Propulsion subsystem for a stand-alone interplanetary CubeSat

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Abstract: A number of studies performed within the frame of the ESA General Studies Programme on lunar and interplanetary CubeSat missions, have identified two main concepts. These involve either mother–daughter system architectures, where the CubeSats are carried to a target destination on a larger spacecraft, or a completely stand-alone system executing its own mission. The mother–daughter architecture alleviates the principal technical challenges of propulsion, long-range communication and deep-space environment survivability, because the host spacecraft provides these resources. However this reliance on a larger mission significantly limits the number of mission opportunities and places the CubeSat mission team at the behest of the larger programme. Stand-alone missions, i.e. where there is no mothercraft, unshackles the CubeSat objectives from these constraints if these principal technical challenges can be addressed. This paper discusses the propulsion subsystem requirements and some possible (propulsion subsystem) architectures for such a stand-alone interplanetary CubeSat. Based around a small high thrust/high specific impulse thruster.

Nomenclature

A	=	amplitude of oscillation
a	=	cylinder diameter

I.Introduction

With the MarCO (NASA's Mars Cube One) mission, a new vast area of application for Cubesats is opening, namely interplanetary missions using Cubesats. Piggy-back 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 (MarCO) was launched in May 2018 as part of the InSight mission (Klesh & Krajewski 2018). 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.

A number of studies performed within the frame of the ESA General Studies Programme on lunar and interplanetary CubeSat missions, have identified two main concepts. These involve either mother–daughter system architectures, where the CubeSats are carried to a target destination on a larger spacecraft, or a completely stand-alone system executing its own mission. The mother–daughter architecture alleviates the principal technical challenges of propulsion, long-range communication and deep-space environment survivability, because the host spacecraft provides these resources. However this reliance on a larger mission significantly limits the number of mission opportunities and places the CubeSat mission team at the behest of the larger programme. Stand-alone missions, i.e.

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where there is no mothercraft, unshackles the CubeSat objectives from these constraints if these principal technical challenges can be addressed.

The system is highly integrated in order to package it within the tight physical constraints of a 12U CubeSat platform whilst ensuring adequate thermal interfaces for heat rejection. A 'sun-beam-to-ion-beam' design philosophy has been followed in the electric power generation and conditioning chain in order to minimise efficiency losses, thereby maximising available thrust at high specific impulse. This is necessary to ensure acceptable interplanetary transfer times to e.g. rendezvous with asteroids or reach other destinations in the inner solar system.

Table 1. Mission scenario assessment 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 swingby	Chemical mono-propellant	Yes (4 x 3U CubeSat entry probes with micro-carrier)
Launcher	Soyuz/ Ariane 6/ Falcon 9/ Atlas V	Various telecom	Geo-stationary Transfer Orbit	Lunar orbit/ Near Earth Object rendezvous	Electric	No (excessive delta-V, time & rad 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/ Near Earth Object rendezvous	Electric	Yes
Launcher	Space Launch System	Orion	Lunar transfer trajectory	Lunar orbit/ Near Earth Object rendezvous	Chemical	No (excessive delta-V)
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 delta-V)

II.MISSION SCENARIO ASSESSMENT

There are presently no dedicated small launchers that can cost-effectively inject nano-spacecraft onto specific near Earth escape or Earth escape (C3>0) trajectories. For the time being at least, piggyback flight opportunities must be relied upon either on a launcher upper stage carrying a primary spacecraft or on the primary spacecraft itself as part of its mission. However, these opportunities may not be frequent, and the constraints on nano-spacecraft deployment can significantly influence delta-V and transfer window (hence the feasibility) to reach the final mission destination. Therefore, as shown in Table 1, an assessment was made on the range of missions that could be performed beyond Earth orbit for a given set of piggyback flight opportunities.

III.STAND-ALONE DISTRIBUTED SYSTEM ARCHITECTURES

A. Deep Space Nano-spacecraft Design

Assuming a piggyback launch opportunity to near Earth escape as a starting point, truly stand-alone nano-spacecraft (i.e. those which have no larger mothercraft) have to be independently capable of reaching their final target destination using on-board propulsion with several km/s of delta-V, and communicating directly back to Earth ground stations over distances of up to 1-2 AU. A deep space CubeSat system called M-ARGO (Miniaturised Asteroid Remote Geophysical Observer) [7] has been designed, using the requirements for a NEO rendezvous reference mission as a design driver.

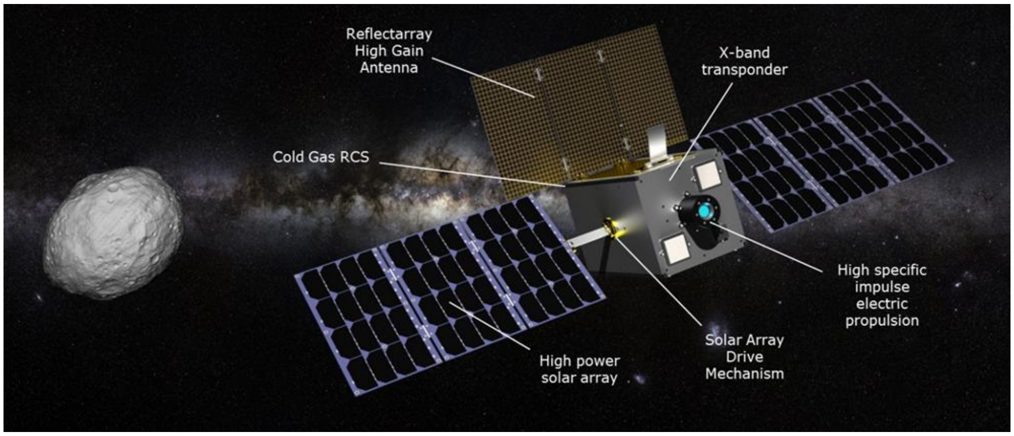

Figure 1. Deep space CubeSat configuration and key technologies

A bottom-up approach was utilised to investigate how much propulsive delta-V could be achieved within a 12U CubeSat, whilst 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 delta-V requires the use of electric propulsion, and since 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 maximise the thrust within the mass constraints whilst achieving a high specific impulse (Isp) to minimise propellant storage mass. This involved a) maximising power generation; b) selecting an available propulsion technology with a moderate power-to-thrust ratio for high Isp; and c) minimising electrical losses from sun beam to ion beam. This trade-off and optimisation process led to the engine model in Figure 5 for two different solar array configurations (6 panel or 8 panel array with extra body mounted panel) and the selection of a miniaturised gridded ion engine.

Table 4. Overview of the M-ARGO Deep Space CubeSat design

Propulsion	Gridded Ion Engine, gimbal, PPU, neutraliser 2 Xenon propellant tanks & feed system (Max. 2.8 kg)	
Communications	X-band DS transponder with ranging/doppler (2 Rx, 3 Tx channels) 4x patch antennas for omni-directional TT&C Deployable HGA reflect-array for P/L data	
Power	Single body mounted panel (6U face) + Li-ion battery 2-wing deployable solar array with SADM	
Power to EP @ 1 AU	93 W (6 panels)	120 W (8 panels)
Thrust @ 1 AU	1.7 mN	2.4 mN

Isp @ 1 AU	3050 s	3180 s
Delta-V (low thrust)	~3750 km/s	
S/C wet mass (w/ margins)	21.6 kg	22.3 kg
AOCS	Sensors: visnav camera, star tracker, 6 sun sensors, IMU Actuators: 3 reaction wheels, 8 Xe cold gas RCS thrusters	
Data handling	Modular avionics with payload data processing	
Thermal control	Passive with radiator panels & heaters	

This engine model was used as a key input to the low-thrust trajectory optimisation during the analysis of specific mission scenarios. For direct-to-Earth communications, the downlink performance in **Error! Reference source not found.** 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 15W) as a function of spacecraft-Earth distance. With 15W RF 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.

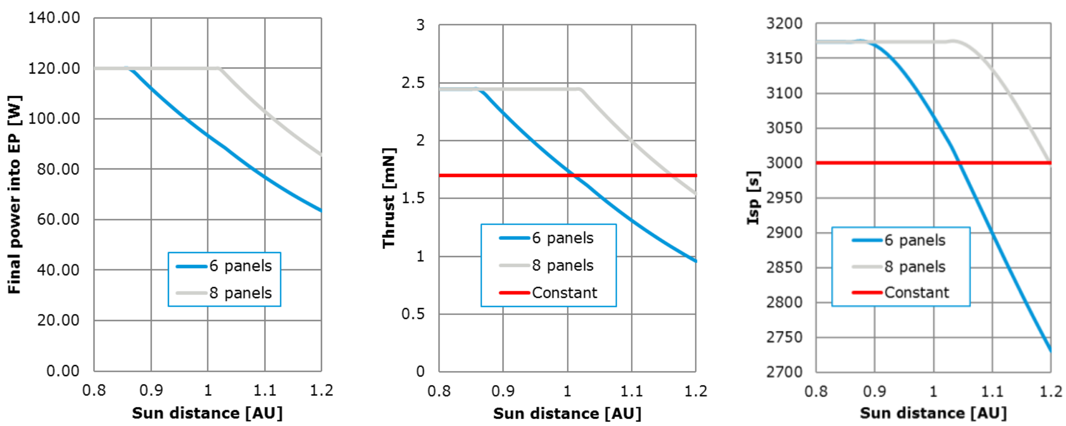

Figure 2. Sun beam to Ion Beam optimisation for the M-ARGO CubeSat using gridded ion engine

From an initial thermal analysis, it was found that the heat dissipation from the electric propulsion PPU @120W input or the X-band transponder 15W power amplifiers could be marginally sustained with a passive thermal control using a large area of the spacecraft body as radiator panels.

B. Wide Survey of the Near Earth Object Population

Science Case

While quite a number of asteroids have been visited by spacecraft, so far only three are NEO: (433) Eros, (25143) Itokawa, and (4179) Toutatis. All of these are several hundreds of meters in diameter.

The M-ARGO spacecraft design would enable the unique opportunity to rendezvous with one of the small asteroids. With a fleet of M-ARGO nano-spacecraft deployed from the same piggyback launch opportunity (each targeting a different NEO), a wide survey of near-Earth asteroids could be achieved very cost-effectively. This would reveal the fundamental differences between these small objects and the previously visited larger asteroids, as well as potentially the diversity between different types of asteroid in terms of composition. Additionally, with commercial interest in space resource exploitation emerging, conducting a wide survey of the closest asteroids to Earth (in delta-V 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.

Table 1. 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 w/ Doppler tracking	0	0	Part of comms subsystem devt
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
TIR Imager	TIR (10-14 um)	1.5	1.5	Some adaptation
TIR Spectrometer	2-25 microns (IR)	1	1.5	MIMA devt for ExoMars

Payload Suite

A payload assessment was carried out to better understand the availability and capabilities of currently existing or projected miniaturised payloads that can fulfil the science objectives of NEO physical characterisation. The results are presented in Table 1. Since the spacecraft design can only accommodate up to 2 kg and 1.5U of payload, the baseline design includes all options with exception of the low frequency radar and thermal infrared instruments. If they can be further miniaturised, or later on the payload resources become available, then it would be possible to accommodate them.

Mission Scenario

It is assumed that the M-ARGO nano-spacecraft fleet is released after the main payload in 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.

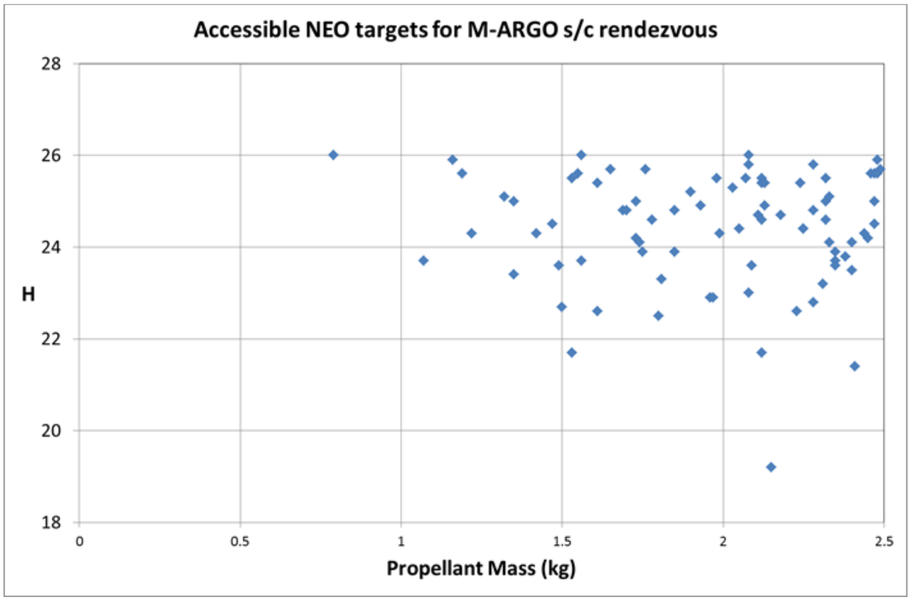

Figure 3. Results from the NEO target screening process (visual magnitude vs. propellant mass)

NEO Target Screening

In order to identify the possible NEOs accessible by the M-ARGO fleet, a complete NEO target screening process [8] was performed within the mission/system design constraints presented above, considering a launch in the 2020-2023 period, start at L2 parking orbit and transfer duration <3 years. Firstly, the complete MPCORB database of >700,000 objects was taken, and a 3-impulse chemical approximation used as a pre-filter. This resulted in 143 objects (delta-V <3.9 km/s, >40 observations, magnitude H<26). Ephemerides of these objects were input into a global low-thrust trajectory optimisation tool in order to calculate rendezvous trajectories for each target optimising for minimum propellant mass within the launch window and transfer duration constraints. This step resulted in 83 different targets with a propellant mass of <2.5 kg (the maximum capacity of the Xe tanks accounting for RCS propellant). The distribution of these accessible NEO targets over visual magnitude (hence size) and propellant mass is given in Figure 3. It can be seen that most targets are within the 22-26 visual magnitude range (i.e. 15-250 m diameter depending on the albedo).

Mission analysis for selected targets

Low-thrust trajectories for a few targets were then further optimised in a low-thrust trajectory local optimisation tool with numerical integration. An example trajectory for the object 2012 UV136 can be seen in Figure 4, along with distance to Earth/Sun, Sun-S/C-Earth angle and solar aspect angle. A 6-month phase for science is assumed, and in order to minimise 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 the M-ARGO nano-spacecraft in a square trajectory relative to the NEO target on its sunward side, with small manoeuvres with the electric propulsion system executed offline at the corners. Between manoeuvres, the spacecraft flies on passively safe hyperbolic arcs. Close approach points for multi-spectral imager and laser altimeter science observations are mid-way through the coast arcs with navigation and radio science performed in between manoeuvres and close approach points, as can be seen in Figure for object 2012 UV136.

The navigation approach includes ground-based navigation (á la 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.

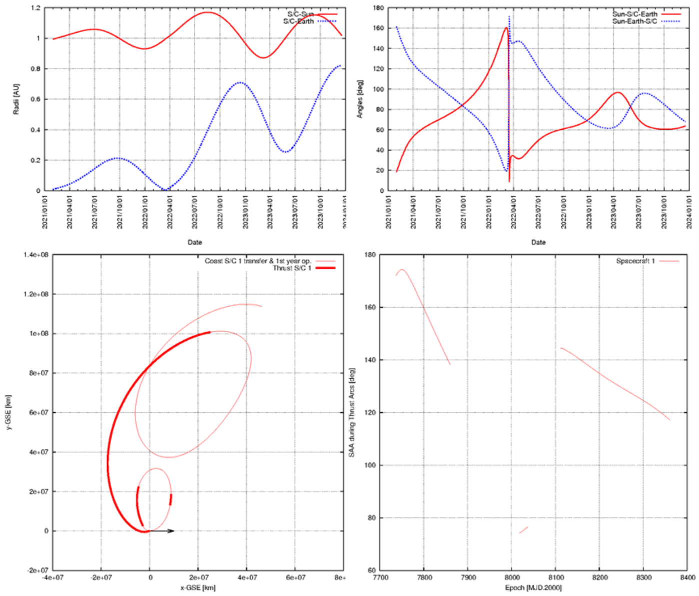

Figure 4. Low-thrust trajectory optimisation for rendezvous with NEO 2012 UV136

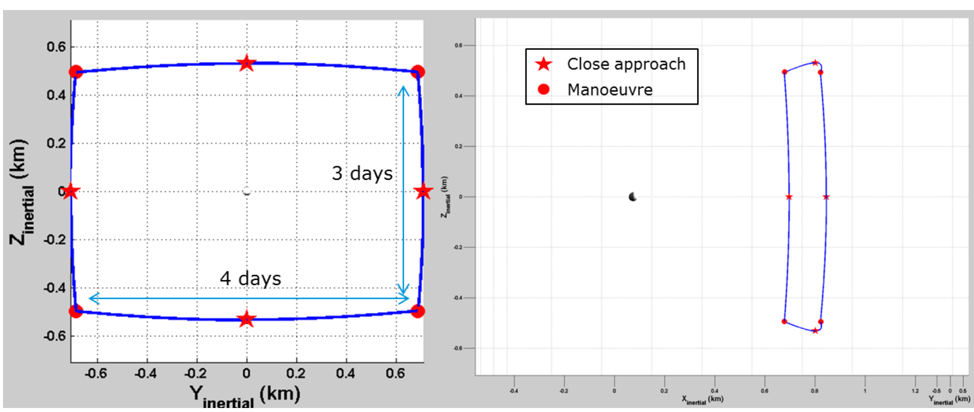

Figure 5. Two week repeat cycle for close proximity operations near NEO 2012 UV136

C. Candidate Electric Propulsion systems

Taking into account the requirements electric propulsion system, especially high Isp, very high life time by a high thrust of over 2.4mN limit the choices to few types of thrusters, mostly Gridded Ion Engines (GIE). In Europe three thrusters show to satisfy the high demands of M-ARGO mission.

Electric Propulsion System Architecture

To be able to function, a thruster needs some utilities. Therefore an electric propulsion subsystem is normally includes:

- Thruster together with Discharge Unit
- Neutraliser

- Power Processing Unit
- Propellant Management System
- Harness and Piping

Figure … shows the schematic of an electric propulsion system for gridded ion thrusters.

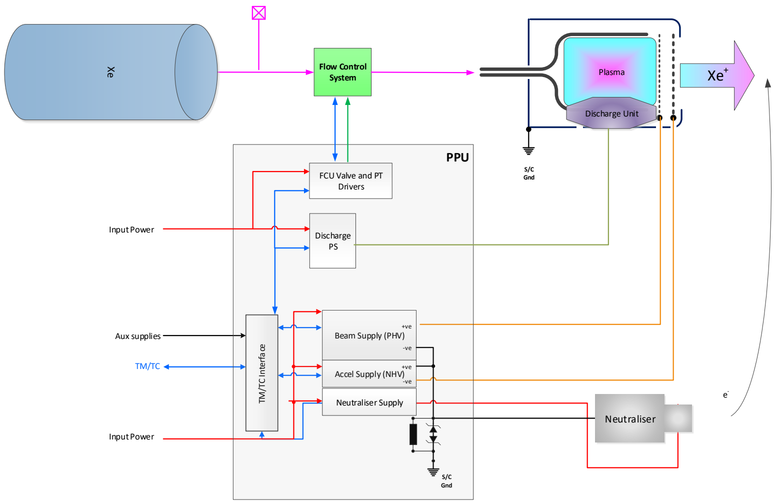

Figure 6 : the schematic of the Electric Propulsion System Architecture

In the following short descriptions are given for each subsystem :

Thruster including the Discharge Utility Unit

The thrusters should be able to deliver the required thrust and total impulse for the mission. Near the discharge chamber, gas inlet and the grid system, thrusters need a discharge unit to provide the plasma. Depending on the type of the thruster a cathode/anode or a RFG/Coil combination can be used. In Kaufman type thrusters a discharge between the cathode and the Anode provides the discharge. In RF thrusters, an RFG converts DC current into the required AC current for the coil outside the thruster. The RFG includes the matching ability to ensure the maximum power transfer to the plasma in different thruster performance points.

Neutraliser

Gridded ion thrusters extract and accelerate solely positive ions. Therefore the thruster's ion current has to be compensated with an equivalent amount of electron current. Hollow cathode type neutralizers are normally used for this purpose.

Power Processing Unit

The thruster needs one positive and one negative high voltage for the grid system and a discharge power supply for the thrusters discharge unit. The PPU includes also the necessary power supplies for the neutraliser. The PPU has to provide all voltages required by the electric propulsion sub-system. The PPU interfaces with the power bus and the spacecraft's data bus. It receives high level commands and translates them into operation sequences. Also autonomous handling is implemented.

Propellant Storage and Management System

The system includes propellant storage system (tanks) and the propellant flow management system. The propellant flow management consists of two steps: Pressure regulation and flow control. The pressure regulator reduces the high pressure inside the xenon tanks down to typically few bars, which is necessary for the function of the Low Pressure Flow Control System. The constant pressure is fed to the flow control units ("FCU") for the individual thrusters and neutralisers. The FCU devices regulate the mass flow to each thruster and neutraliser.