ASTEROID FLYBY FLEET MISSION BY E-SAIL PROPULSION

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KEYWORDS

Single tether electric sail, nanospacecraft fleet.

Abstract

We present a mission concept to make a flyby study of 300 asteroids at low cost using a fleet of E-sail propelled cubesats. The spacecraft are launched to marginal escape orbit or other interplanetary orbit either as a single group or in smaller batches. Each spacecraft uses its E-sail tether to make a tour of the asteroid main belt, performing 6-7 asteroid flybys at 1000 km distance. The spacecraft carries a 4 cm telescope and a near infrared (NIR) spectrometer. The telescope is also used for optical navigation during cruise. To solve the problem of deep-space telemetry, each spacecraft makes its 3.2 year long tour autonomously and transfers the gathered data to ground during a final Earth flyby. In this way, only 3 hours of deep space network telemetry is needed per spacecraft, to return up to 10 gigabytes of data per spacecraft. Together with the launch adaptor, the launch mass per nanospacecraft to marginal escape orbit is about 10 kg.

1 INTRODUCTION

The solar wind electric sail (E-sail) is a concept for propelling a spacecraft in the solar system using the natural solar wind [1, 2, 3, 4, 5]. The E-sail uses a number of thin metallic, centrifugally stretched tethers that are biased at high positive potential (Fig. 1). The biasing is actuated by an onboard electron emitter which pumps out negative charge from the system, at the same rate that the positively biased tethers gather electrons from the surrounding solar wind plasma.

In this paper we consider a single-tether variant of the E-sail to implement a nanospacecraft fleet mission to the asteroid belt. A particular variant of the concept has been proposed to ESA as "Multi-asteroid touring" (MAT) [6]. The structure of the paper is as follows. We introduce the single-tether E-sail and explain how spacecraft pointing can be done. We treat some aspects of spacecraft engineering and present the mission sequence. The paper ends with summary and discussion.

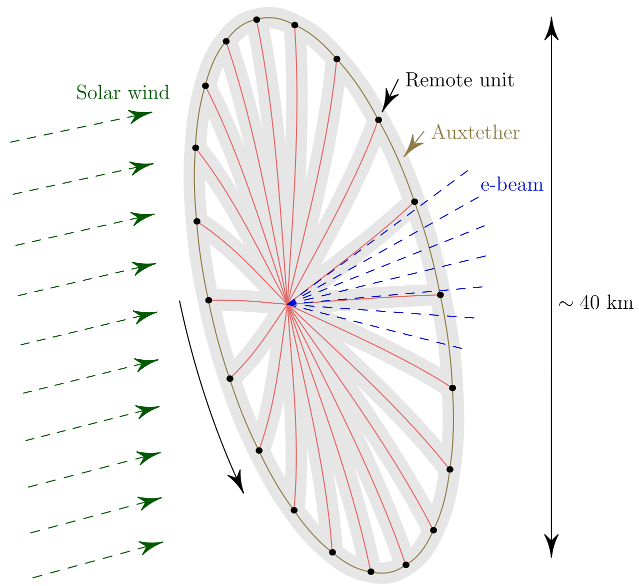

Figure 1: Multi-tether E-sail.

2 NANOSPACECRAFT SINGLE-TETHER E-SAIL

The single-tether E-sail consists of the spacecraft, a thin metallic tether and a remote unit at the tether's tip (Fig. 2). The tether rotates slowly with a spin period of ~ 1 h. The spin period is chosen so that the tether has a suitable centrifugal tension of ~ 3 cN (grams) that overcomes the gathered solar wind thrust by factor ~ 3. The expected solar wind thrust is 500 nN/m in average solar wind at 1 au if 20 kV tether voltage is used so that a 20 km long tether produces 1 cN of thrust. The thrust scales approximately linearly with the employed voltage, and it is inversely proportional to the heliocentric distance[1].

[1]The E-sail's $1/r$ heliocentric distance scaling is more favourable than for solar electric propulsion and photonic sail, because for them the thrust decays as $1/r^2$. The difference is due to plasma Debye length scaling of the the E-sail's effective sail area[5].

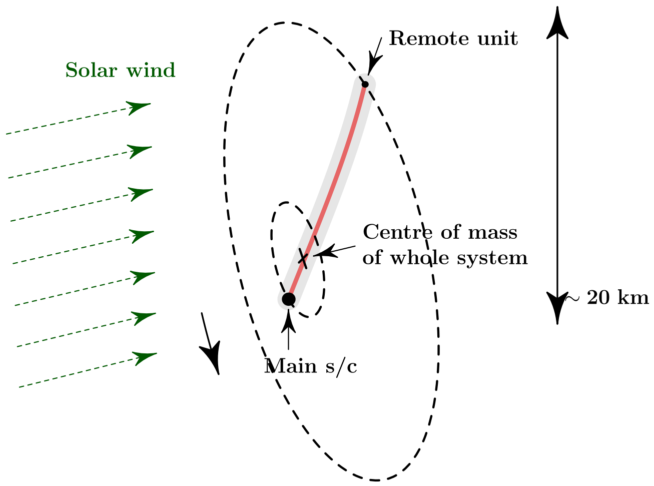

Figure 2: Single-tether E-sail.

2.1 Spacecraft pointing

For an asteroid flyby mission, we need a capability to point the spacecraft towards the target during the flyby so that the optical instrument does not need a gimbal.

The tether is very long and thus it produces negligible resistance to mechanical twisting. This enables easy control of the spacecraft's attitude around the tether direction (Fig. 3) by using a reaction wheel. The spacecraft body is a 3-U cubesat ($30 \times 10 \times 10$ cm) and on its roof there is a deployed solar panel wing extending to the direction of the tether. One end of the spacecraft contains the telescope. The telescope is used for optical navigation during the mission and as a scientific instrument during the asteroid flybys.

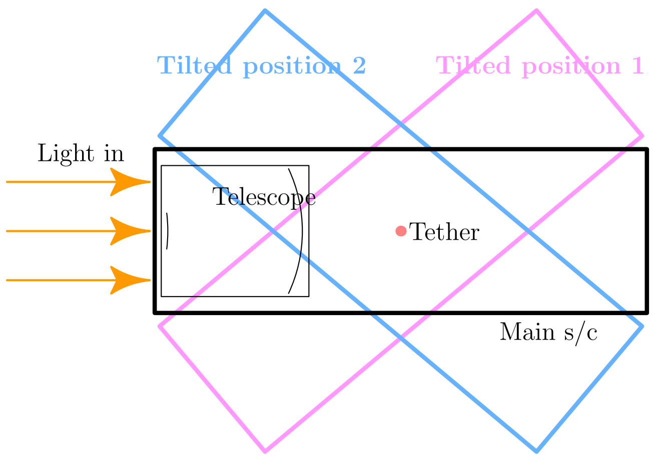

Figure 3: View of spacecraft along tether direction. The tether is very long and thus produces negligible resistance to twisting, which enables easy control of the spacecraft's attitude around the tether direction.

During one ~ 1 h spin period of the tether, the spacecraft's telescope would be able to scan the entire sky already by the around-tether attitude control. However, the closest asteroid flyby lasts only a couple of minutes, so an ability to control the attitude in the other angle is also called for. This can be achieved if the tether is attached at the centre of mass of the spacecraft. The spacecraft now rotates easily around the spin axis of the main tether rotation, only resisted by its own moment of inertia (Fig. 4). A prism-shaped indentation must be made for the tether so that it can be attached to the centre of mass. The tether is deployed from the remote unit end so that the tether reel resides there. On the main spacecraft side the tether only needs an attachment point.

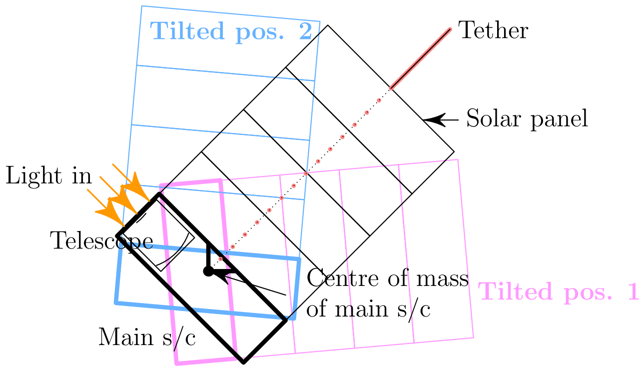

Figure 4: Top view of spacecraft. Attachment of the tether at the centre of mass of the spacecraft allows easy control of the spacecraft's attitude around the vertical axis (spin axis of the tether's rotation).

Figure 5 shows the side view. The tether branches into two in order to prevent the spacecraft from oscillating around its long axis (i.e., around the axis which is perpendicular to Fig. 5). The upper tether branch is made of insulating material so that the high-voltage part of the tether (red) has sufficient clearance with respect to the backside of the solar panel wing.

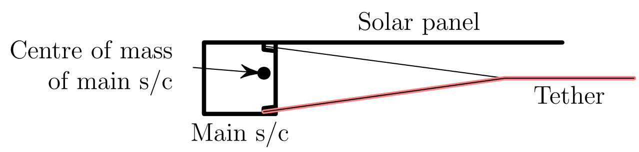

Figure 5: Side view of spacecraft. The tether's double-branched ending prevents the spacecraft from oscillating along axis perpendicular to plot. High voltage tether part (red) remains at safe distance from backside of the solar panel wing.

3 SPACECRAFT ENGINEERING

The main hosted items onboard the spacecraft and the remote unit are listed in Table 1. For a fuller account on spacecraft engineering, see Slavinskis et al. 2018[7].

3.1 Thrusters and delta-v budget

The remote unit needs thrusters for producing the angular momentum during tether deployment. Production

Table 1: Main hosted items onboard spacecraft and remote unit.

Spacecraft
Telescope Tether attachment High voltage subsystem and electron emitter Communication with remote unit Communication with ground station when near Earth Thrusters (by default)
Remote unit
Tether reel Communication with spacecraft ACS with reaction wheels Thrusters

of angular momentum is efficient because thrusting is made at the tip of the tether. During flight, one must also apply thrusting to overcome the secular spinrate change of the tether's rotation which is due to the orbital Coriolis effect [8]. The Coriolis effect arises because the tether's spinplane has to track the Sun when the spacecraft orbits the Sun. The Sun tracking requires changing the tether's angular momentum vector (i.e. tilting the spinplane) which tends to stay fixed with respect to distance stars (i.e. the inertial frame). Tilting the spinplane can be done by modulating the tether voltage in sync with the rotation, but as a byproduct of the process, the spinrate ω changes according to

$$\frac{d\omega(t)}{dt} = \omega(t)\frac{d\varphi(t)}{dt}\tan\alpha \tag{1}$$

where φ is the heliocentric orbital phase angle and α is the tether inclination angle with respect to the Sun direction.

To roughly estimate the required delta-v budget of the remote unit, let us assume that the remote unit's mass m is much less than the mass of the spacecraft so that the countermotion of the spacecraft can be neglected. For simplicity, we also consider the tether as massless so that the tether tension is equal to the centrifugal force F_{cf} experienced by the remote unit:

$$F_{\text{cf}} = mR\omega^2 \tag{2}$$

where R is the tether length (20 km). Due to $d\omega/dt$, the secular acceleration of the remote unit is

$$a = R\frac{d\omega}{dt} = R\sqrt{\frac{F_{\text{cf}}}{mR}}\frac{d\varphi}{dt}\tan\alpha. \tag{3}$$

To overcome the secular acceleration, one must apply thrusting, and the delta-v requirement Δv is $\int a dt$ so that overall,

$$\Delta v = \sqrt{\frac{F_{\text{cf}}R}{m}}\Delta\varphi\tan\alpha. \tag{4}$$

For $R = 20\,\text{km}$, $F_{\text{cf}} = 3\,\text{cN}$, $m = 0.662\,\text{kg}$, $\alpha = 16.5°$ and $\Delta\varphi = 200°$ we obtain $\Delta v = 31$ m/s. These values correspond to the thrusting arc of the mission shown in Fig. 8 below. In addition to the flighttime delta-v, we need the spinup delta-v. A lower limit for the spinup delta-v is the rotational velocity of the tether tip,

$$v_{\text{tip}} = R\omega = \sqrt{\frac{F_{\text{cf}}R}{m}} = 30\text{m/s}. \tag{5}$$

The lower limit for the spinup delta-v is not far from the actual value if tether tension during deployment is kept clearly smaller than 3 cN. We estimate that the remote unit delta-v value of 80 m/s should be sufficient.

If one wants to reduce the total delta-v, it may also be possible to use a lower tether tension initially and let the Coriolis effect accelerate the spin. In this way, both the spinup and flighttime delta-v values can diminish.

Our baseline thruster is an ionic liquid thruster such as TILE 50 of Accion Systems [9] and a cold gas thruster such as NanoProp CGP3 of GOMSpace[10] as a backup choice. The specific impulse of TILE 50 is 1250 s so that to give the 0.662 kg remote unit a delta-v capability of 80 m/s requires only 4.3 grams of propellant. With CGP3 (specific impulse 60 s if not using heating) one would need 96 grams of liquid butane.

3.2 Other sparts of the spacecraft

Figures 6 and 7 show exploded views of the spacecraft and the remote units. During stowed configuration, the two components fit in a 6-U envelope, without using it fully. During flight, the main spacecraft is of 3-U size and the remote unit of 1-U size.

Desaturation of the reaction wheel which controls attitude around the tether's spin axis can be done without thrusting, because the tether is a large reservoir of angular momentum. Desaturation of the reaction wheel which controls attitude around the tether direction probably requires thrusting, however. Hence we include a thruster system also in the main spacecraft. The thruster system also makes it straightforward to make small orbital corrections in the final approach phase of an asteroid. The E-sail is the main propulsion system used to control the orbit, but vectoring the E-sail takes a couple of days because it involves tilting of the sail. Making the final orbital corrections could be done by the remote unit thrusters alone, but thrusting (also) from the main spacecraft side makes the manoeuvre simpler because then the E-sail rotational state is not disturbed.

The power system was designed such that the beginning of life (BOL) maximum electric power is 40-47 W at 1 au and 10-12 W at 2 au.

Concerning the total radiation dose, the components are solar protons and to a lesser extent galactic cosmic rays. Of these, only solar protons are shieldable in the nanosatellite context. The flux of solar protons diminishes when going outward from the Sun. The decay does not obey any simple law, but could be roughly approximated by a $1/r^{1.5}$ dependence (Rami Vainio, private communication). Because a spacecraft in a quasi-elliptic heliocentric orbit spends most of its time near the

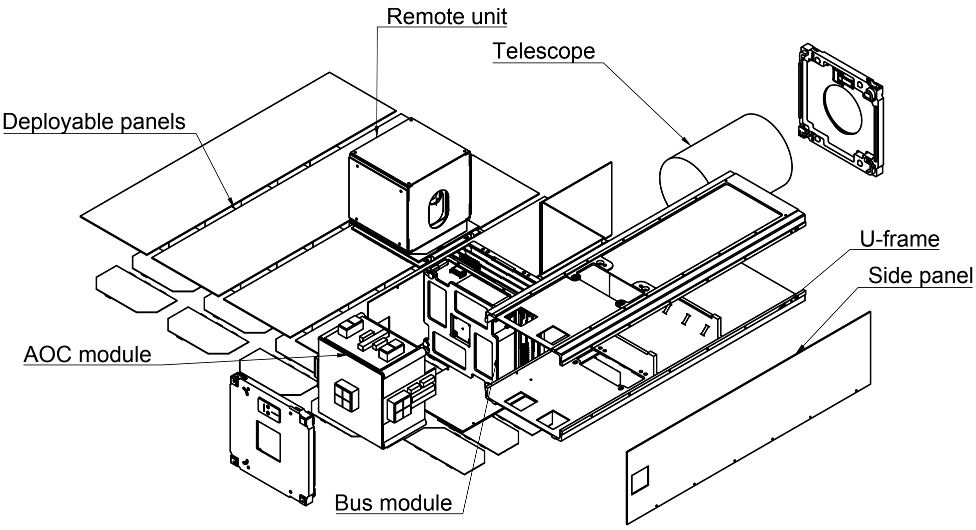

Figure 6: Exploded view of the spacecraft and remote unit in stowed configuration[7].

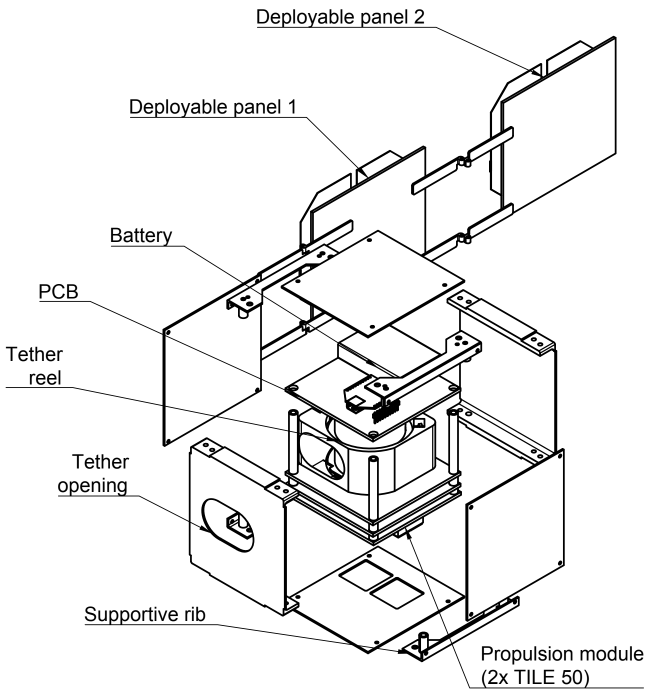

Figure 7: Exploded view of the remote unit[7].

aphelion (this can be seen also in Fig. 8), the net result is that our main belt touring nanospacecraft that spends 3.2 years in space receives about the same total solar proton dose than an interplanetary cubesat at 1 au in a 1 year mission. In other words, our radiation tolerance requirements are similar to a cubesat that orbits (e.g.) the Moon for one year.

Table 2 shows a breakdown of the mass budget for the main spacecraft and the remote unit[7]. By adding a 20 % margin we obtain 6 kg mass for the total flying package (main spacecraft, remote unit and tether). To this mass one has to add the mass of the separators from the carrying mother spacecraft.

Table 2: Mass budget (PCB=printed circuit board, HV=high voltage, RW=reaction wheel, AOC=attitude and orbit control, TILE 50=TILE 50 propulsion module).

Component	Mass/g	Count	Total/g
Spacecraft			
Bus PCB	75	4	300
Battery	80	4	320
Battery PCB	40	2	80
RW200-15	21	1	21
RW-0.01	120	2	240
Sun sensor	5	6	30
Patch antenna	64	1	64
Dipole antenna	100	1	100
HV source PCB	75	1	75
HV insulation	23	1	23
TILE 50	55	5	275
Deployable panels	102	4	408
Hinges	5	16	80
U-frame	184	2	368
Side panels	62	5	310
Frame	182	1	182
AOC structure	112	1	112
Screws, nuts, inserts	100	1	100
Telescope	850	1	850
Framing camera	150	1	150
Total for spacecraft			4088
Remote unit			
PCB	50	2	100
Communications chip	30	1	30
Reel and motor	150	1	150
TILE 50	60	2	120
Deployable panels	42	2	84
Battery	38	1	38
Structure	140	1	140
Total for remote unit			662
Tether (20 km)	200	1	200
Total for spacecraft, remote unit and tether			4950
Total with 20 % margin			5940

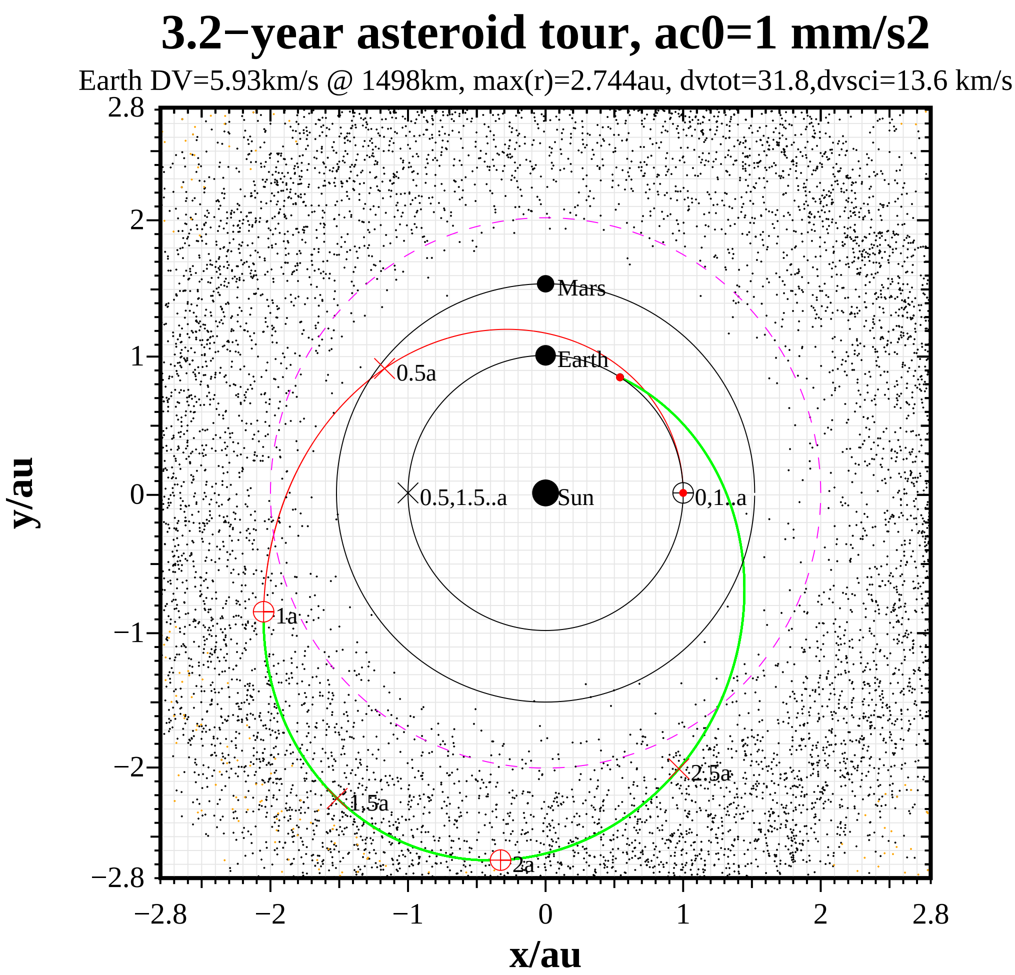

Figure 8: Exemplary tour through the main asteroid belt lasting 3.2 years[6].

4 MAIN BELT FLEET MISSION

Consider a fleet of 50 spacecraft, launched as a ~ 500 kg package to a marginal escape orbit. Each spacecraft weighs 6 kg (including 20 % margin), hence reserving 10 kg for each to cover the additional mass due to the cubesat release mechanisms is adequate. Each spacecraft is ejected from the mother spacecraft, and initially it orbits the Sun at 1 au. The spacecraft deploys its E-sail and thrusts for about one year (Fig. 8). After the thrusting phase is over, the spacecraft is at ~ 2.2 au heliocentric distance and has just entered the main belt. Then a science phase is started where the E-sail is used to maximise the number of flybys with interesting asteroids. The asteroid tour is preprogrammed and employs automatic optical navigation based on nearby known asteroids and planets. The aphelion distance of 2.7 au is reached and after 2.7 years from the beginning the spacecraft exits the main belt and uses the E-sail from that point onwards to ensure a successful Earth flyby, during which the scientific data are downlinked. The downlinking occurs using large (30 m) ground antennas, but only ~ 3 hours of antenna tracking time is required per spacecraft to transfer ~ 10 GB of data at 10 Mbit/s data rate. In this way, no directional antenna is needed on the nanospacecraft and very low cost of science data per bit is realised because the data rate is high. The whole tour lasts 3.2 years in this case and requires, including some margin, 1.0 mm/s^2 of E-sail characteristic acceleration at 1 au.

Figure 9 shows an artistic rendering of the main spacecraft, with its tether deployed.

During the science phase, the inclination angle of the E-sail typically alternates in sign according to the specific needs to reach the wanted asteroids. The altering of the sign typically implies that propulsive spinrate management is not needed because on average, manoeuvres of both signs tend to cancel out each other regarding the secular spinrate change.

Each spacecraft makes its own preprogrammed tour and the fleet of 50 can, as a whole, make a flyby of more than 300 asteroids. To avoid members of the fleet arriving simultaneously to downlink their data, some of the fleet members are programmed to wait for few weeks or months before starting their E-sail. This strategy also provides more spread in the asteroids that can be covered.

Designing a scientifically optimal tour for visiting 300

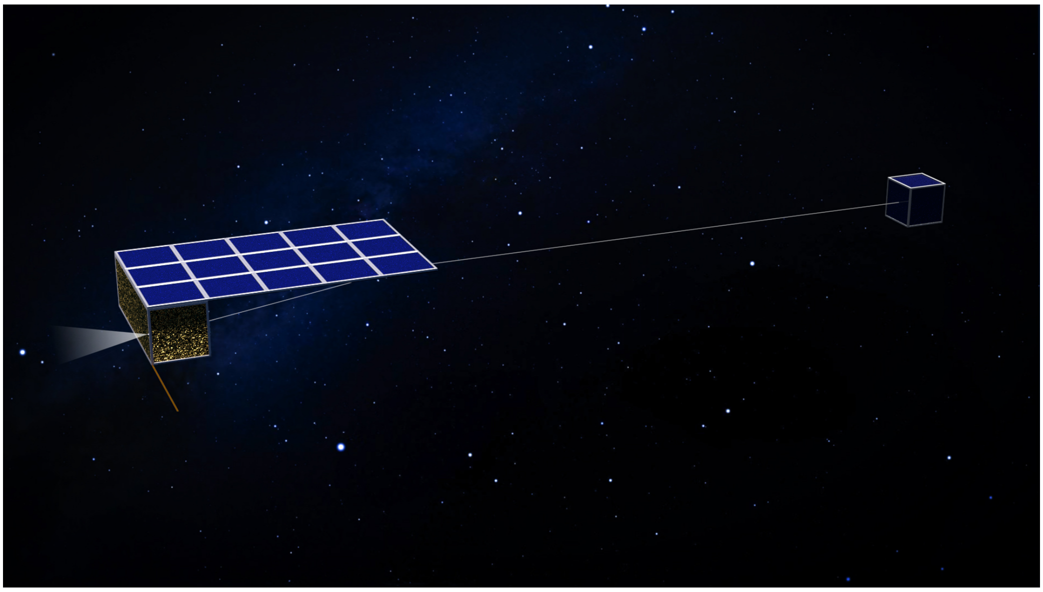

Figure 9: Artistic rendering of the MAT spacecraft.

asteroids would be a daunting task of orbit calculation and scientific prioritisation. To avoid a combinatorial explosion of options, one can select for each spacecraft a single primary asteroid target and then a number of secondary targets that a flown by along the way. The primary target can be selected freely among the globally most interesting objects in the asteroid belt. The secondary targets cannot be selected freely because they must be asteroids that happen to be in the right place at the right time, within the E-sail's capability to manoeuvre. In the example orbit shown in Fig. 8, the E-sail gives a delta-v budget of 13.6 km/s for asteroid to asteroid manoeuvres.

The fleet size of fully scalable. We considered a fleet of 50 mainly because the smallest launch vehicles that can reach marginal escape orbit (such as India's Polar Satellite Launch Vehicle, PSLV) can lift about 500 kg to the marginal escape orbit. The science output of course grows with the size of the fleet, as does the budget. At lower cost categories, one can use piggyback opportunities to launch much smaller fleets than 50. Such piggyback launched mission can also act as a precursor or pathfinder (technology demonstration) for a follow-up full-scale fleet mission. The members in the fleet are completely independent of each other and the science output does not depend on how they are launched (individually, in groups or all at once).

5 NEO MISSION POSSIBILITY

Reaching the main belt in a single orbit around the Sun requires E-sail characteristic acceleration of 1.0 mm/s^2 or slightly less. By characteristic acceleration we mean the acceleration of the spacecraft due to the E-sail effect in nominal solar wind at 1 au when the tether's spinplane is perpendicular to the Sun direction. If the characteristic acceleration is lower, reaching the main belt would be possible using two or more rounds and longer mission duration. Also the remote unit delta-v requirement due to spinrate management would increase because the spinrate management delta-v is proportional to the number of times the spacecraft goes around the Sun with its sail inclined.

A similar mission architecture could also be used for NEOs. In the NEO case, the E-sail characteristic acceleration requirement is less than 1.0 mm/s^2 and consequently one could make the spacecraft heavier (if doing so makes scientific and economic sense) without losing the simplicit benefits of a single-tether E-sail. Besides for science, a NEO version of E-sail asteroid touring would be useful also for planetary defence and for asteroid mineral prospecting as prepration for asteroid mining.

6 SUMMARY AND DISCUSSION

We have given some details of our proposed mission "Multi-asteroid Touring" (MAT) [6]. The mission enables obtaining flyby data (images, spectra) of a large number of asteroids, with low cost per asteroid and low cost of data returned per bit. The main innovative aspects of the mission are the following:

- If one wants to image a large number of asteroids at close range, the only cost-effective way is to use a small spacecraft. The single-tether E-sail makes it possible because it gives a large delta-v capability to a nanospacecraft.

- By making a final Earth flyby we can return the data at low ground operation cost per retrieved bit, and still the spacecraft does not need a directional antenna.

The mission architecture is fully scalable in terms of the number of spacecraft. As a precursor of pathfinder mission, launching a single spacecraft or a few spacecraft as a piggyback payload of another mission could be a useful option.

A similar mission architecture could also be used for NEOs and asteroid mining prospecting, with optionally heavier spacecraft because the E-sail characteristic acceleration requirement would be lower.

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