All EP Platform: Mission Design Challenges and Subsystem Design Opportunities

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A new vision within OHB System AG is aimed towards the development of a full electric propulsion platform. With the use of only electric propulsion on station, OHB-SY (as a Prime) and OHB-SE (as a propulsion subsystem and Attitude and Orbit Control system supplier) will push the SGEO platform one step further. The small launchers market and available electric propulsion (EP) technology allow for a cost efficient mission design with a range of S/C masses and orbit raising time. Depending on the thrust from a particular EP solution, different transfer times and propellant masses are foreseen. The main challenge in the implementation of EP for all spacecraft maneuvers is securing the orbit raising phase in terms of duration and spacecraft exposure to radiation during this phase. All EP platforms open an opportunity for the development of pointing mechanisms that will meet the requirements for high pointing accuracy, thermal transfer and thruster-spacecraft interaction while being capable of a large range of motions. In this paper some results of the orbit raising strategy feasibility study for a number of injection orbits will be presented and discussed. In addition, an overview of the electric propulsion pointing mechanism requirements with respect to the all-EP mission demands will be given.

Acronyms and Abbreviations

BOL	Beginning Of Life	OR	Orbit Raising
CP	Chemical Propulsion	PPP	Public Private Partnership
EOL	End Of Life	P/L	PayLoad
EPPM	Electric Propulsion Pointing Mechanism	S/C	SpaceCraft
EW	East-West	SGTO	Sub-Geostationary Transfer Orbit
GEO	GEOstationary orbit	SK	Station Keeping
GIE	Gridded Ion Engine	SSTO	Super-Synchronous Transfer Orbit
GTO	Geostationary Transfer Orbit	TID	Total Ionizing Dose
NS	North-South		

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I. Introduction

It is well known that electric propulsion can be used for a wide number of space mission types, from scientific such as GOCE, SMART-1, Artemis[1–3] to commercial satellites such as Boeing 70, SSL 1300, Astrium Eurostar 3000, Thales Spacebus 4000, OHB SGEO etc.[4–8] The most common application of EP is station keeping (SK) of geostationary satellites (GEO). The reliability, thrust characteristics and mass saving properties of EP tend to make it a common choice over chemical propulsion (CP) for SK application. CP has been generally used for orbit raising (OR) due to its high thrust with the only exception of Boeings 702HP platform equipped with XIPS 25 for both OR and SK.[5] Various European geostationary built satellites are using EP (Eurostar, Alphabus, Artemis) of which at present none is using electric propulsion (EP) for nominal orbit raising maneuvers. OHB System AG has the opportunity of designing and possibly developing the first European fully-electric propulsion platform. The future platform design is called Electra, and is currently being evaluated against mission requirements, technical constraints and market opportunity. The first launch of an Electra mission is planned for the 2018-2019 timeframe. One of the main challenges will be to find a suitable electric propulsion system fully qualified for the program before the space craft assembly starts. This paper begins with background information on the Electra project, followed by discussions of the impacts of the Electra mission design on possible launch strategies, electric propulsion thruster types, and the electric propulsion pointing mechanism (EPPM). These discussions include optimization within the constraints of the predefined limits on transfer time and spacecraft (S/C) mass, as well as the increased radiation exposure. The paper concludes with a discussion of future developments in EP and the possible impacts of its increased usage and expanded application.

II. Electra: PPP between ESA and SES

The Electra PPP (Public Private Partnership) project was initiated when SES S.A. (Euronext Paris and Luxembourg Stock Exchange: SESG) decided to participate in the Artes-33 programme of the European Space Agency (ESA) in 2012. Under the Electra programme SES and ESA will establish a public-private partnership with the goal of developing a fully-electric propulsion European satellite platform. SES will lead the satellite design definition phase by providing customer system requirements and working closely with OHB System AG of Bremen, who will act as the prime contractor to SES. OHB Sweden will be responsible for the Electric Propulsion Subsystem (EPPS) and Attitude and Orbit Control Subsystem (AOCS) including thruster assembly, propellant supply assembly, Xe tank and pointing mechanism. The project is being currently in the proposal phase, with supplier selection expected in the beginning of 2014.

III. Mission Analysis Results

A discussion of the mission analysis can be found in this section with a focus on orbit transfer in lieu of station keeping in light of the well-known application of EP to the latter. Three transfer strategies have been initially chosen for OR: GTO to GEO, SSTO to GEO and SGTO to GEO. This approach is based on free-apogee insertion into an elliptical orbit, followed by orbit topping. For each orbit a representative launch mass is used in the determination of the required delta-v for transfer to the final GEO. Depending on the delta-v, both the transfer duration and propellant mass can be determined. Orbit simulation results were used as inputs to a radiation analysis, the results of which influence the technical and financial budgets.

A. Trajectory

The GTO to GEO trajectory which is shown in Figure 1 has its apogee close to the GEO-belt and thus only requires orbit topping and raising of the perigee to reach the final orbit. For this trajectory, restraint of the apogee height can avoid unwanted GEO belt crossings, but not without a certain propellant penalty.

The SSTO to GEO trajectory shown in Figure 2. has its apogee outside the GEO belt. The orbit topping involves raising the perigee while lowering the apogee, and also ensuring that the crossings of the geostationary belt occur at inclinations larger than the geostationary slots to minimize the residence time. Although particular attention is required at the end of this transfer to simultaneously raise perigee and lower apogee to not only place the S/C into a specific orbit, but a specific location within that orbit (i.e. the final GEO slot), the excess initial energy of the high apogee altitude can be used for plane changes and perigee

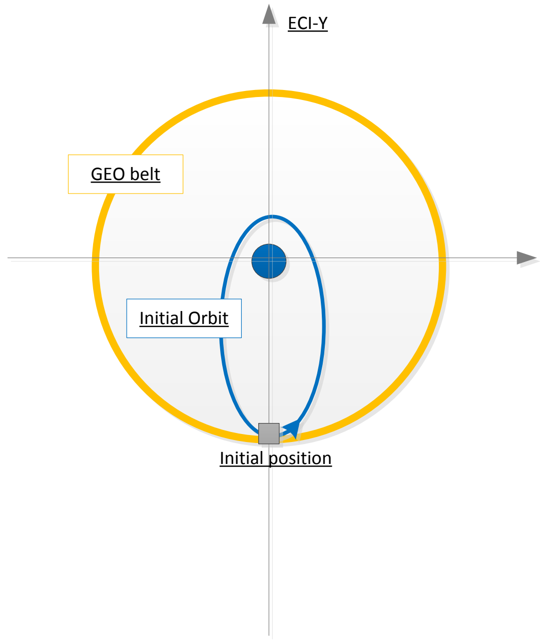

Figure 1. GTO-GEO strategy.

raising, lowering the total required delta-v.

The Sub-GTO (SGTO) to GEO trajectory, the beginning and final orbits of which are shown in Figure 3, includes orbit topping and OR of both perigee and apogee, but has the added benefit of more easily avoiding GEO belt crossings as they only come into consideration towards the end of the trajectory. The presented trajectories are particularly interesting for missions seeking reduced launch costs by using small launchers only capable of lifting small and medium-sized satellites.

B. Transfer Duration and Delta-V

S/C launch masses in the range of 2 to 3 tons were used for calculations of mission parameters. Launcher capabilities have been used to define initial apogee altitudes at a fixed perigee altitude of 250 km, to calculate the required EP delta-v for different launch masses. The resulting delta-v figures range from 2.2 km/s to 3.5 km/s, depending on the launch mass. The transfer duration and propellant mass requirement corresponding to each delta-v calculation from the mission analysis can be estimated using typical thrust and specific impulse values for a Hall Effect Thruster (HET) and a Gridded Ion Engine (GIE). Consideration of both technologies enables accommodation of a wider range of S/C mass (2000 3000 kg) while still abiding by the commercially imposed limit on the delay prior to satellite operation. In the case of the launchers listed above and observing the limit on transfer time, use of the HET results in possible accommodation of up to 2 tons S/C dry mass. Use of the GIE and its reduced thrust (as compared to the HET) will result in a lower possible S/C mass accommodation to meet the transfer time limit, but its higher specific impulse (Isp) (as compared to HET) reduces the propellant mass requirement, allowing injection into a higher orbit and reducing the required delta-v. If the transfer time is allowed to vary within the imposed limit, it is found that ion technology offers lower consumption of propellant and longer transfer duration, whilst a HET will deliver the satellite to the final orbit faster with a propellant penalty. Some discussions on the advantages and disadvantages of both options follow in Section IV.

C. Radiation Exposure

Low thrust transfer, meaning long transfer, brings additional radiation issues to be considered when designing a space mission. The average transfer duration as estimated by a trajectory study ranges from little less than 100 to just above 250 days depending on the EP type and delta-v. The slower the orbit raising the more

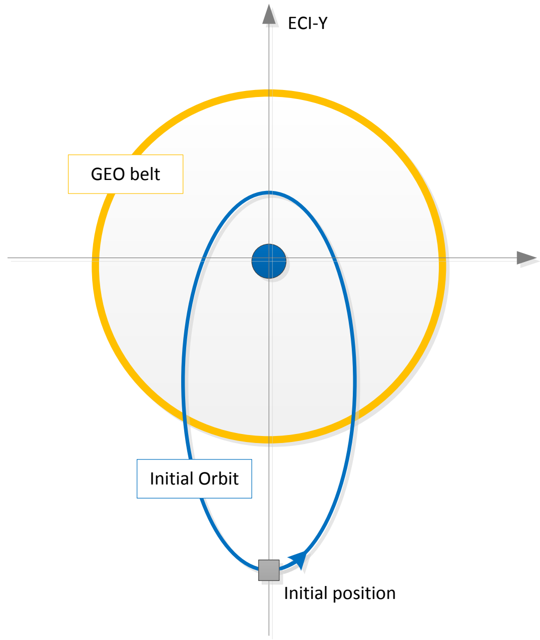

Figure 2. SSTO-GEO strategy.

time will be spent in the proton belts for identical OR trajectories. For the same thrust, however, a judicious choice of orbit transfer can reduce the radiation exposure. Another contribution to the radiation exposure is the solar activity. This is an important factor for all EP missions, since the effect of the radiation belts on the S/C will intensify if it is launched at the start of solar maximum, leading to more rapid satellite decay.[9] The radiation exposure issue is not new to the space engineering world; previous, slow transfers through radiation belts have been performed by SMART-1 under OHB-Sweden prime responsibility. The main danger during the transition is the high energy proton belt located between 5 000- 10 000 km from the Earths surface. Low perigee -250 km/high apogee insertion means that the satellite will spend some time in the proton belts regardless of the apogee altitude. Using the SPENVIS software[10] the equivalent 1 MeV lectern fluence for the short circuit degradation of multi-junction solar cells was calculated for various trajectories and launch masses. The proton belts effect on the spacecraft will be mostly visible on the solar cells. The higher the fluence, the thicker the required cover glass would be to maintain the cells performance. A reasonably thicker cover glass and an optimized transfer orbit can significantly reduce the extent of degradation.

D. Power and Mass Budget

A range of 2 000- 3 000 kg of S/C launch mass for an all EP satellite defines the power available for the payload during the operation. Taking into account solar cell degradation, the power available for the payload is estimated to be around 8 kW (BOL), with a mass of 700 kg, placing it in the same category as current mid-sized telecom satellites with typical launch masses around 5 tons. Thanks to the use of electric propulsion also for the orbit raising phase, the launch mass for Electra can be reduced to the range 2-3 tonnes, thereby placing it within the capabilities of several low-cost launch alternatives.

E. Conclusion

The initial sizing of the propulsion subsystem is an intricate question where mass, launcher selection, radiation exposure and EOL power have to be jointly considered. From the above and with guidance of the end user on weighing the priorities, a general architecture could be established. Preliminary radiation analysis showed that an all-EP mission with a long orbit transfer would have additional radiation requirements for all subsystems compared to conventional 15 year GEO mission.

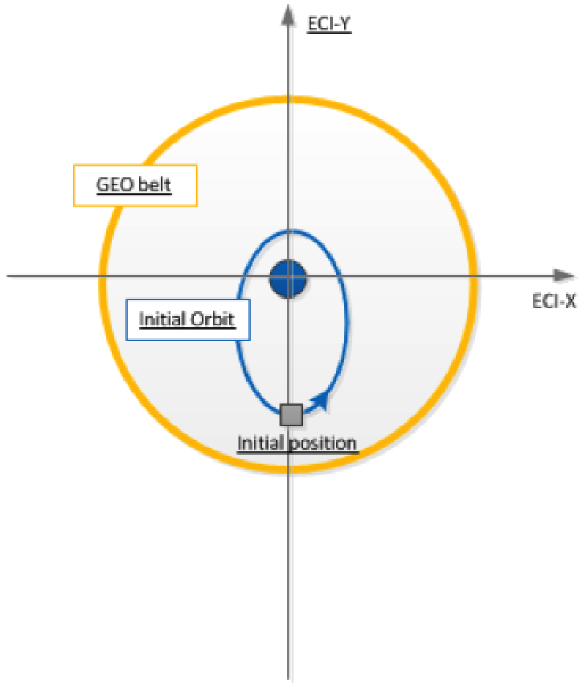

Figure 3. SGTO-GEO strategy.

IV. Electric Propulsion System

This section presents considerations involved in selection of the electric propulsion thruster type, with a focus on the tradeoffs between high thrust and high specific impulse (Isp), and between high qualification levels and high performance levels.

A. High Specific Impulse versus High Thrust

Up to date EP technologies have proved high reliability and robustness. They can be very beneficial in terms of increased payload capacity of a telecom spacecraft. To decrease the transfer time an EP with higher thrust should be applied. Ion thrusters have higher specific impulse and lower thrust compared to Hall Effect thrusters for a fixed power input, and therefore can be suitable for smaller satellites where the power available might be proportionally larger, whereas HETs can be used for heavier satellites. Both technologies have flight heritage of being the only propulsion on the spacecraft. Ion thrusters were successfully used by GOCE and Boeing 702HP satellites, while the heritage of the HET is SMART-1. Ion thrusters represented by XIPS perform OR of a 2000 kg S/C in 80 days. On the other hand HETs have been widely used for SK maneuvers, with tens of HET-equipped GEO satellites on orbit presently. For the time being there is no European EP qualified thruster for a 15 year mission capable of long orbit transfer, making it challenging for the OHB group to have their first satellite ready for launch in the year 2018/2019. Potential HET or GIE candidates must be qualified in lifetime duration and number of cycles. At present, the critical component in GIE systems is the grid assembly due to erosion of the grids, while in HET systems the lifetime is limited by erosion of the ceramic insulation. For both systems a high level of thrust and specific impulse stability throughout the lifetime is required.

B. Mature versus New Development

The use of mature hardware is safe in that it could fulfill mission requirements, but it would add mass, have higher costs and require more room for accommodation of a larger number of required thruster units as compared to newly-developed, high-power alternatives.

C. Conclusion

In spite of the number of existing EP technologies, it is a great challenge to choose one that can be suitable for an all-EP GEO satellite. Apart from the technical trade-off, the qualification strategy should be carefully adapted to the lifetime requirements and schedule. However, OHB Sweden believes that it is possible to deliver an EP subsystem even within a tight delivery schedule.

V. Electric Propulsion Pointing Mechanism

The Electric Propulsion Pointing Mechanism (EPPM) plays a multifunctional role in the propulsion subsystem. It is a supporting structure for the EP thruster assembly and harness, and it is responsible for the pointing accuracy of the thrust vector. It defines the distance between the thruster and the various spacecraft components, the most important of which is the distance to the solar arrays.[11] Designs of existing pointing mechanisms may be unsuitable for the Electra mission as they were designed to support low- and medium-power thrusters during missions where EP was either exclusively intended for SK, or where EP supported OR but with a transfer time less than half of the customer-defined limit for Electra.[12] For an all EP S/c, EPPMs should comply with a higher range of motions, accuracy and lifetime requirements. Let us discuss some major driving factors. This section discusses constraints on the EPPM design.

A. Degree of Freedom

An all-EP satellite assumes that only EP will be used for all maneuvers throughout the mission, consisting of not only OR but also SK, which includes NS and EW SK. Therefore the EPPM must be capable of motion in all planes to avoid a penalty of propellant or time in the case of an EP thruster failure. The clearance distance between the thruster and the solar arrays defines the cant angle, which is driven by the angle of the thruster plume divergence. For example, as HETs have a larger plume divergence angle than GIEs, a larger cant angle must be implemented if the propulsion subsystem is equipped with a HET versus a GIE.

B. Pointing Accuracy

Pointing accuracy influences the EPPM contribution to parasitic torques, and should be kept to a minimum. The larger the parasitic torques, the larger the mass required for other AOCS actuators that must then compensate for the residual torques to ensure sufficient feedback on positioning. For example stepper motors will have to handle parasitic torque from the EPPM in addition to the detent torque required to steady the boom against the spring effect of Xenon lines or electrical harness. If the parasitic torque surpases the load capacity of the stepper motor, the boom actuators will have to be constantly powered to compensate for the lost functionality. These aspects will be jointly traded and traced in several technical notes.

C. Thermal Loads

The use of EP for orbit transfer implies almost constant firing of the thruster for a duration of approximately 100-250 days, during which time the EPPM can experience thermal loads from the thrusters, temperature gradients due to eclipses and radiation exposure. A thermal model will be used in the design of the EPPM, the results of which can be verified by analysis or test. A more detailed analysis might become necessary to estimate shadowing effects or define thruster assembly constraints on various thermal control system designs (e.g. radiators) to compensate for thermal loads.

D. Mechanical Loads

While the actuators need to be robust enough to support the launch loads without clamping, dampers may be required to protect in the launch configuration the fragile parts of the thrusters. This trade-off will involve input from the spacecraft manufacturer regarding definition of the structure, the boom designer regarding the damping and hold down mechanism design, as well as the thruster supplier regarding possible notchings.

E. Radiation Loads

Subsystem survival of a longer exposure to radiation requires additional shielding of the electrical cables. In the already-thickly shielded high voltage cables of the GIE, the additional radiation shielding may compromise compliance with required lower limits on flexibility and upper limits on mass. Design optimization with respect to the challenge of increases radiation exposure requires involvement of many parties, including harness suppliers and test suppliers, in addition to accurate predictions of product performance backed by representative testing.

F. Developments

The type of EP influences both the mechanical and thermal EPPM design while additional requirements come from the mission design. Compliance to most of these requirements entails detailed trade-offs and compromises, some with the support of the AOCS supplier, some with the support of the spacecraft prime or thruster supplier, and all coordinated by the EPPS supplier. Another consequence of the use of high-power EP technology on commercial satellites is a redesign of the power supply unit to take advantage of the lower currents associated with a higher regulated voltage. This allows for smaller cable cross-sections resulting in certain mass savings that can help offset increases in cable mass required for the Electra mission due to increased radiation hardening. Wide application of EP technology will increase the space sectors Xenon demand which, because of the limited amount of annually-produced Xenon, will increase the price per liter. This trend will continue in the long term if EP platforms continue to gain popularity in the commercial space industry. A need for a cheaper fuel or mixed fuel has already begun to be addressed via testing and exploration of feasible alternatives. This long yet exciting and necessary undertaking will require close cooperation between agencies and industry.

VI. Conclusion

A growing family of small commercial launchers offers a delivery of small- to medium-sized spacecraft into intermediate or transfer orbits cheaper and faster than conventional heavy lifters providing final orbit insertion. This will create a niche in the satellite market for S/C capable of orbit transfer while maintaining a mass compatible with these launchers. Optimized transfer strategies will allow for reducing the orbit raising time and radiation exposure inside the proton belts. A trade-off between spacecraft dry mass and permitted transfer duration will affect the choice of thruster technology. The choice of implementing high-power EP will have consequences beyond the propulsion components development and optimization, for example the regulation accuracy of the power supply units will have to be revisited and reassessed. An all-EP mission results in new lifetime and performance requirements on the propulsion pointing mechanism design, which will open opportunities for developing a new family of EPPMs. With the next generation of high power EP coming to the satellite industry it is likely that more platforms will start using EP thrusters as the main propulsion system making it possible to exclude chemical propulsion from the propulsion subsystem and reach the small S/C mass range. It is likely that the next evolution of EP technology will aim at higher thrust and slightly higher power while keeping specific impulse high and subsystem dry mass low. These developments while increasing requirements on solar cell efficiency, can reduce fuel consumption and transfer duration.

References

[1]Drinkwater, M. R. Floberghagen, R. Haagmans, R. Muzi, D. Popescu, A. GOCE:ESA's First Earth Explorer Core Mission. *Proceedings of an ISSI Workshop*, pages 419–432, March 2002.

[2]Racca, G. D. and Rathsman, P. SMART-1 team, SMART-1 mission description and development status. *Planetary and Space Science*, pages 1323–1337, 2002.

[3]Angelopoulos, V. The ARTEMIS Mission. *Space Science Reviews*, 165:3–25, 2010.

[4]Corey, R.L. and Pidgeon, D.J. Elactic Prepulsion at Space Systems/Loral. *Proceedings of The 31st Iinternational Elactic Prepulsion Conference*, (IEPC-2009-270).

[5]Chien, K.R. Hart, S.L. Tighe, W.G. De Pano, M.K. Bond, T.A. and Spears, R. L-3 Communications ETI Electric Propulsion Overview. (IEPC-2005-315), 2005.

[6]Demair, A. and Gray, H.L. Plasma Propulsion System Functional Chain First Three Years in Orbit on Eurostar 3000. *Proceedings of The 3rd Iinternational Elactic Propulsion Conference*, (IEPC-2007-00060), 2007.

[7]De Tata, M. Frigot, P.E. Beekmans, S. Lubberstedt, H. Birreck, D. SGEO Denvelopment Status and Oopportunities for the EP-based Small Eutopian Telecommunicaions Platform. *Proceedings of The 32nd Iinternational Electric Propulsion Conference*, (IEPC-2011-203), 2011.

[8]Huddleson, J. Brandon-Cox, J. Wallace, N. Palencia, J. An Overve of the T6 Gridded Ion Prepulsion System Pre-Development Activities for Alpha-Bus. *Proceedings of The 4th Iinternational Sspacecraft Propulsion Conference*, (ESA-SP-555):2–9, June 2004.

[9]Wertz, J.R. and Larson, W.J. *Space Mmission Analysis and Design.* Microcosm Press, 3 edition, 2003.

[10]BIRA-IASB ESA. The Space Environment Information System Soft ware Package. 1996.

[11]David, Y.O. Hastings, D.E. Marrese, C.M. Haas, J.M. Gallimore, A.D. Modeling of Stuitionary Plasma Thruster-100 Thruster Plume and Implications for Satellite Design. *Jjournal of Propulsion and Power*, 15(2), March-April 1999.

[12]ESA. Eutopian EPPM suppliers. *http://www.esa.int/TEC/mechanisms/*, 2011.