

HT5k Hall Thruster to Improve Small Launcher Capabilities

IEPC-2013-279

*Presented at the 33rd International Electric Propulsion Conference,
The George Washington University • Washington, D.C. • USA
October 6 – 10, 2013*

Tommaso Misuri¹, Pierpaolo Pergola²
Alta S.p.A., Ospedaletto, Pisa, 56121, Italy

and

Mariano Andrenucci³
Alta S.p.A., Ospedaletto, Pisa, 56121, Italy

Abstract: To enhance the capabilities of small launchers, such as Vega, an upper stage propelled with one or more high power Hall Effect Thrusters (HET) has been identified as a valuable solution. Among electric thrusters, Hall Effect thrusters offer the best compromise between thrust level and specific impulse. High power HETs have an efficiency in excess of 55%, and their fairly high specific impulse allows for important savings in terms of propellant mass for a given mission Δv . Besides, the thrust they provide is significantly higher than other electric devices (i.e. Gridded Ion Engines), thus allowing for faster orbit transfers and making HET especially suitable for orbit raising tasks. With a 5 kW HET, thrust levels up to 0.3N are easily achievable and a cluster of five thrusters offers up to 1.5N. Specific impulse is close to two thousand seconds in the high-thrust mode and it can be raised up to 2500s at the expense of the total thrust provided (high specific impulse mode). With such performance and considering an available power of 10kW, it is possible to lift a mid-size spacecraft (1000–2000kg) from a low Earth orbit to the geostationary Earth orbit in about six months and to a highly inclined medium Earth orbit (like the one used by the Galileo constellation) in less than nine months, with consistent mass savings if compared to traditional chemical propulsion systems. In this study several mission profiles are proposed and analyzed considering the Vega as baseline launcher and showing the benefits of adopting electric propulsion to enhance its current capabilities. Different spacecraft configurations are considered, with initial masses compatible with Vega performance and on-board power between 5 and 25kW. The thruster unit considered for this analysis is Alta 5 kW HET.

Nomenclature

a	=	orbit semi-axis
a_t, a_r, a_h	=	spacecraft acceleration (tangential, radial, normal)
i	=	orbit inclination
m	=	spacecraft mass
t	=	firing time
T	=	thrust
V	=	velocity

¹ Project Manager, Research Engineer, Alta S.p.A.

² Project Manager, Research Engineer, Alta S.p.A.

³ President, Alta SpA; Professor of Electric Propulsion and Space Systems, University of Pisa

β	=	yaw angle
Δv	=	mission velocity change
μ_{Earth}	=	Earth gravitational parameter

Acronyms

EP:	Electric Propulsion
GEO:	Geostationary Earth Orbit
GIE:	Gridded Ion Engine
HET:	Hall Effect Thruster
MEO:	Medium Earth Orbit
LEO:	Low Earth Orbit

I. Introduction

In the fast-growing market of small satellites, including small telecommunication platforms (a field pioneered by missions like Small-GEO¹), there is a vast interest in finding cost-effective strategies to place them in their target orbits. To reach far orbits, such as the Geostationary Earth Orbit (GEO) or the ones required for navigation satellites (usually Medium Earth Orbits, MEO), an appealing option is to use a small launcher to bring the satellite into a Low Earth Orbit (LEO) and then raise the orbital altitude by means of an Electric Propulsion (EP) system^{2,3,4}.

Nowadays electric propulsion has a sufficient degree of maturity that makes it ready to take on the challenge of orbit raising tasks. This aspect combined with the fact that small launchers are much cheaper than heavy ones and that the available electric power is rapidly increasing on-board modern satellites, provides a credible alternative to traditional apogee kick motors.

With a specific impulse six-times higher than chemical rockets commonly used for orbit transfers, electric thrusters allow for important savings in terms of propellant mass. This means that a larger payload mass fraction can be raised into the target orbit or that lighter spacecraft can be used to accomplish the same mission.

In addition, once the satellite has reached its final orbit, a substantial part of the orbit raising propulsion subsystem can be still used to feed another electric propulsion subsystem devoted to orbit maintenance and control tasks. This concept, known as ‘All-Electric’ spacecraft^{5,6}, is generally based on two different electric propulsion subsystems:

- high power thrusters are used for orbit transfer, when almost all available electric power can be used for propulsion to minimize the transfer time;
- mid-low power thrusters are used once the satellite is in the proper orbit, the maneuver time is not an issue anymore and a reduced power amount can be allocated to perform station-keeping tasks.

It goes without saying that a crucial choice for such kind of missions is the electric propulsion device to be adopted. Among electric thrusters, Hall Effect thrusters seem to offer the best compromise between thrust provided and specific impulse. High power HETs have a very good efficiency, exceeding 55%, and, for a give mission, their fairly high specific impulse allows for important savings in terms of propellant mass. The advantage of electric thruster results particularly clear when the mission total impulse is sufficiently high and thus an increased thruster specific impulse allows for significant propellant mass savings. In addition also the propulsion system and power generation system masses are not significantly heavier than a classical chemical system and, especially in the case of geostationary platforms, part of the resources already available on-board (e.g. the some kW power generation system) might be exploited also during the transfer phase.

For a low-thrust transfer in a strong gravity field, like the one affecting a LEO-GEO/MEO, the 0.3–1.5N offered by high power HET together with few tons of initial spacecraft mass allows for a limited transfer time, of the order of few months, that is significantly longer of the one resulting by chemical thruster, but is still within acceptable limits. As a comparison, a Gridded Ion Engine offering 200 mN thrust might accomplish the same transfers here analyzed in more than one year, thus posing significant challenges to the ground segment, collision probabilities, in-orbit spacecraft system deterioration and so on.

In the present study VEGA launcher has been adopted as baseline in combination with Alta HT5k Hall thruster (both, in single-thruster and cluster configuration). Main features of both, the launcher and the thruster, are recalled in section II.

LEO-GEO transfers for small geostationary platforms and LEO-MEO transfers for Galileo-class satellites have been chosen as reference mission scenarios for our analysis. Details of target orbits and analysis results are presented in sections III and IV respectively. It is worth stressing, however, that these scenarios have been chosen

only because they belong to the most common commercial applications of small spacecraft, but the opportunities offered by a Vega electric upper stage are much broader.

Section V presents some possible system configurations for the transfer system architecture. Mainly two cases are analyzed; the electric propulsion subsystem is directly integrated in the spacecraft or it is part of a dedicated module that serves as Vega upper stage to bring the payload into the desired orbit. While the first configuration allows for a more wise exploitation of the main subsystem already available on-board of the satellite (where only the electric propulsion system has to be integrated and the power generation system possibly adapted), the second scenario allows for a more flexible solution. In this case, indeed, the electric powered upper stage is completely independent from the payload. In general such an upper stage would broaden the small launcher market allowing for missions neither conceivable with the current configurations.

Finally section VI presents a further comparison of the mission architecture conceived by considering alternative propellants for the HET system.

II. Vega Launcher and HT5k Thruster Characteristics

The Vega launcher system is an expandable system jointly developed by Arianspace and the Italian space agency and its first launch took place on February 2012 from the French Guiana. It has been developed in response of the need of a small not-expansive launcher to place 300–2000kg satellites into polar and low Earth orbits. Typical applications are Earth observation and scientific missions; although the modern trend in the spacecraft miniaturization might also consider this launcher for small geostationary platforms and communication constellation.

Vega launch vehicle is a small single-body launcher 30m in height, 3m in diameter and with a liftoff mass of 137tons⁷. The vehicle is designed to lift up to 2500 in a 300km almost equatorial orbit and up to 900kg in a 1500km Sun-synchronous almost-polar orbit. Figure 1 presents an up to date Vega performance map.

The propulsive system consists of three solid-propellant stages (P80, Zefiro-23, Zefiro-9) and a re-startable liquid-propellant upper stage, known as Attitude and Vernier Upper Module (AVUM). Thanks to this module, Vega is able, unlike most small launchers, to place multiple payload on orbit, but the orbital altitudes it can reach are rather limited by the main thruster performance. The current launcher configuration does not include an upper stage (besides the AVUM).

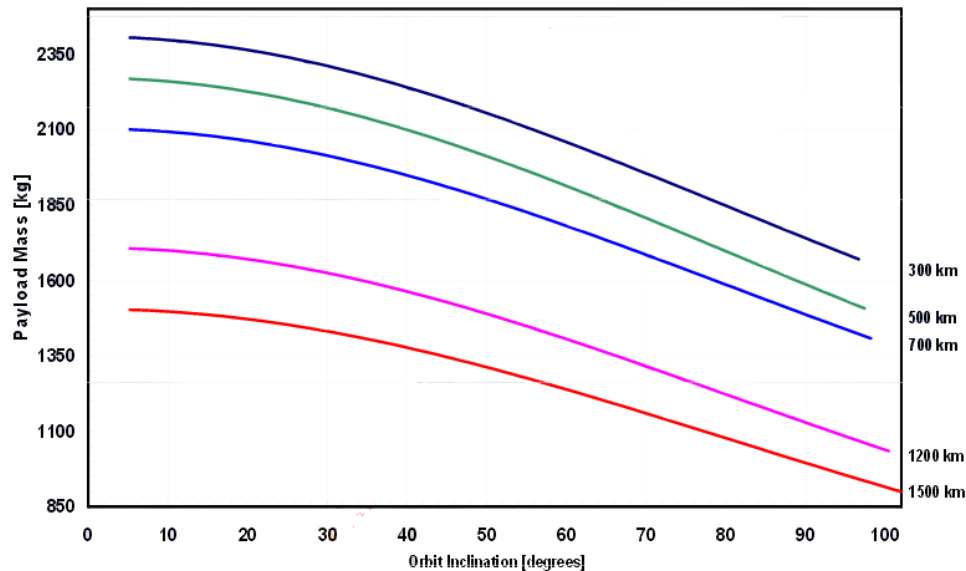


Figure 1: Vega performance map for the whole range of reachable inclinations and altitudes.

A program to upgrade Vega performance of about 30% (LYRA program) is currently ongoing and it is focused on a substantial re-design of the 3rd and 4th stage. In our analysis such an improved configuration has not been considered, but it could enable much more ambitious transfers when coupled to an electric propulsion upper stage, bringing to GEO payloads close to 2 tons.

The mission scenario deepened in the paper is to combine the potentialities of such a small launch system with a high efficient HET. In particular the idea is to analyze the possibility to use the Vega launcher for commercial purposes considering typical payload on classical target orbits, like GEO and MEO.

In order to limit the transfer time required, considering the Vega performance in LEO, a thrust level of at least 0.25–0.30N is a recommended feature for the candidate propulsion device. Of course, for a given initial mass the higher the thrust the shorter the transfer time, thus more appealing the solution proposed. A higher thrust, however, requires a larger electrical power, thus a heavier power generation system and larger solar arrays. A tradeoff between these two needs is the core of the analysis here detailed.

Assuming 0.3N as reference thrust level, the choice of the electric propulsion system naturally fell on the 5kW Hall Effect Thruster such as Alta HT5k, see Figure 2. With a single thruster unit 0.33N can be achieved, while a cluster of five units (thus 25kW, considered as a realistic short-term upper limit for the on-board power generation system) can provide a thrust in excess of 1.5N. HT5k, as well as any Hall thruster, can be operated in two distinct modes for a target power level:

- *High specific impulse mode*: increased discharge voltage and reduced mass flow rate. This directly translates in faster ions and specific impulse improvement, while the thrust level decreases. The presence of high energy ions in the beam enhances the chamber wall erosion, reducing the thruster lifetime.
- *High thrust mode*: increased mass flow rate and a decreased voltage. The thrust provided is increased and the “lifetime-to-mission_time” ratio is improved as well. For orbit raising tasks this is the preferred mode, since a higher thrust abates the total transfer time.

HT5k performance in both modes is summarized in Table 1.

	High Thrust Mode	High I_{sp} Mode
Power [W]	5000	5000
Thrust [N]	0.33	0.25
Specific Impulse [s]	1700	2400
Expected lifetime [h]	~7500	~7500

Table 1: HT5k performance in high-thrust and high- I_{sp} modes

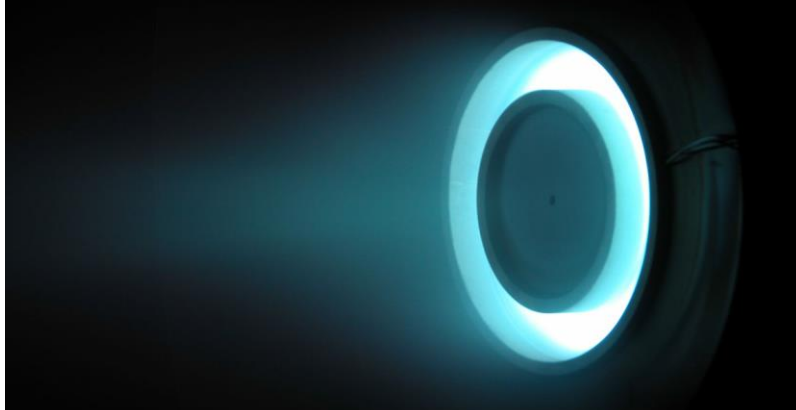


Figure 2: HT5k prototype firing in Alta’s laboratories

III. Mission Scenarios: LEO–GEO and LEO–MEO Transfers

In this study two scenarios are identified as the most interesting for developing an electric propulsion system for LEO raising. Assuming to use the Vega launch system as reference (although the analysis might be done for any small launcher), we assumed Kourou as reference launch site (latitude 5.2deg). This is a quasi-optimal choice for LEO–GEO transfers, while for Galileo satellite target orbits (inclination 56deg) the larger part of the propellant is used for the plane-change maneuver (in this scenario more advantageous launch sites might be identified).

In the analysis initial orbits between 300 and 1500km are analyzed. The initial mass in each of this orbit is taken from the Vega performance map (see Figure 1) assuming 5.2deg of orbit inclination. This assumption results from the consideration that if any inclination change maneuver is required it is more convenient to use the more efficient low-thrust device rather than reduce the launcher payload. In particular the initial masses considered in LEOs are given in Table 2.

Altitude [km]	Vega payload mass [kg]
300	2300
500	2210
700	2075
1000	1810
1200	1670
1500	1475

Table 2: Vega capabilities (orbit height vs payload mass) at 5.2deg inclination

Two target orbits are analyzed:

- classical GEO with 35768km altitude and 0deg inclination. This is the standard orbit for geostationary satellites. Such a kind of satellites usually weights several tons and provides some tens of kW of on-board power. Recently, however, small geo platforms have been developed as valuable bus for low cost GEO missions. These platforms weigh less than 2tons and generate 3.5kW of power (End of Life)¹. One of the aims of the analysis is to assess if such a kind of payload can be delivered in GEO starting from a Vega launch in LEO and using the HT5k equipped upper stage.
- circular MEO with 22322km altitude and 56deg inclination. These are the orbital characteristics of the Galileo constellation⁸. Galileo is a global positioning system jointly developed by the European Union and the European Space Agency. It is composed of 30 satellites (27 active and 3 spare) distributed in 3 orbital planes with ascending nodes separated by 120deg latitude. Each satellite weights around 675kg, generates 1.5kW end-of-life and is designed for a lifetime of 12 years. The constellations should be completed in the 2019. One of the aims of the study is to assess if the conceived system configurations, composed of the Vega and high power HET, are suitable for delivering such a kind of satellite into the target orbit.

IV. Mission Analysis

All mission scenarios described in the previous section are analyzed with a two step approach. In a first phase the low thrust mission total velocity change (Δv) and transfer time are assessed and as second step these data are used to estimate the actual payload mass into the target orbit (see Sec V).

The mission analysis approach follows a numerical method deriving from the Kechichian modification⁹ to the Edelbaum algorithm¹⁰. The method avoids the direct integration of equations of motion (highly time consuming) and provides an alternative method for optimizing the thrust law. The idea is to approximate the low-thrust transfer with the assumptions that the transfer orbits remains always almost circular and the acceleration magnitude is constant. The thrust is considered to be constrained in the plane containing the orbital velocity and the orbital angular momentum vector. The reference frame used is a classical radial-tangential-normal one ($[r, t, h]$) where the thrust acceleration a_T is given as a function of a single angle, the yaw angle β :

$$a_T = [a_r, a_t, a_h]^T = \left[0, \frac{T}{m} \cos \beta, \frac{T}{m} \sin \beta \right]^T$$

where T is the thrust magnitude and m the spacecraft mass.

The Edelbaum approach linearizes the Lagrange planetary equations¹¹, providing the variation of classical orbital elements under a given perturbation (as the low-thrust acceleration). As the transfer is assumed to be composed of almost circular orbits, the two relevant orbital parameters to take into account are inclination and semi-major axis. Thus from the linearization of the relevant equations for these two parameters and averaging out the angular position along a given orbit, it is possible to determine the optimum thrust angle performing the required semi-major axis and inclination change at once. In particular it results that the β starts at a value β_0 (defined only by initial final orbit characteristics and evolves in time t with the following law:

$$\beta_0 = \tan^{-1} \left(\frac{\sin \left(\frac{\pi}{2} \Delta i_0 \right)}{\frac{V_0}{V_f} - \cos \left(\frac{\pi}{2} \Delta i_0 \right)} \right) \quad \beta = \tan^{-1} \left(\frac{V_0 \sin(\beta_0)}{V_0 \cos(\beta_0) - a_r t} \right)$$

Where Δi_0 is the total inclination change between initial and final orbit and V_0 and V_f the circular velocities on these orbits, respectively.

The thrust law defined in this way provides an inclination and semi-major axis change given by:

$$\Delta i(t) = \frac{2}{\pi} \left(\tan^{-1} \left(\frac{a_T t - V_0 \cos(\beta_0)}{V_0 \sin(\beta_0)} \right) + \frac{\pi}{2} - \beta_0 \right) \quad \Delta a(t) = \frac{\mu_{Earth}}{V_0^2 - 2V_0 a_T t \cos(\beta_0) + (a_T t)^2}$$

The algorithm is implemented for a discrete number of time steps until the final inclination and semi-major axis match the ones of the target orbit. Moreover the algorithm has been slightly modified in order to have the acceleration constant only for a given set of orbits (thus satisfying the assumptions on which the Edelbaum/Kechichian equations are based on), but changing from set to set. The spacecraft mass is indeed updated from step to step and so the inclination and semi-major axis variations. The algorithm returns the thrust time and the propellant mass consumption (and from this the Tsiolkowsky equation returns the mission Δv).

The algorithm is further adapted assuming to have no batteries on-board and thus turning off the thruster during eclipse periods. Eclipse periods are evaluated each time step by assuming a simple cylindrical shadow model in the worst conditions (as if the orbit lies on the ecliptic plane).

Atmospheric drag has been neglected, as it amounts to a maximum value of 15% of the thrust level at 300km and quickly (exponentially) vanishes for higher orbits. This stands independently of the power level considered, since both thrust and atmospheric drag are roughly proportional to the power level itself (twice the power means twice the thrust but also twice the solar arrays surface).

Fix thruster characteristics have been used, but in a power range between 5 and 25kW. As a consequence we assumed to use from a single thruster up to a cluster of five units. Each thruster is assumed to provide 0.33N with a specific impulse of 1700s (high thrust mode operation, see Sec. II).

A typical mission profile for a 300km LEO-GEO transfer is shown in Figure 3. Thrust angle, β , ranges from 5.3deg to 13.5deg, both inclination and semi-major axis reach the target values and the eclipse duration ranges from 40% of orbital period to 5%.

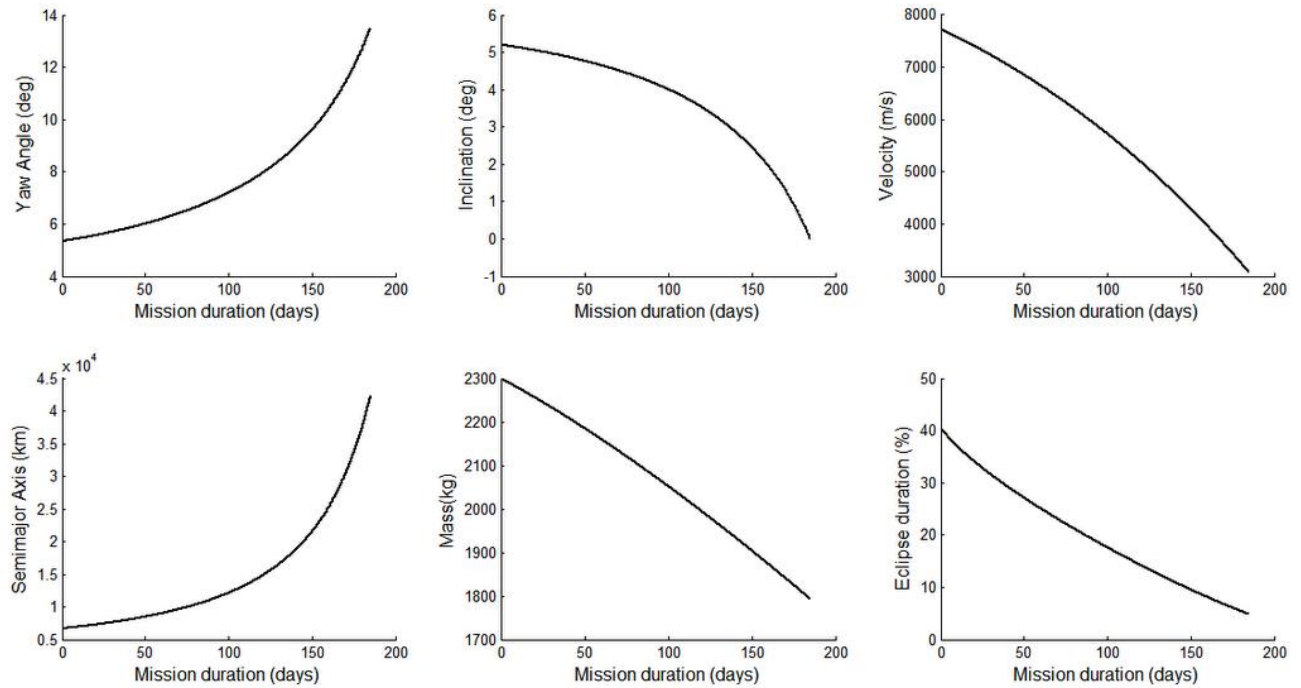


Figure 3: Time evolution of relevant mission parameters controlled by the algorithm implemented

A. LEO–GEO Transfers

The first group of cases here presented is referring to LEO–GEO transfers. Results in terms of final mass that can be brought to GEO are displayed in Figure 4 compared to the final mass that could be transferred in the same orbit by means of the chemical propulsion option.

Starting to use EP from a lower orbit is advantageous in terms of transferred mass since a larger portion of the transfer is performed by means of electric propulsion instead of using Vega’s chemical rockets.

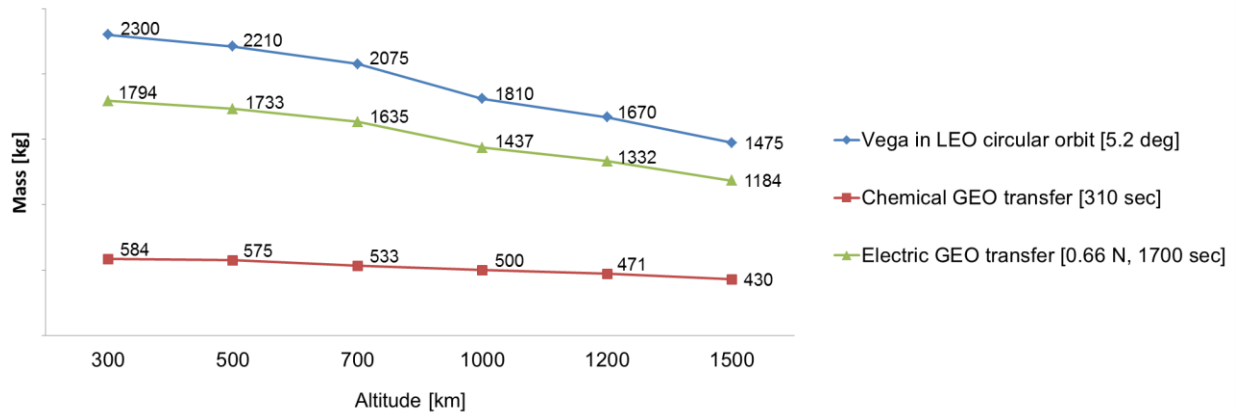


Figure 4: Final mass in LEO (Vega performance) and GEO (chemical and electric comparison)

As a comparison in Figure 4 also the Vega performance in LEO is given. It results that from 1184 to 1794kg can be delivered in GEO, while the chemical case only allows for 430–584kg in the same orbit. Thus with the EP configuration there is an average gain of 52% of the initial mass delivered in GEO.

Following the approach outlined, results shown in Figure 4 are independent of the power level adopted, since the specific impulse is invariant. According to the power level available, an array of thrusters can be installed on-board enhancing the total thrust but not affecting the specific impulse of the EP subsystem. As a consequence, power only affects the transfer time, see Figure 5. For a chosen power level, going to GEO from a 300km LEO takes nearly twice the time than starting from a 1500km LEO. However, as explained, this is counterbalanced by a larger final mass placed in the target orbit.

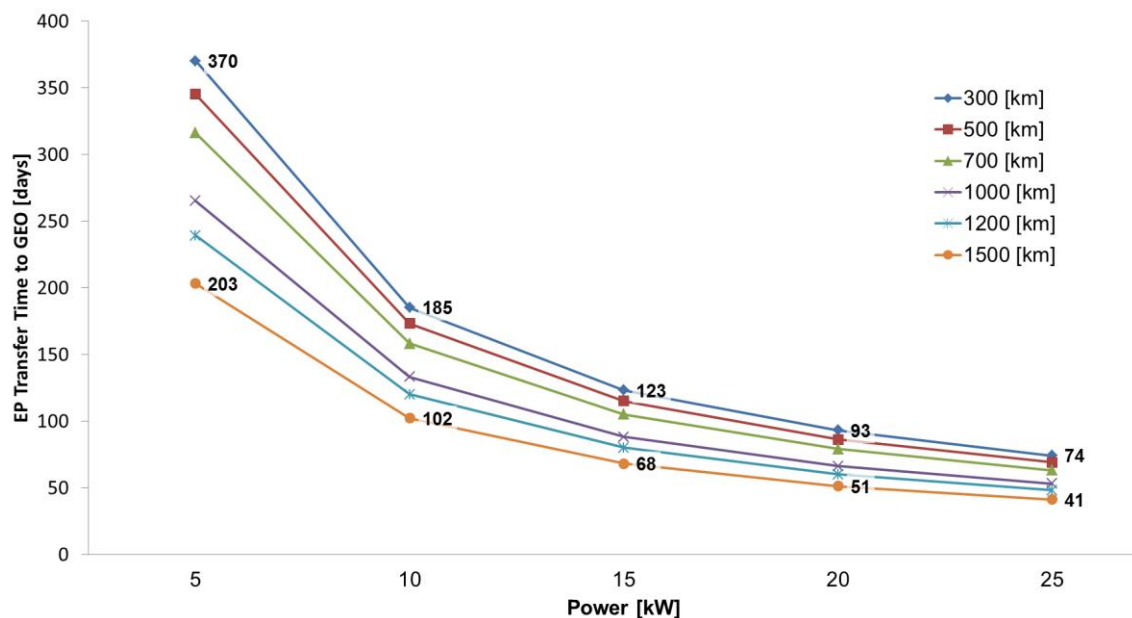


Figure 5: EP transfer times to GEO for different power levels and starting LEO altitudes

Figure 5 shows that in the worst case (5kW, 300km LEO) almost 1 year is required to reach GEO, while in the best case (25kW, 1500km LEO) slightly more than one month is sufficient to reach the target orbit. For a given initial orbit altitude, transfer time curves decrease for growing power levels, but not linearly. In particular for all initial altitude considered the maximum transfer time decreasing can be appreciated around 10kW. Accordingly this is the reference power level used in the following analysis.

B. LEO–MEO Transfers

In analogy with the LEO–GEO transfers, the same analysis is extended to LEO–MEO transfers. Results are shown in Figure 6 and Figure 7.

From Figure 6, it results that from 1020 to 1557kg can be delivered in MEO, while the chemical case only allows for 187–249kg in the same orbit. Thus with the EP configuration there is an average gain of 56% of the initial mass delivered in GEO. In this case the propellant mass required is higher than the LEO–GEO case due to the high inclination change required and, as a consequence, the high efficiency of the EP system allows for a higher propellant mass saving percentage.

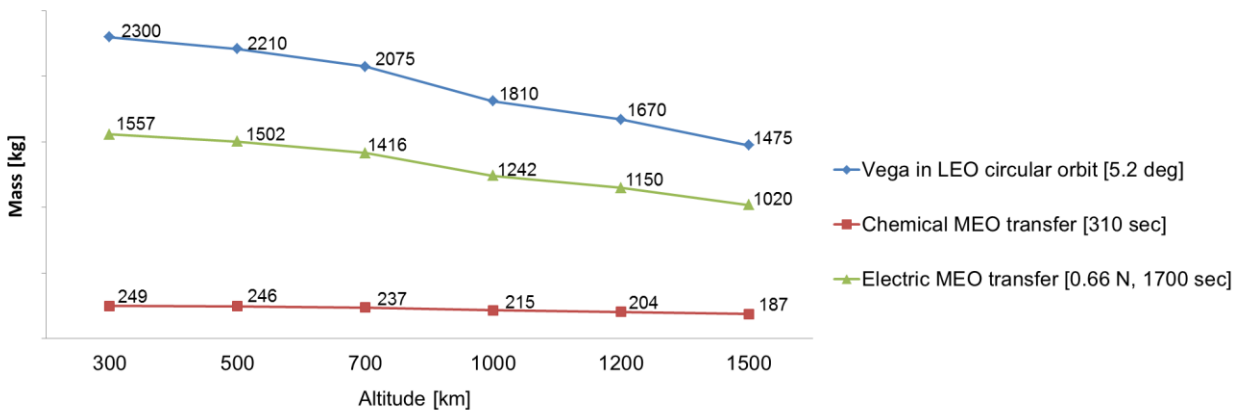


Figure 6: Final mass in LEO (Vega performance) and MEO (chemical and electric comparison)

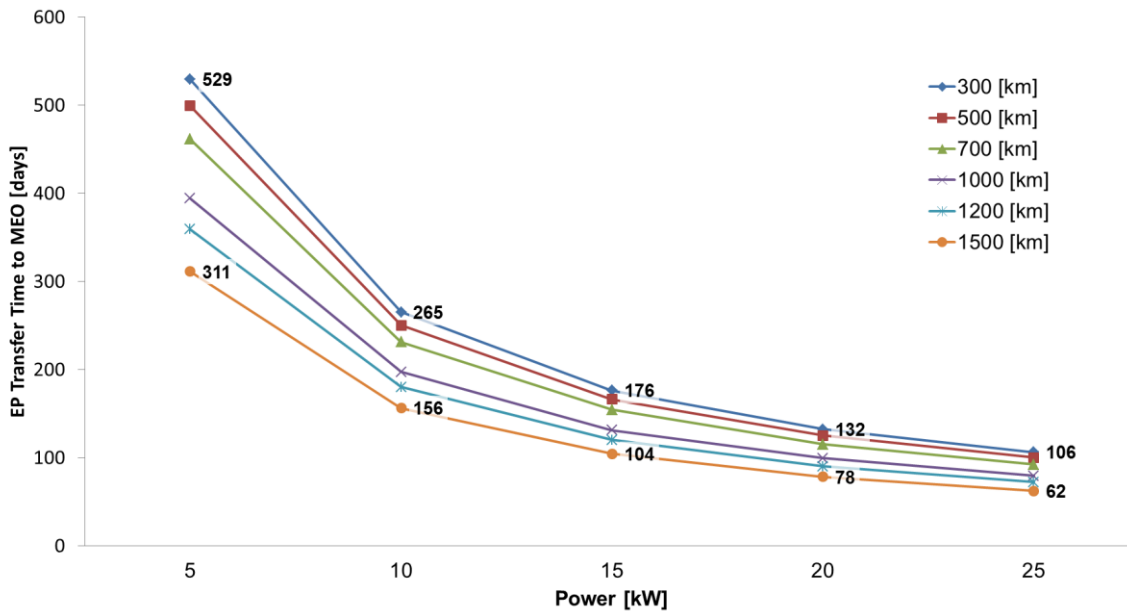


Figure 7: EP transfer times to MEO for different power levels and starting LEO altitudes

Figure 7 shows that in the worst case (5kW, 300km LEO) almost 1.5 year is required to reach GEO, while in the best case (25kW, 1500km LEO) slightly more than two months are sufficient to reach the target orbit. As in Figure 5, transfer time curves decrease for growing power levels, but not linearly. Also in this case the maximum transfer time decreasing can be appreciated for a power around 10kW. As a consequence also for this scenario this power level is used as reference in the following analysis.

By means of EP it is possible to transfer a large mass into this target orbit, making it feasible to raise a typical navigation satellite of the Galileo constellation. This will be shown more clearly in the next section (see Figure 11), where the net payload mass is separated from the EP subsystem mass: using electric propulsion from less than 700km, a payload mass larger than 850kg can be delivered to Galileo orbit (thus allowing also for some margin, Galileo satellites typical mass is around 700kg).

C. Thruster Lifetime

A crucial aspect to take into account when designing an EP transfer is the actual thruster firing time (total mission time without eclipse periods). Considering the candidate thruster lifetime (see Table 1), low-thrust transfers have to last no more than 312 days.

For LEO-GEO transfers here analyzed this condition is always satisfied, while for LEO-MEO transfers there are some scenarios where an additional thruster unit has to be added to reach the total firing time required. Figure 8 shows the actual firing time for LEO-MEO transfers compared to HT5k lifetime. It is shown that starting from an altitude lower than 1200km, for a power level of 5kW, a single thruster unit is not sufficient to accomplish the target mission.

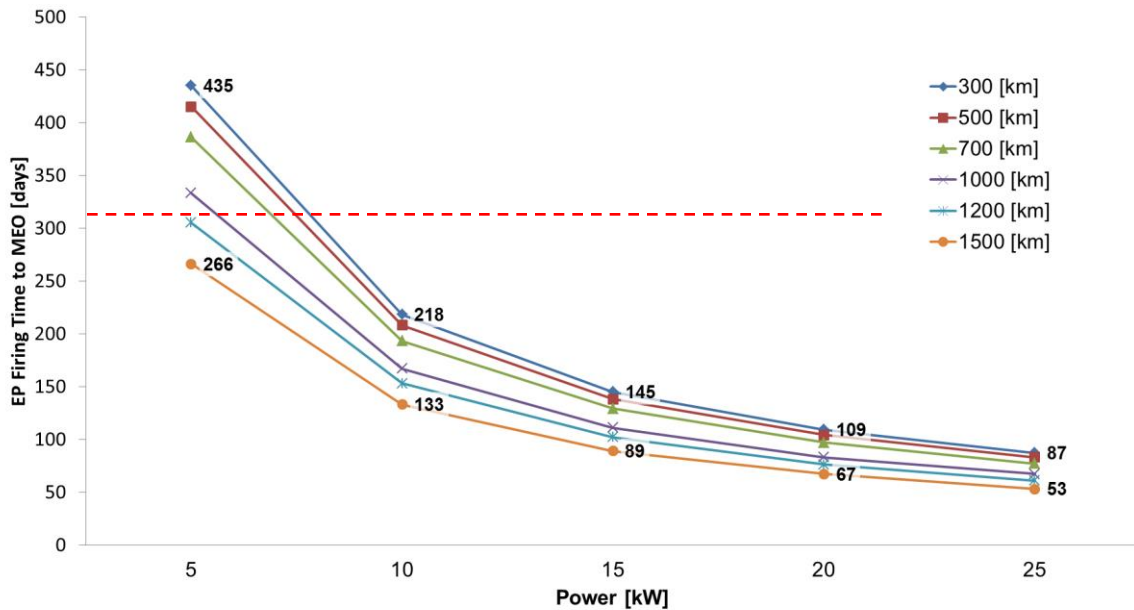


Figure 8: EP firing times to MEO for different power levels and starting LEO altitudes; red dashed line indicates HT5k lifetime

V. Launcher/Spacecraft Configuration Options

Once defined the final mass that can be transported to the target orbit, the effective payload has to be determined. Actually part of that mass is constituted by the spacecraft subsystems, including the electric propulsion system and the power generation system. The mass of the EP subsystem, in particular, depends on the adopted spacecraft configuration. Two different options are considered in our analysis:

- 1) an EP system directly integrated in the satellite: in this case it cannot be jettisoned when target orbit is reached, but it could be used at the satellite end-of-life for deorbiting maneuvers.
- 2) a separate EP module: in this case the module would act as Vega 5th stage and is separated from the satellite at the end of the orbit raising phase.

A schematic representation of the two options is displayed in Figure 9, while split mass budgets are reported in Table 3.

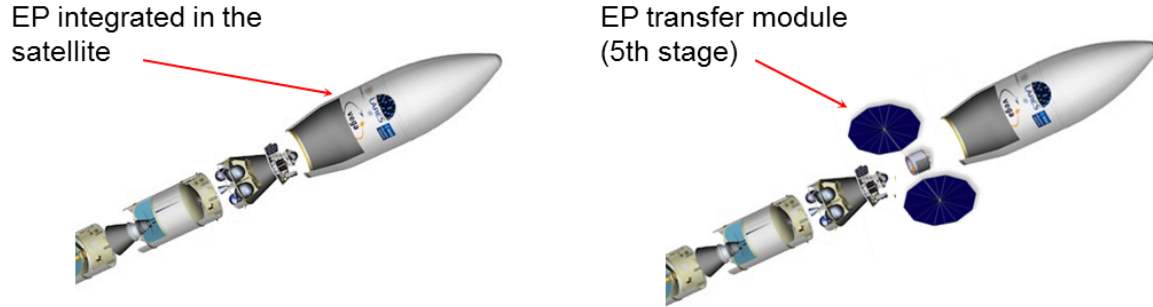


Figure 9: Possible launcher/spacecraft configurations for EP orbit raising

The first option is more effective in terms of useful payload brought into target orbit, but it has a lower flexibility because an integrated EP subsystem heavily affects the whole satellite design. The second option, instead, is much more flexible in this sense, as the electric propulsion stage is by all means part of the launcher and the satellite is just a black-box on top of it. The satellite manufacturer has only to know interfaces and allowable mass and size and then the satellite design can proceed independently.

In this analysis the Vega's 4th stage, AVUM, is always kept as part of the launcher. An interesting possibility, not considered in the present analysis, would be to integrate the electric propulsion subsystem within the AVUM, obtaining a single separate EP transfer module. Such a solution is likely to be lighter than 'option 2' previously described (thus capable of transporting a larger payload) maintaining the very same flexibility.

Electric propulsion subsystem: Mass Estimation (option 1)		
Tank	100	kg
PPU	50	kg
Harness	10	kg
Xe Flow Control	6	kg
Pre Unit, BPRU, Electrical FU	6	kg
Thruster and cathode	20	kg
Electric propulsion subsystem Total Mass	192	kg

EP Transfer Module Mass Estimation (option 2)		
Structure + shield (25% of module mass)	108	kg
Solar Arrays (150 W/kg)	67	kg
AOCS (6% of module mass)	26	kg
C&DH (5% of module mass)	21	kg
Thermal (2% of module mass)	8.5	kg
SCS (2% of module mass)	8.5	kg
Electric propulsion subsystem	192	kg
EP Transfer Module Total Mass	431	kg

Table 3: Mass budget for EP subsystem and EP transfer module (10kW case)

A considerable percentage of the EP transfer module mass is constituted by the aluminum shield necessary to protect the payload and the module itself from heavy radiations encountered during the slow transition through the Van Allen belts. With this regard, it is especially interesting to analyze the mass budget of EP transfer module at different power levels. Table 3 refers to a power level of 10kW, but it can be shown that the overall mass is not much different if we change power level, at least in the range considered. The reason is that there are two opposing effects counterbalancing each other: at higher power levels a larger solar array mass and more thruster units are required, but at the same time the mass of the protecting shield gets lower as the time spent in the Van Allen belts is reduced.

Figure 10 and Figure 11 show the effective payload mass that can be delivered in GEO and in MEO in the two configurations described together with a chemical propulsion comparison. The chemical propulsion system dry mass is estimated starting from the AVUM propulsion system characteristics and is assumed to be around 230kg. It is worth noting that no payload can be brought in a Galileo-like orbit launching with Vega if chemical propulsion is adopted for the orbit raising; values close to zero for the payload mass mean that all the available mass has to be used by the spacecraft subsystems to reach the target orbit and sometimes this may not be enough anyway (negative payload values).

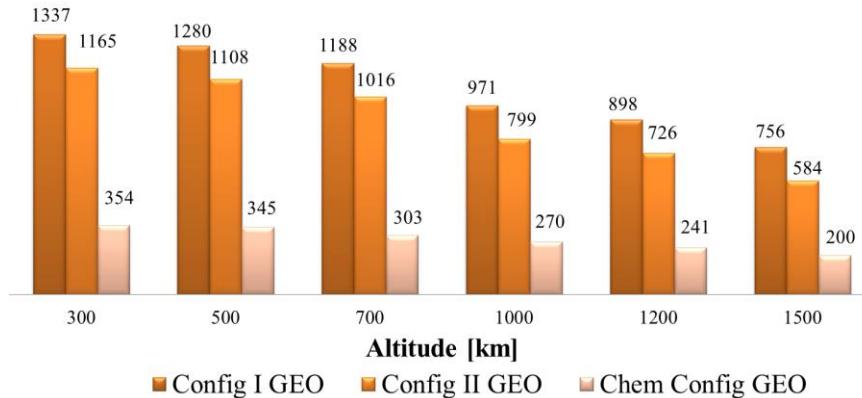


Figure 10: Payload mass in GEO (Vega launcher, LEO-GEO transfers)

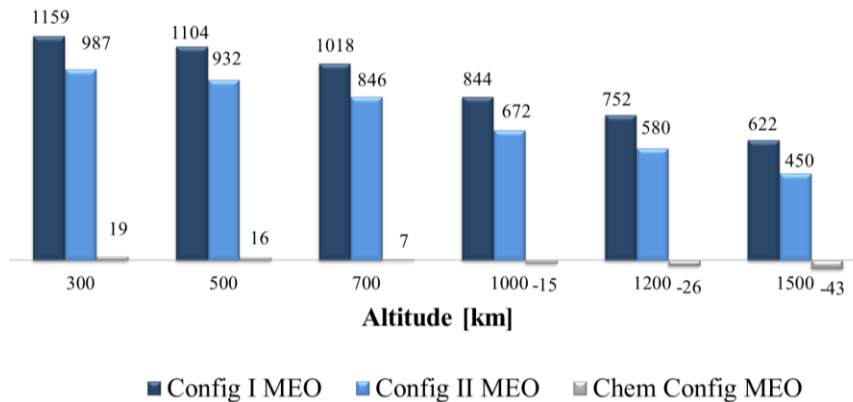


Figure 11: Payload mass in MEO (Vega launcher, LEO-MEO transfers)

VI. HET alternative propellants

An further interesting option that is now being investigated by space agencies with the aim of further reducing missions costs, is to adopt alternative propellants for the EP system. HET usually work with Xenon which is a rare gas and is becoming more and more expensive, a reference cost is around 1000–1200\$ per kg. Among the possible choices, Krypton sounds as the most attractive one¹². The main reasons are that it does not pose special risks; it does not require substantial re-design of the propellant management assembly; it does not exactly match Xe performance but it comes at a price that can be about three times lower. This means that for a mission consuming about 700kg of propellant, more than 500k\$ could be saved. This figure is even more relevant if we consider that Vega's capabilities are going to be enhanced in the near future (allowing for more ambitious transfers) and that Kr might be also used for the station-keeping maneuvers.

A reasonable price for a 5–10kW propulsion system which can be competitive on the market seems to be around 4M\$. Then, by using Kr, it is possible to reduce the overall cost of a sound 10-15%. Of course, when operating in the same V-I conditions, a Hall thruster fed with Kr has the disadvantage of providing a lower thrust level. This has

a direct impact on the transfer time which gets longer, although the total propellant consumption does not grow up accordingly since the specific impulse with Kr is higher. Kr is 40% lighter than Xe, thus resulting in a 30% specific impulse increase for the same discharge voltage; actually, the net benefit is slightly lower because of a lower overall thruster efficiency is obtained when HET is operating with Kr instead of Xe.

Figure 12 shows the benefit in terms of cost and final payload mass if Kr is used for a transfer from LEO–MEO orbit. The HT5kW discharge voltage when operating with Kr is assumed to be higher (500V), to avoid an excessive reduction of the overall thruster efficiency. Under this condition, from theoretical scaling laws, a specific impulse of 2500s and a thrust of 0.39N are expected. This brings a further saving in terms of propellant mass, but also leads to transfer times about 80% longer.

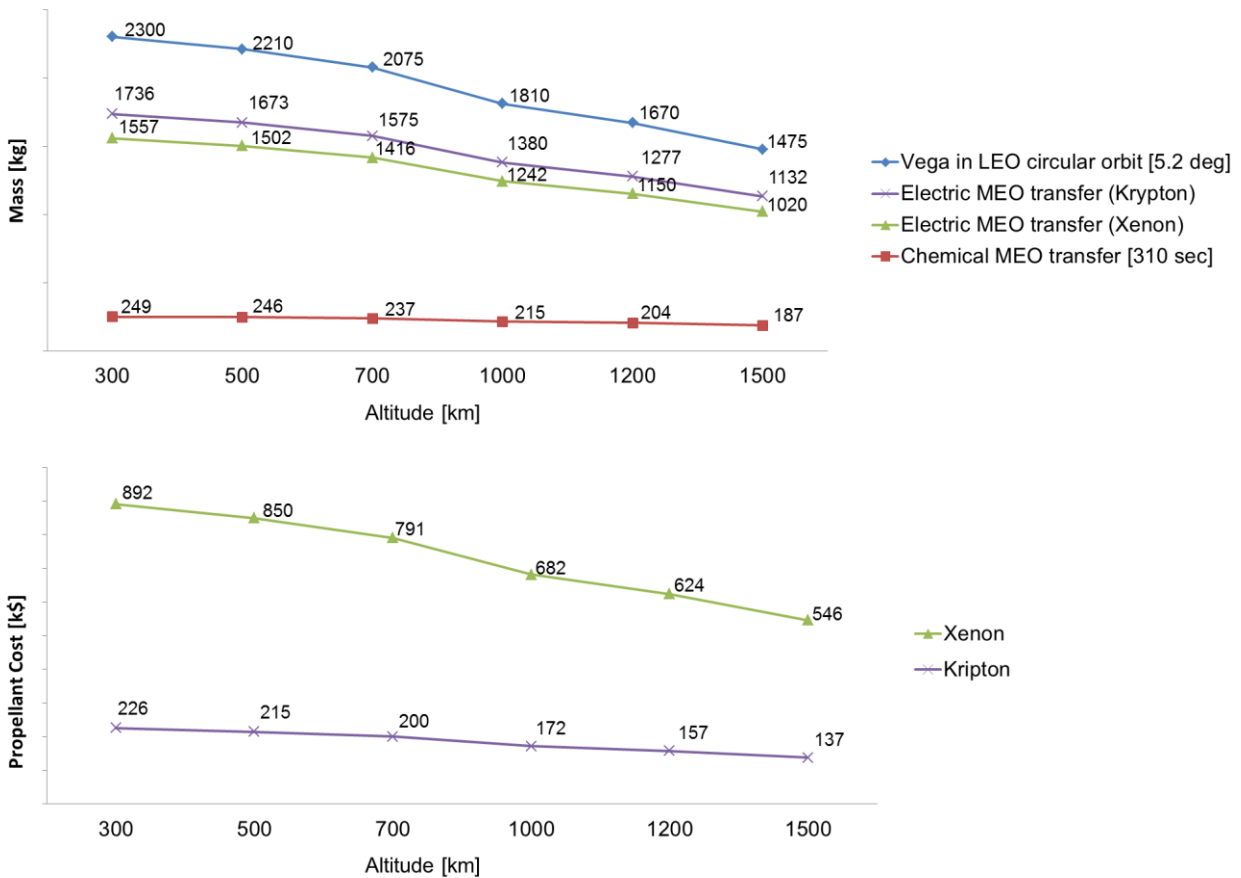


Figure 12: Comparison between Xe and Kr for an orbit transfer from LEO–MEO. Final mass (up), propellant cost (bottom)

VII. Conclusions

The analysis here presented shows a number of interesting possibilities that open up when coupling Vega launcher with an electric–propulsion–driven upper stage. By means of EP, a small launcher like Vega can extend its capabilities and successfully place mid–size spacecraft in higher orbits. Up to nearly 1350kg of payload can be placed in GEO, a mass which is consistent with the present size of small geostationary satellites. In the same way, there is the possibility of bringing payloads in excess of 1–ton in the orbit used for Galileo satellite navigation constellation. Launching from Kourou, this orbit is even harder to reach (in terms of Δv) than the GEO because of its high inclination.

Of course the drawback of using EP for LEO–GEO and LEO–MEO transfers is the transfer time, which last always at least a few months. However, increasing the power level and using a cluster of thruster the transfer time can be consistently reduced. An interesting result emerging from this study is that the total mass of the electric propulsion subsystem is scarcely dependent on the power level (at least in the 5–25 kW range here considered),

because higher power levels allow for lower transfer times and, as a consequence, for a significant reduction of shield mass necessary to protect the spacecraft from radiation during the orbit raising maneuver.

Last, two different options have been analyzed to install an EP subsystem on Vega launcher: directly integrated in the satellite, or as an independent 5th stage. Both solutions present pro and cons, but it would be probably more advantageous to integrate the EP subsystem in Vega's 4th stage (AVUM) in order to obtain the maximum flexibility at a minimum structural mass cost.

To further reduce the mission costs, a convenient option could be to use Kr instead of Xe as propellant. Our analysis shows that adopting this solution, at the price of slightly reduced performance, it is possible to save 0.5M€ per launch.

References

¹ De Tata M. et al., “*SGEO Development Status and Electric Propulsion Subsystem Overview*”, ID:2554314, Space Propulsion Conference, Bordeaux, France, 2012

² Casaregola C. et al., “*A VEGA Dedicated Electric Propulsion Transfer Module To The Moon*”, IEPC–2007–306, International Electric Propulsion Conference, Firenze, Italy, 2007

³ Finocchietti C. et al., “*Low–thrust Transfers for the VEGA Upper Stage*”, IAC–12–D2.3.12, Napoli, Italy, 2012

⁴ Pergola P. et al., “*E–Plus: Electric Propulsion Equipped Launcher Upper Stage*”, ID: 2394190, Space Propulsion Conference, Bordeaux, France, 2012

⁵ Boeing, “*Boeing 702SP Spacecraft*”, 2013

⁶ Duchemin O. et al., “*Next generation of thruster module assembly (TMA–NG) : advanced solution for electric orbit raising and station keeping*”, ID: 2394266, Space Propulsion Conference, Bordeaux, France, 2012

⁷ Arianespace, “*Vega User’s Manual*”, Issue 3, 2006

⁸ “*Birth of the European Satellite Navigation Constellation – Galileo In–orbit Validation*”, Galileo Factsheet, ESA

⁹ J. Kechichian J., “*The reformulation of Edelbaum's low-thrust transfer problem using optimal control theory*”, AIAA–92– 4567–CP, AIAA/AAS Astrodynamics Conference, Hilton Head, South Carolina, 1992

¹⁰ Edelbaum, T. N., “*Propulsion Requirements for Controllable Satellites*”, American Rocket Society Journal, Vol. 31, 1961, pp. 1079–1089

¹¹ Chobotov, V. A., *Orbital Mechanics*, AIAA Inc., Reston, Virginia, 2nd ed., 1996

¹² Nakles M.R. et al. “*A Performance Comparison of Xenon and Krypton Propellant on an SPT–100 Hall Thruster*”, IEPC–2011–003, International Electric Propulsion Conference, Wiesbaden, Germany, 2011