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SOIL TO SAIL – ASTEROID LANDERS ON NEAR-TERM SAILCRAFT AS AN EVOLUTION OF THE GOSSAMER SMALL SPACECRAFT SOLAR SAIL CONCEPT FOR IN-SITU CHARACTERIZATION

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ABSTRACT

Any effort which intends to physically interact with specific asteroids requires understanding at least of the composition and multi-scale structure of the surface layers, sometimes also of the interior. Therefore, it is necessary first to characterize each target object sufficiently by a precursor mission to design the mission which then interacts with the object. In small solar system body (SSSB) science missions, this trend towards landing and sample-return missions is most apparent. It also has led to much interest in MASCOT-like landing modules and instrument carriers. [1][2][3][4] They integrate at the instrument level to their mothership and by their size are compatible even with small interplanetary missions. [5][6]

The DLR-ESTEC GOSSAMER Roadmap NEA Science Working Groups‘ studies identified Multiple NEA Rendezvous (MNR) as one of the space science missions only feasible with solar sail propulsion. [7] The parallel Solar Polar Orbiter (SPO) study showed the ability to access any inclination and a wide range of heliocentric distances. It used a separable payload module conducting the SPO mission after delivery by sail to the proper orbit. [8] The Displaced L_1 (DL1), spaceweather early warning mission study, outlined a very lightweight sailcraft operating close to Earth, where all objects of interest to planetary defence must pass. [9]

These and many other studies outline the unique capability of solar sails to provide access to all SSSB, at least within the orbit of Jupiter. Since the original MNR study, significant progress has been made to explore the performance envelope of near-term solar sails for multiple NEA rendezvous. [10]

However, although it is comparatively easy for solar sails to reach and rendezvous with objects in any inclination and in the complete range of semi-major axis and eccentricity relevant to NEOs and PHOs, it remains notoriously difficult for sailcraft to interact physically with a SSSB target object as e.g. the HAYABUSA missions do.

The German Aerospace Center, DLR, recently brought the GOSSAMER solar sail deployment technology to qualification status in the GOSSAMER-1 project [11] and continues the development of closely related technologies for very large deployable membrane-based photovoltaic arrays in the GOSOLAR project, on which we report separately. [12][13]

We expand the philosophy of the GOSSAMER solar sail concept of efficient multiple sub-spacecraft integration to also include landers for one-way in-situ investigations and sample-return missions. These are equally useful for planetary defence scenarios, SSSB science and NEO utilization. We outline the technological concept used to complete such missions and the synergetic integration and operation of sail and lander.
We similarly extend the philosophy of MASCOT [1] and use its characteristic features as well as the concept of Constraints-Driven Engineering for a wider range of operations. For example, the MASCOT Mobility hopping mechanism has already been adapted to the specific needs of MASCOT2. [2] Utilizing sensors as well as predictions, those actuators could in a further development be used to implement anti-bouncing control schemes, by counteracting with the lander's rotation. Furthermore by introducing sudden jerk into the lander by utilization of the mobility, layers of loose regolith can be swirled up for sampling.

INTRODUCTION

The recent achievements in solar sail trajectory design [10] and sailcraft hardware development [14][15][16][17][18][19][20][13] made clear that a point has been reached where a review of the results and ongoing efforts should be made for a determination which road they should take. The development towards this point happened in trajectory analysis and technology development over more than a decade, on the background of a sustained resurgence of interest in small solar system bodies (SSSB). It saw the successful conclusion of the HAYABUSA and ROSETTA/PHILAE missions, the launch of HAYABUSA2 [21] with the small lander MASCOT aboard [1], the launch of OSIRIS-REx [22], the flight of IKAROS [14][15][16][17], and the first steps towards a long-term Solar Power Sail (SPS) sample-return mission to the Trojan asteroids of Jupiter [18][19][20].
Among small solar system bodies, the near-Earth asteroids (NEA) in many ways may hold keys to our future on Earth and in space: for planetary science, they appear to represent a fairly good mix of the building blocks of the terrestrial planets while orbiting at an accessible distance; for planetary defence, they are the reservoir of almost all potential threats which we need to understand to protect Earth from

dangerous impacts; and for the new emerging field of asteroid mining, their surfaces and interiors are the promising terra incognita to be mapped and prospected.

MULTIPLE NEA RENDEZVOUS

A near-term mission scenario for solar sails is the multiple NEA rendezvous (MNR). [10][7] It is a means to increase the knowledge on NEAs by accelerating the exploration of a more representative sample of the NEA population. All asteroid user communities – planetary science, planetary defence, and in-space resource utilisation – have an expressed need or desire to expand their respective body of knowledge on a reasonable time scale.
The MNR mission is presently only feasible by solar sail propulsion.

Current MNR trajectory studies demonstrate the feasibility of exploring 5 different NEAs in a rendezvous scenario for >100 days, each, with near-term first-generation sailcraft. [10] This rendezvous duration is comparable to the mission scenario of AIM [23] and MASCOT2 [2]. It is also demonstrated that the sequence of asteroids to be visited can be changed easily and on a daily basis for any given launch date and even *after* launch and between rendezvous. [10]
Therefore, a sailcraft carrying a set of five MASCOT landers based on a common design but differently equipped with science instruments and landing or mobility related systems appears desirable. Which lander is used can be decided after arrival at and initial study of the respective target asteroid, considering the expectations for the targets still to come. Many features of the MASCOT lander design can be shared with the core sailcraft and its four boom-sail deployment units (BSDU) which – excluding their more extensive and for a realistic sailcraft also more voluminous suite of mechanisms – are all MASCOT-scale spacecraft of their own. Indeed, this sharing of design elements and heritage has been done already, for the GOSSAMER-1 QM BSDU which was developed in parallel to MASCOT. This approach was carried on for the structurally similar ROBEX lunar-analog demonstration mission scientific Remote Units (RU) design. [24] The economy of scale becomes immediately obvious considering that one such mission would already consist of 10 independent sub-spacecraft physically connected at launch but to be separated step by step throughout the mission. The initial connection also enables resource-sharing between all initially connected as well as those still connected throughout cruise.

SEE FIVE – THE MISSION SCENARIO

Peloni et al. [10] set a benchmark MNR objective: to study at least 5 NEAs by a rendezvous of at least 100 days, each, in a mission duration of less than 10 years, and presented a multiple-NEA rendezvous mission through solar sailing. Table 1 shows the mission parameters for the sequence shown in the reference paper. The characteristic acceleration of 0.2 mm/s² assumed in this paper was shown to be within the capability of current and near-term sailcraft technology by Seefeldt et al. [11].

Table 1 – Mission parameters for the considered sequence. (For parameters passed from sequence-search algorithm to optimizer see [10]).

Object	Stay time [days]	Start	End	Time of flight [days]
Earth	//			
		10 May 2025	26 Feb 2027	657
2000 SG_{344}	123			
		29 Jun 2027	06 Sep 2028	436
2015 JD_3	164			
		18 Feb 2029	24 Sep 2030	584
2012 KB_4	160			
		04 Mar 2031	29 Sep 2032	576
2008 EV_5	171			
		20 Mar 2033	30 Sep 2034	560
2014 MP	//			

It is worthwhile to note that the arrival at 2014 MP after 3431 days or nearly 9.4 years is not necessarily the end of the mission, nor is it the 222-day stay there still within the 10-year trajectory design goal. The visit at 2014 MP may well be followed by another departure and more journeys to and stays at other NEAs, as long as the sailcraft remains flightworthy. The duration of the mission does not depend on a finite amount of fuel aboard. It only depends on the creativity and attention to detail of the spacecraft designers, the skill and care of the hardware integrators, the means put at their disposal by 'programmatics', the ingenuity and patience of the operators on the ground to get smarter, faster than the sailcraft mechanisms wear out and age in space, and the will to pay them a while longer for their effort.

For one, Pioneer 6 was designed to last about 6 months counting from its launch on December 16th, 1965. It was last operated on December 8th, 2000 – 35 years later. In 1997, three of its instruments still worked well. Two of its three companions fared similarly well; Pioneer 7 successfully participated in the Halley campaign of 1986 and in 1995 one of its instruments was still working, as for Pioneer 8 in 1996. Only Pioneer 9 is known to have failed in 1983. Thereafter, we only know that Earth did not call any of them again …, yet… [25]

Table 2 – Orbital parameters and size of the bodies in the MNR sequences [26]

Object	Earth	2000 SG_{344}	2015 JD_3	2012 KB_4	2008 EV_5	2014 MP
Orbital type	-	Aten	Amor	Amor	Aten	Amor
Semi-major axis [AU]	1	0.977	1.058	1.093	0.958	1.050
Eccentricity	0	0.067	0.009	0.061	0.083	0.029
Inclination [deg]	0	0.111	2.730	6.328	7.437	9.563
Absolute magnitude [mag]	-	24.7	25.6	25.3	20	26
Estimated size [m]	-	35 – 75	20 – 50	20 – 50	260 – 590	17 – 37
EMOID [AU]	-	0.0008	0.054	0.073	0.014	0.020
PHA	-	no	no	no	yes	no
NHATS	-	yes	yes	yes	yes	yes

The asteroids selected by the sequence-search algorithm do tend to have fairly Earth-like orbits, however, the catalog was restricted to NHATS-listed asteroids and PHAs of which a larger fraction populates this region. But MNR missions or solar sails are not at all restricted to targets near the ecliptic or near 1 AU. In earlier studies, the capabilities of solar sails in closely Earth-co-orbital [9], very high heliocentric inclination [8], and even fully retrograde orbits [27][28][29], for similarly demanding ≈10-year missions have been demonstrated for near-term sails. Thus, the combination of micro spacecraft solar sail and nano-lander makes every small solar system body accessible within reasonable mission duration, at least out to the orbit of Jupiter.

Solar sailing has the advantage of continuous target asteroid flexibility. For each launch date, hundreds of accessible NEA target sequences exist even within the restricted database of targets. The targets do not have to be selected before launch and they can be changed en route, for example when scientific or commercial interest changes over the years of the mission or when a new target of particular interest appears.

Going For The One – On the Return Leg

To study the potential for a multiple NEA sample return mission, the last leg to 2014 MP has been removed and substituted with a return leg to the Earth. The same methodology described in Peloni et al. [10] was used to compute the return leg to the Earth. The total mission duration is now 4131 days, about 11.3 years. The complete trajectory of the overall sequence is shown in Figs. 1 & 2, whereas Table 3 shows the updated mission parameters.

Table 3 – Mission parameters for the considered sequence with the last leg to the Earth.

Object	Stay time [days]	Start	End	Time of flight [days]
Earth	//			
		10 May 2025	26 Feb 2027	657
2000 SG_{344}	123			
		29 Jun 2027	06 Sep 2028	436
2015 JD_3	164			
		18 Feb 2029	24 Sep 2030	584
2012 KB_4	160			
		04 Mar 2031	29 Sep 2032	576
2008 EV_5	*160*			
		18 Mar 2033	*22 May 2036*	*1161*
Earth	∞			

It is important to note that the sequence still contains 2008 EV_5, which is classified as a PHA and was selected as one of the candidate targets for the ARRM mission by NASA [54]. Also, although departure from it comes 11 days earlier, the stay time at 2008 EV_5 remains well beyong 100 days and among the longest of this particular sequence of asteroids.

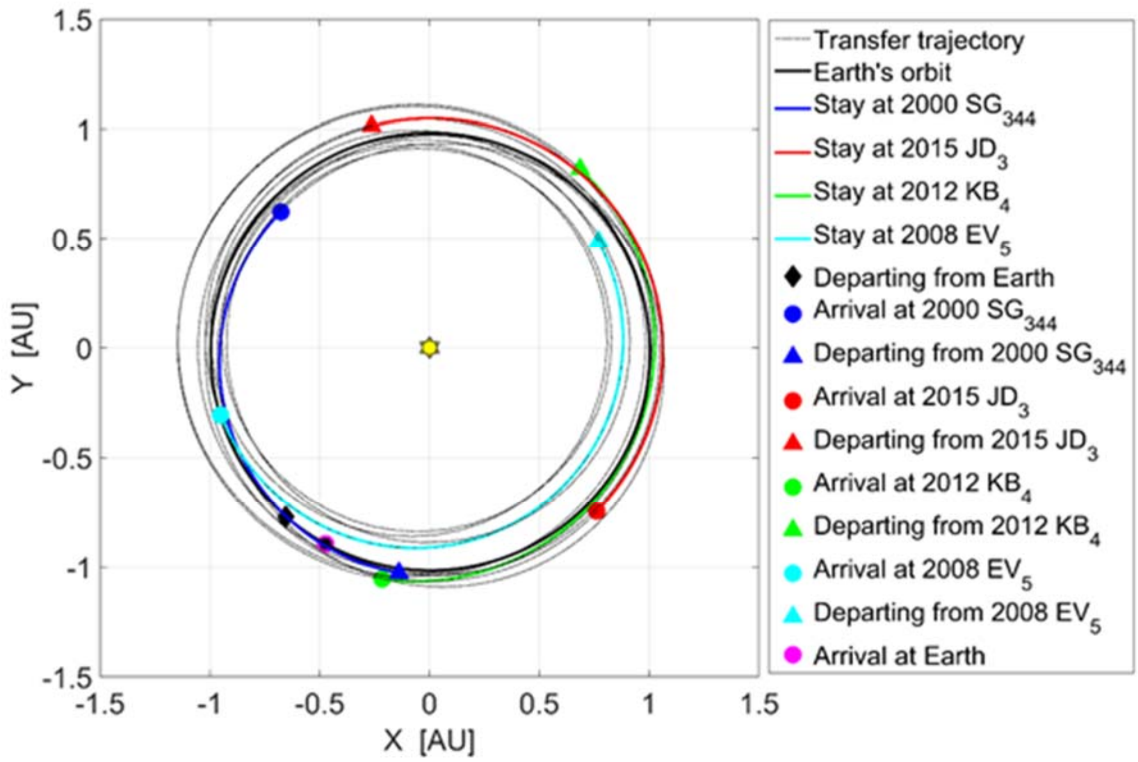


Figure 1 – Heliocentric view of the complete three-dimensional trajectory of the considered sequence.

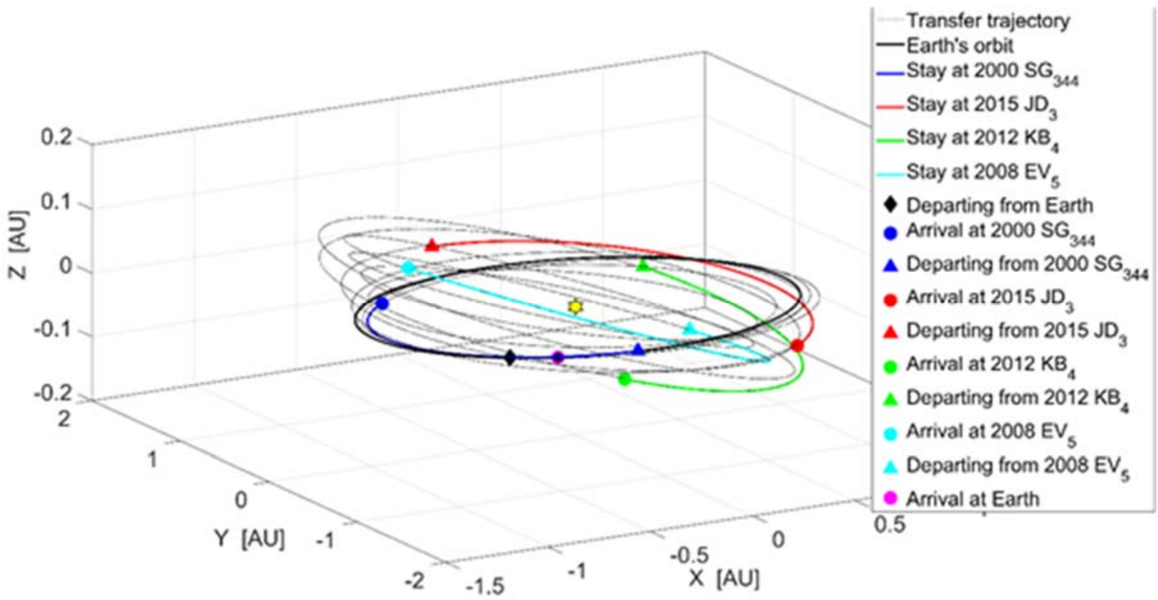


Figure 2 – Three-dimensional view of the complete three-dimensional trajectory of the considered sequence.

FLY FIVE – THE GOSSAMER PRINCIPLE OF SAILCRAFT DESIGN

The DLR GOSSAMER solar sail design is based on a crossed boom configuration with triangular sail segments made of a membrane manufactured from aluminized polyimide foil. A specifically designed combination of folding and coiling ensures that the deployed sail area can be held taut between the partly deployed booms.

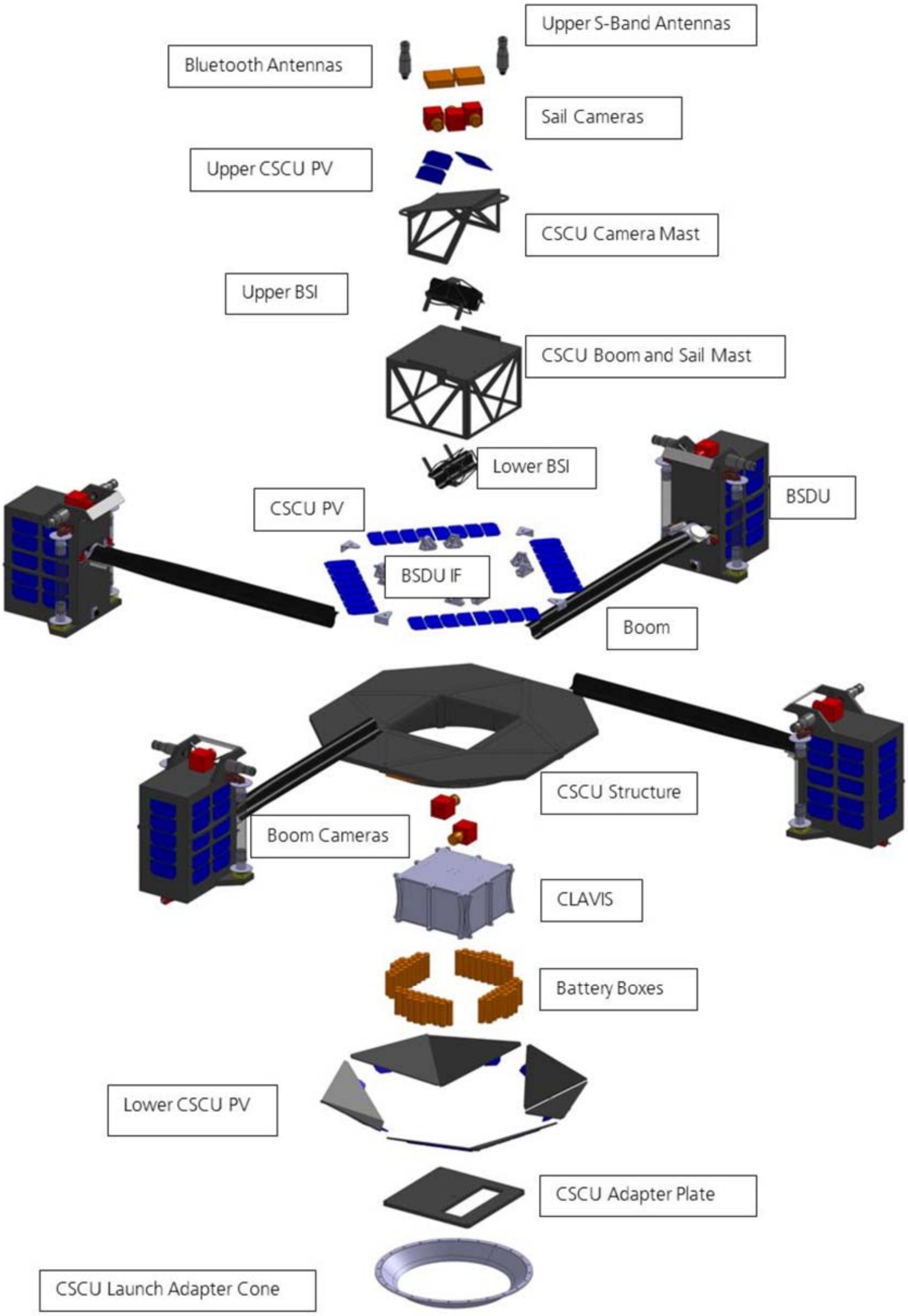


Figure 3 – GOSSAMER-1 PFM final design exploded view

A key design feature of the GOSSAMER solar sail is the Boom Sail Deployment Unit (BSDU) which is moving away from the Central Sailcraft Unit (CSCU) to uncoil the booms and unroll and unfold the sail segments. During deployment, four BSDUs synchronously move away from the central bus unit, each with two spools on which one half of either adjacent sail is stowed. (For a detailled discussion see [11][13][30][12] and references therein.)

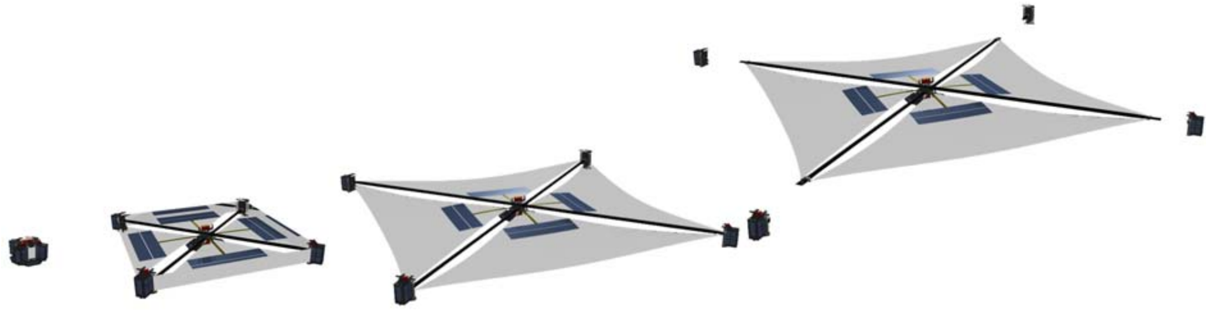


Figure 4 - GOSSAMER deployment sequence with BSDU separation

The BSDUs communicate through a wired interface while attached to the CSCU. After the connections are separated, the 5 sub-spacecraft communicate in a wireless network. The Umbilical connector and other harness technologies were jointly developed with the MASCOT project, the wireless communication concept and much of the BSDU electronics were re-used in the ROBEX project's Remote Unit.

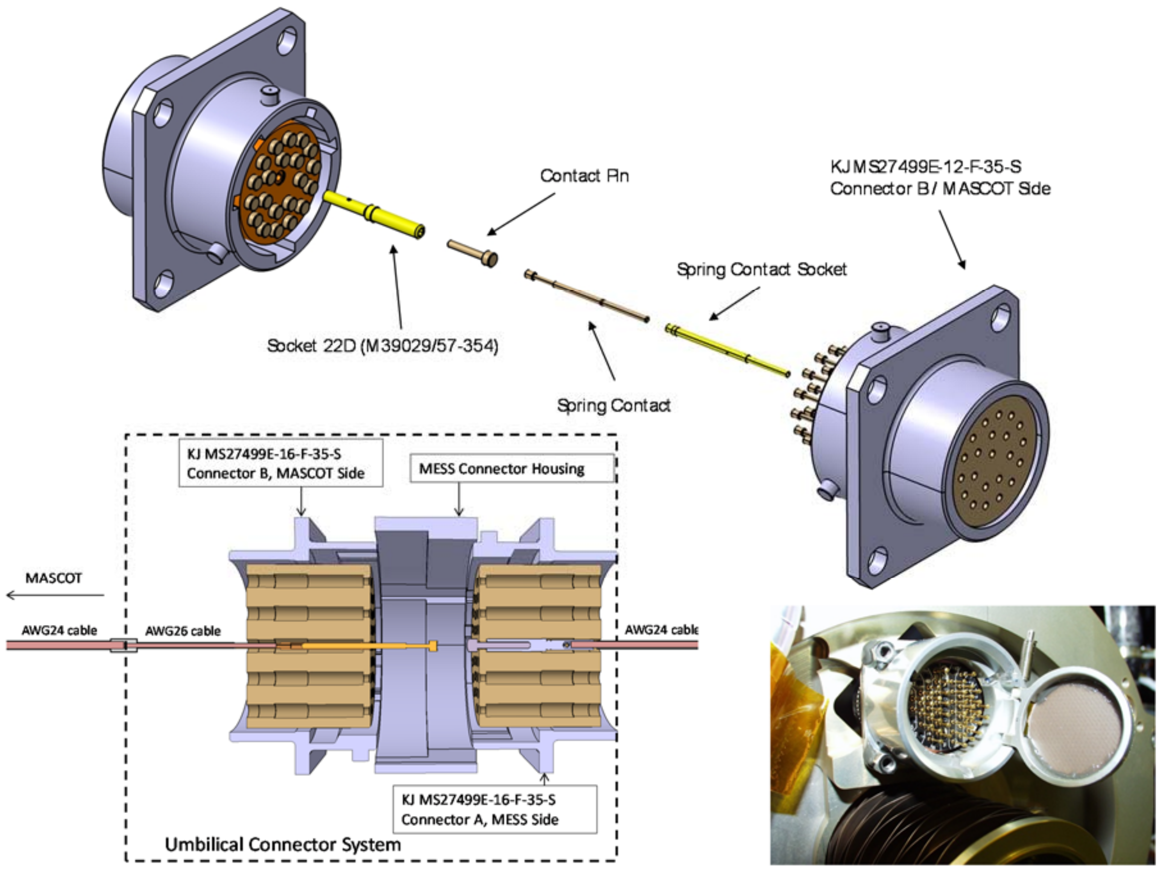


Figure 5 – Umbilical connector: PHILAE and MASCOT heritage

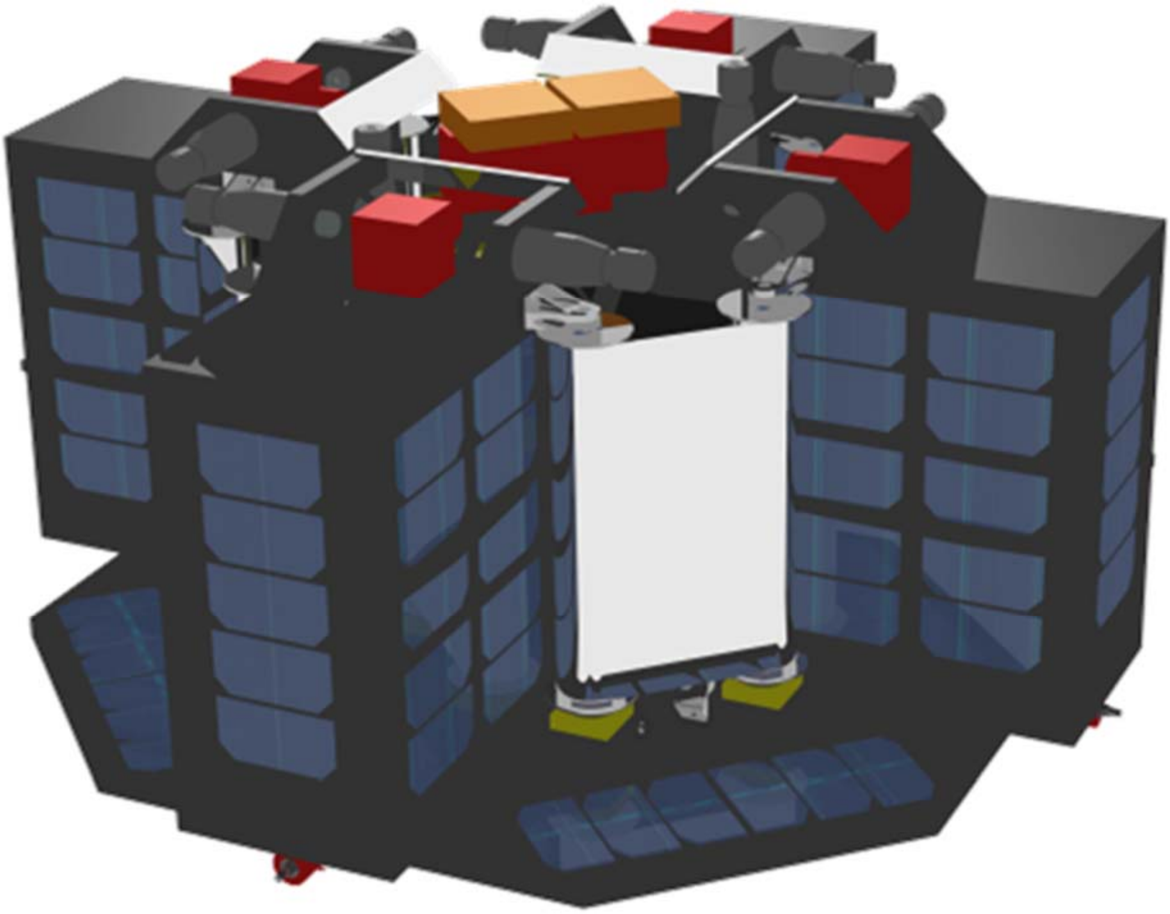


Figure 6 – GOSSAMER-1 independent free-flyer design launch configuration (CAD view of final PFM design status)

Un pour tous, tous pour un – shared resources multi-sub-spacecraft design

The controlled GOSSAMER deployment concept [11][31] requires synchronized operation of the four BSDUs moving away from the CSCU, and thus coordinated communication of all five elements. After separation of the BSDUs, a wireless network is used. [32][33][34][35]

Before separation, communication is also possible via wired connection through umbilical connectors from each BSDU to the CSCU. This interface between sub-spacecraft also supports power transfer from each BSDU to the CSCU. Either sub-spacecraft can provide power to the other and receive power from it. At the same time, it is in control of its own energy budget through control of the switch in the Power Distribution (PD) unit which feeds power to the Charging Network (CN).

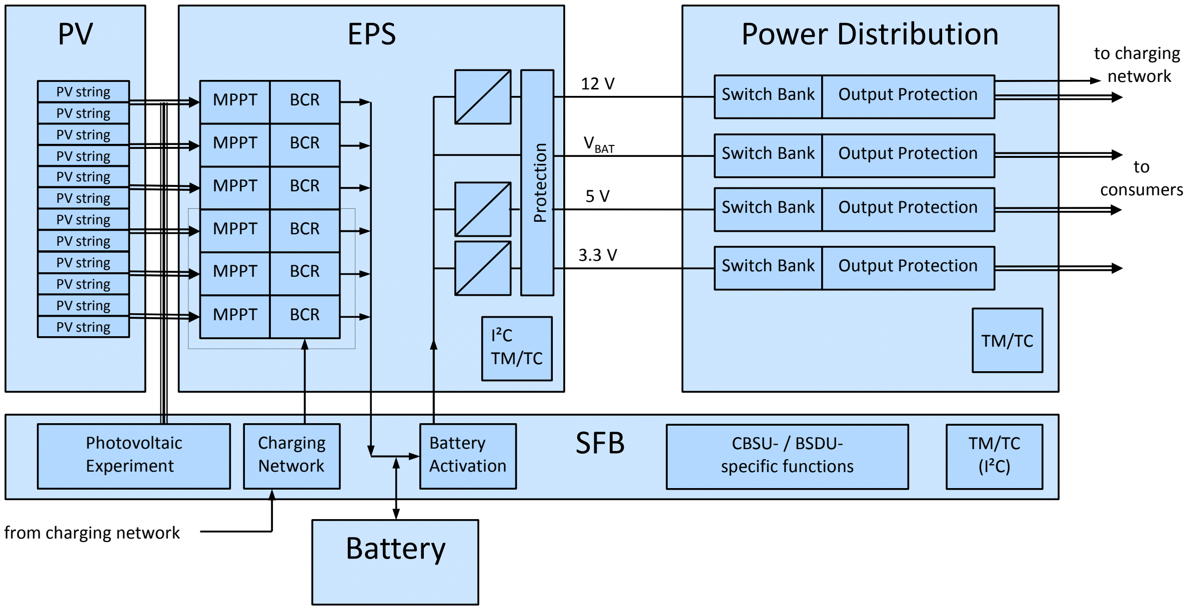


Figure 7 – GOSSAMER-1 common Electrical Power Subsystem (EPS) architecture of CSCU and BSDUs with Charging Network (CN) interfaces routed through Power Distribution (PD) units and a Special Functions Board (SFB) connecting to Photovoltaic arrays (PV) and the battery (Bat.)

The power receiving interface connects through a Special Function Board (SFB) to the same Battery Charge Regulator(s) (BCR) fed by the Photovoltaic (PV) arrays on the surface of each unit, similar to a maintenance charging connection from the Electroical Ground Support Equipment (EGSE) or the launch vehicle (LV). Each side of either interface is protected in a fail-safe manner, against energy loss and deep discharge of the feeding side's battery as well as against complete loss of energy flow.

The charging network effectively creates one spacecraft power subsystem from the energy gerneration, storage and distribution units of five self-sufficient spacecraft with their own complete and independent power subsystems and control units. This concept grew from the secondary passenger ("piggy-back") launch envisaged for GOSSAMER-1 with the QB50 project and the mission objectives assigned to it in the GOSSAMER Roadmap. For secondary passengers, a pre-determined separation attitude can not always be provided. In the Roadmap context, GOSSAMER-1 only has to demonstrate membrane deployment which does not require attitude control.

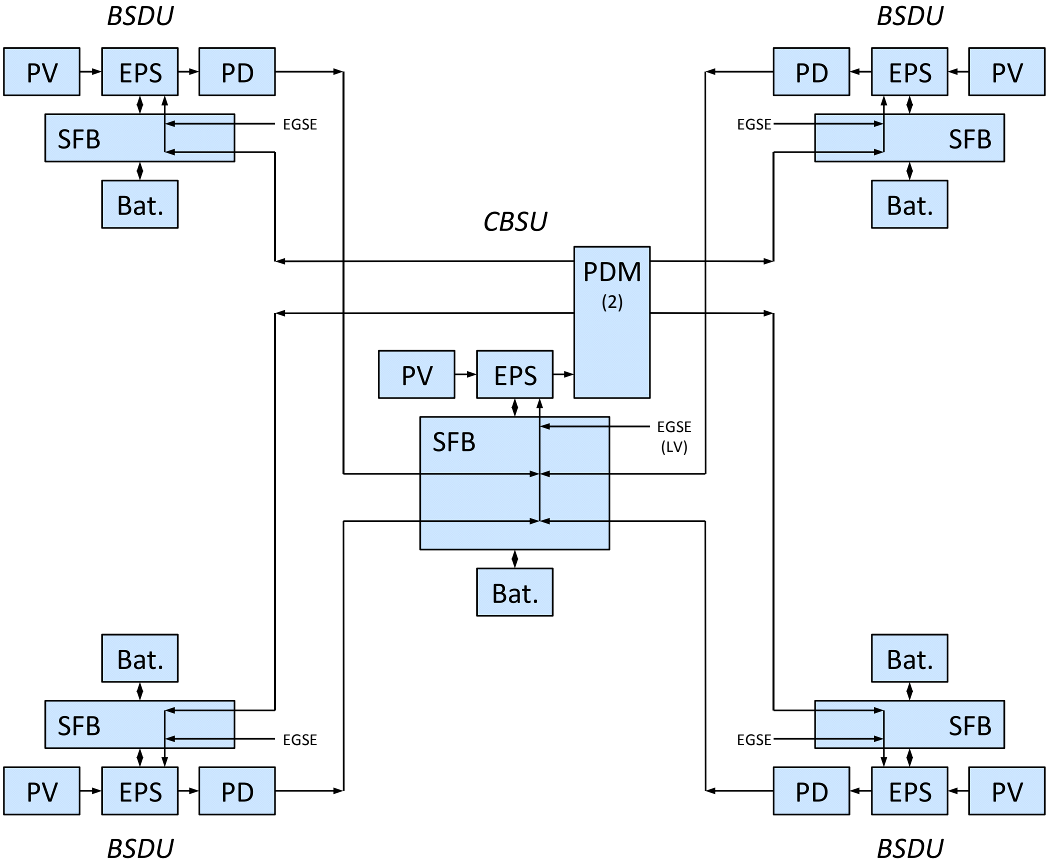


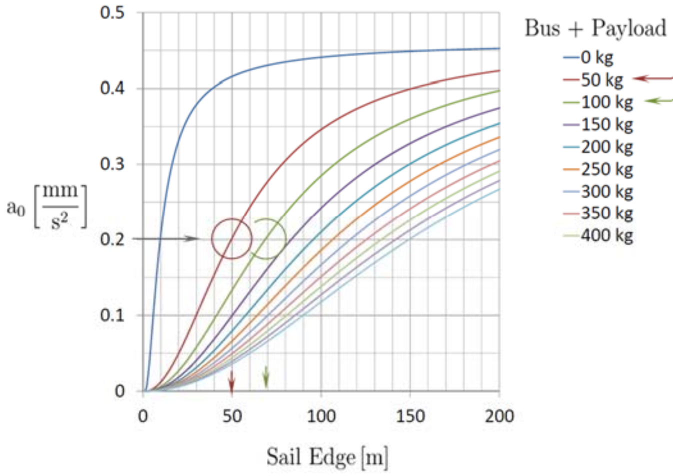
Figure 8 – GOSSAMER-1 Charging Network (CN) architecture with central Launch Vehicle (LV) interface, separate Electrical Ground Support Equipment (EGSE) interfaces to each unit

Thus, it was entirely possible that GOSSAMER-1 was deployed by the launcher such that e.g. only one BSDU is fully illuminated by the Sun. Even in this case, sufficient power supply for deployment could be achieved without the need to carry an excessively large and fully pre-charged battery.

Size Matters – Mission Design for a Realistic Near-Term Sail

The MNR mission scenario by Peloni et al. [10] is feasible using near-term solar sails with a characteristic acceleration of only a_c = 0.2 mm/s. In currently available technology such as introduced by GOSSAMER-1, this corresponds approximately to a (50 m)² sail, i.e. one of square shape and 50 m side length, carrying a science payload of approximately 20 kg, or a (70 m)² sail carrying about 60 kg. This science payload could be composed of heritage remote sensing instruments such as flown on conventional planetary science missions like ROSETTA or CASSINI.
However, the current state of small body science demands in-situ measurements for significant progress. Sailcraft due to their huge size and inherent agility limits can not as easily land or even perform a touch & go like HAYABUSA. But applying recent technology and MASCOT-style integration concepts, a combination of approximately 10 kg ‘orbiter‘ science payload and one or, respectively, five MASCOT landers of approimately 10 kg, each, appears feasible. By this combination, the gap between sail and soil can be closed, and the access frequency of asteroids to landers dramatically increased. Sail-based sample-return missions have also been studied

for many years, recently with focus on JAXA's Solar Power Sail with its PHILAE-sized lander. [5]



Bus (Gos-1 + X-Band)	≈30 kg
Mother Spacecraft Science Instruments	≈10 kg

Lander	
PHILAE	98 kg
SPS Lander	100kg
1 MASCOT	10 kg
5 MASCOTs	50 kg

Figure 9 – Payload performance of near-term solar sails based on GOSSAMER-1 technology [5]

Up and Away – Launching a Small Spacecraft to Escape Velocity

Due to the stringent mass requirements of solar sailing and the need to deploy very large structures, anyway, the resulting spacecraft launch configuration can be very compact and lightweight. A typical MNR design would fit the current standard 'micro-payload' secondary passenger slots of launch vehicles flying to GTO or other high altitude orbits, e.g. ASAP on European or ESPA on U.S. launchers. From Navsat-MEO, GEO or other high and moderately eccentric orbits, the sail could comfortable depart from Earth under its own thrust. With the high frequency of GTO launches, a reliable and affordable access to Earth departure becomes available at the expense of a small propulsion module for substantial perigee-lifting for easy spiral-out from Earth orbit or direct escape from GTO to $c_3 > 0$.

Dedicated launches would be an option in the case of missions requiring an extremely high c_3. Based on the current performance of Ariane 5 ECA [36], the performance for a maximum velocity escape trajectory has been calculated. For a dedicated launch, unnecessary standard equipment units such as the double launch adapter Sylda are removed. The performance for different c_3 values and an inclination of 6°, in case of launches from the Kourou spaceport are plotted in Figure 11. Payloads of 500 kg, 250 kg, and 50 kg, respectively, can be injected on escape trajectories with a c_3 of up to approximately 56 km²/s², 60 km²/s², and 64 km²/s².

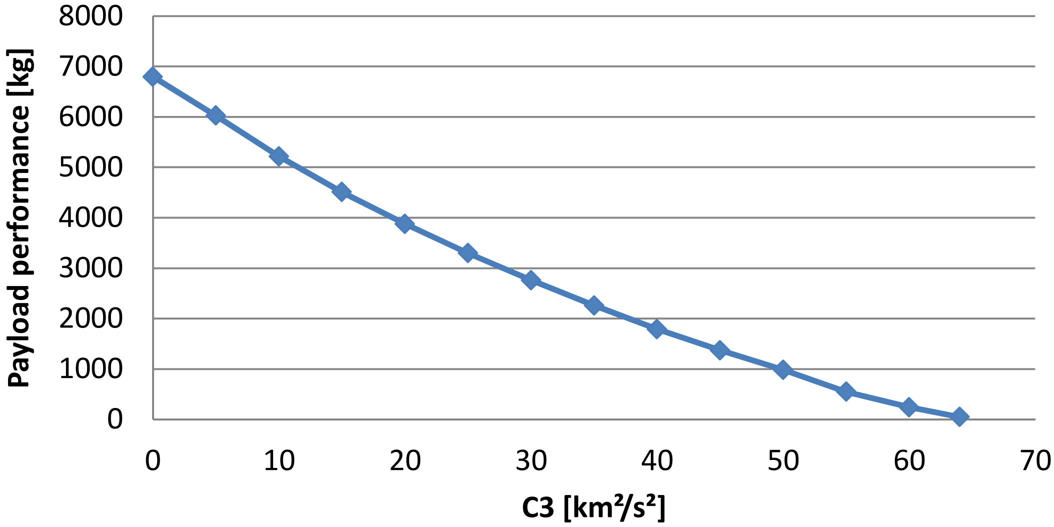


Figure 11 – Payload performance of Ariane 5 ECA for different C3 values and an inclination of 6°

The payload masses of 500 kg, 250 kg, and 50 kg, respectively, correspond approximately to a dual MNR launch (or a HAYABUSA2 reflight), a single MNR launch (or a NEW HORIZONS reflight), and a minimum sailcraft e.g. similar to NEAscout [37] (or a MASCOT-style high-density design chemical propulsion flyby spacecraft) with minimum deep space communication equipment added.

LANDERS

It is assumed that landers are separated like MASCOT, by a pre-set spring force. The solar sail trajectory is modified such for lander separation that the initial state vector relative to the asteroid ensures that the separated lander hits its mark, similar to MASCOT2 and AIM. [2] The sail may be in very slow fly-by, or in a stable solar-radiation-pressure displaced orbit or station-keeping. [38][39]

Multiple MASCOTs

Since the delivery of MASCOT for HAYABUSA2, several studies for MASCOT-style landers on other small solar system bodies have been conducted. [5][6] Also, there have been precursor studies for the development of MASCOT aboard MARCOPOLO, including payload concepts similar to MASCOT2. [40] These studies have created a wide repertoire for variation of the science instruments carried. Lander instrumentation can be adjusted e.g. for regolith investigations (MASCOT), radar tomography (MASCOT2), mineralogy, thermal surface properties charachterization related to the Yarkovsky and YORP effects, etc. It is then possible to select the lander most appropriate for the asteroid when it has been characterized remotely on or after arrival of the sailcraft.
Akin to the HAYABUSA2 MINERVA landers, [41] it is also possble to divide the payload mass down further, e.g. for CubeSat format landers or Mini-MASCOTs for very reduced tasks. A typical planetary defence related minimum science payload could consist of a planetary radar beacon, a miniature camera similar to those

qualified for GOSSAMER-1 [42] [43] [44], and a version of MARA [45] [46] adapted to the direct requirements.

Sample-Return Landers

NEA samples of the five asteroids visited can be returned by one larger lander shuttling between the NEA surfaces and the sailcraft. A reasonable design goal would be to pick up at least 2 samples per NEA and transfer them to a re-entry capsule aboard the sail. The technology to pick up and transfer asteroid samples already exists in several forms. It was demonstrated by the HAYABUSA mission, and has been further developed for HAYABUSA2 and OSIRIS-REx.

Solar Power Sail Lander Derivate

We evolve our design from the lander design for the JAXA Solar Power Sail mission to pick up samples from a Jupiter Trojan asteroid. [47][48] This design emphasizes in-situ analysis of samples due to the very long duration return journey from the orbit of Jupiter.

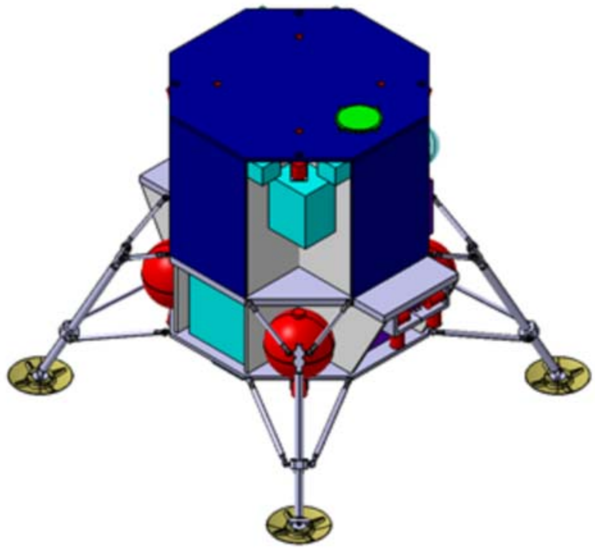


Figure 12 – 100 kg Trojan Asteroid Lander investigated as part of the JAXA Solar Power Sail (SPS) mission.

For the MNR scenario, a reduced in-situ suite of instruments can be considered due to shorter mission duration.
To account for added mass of a system capable of collecting a larger amount of samples multiple times, the original Trojan lander science suite is modified in order to remain within mass constraints.

- the in-situ science suite is cut down, reducing science payload mass, making it available for the sample collection and transfer system
- the collection of sub-surface samples is dropped, as this piece of hardware is the main mass-driver of the sampling/science suite. Subsurface sampling requires counter-thrusting systems, as well as being heavy on its own.

Samples would be collected and stored on the Lander and either transferred to the solar sail after each NEA rendezvous, or as a final package at the end of the multi-rendezvous mission. Both options have their advantages, resulting in a trade between simple systems (one-time transfer) and mission success / safety against failure during one of the NEA encounters (individual transfer after each mission).

The propulsion system is reevaluated for the multi-encounter mission and the lower gravity of the asteroids, compared to a 20…30km Trojan asteroid.
Based on the asteroid data listed in Table 4, a first estimate of the delta-v and thrust requirements is performed. Results are listed in Table BB, based on the worst-case assumption of parameters, as little information is available on the proposed asteroids; the targets may also change in the course of the mission design, or even during its execution. By using worst case assumptions, and adding sufficient margins (100%), the Lander allows for mission flexibility.

Table 4 – Asteroid Data. In most cases rotation rate is unknown (assumed = 0.1hrs). Gravity is based on worst case assumptions and albedo range where unknown is set as range for asteroids.

Asteroid data								unknown = 0.1 hrs	
name	magnitude	diameter max [m]	diameter min [m]	density (max) [kg m-3]	mu [m3 s-2]	eroid albedo rar		rotation period [sec]	gravity [m s-2]
2000SG344	24.7	76.29	24.13	4000.00	0.06208	0.04	0.4	360	4.26584428377671E-05
2015JD3	25.6	50.41	15.94	4000.00	0.01790	0.04	0.4	360	2.81841536851192E-05
2012KB4	25.3	57.88	18.30	4000.00	0.02710	0.04	0.4	360	3.23597381223943E-05
2008EV5	20	359.06	359.06	4000.00	6.47059	0.137	0.137	13410	0.00020075885516
2014MP	26	41.93	13.26	4000.00	0.01030	0.04	0.4	360	2.3442557954472E-05

Table 5 – d elta-v budget for 5 asteroid landings and sample-retrievals. The fifth is added as additional margin. All delta-v are estimated according to the worst case scenario for horizontal and vertical velocity. Included are three 10m hops per asteroid. The total margin added is 100% to increase system flexibility for large asteroids and faster rotating targets.

delta-v budget	hopping (3-times)	descent (vertical+horizontal)	ascent (vertical + horizontal)	margin [%]	total [m/s]
2000SG344	0.087627163087698	0.105617932801145	1.33159560436816	100	3.04968140051402
2015JD3	0.07122603220257	0.062900698082018	0.278209754406865	100	0.824672969382905
2012KB4	0.076320068540529	0.074943293696123	1.01011856448112	100	2.32276385343554
2008EV5	0.190096275420735	0.355258458889765	0.16823478127909	100	1.42717903117918
2014MP	0.064958913413056	0.049614864636168	0.731766212049452	100	1.69267998019735
TOTAL					**9.316977234709**

The delta-v requirements include for every asteroid, delta-v to cancel out horizontal and vertical velocity from a release position (5000m altitude) to the landing site of the asteroid, as well as the reverse transfer to return to the solar sail. Also included is delta-v to perform a number of hops (three times) on the asteroid surface with a distance of ~10m.

Due to the multi-rendezvous mission duration, propulsion system leakage becomes an issue. The use of isolation valves during cruise phase allowed for low leakage in a high-pressure cold gas system on the Trojan SPS mission. However if multiple landings are performed at different times throughout the entire mission duration, this approach is not possible. The use of liquid propellant systems is one alternative to reduce leak-rates, as liquid stored propellants are more suited for long-term storage.

An analysis of propellants is performed shown propellant mass, volume, and power requirements in a system trade. Power requirements are based on the heating enthalpy of the liquid and the mass-flow rate needed to provide sufficient thrust for both hovering and ascent from the asteroid surface. Thrust is designed according to the maximum gravity asteroid (in this case 2000 EV_5); it is 40 mN for a thrust capable of providing 2 times the gravitational acceleration (2 x 0.18 m/s²) to a 100 kg lander system.

Propellant mass is traded with propellant volume and power requirements for a number of potential propellants.
Different options can be considered as best suited for use as propellant in the NEA Lander.
Propellants such as Hexafluoroethane (R116) or Sulfurhexafluoride offer low power and storage volume, with reasonable propellant mass requirements. Lower propellant mass is available by taking higher heating power requirements into account (see Table 6 for comparison of a number of suitable propellant options)

Table 6 – Propellant-dependent parameters.

fluid	CarbonDioxide	SulfurHexafluoride	NitrousOxide	n-Butane	R116	R134a
operating pressure (gas) [bar]	21.374571	1.268313	18.105941	0.1	6.600507	0.1
operating pressure (liquid) [bar]	91.1503	51.304813	89.695	51.756	43.528	54.65208
ISP [sec]	52.756973	45.039218	55.114462	72.23125	44.366721	53.322624
lq. propellant density (low temp) [kg/m3]	844.708806	1471.472485	841.961154	586.24228	1013.37774	1249.72769
maximum heating temperature [K]	352.731089	346.125581	356.9907	423.000624	328.798742	393.015567
ideal heating power [W]	19.90552	8.516516	19.016496	21.430974	7.411362	15.388033
propellant mass [kg]	1.925384	2.251571	1.843791	1.409961	2.285306	1.905155
propellant volume [cc]	2279.346578	1530.147881	2189.876586	2405.082687	2255.137494	1524.456364
propellant heating energy [Wh]	140.074759	59.796556	133.877998	151.127809	52.038597	108.245559
propellant mass fraction [-]	0.019254	0.022516	0.018438	0.0141	0.022853	0.019052

Based on these results and an adapted propulsion system design of the Trojan lander for use with liquid stored propellants is considered (Figure 13). The selected propellant is Sulfurhexafluoride due to low volume and heating power requirements. Although other options such as R116, R134a and n-Butane are not that different.
Operating pressure is low, at 51 bar at a nominal operating temperature range around 10°C for the liquid stored propellant, and 1.3 bar on the generated gas side of the system.
The required propellant mass of 2.25 kg easily fits into the 100 kg lander mass budget. The total propulsion system dry mass is 10.0 kg, including propellant tanks, valves, tubing and battery for heating the propellant. Tank Volume is 1630 cm³, including 100 cm³ of buffer tank volume. Power requirements (considering a conservative 70% system efficiency) are 12 W (85 Wh total). This can be handled by the on-board battery without issues, especially when considering the use of low power solenoid valves is possible at these lower operating pressures.
The main reason for these low power requirements is the low gravity and therefore thrust requirements needed to allow the lander to operate around the targeted NEAs.

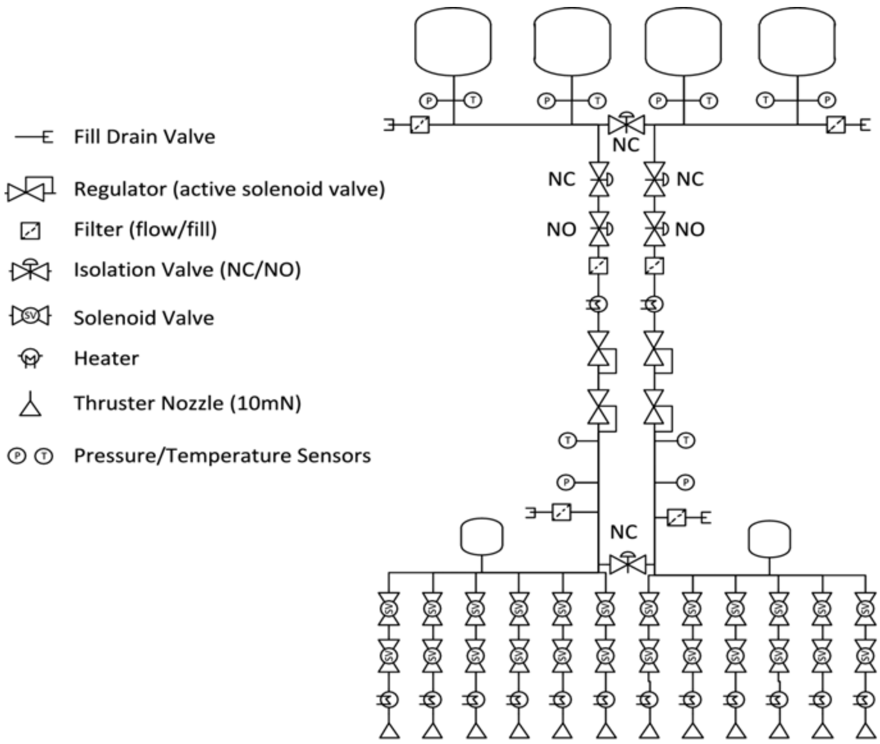


Figure 13 – Schematic of a liquid propellant based lander propulsion system operating at vapor-liquid equilibrium pressure.

This system provides an alternative to the N_2 system considered for the Trojan SPS Lander, with the added benefits of low pressure and leakage for the cost of additional heating power. The system mass is reduced to 12.3 kg, compared to the 26 kg needed in the SPS Trojan mission.

While the Trojan lander had to trade descent/ascent times with propellant usage for powered descent, this lander can rely on solar power for operation due to its orbit in proximity to 1 AU; not 5.2 AU.

Resource Sharing of Lander(s) and Sailcraft

Following the BSDU-CSCU concept of GOSSAMER-1, many resources can be shared with the CSCU in cruise and the CSCU-BSDUs before sail deployment.

Landers which have to expect rough terrain and unexpected shadowed areas (cf. PHILAE) require a relatively large battery while a deployed sailcraft operating in deep space in almost all cases of nominal operation only needs a relatively small capacity battery to buffer brief high-power peaks. Thus, the batteries of the still attached lander(s) can support the CSCU during deployment when the BSDUs have already separated from it.
Similarly, the sailcraft can generate its power after deployment from ultra-lightweight membrane-mounted photovoltaics similar to the GOSSAMER-1 Photovoltaics Experiment (PVX) or the GOSOLAR technology currently under development. [12] These thin-film CIGS photovoltaic cells have a lower efficiency, currently ~12%, than rigid triple-junction photovoltaic cells which are currently approaching 30%. Thus, thin-film generators, although still significantly lighter, require about three times the

array area for a given power output. Rigid triple-junction photovoltaic cells are therefore used for the pre-deployment photovoltaic generators of high-density small spacecraft design GOSSAMER-style sailcraft for secondary payload flight opportunities. Area-efficient photovoltaics are also required for mobile asteroid landers. The landers' photovoltaics generators exposed to the outside in launch configuration and after BSDU separation can therefore be used as a significant part of the pre-deployment, respectively in-deploment photovoltaics of the CSCU.

Science instruments of the landers, in particular panoramic cameras and thermal infrared sensors, can provide services on an operational spacecraft which are normally only designed into demonstrator spacecraft to monitor sail deployment and membrane ageing, cf. [13][49].

Suitably designed and/or oriented instruments of the landers still attached can also double as 'orbiter' instruments, e.g. to monitor the asteroid in the vicinity of the sailcraft without the need to turn for the pointing of a boresighted sailcraft camera.

These and more opportunities for resource sharing can be used to adapt lander designs similar to MASCOT, PHILAE, or the Solar Power Sail Trojan lander into GOSSAMER-style integrated sub-spacecraft performing a common mission.

EXERCISE

Solar Sail – Online Change of the Mission

Diversion to fly-by or rendezvous

Due to the early impact in 2027, this year's fictitious impactor 2017PDC requires more extensive modifications of the sequence presented above, which we for now have to relegate to future work. However, the asteroid 2011 AG_5 used for the PDC'13 exercise [50] easily matches the existing 5-NEA-sequence. The modified sequence also demonstrates the target flexibility unique to solar sailing.

The last leg to 2014 MP shown in Table 1 has again been removed to add a leg to the potentially-hazardous asteroid 2011 AG_5, which was one of the two case studies considered during the Planetary Defense Conference 2013. At the time of the conference, the potential impact was expected to be on February 3rd, 2040. Table 7 shows the properties of all the encountered bodies of the new considered sequence.

Table 7 – Properties of all the encounters of the new considered sequence.

Object	2000 SG_{344}	2015 JD_3	2012 KB_4	2008 EV_5	2011 AG_5
Orbital type	Aten	Amor	Amor	Aten	Apollo
Semi-major axis [AU]	0.977	1.058	1.093	0.958	1.431
Eccentricity	0.067	0.009	0.061	0.083	0.390
Inclination [deg]	0.111	2.730	6.328	7.437	3.681
Absolute magnitude [mag]	24.7	25.6	25.3	20	21.8
Estimated size [m]	35 – 75	20 – 50	20 – 50	260 – 590	110– 240
EMOID [AU]	0.0008	0.054	0.073	0.014	0.0002
PHA	no	no	no	yes	yes
NHATS	yes	yes	yes	yes	no

A methodology similar to the one described in Sullo et al. [51] has been used for this study to compute the leg to 2011 AG_5. First, a constant-mass low-thrust transfer between 2008 EV_5 and 2011 AG_5 has been computed by means of the indirect optimization approach. The time of flight and the initial values of the costates have been determined through a particle swarm optimization (PSO) [52]. For this scenario, the orbits of both objects are considered coplanar. That is, the orbital plane of 2011 AG_5 has been rotated and projected onto the one of 2008 EV_5. Moreover, the maximum acceleration given by the propulsion system was set to a_{max} = 1 mm/s². Starting from the low-thrust solution, the homotopy-continuation approach described in [51] has been used to find a coplanar solar-sail transfer with a_c = 0.2 mm/s². Then, the Automated Trajectory Optimiser for Solar Sailing (ATOSS) [53] has been used to find the final three-dimensional (3-D) trajectory by first changing the orientation of the orbital plane and then changing its inclination.

The total mission duration is now 4398 days, about 12 years, and the sailcraft arrives at the final target object on 25 May 2037, about 3 years before the potential impact. The complete trajectory of the overall sequence is shown in Figure 14, whereas Figure 15 shows only the last transfer leg between 2008 EV_5 and 2011 AG_5. Figure 16 shows the acceleration history in the orbital reference frame needed during this last leg, whereas Table 8 shows the updated mission parameters. It is important to note that the sequence still contains 2008 EV_5, which is classified as a PHA and was selected as one of the candidate targets for the ARRM mission by NASA [54].

Table 8 – Mission parameters for the considered sequence with the last leg to 2011 AG_5.

Object	Stay time [days]	Start	End	Time of flight [days]
Earth	//	10 May 2025	26 Feb 2027	657
2000 SG_{344}	123	29 Jun 2027	06 Sep 2028	436
2015 JD_3	164	18 Feb 2029	24 Sep 2030	584
2012 KB_4	160	04 Mar 2031	29 Sep 2032	576
2008 EV_5	*7.5*	*07 Oct 2032*	*25 May 2037*	*1691*
2011 AG_5	*987 to ⊕*			

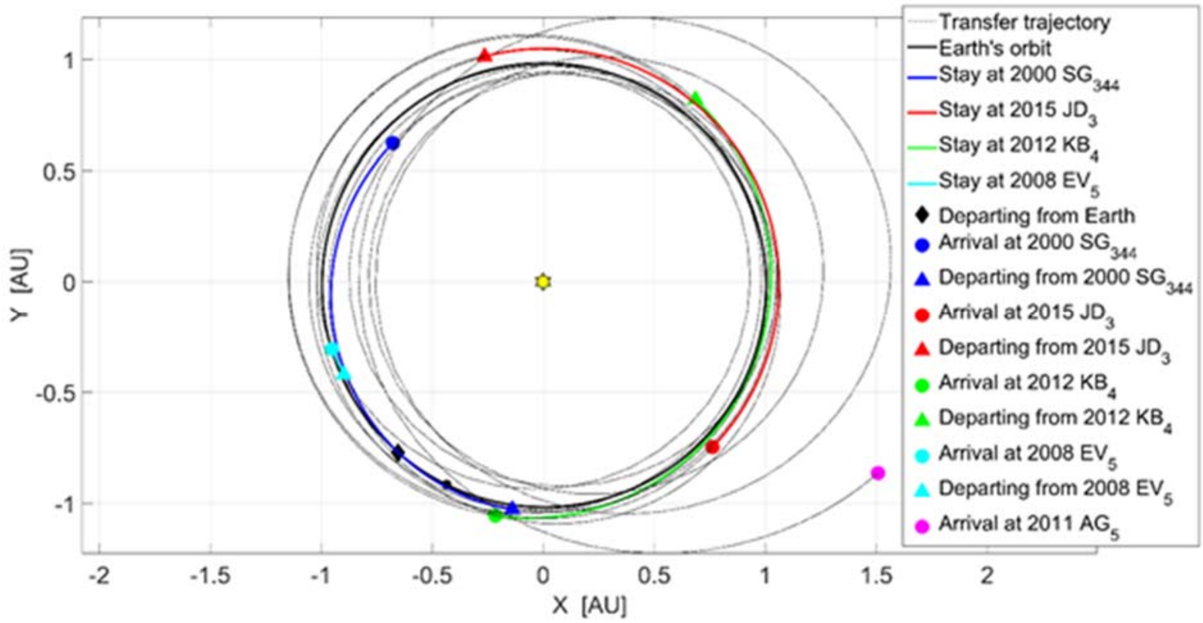


Figure 14 – Heliocentric view of the complete three-dimensional trajectory of the considered sequence. Ecliptic-plane view.

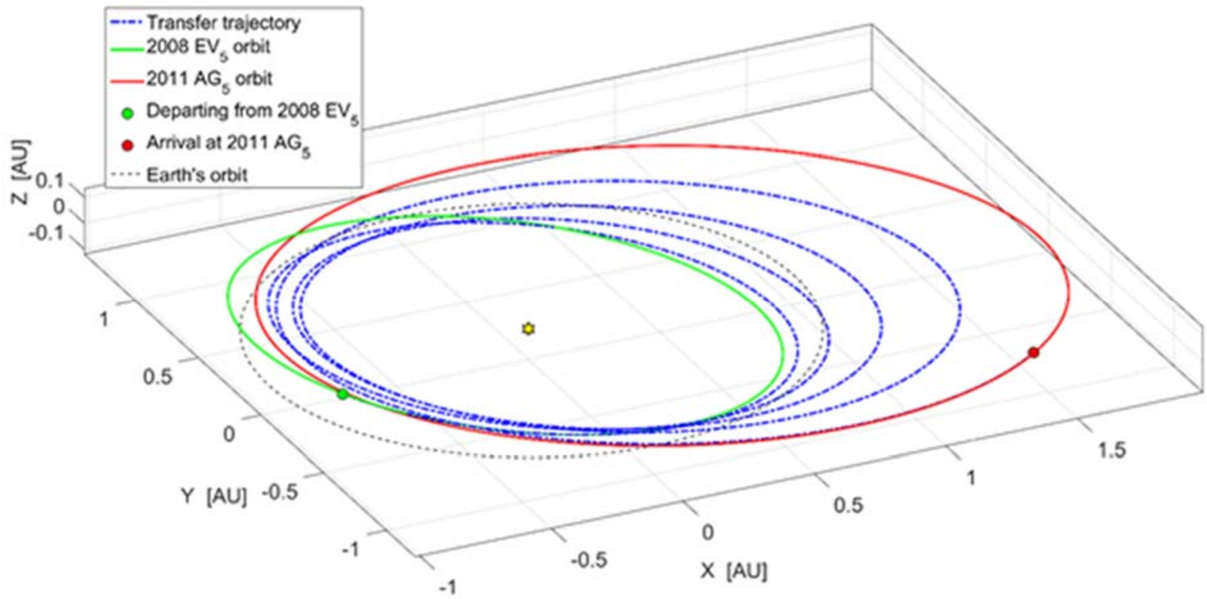


Figure 15 – Heliocentric view of the last transfer leg between 2008 EV_5 and 2011 AG_5.

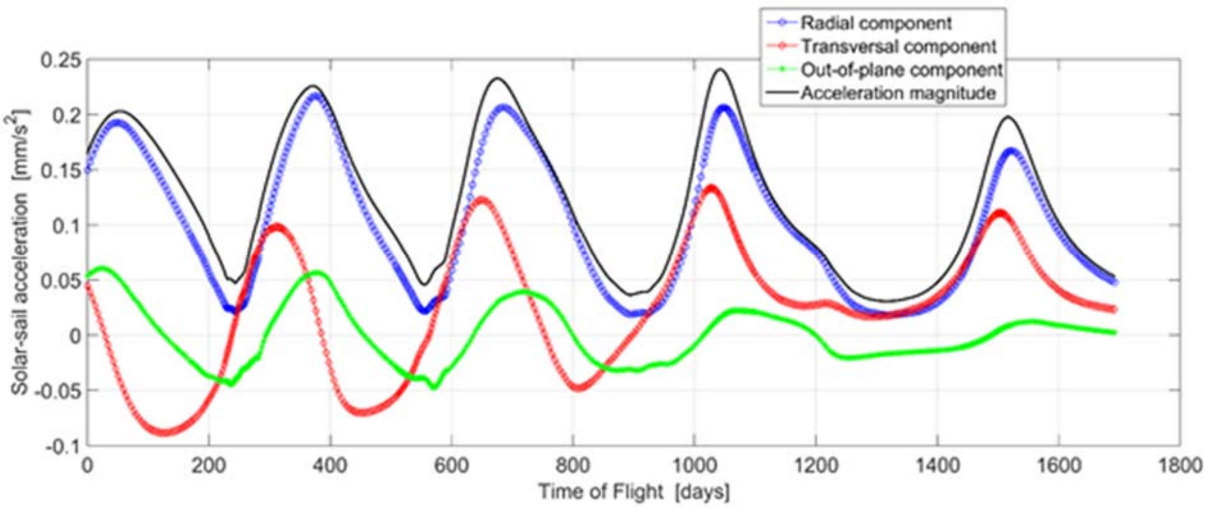


Figure 16 – Acceleration history during the last transfer leg between 2008 EV_5 and 2011 AG_5.

PDC Exercise 2017 Fictitious Impactor

A second case study is considered which targets the fictitious potentially-hazardous asteroid 2017 PDC introduced at the Planetary Defense Conference 2017. The potential impact of such fictitious object is expected to be on July 2027 [55]. Table 9 shows the ephemerides of 2017 PDC.

Table 9 – Ephemerides of 2017 PDC.

Object	2017 PDC
Semi-major axis [AU]	2.24
Eccentricity	0.607
Inclination [deg]	6.297
Right ascension of the ascending node [deg]	298
Argument of periapsis [deg]	312
Mean anomaly [deg]	332
Epoch [MJD]	57940
Absolute magnitude [mag]	21.9
Estimated size [m]	110 – 240

Because of the date of impact, the multiple-NEA rendezvous mission presented in [10] is not a good candidate. Therefore, from the same study presented in [10], a different sequence has been optimized and considered as a potential starting point for a leg to 2017 PDC. Table 10 shows the mission parameters for such sequence.

Table 10 – Mission parameters for the original sequence for 2017 PDC case study.

Object	Stay time [days]	Start	End	Time of flight [days]
Earth	//			
		13 Aug 2020	26 Apr 2022	621
2005 TG_{50}	128			
		02 Sep 2022	13 Jan 2024	498
2015 JF_{11}	104			
		25 Apr 2024	10 Jun 2026	776
2012 BB_{14}	139			
		28 Oct 2026	02 Aug 2028	644
2014 YN	//			

As for the previous case study, the mission is changed after the second leg to go towards 2017 PDC. The same methodology to find a transfer leg from 2015 JF_{11} to 2017 PDC has been used in this case, starting from a low-thrust solution with a_{max} = 2 mm/s². Nevertheless, this time the orbit of the target asteroid is too different from the one of the departing object and no good solution has been found for this case study. In fact, a solar-sail transfer leg with a_c = 0.73 mm/s², considering the orbit of 2017 PDC coplanar with the one of 2015 JF_{11}, needs more than 2000 days to be performed. That is, a sailcraft with a much larger characteristic acceleration than the one considered in this study would arrive at the target asteroid on 21 August 2030, which is about three years after the predicted impact with the Earth. Figure 17 shows the aforementioned transfer leg with the non dimensional acceleration vector.

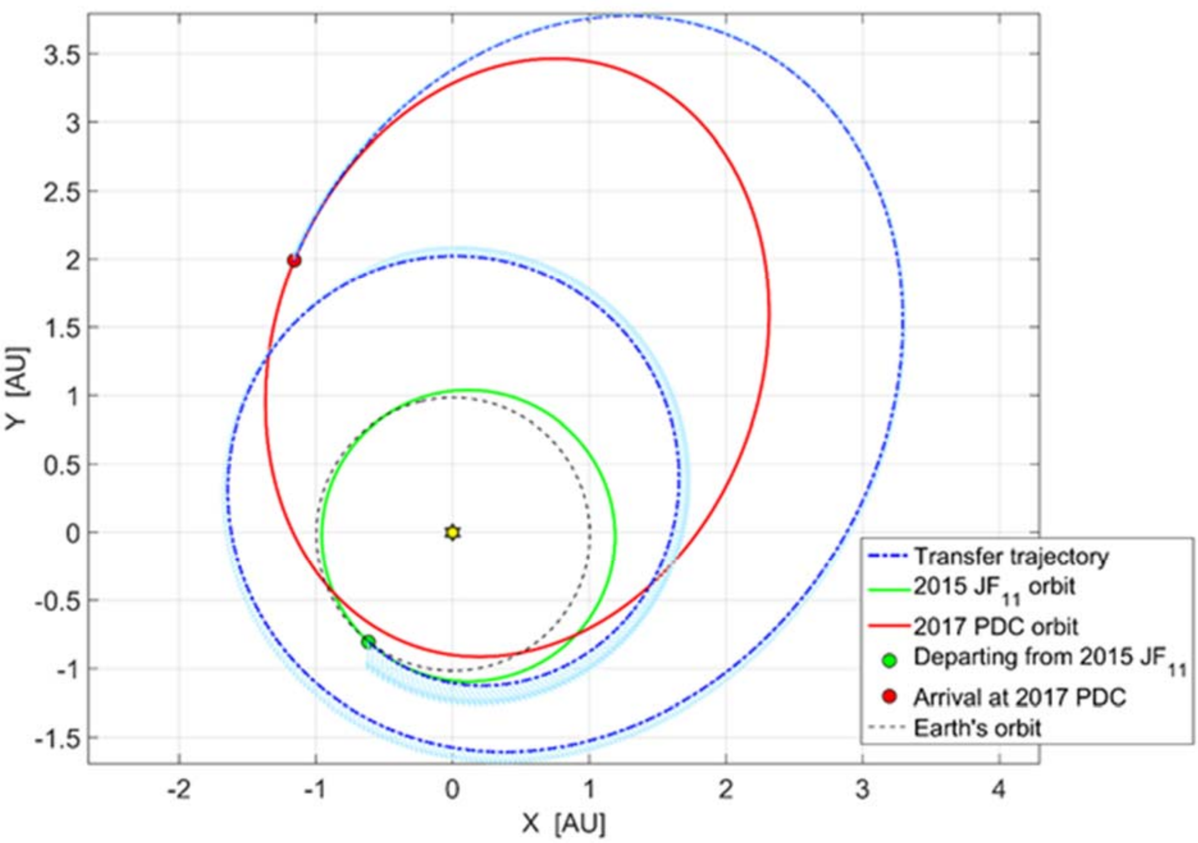


Figure 17 – Heliocentric view of the coplanar leg between 2015 JF_{11} and 2017 PDC. Characteristic acceleration $a_c = 0.73$ mm/s^2.

CONCLUSIONS

We outlined a synergetic development path of small spacecraft solar sails and nano-scale asteroid landers enabling a substantial increase in the number of NEAs studied by planetary science in a dynamic manner which allows in-flight adjustment of the choice of rendezvous targets. The capability to change targets in flight also allows a mission already in flight to respond to extreme events such as a probable Earth impactor being discovered. It may also follow changing commercial interest in this manner. Within the capabilities of near-term first-generation sailcraft technolgy, the small spacecraft design concepts of GOSSAMER-1 and MASCOT enable a sailcraft performance sufficient to achieve 5 NEA rendezvous of at least 100 days, each, in 10 years by one spacecraft. Each rendezvous includes a target-adapted one-way nano-lander delivery or a sample pick-up at each target by a larger shuttling lander. The small spacecraft approach enables the use of surplus launcher payload capability in the geostationary and high Earth orbit market with a potential of 10's of launches per year. If the spacecraft concept here presented were serialized in a manner akin to similar-sized communication satellite constellation spacecraft, the number of NEAs visited and studied in-situ could be increased by orders of magnitude within a few decades.

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