HORIZON 2020

EPIC

Report

D2.2 Database of EPS key requirements and technical specifications for current and future missions

Due date of deliverable:	02/2015
Actual submission date:	
Start date of project:	2/10/2014
Work package/Task	WP2/T2.2
Lead Beneficiary	Centre National d'Etudes Spatiales
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Status	Second issue
Dissemination Level	DRAFT- CONFIDENTIAL, Final version-Public
Reference	EPIC-CNES-2.2-RP-D2.2-1.2

This project has received funding from the European Union's Horizon 2020 research and innovation programme under grant agreement No 640199

APPROVAL

Title D2.2 Database of EPS key requirements and technical specifications for current and future missions	
Issue 1.2	**Rev** 2
Author Nicolas ArcisAlexandra Bulit	**Date** 21/04/2015
Approved by	**Date**

CHANGE LOG

Reason for change	Issue	Date
Creation	1	15/12/2014
Comments received from Advisory Board	1.1	28/01/2015
Comments received from REA	1.2	21/04/2015

CHANGE RECORD

Issue			
Reason for change	**Date**	**Pages**	**Paragraph(s)**
Comments received from Advisory Board	28/01/2015	All	All
Comments received from REA	21/04/2015	15-17, 19, 34	§4 and §7

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1 INTRODUCTION

In the frame of the Electric Propulsion Innovation & Competitiveness (EPIC) project, (grant number 640199) and more concretely its Work Package 2 "Technology Mapping & Application Requirements", this document has been produced with the aim to be the main output of Task 2.2 "Collection of Requirements from Commercial and Institutional actors, European primes and Agencies".

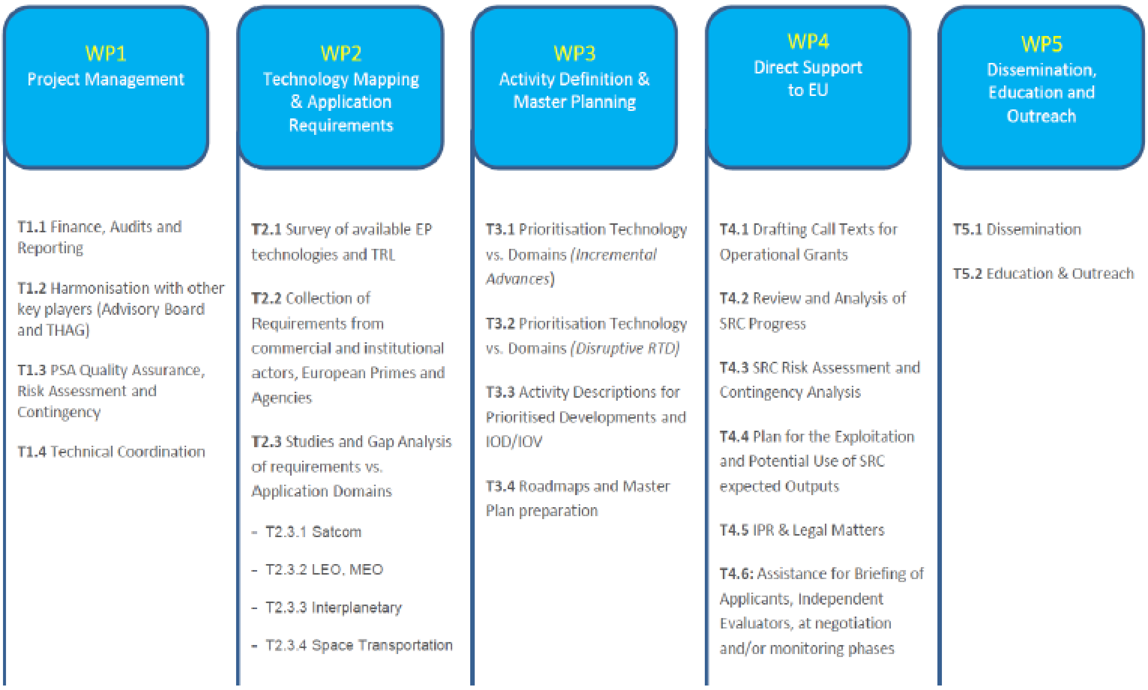

Figure 1-1: EPIC Work Logic

This document gives an overview of EP-related requirements. Information has been gathered through national space agencies, the related European Space Technology Harmonisation dossiers [RD1][RD2][RD3], direct consultation with satellite primes or mission designers, and outputs of a dedicated Workshop organized in Brussels on November 25-28 2014 in the frame of the EPIC project [RD7]. The overall objective is not to provide detailed technical specifications, since those are too spacecraft-specific, but rather to provide high-level requirements for future generic space missions in the following areas:

- LEO (e.g. Earth Observation, Earth Science)
- MEO (e.g. GNSS)
- GEO (e.g. telecommunications)
- Space transportation (e.g. launcher kick stages, space tugs)
- Interplanetary and science.

Those requirements should be to encourage competition between potentially eligible technologies so that Europe could develop the most innovative and globally competitive solutions.

2 REFERENCE DOCUMENTS

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[RD3] *European Space Technology Harmonisation Technical Dossier – Power Management and Distribution*, Ref. ESA/IPC/THAG(2013)7, Issue 3, Technical Note.

[RD4] *EPIC D2.1 EP and EP-related technologies database, Issue 1.2*, EPIC-CNES-2.1-RP-D2.1-1.2

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[RD6] H. Gray, O. Sutherland, *Development and Qualification Status of the Electric Propulsion System for the BepiColombo mission*, IEPC-2013-114, 33rd International Electric Propulsion Conference, George Washington University, USA, 2013.

[RD7] *EPIC D2.3 Workshop 1 Report, Issue 1.1*, EPIC-CNES-2.3-RP-D2.3-1.1

[RD8] *Mission concepts for a Polar observation system*, Executive summary, POS.UofS.ES.001, issue 1.0, 27/11/2014.

[RD9] *CDF Study Report, RAM-EP, Advanced Electric Propulsion for very low Earth Orbit missions*, CDF-55(A), January 2007.

3 LIST OF ACRONYMS AND ABBREVIATIONS

ADR: Active Debris Removal

Airbus DS: Airbus Defence and Space

AGILE: Advanced Galileo Injection in Low earth-orbit using Electric-propulsion

AOCS: Attitude Orbit Control System

ARM: Asteroid Retrieval Mission

EO: Earth Observation

EOL: End Of Life

EOR: Electric Orbit Raising

EP: Electric Propulsion

EPPM: Electric Propulsion Pointing Mechanism

EWSK: East-West Station Keeping

FCU: Flow Control Unit

FEEP: Field Emission Electric Propulsion

GEO: Geostationary Earth Orbit

GIE: Gridded Ion Engine

GNSS: Global Navigation Satellite System

GOCE: Gravity field and steady-state and Ocean Circulation Explorer

HEMPT: High Efficiency Multistage Plasma Thruster

HEO: Heliosynchronous Earth Orbit

HET: Hall Effect Thruster

HPT: Helicon Plasma Thruster

Isp: Specific Impulse

LEO: Low Earth Orbit

LEOP: Launch and Early Orbit Phase

MEO: Medium Earth Orbit

MPD: Magnetoplasmadynamic thruster

NGGM: Next Generation Gravity Mission

OTV: Orbital Transfer Vehicle

PPT: Pulsed Plasma Thruster

PPU: Power Processing Unit

QCT: Quad Confinement Thruster

RAAN: Right Anomaly Ascending Node

SSO: Sun-Synchronous Orbit

TAS: Thales Alenia Space

TC: Telecommand

TM: Telemetry

VLEO: Very Low Earth Orbit

XIPS: Xenon Ion Propulsion System

4 LEO/VLEO MISSIONS

4.1 Introduction

Most Low Earth Orbit (LEO) satellites are currently equipped with monopropellant propulsion. Delta-V (velocity increment to be provided to the spacecraft) sizing is quite low. Pending type of platforms, it goes from 0 (nano satellites, some micro-satellites) to 150-180 m/s (big satellites). However, several new needs are pushing toward an increased Delta-V capacity, which will fully justify the electrical propulsion emergence.

- Very Low Earth Orbit (VLEO) needs

VLEO orbit corresponds to orbit altitude lower than 400-500 km.. This orbit is very useful for Earth Observation (due to the constraints on the optics but also the Lidar and Radar). At these altitudes, the atmospheric drag becomes very high and the Delta-V required to compensate for this drag becomes important. Some missions (such as GOCE) are based on drag compensation (i.e. that a continuous thrust is required in order to compensate in real time the atmospheric drag). Electric propulsion (EP) has already proven during the GOCE mission that it is a very suitable candidate fulfilling this need because it allows for continuous operation and for very small and fine thrusting capability.

- Launcher offer

Only few launchers are available on the market. The price differences can be very high (eg 40 M€ for VEGA, 80 M€ for Soyuz and 150 M€ for Ariane 5 in case of European preference, and 50 M€ for Falcon 9 in the US, etc.) and there are obvious threshold effects when satellite mass forces to go from one launcher to the upper class one. Using electric propulsion can allow important mass savings allowing using lower class launchers.

- LEO constellations

A new market for constellation of LEO satellites is appearing for global internet which will lead to an enormous amount of EP systems to be built and at very low cost.

- Multiple passenger flexibility

In order to get launch prices as low as possible, multiple payload can be launched at the same time but with mission orbit quite different. A large Delta-V is then required to go from the injection orbit to the mission orbit.

- Space debris mitigation rules

Space agencies and member states are now issuing rules or law to mitigate space debris to prevent both in orbit collisions and casualties on-ground. These new rules impose in some cases to increase drastically the on-board available Delta-V and electric propulsion could partially fulfill this need.

- LEO to high orbits

This mission case rational is to use a low cost launcher able to separate the spacecraft in LEO orbit while the mission orbit is in MEO, GEO or even L2. A very large Delta-V is therefore required in this case to perform the orbit transfer. However this transfer is more complex and less rentable because of the long duration in the Van Allen Belts increasing the irradiation time and dose on the spacecraft (only 1 or 2 kRad for GTO to GEO and up to 1 MRad if going through the Van Allen Belts).

- Specific nanosat missions needs

Current nanosat missions do not implement propulsion but the evolution of available technologies, the miniaturisation of other nanosat bricks together with the increase of nanosat range should enable the use of electric propulsion in this type of satellite. This will enable nanosats and CubeSats to manoeuvre for the first time and thus significantly increase the mission utility and flexibility.

4.2 Proposal for different mission cases

In the following paragraphs examples of mission studies are presented, allowing the elaboration of ranges for the requirements to be defined. Therefore the numbers given hereafter should not be seen as constrains for the derivation of those requirements.

4.2.1 VLEO needs

For some missions, it can be very interesting to use very low Earth orbits going down to 200-250 km altitude. For these missions, the main driver for propulsion is the total Delta-V (required to compensate for high drag).

An example of Delta-V required is depicted in Figure 4-1 (pending S/m ratio). One can rapidly see that for several year missions, and very low altitudes the required Delta-V can be higher than 1 km/s. This cannot be met with current chemical propulsion systems.

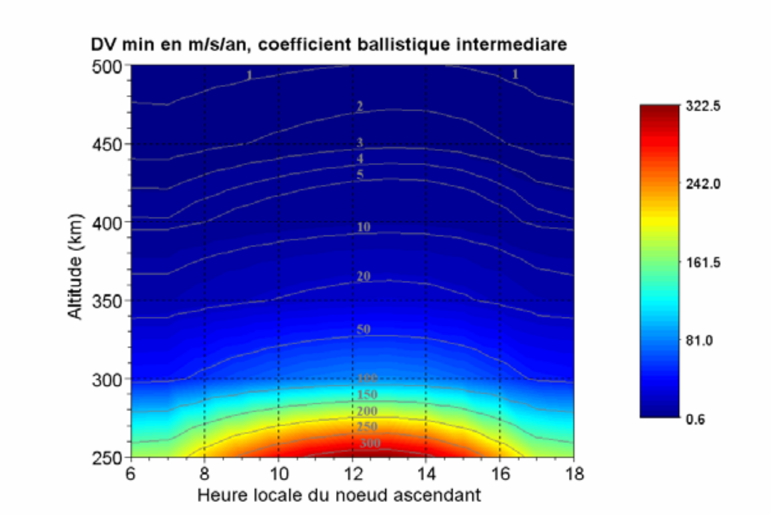


Figure 4-1: Delta-V as a function of altitude and ascending node for a given S/m

This Delta-V requirement can be converted in:

- Thruster total impulse requirement
- Xenon tank capacity
- Specific Impulse (Isp).

VLEO mission design is driven by the drag forces to be overcome and the solar power available. The drag is determined by the altitude and the shape of the spacecraft which may include deployed solar arrays. The solar power available is constrained by the shape of the spacecraft and atomic oxygen effects. Body mounted solar panels are preferred if the spacecraft has to execute rapid manoeuvres.
different propulsion solutions are likely to be appropriate for different sizes of spacecraft and different mission durations. A small spacecraft in the 200 kg range with limited solar power may need high thrust to power and be prepared to sacrifice mission duration because of the higher propellant utilisation. A spacecraft in the 1000 kg range may be able to generate sufficient power to give high specific impulse and low propellant utilisation and therefore a longer mission.
The power requirements will also be determined by the mission objectives and payload requirements. A low-power optical or infrared sensor with a limited duty cycle is likely to have modest power demands. A synthetic aperture radar or a high duty cycle sensor with very high data rate downloads will have much greater power requirements.

Thus the choice of EPS is very mission dependant as illustrated by the examples below taken from the SAT250 phase ø project (CNES).

For a satellite flying at 250 km during 7 years, the following figures can be derived (it has to be noticed that drag being proportional to S/m, the required thruster characteristics is not function of the satellite mass, but only to surface which can be optimised in a certain way, but will be anyway larger for big satellite than for small satellite, considered only 50% bigger in this case for 400% heavier satellite) :

- For a 200 kg satellite, no more than 40 W average can be dedicated to electric propulsion (400 W can be envisaged with 10% duty cycle). For a 1000 kg VLEO satellite, 200 W-300 W average can be envisaged.
- Thrust : > 2 to 3 mN in case of permanent thrust,
- Thrust : > 20 to 30 mN if the thrust duration shall be limited to 10% of the orbit.
- Total impulse: 0.42 to 0.63 MN.s (note: It can however be envisaged to install several thrusters in order to fulfil the total impulse requirement, assuming that overall propulsion system is able to cross strap the functions).
- Cycling requirements: <2000 on/off in case of permanent thrust, >100,000 on/off in case of 2 manoeuvres per orbit. It should be noted that this last figure can be decreased using a 0.8 kWh battery, enabling to make one firing per day, or one full orbit every 10 orbits.
- Specific thrust: >50 mN/kW; this figure is obtained for a small satellite (200 kg/200 W class that cannot dedicate more than 20% of average power to the propulsion system). It could be decreased for a bigger satellite that gets more power for the same thrust requirements (e.g.: 20 mN/kW is acceptable for a 1000 kg satellite).
- Propulsion subsystem mass: this is not only the thruster that is the criterion but the overall propulsion system wet mass (including propellant, thrusters, power processing units, orientation mechanism if used, and propellant management system). This wet mass shall be limited to 30% of the satellite mass, at first order. This means that for a satellite of 200 kg and an Isp of 1000 s, the dry-mass shall be limited to 13 kg while for 2000 s Isp it could reach 40 kg (Note: this could be much higher for the 1000 kg class satellite).

The main figures are summarized in the table hereafter. Of course, they are examples and are not hard requirements.

	Duty cycle [%]	Power [W]	Total Impulse [MNs]	Thrust [mN]	Power/ Thrust [W/mN]	Isp [s]	Cycling	Dry mass * [kg]
VLEO small	100	< 40	> 0.42	> 2	< 20	> 1000	~ 2000	15–40
VLEO small	10	< 400	> 0.42	> 20	< 20	> 1000	~ 100,000	15 40
VLEO big	100	< 200	> 0.63	> 3	< 50	> 1000	~ 2000	100–200
VLEO big	20	< 1000	> 0.63	> 30	< 50	> 1000	~ 50,000	100-200

* Pending Isp, the driver being the wet mass of the subsystem, better Isp allows for higher dry mass.

Table 4-1: Example of EP requirements for a small (200 kg) and large (1 t) VLEO spacecraft, for a 7 year mission

VLEO needs have also been analysed by OHB for a 400 kg type satellite [RD7] and are consistant with hereabove figures corresponding with a satellite that would not use continuous thrust. They are presented in Table 4-2.

Parameter	Requirement
Power [W]	300
Isp [s]	≥ 1500
Thrust [mN]	≥ 18
Efficiency [-]	≥ 40%
Xe throughput [kg]	≥ 48 (12 per thruster)
Total firing time [h]	≥ 9600 (2400 per thruster)
ON/OFF cycles	≥ 10400 (2600 per thruster)

Table 4-2: OHB-CGS thruster needs for a 400kg class V-LEO satellite

In the same way [RD7], QinetiQ suggest to use the existing T5 system to fly an EP modified PROBA platform in VLEO with a maximum peak power available for electrical propulsion bound by 400 W and cycling limited to 8000 cycles.
The proposed minimum altitude being limited to 350 km it is needed to perform maneuvers 3 times a day but lower altitude would require 1 manoeuvre per orbit (5 times more).

Airbus DS proposal [RD7] is also consistent with the previous figures regarding the high level requirements envelope (for the main propulsion system):

- Nominal thrust range: from 4 to 50 mN
- Optimal Isp range : 3000 s to 5000 s
- Thrust modulation: from 5% to 100 %
- Firing duration: continuous along the operative mission duration.

The possibility of air breathing to feed the EP thruster is also considered for this kind of missions, allowing decreasing the propellant mass for the EP system and increasing the mission duration (because no propellant, except maybe for the neutraliser to be used to compensate for the spacecraft charging, needs to be stored on board the satellite).

4.2.2 Launcher offer

The design of LEO and VLEO missions is also driven by the launcher performance and its associated cost. Indeed a launcher can only inject on a certain orbit with a certain Delta-V, a certain class of satellite defined mainly by its mass at launch and payload mass in orbit. Therefore, to decrease the costs of the mission, it might be necessary to change to a cheaper launcher reducing the wet mass of the entire system using EP.

For instance, an analysis was done on several early project phases (SAMUEL) envisaged for CNES missions. Two main cases can be envisaged. The first one is typically to stay under standard European launcher performance. For instance, Vega launcher is able to launch roughly 1500 kg payload to LEO orbit for a price of 40 M€. If we want to keep the European preference for a satellite (or a set of satellites) that would weight 1600 kg, this would force the use of Soyuz launcher for a price of 80 M€. This means this is mainly related to:

- Subsystem cost (to be challenged with launcher cost savings),
- Subsystem wet mass for standard Delta-V.

For this analysis, the following figures are taken into account. The typical Delta-V required for a LEO mission (that does not require controlled re-entry) is around 150 m/s for LEOP, station keeping and deorbiting (for less than 25 years re-entry). For a 1600 kg satellite, it represents 130 kg of hydrazine and a propulsion subsystem weighting around 30 kg. In order to save 100 kg of mass and come back to VEGA launch capability, this wet mass shall be brought back to less than 60 kg using electric propulsion as long as the EP system is capable to perform the required manoeuvres of the mission (assuming adequate electrical power and thrust levels are mission dependent). Since the Delta-V is not very high, Isp is less important than dry mass even though the overall wet subsystem mass shall be the criterion. An example would be a 37 kg dry subsystem with 23 kg of Xe and an Isp of 1000 s, or 49 kg dry for 11 kg Xe and an Isp of 2000 s. It should be noted that the higher the Isp, the longer the transfer time, and that the duration allowed for the transfer is mission specific.

As far as power is concerned, most heavy LEO missions are able to provide more than 1-2 kW to electric propulsion in peak. In order to avoid mission outages as far as possible and also LEOP/deorbiting durations, the thrust shall be high enough to perform the manoeuvers in a limited duration.

One can therefore derive the following drivers for the presented example:

- Isp > 1000 s
- Dry mass as low as possible
- Power for EP < 1000-2000 W
- Thrust > 30 mN (allowing to reach target orbit in less than 2 months for 50 m/s and 50% duty-cycle).

	Dutycycle	Power	Total Impulse	Thrust	Isp	cycling	dry mass	Price
LEO medium/big	50%	1000-2000W	-	> 30mN	> 1000 s	~ 5.000	30-50 kg *	< 10M€

* Pending Isp, the driver being the wet mass of the subsystem, better Isp allows for higher dry mass.

Table 4-3: Example of EP requirements for a LEO spacecraft in the range 500-1500 kg

These high level requirements are supported by Airbus DS with the following proposal [RD7]:

- Maximum available power: from 200 W (300 kg class spacecraft) to 2 kW (1 ton class spacecraft)
- Ideal range of power-to-thrust ratio: around 10 W/mN (20 W/mN could still be viable in some cases)
- On/off cycles: up to 30 per day for the whole mission duration
- Cost optimization: being an "opportunity" market, i.e. in direct competition with chemical propulsion, EP cost shall be as low as possible in order to have a clear business case for it.

For the case of satellite constellations (see next paragraph), the launch price is an important driver for constellation deployment. The satellite architecture can be optimized in a way to be able to fly as many satellites as possible in a single launcher (optimized geometry to fill the fairing, specific satellite/launcher interface on dispensers, etc.).

Electric propulsion is a very efficient way to decrease the satellite mass and, therefore, to launch more satellites on the same launcher. The requirements will depend on the design of the constellation

In this constellation case, the launched cost will be the driver and thus, the recurring cost of electric propulsion sub-system is a major driver.

4.2.3 LEO constellations

Very recently the American SpaceX has announced its global satellite Internet project which would include some 4000 satellites in LEO (altitude around 1100 km) to be operational within five years. They would need to avoid interfering with signals from the GEO satellites. They plan to build spacecraft of several hundred kilograms at launch and use HETs for station keeping manoeuvres. The objective of this SpaceX network is to feature inexpensive antennas to deliver broadband to areas that are unlikely to be served by terrestrial broadband in the near future.

This announcement followed the one of OneWeb LLC that Virgin Galactic of London and Qualcomm Inc of San Diego had agreed to invest in OneWeb's 650- satellite system, that will use the Ku-band network at 1200 km altitude.

Therefore in addition to the reduction of the launch cost, the design of a constellation of LEO spacecraft will require an industralisation of the EP system manufacturing and acceptance testing processes, allowing the rapid availability of hundreds of EP systems at low cost.

4.2.4 Multiple passenger flexibility

This mission case is typically related with a micro/mini-satellite flying as a co-passenger on a launcher like Soyuz or Vega; but it can also be two small satellites on a dual launch with the Falcon 9 launcher Satellite pairing is not an easy task for launcher authority and launch opportunities do not occur very often if the co-passenger orbit is not flexible to fulfil mission needs.

In this case, the satellite equipped with electric propulsion is separated on an orbit far from it mission orbit and it shall reach it thanks to its propulsion system. This mission case has been analysed in the frame of SAMUEL early phase in CNES.

Two drivers will be here considered, the first one is the thrust, because, it was chosen to consider LEOP durations not longer than typically 6 months, together with a subsystem wet mass as low as possible in order to keep the satellite within co-passenger allocations. According to Arianespace information, two main points can be considered: 200 kg microsatellites (for Soyuz external ASAP) and 400 kg mini-satellites (for Soyuz internal ASAP and VEGA Vespa).

It will anyway be very difficult to modify in a reasonable time the injection plane (big inclination change or Right Anomaly Ascending Node – RAAN - change), but altitude/eccentricity and small inclination changes can be envisaged and would allow for instance going from a Sun-Synchronous Orbit (SSO) 6 am orbit to another SSO 6 am orbit but with 150 km altitude difference. This represents a Delta-V of 150 m/s (electrical cost to be compared with 110 m/s for an impulsion chemical propulsion) for the LEOP only (to be compared with typical 10 to 20 m/s in case of big launcher dispersions) and doubles the overall satellite Delta-V need at first order.

For instance, for a 400 kg satellite, in order to be able to perform these 150 m/s in less than 6 months, the thrust shall be higher than 4 mN if the thrust is continuous. But assuming that power available to propulsion during this phase is limited to typically 150 to 200 W in average, the power-to- thrust ratio shall be higher than 40 to 50 W/mN. The maximum power being limited to 75% of the solar array peak power, the thruster power range shall be at maximum 500 W to 600 W.

For a 200 kg satellite, in order to be able to perform these 150 m/s in less than 6 months, the thrust shall be higher than 2 mN if the thrust is continuous. But assuming that power available to propulsion during this phase is limited to typically 80 to 100 W in average, power-to- thrust ratio shall be higher than 40 to 50 W/mN. The maximum power being limited to 75% of the solar array peak power, the thruster power range shall be at maximum 250 W to 300 W.

	Dutycycle	Power	Total Impulse	Thrust	Isp	cycling	dry mass
Multiple micro	20%	400W	-	10mN	> 1000 s	~ 10.000	10-15 kg *
Multiple mini	20%	800W	-	20mN	> 1000 s	~ 10.000	30-50 kg *

* Pending Isp, the driver being the wet mass of the subsystem, better Isp allows for higher dry mass.

Table 4-4: Example of EP requirements for micro- (200 kg) and mini- (400 kg) satellite piggybacking

4.2.5 Space debris mitigation rules

Space Agencies and Member states are currently voting rules or laws to avoid space debris proliferation but also casualty risk on ground at the end of the mission. For this purpose, the electric propulsion can be helpful if:

- Technologies designed for demise during atmospheric re-entry are developed. In particular tank material can possibly be chosen with a low temperature melting point, which is not the case of titanium currently used for hydrazine systems.
- Available Delta-V of the satellite is increased in order to further decrease the disposal orbit perigee/apogee in order to reduce the duration in orbit.

French Space operations Act is probably the most constraining at the moment, because it requires (except for duly justified waiver) a controlled re-entry at the end of life of the satellite. With respect to 25 years in orbit proposed by most agencies, this requires roughly 120 m/s more Delta-V and therefore doubles the standard Delta-V requirement of LEO satellites. However, an important point is that controlled re-entry requires a very high thrust, definitely not achievable by electric propulsion. Anyway, electric propulsion can be a very good candidate to reduce the orbit perigee down to typically 200 km and then another high thrust propulsion system could take the lead.

It was confirmed by TAS that controlled reentry will require 0.05 to 0.1 N/kg (500 kg satellite would require 25 to 50 N of thrust) to perform the final boost; but for the rest of deorbiting needs, the following figures are proposed for small satellite (< 500 – 1000 kg) [RD7]:

- Propulsion power supply scalable : 250 to 1000 W
- Thrust level: 20 to 100 mN (target 0.2 N)
- Isp: > 500 s (target of 1500 s)
    - Objective to minimize mass and volume of the tank requires a high Isp
    - But thrust could be preferred to Isp to decrease the time to deorbit
- Delta-V: 300 – 600 m/s.

Alternatively, strategies of semi controlled re-entry, entirely based on electric propulsion are under analysis and would allow to target casualty area not on a given small zone on the Earth, but over few orbits that could be correctly chosen to avoid high population density areas. This strategy requires anyway to get a relatively high thrust in order to have time at each apogee of the orbit to reduce enough the perigee altitude and to limit the end-of life (EOL) operations within a limited timeframe (maximum 6 months considered in SRL early project).

The main drivers of this mission examples would be the following. For power, it shall represent the available power for mission since the mission will be switched-off for disposal and it would be wise not to oversize the system for EOL purpose only. Therefore, a typical figure would be 0.5 W/kg (100 W average for a 200 kg satellite, 500 W average for a 1000 kg satellite). For thrust, the rationale is to be able to decrease the perigee altitude by at least 4 km per day which ends up in 180 days to deorbit a satellite on a 800/800 km mission orbit.

	Dutycycle	Power	Total Impulse	Thrust	Isp	cycling	dry mass
Improved Disposal small	20%	250W	-	20mN	> 1000 s	~ 5.000	10-15 kg *
Improved Disposal big	33%	1500W	-	100mN	> 1000 s	~ 5.000	30-50 kg *

* Pending Isp, the driver being the wet mass of the subsystem, better Isp allows for higher dry mass.

Table 4-5: Example of EP requirements due to EOL constraints (e.g. de-orbit regulations)

Contactless orbit modification of space debris and asteroids with the aid of EP energetic plumes emitted by a spacecraft at a safe distance (known as the Ion Beam Shepherd concept) is being the subject of different FP7 (LEOSWEEP) or ESA funded (IBSIOD, MOSAIC) projects. Low plume divergence is a central requisite for this transversal application of EP.

4.2.6 Candidate thrusters for VLEO-LEO applications

Based on the high-level requirements presented in the previous sections and on [RD4], one can identify the following candidate thrusters for LEO-VLEO applications:

Thruster type	Name	Power to Thrust Ratio (W/mN)	Thrust (mN)	Isp (s)	Power (W)	I tot. (MN.s)	TRL

HET	PPS®1350-G	17	40 -88	1650	700 - 1500	3.4	9
GIE	UK-10	30	25	3000	750	1	9
GIE	RIT10	30	15	3300	470	>1.5	9
GIE	T5	32	0.6-25	417-2926	<600	>3.5	9
HEMPT	HTM3050	20-35	10-70	2000-3500	<1450	>4	7
HET	HT-100	20	6-18	1000-1600	80-300	0.05	4
HET	HET-70	20	2-5	900-1000	50-100		4
QCT	QCT-200	20	10.2	1200	<200		4
HET	EPS-500	15	15-27	1200-1500	300 -700	0.5	3
HET	PPI	17	12	1200-1400	200		3
HET	HT-400	20	20-50	1100-1850	200-1000	0.2	3
HET	KLIMT	20	30		200-600		3
GIE	ALPHIE	22	0.1 – 4.5	1800 - 2500	< 200		3
GIE	PEGASES	30-40	5.5		<200		3

Table 4-6: candidate thrusters for LEO-VLEO applications

4.2.7 Drag compensation missions

LEO drag compensation missions are very specific and can require specific thruster development and its associated power supply.

Indeed, in order to compensate in real time the drag, in both force and torque domain, it is required to get really small thrust capability, with a capability to control this thrust with a good quantification. Pending the orbit altitude and solar activity, the need for thrust goes from 0.1 mN to 15 mN with typical quantification of 5% of the nominal thrust range.

This mission case is very specific and will probably deserve a dedicated development on a mission basis, as part of the payload. However, it can be important to develop building blocks.

For instance, after the GOCE and GRACE missions, the ESA Next Generation Gravity Mission (NGGM) will monitor the temporal variation of the Earth gravity field over a long time span and with a higher spatial resolution, and with improved temporal resolution. For this NGGM has to be based on the Low-Low Satellite –to-satellite tracking (SSTL) technique, for which the two satellites fly in a loose formation in which they are free to move under the action of the gravity field and, since the altitude of the satellites must be low (< 350 km or less) to increase the sensitivity to the higher harmonics of the Earth gravity field, the relative motion between the satellites will be perturbed by aerodynamic forces, i.e. air drag, too. The distance variation between their centres of mass (produced by both gravitational and non-gravitational forces together) is measured with high resolution by a distance metrology (laser interferometer instrument) of the order of 10^{-13} m/√Hz, namely 10 nm/√Hz for a typical inter-satellite distance of 100 km. The proper operation of the accelerometers to be installed on this spacecraft requires a “drag-free” control in order to reduce the level of the linear and angular non-gravitational accelerations of each satellite below a certain value. For such a mission, a scalable micro-propulsion system would be able to cope with all the micro-propulsion tasks in this mission: orbit altitude maintenance, drag compensation, fine attitude control, satellite formation control and laser beam control. This mission also requires a very low noise and high resolution system to be able to fulfil all the scientific requirements.

For NGGM, the following main requirements for the micro-propulsion system are given:

- Need of a system capable to operate with very high resolution (< 1 µN) in the micro- Newton range (from 50 µN to 100 µN) during science phase of the mission, and in the milli-Newton range (up to 2.5 mN) in the high thrust mode to perform the slew manoeuvres;
- Very low noise system;
- High total impulse system able to operate in orbit for more than 6.25 years continuously, i.e. an EP system with a lifetime of more than 55,000 hours of operation;

- Light and small system;
- Minimisation of the power consumption and the PPU dissipation for any operating regime;
- Maximisation of the specific power for minimizing the solar panel surface.

Those high level requirements for future long missions combining drag compensation requirements and scientific objectives (pointing accuracy and/or very fine positioning) have been confirmed by TAS-I [RD7]:

- Minimum thrust : < 1 µN
- Thrust range : up to 2 mN
- Resolution: < 0.1 µN
- Response time: better than 100 ms @ 300 mN
- Noise : < 10^{-4} N/sqrt(Hz) @ 10^{-4} Hz; 10^{-6} N/sqrt(Hz) @ 10 Hz
- Specific impulse: > 3000 s
- Firing time: continuous along the operative mission duration
- Cycle life : > 5.10^{8} actuations

and by Airbus DS for the micropropulsion system:

- Thrust range : 500 to 1500 µN
- Thrust modulation: from 0.1 % to 100%
- Resolution: ≤ 0.5 µN
- Response time: better than 100 ms @ 300 mN
- Noise : mission dependent
- Specific impulse: > 1000 s
- Firing time: continuous along the operative mission duration.

Based on these high-level requirements and on [RD4], one can identify the following candidate thrusters for this type of application:

Thruster type	Name	Power to Thrust Ratio (W/mN)	Thrust (mN)	Isp (s)	Power (W)	I tot. (MN.s)	TRL
FEEP	IFS-C	50-55	0.001-0.1	>5000		930h	6
FEEP	IFM-350 Nano	60-80	0.0003 – 2	>6000		150h	6
GIE	RIT-µX	30-100	0.010-2.5	> 300-3000	< 50	0.025	4
GIE	MiDGITS	42.5	0.010-2.5	400-2200	<90	0.040	4
FEEP	IFM-350	60-80	0.0003 – 2	>6000		2000 h	4
Colloid		20	2.5 µN/mm2	2500			4
HEMPT	µHEMPT		0.021 – 0.312	87-620	1-10		3
Colloid		22 @ 1000 s	0.23 mN/cm^2	1000 -5000			3
Colloid		30 @ 5000 s	0.48 mN/cm^2				3
FEEP	NanoFEEP	60-90	0.001 – 0.022	6000		15.10^{-6}	2
FEEP	IFM-3000		0.05-600	Up to 30000			1

Table 4-7: candidate thrusters for drag compensation applications

4.2.8 LEO to high orbits

One specific case which is at the edge of LEO missions and high orbit missions is the specific case of satellites that are injected in LEO orbit and have to reach high altitude orbit. This case has been analysed on GeoOcapi early phase, in order to reduce the launch cost, but can be extended to other scientific or even other missions.

The idea is to use a cheap launcher such as VEGA that will leave the payload in LEO orbit and to use electric propulsion to reach MEO, GEO, HEO or even L2 orbits.

The drivers of this type of mission is clearly the thrust (to limit the LEOP duration) and the Isp/specific thrust (to reduce the satellite mass). The needs are close to those of all electric Telecom satellites that are emerging, but would be dedicated to smaller satellites in the range 500/1500 kg wet mass to fit within VEGA performances in LEO for full/auxiliary payload.

After discussions with Arianespace, for instance, the following VEGA launcher performance that minimise the Delta-V to GEO is the following: 1500 kg satellite is separated on a 2450 km/450 km apogee/perigee orbit inclined at 5.4°. The Delta-V to reach geostationary orbit is then less than 4 km/s and the propulsion system has to be tuned accordingly.

In order to keep a LEOP duration within reasonable limits, 6 months have been considered which drives the thrust to 400 mN minimum. However, in the opposite to telecom missions that require high power on station, a scientific mission in GEO orbit will not need 10 kW solar arrays. For this study, a maximum of 5 kW solar array has been considered as a maximum. Higher levels of radiation due to several months through Van Allen belts are also a concern.

	Dutycycle	Power	Total Impulse	Thrust	Isp	cycling	dry mass
LEO to GEO	100%	5000W	6MNs	400mN	>2000 s	~ 2.000	80kg

Table 4-8: Example of EP requirements for LEO-GEO transfer using VEGA launcher

4.2.9 Nanosat missions

The range of satellite considered is this analysis is based on current nanosat used and forecast for next years. Most current nanosat use between 1U and 3U concepts but 6U and eventually 12U will very soon emerge. After 12U, one considers that one enters in microsatellite range already addressed in other chapters. As said earlier, the nanosat are not equipped with propulsion at the moment, but thanks to mass/size increase the missions should become more and more demanding, in particular in term of station keeping requirements. Therefore, one has here considered that Delta-V requirements would be comparable, except for the duration of operations that are designed to be much smaller (1 to 2 years maximum).

Due to cubesat concepts, and the standardisation of pods to launch them, the size/mass of the system will certainly be drivers for an electric propulsion system.

As a foreword, it is clear that propulsion of a nanosat will be useful only if the nanosat is equipped with 3 axis Attitude Orbit Control System (AOCS) allowing to provide stable direction to the propulsion and then allow for Delta-V in a given direction. At the moment, servitude (or platform/bus) of attitude controlled nanosat takes between 1U and 1.5U and the payload also between 1U and 1.5U. It is then hardly feasible to envisage a propulsion system for a 2U or 3U satellite on top of another mission. But it could be envisaged to demonstrate a propulsion system in orbit using the payload spacing.

The roadmap that could be envisaged would be an electric propulsion using between 1U and 2U for the whole subsystem. It could be demonstrated as a payload on a 3U satellite and used as main control system for 6U and 12U satellites.

Concerning power supply, the most constrained case would be 3U for demonstration or 6U for a full mission. The satellite gets between 24 W peak (3U) and 36 W peak (6U). It can be considered that 100% of this power can be used by electric propulsion in peak (taking power from battery) but keeping only 10% of this power in average (between 2 and 4 W average). By scalability, the minimum thrust should be typically $1/10^{th}$ of proposed one for micro/mini satellite and thus 0.1 to 0.3 mN.

The Delta-V needs are considered equivalent to those of micro satellites, and estimated to be around 80 to 100 m/s maximum. For the maximum mass (12U – 20 kg) and an Isp of 1000 s, this would correspond to 200 g of propellant, while an Isp of 500 s would need 410 g of propellant, which is still acceptable.

The following characteristics are proposed as target for nanosatellites:

- Dimensions of the subsystem: 10*10*15 cm
- Wet mass: 1.5 to 2 kg (Isp to be challenged in order to optimise dry mass vs wet mass)
- Max/Mean power: <20 W/3 W
- Thrust: >0.1 to 0.3 mN
- Cycling: 1 to 2 years with 1 manoeuvre per day: 1,000 on/off cycles.

	Dutycycle	Power	Total Impulse	Thrust	Isp	cycling	dry mass	Dimensions
nanosat	100%	2-4W	-	0.1mN	>500 s	~ 1.000	1kg	10x10x15cm
nanosat	10%	30W	-	1mN	>500 s	~ 1.000	1kg	10x10x15cm

Table 4-9: Main EP requirements for nanosatellites

The following requirements have been proposed by TAS-UK concerning the cubesat propulsion (using pulsed plasma thrusters – PPT) [RD7]:

- Volume : 10 x 10 x 10 cm
- Delta-V : 10 m/s
- Min impulse bit: 0.1 to 0.5 mNs
- Propulsion unit wet mass limit : <1 kg
- Total cubesat system mass : < 10 kg
- Number of thrusters: 1 to 4
- Power: < 5 W
- Cost goal: < 20 k€.

Based on these high-level requirements and on [RD4], one can identify the following candidate thrusters for this type of application:

Thruster type	Name	Power to Thrust Ratio (W/mN)	Thrust (mN)	Isp (s)	Power (W)	I tot. (MN.s)	TRL
PPT	PPTCUP (34 µN)	58	0.040	600	2	$44\text{-}60.10^{-6}$	8
Colloid		20	2.5 µN/mm2	2500			4
PPT	Nano PPT		0.09	600	5	133.10^{-6}	4
PPT	µPPT		0.005-0.030	800	2		4
HEMPT	µHEMPT		0.021 – 0.312	87-620	1-10		3
Colloid		22 @ 1000 s	0.23 mN/cm^2	1000 -5000			3
Colloid		30 @ 5000 s	0.48 mN/cm^2				3
PPT	L-µPPT			400-1200	2	> 500 Ns	3

Table 4-10: candidate thrusters for nanosat applications

5 MEO MISSIONS

The Galileo 2G programme, with the European Union (EU) as main customer/user, is targeting the possibility to increase the Galileo Payload capability without impacting the launch costs (and possibly reducing them). The need to increase the size of the Galileo payload (mass and power) is deriving from system needs, which are considered to become essential in a new scenario of Galileo, starting in 2020 and having its final configuration not earlier than 2030. Today, there are no margins in the Galileo platform to allow an increase in the payload capability, due to mass and power limitations induced by the adopted launch strategy (direct injection). Any increase in the satellite mass would lead to a negative impact on costs (from two to one satellite in Soyuz and from four to three satellites in Ariane-5).

In 2011-2012 a number of ESA internal studies were performed within ESA to assess the feasibility of increasing the Galileo payload capability without impacting the launch costs. These studies were conducted under the acronym of AGILE, Advanced Galileo Injection in Low earth-orbit using Electric propulsion, and constitute the phase Ø of the Galileo 2G programme.

The AGILE studies concluded that the increase in payload capability could be achieved by changing the launch injection strategy and by using EP to transfer the satellite from the injection orbit to the target operational orbit. EP would allow increasing the Galileo payload capability, making the Galileo platform compatible with any launcher of the Arianespace's launcher family and reducing the launch costs by increasing the number of satellites per launch, with the goal to launch:

- 4 or more satellites in Ariane 5 shared launch into standard GTO, leaving a co-passenger mass ≥ 3000 kg
- 3 or more satellites in Soyuz ST dedicated launch
- 1 satellite in Vega dedicated launch.

The AGILE studies have considered the use of several EP Subsystems from different European suppliers. In particular three EP technologies were assessed: GIE, HET, and HEMPT.

Currently, parallel Phase A satellite studies are running. The ultimate selection of the EP system will be based on the need to provide adequate thrust to achieve mission objectives, to maximise the specific impulse (in order to minimise the propellant mass) and to reduce the subsystem power demand. In addition, cost and industrialisation status will be a strong driver for the selection.

Moreover the chosen orbit for the spacecraft (MEO) implies a much more constraining requirement on the cycling of the EP systems than it is today for GEO spacecraft. Indeed many more eclipses will be encountered, increasing the number of on-off cycles on the thrusters and its components; and this will have to be demonstrated during the qualification phase of the EP system.

6 GEO MISSIONS

6.1 Telecommunication

The GEO orbit is mainly used for satellites dedicated to telecommunication.

The trend in GEO telecom satellites has resulted in a considerable increase of electric power to satisfy the payload needs, an increase in platform size to accommodate larger payload and longer mission duration up to 15 years. All the major Satcom manufacturers (i.e. Boeing, TAS, Space Systems Loral and Airbus DS) have already implemented EP systems for North-South Station Keeping (NSSK). It is a first step for the penetration of this technology in the telecom market.

In view of the increasing mass and mission duration of new GEO platforms (15 years and more), the requirements on the Total Impulse to be provided by the EP systems are constantly increasing. Qualified thrusters might not be able in the near future to fulfil the Total Impulse specification of future commercial missions. Faced with the US competition, it is reasonable to expect that the market will over time demand that European platform providers match the worldwide competition, by offering orbit topping on its large platforms and developing small platforms with 'all-electric' orbit raising and station keeping functions.

Table 6-1 presents some of the present and near-future European GEO telecommunication platforms using EP. Today HETs are the EP thrusters embarked on European telecommunication platforms to perform the desired manoeuvres.

Platform	Supplier	Status	Platform Mass Range (tons)	Platform Power Range (kW)	EP Function
Eurostar E3000	Airbus DS	Flight Proven	4.5 – 6.0	9 - 16	NSSK
SpaceBus NEO	TAS	Under Development			NSSK, Orbit Raising
AlphaBus	Airbus DS / TAS	Flight proven (Alphasat)	6.0 – 6.5	12 - 18	NSSK
AlphaBus Extension	Airbus DS / TAS	Under Development	<8.4	12-22	NSSK, Orbit Topping
SmallGEO	OHB	Under integration	3.2	6.5	NSSK, EWSK, Momentum Management, repositioning, EOL disposal
NEOSAT	Airbus DS/ TAS	Under Development			Orbit raising, SK
Electra	OHB	Under Development	3	10	Orbit raising, NSSK, EWSK, Momentum Management, repositioning, EOL disposal

Table 6-1: Summary of European Geostationary Telecommunication Platforms using Electric Propulsion

The European Large Platform Development AlphaBus uses HET EP for NSSK. Although orbit topping is feasible, it would provide only a very small fraction (50 to 110 m/s) of the needed transfer velocity increment (1.5 km/s). This option may be attractive on the smallest of the AlphaBus range when the spacecraft mass budget is marginal for one of the smaller launcher. For the nominal AlphaBus range, work has started on incorporating an Electric Orbit Raising (EOR) capability to improve competitiveness and capability. These improvements are based on the extension of the current HET performance to enable orbit raising at a higher power operating point. Both Airbus DS and TAS have recently embarked on upgrade programmes to extend the capabilities of their existing Eurostar and SpaceBus platforms. In the case of Airbus DS, they have taken the decision to embark electric propulsion on their smaller platform configurations. This activity is targeted at achieving compatibility with emerging low-cost launchers, with the electric propulsion system only used to meet NSSK requirements and at squeezing more performance from the existing platform designs.

OHB's SmallGEO platform has been designed from the beginning to utilise EP system for all orbital manoeuvres thus getting rid of the chemical propulsion utilization while in GEO orbit. With eight fixed SPT-100 thrusters mounted in pairs on the East-West-North-South edges of the spacecraft, the 15 years NS and EW station keeping, the momentum control, the repositioning, and the EOL disposal manoeuvres are provided with this EP system. Auxiliary propulsion with Xenon cold gas is also implemented to provide the initial detumbling after separation and the emergency safe mode attitude control functions.

Moreover, the European operator Eutelsat will have 7 further satellites to be launched by 2017 whose 3 will be full EP satellites: Eutelsat 115 West B, Eutelsat 117 West B and Eutelsat 172B. The first two will be based on the Boeing 702 SP full EP platform (US, based on 4 XIPS-25) and will have an expected orbit lifetime of more than 15 years.
Eutelsat 115 West B is planned for launch in 2015 in SSTO with Falcon 9 L1 on a dual launch (see Figure 6-1). EOR time is expected to be between 7 and 9 months. The mass at launch is 2200 kg with an 8 kW payload. Eutelsat 117 West B is the same as the previous one and is planned for launch at the end of 2015 also in a stacked configuration.

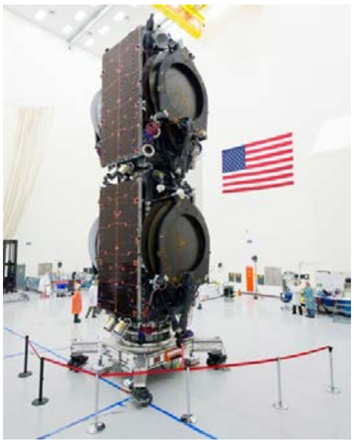


Figure 6-1: Stacked configuration of Boeing 702 SP satellites (www.gizmag.com/boeing-conjointed-satellites/34755/)

On the other side Eutelsat 172 B will be based on the Eurostar 3000 European platform (with HETs) and is planned for launch on Ariane 5 lower position in GTO in 2017. The mass at launch will be about 3500 kg for a maximum power available of 13 kW. It will perform the EOR in about 4 months. Its expected lifetime is more than 15 years in orbit. On the three spacecraft, the EP system will be used to perform the EOR, NSSK/EWSK, momentum damping, station relocation and final re-orbitation.

As well, the operator SES has recently ordered from Airbus DS a full EP platform: SES-12 will be the most powerful European satellite with full electric orbit transfer with a payload mass of 1500 kg, a payload power of 15 kW, and launch mass of 5.3 tons. It is planned to be launched in 2017.

In the last two years ESA has endorsed the approach to propose in telecom satellite market, platforms with electric propulsion thrusters (5 kW or clusters of electric thrusters) to perform orbit topping and station keeping functions: NEOSAT and Electra. European producers of electric thrusters have already initiated development programmes to qualify in the short-term high power version of existing thrusters (up to 5 kW) to be embarked on NEOSAT and Electra. In the full EP platform, Electra (see Table 6-2), from OHB, the EP system will be able to perform all the manoeuvres: EOR, NSSK, EWSK, momentum damping, station relocation and final re-orbitation.

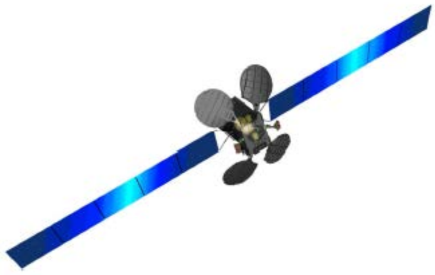
Electra Mission – Small GEO Flex
- < 3 ton launch mass
- About 10 kW of spacecraft power at BOL
- 600 kg communication payload
- 6 months for EOR
- 15 years design lifetime
- Thrusters on boom
- Orbit raising mode
    o > 9 kW total for orbit raising
    o Thrust: 330 – 500 mN
    o Isp: 1700 – 3500 s
- Station keeping and wheel unloading
    o < 3 kW
	

Table 6-2: Characteristics of Electra platform (OHB)

It is not expected that the maximum power available on the telecom spacecraft will be increased higher than 18 kW by 2025. To be able to perform the orbit raising in future GEO telecom platforms, the transfer velocity increment needed is higher than 1.5 km/s and can even reach about 3.5 km/s depending on the launcher performances and on the transfer strategy. For an orbit lifetime requirement of at least 15 years, and depending on the inclination of the insertion orbit (GTO) by the launchers the Total Impulse to be delivered by the EP system should be between 5.4 MNs (with Proton) and up to 16 MNs (with Ariane 5).
The use of EP for EOR from GTO to GEO has also an impact on the launcher performances. With the apparition of the Falcon 9 launcher the expectations in launcher performance have been redefined. Indeed EP offers EOR capabilities giving the satellite a better raising efficiency than the Launcher: the usual 1.5 km/s of Delta-V from GTO could be revisited for a higher Delta-V from the satellite. Falcon 9 can stack or launch very heavy satellites, trend which could be followed by future European launchers.

6.1.1 Orbit topping and thrust-to-power ratio

Future EP systems shall strive to enable all-electric satellites and to avoid the complication induced by an additional chemical propulsion system. Orbit topping may, however, remain to be an option to the extent that the optimum configuration depends on a number of parameters and will be assessed on a case by case basis.
As a consequence, the thrust of EP systems shall be high enough to allow the transfer of a satellite into GSO within reasonable time. In this context the power-to-thrust ratio is becoming important. The satellite to be raised to GSO can provide a certain amount of electric power to the EP system, depending on its architecture and the telecommunication payload requirements, for which it is designed. Assuming that EP systems could be scaled such as to fully exploit the offered electric power, then the transfer time will depend on the power-to-thrust ratio (specific power) of the EP system. This is illustrated in Figure 6-2 below. The solid and the dashed lines correspond to the power efficiency of the EP system with the bold solid line representing the theoretical limit of 100%. The optimum EP system would have a high thrust-to-power ratio, so that transfer time is relatively short. A very high Isp is not strictly required for orbit raising as long as it is significantly higher than for chemical propulsion to benefit from the mass savings. In Figure 6-3, the Isp is plotted against available satellite power. For orbit raising the range of useful systems is defined by minimal and maximum available power (2 kW - 18 kW) vs. Isp in the range 1100 s – 4200 s. The Isp range reflects the Isp targets of Figure 6-3.

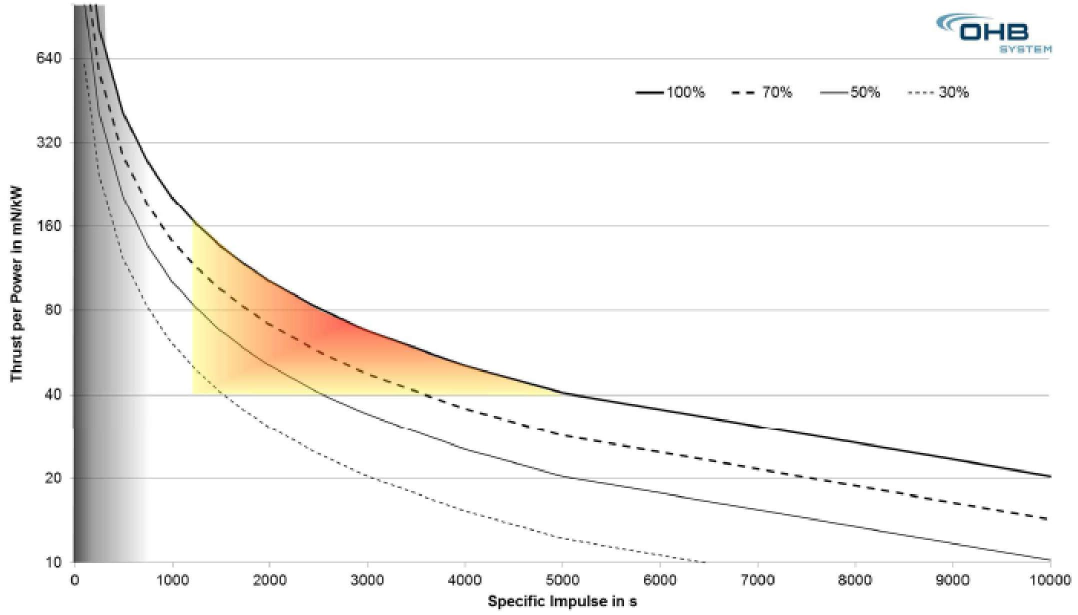


Figure 6-2: Thrust per power requirements for EOR for GEO satellites [RD7, Credit: OHB]

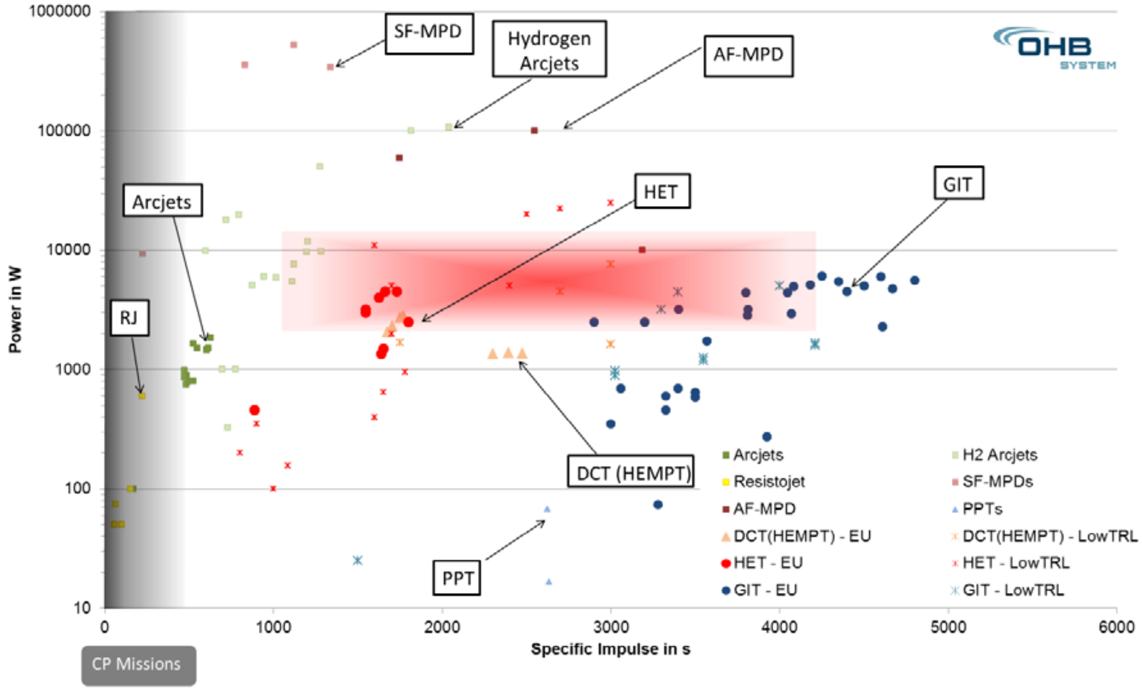


Figure 6-3: Orbit raising of geostationary satellites with current EP technologies [RD7, Credit: OHB]

In addition, the level of on-board autonomy shall permit the spacecraft to be autonomous for long periods during the orbit transfer phase and during all phases of its mission. This is a necessary condition for long EOR in which the visibility of ground stations for telecommand (TC) is not always visible.

6.1.2 Direct injection, station keeping only

A different scenario applies for station keeping applications, where Isp is more important than thrust, because the time for the operation is less critical. For the selection of thruster technology the scenery can change completely, if e.g. the launcher allows direct injection. This depends on the business model of the satellite operator, who may prefer not to use the mass savings due to EP for additional payload, but rather for cost saving by a smaller launcher or an early start of operation by using a launcher with direct injection capability. In this case a thruster technology, which offers highest Isp at a modest thrust-to-power ratio would offer the highest mass savings and thus the easiest access to the GSO position.
Figure 6-4 is again a plot of thrust–to-power ratio over Isp. In the station keeping case, the target range for power-to-thrust ratios is extended, and also Isp values in a very broad range from nearly 1000 s to well beyond 5000 s may be useful. This target range is highlighted in green in the same figure. For station keeping Isp is more important than thrust.

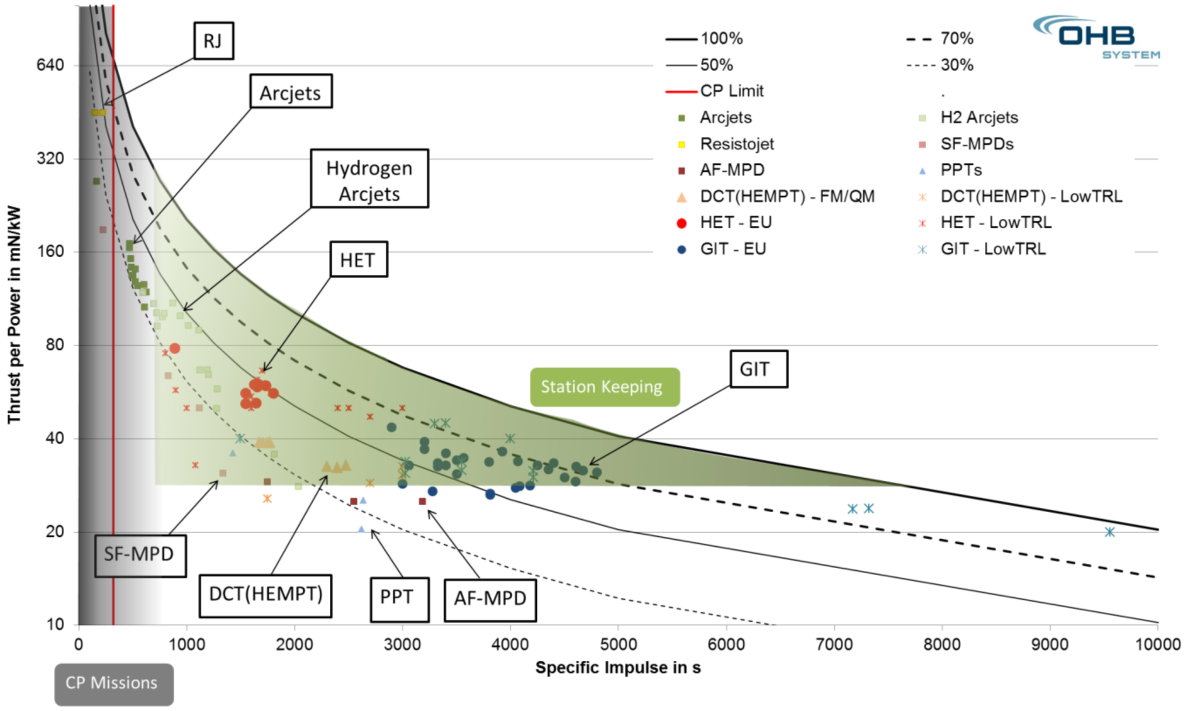


Figure 6-4: Desired Isp and thrust-to-power ratio for station keeping manoeuvres [RD7, Credit: OHB]

6.1.3 Cost and complexity

For the final decision of a satellite prime, which EP system to implement for orbit raising and/or station keeping, additional factors are taken into account. As stated already, overall cost remain to be the most important factor. This is not just the price for the thruster, but of the thruster orientation mechanisms and the power processing unit normally are the most expensive part of the EP system. Therefore the development of power processing units towards higher power and lower cost has to be considered at the same time. Both trends are under development and 5 kW PPUs are within reach, but industrialization and cost reduction for PPUs has just started and TRL is still low.

It has to be noted that different thruster technologies may have different requirements. For example, GIE require several high voltages, whereas HEMP thrusters only need one. For HETs the high voltage must be switched between thrusters. Orbit raising and station keeping with the same thruster may only be meaningful with a thruster working at different operational points, i.e dual mode operation. In that case the PPU must be able to switch between these different voltages and needs to exhibit high current capabilities.

When selecting the propulsion system for a future telecom satellite, additional consideration will be given to system complexity and cost of integration. Indeed, both due to development and hardware production and qualification costs, generally EP systems are more expensive than conventional chemical propulsion systems. A large use of EP systems for commercial applications will generate an economy of scale resulting in a potential reduction of EP platforms costs.

Moreover, reliability and risk mitigation shall be taken into account: as always, technical risks for implementation of EP on the spacecraft need to be minimized. The reliability of proposed EP platforms shall be equal or better than full chemical platforms.

As well, the impact at system level of the EOR environment of the EP thrusters shall be carefully assessed in order to confirm the acceptability of the proposed design for the EOR and the in-orbit operations including any interference with the payload.

It also has to be noted that, in general for an operator, the shorter the transfer time and so the time to orbit, the better. However, the time for EOR can be increased if the launch costs are reduced. It depends on the type of mission: either to increase the payload mass, either to decrease the mass at launch. EOR does not provide any incomes to the operator, on the contrary, due to the increased transfer time, additional investment costs have to be considered. This way the large benefit are the cost savings for the launch vehicle and a possibly increased payload mass. The solution to go for a dual launch is also a good option to consider longer EOR time (and as a consequence the adoption of GIE or HEMPT technologies instead of the currently considered HETs on European platforms). Therefore, compatibility to multiple launch vehicles will be preferred: this potentially represents an essential requirement from the operator point of view. A new design that is constrained to one launch vehicle is not desirable as it will limit the possibility to obtain competitive prices from different launch services and introduce exposure in the case of a launch failure or delays.

6.2 GEO-like for Earth-Observation and high elliptical orbit

Some telecommunication customers now focus on orbits covering the poles and consider reaching an highly elliptical orbit. This will redefine the way EOR is performed for GEO as well as on-station manoeuvres and as a consequence the architecture, hardware and expectable performance of the EP system and satellites.

Recently, a consortium between the University of Strathclyde Engineering, ESA, the University of Reading (UK) and Airbus DS, has performed a study [RD8] to provide geostationary-equivalent observation of the polar regions, along with the mission and system design, and technology required to enable such a concept. The main objectives of such a geostationary-like polar platform would be the cryosphere, geophysical data products and numerical weather prediction in the region poleward of 55 degrees latitude (see Figure 6-5).

A newly developed highly-elliptical orbit was examined to provide the required GEO-like observations. It uses the acceleration from an EP system to null the dominant orbit perturbation for the non-spherical nature of the Earth, allowing free selection of the orbit inclination.

The most beneficial concept required two spacecraft to give continuous composite coverage to 55 degrees latitude, and four spacecraft for continuous single-image coverage for a mission duration of 8.5 years (to match the Meteosat Third Generation platforms). The most likely launch case was identified as two spacecraft on an Ariane 5 launcher to enable the composite image concept, with an optimal additional launch giving complete single-image coverage.

Finally, the use of EP for orbit topping would allow significantly increasing the payload mass available due to the selected launch vehicle, ranging from 340- 849 kg, or decreasing the launch costs for a minimum payload mass.

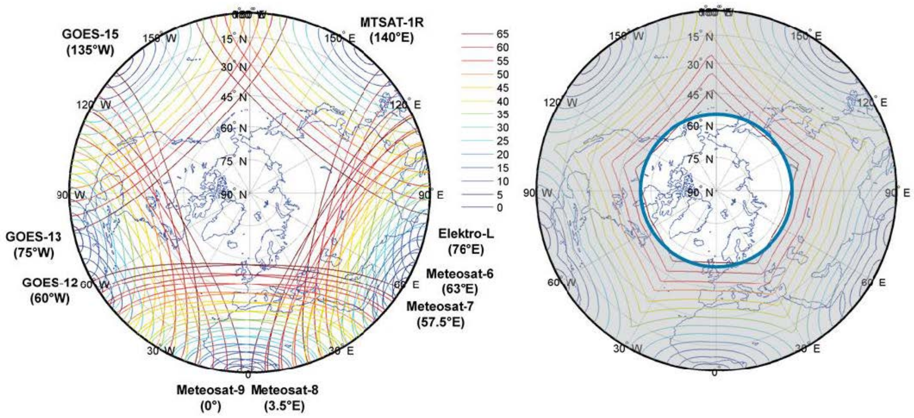


Figure 6-5: Polar stereographic plot of view zenith angle contours (in degrees) of nine GEO spacecraft with region poleward of 55 degrees highlighted on right [RD8]

For such type of missions which would use the GEO for Earth Observation, the same requirements as for telecommunication satellites would apply, in terms of station keeping and orbit raising, even if, as the available power on board is less, the transfer time could be more relaxed; the advantage of EOR needs to be analysed on a case by case basis.

And finally future Earth Observation missions in that orbit could extend to high resolution observation; in this case EP would be the enabling technology because micro-propulsion would allow achieving a quick repointing and ultra-stable pointing during image acquisition. The requirements for such system would be in the same range of these of LEO formation flying (NGGM for instance).

6.3 Technical requirements

Following the trends and requirements for future GEO telecommunication satellites presented above, the future EP system shall:

- be able to use the power available on board the spacecraft during the orbit transfer to reduce the transfer time
- be able to comply with the increase of Total impulse (> 15 MNs), i.e. meet the increase lifetime requirements (between 10,000 and 20,000 hours of operation) and comply with the xenon throughput increase
- be able to be optimised for EOR and SK manoeuvres:
    - high specific impulse
    - dual mode operation
    - improved thrust-to-power ratio
    - thrust orientation mechanisms (boom or not)
- be more efficient in terms of power-to-thrust ratio and for the entire system
- be scalable if needed (even if the number of thrusters will be limited by the thruster orientation mechanisms)
- having the flexibility and capability to apply thrust vector steering in order to save the costs for thruster pointing mechanisms
- be low cost to be competitive with American and Russian providers
- be available and traceable to avoid any problem of embargo (for US equipment for instance) and issues with ITAR restrictions
- have common building blocks in case of use for station keeping only to optimise the return on investment.

7 SPACE TRANSPORTATION

7.1 Mission requirements

Space transportation vehicles can take great advantage of the growing maturation of electric propulsion systems and increasing capabilities (power rising) of such propulsion devices.

New applications and missions become possible, bringing the potential for new services in space, for the benefit of European industry.

These applications are being gradually evaluated with a more and more detailed level of analysis, and it is today possible to describe the potential applications under the three following directions of evolution.

Electric propulsion can:

- decrease the cost of access to space
- provide new in orbit services
- increase launchers' capacities and performances

These directions of evolution are necessary for European Space industry in order to preserve an affordable and autonomous European access to space and to develop its capabilities for in orbit services.

7.1.1 Electric propulsion to decrease access to space cost

The continuous quest for reducing launchers' cost reaches the limits of incremental innovation with maximum standardization of subsystems (engines, tanks, ...), optimization of organization (reduction of interfaces, simplifications of processes and management for design / fabrication / assembly / control / launch).

Electric propulsion is a disruptive technology for space transportation, which allows conceptual evolutions.

A first possibility to be studied is the "re-usable upper stage" concept because upper stage is one of the most expensive parts of a launcher. As it is very complex and costly to re-use an upper stage after its return back on ground, such as the Space Shuttle for example, the idea is to keep it in orbit and use it several times.

In such a concept, a simplified launcher would inject payloads in LEO (where the requested performance is limited), then the "re-usable upper stage" (also called space tug) would transfer the payload from LEO to the final destination, and come back.

Using electric propulsion on this space tug would drastically cut the propellant consumption with a beyond 3 factor from LEO to final orbit.

As a consequence, the launcher has to provide less energy to the payload leading to a reduced need of performance and cost. And the space tug, because of electric propulsion use, can perform the final phase of the mission without frequent refueling. The more powerful the electric propulsion device is the faster the final orbit will be reached.

7.1.2 Electric propulsion to provide new in orbit services

The past and current periods show many in progress activities about Orbital Transfer Vehicles (OTV). Thus several studies have been already performed with propelled vehicles by electric propulsion systems. For instance:

- Chaser for ADR (Active Debris Removal). Different studies have been performed through, e.g. ESA, NASA, or CNES programmatic frames.

The figure below (Figure 7-1) shows an example of a concept for an ADR chaser that is propelled by an electric propulsion system and which delivers propulsion kits to the different debris it visits. This study considers PPS®5000 as propulsion engine. The principle is to accelerate the path between two debris.

Figure 7-1: Chaser for ADR (Airbus DS concept)

Another example is the NASA SPECTER study for ENVISAT relocation which suggests usage of four PPS®1350 on gimbals to move the satellite up to a storage orbit (2100 km) in 9 months.

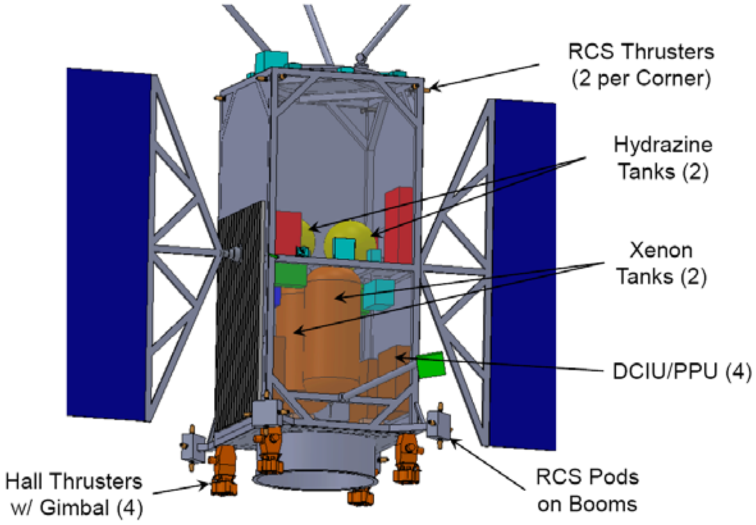


Figure 7-2: NASA concept to move ENVISAT

- Asteroid Retrieval Mission (ARM) from NASA suggests to move an asteroid towards a cis-lunar orbit with 50 kW engines and 3000 s Isp.

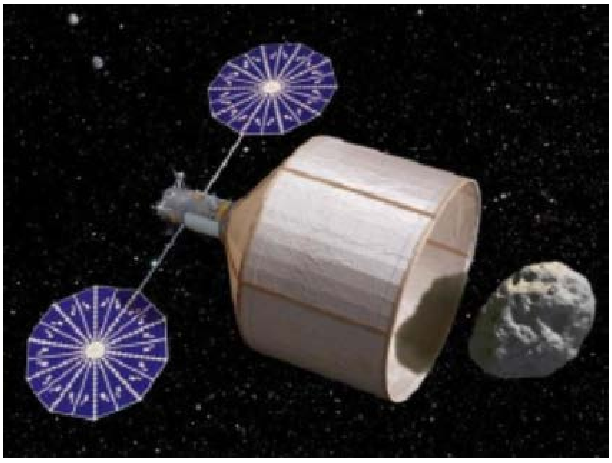


Figure 7-3: NASA concept for ARM

- Other applicative missions with electric propulsion were also studied as tugs for interorbital servicing. With international effort about electric propulsion, commercial applications are now also examined with details for maintenance or refuelling of orbital objects. Vehicles for cargo supplying beyond LEO are also envisaged to provide heavy loads to far destinations as Mars or its moons. The idea of possible standard service module for interplanetary probes is also examined

  As a matter of example a concept of generic service module for interplanetary mission has been studied with five PPS®5000 operating at 7 kW.

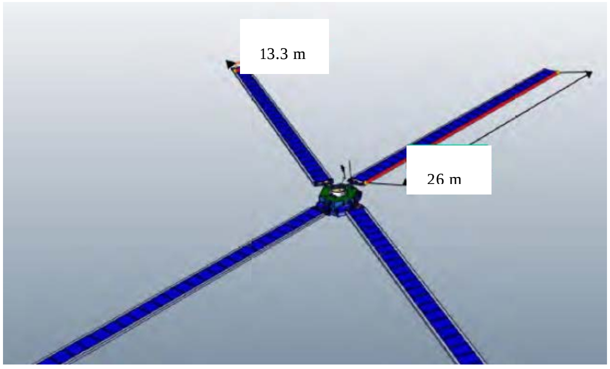


Figure 7-4: TAS concept of a 40 kW generic service module

These missions are only examples and could be enlarged to a wide scope of both institutional and commercial missions that need moving masses in space and providing opportunities for inspection and/or maintenance of orbital systems.

Electric propulsion is particularly well adapted to these types of missions because of its limited propellant consumption and the capability to prepare the transfer with management of their duration. Of course higher thrust would be of interest to reduce the transfer time duration.

7.1.3 Electric kick stages

Several studies have already been performed to increase the performances of launchers by adding an electrical kick stage to a conventional architecture chemically propelled.

As a matter of example Italian studies evaluated capabilities of the Vega launcher with an electrical kick stage (see Figure 7-5). This study (IEPC – 2013 – 279), presented at the International Electric Propulsion Conference in October 2013, relies on a cluster of five HET engines (4 active and 1 in cold redundancy), each of them delivering a 0.3 N thrust with a 5 kW power requirement. Such a concept would be able to lift a mid-size spacecraft (1000 – 2000 kg) from LEO to GEO.

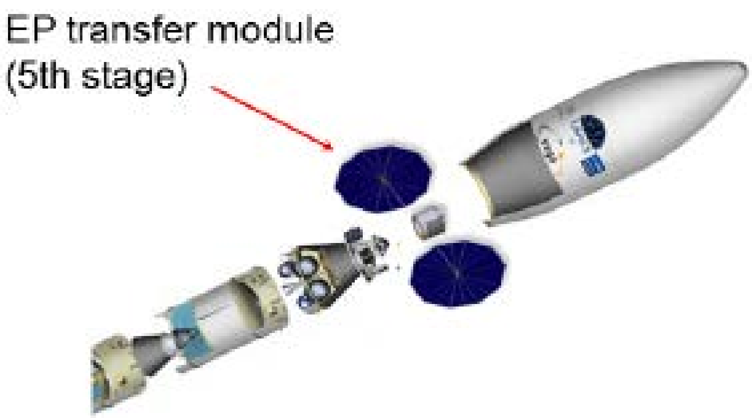


Figure 7-5: Vega Launcher with an electrical kick stage

Another possibility has been studied by CNES for implementation in Ariane launchers (EASE concept, Figure 7-6) and provides a significant increase of performance that can be appreciated for liberation missions for instance.

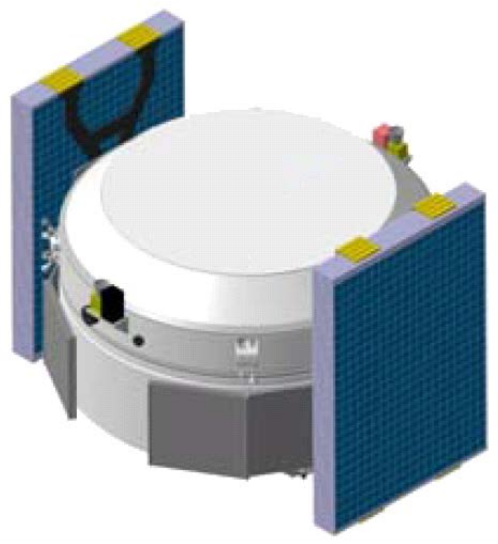


Figure 7-6: CNES EASE concept for kick stage

Because of unavailability of higher thrust, the first study only considered cluster of PPS®5000 engines (5 kW thrusters).

Globally speaking these architectures clearly offer to launchers significant gains of performance for a limited cost. This gain is especially important when the thrust is increased. It is the reason why such potential applications are looking for more powerful electric thrusters to limit the number of engines to be assembled in a cluster.

7.1.3.1 Towards high power electric propulsion

Some of these new missions will require important electric power. Today photovoltaic panels are very good products to generate some tens (satellites' scale) or hundreds of kW (Space infrastructures): technology is mature and flight-proven with activities in progress to still improve the performances. Nevertheless this technology is limited by:

- The requested surface to generate great amount of electrical power
- The decreasing efficiency with the increasing distance to the Sun is penalizing
- The decreasing efficiency with the duration because of UV damage
- The actual power conversion technology (PPU).

High power electric propulsion (which power source can be either solar or nuclear) can be a game changer for robotic exploration missions, allowing higher payload mass ratios and faster missions than what could be done with chemical propulsion.

These are the reasons why activities are performed in order to identify, study and mature alternative means to generate great amount of electrical power in space (some hundreds of kW and more). Among these alternatives, space electronuclear generator is interesting and requires international cooperation to share the effort.

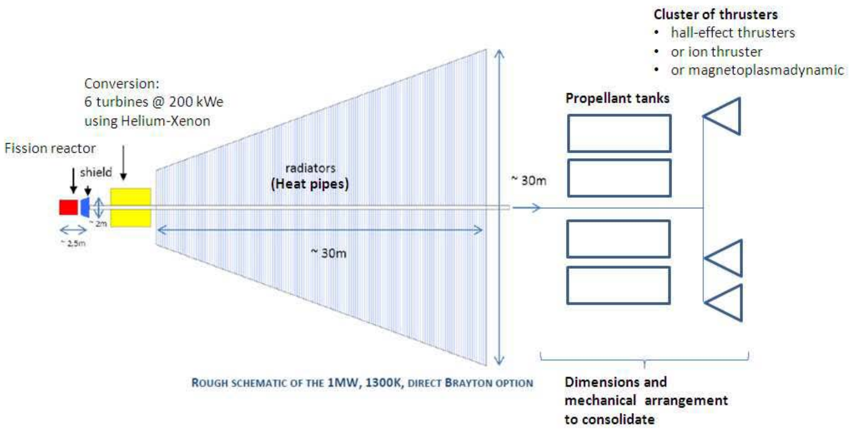


Figure 7-7: Conceptual architecture of a 1 MW electronuclear space generator

Achievable missions depend on the power available. Performed activities in the frame of DiPoP (considering 25 kW and MW thruster capabilities) and Megahit FP7 projects identified potential applications with nuclear electric propulsion like:

- robotic missions towards the giant planets with more potentialities of performances than the reachable one with conventional technologies
- mission for asteroid deflection by sending required payloads to destination

- logistic mission to send heavy cargo payload to support crewed missions around Moon or Mars.

7.1.4 Conclusions

This very quick overview demonstrates an important potential for future missions. The already promising results would be greatly improved if more powerful engines was available. For low cost access to space, interorbital missions and launchers' kick stages the main requirement is generally to increase the available thrust. For interplanetary probes the search clearly focuses on the maximization of the Isp.

With the FP7 HIPER project background, which allowed testing a 20 kW HET electric thruster (1 N of thrust with 2400 s Isp), Europe has gained capabilities to better address the first class of missions.

It is also of interest to consider the possible merging of the different considered vehicles around a common basis which would be customized for each mission. Such a family concept would allow to change cost data for access to space and in space transportation for a wide scope of missions. Finally, the set of potential applications will be more developed when cost reduction about electric propulsion will be effective.

7.2 Technical requirements

As previously mentioned the main concern about electric propulsion for space transportation vehicles has to be focused on high thrust for a limited cost. Naturally, high Isp is of interest but because of requested movements in the vicinity of Earth (kick stages, OTV in LEO, ...) or heavy masses to be transported beyond LEO, the first requirement is the increase of the thrust.

7.2.1 Cost reduction

In parallel with efforts to reduce the cost of today existing technologies, four possible tracks are suggested:

- High voltage solar arrays

  Efforts have to be focused on the solar arrays and the electric interfaces which are the more costly part of the global electric propulsion device.

  One possible way to reduce the PPU cost (which remains important) is to use the same electric voltage between the delivered power by the solar arrays and the delivered power to the electric thruster. Consequently research for high voltage solar arrays would be of interest as well as investigation of new spacecraft power concepts as solar power, EP power and platform power need to be managed in a different way.

- Low cost solar arrays

  New concepts of solar arrays have to be studied, for instance investigations about organic solar arrays (OPV) could be performed to evaluate opportunity to consider this technology. Potential benefits would be an easier unfolding sequence with an allowed larger surface to be implemented under the fairings of the launchers. Though the today efficiency is limited the low mass may counterbalance this disadvantage. Evaluations of electrical performances and behaviour in space of organic material would also allow to status about the possible interest of such a concept. Of course other low cost concepts could be considered.

- Cluster of thrusters instead of high power electric thruster

  For high performance demanding missions a usual approach is to envisage more powerful thruster. Of course it is necessary to increase the power of the thruster because the today 5 – 7 kW range considered for orbital systems is not powerful enough for applications at space transportation level. Nevertheless cluster of thrusters offer a more reliable configuration than a single one because of tolerance to failure. In addition clustered configurations exhibit significantly reduced qualification costs and an increased flexibility in the choice of testing facility. Hence it is of interest to gain more experimental data about cluster of thrusters with interactions of plume and possible edge effects. As a consequence a combined approach with high power thrusters (for instance 20 kW) used in a cluster might be considered to progress on such configurations.

- Alternative propellants

Xenon is today generally used for electric propulsion. More than its scarcity its production cost (because of the necessary process) is a potential brake to a more expendable usage. It is the reason why alternatives have to be looked for as krypton or argon with evaluation of their impact on the global performances of the entire EP system. Moreover residual and atmospheric gases could also be used as propellant for the EP system if the efficiency and performance of the system is not decreased too much.

7.2.2 Increase of performances

In parallel with scalability studies of existing thruster, this quest unavoidably needs an increasing life duration. This also leads to the need for very high Isp (10,000 to 15,000 s) systems to reduce propellant consumption and therefore the need to develop a high thrust extremely high Isp system. Other tracks are also to be investigated as the technological ways to increase the speed of the propellant such as to increase the efficiency of the thruster. The storage phase (gaseous, super critical, ...) might also be of interest to be studied in order to limit the volume and the mass of the tanks for demanding missions.

Based on the high-level requirements presented in the previous sections and on [RD4], one can identify the following candidate thrusters for space transportation applications:

Thruster type	Name	Power to Thrust Ratio (W/mN)	Thrust (mN)	Isp (s)	Power (W)	I tot. (MN.s)	TRL
HET	PPS®5000	15-20	90-400	1400-2000	2000-7000	11.6	6
HET	PPS-20k		300-1000	1200-2500	5000-20000	70	4
HET	HT-5k	22	150-350	1700-2800	1650-7500		3
HEMPT	HTM 30250	20-35	140-320	2000-3500	4 k-10 k (anode power)	>20	3
MPD	SF-MPD		10 k-70 k	10000	300 kW – 5 MW		3
MPD	SF-MPD with heated cathode		20 k – 150 k	2000 – 4000	1 MW – 5 MW		3
MPD	HPT MPD		Up to 14 k	Up to 2500	100-800 kW		3
MPD	AF-MPDT		1000-20000	1500 – 3000	50-250 kW		3
MPD	AF MPD ZT1	40	250	3000	10000		3
MPD	AFMPD ZT2	40	2500	3000	100000		3
MPD	ZT and DT thrusters	19	26000		500000		3
GIE	DS3G/DS4G	32 -63	200-400	5000-10000	6-25 kW		2

Table 7-1: candidate thrusters for space transportation

8 INTERPLANETARY AND SCIENCE MISSIONS

8.1 Exploration & interplanetary missions

The use of EP for orbital transfer between bodies with the solar system is now established, with its used on Deep Space 1 (NASA), Smart 1 (ESA), Hayabusa Explorer (JAXA) and Dawn Programmes. Several studies were performed at ESA, NASA and JAXA regarding concepts for exploration and interplanetary missions, to a planet or an asteroid, trying to assess the possibility to use EP mainly to perform the transfer such as to increase the dry mass at the arrival at those bodies. Several examples of such type of European missions will be presented in the next paragraphs.

8.1.1 BepiColombo

The BepiColombo mission is an example. It is the Europe's first mission to Mercury that will set off in 2016 on a journey to the smallest and least explored terrestrial planet of the Solar System. When it arrives at Mercury in 2024, it will endure temperatures in excess of 350 °C and gather data during its 1 year nominal mission, with a possible 1-year extension. This mission comprises two spacecraft: the Mercury Planet Orbiter (MPO, provided by ESA) and the Mercury Magnetospheric Orbiter (MMO, provided by JAXA). Those two orbiters will be carried to their destination by the Mercury Transfer Module (MTM) which includes an Electric Propulsion system based on the gridded ion engine technology to perform the 7.5 years journey.

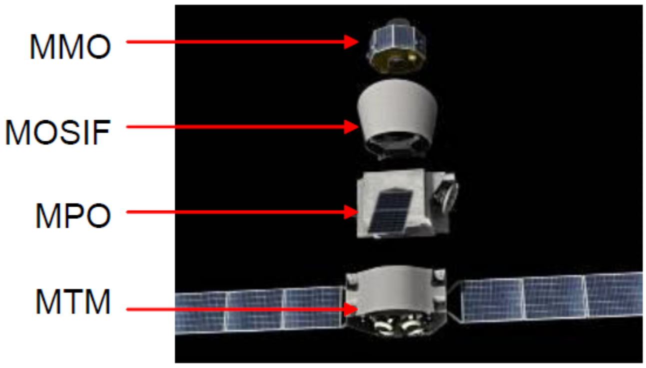


Figure 8-1: BepiColombo spacecraft and modules

To enable this mission, a specific EP system has to be developed able to comply with the following main requirements [RD6]:

- The EP system (composed of 1 thruster or a cluster of thrusters) shall be capable of being throttled over a thrust range between 120 and 290 mN at any point in the accumulation of the total impulse requirement;
- The total impulse to be delivered by the EP system shall be 22.5 MNs.
- Thrust vector stability of one thruster shall be <1.0° thrust vector deviation (defined as the maximum vector movement w.r.t. the initial thrust vector, i.e. the thrust vector measured immediately after the ion beam is enabled)
- The thruster shall not generate any torque around the thrust axis larger than 0.0435 mNm at 145 mN operation, and 0.0225 mNm at 75 mN operation;
- In case and during simultaneous twin thruster firing operations, the thrust vectors of the two operating thrusters shall be nominally parallel, with each thrust vector varying by up to +/-2.5 degrees around the nominal overall thrust vector centreline. Any interactions between the two operating thrusters shall not cause deviations from values identified in this specification.

- It shall be possible to operate the thrusters singly or in any pair combination compatible with the available power level;
- Each neutraliser shall be able to deliver independently the necessary neutralisation current corresponding to a single thruster operating at the maximum thrust level, plus any additional spacecraft/solar array coupling currents;
- The specific impulse for each thruster + FCU (and corresponding power supplies as appropriate) shall be greater than the limits expressed by the following polynomial (worst case, including all margins) across the specified thrust range:

$$I_{sp} \geq a_2 \times Thrust^2\ (mN) + a_1 \times Thrust\ (mN) + a_0$$

$$The\ corresponding\ coefficients\ are:$$

		a_0	a_1	a_2
Nominal	BOL	3263.8	14.962	-0.04848
	EOL	3091.6	14.908	-0.04851
Worst Case	BOL	3019.8	16.464	-0.05378
	EOL	2856.6	16.43	-0.05436

The above requirements shall include all flows, i.e. including cathode, neutraliser and main flows.

- The power required by each thruster + FCU, in conjunction with its power supplies (at the PPU input, assuming worst case and including all margins), shall be less the limits expressed by the following polynomial:

$$P \leq a_2 \times Thrust^2\ (mN) + a_1 \times Thrust\ (mN) + a_0$$

$$The\ corresponding\ coefficients\ are:$$

		a_0	a_1	a_2
BOL	Nominal (local thruster)	413.2	31.895	0.0054
	Switched (remote thruster)	428.8	31.968	0.004
EOL	Nominal (local thruster)	431.5	32.362	0.00594
	Switched (remote thruster)	447.6	32.434	0.00458

- The power dissipation from a single PPU when operating a single thruster and FCU (assuming worst case and including all margins), shall be less than the limits expressed by the following polynomial:

$$P \leq a_2 \times Thrust^2\ (mN) + a_1 \times Thrust\ (mN) + a_0$$

$$The\ corresponding\ coefficients\ are:$$

		a_0	a_1	a_2
BOL	Nominal (local thruster)	114.475	0.593	0.0092
	Switched (remote thruster)	127.022	0.58	0.0072
EOL	Nominal (local thruster)	132.993	0.913	0.0097
	Switched (remote thruster)	146.072	0.896	0.0078

- Each individual thruster and FCU (and dedicated electronics where appropriate) shall be able to achieve the following total impulse and total operating life requirements (the following figures include qualification factor of 1.5): 11.5 MNs total impulse and 25,000 hours of operation;
- The mass of each thruster shall be less than 8.4 kg without harnesses; this shall include all margins;

At the time of the selection the T6 ion engine was chosen based on its average specific impulse of 4200 s. This implied the need to have an input power at maximum thrust of 10.5 kW available for the EP system, and the amount of propellant needed (580 kg) drove the design of the Xenon storage and feed system accordingly.

To be able to cope with such requirements as well as the environmental and mechanical interfaces ones, the harnesses between the PPU, thruster and FCU were recognised to be a critical system element which required a design in parallel. The design drivers are the high voltage, the high current and high temperature operation in conjunction with the need to travel the moving interface presented by the pointing mechanism (needed to comply with the requirement on thrust vector). The same happened for the fluidic connections between the FCU and the ion engine thruster. The design drivers are the low mass flow rates, very low global leak rates and high temperature operation in conjunction with the need to traverse the moving interface presented by the pointing mechanism.
Moreover, during the design phase of the EP system, the choice to use of several thrusters at the same time linked to several considerations regarding system power consumption: it is known that system power consumption has a dependence on the ageing of the thruster and PPU and the harness length and temperature. The nominal configuration corresponds to operations where a thruster is operated using the nearest PPU, i.e. the harness is the shortest. The switched configuration corresponds to operations where a thruster is operated using the other PPU, which in the case of BepiColombo is located on the other side of the spacecraft via a-cross strapping harness. A switched configuration is only envisaged to be employed in case of any failure which prohibits use of a local PPU on one thruster.

8.1.2 MarcoPolo-R

A CDF study was performed by ESA in 2011 with the objective to design a mission that returns a sample from a near-Earth asteroid of a primitive class, while providing context information on the asteroid itself. The Marco Polo mission was a Cosmic Vision M-Class and was first studied in the CDF in 2008. The changes wrt to that study were a change in the selected asteroid target; and a severe cost reduction, the price being the main reason of non-selection in the cosmic vision M1/M2 down selection in 2010.

The study objective was to reduce the cost wrt the existing MarcoPolo design (ideally by 25 % to meet the cost target), while maintaining a feasible design in order to reach the new asteroid target. In particular, the following requirements and design drivers aspects were to be studied:

- Requirement of a strong cost reduction of the existing mission , by 25 %
- Reduction of the stay time in orbit and on the asteroid surface
- Simplification of the sampling system by the removal of robotic arms
- Simplification of the landing strategy by removal of low TRL GNC equipment
- Application of a commercial platform to the interplanetary design
- Selection of a new target asteroid, increasing the transfer Delta-V.

The system main characteristics are presented in Table 8-1.

Mass (incl. Margin)	Dry mass: 1170 kg (max) Science payload mass: 25 kg Max EP propellant mass : 267 kg xenon Max CP propulsion mass: 104 kg hydrazine
Spacecraft main components	Main sampling Spacecraft (including science payload) Earth Re-entry capsule

Table 8-1: System main characteristics

The following assumptions for the use of EP were used:

- SEP trade-offs were performed in combination with a SOYUZ direct escape
- Only one thruster would be active at a time during the SEP thrust phases
- The solar arrays were sized to provide 3 kW of SEP input power at a solar range of 1 AU
- SEP is used to cruise and return from the NE.
- Whenever possible, mass, volume and power demand shall be minimised.

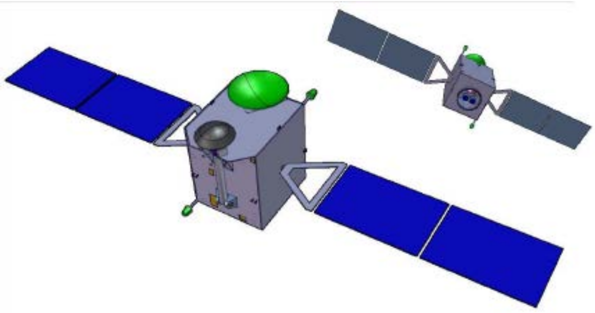


Figure 8-2: MarcoPolo-R Spacecraft in deployed configuration

To perform the requested mission, the necessary EP system shall comply with the following requirements:

- To comply with the dry mass requirements, high specific impulse thruster is needed to reduce as much as possible the xenon propellant consumption
- Very significant Delta-v budget > 4.9 km/s depending on the option
- The cost drives the choice of the quantity and of the propulsion system (complexity and number of propulsion systems): for EP for transfer and CP for SK manoeuvres. Full electric propulsion system was discarded due to both limited performance and controllability issues related to the maximum thrust achievable by EP resistojets (< 1 N, presenting a problem when in the proximity of the asteroid)
- High total impulse requirement (11- 15 MNs)
- Importance to have a good throttleability range of the thruster
- Xenon consumption based on the T6 is comprised between 188 kg and 243 kg depending on the options
- Increase of autonomy in thruster operation.

Following this study, European industry was contracted to perform an assessment study and a preliminary requirement review for this mission took place in 2013. The mission was planned to be launched in December 2022. An EP system based on the heritage of SMART-1 and AlphaBus spacecraft was chosen to decrease the costs of the mission as requested. MarcoPolo-R was not selected to continue to phase B following the decision to select PLATO as the next M3 mission by the SPC.

However without the cost reduction constraints, the overview of this type of missions allows drawing the following conclusions on the EP system:

- High Isp and higher power EP thrusters would allow reducing the propellant mass as well as the transfer time to arrive to the asteroid
- To develop long lifetime EP systems able to cope with the increasing requirements on the total impulse
- To improve the specific power of the thruster
- To have a propulsion system based completely on EP technologies, it is necessary to develop high thrust EP engines system (equivalent to resistojets performances in term of Isp)
- To have sizes of tanks corresponding to the needs of the mission (and not oversized) such as to save dry mass as much as possible
- 100 V Unregulated power bus could be used to reduce the mass on the power system (like BepiColombo). This implies that the protection of the EP subsystem is provided internally to this subsystem.

8.1.3 Mars Electric Propulsion (MEP)

Exploration of the Martian system has been identified by ESA as one of the most interesting targets for future space exploration missions. This ESA study was performed in 2008 in the ESA Concurrent Design Facility (CDF). Its goal was to study the impact of using Electric Propulsion on future missions to Mars, to assess the impact of EP on overall useful mass at Mars and to identify the technical requirements at system level. This study gave an indication on potential advantages of using this kind of propulsion.

In order to achieve the study objectives and evaluate the impact of the use of EP in interplanetary mission to Mars, a reference mission was chosen. By assessing the effect of using solar EP in the different phases and manoeuvres the MEP aims to reach exploitable and relevant results for other Mars missions. The selected reference mission was the MARS NExT.
The spacecraft was divided in two elements: it used the BepiColombo's Mercury Transfer Module (MTM) for the propulsion module and the design reference system was Mars Next (ref CDF-63(A)) for the Orbiter.

The study objectives were to design a mission and a spacecraft to:

- Deploy a network of net-science probes on the surface of Mars
- Provide communication relay and health monitoring for net-science probes
- Demonstrated MSR autonomous rendez-vous and capture.

The requirements and design drivers were the following:

- The launcher to be used is Soyuz-Fregat 2-1b.
- The launch window and transfer option: at the time of the study an arrival into the target orbit was planned not later than mid 2018. SEP should be used to a maximum extent when possible for all the phases of the mission.
- Lander deployment and Entry: the hyperbolic arrival velocity shall not exceed 3 km/s and the mass of the NetLanders shall be more than 450 kg.
- Mars Global Dust storm season: the exclusion zone extend from LS=180° to LS-340°. The requirement defined for the mission is that the Landers release should not take place close to the beginning of the exclusion zone.
- Target Orbit: the target orbit shall be a 500 km altitude circular orbit at 45° inclination. The mass into the target orbit shall be greater than the mass into orbit obtained in the MARS NExT mission design.
- The maximum mass of the NetLanders shall be maximised (around 450 kg).
- Possibility to have a full Xenon system to reduce cost and mass (common pressure control system and tanks and minimum equipment for AOCS).

Three (3) baseline options were studied with launch windows starting from May 2015 and arriving at the latest in June 2017. In those three options the transfer time was between 630 and 640 days, with a Launch capacity at escape of 2355 kg for options 2 and 3 and a launch mass of 3055 kg (excluding the adapter at insertion) for option 1.
Another option with a max power available was allowing a transfer time of 420 days with a 2355 kg launch capacity at escape.
The maximum power available at 1 AU was 12.7 kW.
The following assumptions were made:

- The main propulsion system for the interplanetary transfer to Mars was assumed to be a low-thrust propulsion.
- The capture at Mars was assumed to be achieved by low thrust propulsion; but in order to achieve capture in a reasonable time at least an average thrust of 150 mN has to be available at Mars. This sizes the power and the thrust level at 1 AU, which is approximately assumed to be at least 300 mN, as the average power available at Mars is half the power at 1 AU.

During the study the advantages and drawbacks of using EP for Mars missions have been assessed.
The following conclusions could be drawn:

- The engines to be used for this kind of mission shall be capable of high specific impulse and to provide a thrust between 0 and 350 mN;
- A full, complete xenon system with the possible use of Xenon cold gas thrusters or resistojets to perform the NetLanders release and rendez-vous operation was discarded because of the low specific impulse that would imply a high propellant consumption, and an increase in complexity off the overall system design (high number of thrusters to perform the manoeuvres for the landers).
- This kind of mission is very demanding for the power subsystem since the EP is power demanding, and hence if solar arrays are used, their area shall be quite large and the power subsystem shall be developed accordingly.
- MEP is not very demanding for the batteries because there are no eclipses during the cruise to Mars.

It has to be noted that at the time of the study and in the timeframe for the mission launch, options with European technologies with high TRL (T6 and PPS®5000) were considered, studied and proven to be achievable.

However Mars exploration is now considered in the Medium to Long term and could use thrusters of higher power than the ones studied; for instance thruster with power of 10 kW allowing then to reduce the transfer time as well as the capture as long as the thrust is low enough for the last phase. Solar cells efficiency shall be increased to decrease the necessary area and weight shall be decreased to save mass. Regarding the power topology, performance could be greatly improved if the thrusters could be directly connected to the solar array via a single processor: Direct Drive Concept. In this case the total solar array surface could be reduced as well as the mass.

8.1.4 Missions to Jupiter and beyond

Several concepts were studied for exploration missions to Jupiter and farther planets and bodies. Such a mission is challenging in terms of communication, power and mission design. However it would yield paradigm-changing advances in multiple fields of planetary science.

A study for Exploration of Neptune and Triton [RD5] was done by an ESA-JAXA team. It was motivated by the call from ESA in 2013 to define science themes for the next L-class missions. Among the enabling technologies for an ESA-led mission, a Solar Electric Propulsion (SEP) module would be used in the first part of the interplanetary transfer and ejected before approaching Neptune. The dry mass after separation of the SEP module would vary from 1400 to 1900 kg depending on the propulsion system used for Neptune orbit insertion, which is comparable to the dry mass of planetary orbiters to Jupiter and Saturn, and of the most recent studies for a NASA's Neptune orbiter.

Several options for the interplanetary transfer using chemical or SEP were analysed, but only the SEP options resulted in a sufficiently large dry mass for a L-class Neptune orbiter. A launch with Ariane 5 ECA and use of European technology for the SEP system were assumed. At Neptune arrival, the orbit insertion was done using chemical engines.

For any mission beyond Jupiter, RTGs (Radioisotope Thermoelectric Generator) are mission enabling technologies. The European program to develop radioisotope space nuclear power systems is currently at TRL 3. The radioactive isotope chosen for the program is Americium-241, which has a longer half-life than the Plutonium-238 used in conventional RTGs for space exploration. The current European RTG lifetime requirement is 20 years (3-year pre-launch ground phase and 17 years post launch), which is a requirements for the mission studied here. An alternative source of power could be the European SRGs (Stirling Radioisotope Generator), also currently under development; each unit is able to provide twice the power than the European RTGs.

The RTGs lifetime leads to a constrain on the interplanetary transfer time. The SEP module is ejected some time before approaching Neptune, and is only used in the earlier part of the interplanetary transfer, where it provides large Delta-Vs with a small propellant mass thanks to the high specific impulse. The power for the EP system is assumed to be provided by the solar arrays.

To perform such a mission, an EP system and launchers capabilities shall be based on the following requirements:

- High specific impulse thrusters :> 4000 s
- High thrust and high power EP system: > 450 mN and P > 25 kW (it can be one thruster or a combination of thrusters)
- Increase of the specific power that can be provided by the power subsystem would allow to decrease the mass at launch and as a consequence the costs of the mission, or would allow increasing the payload mass at Neptune by about 50 kg → increase the efficiency of the solar cells and reduce the weight of the panels
- An Ariane 5 ME launcher (higher performance, with a restartable upper stage) would not only increase the dry mass at Neptune orbit, but would also decrease the launch cost by around 20 %.
- Dry Mass at Neptune would be increased using several lunar gravity assists, at the cost of increased complexity. Further mass would be saved when launching the spacecraft into a highly elliptical orbit, and raising the apogee with the onboard engine until escape or lunar GA.

A NASA study JIMO (Jupiter Icy Moons Orbiter) was performed in 2003 to design a mission and spacecraft to explore the icy moons of Jupiter. The main target was Europa, Ganymede and Callisto. JIMO was to have a large number of revolutionary features: through its main voyage to the Jupiter moons, it was to be propelled by an ion propulsion system and powered by a small fission reactor.

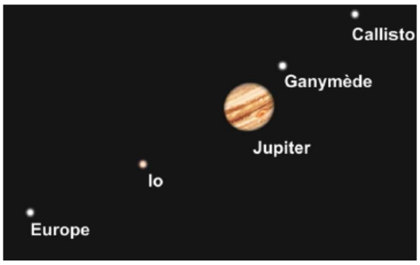


Figure 8-3: Jupiter and its moons

The ESA mission JUICE (Jupiter Icy Moons Explorer) which is the first Large-class mission in ESA's Cosmic Vision 2015-2025 programme has exactly the same targets. It is planned for launch in 2022 on Ariane 5 ECA (Delta-V at escape of 3.15 km/s) for an arrival at Jupiter in 2020. The spacecraft will spend at least three years making detailed observation of the biggest planet of the Solar System and three of its largest moons. Today the transfer cruise to the planet is planned to be performed with chemical propulsion thrusters, to reduce the transfer time.

However, as specified for JIMO and the previous study to Neptune, the spacecraft could be propelled by an Electric Propulsion system which would allow saving mass at launch (which is critical for any high Delta-V mission) or increasing the payload mass arriving at Jupiter. A high level study was recently performed by Snecma to use EP based on HET technology as an alternative for the JUICE mission. With the available power on board (Figure 8-4), they demonstrated that using EP for such a mission would allow reducing the proportion of propellant mass (400 kg of xenon) vs launch mass down to 18 % with respect of the 60 % for one of the chemical propulsion option. It is clearly demonstrated as well, following the proposed mission analysis for JUICE (not optimised for EP) that the insertion into Jupiter orbit is also feasible (Delta-V = 0.2 km/s), adjusting the angle and amplitude of the thrust vector obtained at the arrival at Jupiter (Delta-V = 0.9 km/s between Earth and Jupiter).

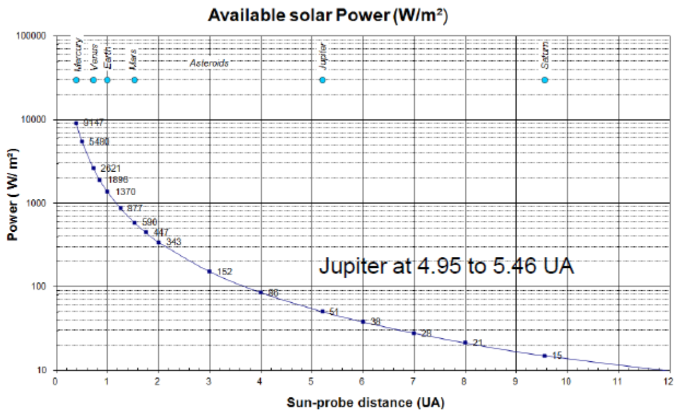


Figure 8-4: Available solar Power during the mission to Jupiter. (Credit: Snecma, RD7)

Taking into account the limited power around Jupiter (636 W), the following high level requirements for the EP system could be deduced from the Juice EP-type and from the JIMO studies:

- The required Total Impulse should be higher than 6.2 MNs;
- The power of the EP system shall be flexible;
- The thruster itself could be in the 20-50 kW class (with the corresponding power generation system), with a specific impulse of 6000 -9000 s, and able to deliver about 500 mN of thrust;
- The propellant throughput capability of the system shall exceed 100 kg/kW.

As well, mitigation options of mass increases exist by using a high performance launcher (better then Ariane 5 ECA). And, if a long interplanetary transfer is allowed, the escape velocity can be reduced, changing the mission profile and allowing higher available mass capability.

8.1.5 Hybrid transfer : an option for exploration missions

This chapter is intended to show the advantage of “hybrid” transfer for planetary missions using :

- EP for orbit raising
- Chemical injection to interplanetary speed
- EP thrust arc in interplanetary space.

An example of procedure and benefits of using EP are shown for a Mars Sample Return mission below (excerpts from “Single Launch, Direct Earth Return MSR Mission. 42-081, AAAF Space Propulsion 2008, Heraklion, 5-8 May 2008”) with the following drivers and requirements:

- Heavy platform toward Mars.
- Launch in GTO like orbit (inclination adapted to ecliptic plane).
- GTO mass : 12000 kg (AR5 ECB)
- Power: 25 kW

- Isp: 1600 s
- Thrust: 1.33 N
- Delta V: 1000 m/s (700 m/s if chemical)
- Propellant mass : 740 kg
- Mass in HEO: 11259 kg
- Delta V for Mars injection reduced from 1300 m/s to 600 m/s

For a Low Cost Cryogenic Propulsion interplanetary injection with an Isp of 440 s, the following results were obtained:

- Mass in interplanetary orbit : 9797 kg
- Dry mass (Electric and LCCP) : 870 kg
- Useful mass in interplanetary orbit : 8927 kg

If the same procedure is applied with bipropellant propulsion, then one obtains:

- Mass in interplanetary orbit :9299,74 kg
- Dry mass ;718 kg
- Useful mass in interplanetary orbit : 8927 kg

A comparison with an all chemical bipropellant injection was performed with the following drivers :

- Isp : 320 s
- Delta V : 1300 m/s
- Mass in interplanetary : 7980 kg
- Propellant : 4069 kg
- Dry : 450 kg
- Useful mass in interplanetary orbit: 7530 kg

i.e that there is a loss of mass of 18.5%w.r.t the EP and CRYO option.

However using EP alone from LEO to interplanetary is not efficient. Two examples are given below:

- Example 1: Escape to Mars from escape : 3 km/s (instead of 0.5 km/s impulsive)
- Example 2: Spiralling from LEO to escape
    - Delta V = 7.7 km/s
    - Impulsive LEO – escape = 3.9 km/s.

8.2 Air breathing

To perform such long duration missions or simply to increase their lifetime, several studies were performed regarding the so-called Air breathing technique where the propulsion system would use the propellant available in the atmosphere or in the close vicinity of the visited planet.

At the end of 2006, ESA performed an internal study, called RAM-EP [RD9], on the use of advanced EP for very low Earth Orbit missions using atmospheric gases at propellants. The main requirements for the definition of the technology demonstration mission were as follow:

- Fly a spacecraft at orbital ranges consistent with the performance of drag compensation and altitude control by RAM-EP propulsion
- Have a duration sufficient to prove the lifetime increase benefits from the RAM-EP concept
- Rely on the Ram-EP propulsion subsystem during the nominal mode of operation when drag compensation is performed
- Perform the nominal operations with the RAM-EP thruster at least for 90 % of the mission lifetime
- Include a payload representative of an EO mission
- Be able to operate the EO payload at the same time as the RAM-EP thruster.

The study was performed with HET and GIE technologies and was demonstrated to be feasible at the condition that the thrusters could cope with the lifetime requirements; the overall thruster lifetime was about 7 years (> 60,000 hours), as

the RAM-EP engine shall practically be always on. Also it is necessary that the EP system is able to operate with different gas mixture compositions due to the variation in altitude, and in particular, with different oxygen fractions.
Following this study, experimental investigations concluded that HETs and GIEs would be able to work with such kind of propellants with a few design changes. Indeed today the efficiency of working with such propellants is much lower for both types of thruster and the lifetime reduced due to oxidation of some materials.

The same principle can apply to any other types of atmospheres (from other planets for instance). The necessary conditions are:

- To be able to collect the said propellant in a kind of collector
- To bring the collected propellant up to the thruster inlet
- To have a suitable EP system (thruster, neutraliser, fluidic system, etc.) able to work with such propellant for a long time.

It should be noted that an hybrid system can be of interest here, i.e. the main flow to the EP thruster would be provided by the Ram air and the cathode(s) flow provided by the noble gas. The embarked propellant mass would be divided by a factor of about 10 wrt conventional EP systems.

8.3 Science missions

In the case of science missions, the scientific requirements drive the propulsion system to be installed on board of the spacecraft and in certain cases EP is the enabling technology for this kind of mission. It provides a better resolution, accuracy and savings of propellants.

Several ESA scientific missions are scheduled such as Lisa PathFinder or Euclid, or being studied such as NGO (New Gravitational wave Observatory) for which EP was or is-being considered to be used. The scientific requirements can be so stringent that specific EP systems may have to be developed to cover the needs of the mission or to enable it.

8.3.1 Euclid

The ESA Euclid mission is a cosmology mission, planned for launch by 2020 in the Soyuz ST 2-1 B from French Guiana. It has for main scientific mission to understand the nature of Dark energy and Dark Matter and to map the geometry of the dark Universe. To do this, Euclid will investigate the distance-redshift relationship and the evolution of cosmic structures by measuring shapes and redshifts of distant galaxies out to redshifts ~ 2, or equivalently by looking back on 10 billion year of cosmic history. Euclid spacecraft is composed of three modules: the Sunshield, the Payload Module and the Service Module. The last spacecraft comprises the spacecraft subsystems supporting the payload operation, hosts the payload warm electronics and provide structural interfaces to the payload Module, The Sunshield and the launch Vehicle.

To be able to comply with the stringent scientific requirements on precision and pointing accuracy, a micro-propulsion is devoted to provide the finely controlled torques during the science pointing sessions when exceptional attitude stability is required. The following main requirements for the micro-propulsion system are given:

- Need of a system capable to operate with very high resolution (< 1 µN) in the micro- Newton range (from 50 µN to 100 µN) during science phase of the mission, and in the milli-Newton range (up to 2.5 mN) in the high thrust mode. .This broad throttleable range would allow completely removing the reaction wheels on the spacecraft.
- Very low noise system
- High total impulse system able to operate in orbit for more than 6.25 years continuously, i.e an EP system with a lifetime of more than 55,000 hours of operation
- Light and small system
- Minimisation of the power consumption and the PPU dissipation for any operating regime.

Presently EP is considered as a back-up solution to a micro-propulsion system based on the GAIA micro cold gas thrusters' heritage.

8.3.2 New Gravitational wave Observatory (NGO)

This ESA mission was designed to measure gravitational radiation over a broad band at low frequencies, from about 100 μHz to 1 Hz, a band where the Universe is richly populated by strong sources of gravitational waves. It was the result of the reformulation, in 2011, of the LISA mission.

Even if this mission was not selected to continue into the definition phase, it can serve as a good basis to derive the requirements that would be necessary for an EP system to be chosen as a baseline for such a mission.

The New Gravitational wave Observatory (NGO) mission planned to employ three spacecraft deployed in a V configuration with an arm length of 10^9 metres, positioned in heliocentric, Earth trailing orbits (HETOs) with the plane of the constellation inclined at 60° to the ecliptic. Gravitational waves would be detected by measuring changes in path length between free falling test masses housed in each of the spacecraft to picometre accuracy.

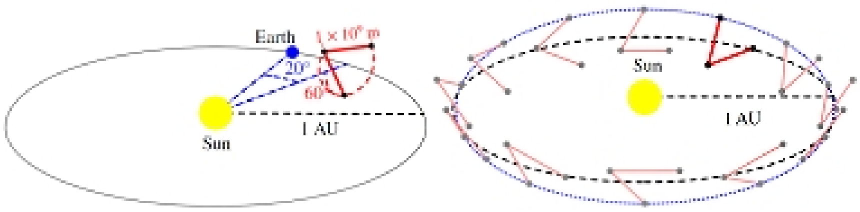


Figure 8-5: Schematic diagram of NGO's science orbit. Credit ESA

The spacecraft at the apex of the 'V' is referred to as the 'Mother' spacecraft; those at the ends of the arms are 'Daughter' spacecraft. The mission was planned to be launched by two Soyuz vehicles, one carrying the Mother spacecraft, the other the two Daughter spacecraft. Each spacecraft would be equipped with a propulsion module that would manoeuvre it during a 14-month transfer to its operational orbit. Once on orbit, the propulsion modules would be separated to ensure that they do not disturb the payload. The spacecraft itself forms the science instrument. The two sides of the equilateral triangle, from the apex to the two vertices with a spacecraft at each end, form two measurement arms. Each spacecraft carries the interferometry equipment needed to measure changes in arm length. The spacecraft at the apex, or the 'mother' spacecraft, houses two send/receive laser ranging terminals and two free-falling test masses, forming one set of endpoints of the two measurement arms. The other two spacecraft at the vertices of the triangle, the 'daughter' spacecraft, each housing one laser ranging terminal and free-floating test mass, form the two other endpoints of the measurement arms. The mother and daughter spacecraft are identical, but for the number of payloads they carry.

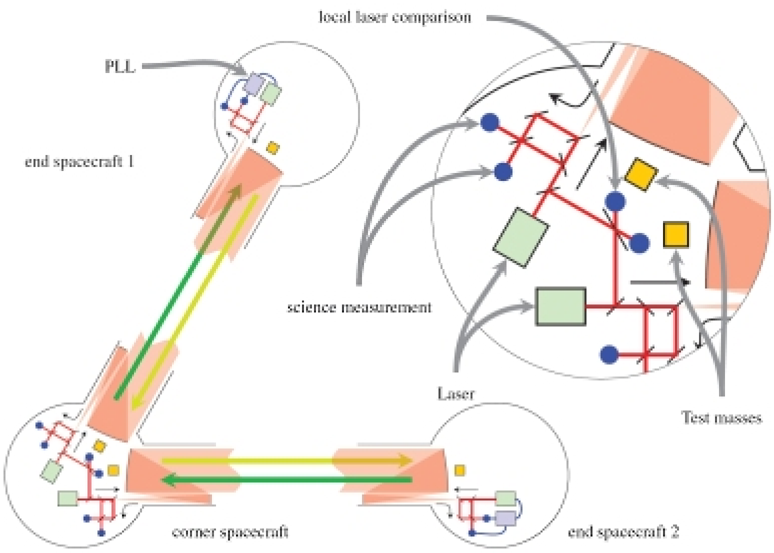


Figure 8-6: Measurement concept: To measure the strain, or deformation, on the fabric of spacetime caused by gravitational waves, the mission uses precision laser interferometry. Credit: ESA

Each NGO spacecraft carries an identical payload (the Mother spacecraft carries two, the Daughters one each), provides resources for the payload, and accommodates the micropropulsion system. The propulsion modules, which are responsible for delivering the spacecraft to their operational orbits, will be separated following final orbit acquisition to ensure that no disturbance generated by them will affect the payload.

The NGO payload is composed of elements that are either inherited from LISA Pathfinder (LPF) or are the object of ongoing dedicated technology development activities performed by European industries and laboratories.

The main objective of the Disturbance Reduction System is to maintain the free fall of the test mass that serves as nominal reference point for the measurement of the inter-spacecraft distance. To do so, the DRS controls a drag-free attitude control system (DFACS) by measuring the position and orientation of the test mass with respect to the spacecraft, applying a control law and commanding micro-newton thrusters such that the test mass remains in its nominal position with respect to the spacecraft. As for the payload, the micro-propulsion can be inherited from LPF or can be developed to comply with the stringent requirements, such as the one on the thrust noise (see Figure 8-7).

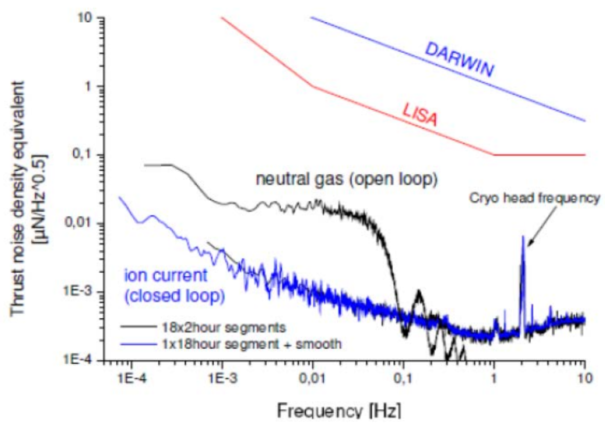


Figure 8-7: Thrust noise density vs frequency for the NGO (previous LISA) and Darwin missions

8.4 Technical requirements

Most of the future missions presented here are based on existing technologies with a TRL high enough that they could be considered to be ready at the time of the mission.

In general, the longer the orbital period, the more it can be done by EP along the orbit. This is why interplanetary missions are in general a good application field for EP, both for the enabling and opportunity segments.

The current limit of use for interplanetary missions derives from the power source: all European EP missions are using substantially the solar EP type, the energy source being the Sun and the harvesting device being the solar cell arrays. De facto this limits the use of EP to the inner solar system, i.e. up to about asteroids belt. Beyond that point the solar array sizing would be unfeasible with the current technology.

Typical interplanetary missions are characterised by the availability of a lot of firing time. Therefore the propellant budget minimisation is a key feasibility parameter. High specific impulse system (> 1000 s) is required due to high velocity increments. As well, lower thrust-to-power ratio is acceptable due to long transfer phases (> 20 mN/kW) as depicted in Figure 8-8. Moreover, development risk minimisation is also important but is not key for this kind of missions.

Exploration missions have specific performance requirements, and in particular, future missions may require higher specific impulse and total impulse levels currently only available with ion engines.

Availability of very-high-power HET and/or HEMPT and/or GIE systems as well as new disruptive technologies will enhance future robotic exploration missions. Such system will provide remarkable mass gain, which can be converted into either additional payload or a reduced launch mass, with acceptable transfer duration.

A class of interplanetary missions are scientific observatories which need very low disturbances: there, enabling EP technology is very low noise micro-propulsion. The corresponding high level requirements are the same as for LEO micro-propulsion, with the addition of stricter requirements on thrust noise.

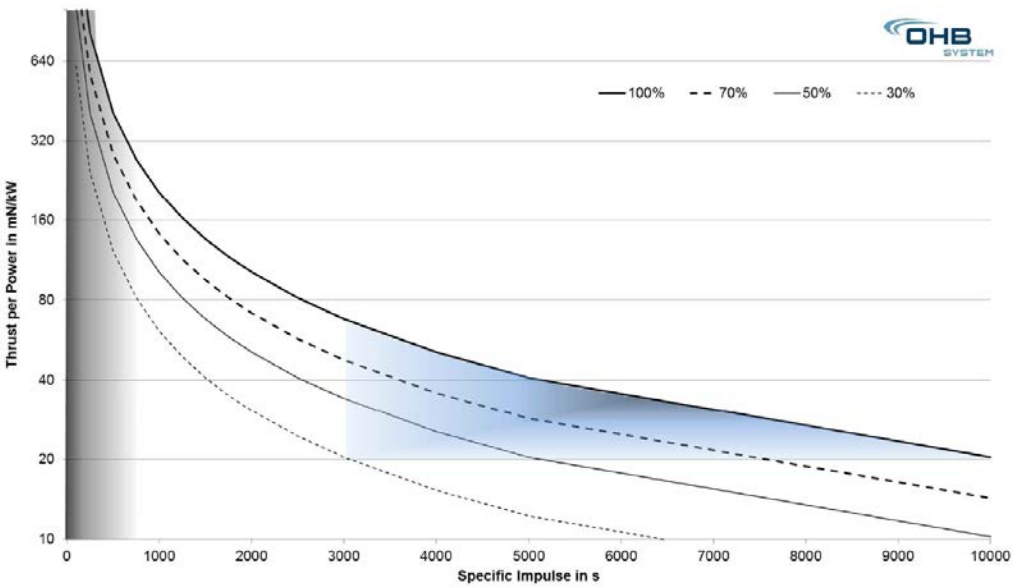


Figure 8-8: Thrust-to-power ratio vs Isp for interplanetary missions [RD7, Credit OHB]

9 CONCLUSIONS

A non-exhaustive review of several studies done by Agencies, European primes and industries, was performed for different types of applications: Telecommunication platforms, LEO and MEO spacecraft, Space transportation area and Exploration and science missions. From this review, the EP technology is facing operational needs which can cover:

- Electric transfer from GTO to GEO
- Station keeping
- Interplanetary cruise
- Continuous LEO operations (air-drag control)
- (extreme) fine and/or highly agile attitude and positioning control (Earth observation and science missions)
- Long-endurance missions
- Interorbital transfer.

High level requirements that the propulsion system and particular an EP system should comply to, could be derived from those applications and needs.

The figures provided for each requirement should not be taken as strict figures as they come from specific examples/missions but should be included in broader ranges. As well, some examples were studied and then provided for a specific type of thruster; however several technologies could be well answering the same needs and requirements. Those requirements should as a consequence be flexible enough to select the most suitable technology for the specific mission taking into account of economic, technical and programmatic factors.

And finally it should be highlighted that in the near future no big European exploration missions using Electric Propulsion are planned by ESA or National Agencies, maybe because of the unavailability of adequate EP systems in terms of power or lifetime to perform extremely challenging missions.