Electric Propulsion Module for the extension of VEGA Launch Vehicle Payload Capability

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Abstract: VEGA successful maiden flight in February 2012 and the following successful flights opened a new efficient, reliable and cost effective access to space for the European and worldwide space industry. In a constant effort devoted to performance enhancement, costs reduction and adaptation to evolving market needs, a system based on electric propulsion has been conceived to extend the VEGA launch system mission from its current LEO capability to MEO, GEO, HEO and interplanetary missions, today prerogative of middle and heavy launchers. A fully autonomous EP module, thought as an add-on to the VEGA launcher, has been preliminarily studied in terms of mission analysis, functional architecture and configuration, defining the main system level requirements and providing some insight on the most critical subsystems. Conclusion of the study confirms this application as an attractive business case, where important enhancements are brought to the launch service in terms of improved performance and flexibility, keys to satisfy the wide satellite market that requires efficient solutions.

Nomenclature

ACS	=	Attitude Control Sub-System
BOL	=	Begin Of Life
CFRP	=	Carbon Fiber Reinforced Plastic
COPV	=	Composite Overwrapped Pressure Vessel
CPS	=	Chemical Propulsion Sub-System
CSG	=	Centre Spatiale Guyanes
ΔV	=	Delta Velocity
EOL	=	End Of Life

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EP	=	Electric Propulsion
EPM	=	Electric Propulsion Module
GEO	=	Geosynchronous Earth Orbit
GNC	=	Guidance, Navigation & Control
GNSS	=	Global Navigation Satellite System
HEO	=	Highly Elliptic Earth Orbit
HET	=	Hall Effect Thruster
H/W	=	Hardware
I/F	=	Interface
I_{sp}	=	Specific Impulse
LEO	=	Low Earth Orbit
MEO	=	Medium Earth Orbit
MIR	=	Module Interface Ring
OCS	=	Orbit Control Sub-System
OR	=	Orbit Rising
OTS	=	Off-The-Shelf
PCDU	=	Power Control & Distribution Unit
P/L	=	Payload
PMA	=	Pressure Management Assembly
PPU	=	Power Processing Unit
RCS	=	Reaction Control Sub-System
RJT	=	Resistojet Thruster
SA	=	Solar Array
S/C	=	Spacecraft
SEP	=	Solar Electric Propulsion
S/S	=	Sub-System
SWU	=	SWitching Unit
T	=	Thrust
TRL	=	Technology Readiness Level
Xe	=	Xenon
XFC	=	Xenon Flow Controller

I. Introduction

VEGA successful maiden flight in February 2012 and the following successful flights opened new opportunities for the European space industry. Europe has now at its disposal a flexible small launcher able to inject payloads from 300 to 2500 kg into a wide range of LEO orbits, from equatorial to sun-synchronous. A key-strategy to successfully compete in the space transportation field is represented by a constant effort in performance enhancement, costs reduction and adaptation to evolving market needs: in this frame, Electric Propulsion can play a fundamental role.

The idea of using an additional transfer module for high delta-V missions, installed inside the payload fairing, is already a reality in Vega, but with chemical propulsion. The 500-kg class LISA Pathfinder interplanetary spacecraft will be launched this year on a Vega launcher from Europe's Spaceport, reaching its final orbit in L1 via 1.5-ton chemical propulsion module.

Along this principle, CGS is today looking to electric propulsion as one of the most promising technologies, enabling savings of mass and cost through better performance such that, in cooperation with the European Space Agency and Sapienza Università di Roma, its application to the launch services segment has been conceived.

A fully autonomous module based on electric propulsion has been thought as an add-on of VEGA launcher, with the purpose of extending the launcher mission from its current capability up to a new and wider set of final orbits.

In the present paper, the output of the EP module pre-feasibility study performed under an ESA contract is outlined: the business case and the associated mission requirements are firstly defined, then the preliminary design of the EP module is described, in terms of mission analyses results, S/C functional architecture, configuration and layout. In the frame of the above-mentioned study, CGS is the mission/system architect and responsible for the S/C design, whereas Sapienza Università di Roma developed the low-thrust trajectory optimization tool and performed the mission analyses.

II. Business case

A. Opportunity in the European launchers' market

The European market of small satellites (launch mass up to about 1 metric ton) injection beyond LEO is currently served by the large and mid-sized launchers Ariane 5 and Soyuz from CSG, which is not an optimal solution since:

1) a single-payload launch would be very expensive and over-dimensioned in terms of lifting capability;
2) a multi-payload launch would be more cost effective but strictly constrained on the mission profile and launch window, which should be similar to the main payload ones.

The small satellite class is common to a wide range of missions and applications that require orbits other than LEO, e.g. navigation services in MEO, Data/Telecom and Data Relay services in MEO and GEO, scientific missions in unconventional orbits such as HEO and Sun-Earth or Earth-Moon Lagrange points.

Due to the scenario depicted above, the opportunity was identified of enhancing the VEGA mission capability to provide to such small spacecrafts with the access to orbits beyond LEO at an affordable price, by means of a flexible add-on recurrent service to the standard VEGA launch, which must require no modification to the current VEGA I/F and take profit from VEGA-C, the first evolution step of VEGA which will improve the launcher performance by means of a more powerful first stage solid rocket engine.

In order to be cost effective and competitive on the European launchers' market, a raw estimation of the EPM recurrent cost sets it around the 10% of the VEGA standard launch cost, so to match an overall launch service cost comparable to or lower than the direct competitor one, i.e. the multi-payload Soyuz launch, without the drawbacks of the typical multi-payload missions. A design-to-cost approach is deemed necessary to match the EPM target cost figure, relying on OTS components, a simple and modular design, a fast and effective design, development and qualification plan and a qualification flight on board of VEGA-C maiden flight.

B. Mission scenarios

Several missions have been already identified that suits well the VEGA+EPM service, taking benefit of its flexibility:

1) Galileo constellation replenishment: MEO, h= 23222 km, i= 56°;
2) O3B commercial constellation replenishment: MEO, h= 8062 km, i= 0°;
3) PROBA-3 scientific mission deployment: HEO, h= 600 x 60530 km, i= 59°;

but the designed mission flexibility allows a wider range of target orbits, including:

4) Data/telecom small satellites deployment to GEO
5) Scientific missions deployment to Sun-Earth and Earth-Moon Lagrange points
6) Scientific rendez-vous missions with a Near Earth Asteroid (NEA)
7) Active debris removal/towing

As far as the Galileo constellation maintenance/replenishment is concerned, it would strongly benefit from the single launch configuration, being the EPM able to target each satellite to its own final orbit, with a corresponding propellant mass saving or lifetime extension (currently, multi-payload launches require the satellites to autonomously maneuver to reach their own final orbit, from the satellite release one). On the other hand, PROBA-3 represents the typical mission that would realize great saving on the launch cost, when lifted-off by VEGA+EPM: in fact, Soyuz from Kourou (single payload) is currently selected as the baseline launch service provider since, even if the S/C is actually compatible with the VEGA fairing, the peculiar HEO orbit would be out of its performance range, Soyuz representing the only feasible option, although non optimal in terms of over-dimensioned launch mass capability and huge cost (no other S/C has been identified so far sharing the same peculiar elliptic orbit to allow a dual-payload launch opportunity). The drawback of using the electric propulsion mainly deals with the low thrust delivered, which makes the orbit transfer phase duration last several months and the passenger S/C total radiation dose increase by a relevant quantity: while the latter issue can be technically assessed and managed, the first one becomes acceptable to the Customer as far as the delay in the beginning of the operational phase does not represent a cost or such a cost is lower than the saving achieved on the launch cost.

III. Mission and system concepts

A. Generalities

As it will be shown later on, for the above depicted mission scenarios the required ΔV always exceeds 4.0 km/s, meaning that, given the constrains on the VEGA-C payload mass at lift-off, a very high specific impulse figure is required in order to reduce the propellant mass fraction and thus guarantee the mission feasibility: this poses electric propulsion as an enabling technology for this class of orbit raising missions, which would not be feasible with traditional chemical propulsion (see Fig. 1). Figure 2 shows the benefit that an actually designed mission such as LISA Pathfinder would have experienced whether SEP was implemented onto its propulsion module instead of the traditional CPS (different P/L masses are delivered into orbit by VEGA, depending on the P/L release orbit).

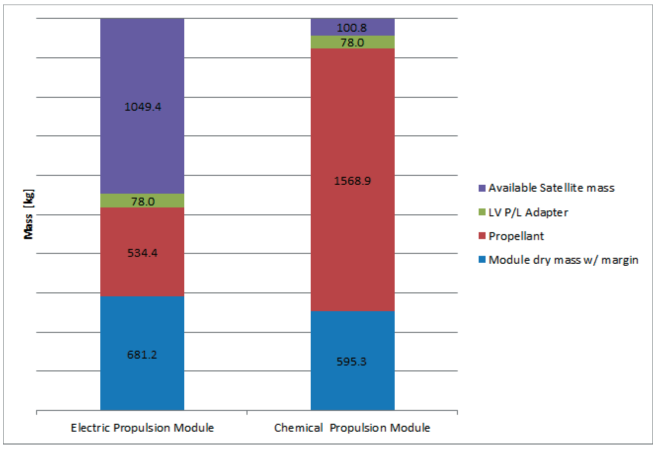

Figure 1. SEP vs. CPS performance comparison: generic 4.0 km/s ΔV orbit raising mission, launched by VEGA-C.

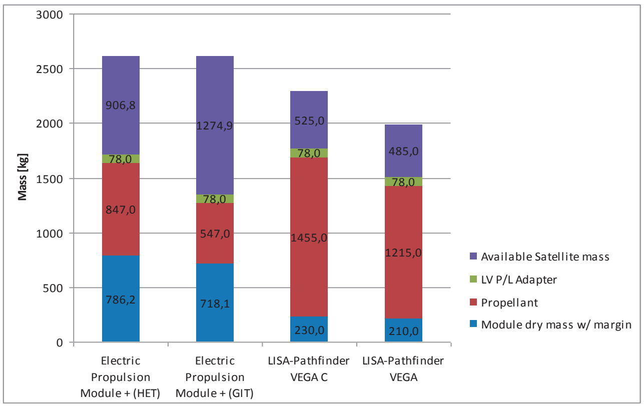

Figure 2. SEP vs. CPS performance comparison: LISA Pathfinder mission (S/E L1 Lagrange Point).

Since an effort is devoted to keep the orbit raising duration within the relevant mission requirement, Hall effect thrusters are preferred over ion engines due to their higher thrust-to-power ratio.

The EPM is designed as an autonomous module which is connected to the launch vehicle adapter and carries atop the passenger S/C. Once released into the initial parking orbit by the launcher, the EPM performs a low-thrust orbit raising up to the passenger requested final orbit, then releases it and proceeds with the re-entry or re-boost to graveyard orbit, before final passivation.

The mission analysis, carried out for each selected mission scenario, identifies the initial parking orbit which maximizes the available passenger S/C mass while respecting the constrains on the launcher performance map and the maximum transfer duration, defines the optimal thrust law and provides the information required to perform the power generation S/S dimensioning.

The Galileo orbit injection mission is identified as the reference mission scenario to drive the EPM main dimensioning, as per ESA agreement: a 9 months maximum transfer time and a 910 kg minimum wet mass of the passenger satellite (+30% on mass-at-launch over the current Galileo FOC satellite) are agreed as high level mission requirements, as well.

B. Mission analysis tools, hypotheses and assumptions

The mission analysis tool, developed as a fast and efficient tool able to perform complex parametric analyses with a minimum computational effort, is based on the optimal control theory optimization strategy, numerically implementing the methods proposed by Kluever[1,2]. Since the propulsive acceleration is much smaller with respect to the gravitational one, it can be considered as a perturbation, so that the Gauss' variational equations written as function of the classical orbital elements are numerically integrated. Perturbative effects of the space environment on the orbital propagation are taken into account, including Earth's shadow[3] (which is used as switch-off flag for propulsion), Earth's oblateness[4] and atmospheric drag. A differential evolution algorithm is employed in order to compute the optimization variables, i.e. the parameters of the guidance law and the transfer time, which is unknown. The objective function to be minimized can be the transfer time as well as the propellant mass. Equality constraints are considered since the spacecraft has to match the final conditions imposed by the target orbit.

The optimization tool has been validated using test cases provided by literature[1,2].

On-board power degradation due to solar cells exposure to energetic particles during the Van Allen belts passages is modelled within the tool, as well. According to Ref. 5, the 1 MeV Equivalent Electron Fluence is first evaluated in order to further compute a damage function which represent the fraction of the available on-board power at any time. Such model is fine-tuned using data from the actual SMART-1 mission[6] and from state-of-the-art triple junction solar cells' datasheets (see Fig. 3).

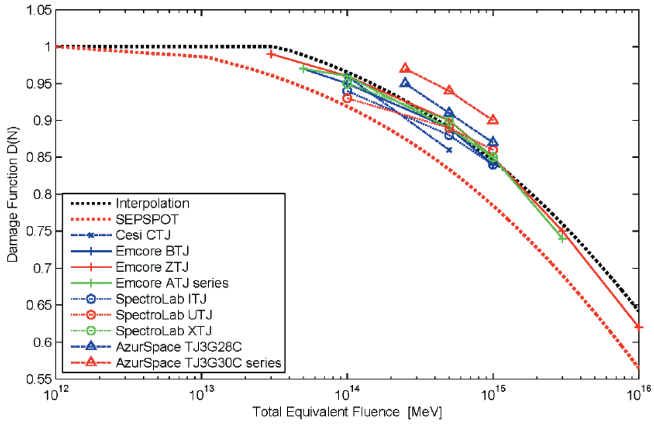

Figure 3. Solar cells damage function vs. total equivalent fluence, actual and interpolated data.

Taking into account the power degradation by a traditional EOL dimensioning approach (to guarantee sufficient power to generate the nominal thrust at EOL) would likely bring to an over-dimensioning of the solar array installed power, leading to several technical and programmatic criticalities. In order to mitigate such a risk, a different mission and power management scenario is formulated and proposed, which, at the cost of some technological improvement, could guarantee the mission technical feasibility: it could be referred to as a sort of *blow-down propulsive power regime*, as detailed here below.

The on-board power breakdown can be defined as the main propulsive component (electrical power converted into thrust) plus the other bus sub-systems ancillary feeding (electrical power used by the S/C while in no-thrust stand-by mode and battery charging power during sunlight phases). The PPU efficiency is accounted in the main propulsive power budget. The assumption on which the *blow-down propulsive power regime* is based on foresees that, since the power degrades during the mission and the ancillary power budget is nearly constant throughout the mission duration (conservatively set to 850 W), the power degradation affects the propulsive subsystem only, which means that the performance of the thrusters are degrading during the mission. As a preliminary assumption, the PPU is assumed to be able to control the thruster parameters in order to have a decreasing thrust at constant I_{sp} while the power input decreases.

Such logic is implemented in the mission analysis tool, which computes the BOL value of the on-board power which allows to meet the constraint on the max transfer duration, thus minimizing the required installed power and solar panels surface w.r.t. the EOL dimensioning approach. An example of this mission philosophy is showed in Fig. 4, in which the instantaneous power is reported as a function of the transfer time.

Since a 0.5 N minimum thrust is necessary to match the transfer duration constraint, a cluster of 5.0 kW class HET is baselined for the EPM configuration, trying to limit the thruster number to two, as far as practicable, for cost reasons. Among the OTS 5 kW HET currently available in the worldwide market, the Aerojet-Rocketdyne BPT-4000/XR-5 has been used for defining the reference performance since it is the only one with flight heritage and publicly available data. Nevertheless, alternative engines such as the Snecma PPS-5000, the Alta/Sitael HT-5k and the Fakel SPT-140 are analyzed to confirm the robustness of mission design to the use of different thrusters. Being all the above mentioned items able to work at multiple set points, both high-thrust (low discharge voltage) and high-I_{sp} modes (high discharge voltage) are evaluated.

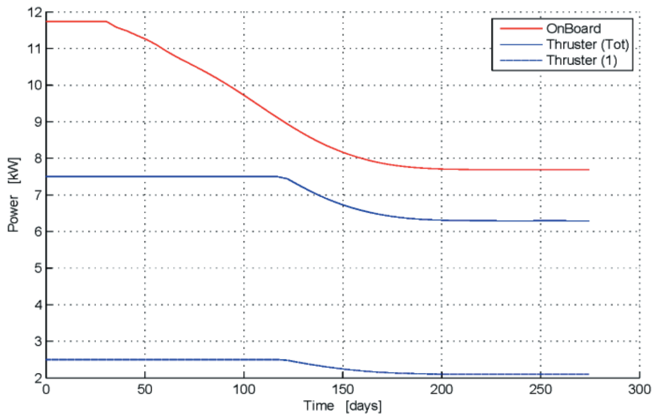

Figure 4. Blow-down propulsive power logic, typical behavior.

C. Mission design and high level requirements

1. Initial parking orbit

Table 1 shows the results of the parametric analysis carried out to estimate the optimal initial parking orbit, on the Galileo reference mission (results are of general validity). Two findings arise:

1) the parking orbit shall be at the same inclination of the target orbit, since it is too onerous for the EPM performing the inclination plane change manoeuver within the 9 months timeframe, in terms of both thrust value (3 thrusters required instead of 2), BOL power and propellant mass;
2) the parking orbit shall be at the lowest practical altitude to maximize the overall lifting capacity (EPM dry mass + passenger satellite wet mass).

For this reason, a 300x300 km circular orbit at the inclination of the final target orbit is selected as initial parking orbit for any mission scenario.

Parking orbit inclination, deg	Parking orbit altitude, km	VEGA-C performance, kg (*)	Propellant mass, kg	BOL power, kW	Number of thrusters	Usable mass, kg (†)
5.2	300 x 300	2533	829.3	22.3	3	1703.7
5.2	700 x 700	2257	770.0	20.1	3	1487.0
5.2	200 X 1200	2257	749.2	19.1	3	1507.8
56.0	**300 x 300**	**2265**	**435.8**	**10.7**	**2**	**1829.2**
56.0	700 x 700	2008	365.5	8.9	2	1642.5
56.0	200 X 1200	2008	366.2	9.0	2	1641.8

(*) VEGA performance map + 300 kg VEGA-C improvement -78 kg P/L adapter mass
(†) Available for EPM dry mass + passenger satellite mass at launch

Table 1. Effect of the initial orbit on Galileo mission (BPT-4000 Hi-T mode, on-board power computed for a constrained 9 months OR duration).

2. Power and SEP dimensioning

Three main parameters are involved in the system preliminary dimensioning process: installed thrust, propellant capacity and installed solar power supply, given the required 9 months transfer duration.

Their evaluation has been carried out in two steps: first basing on the sole Galileo reference mission, then taking into account the full range of application identified.

Concerning the minimum installed thrust at BOL, a rough order of magnitude of 500 mN has been identified by means of several preliminary analysis runs, which can be obtained either by a cluster of three 5 kW class HET operating in hot redundancy at 3 kW, or by a cluster of two operating at their maximum input power. It is straightforward that redundancy level and H/W cost are higher for the first option, which is thus discarded due to the design-to-cost requirement, at least during the first evaluation step.

Mission analysis of the Galileo reference case, with two BPT-4000 operating at 4.5 kW nominal power, either at 300 V Hi-T or 400 V Hi-I_{sp} modes gives an insight on the power and propellant budgets, as summarized in Table 2.

When one tries to accomplish the 9 months orbit raising with the Hi-T mode, a higher Xenon mass and a lower power at BOL are required, whereas the opposite occurs for the Hi-I_{sp} mode. The BOL power trend is justified by the fact that, when the nominal thrust at BOL decreases, in order to match the same transfer duration target a longer time must be spent at full propulsive power, before the solar cells degradation begins affecting the propulsive power and thus thrust level.

In order to guarantee the highest mission flexibility for the EPM, the dimensioning is accomplished enveloping both thruster's functioning modes, thus EPM design values are the ones highlighted in green in Table 2.

Configuration	VEGA-C performance, kg	Propellant mass, kg	BOL power, kW	EOL power degradation	Solar array surface, m2	Transfer duration in contingency mode, months
High Thrust	2265	**484.7**	10.9	35.2%	30	12.2
High I_{sp}	2265	434.7	**12.5**	35.1%	35	14.2

Table 2. EPM dimensioning cases (constrained 9 months transfer duration).

During the second evaluation step, it is recognized that the Xenon mass coming from the first design step is not able to cover the needs of GEO and interplanetary missions, for which a capacity extension up to about 1 metric ton of Xenon is required, whereas it is traded-off that the 12.5 kW installed power, already related to a very high solar panels area, can be maintained as it is, given that, on one hand, a small penalty on the transfer duration to GEO is acceptable and that, on the other hand, transfer time is not a driver for interplanetary scientific missions. On the thruster cluster side, simple considerations about the typical Xenon throughput (e.g., BPT-4000 is qualified for up to 285 kg of Xe[7]) suggest that two 5 kW thrusters would not be able to fulfill such extended missions, so bigger clusters have to be taken into account. In order to keep under control cost and mass of the SEP S/S (in particular for those missions that would not require additional thrusters) a configuration of two PPU's to four HET's by means of one switching unit (SWU) is conceived, which is able to fire two thrusters per time, switching at any time from one couple to the other. For MEO missions, the two additional HET's and the SWU could not be installed, saving mass and money with no major modification to the SEP configuration.

3. VEGA+EPM performance summary

Once the main high level requirements of the EPM are identified, performance evaluation of the VEGA+EPM launch system on each mission scenario has been performed. This time, the mission analysis tool is set to compute the transfer duration given the fixed BOL power level and, for sure, the desired firing mode and initial parking orbit.

Table 3 summarizes the mission analyses results for the reference thruster, basing on a EPM dry mass estimation that includes both standard maturity margins for each component plus an overall 20% system margin.

Orbit	Orbital elements	Application	Parking orbit	EPM dry mass, kg	Firing mode	Transfer time, months	Xenon mass, kg	P/L mass, kg
MEO	h= 8062 km i= 0 deg ; e= 0	O3b	h= 300 km i= 5.2 deg ; e= 0	740.6	Hi-T	6.6	415	1378
				740.6	Hi-Isp	7.5	363	1430
MEO	h= 23222 km i= 56 deg ; e= 0	Galileo	h= 300 km i= 56 deg ; e= 0	740.6	Hi-T	7.3	531	993
				740.6	Hi-Isp	8.7	477	1047
GEO	h= 35786 km i= 0 deg ; e= 0	Telecom, Data	h= 300 km i= 5.2 deg ; e= 0	787.0	Hi-T	12.0	722	1024
				787.0	Hi-Isp	14.0	651	1095
HEO	h= 600 x 60530 km i= 59 deg ; e≈ 0.81	PROBA-3	h= 300 km i= 59 deg ; e= 0	740.6	Hi-Isp	11.8	613	848
Sun/Earth L1	d= 1491110 km	Scientific	h= 300 km i= 5.2 deg ; e= 0	787.0	Hi-Isp	19.1	943	803
Earth/Moon L1	d= 326370 km	Scientific, Communication, Infrastructure	h= 300 km i= 5.2 deg ; e= 0	787.0	Hi-Isp	16.7	850	896
NEA	d= 2047990 km	Scientific	h= 300 km i= 5.2 deg ; e= 0	787.0	Hi-Isp	19.8	1015	731

Table 3. VEGA+EPM launch system performance summary.

When the system performance is evaluated considering the alternative HET models identified (of which only limited performance data are available), a variation of the P/L mass <13% is found, which do not modify in any way the technical feasibility of the mission.

IV. Module configuration and preliminary design

A. Configuration

1. Design guidelines

Since the EPM and its passenger satellite shall be accommodated within the current VEGA fairing, stringent constraints to envelope and dimensions are given and an effort in minimization of the module stowed envelope is necessary, with particular attention paid to its dynamic envelope that must not interfere with the passenger satellite one, in order not to damage it.

The EPM shall feature full autonomy during all the mission phases: on-board computation, reconfiguration from ground by telecommand, solar power generation, thermal and attitude control and all the typical functionalities of a spacecraft shall be provided; it is assumed that the passenger satellite is carried in an off or safe & stowed configuration, the EPM providing to the satellite, through a dedicated I/F, the minimum power required to its survival during the orbit transfer duration and a data-link to download a predefined set of housekeeping data.

Thus, the EPM features a configuration similar to the one of a standard S/C, but, taking into account its mission peculiarities and the electric propulsion system capability, some optimizations to the S/S configurations have been carried out, in order to reduce mass (which is a driver for the mission feasibility) and power budgets and to increase reliability.

Redundancy concept is based on an hybrid approach between the S/C and the launcher ones, so duplications of components is reduced to a minimum, preferring hot redundancies, in order to meet a reliability figure in line with the current VEGA requirement and keep mass low.

Flexibility, which is a key feature of the EPM, is pursued by means of the above described dimensioning of the

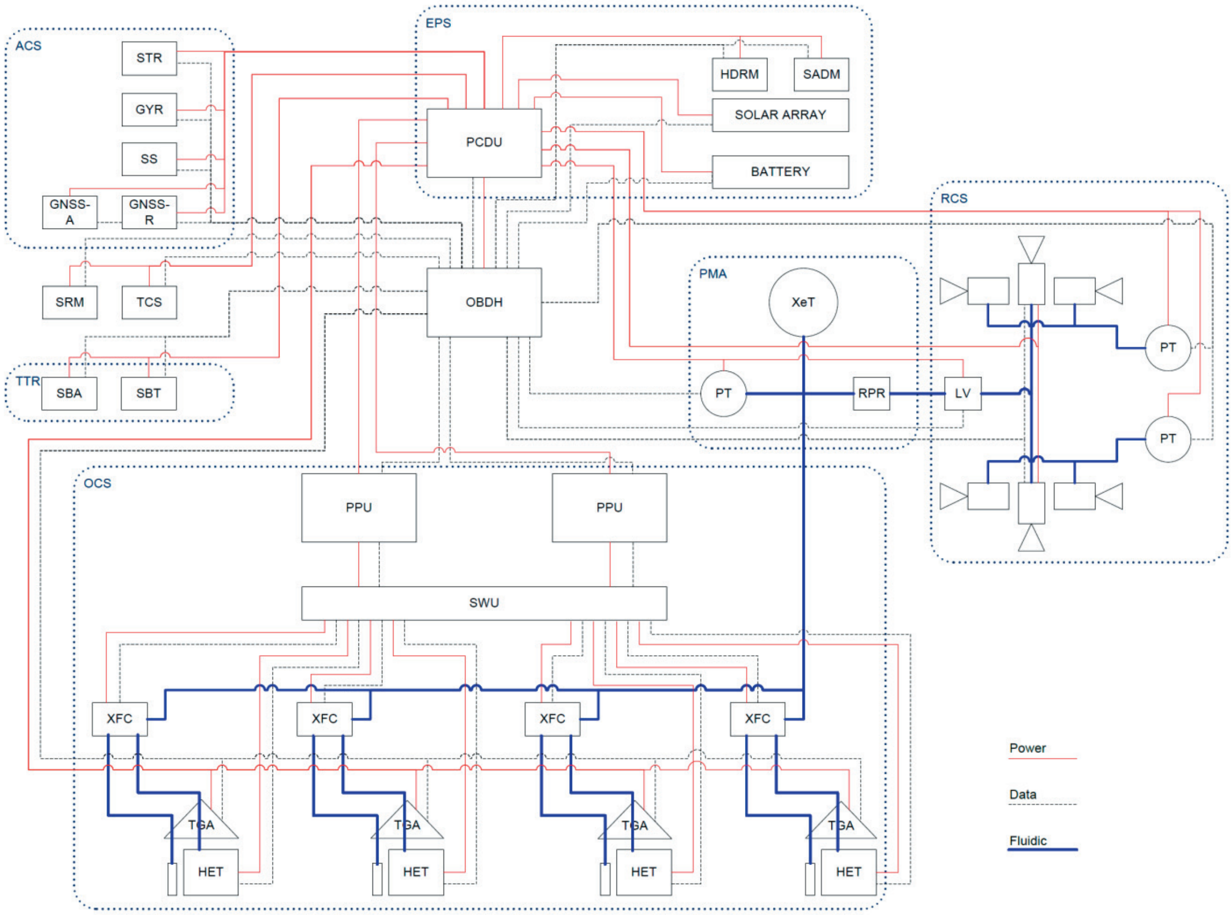

Figure 5. EPM functional block diagram.

power and SEP subsystems, since tank capacity and propellant budget envelope any possible mission and solar power generation allows both high thrust and high I_{sp} modes, depending on the mission priority (transfer duration minimization or ΔV maximization, respectively).

2. Sub-Systems description

The EPM product tree, reported in Table 4, is the final result of an iterative process, carried out in the system design loops in coupling with the performance estimation, output of the mission analysis. Figure 5 reports the functional block diagram that shows the data, power and fluidic connections between each S/S.

Here below, a synthetic description of each sub-system is provided.

<u>Electric Power S/S</u>

- The estimated overall power need is 10,9 kW (for baseline Hi-T thruster mode), nevertheless solar array are dimensioned in order to satisfy up to 12.5 kW power generation required by the Hi-I_{sp} thruster mode.
- High performance 29% efficiency Ga-As triple junction cells are considered in order to minimize the SA surface, which is estimated about 35 m^2.
- Dimensioning accounts for the power degradation induced to the solar arrays by the Van Allen radiation belts, in compliance with the requirement on transfer time (9 months for Galileo reference case).
- SA drive mechanisms guarantee the constant alignment of SA to the sunlight direction, in order to maximize the power generation.
- Batteries are dimensioned under the hypothesis that HET are powered off during eclipse phases.

C.I. number	Level	Item	Acronym	Number of Items	Note
1	**1**	**VEGA EP Module**	**EPM**		
1.1	**2**	**Structure**	**STRC**	1	
1.2	**2**	**LV to EPM I/F ring**	**MIR**	1	
1.3	**2**	**Satellite Release Mechanism**	**SRM**	1	
1.4	**2**	**Thermal Control System**	**TCS**		
1.4.1	3	Multi-layer Insulation	MLI	AR	
1.4.2	3	Heater	HTR	AR	
1.4.3	3	Temperature sensors	TS	AR	
1.4.4	3	Radiator panels	RDT	AR	
1.4.5	3	Heat pipes	HPIP	4	
1.5	**2**	**On board Data handling**	**OBDH**	1	
1.6	**2**	**Electrical Power System**	**EPS**		
1.6.1	3	Power Control & Distribution Unit	PCDU	1	
1.6.2	3	Solar Array	SA	2	
1.6.3	3	Battery	BAT	2	
1.6.4	3	SA Hold-down & release mechanism	HDRM	2	
1.6.5	3	SA Drive Mechanism	SADM	2	
1.7	**2**	**Attitude Control System**	**ACS**		
1.7.1	3	Star Tracker	STR	2	
1.7.2	3	Gyroscope	GYR	1	
1.7.3	3	Sun Sensor	SS	12	
1.7.4	3	Global Navigation Satellite System	GNSS		
1.7.4.1	4	GNSS Antenna	GNSS-A	2	
1.7.4.2	4	GNSS Receiver	GNSS-R	1	
1.8	**2**	**Telemetry Telecommand & Ranging**	**TTR**		
1.8.1	3	S-band Antenna	SBA	2	
1.8.2	3	S-band Transponder	SBT	1	
1.9	**2**	**Propulsion System**	**PS**		
1.9.1	3	Pressure Management Assy	PMA		
1.9.1.1	4	RCS Pressure Regulator	RPR	1	
1.9.1.2	4	High Pressure Xenon Tank	XeT	4	
1.9.1.3	4	Fill&Vent Valve	FVV	2	
1.9.1.4	4	Pressure Transducer	PT	1	
1.9.1.5	4	Piping, Supports		1	
1.9.2	3	Orbit Control System	OCS		
1.9.2.1	4	Hall Effect Thruster	HET	4	2, MEO mission only
1.9.2.2	4	Power Processing Unit	PPU	2	
1.9.2.3	4	Hi-Press Xenon Flow Controller	XFC	4	2, MEO mission only
1.9.2.4	4	Electrical Filter Unit	FU	0	2 or 4, Snecma PPS-5000 only
1.9.2.5	4	Switching Unit	SWU	1	0, MEO mission only
1.9.2.6	4	Thruster Gimbal Assembly	TGA	4	2, MEO mission only
1.9.2.7	4	Piping, Supports		1	
1.9.3	3	Reaction Control System	RCS		
1.9.3.1	4	Resistojet Thruster	RJT	6	
1.9.3.2	4	Solenoid valve	SV	6	
1.9.3.3	4	Latching valve	LV	1	
1.9.3.4	4	Fill&Vent Valve	FVV	2	
1.9.3.5	4	Pressure Transducer	PT	2	
1.9.3.6	4	Piping, Supports		1	
1.A	**2**	**Harness**	**HRN**	AR	
1.B	**2**	**Software**	**S/W**	N/A	

Table 4. EPM product tree.

- Bus voltage is preliminary set to 100 V, sun-light regulated.
- The distribution logic implemented in the PCDU feeds the bus sub-systems excluding OCS with up to 850 W peak, while providing, in sun-light condition only, all the rest of the available power to the two PPU, managing the power reduction related to the solar cells progressive degradation.
- Due to the high SA surface, the low allowable volumes inside the VEGA fairing and the stringent mass constraint, a big challenge is represented by the SA accommodation. A traditional configuration with honeycomb panels as structural substrate is deemed critical in terms of static and dynamic envelope, but, above all, in terms of mass, since a 3 m long cantilever panel assembly could easily hit the fairing or the Galileo satellite, which is not an acceptable scenario, whereas a SA wing mass exceeding 100 kg could endanger the mission feasibility. Thus, to guarantee the feasibility a break-through technology such as flexible solar panels may be envisaged, such the one proposed in the present study and currently developed by Orbital-ATK in US (Ultraflex solar array, flight proven on NASA Mars Phoenix Lander – similar diameter as EPM need – and considered also for NASA/ESA Orion MPCV and Orbital-ATK Cygnus applications).

<u>Propulsion S/S</u>

- The main propulsion for orbit raising (OCS) configuration foresees four 4.5 kW Hall Effect Thrusters, fed by one XFC each and two PPU's that can operate one HET each, HET selection being in charge to a dedicated SWU; only two thrusters operate at a time.
- BOL operating point of the baseline BPT-4000 (now designated as XR-5) is:
    - Hi-T mode: 291 mN, 1788 s @ 4.5 kW, 300 V ;
    - Hi-I_{sp} mode: 253 mN, 2020 s @ 4.5 kW, 400 V.
- During the mission, a progressive reduction of the power fed to the PPUs occurs due to the solar cells degradation, which shall be managed by the PPUs by switching to several different set-points that feature lower power and thrust at constant I_{sp}.
- HET are mounted on gimbal devices in order to vectorize thrust for GNC purpose.
- A Xenon resistojet based RCS has been integrated into the propulsion system, in order to manage the module attitude throughout the mission, when rapid slews are required and when OCS is switched off during eclipses; such choice, even if penalized by a low I_{sp}, benefits from the common propellant feeding with the OCS, that allows relevant mass optimization.
- 6 resistojets arranged in triads guarantee the three axes control of the EPM.
- Typical figures of OTS resistojets, when Xenon fed, are considered: $I_{sp} \sim 50$ s and thurst ≤ 100 mN.
- The PMA, in charge to store the Xenon and feed it at low pressure, is common to both OCS and RCS. A Pressure Regulation Unit is not included in the product tree, since the high pressure XFC's are able to receive high pressure gas inlet, simultaneously reducing pressure and metering the flowrate of the Xenon, thanks to the innovative proportional piezoelectric valves which are based on.
- A mechanical Xenon pressure regulator feeds the resistojet triads, allowing the contemporary use of OCS and RCS, when required.
- Xenon is stored in supercritical conditions into a set of four leak-before-burst, 132.7 l each, high pressure COPV tanks of Orbital-ATK, which are able to store up to 1018 kg of propellant at 180 bar and 40°C.

<u>Attitude Control S/S</u>

- A first analysis loop confirm then feasibility of avoiding the use of reaction wheels if gimballing of the HET is foreseen: during the propulsive phase a thrust misalignment is commanded in closed control loop to keep the EPM axis and the thrust vector parallel to the EPM velocity vector, providing a small and constant torque during every propulsed arcs.
- Resistojet triads are used to provide torque to:
    - recover the thrusting attitude coming out from an eclipse phase (during which the EPM is assumed to be in inertial attitude);
    - control the roll velocity of the EPM (required both to ensure an acceptable thermal environment to the passenger S/C and to track the Sun direction for solar power generation).
- Magnetometers and magneto-torquers are not included in the ACS configuration since their use is not recommended beyond LEO altitudes.
- Attitude and position knowledge is guaranteed by:
    - 2 Star Trackers;
    - 1 Gyroscope unit;
    - Sun sensors;

- 1 Global Navigation Satellite System.

- The GNSS is preliminary included in the configuration for sake of conservativeness, but, since it is expected that its use will be no more possible over MEO orbits, and thus could not serve during large parts of missions to GEO, HEO and Lagrange points, it is reasonable to expect that it will be discarded in the following design loops, being its function guaranteed by Gyro acceleration data integration and periodic ranging of the EPM.

Telemetry, telecommand & Ranging S/S
- One S-band transponder with ranging functionality is coupled to two S-band antennas.
- Dimensioning is performed taking into account the most demanding Sun-Earth Lagrange points mission scenarios.

Satellite Release Mechanism
- A compact, flexible and flight qualified OTS solution has been preliminarily identified in the US Planetary System Corp. MKII LightBand non-explosive separation system, for interfacing a generic passenger satellite to its interface ring, with no modification to the mechanical layout required to be mounted on top of VEGA without EPM.

Thermal Control S/S
- It is based on a standard satellite concept, with an effort to keep the thermal control as much passive as possible, minimizing the use of heaters.
- Particular attention is given to the HET heat dissipation management: assuming a 90% PPU efficiency and a 50% HET efficiency, whose 70% is carried away by the plume, it is estimated that up to 14% of the propulsive electric power is dissipated as heat to the EPM structure (about 1.2 kW). A dedicated analysis shall be set up in order to assess the feasibility of using heat pipes to move such heat towards structure close-up lateral panels with radiating features. At the time being, use of loop heat pipes and of a deployable radiator could not be excluded.

Structure
- The EPM structure is composed of two central shear panels that represent the main load path from the upper plate where the passenger satellite is supported to the module I/F ring and the launcher adapter. A set of lightweight lateral panels, connected through an aluminum frame, close the module and provide mechanical I/F to the EPM equipment. Both the shear panels and the close-out ones are made of an aluminum honeycomb core bonded to CFRP or aluminum sheets (depending whether the panel needs to work as a thermal radiator or not).
- Structural design of the EPM is more difficult with respect to what is usual for standard satellites, since the EPM structure must survive to the launch loads sustaining not only its own weight, but the passenger satellite one as well, with the further disadvantage that such satellite mass shows a 2.0+ m arm from the EPM baseplate, thus inducing huge loads on the shear panels, on the baseplate and the module I/F ring, which require a very careful design-to-mass approach in order to keep the structural mass under control.
- A truss configuration is proposed for the module I/F ring and the tanks' lower support, which is the optimal solution to minimize the structural mass when demanding loads must be sustained.

On-Board Data Handling
- It is based on a standard satellite concept.

B. Layout and accommodation
A preliminary layout of the EPM has been defined and an accommodation study inside VEGA fairing has been performed.

Figure 6 shows the EPM in operational in-orbit configurations, the P/L satellite on top and deployed solar arrays.

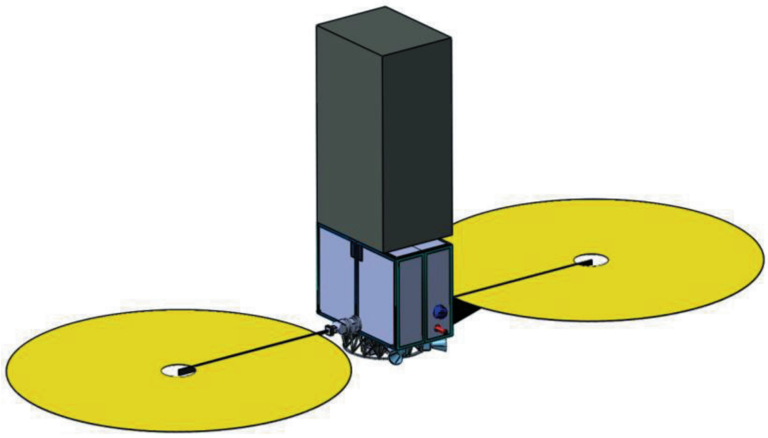

Figure 6. EPM deployed configuration.

Figure 7 shows the accommodation of the EPM with the P/L satellite on top of it into the current VEGA fairing: sufficient margin exists in the accommodation of the static envelope, such that it is likely that the dynamic analysis to identify the dynamic envelope of the composite will provide a volume still accomplishing the fairing allowed dimensions. A solar panel launch locking device which connects the stowed wing to the EPM top plate is not modelled but certainly required in order to stiffen the wing assembly and reduce its displacement under launch loads.

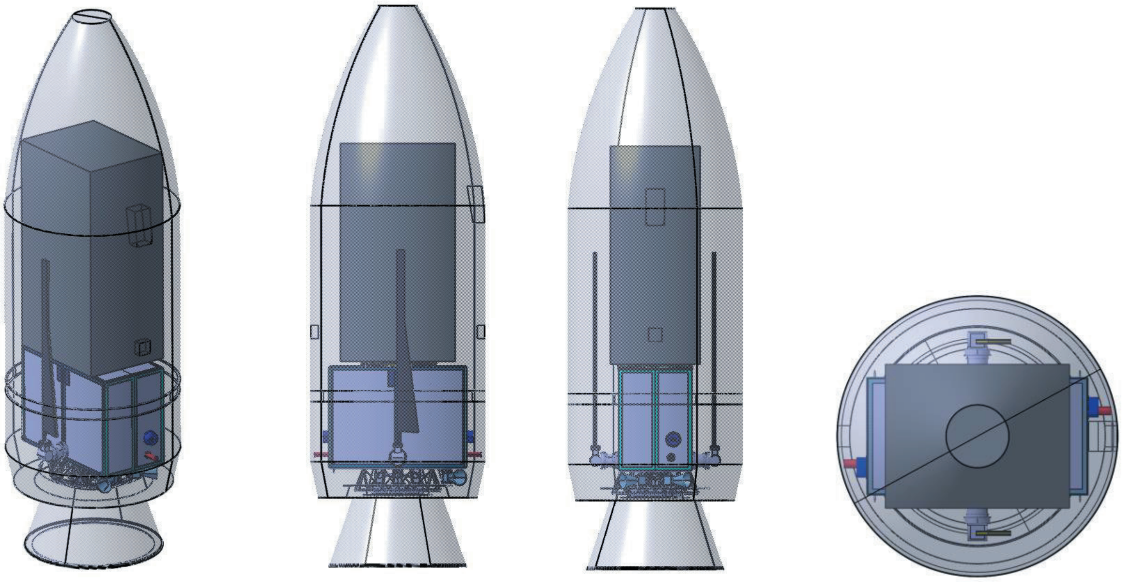

Figure 7. EPM w/ P/L satellite accommodation into VEGA fairing.

The layout of the HETs and of the resistojet triads on the outer face of the baseplate is shown in Fig. 8. HETs are placed in a cross configuration, just one couple of opposite thruster being operated at a time; missions featuring a total propellant throughput less than about 570 kg can be equipped with two HETs only. RJT triads are placed on the short side of the baseplate, in order to minimize their plume impingement onto star trackers and solar panels. Details of the 120° layout of the star trackers and of the design of the round-to-squared truss MIR can be observed, too.

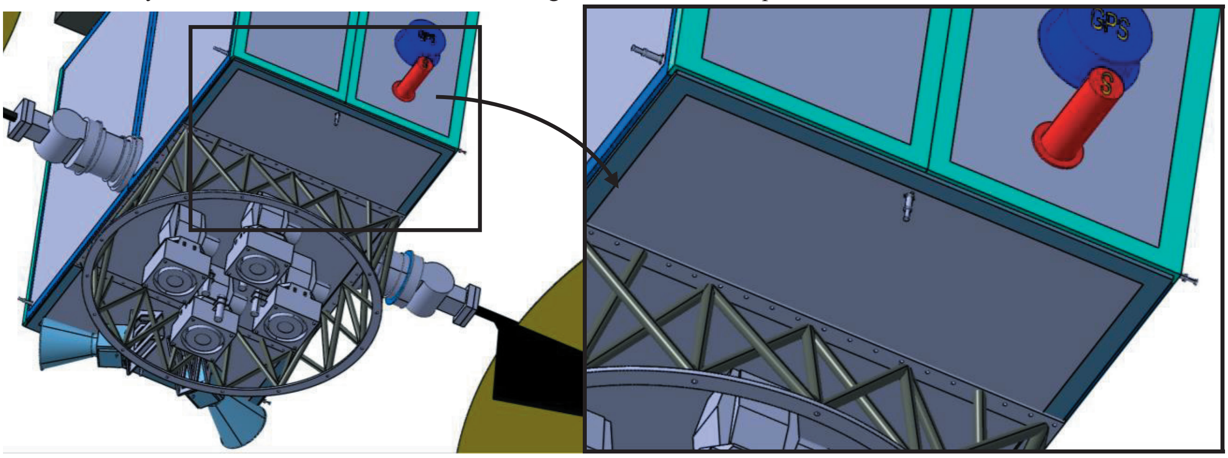

Figure 8. HET and RJT triads layout.

Figures 9 and 10 provide an overview of the equipment accommodation inside the EPM structure. All the components of the propulsion S/S lay on the baseplate, including the Xenon tanks which are connected on their ported bosses to high strength truss supports, which mainly sustain the launch loads, whereas the upper bosses are connected to the upper plate by means of sliding sleeves that avoid any induced stress while tanks are pressurized. The lateral close-out panels host the avionic boxes and the battery modules on their inner faces, the antennae and the solar arrays on their outer faces.

Both panels hosting the SA are radiator panels with aluminum skins (the remaining panels featuring CFRP skins), equipped with heat pipes to distribute the heat coming from PPUs and HETs.

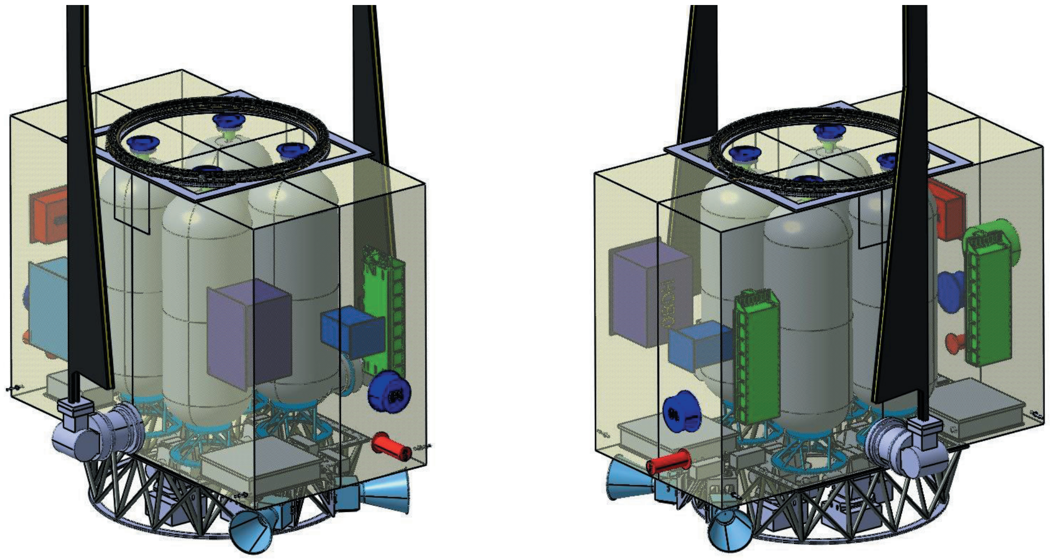

Figure 9. Equipment internal layout.

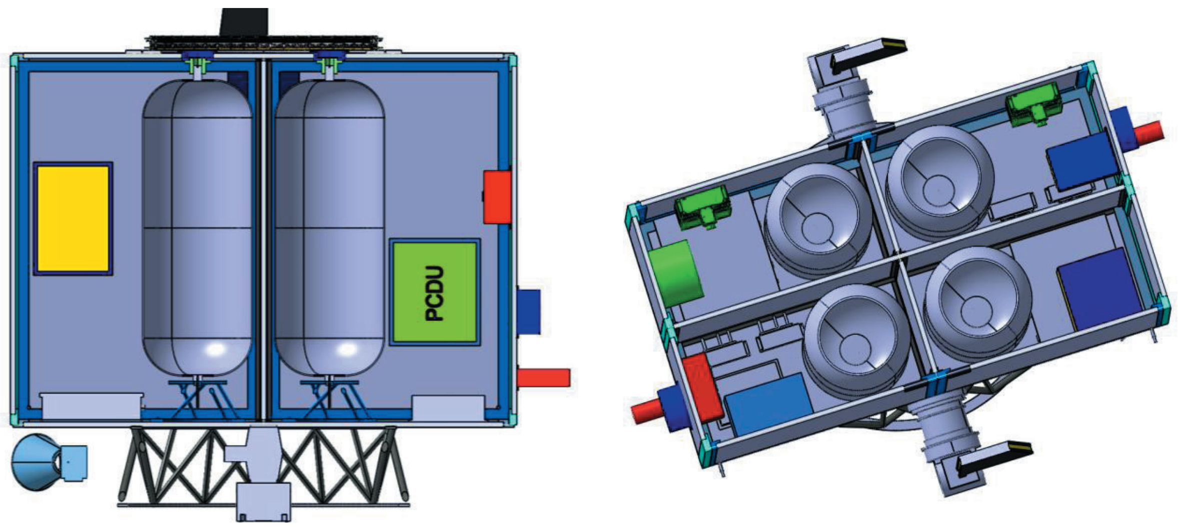

Figure 10.EPM cut-out views.

V. Conclusion

A brief summary of the pre-feasibility study of a fully autonomous module based on electric propulsion, thought as an add-on of VEGA launcher with the purpose of extending the launcher mission from its current capability up to a new and wider set of final orbits, has been presented. The technical feasibility has been assessed in terms of mission performance, S/C functional architecture, configuration and layout. The main technical criticalities that has been identified deal with the high mechanical loads that the structure shall sustain while keeping the mass under control in order not to put at risk the mission feasibility. A similar stringent mass constraint applies to the solar arrays, whose implementation by means of standard rigid substrates could not fit the relevant mass budget allocation: a flight qualified flexible solar array technology (US only, not available in Europe at the time being) is identified as an enabling factor for the mission feasibility.

5 kW class Hall effect thrusters have been identified as suitable to the EPM application, being preferred to ion thrusters due to their higher thrust-to-power ratio that allow shorter transfer durations, when this is required. Currently, the only flight qualified thruster is the former Aerojet–Rocketdyne (US) BPT-4000, now available in Europe by ESP (UK) and designated as XR-5E, which has been baselined in the EPM configuration. Nevertheless, several European HET are available as well, currently under development (Alta/Sitael HT-5k) or qualification (Snecma PPS-5000).

Development of the EPM shall comply to a design-to-cost requirement, as well, in order to fit the aggressive target recurrent price required by ESA and needed to ensure the economic sustainability of the VEGA+EPM launch service. Moreover, the EPM maiden flight shall be coordinated with VEGA-C maiden flight in 2018, to save the launch cost. To achieve such goals, a design based on OTS components that minimize the development effort has been conceived and a fast and effective system development and qualification roadmap shall be outlined in order to match the development logic.

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