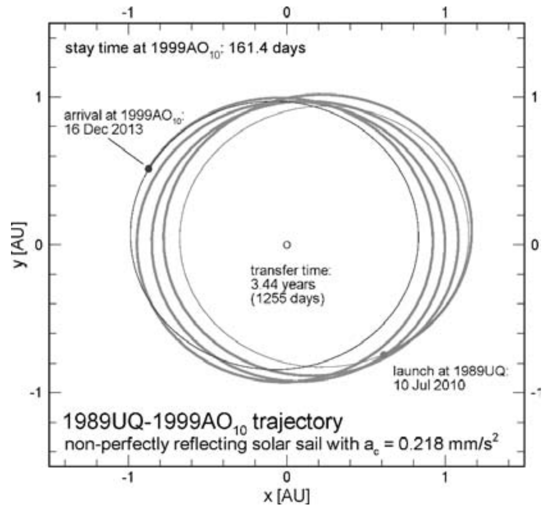

Figure 13: ENEAS+ trajectory option for 1989UQ-1999AO$_{10}$-leg

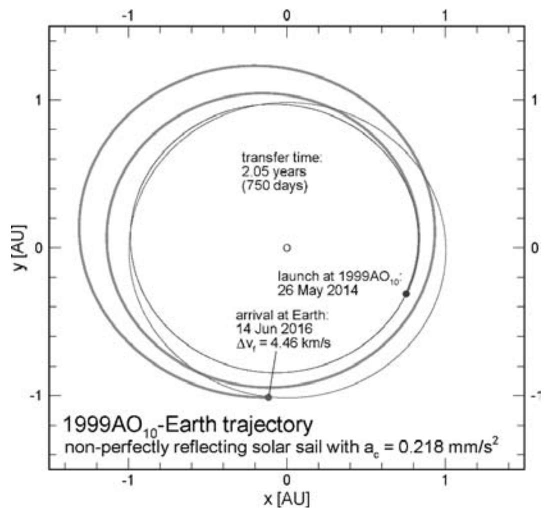

Figure 14: ENEAS+SR trajectory option for 1999AO$_{10}$-Earth-leg

turn mission[5] that employs a spacecraft with a cluster of three NSTAR thrusters to return samples from three different NEAs to Earth within a mission duration of about 4.8 years (676 kg dry mass; 1102 kg total mass), have been adopted [19]. InTrance has been used to calculate the transfer times between the targets for various launch dates to find the optimal target sequence. The best found sequence is illustrated in Figures 11 to 13.

The optional trajectory from 1999AO$_{10}$ to Earth, shown in Figure 14, is not part of the actual ENEAS+ mission. Nevertheless, the trajectories are valid for any solar sailcraft that accomplishes a characteristic acceleration of $0.218\,\mathrm{mm/s^2}$. Using a larger solar sail ($s = 139\,\mathrm{m}$), this mission might be also performed with the ENEAS-SR payload. With such a

[5] mission proposal to NASA under the lead of the Arkansas-Oklahoma Center for Space and Planetary Science

solar sailcraft, all three visited NEAs could be sampled, and the samples could be returned to Earth within 10.1 years (this mission might be termed ENEAS+SR). Regarding the question, whether solar sail propulsion or SEP is superior for such a mission, the same conclusions as in the previous section can be drawn.

6. SUMMARY AND CONCLUSIONS

It was shown that even with moderate-performance solar sails of the first generation, challenging scientific missions, like a sample return mission or a multiple rendezvous mission to near-Earth asteroids, are feasible at relatively low cost. It was shown that a $(70\,\mathrm{m})^2$ solar sail with a specific weight of about $23\,\mathrm{g/m}^2$ is able to transport a 300 kg-spacecraft (incl. a lander and a sample return capsule) to a near-Earth asteroid and to return a sample to Earth within 10 years. With the same solar sail, a triple NEA rendezvous mission with a 75 kg-payload is feasible within 7.6 years, spending about 200 days at each asteroid.

The obtained results demonstrate that near-term solar sailcraft are outperformed by the SEP option, if only the flight duration is considered, whereas, if only the launch costs are considered, the solar sail might be the favorable option. Thus, if the longer arising ground operation costs are lower than the savings in launch costs, and if the mission duration plays a subordinate role with respect to cost, the solar sail might be the better propulsion option for the missions considered within this paper. Nevertheless, on the way to more advanced solar sailcraft, as they are required for high-ΔV missions, the development of solar sails with moderate performance is an indispensable first stepping stone.

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