Deep-space CubeSats: thinking inside the box

Roger Walker and colleagues consider the potential for sending nanospacecraft into deep space.

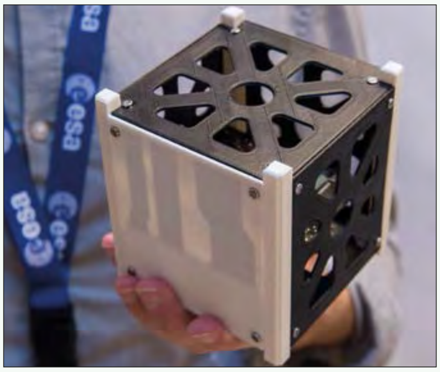
Since the introduction of the CubeSat standard in the early 2000s, there has been a proliferation of nano-/small microsatellites in low Earth orbit, with 100–300 or more launched annually and at a growing rate (according to reports from SpaceWorks and Euroconsult). CubeSats have reduced entry-level costs for space missions in low Earth orbit (LEO) by more than an order of magnitude (see box "CubeSats: small but perfectly formed"). This has brought new players into the space sector and launch opportunities for CubeSats have increased significantly in the last decade to address the associated demand. With more piggyback opportunities, platform capabilities and small payloads have rapidly advanced to a level that are suitable for real operational missions. Niche commercial applications are emerging based on operating multiple CubeSats together in distributed systems, e.g. constellations and swarms.

How far can this new paradigm be extended from the safety of LEO out to cis-lunar and deep space, and what unique new missions can be performed? Piggyback opportunities for CubeSats to lunar orbit and interplanetary space are already becoming available, and innovative miniaturized technologies are being developed to overcome the severe technical challenges of deep-space missions. Performance has reached a level where the first interplanetary nanospacecraft mission (NASA's Mars Cube One – MarCO) was launched in May 2018 as part of the InSight mission (Klesh & Krajewski 2018). And, as was the case for LEO, it is expected that there will be an order of magnitude reduction in the entry-level cost of interplanetary missions, thus paving the way to new mission applications and architectures based on distributed systems of deep-space nanospacecraft. This article addresses some of the missions that may be performed with such distributed systems, their system architectures and enabling technologies, based on a number of studies completely within the frame of the ESA General Studies Programme on lunar and interplanetary CubeSat mission concepts.

CubeSats: small but perfectly formed

CubeSats are modular spacecraft, using multiple standard-sized units of 10×10×10 cm (figure 1); the size of the resulting spacecraft is measured by the number of units, e.g. 3U or 6U. The capabilities of these satellites have increased significantly in recent years, notably in pointing, propulsion and communications as well as the availability of compact optical/radio frequency/environment payloads.

CubeSats are launched inside a standard container which simplifies launcher accommodation while ensuring safety for the primary passenger, thus facilitating widespread low-cost piggyback launch opportunities on many different launch vehicles worldwide. At the same time, the extensive use of miniaturized electronics, sensors/actuators and a stacked integration approach has cut spacecraft production and integration costs. The dramatically lower entry-level costs mean that the true power of CubeSat-based nano- or small microsatellites lies in their operation as distributed systems, such as constellations or swarm formations.



1 Example of a 1U CubeSat shell. There is a weight standard too – the finished launch unit must not exceed 1.33 kg. (ESA/G Porter)

These studies have involved either mother–daughter system architectures, where the CubeSats are carried to a target destination such as lunar orbit or to a near-Earth object (NEO) on a larger spacecraft and deployed at the target in order to fulfil their mission, or a completely stand-alone system executing its own mission. The mother–daughter architecture alleviates the technical challenges of propulsion, long-range communication and deep-space environment survivability, because the host spacecraft provides resources and accommodation during the cruise, as well as communications to Earth ground stations, in conjunction with local inter-satellite links with the deployed CubeSats. For a stand-alone deep-space CubeSat (i.e. where there is no mothercraft), these challenges have to be tackled and suitable technology solutions identified and/or developed.

"The first interplanetary nanospacecraft mission was launched in May, to Mars"

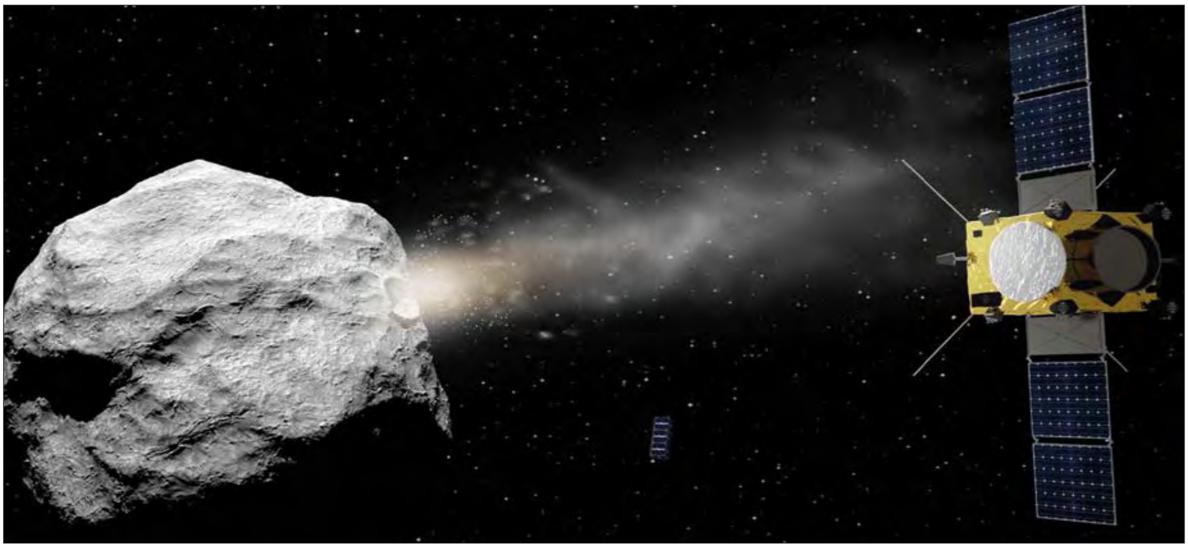
There are presently no dedicated small launchers that can cost-effectively inject nanospacecraft onto specific near-Earth escape or Earth escape trajectories. For the time being, piggyback flight opportunities must be found, either on a launcher upper stage carrying a primary spacecraft, or on the primary spacecraft itself as part of its mission. However, these opportunities are rare, and the constraints on nanospacecraft deployment can significantly influence the Δv (velocity impulse) and transfer window needed to reach the final mission destination – and hence the feasibility of such a mission. Table 1 shows our assessment of a range of missions that could be performed beyond Earth orbit for a given set of piggyback flight opportunities.

Asteroid Impact Mission

Within the ESA General Studies Programme, the third edition of the Sysnova technical challenge focused on Cubesat Opportunity Payload of Intersatellite Networking Sensors (COPINS). The overall objective of the challenge was to investigate new mission concepts involving a number of CubeSats operating together in interplanetary space in support of the objectives of the proposed ESA Asteroid Impact

1 Mission scenarios with respect to different piggyback flight opportunities

piggyback on	launcher	primary spacecraft	CubeSat deployment	CubeSat final destination	propulsion type	feasibility (≤12U)
spacecraft	PSLV/Ariane 6	Lunar Pathfinder	highly eccentric lunar orbit	low lunar orbit/ Earth–Moon L2	chemical mono-propellant	yes
spacecraft	Soyuz/Ariane 6	HERA	Didymos binary NEO	Didymos binary NEO	cold gas	yes
launcher	Ariane 5	JUICE	Venus-bound trajectory	Venus high-altitude atmosphere swing-by	chemical mono-propellant	yes (4 × 3U CubeSat entry probes with microcarrier)
launcher	Soyuz/Ariane 6/ Falcon 9/Atlas V	various telecom	geostationary transfer orbit	lunar orbit/ NEO rendezvous	electric	no (excessive Δv, time and radiation dose)
launcher	Ariane 6	various astronomy	transfer to Sun–Earth L2	NEO rendezvous/ Sun–Earth L5/ heliocentric (<1 au)	electric	yes (use of L2 halo orbit for waiting until transfer to target)
launcher	Space Launch System	Orion	lunar transfer trajectory	lunar orbit/ NEO rendezvous	electric	yes
launcher	Space Launch System	Orion	lunar transfer trajectory	lunar orbit/ NEO rendezvous	chemical	no (excessive Δv)
launcher	Proton/Atlas V	ExoMars/ Mars 2020	Mars transfer trajectory	Mars orbit	electric	yes (but excessive duration for spiral down)
launcher	Proton/Atlas V	ExoMars/ Mars 2020	Mars transfer trajectory	Mars orbit	chemical	no (excessive Δv)



2 Illustration of the AIM spacecraft and deployed CubeSats at asteroid Didymos during the DART hypervelocity impact. (ESA)

Mission (AIM). AIM had three different objectives relating to technology flight demonstration in interplanetary space, investigation of NEO mitigation techniques and acquiring new scientific knowledge for understanding solar system evolution. The proposed COPINS payload consisted of two or more CubeSats carried on the main AIM spacecraft in their deployment systems (two 3U deployers), and released in the vicinity of the target asteroid Didymos, where the main AIM spacecraft acted as a communications data relay between the CubeSats and Earth ground stations (figure 2).

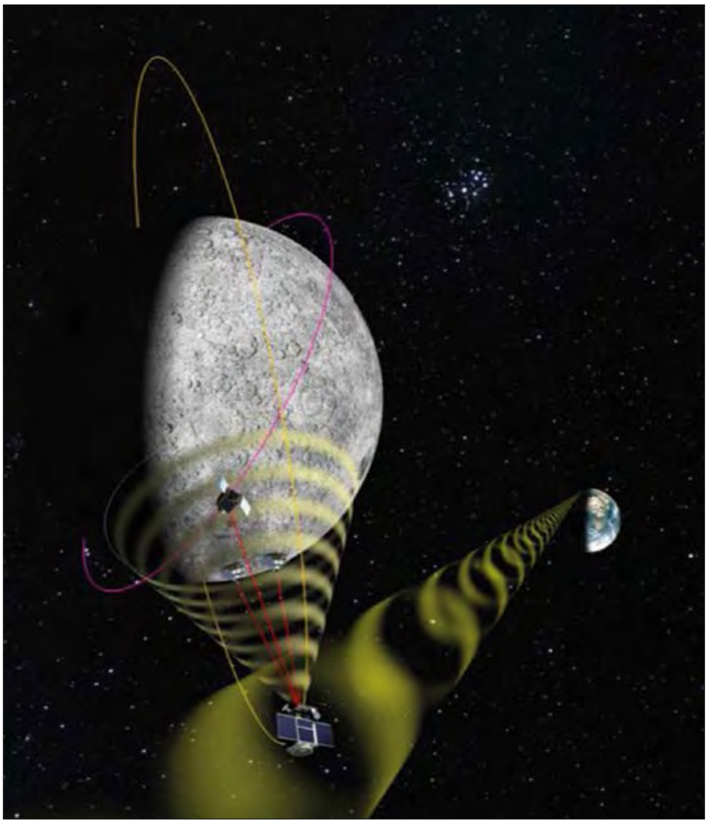
The ESA AIM mission (ESA AIM Team 2015) was proposed as part of the AIDA international cooperation, together with a US spacecraft, Double Asteroid Redirection Test (DART). DART is planned to impact the smaller component of the Didymos binary asteroid (nicknamed Didymoon) at very high velocity; AIM was envisaged as a means to rendezvous with the target asteroid in advance, in April 2022, and characterize the binary system before, during and after the impact event. The launch of the AIM spacecraft was proposed for 2020 on the Soyuz/Fregat launcher from Kourou into a direct escape trajectory, with arrival at the asteroid after about 22 months. During the rendezvous phase, the distance to the Sun and Earth is close to 1 au and from 0.5 to 0.1 au respectively.

After an initial measurement phase, the AIM spacecraft would be manoeuvred to within 10 km of the binary asteroid. Before DART's impact, the COPINS CubeSats (Walker *et al.* 2016) would be released and the main spacecraft would then retreat to a distance of 100 km, ready for the impact up to a month later. The CubeSats would perform their mission up until two months after the impact date, i.e. for three months in total, using a 1 Mbps S-band inter-satellite link (ISL) with the main spacecraft's communications system (and each other) for telecommand, housekeeping and payload telemetry transfer. Five different CubeSat concepts were selected by ESA for parallel studies via an open competitive announcement of opportunity (AO) process, as described in table 2.

The AIM mission did not receive the required funding for implementation, but the objectives and concept remain valid. Therefore a new mission proposal called HERA has since been formulated based

2 Overview of the five AIM COPINS CubeSat mission concepts

name	organization	system	mission	payload
ASPECT: Asteroid Spectral Imaging	VTT, Uni. Helsinki, Aalto Uni.	single 3U CubeSat with <1° pointing, cold gas propulsion 1 $m\,s^{-1}$ Δv	spectral imaging of Didymoon before/after impact from 4 km orbit	imaging spectrometer in VIS/NIR: 1–2 m GSD; SWIR spectrometer
DustCube	Uni. Vigo, Uni. Bologna, MICOS	single 3U CubeSat with cold gas propulsion 2 $m\,s^{-1}$ Δv, optical IR rel. navigation	characterize ejected dust plume after impact from 3–5 km orbit, transfer to L4/L5 orbit pre-impact, DRO 280 m alt. post-impact	*in situ* nephelometer, remote nephelometer
CUBATA	GMV, Uni. La Sapienza, INTA	two 3U CubeSats with cold gas propulsion Δv 1.5 $m\,s^{-1}$, <1° pointing, optical rel. navigation	gravity field determination of Didymos system before and after impact	radio science: Cube-Cube LoS Doppler tracking with S-band transponder and ultrastable oscillator
PALS: Payload of Advanced Little Satellites	Swedish Institute of Space Physics, KTH, DLR, IEEC, AAC Microtec	two 3U CubeSats with cold gas propulsion 12.5 $m\,s^{-1}$ Δv, <1° pointing, optical rel. navigation	magnetization, bulk chemical composition, presence of volatiles, super-resolution surface imaging of Didymos components impact ejecta via tour of Didymos system	narrow angle camera, volatile composition analyser, fluxgate magnetometer, video emission spectrometer
AGEX: Asteroid Geophysical Explorer	ROB, ISAE Supaero, Emxys, Antwerp Space	two 3U CubeSats: one lander, one orbiter	determination of dynamical state, geophysical surface properties, subsurface structure of Didymoon before/after impact	lander: three-axis seismometer, accelerometers, three-axis gravimeter; orbiter: 30 chipsats deployed to surface



3 Illustration of the Surrey Satellite Technology Ltd/Goonhilly Earth Station Lunar Pathfinder and deployed lunar CubeSats. (SSTL)

on a consolidation of the extensive technical work performed for AIM, focused on the asteroid deflection objectives. This has led to a change in the CubeSat payload to a single 6U CubeSat with a payload to be refined during 2018. During the COPINS studies, a number of key technologies have been identified that would require development, including short-range S-band ISLs with ranging (1 Mbps, 1 m accuracy) and time synchronization, a low-velocity (<5 $cm\,s^{-1} \pm 20\%$) CubeSat deployer with power/data interfaces, and optical relative navigation techniques such as centre of brightness determination.

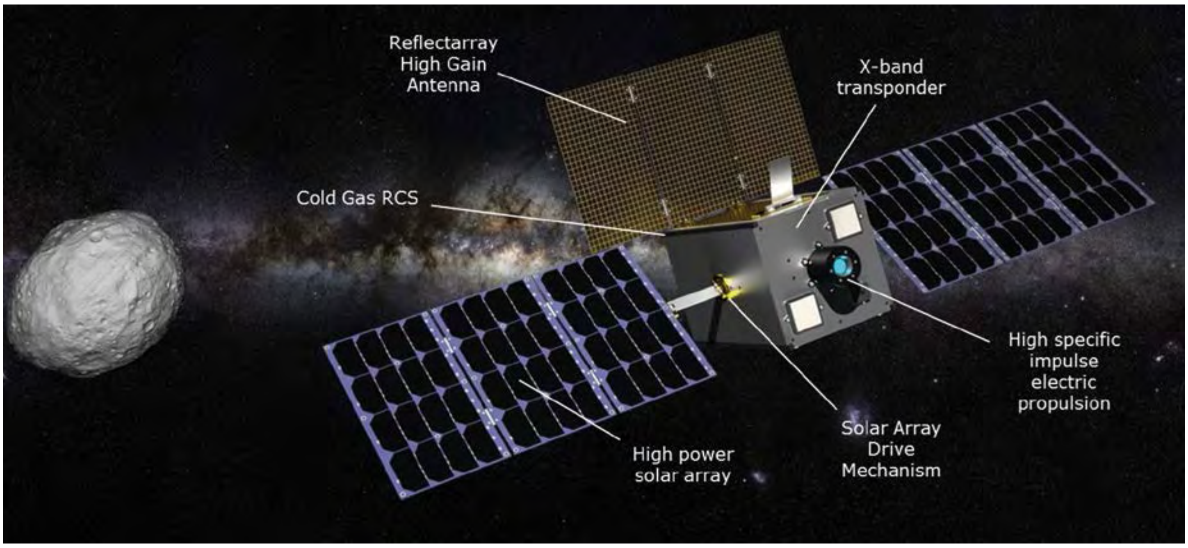
LUnar CubeSats for Exploration

Within the ESA General Studies Programme, the fourth edition of the Sysnova technical challenge focused on "LUnar CubeSats for Exploration (LUCE)". The overall rationale for the challenge is to prepare European teams for flight opportunities to the Moon that may arise, while addressing and supporting ESA's lunar exploration objectives. The activity was run through an open competitive AO process. Proposals responding to the AO were able to address one or more themes in their mission concepts and were required to comply with a common set of mission and system constraints. Single spacecraft as well as multiple distributed spacecraft were permitted, provided they complied with the constraints. It was assumed that a larger lunar orbiter spacecraft would provide transportation for up to 60 kg of CubeSat payload (a maximum of 12U per spacecraft) to a >500 km circular, >50° inclination lunar orbit, and data relay services to/from Earth ground stations via a predefined ISL operating at UHF frequencies with Proximity-1 protocol, providing maximum data rate of 512 kbps with one hour per day link availability over one year. The SSTL/GES Lunar Pathfinder (Saunders *et al.* 2016) mission (see figure 3), proposed to be undertaken in partnership with ESA, is aimed at providing these capabilities for lunar CubeSats as a commercial service, and is planned for launch in the 2023 timeframe. Hence, the associated propulsion and communications challenges were alleviated for the deployed CubeSats.

More than 30 innovative CubeSat mission concept proposals were submitted to the AO, covering a diverse range of operations concepts, platform designs and instruments, demonstrating a significant

3 Overview of LUCE mission concept studies

name	organization	system	mission	payload
LUMIO: Lunar Meteoroid Impact Observer	Politecnico Di Milano, TU Delft, EPFL, S[&]T Norway, Leonardo S.p.A, Uni. Arizona	single 12U CubeSat with chemical propulsion for L2 transfer, lunar disc optical navigation	meteoroid impact flash monitoring of the far side from an Earth–Moon L2 halo orbit	optical camera with 3.4 km/pixel spatial resolution, 3 Hz frame rate, FOV 3.5°
MoonCARE: Moon CubeSat for the Analysis of the Radiation Environment	Von Karman Institute, DLR, Tyvak International, Politecnico di Torino	three 12U CubeSats with chemical propulsion for orbit transfer	lunar radiation environment characterization, study radiation effects on specific microorganisms	radiation detector (0.06–200 MeV), astrobiology experiment (VIS/UV spectrometer, four microorganisms)
CLE: CubeSat Low frequency Explorer	ISIS bv, ASTRON, Radboud Uni. Nijmegen, Uni. Twente, TU Delft	three 12U CubeSats with inter-satellite link for ranging/time synchronization/data transfer for interferometry, propulsion for loose formation flying	technology demonstrator for distributed low-frequency radio interferometric telescopes in lunar orbit, operated in radio quiet zone around far side	<30 MHz software-defined radio receiver, three deployable monopole antennae per spacecraft
VMMO: Lunar Volatile and Mineralogy Mapping Orbiter	MPB Communications, Uni. Surrey, Lens R&D, Uni. Winnipeg	single 12U CubeSat with electric propulsion for transfer to very low lunar orbit	content of water ice deposits in permanently shadowed craters at south pole, other volatiles (e.g. ilmenite) on the day side, lunar radiation environment	active fibre laser at 1560 and 530 nm with 10 m spot size, VIS/NIR spectrometer, radiation environment sensor



4 Deep-space CubeSat configuration and key technologies. (ESA)

interest in lunar CubeSats. As a result, four mission concepts (Walker *et al.* 2017a and summarized in table 3) were awarded parallel six-month study contracts.

Based on an evaluation of the parallel study results, an ESA panel selected the LUMIO and VMMO mission concepts as joint winners of the Sysnova challenge. The winners received a prize consisting of a further study on their concept in the ESA Concurrent Design Facility (CDF) supported by an ESA engineering team.

"Four mission concepts were awarded parallel six-month study contracts"

The main refinements made were on the mission design to ensure compatibility with feasible injection from Lunar Pathfinder, the resulting propulsion subsystem selection (chemical propulsion >200 m s^{-1} for both), increase in power, and increase of the baffle length for the optical payload (both LUMIO and VMMO) for improved stray light rejection. Additionally, a near-infrared (NIR) channel was added to the LUMIO payload for improved detection of impact flashes and determination of impact energy.

The CDF studies identified several key technologies for lunar CubeSats operating in mother–daughter system architectures. These include chemical propulsion with a Δv capability of >200 m s^{-1} in order to either transfer from the injection orbit provided by, e.g. Lunar Pathfinder, to final operation orbit (LUMIO) or maintain the orbit to ensure perilune at low altitude over the south pole region (VMMO). Additionally, long-range (4000–8000 km) UHF or S-band ISLs with the mothercraft are required for communication back to Earth. For navigation, there are different options depending on the accuracy requirements including: ISL with ranging/Doppler, optical relative navigation (e.g. full lunar disc at high altitude, or feature tracking at low altitude), X-band transponder with ranging directly from/to Earth ground stations, or GNSS receiver with medium gain antenna.

M-ARGO

For deep-space missions, assuming a piggyback launch opportunity to near-Earth escape as a starting point, truly stand-alone nanospacecraft – those with no larger mothercraft – have to be independently capable of reaching their target destination using on-board propulsion with several km s^{-1} of Δv, and communicating directly back to Earth over distances of up to 1–2 au. A deep-space CubeSat system called M-ARGO (Miniaturised Asteroid Remote Geophysical Observer; Walker *et al.* 2017b) has been designed, using the requirements for a near-Earth object (NEO) rendezvous reference mission as a design driver.

A bottom-up approach was used to investigate how much propulsive Δv could be achieved within a 12U CubeSat, while still accommodating a science payload of 1–2U and downlinking the science data back to Earth at a reasonable data rate when the spacecraft is operating in close proximity to the NEO. Figure 4 shows the spacecraft design and key enabling technologies, and the design specifications are provided in table 4. Achieving very high Δv requires electric propulsion. Because the duration of the interplanetary transfer phase using an electric propulsion system is proportional to the spacecraft thrust-to-mass ratio, a great deal of effort was made to maximize the thrust within the mass constraints while achieving a high specific impulse (I_{sp}) to minimize propellant storage mass. This involved: maximizing power generation; selecting an available propulsion technology with a moderate power-to-thrust ratio for high I_{sp}; and minimizing electrical losses from sunbeam to ion beam. This trade-off and optimization process led to the engine model shown in figure 5 for two different solar array configurations (six-panel or eight-panel array with extra body-mounted panel) for the selected miniaturized gridded ion engine.

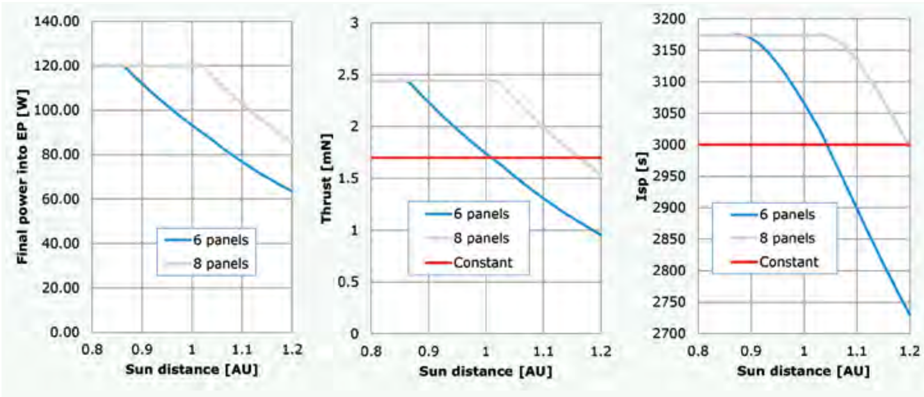
This engine model was used as a key input to the low-thrust trajectory optimization during the analysis of specific mission scenarios. For direct-to-Earth communications, the downlink performance in figure 6 was investigated for different antenna diameters of the ESTRACK deep-space ground station network (15 and 35 m), as well as the Sardinia Radio Telescope (64 m), and for different RF transmit power levels (5 and 15 W) as a function of spacecraft–Earth distance. With 15 W radio frequency power, it is possible to achieve rates of 25 kbps and 7 kbps for the 64 m and 35 m antennas respectively at 1 au from Earth. Initial thermal analysis suggests that the heat dissipation from the electric propulsion could be marginally sustained with a passive thermal control using a large area of the spacecraft body as radiator panels.

Apart from the electric propulsion and X-band communications system, other key enabling technologies identified during the study include a cold gas reaction control system for detumbling/wheel offloading, and a solar array drive mechanism to maximize power generation. With the system design presented in this section, a number of different mission scenarios for stand-alone distributed nanospacecraft systems were investigated.

4 Overview of M-ARGO deep-space CubeSat design

attribute	specification	
volume	12U form factor (226 x 226 x 340 mm)	
propulsion	mini-RIT gridded ion engine, gimbal, PPU, neutralizer, two Xenon propellant tanks and feed system (max. 2.8 kg)	
communications	X-band deep space transponder with ranging/Doppler (two receive, three transmit channels), four patch antennas for omni-directional TT&C, deployable high gain antenna reflect-array for payload data	
power	single body-mounted panel (6U face) + Li-ion battery two-wing deployable solar array with drive mechanism	
power to EP at 1 au	93 W (six panels)	120 W (eight panels)
thrust at 1 au	1.7 mN	2.4 mN
I_{sp} at 1 au	3050 s	3180 s
Δv (low thrust)	~3750 km s^{-1}	
wet mass (w/margins)	21.6 kg	22.3 kg
altitude and orbit control system	sensors: visnav camera, star tracker, six Sun sensors, IMU actuators: three reaction wheels, eight Xe cold gas RCS thrusters	
data handling	modular avionics with payload data processing	
thermal control	passive with radiator panels and heaters	

Key: RIT = radio frequency ion thruster. PPU = power processing unit. TT&C = telemetry, tracking and command. EP = electric propulsion. IMU = inertial measurement unit. RCS = reaction control system.



5 Sunbeam to ion beam optimization for the M-ARGO CubeSat using gridded ion engine.

The near-Earth object population

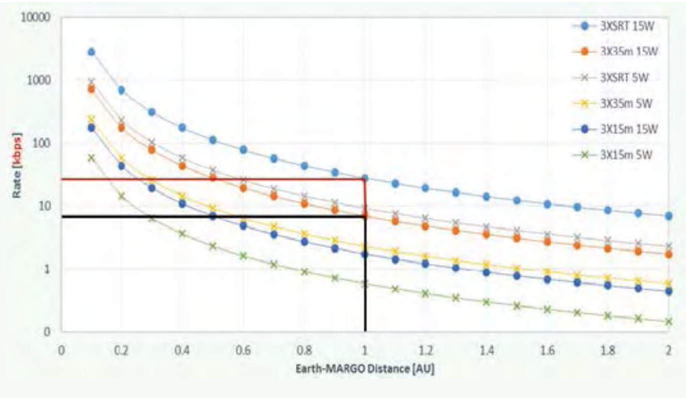
While quite a number of asteroids have been visited by spacecraft, so far only four are NEO: (433) Eros, (25143) Itokawa, (4179) Toutatis and (162173) Ryugu, all up to a kilometre in diameter. Almost no objects larger than about 100 m in diameter rotate faster than a period of a few hours. It is assumed that this is because the centrifugal force arising from the rotation is larger than the tensile strength of the object. Some models suggest that the larger objects are typically built up from small "monolithic" objects, which can have larger spin rates. It can be expected that these smaller objects do not have regolith on their surface and are more homogenous than larger asteroids. Just confirming the absence of regolith on a smaller asteroid would contribute to refining formation models of asteroids. With a NIR spectral imager one could check the mineralogical composition of the object and determine its level of homogeneity. Tracking the spacecraft with high accuracy using radio science during very close fly-bys, the mission can constrain the internal structure of the object, again showing whether it is monolithic or not.

The M-ARGO spacecraft design would give a unique opportunity to rendezvous with one of these small asteroids. A fleet of M-ARGO nanospacecraft deployed from the same piggyback launch opportunity (each targeting a different NEO), could achieve a wide survey of near-Earth asteroids very cost-effectively. This would reveal any fundamental differences between these small objects and the previously visited larger asteroids, as well as potentially the compositional diversity among different types of asteroid. Additionally, with commercial interest in space resource exploitation emerging, conducting a wide survey of the closest asteroids to Earth (in Δv terms) would be the first exploratory step to identifying resources *in situ* for later extraction. Apart from minerals, precious metals would be of high interest, and therefore the addition of a magnetometer to measure magnetic field of the asteroid would be relevant.

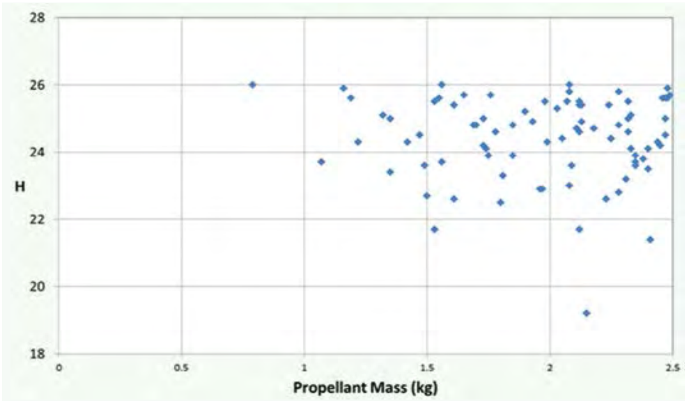
A payload assessment was carried out

5 **Summary of payload instrument options for NEO rendezvous missions**

name	details	mass (kg)	volume (U)	maturity
VIS/IR spectral imager	VIS (500–900 nm), NIR (900–1600 nm), SWIR (1600–2500 nm)	1	1	Aalto-1 (2017), PICASSO (2019), ASPECT
RSE	X-band transponder with Doppler tracking	0	0	part of comms subsystem development
laser altimeter	1.5 km range (10% albedo), 0.5–1 mm accuracy	0.033	0.04	DLEM 20 (Jenoptik) to be space qualified
magnetometer	<2 nT sensitivity, deployed on 1 m boom	0.2	0.2	MAGIC on CINEMA (2012), RadCube (2019)
low-frequency radar	20 MHz with 7.5 m dipole antenna for interior studies	1	1	DISCUS (MPI) to be developed
thermal IR imager	10–14 µm (TIR)	1.5	1.5	some adaptation
thermal IR imager	2–25 µm (IR)	1	1.5	MIMA development for ExoMars



6 Communication downlink data rates as a function of radio frequency power and Earth station antenna.



7 Results from the NEO target screening process (visual magnitude vs propellant mass).

to better understand the availability and capabilities of existing or projected miniaturized payloads that can fulfil the science objectives of NEO physical characterization. The results are presented in table 5. The spacecraft design can only accommodate up to 2 kg and 1.5U of payload, so the baseline design includes all options with exception of the low-frequency radar and thermal infrared instruments. If these can be further miniaturized, or later on the payload resources become available, then it would be possible to accommodate them.

"Several potential piggyback launch opportunities in the 2020s have been identified"

It is assumed that the M-ARGO fleet is released after the main payload on a Sun–Earth L2 transfer trajectory and, after reaching the L2 region, they are inserted into a L2 halo orbit where they wait for their optimum low-thrust transfer window to their specific target. A number of potential piggyback launch opportunities in the 2020s timeframe have been identified related to launch of medium/large-class astronomy missions to L2.

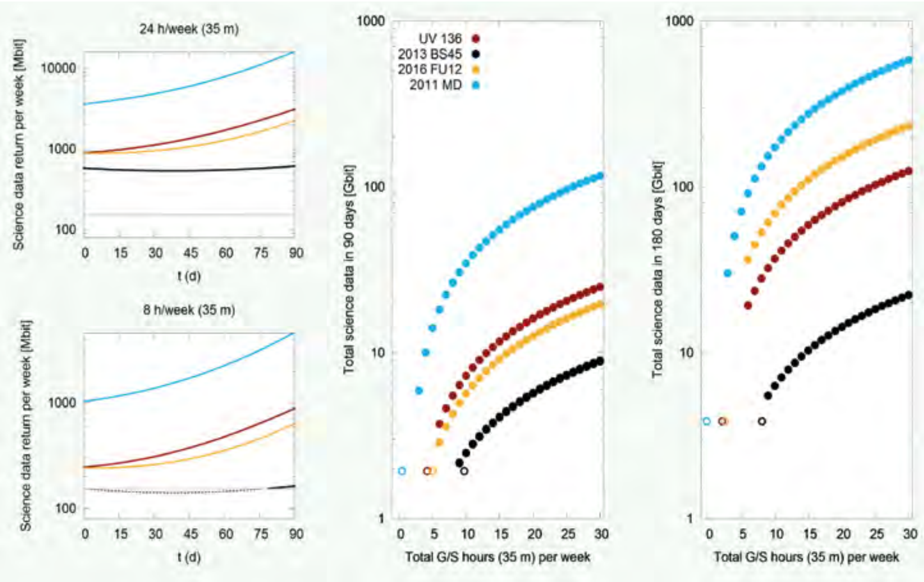
In order to identify the possible NEOs accessible by the M-ARGO fleet, a complete NEO target screening process (Mereta & Izzo 2018) was performed within the mission/system design constraints presented above, considering a launch in the 2020–23 period, starting with an L2 parking orbit and transfer duration less than three years. The complete MPCORB database of more than 700 000 objects was taken as a starting point, and a three-impulse chemical approximation used as a pre-filter. This resulted in 143 objects ($\Delta v < 3.9\,\mathrm{km\,s^{-1}}$, more than 40 observations, magnitude $H < 26$). Ephemerides of these objects were input into a global low-thrust trajectory optimization tool in order to calculate rendezvous trajectories for each target, optimizing for minimum propellant mass within the launch window and transfer duration constraints. This step resulted in 83 different targets using a propellant mass of less than 2.5 kg (the maximum capacity for xenon propellant). The distribution of these accessible NEO targets over visual magnitude (hence size) and propellant mass is given in figure 7. Most targets are within visual magnitude range 22–26 (i.e. 15–250 m diameter depending on albedo).

Low-thrust trajectories for a few targets were then further refined using a low-thrust trajectory local optimization tool with numerical integration. A six-month phase for science is assumed and, in order to minimize the science operations cost, the close proximity operations at the NEO target have been designed to follow a two-week repeat pattern with mission operations centre tasks performed in normal working hours. The two-week pattern is enabled by flying an M-ARGO CubeSat in a square trajectory relative to the NEO target on its sunward side; small manoeuvres with the electric propulsion system are executed offline at the corners. Between manoeuvres, the spacecraft flies on passively safe hyperbolic arcs. Close approach points for multispectral imager and laser altimeter science observations are centred mid-way through the coast arcs, with navigation and radio science performed in between manoeuvres and close approach points. The navigation scheme includes ground-based navigation (as for Rosetta) used as the basis for manoeuvre command generation together with on-board optical navigation (based on centroiding image processing algorithm and unscented Kalman filter) used to improve pointing and for collision risk assessment.

The distance to Earth during the science phase and the communication link performance shown in figure 6 has been used to determine the total payload data volume that can be transmitted to Earth as a function of ground station utilization time per week. This is shown in figure 8 for object 2012 UV136 (0.71 au from Earth), as well as the other studied targets (2013 BS45 at 1 au, 2016 FU12 at 0.75 au, and 2011 MD at 0.39 au from Earth).

6 Payload instrument options for space weather heliocentric missions

name	details	mass (kg)	volume (U)	maturity
radiation telescope	electron energy 0.3–8 MeV, proton energy 3.1–500 MeV, heavy ions 50–1000 MeV/n	0.8	0.8	RadTel on RadCube (2019)
magnetometer	<2 nT sensitivity, deployed on 1 m boom	0.2	0.2	MAGIC on CINEMA (2012), RadCube (2019)
solar X-ray flux monitor	energy range 1.5–25 keV, energy channels 10+, flux accuracy 1%	0.5	1	XFM-CS under development, demo in 2020



8 Science data return for close proximity operations at 2012 UV136 and other objects.

Space weather measurements

Space weather and its effects on Earth-orbiting satellites and terrestrial infrastructure is driven by particle radiation from the Sun, emanating from solar flare events. Continuous *in situ* monitoring of the energetic particle flux, interplanetary magnetic field, and other space weather parameters such as the solar X-ray flux emission at locations other than the Earth, would dramatically increase our understanding of these processes, even with only rather limited measurement capabilities. A location at Sun–Earth L5 would provide a few hours warning of major storm events. Ideally, a constellation of spacecraft in a circular heliocentric orbit inside that of the Earth would allow us to gain a complete simultaneous picture of solar activity as a function of solar longitude, potentially leading to new knowledge on solar–terrestrial physics and even more advance warning of storms.

A payload assessment was carried out in order to identify existing or projected miniaturized payloads that can fulfil the objectives of space weather monitoring. The results are presented in table 6. All of the payloads can be accommodated in the M-ARGO platform design, replacing the NEO payload suite.

"Such a spacecraft may reduce the cost of interplanetary missions by an order of magnitude"

As with the NEO rendezvous mission scenario, it is assumed that a fleet of nanospacecraft like M-ARGO, equipped with the space weather payload suite, are injected onto a Sun–Earth L2 transfer trajectory from a piggyback launch opportunity. From there, the spacecraft are manoeuvred to an L2 parking orbit. One of the spacecraft then performs a transfer from L2 to L5, while the others perform a spiral-in transfer to a circular heliocentric orbit with a semi-major axis of <1 au, one after the other in order to achieve an equal distribution of spacecraft around the orbit.

The transfer from Sun–Earth L2 to L5 has been optimized in a low-thrust trajectory local optimization tool with numerical integration. The transfer takes around 20 months and consumes 1.7 kg of Xe with 342 days of total thruster firing (Δv of 2.6 km s^{-1}). For the other spacecraft in the fleet, each with 2.5 kg of Xe propellant, the lowest semi-major axis that could be achieved for a circular heliocentric orbit (with no plane change) is 0.8 au (orbital period 0.72 years) for departure from Sun–Earth L2, and 0.75 au (orbital period 0.66 years) for departure from Sun–Earth L1. The transfer takes around 440 days and 395 days respectively. For a fleet of 10 M-ARGO-like spacecraft equally spaced around the final orbit, each spacecraft would need to depart from L2 around 94 days apart, or from L1 approximately 70 days apart, based on the synodic period with respect to Earth.

Conclusions

Several opportunities exist to embark CubeSats on larger spacecraft to lunar orbit and to a specific NEO, where they can be deployed locally for deep investigation in a mother–daughter system architecture. Numerous feasible mission concepts have been studied for this approach. A stand-alone deep-space 12U CubeSat nanospacecraft has been designed to be capable of rendezvous and characterization of NEOs or space weather measurements at the Sun–Earth L5 Lagrange point (or other locations inside Earth orbit). Based on exploiting piggyback launch opportunities to near-Earth escape, such a spacecraft is expected to reduce the entry-level cost of deep space missions by an order of magnitude, thus enabling stand-alone distributed systems in deep space with a fleet of nanospacecraft. New missions, such as a cost-effective wide survey of the NEO population for science/resource identification and simultaneous multi-point *in situ* space weather measurements, would be feasible for the first time, allowing hitchhiking nanospacecraft to write a new chapter in the guide to the solar system. ●

AUTHORS

Roger Walker, **David Binns**, **Cristina Bramanti**, **Massimo Casasco**, **Paolo Concari**, **Dario Izzo**, **Davar Feili**, **Pablo Fernandez**, **Jesus Gil Fernandez**, **Philipp Hager**, **Detlef Koschny**, **Vasco Pesquita**, **Neil Wallace**, European Space Agency, ESTEC, Noordwijk, The Netherlands. **Ian Carnelli**, ESA HQ, Paris, France. **Michael Khan**, **Mehdi Scoubeau**, **Daniela Taubert**, ESA, ESOC, Darmstadt, Germany.

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