Electra – Full Electric Propulsion Satellite Platform for GEO Missions

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ABSTRACT

In the frame of the Electra program with ESA and SES, OHB develops an innovative geostationary satellite platform fully based on electric propulsion. Thus, the satellite platform can take advantage of smaller launch vehicles or dual launch capabilities, while carrying payload capabilities equivalent to chemical propulsion based mid-sized satellites in terms of power consumption and number of active transponders.

After an introduction into the OHB SmallGEO satellite family and electric propulsion missions in general, this paper describes Electra mission analysis for orbit raising and station keeping as well as the derived electric propulsion configuration together with an outline of the operational concept.

INTRODUCTION

The OHB SmallGEO family of satellites are designed for small and efficient geostationary missions with a launch mass of up to 3.5 tons. The first mission Hispasat 36W-1 providing Telecom services in Ku- and Ka-Band, was successfully handed over to the end-customer after launch in January 2017. Three further telecom satellites, EDRS-C, Heinrich Hertz and Electra as well as the MTG spacecraft's, all based on the SGEO platform are in development at OHB[1].

After the transfer to GEO performed by chemical propulsion, Hispasat 36W-1 fully relies on electric propulsion for all nominal propulsion tasks in GEO including station keeping and relocation[2]. Electra will be the first all electric SGEO platform including the transfer to GEO. Due to propellant mass savings, Electra more than doubles the payload capacity in terms of mass and power compared to full chemical configurations like EDRS at similar launch mass. Due to the low thrust however several months are needed for the transfer to GEO as compared to a few days for CP configurations with related impact on the radiation hardness driven in particular by the residence time in the inner Van Allen Belt[3].

EP technologies suitable for the envisaged GEO applications include in particular Hall Effect Thrusters (HET) and Gridded Ion Thrusters (GIT). HET provide typically about 50-60 mN thrust per kW input power compared to 30-40 mN/kW for GIT, while operating at a specific impulse in excess of 3000s for GIT compared to less than 2000s for HET[3].

Therefore GITs have been used in particular for high Δv deep space missions like Deep Space 1 (1998-2001)[4], Hayabusa (2003-2010)[5], Dawn (2007-2015)[6] and Bepi Colombo (launch planned 2018), as well as Earth missions with low thrust needs like GOCE (2009-2013)[7] and for Stationkeeping in GEO. As a unique example the ARTEMIS mission[8] launched in 2001 is mentioned. After a malfunction of the Ariane 5 upper stage the final orbit raising to GEO was performed by the low power GITs with less than 20 mN thrust (Airbus RIT-10/ Quinetiq T5).

In 2003, the SMART-1 spacecraft started its 13½ months journey from GTO to the Moon[9] using only a single HET (Snecma PPS-1350), which provided 70 mN of thrust.
After the first Air Force AEHF (Advanced Extremely High Frequency) satellite launched in August 2010 experienced an anomaly of the main chemical engine, the orbit raising was completed by means of a 4.5 kW HET system (Aerojet BPT-4000/ XR-5)[10].

On March 2, 2015, the ABS-3A and Eutelsat 115 West B communication satellites were launched in a stacked configuration on Falcon 9 into a super-synchronous GTO. This was the first pair of four full electric propulsion satellites (Boeing 702SP platform) based on GIT (XIPS-25). 7 month later on October 1, 2015, the heavier S/C Eutelsat West B with a launch mass of ~2.2 tons has completed the ascent from GTO to GEO[11]. The second pair Eutelsat 117 West B and ABS-2A followed on June 15, 2016. Eutelsat 117 West B satellite, has entered into full commercial service 7 month later on January 16, 2017[12].

On June 1, 2017, Eutelsat 172B was launched on Ariane 5 into a standard GTO. The first full electric propulsion satellite (Airbus E3000) based on HET (SPT-140) with a launch mass of ~3.5 tons reached GEO already after four month orbit raising[13].

The increased thrust-to-power ratio make HETs in particular attractive for GEO missions, where the transfer time is still an important constraint, while the Δv need is limited.

NOMENCLATURE

Acronyms

AIT	= Assembly, Integration and Testing
AOCS	= Attitude and Orbit Control System
BOL	= Beginning Of Life
CGT	= Cold Gas Thruster
EMI	= Electro-Magnetic Interference
EOL	= End Of Life
EP	= Electric Propulsion
EPPS	= Electric Propulsion Subsystem
GEO	= Geostationary Earth Orbit
GIT	= Gridded Ion Thruster
GTO	= Geostationary Transfer Orbit
HET	= Hall Effect Thruster
PPU	= Power Processing Unit
S/C	= Spacecraft
SSTO	= Super Synchronous Transfer Orbit

Greek symbols

Δv	= velocity increment

ELECTRA OVERVIEW

In late 2012, ESA, SES and OHB System have established the Electra program, a public-private partnership aimed at developing a full-electric propulsion small/medium sized satellite platform. Specifically, the project aims to develop, implement, launch and commercially operate an innovative geostationary satellite platform that uses electric propulsion for transfer into geostationary orbit as well as for orbit station keeping. Thus, the satellite platform can take advantage of smaller launch vehicles or dual launch options, while carrying payload capabilities equivalent to current mid-sized satellites in terms of power consumption and number of active transponders.

OHB System is acting as satellite prime and is leading the Electra platform development under a contract with SES, who is defining the customer requirements. SES is the mission prime contractor towards ESA. The industrial organization led by OHB System is composed of Luxspace, responsible for the Command Telemetry and Ranging subsystem, and OHB Sweden, responsible for Attitude & Orbit Control System (AOCS), and the Electric Propulsion Subsystem (EPPS), already proved to be effective during the SGEO program.

The Electra development is currently in phase CD. In late 2017 the Platform Accommodation Review (PAR) was successfully completed. Platform CDR is planned for end of 2018. The launch date for the first Electra mission is planned for 2021.

The Electra Satellite platform provides the following key characteristics:

- Payload mass up to 800 *kg*,
- Payload power consumption of up to 10 *kW*,
- Lifetime of 16 years (including orbit raising),
- Satellite launch mass in 3 *t* class,
- Launchers include Ariane 5 (lower passenger), Soyuz CSG, Falcon 9 (Tier 1 & 2) among others,
- Three-axes attitude stabilization relies on a set of star trackers and reaction wheels.

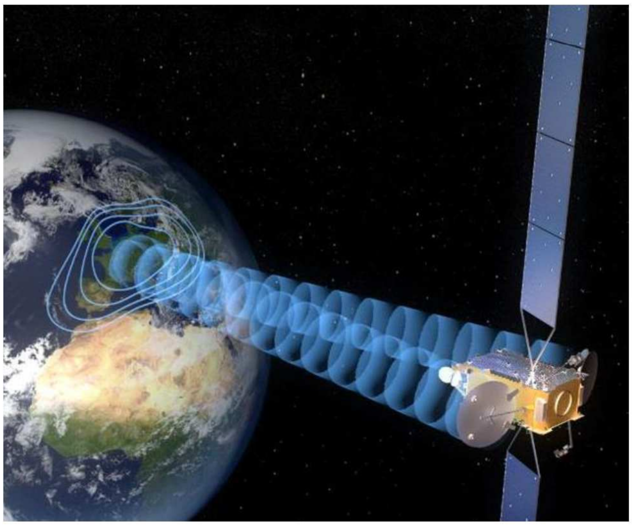

Figure 1 Electra satellite in GEO (artist impression)

ELECTRA MISSION DESIGN

On Electra all propulsive tasks are performed by the Electric Propulsion Subsystem (EPPS) including the transfer phase to the GEO, the final station acquisition and station keeping in GEO as well as repositioning and finally the disposal to the graveyard orbit at end of life. In addition, momentum management during all those phases relies on the EP thrusters. The system is supported by a set of Cold Gas Thrusters (CGT) for de-tumbling of the satellite after launcher injection and to provide attitude control in satellite safe modes when reaction wheels are not available.

Transfer to GEO

The orbit raising manoeuvre imposes design parameters which drive the dimensioning of the satellite in particular in terms of thrust, tank size, power system and radiation hardness. A major advantage of performing the GTO Transfer with EP is the high flexibility of the system to accommodate a broad spectrum of different launch and transfer scenarios ranging from sub GTO up to super-synchronous injections. Launcher overcapacities can be used to minimize transfer time by means of increasing the apogee respectively lowering inclination and increasing the perigee. In Phase B a range of injection orbits from sub-GTO to SSTO have been considered with inclinations up to 28 deg and Δv up to 3.5 km/s [14].

The transfer trajectory is mainly driven by the central gravity field of the Earth, but several other orbital perturbations might significantly modulate the time and even the effort (Δv) devoted to bring each satellite to its corresponding operational slot, i.e.

- Higher-order harmonics of the Earth gravity field;
- Sun and Moon gravity pull (as third-body attractors);
- Sun and Earth radiation pressure;
- Atmospheric drag (only significant below ~900 km).

In order to limit the external disturbance torques at low altitude to not exceed the actuator capacity, a fixed attitude is suggested until the perigee is above a threshold altitude e.g. of 800 km. This approach leads to a small Δv penalty, which is considered a good baseline in the overall trade off between actuator design and operations efforts.

As a design constraint, it is considered to switch EP off during eclipse in order not to oversize the power subsystem. The overall time spent in eclipse can be minimized between zero and a few percent of the transfer time based on the daytime selection of the launch window (Figure 2)[15].

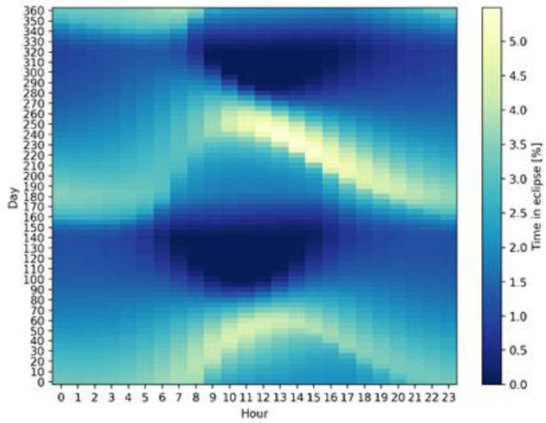

Figure 2: Launch window analysis: total time spent in eclipse (% of transfer time) as function of Hour of the day for each day of the year for Ariane 5 launch

In particular, the longest eclipses are registered for launch dates around the equinox and at midday, whereas the shortest eclipses (or even no eclipse) are obtained for launches around the solstices. Typically the apogee is Sun pointing leading to short eclipses around perigee during the first weeks of the transfer for equinox seasons (black area in Figure 3). During Sun phases, the offset angle between the normal to the solar panels and the satellite-Sun direction needs to be minimized. This is achieved by rotating the S/C around the thrust axis combined with Sun tracking of the solar wing mechanisms. The S/C slew acceleration and rate is however limited by the capability of the reaction wheels, which needs to be considered in the analysis[15].

The classical GTO orbit is characterised by having the apogee in GEO. Previous analyses[14] considered the introduction of a constraint on the evolution of the apogee altitude in order to limit the number of crossings with the GEO belt. However, this approach was found to introduce a significant penalisation in terms of Δv. Therefore the current transfer strategy[15] allows to raise the apogee above the GEO belt as shown in Figure 3 for a typical transfer profile after an Ariane 5 launch. In this case some crossings occur (about 10). Future work will evaluate the avoidance of crossings with the GEO belt. Previous work on the topic has shown that this kind of constraint introduces a penalisation, which is typically much smaller than constraining the apogee[16].

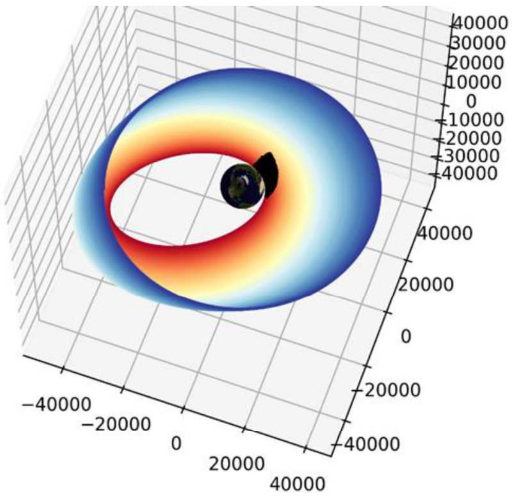

Figure 3 GTO transfer for Ariane 5 launch

The total Δv need for a time optimized low thrust strategy considering above constraints for an Ariane 5 standard GTO launch amounts to 2.2 km/s compared to slightly less than 1.5 km/s for an impulsive transfer scenario. Based on a thrust level of 0.5 N and a launch mass of 2.5 to 3 tons the total transfer time arrives around 4 to 5 month for a worst day of the year launch.

Station Keeping Requirements

The orbit of a satellite in GEO is affected mainly by the following three dynamic effects:

- Earth gravity harmonics, i.e. higher than central terms in the gravitational field of the Earth;
- Gravity pull from third bodies, remarkably Sun & Moon; and
- Solar Radiation Pressure (SRP).

Analysis of these perturbations lead to the need of controlling the satellite both in North/South (inclination vector) and in East/West (eccentricity vector & longitude). The driver for the Δv budget is the inclination control (about 95 %), while the manoeuvre frequency is driven by the longitude and eccentricity control within a few days. Compared to orbit raising from a standard GTO, the station keeping Δv amounts to about 1/3 up to 1/2 of the total budget considering a lifetime of 15 respectively 20 years in GEO.

Electra EP-Thruster Configuration

The Electra design is mainly driven by the optimization of cost, performance and reliability figures. The Electra EP architecture is thus the result of a trade to maximise the satellite performance in terms of payload capacity, transfer time and on station life while minimising the systems and operations complexity. EP systems cost & complexity are in particular driven by the number of thrusters. The Hispasat 36W-1 configuration allows full orbit control and momentum management based on only four thrusters plus another four in cold redundancy, mounted to the S/C body without the need of any mechanism[16]. For Electra the orbit raising requirements determines the design towards higher power thrusters, which even more emphasize the need to minimize their number. To still fulfil above orbit control requirements, mechanisms have been introduced to a flexible architecture with only two thrusters plus two redundant mounted on two articulated booms as shown in Figure 4.

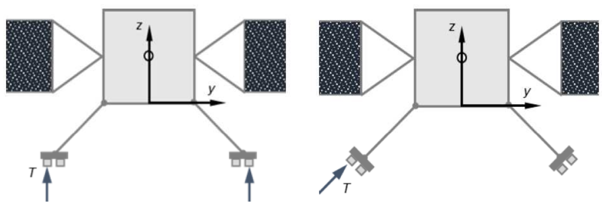

Figure 4 Electra EP thruster configuration schematic: Orbit Raising (left), Station Keeping (right)

During orbit raising two thrusters are operated simultaneously, one on each boom with the thrust directed towards the Z axis. During station keeping only one thruster is operated at the same time with the thrust vector directed mainly towards the CoG. This tilted configuration of the EP thrusters generates a strong cross coupling for any Station Keeping manoeuvre. First and mainly, any normal (North/South) manoeuvre will always produce a radial component towards Nadir (Figure 5). Second, in order to generate tangential impulses it is needed to take the thrust vector out of the YZ body plane. This angle β is small, since the effort for tangential control is significantly smaller than the one needed to control the inclination vector.

Station Keeping Strategy

Based on the orbit disturbances and HISPASAT 36W heritage a combined strategy for N/S and E/W Station Keeping was derived considering the couplings generated by the different thrusters configuration as follows:

- In order to keep every manoeuvre as short as possible with the intention to minimise dispersions, there shall be two manoeuvres per day, one at (approximately) each ascending & descending node, with the objective of keeping the inclination vector.
- A weekly Station Keeping cycle made of up to 6 thrusting days and at least 1 day without thrust for

orbit determination. The Δv in normal direction is distributed more or less equally between the North & South manoeuvres.

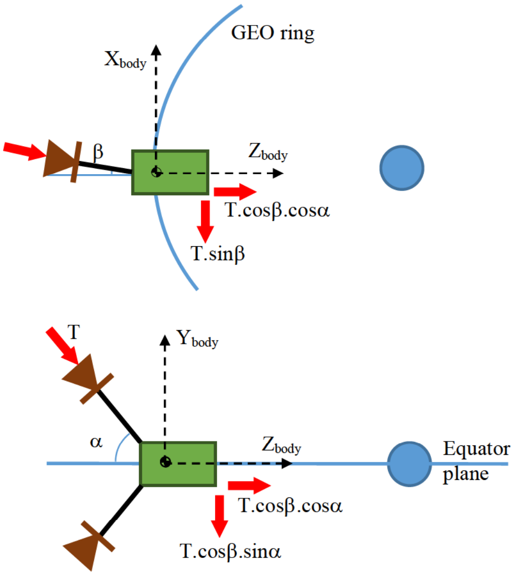

Figure 5: Station Keeping Geometry in GEO

- The consequence of the radial component, is a jump in mean longitude, always Eastwards for an impulse towards nadir, directly proportional to this impulse. This can, if desired, be countered by selecting a semi-major axis slightly above GEO, thereby imposing a slow westward drift.
- The perturbation on the eccentricity vector caused by this radial component would vanish assuming both daily manoeuvres equally sized and happening with a separation of exactly 180°.
- Small asymmetries between the two orbital locations / sizes of the North-South manoeuvres can be used to effectively control the eccentricity vector simultaneously to the longitude control, by combining the radial and tangential impulses.
- The tangential impulses are needed to keep the longitude by acting on the drift of the orbit and to counteract the perturbation of the eccentricity vector by the radial thrust component as described above.

In conclusion the strategy for orbit control consists of one manoeuvre around each orbital node on up to 6 days per week. In addition the Momentum management needs to be considered in above Station Keeping strategy. External and internal disturbances lead to the accumulation of angular momentum in the reaction wheels, which shall be unloaded during the Station Keeping manoeuvres.

Based on the EP thruster configuration, momentum control is possible in both axes perpendicular to the thrust direction but not about the thrust vector itself. In order to control the momentum about the main thrust direction as given by above orbit control strategy following options were identified:

- Select non-parallel orbit locations for the two nodal burns, such that the needed momentum components are achieved.
- Add one burn on the opposite boom in sequence with the nominal orbital control manoeuvre around the nodes dedicated to momentum management about the main thrust axis.
- Add one burn about 90° apart the nominal orbital control manoeuvre around the nodes dedicated to momentum management about the main thrust axis.

The first option leads to the minimum number of manoeuvres (two per orbit) but involves non-optimum SK manoeuvres. The latter two options decouple the momentum management about the main thrust axis from the orbit control needs, but need additional burns, typically one per boom per week. The second strategy is preferred compared to the third, as the thrust can be applied nearly perpendicular to the momentum to be dumped and thereby the burn time and the corresponding parasitic Δv is minimised.

EPPS Definition

As consequently derived from the Mission requirements, the Electra EPPS is based on two EP thrusters mounted on two articulated booms plus a cold redundant branch. The two nominal thrusters are connected each to a single Power Processing Unit (PPU), while the redundant thruster couple is connected to a PPU equipped with a thruster switching unit (Figure 6). Thus, the EPPS guarantees full performance also in case of one thruster or one PPU failure. The available power to the EPPS during the orbit transfer is basically defined by the payload power of about 10 kW, while during the station keeping phase the power is reduced to 3.5 *kW* in order to minimise the impact on the power system. At these input power levels, the EPPS provide thrust in the range of 150 ÷ 550 mN with specific impulse in the order of 1700 – 1800 *s* based on HET technology.

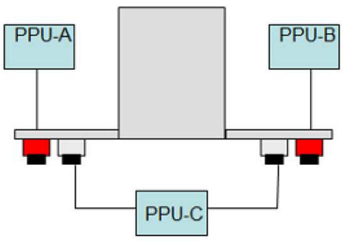

Figure 6 Thrusters-PPUs connection schematic

The EP booms are mounted on the lower edges of the North and South radiator panels, close to the anti-Earth panel (Figure 7). This configuration has been extensively analysed and found to compare favorably against configurations using individually gimballed thrusters as it reduces the total number of actuators. The three actuators per boom allow to position and align the thrust vector to apply forces and torques as required for Orbit and Attitude control. Actuator axis 1 and 2 are implemented in parallel direction to allow decoupling of force and torque in the time critical E/W direction during station keeping.

During orbit raising, the thruster plumes are directed towards the -Z axis, therefore the plume impingement on S/C surfaces is negligible. During station keeping the thrusters are positioned such that the angle between the thrust vector and the North-South direction is minimised thanks to the dimension of the boom while keeping any plume impingement impact on the solar arrays within allowable levels.

The use of EP booms provide further advantages with respect to body mounted configurations in terms of thermal control management and electro-magnetic interferences (EMI) between thrusters and satellite.

The EP boom solution increases also the flexibility during AIT activities at subsystem and system level, as the integration of the EP thrusters as well as harness and piping on the robotic arms can be completed independently of the AIT activities on the platform.

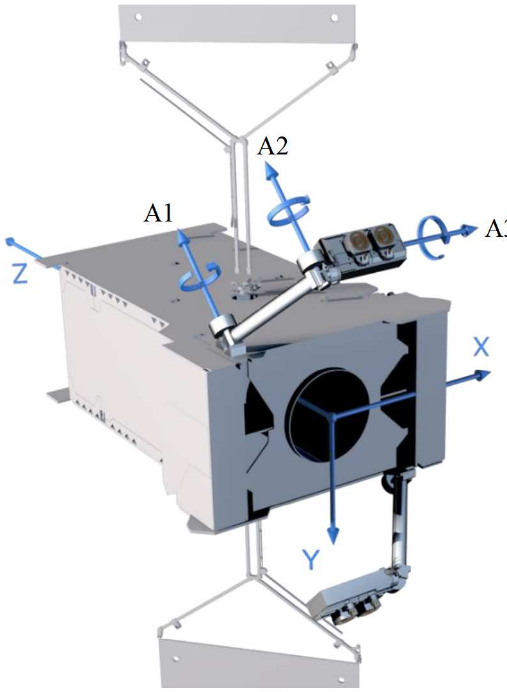

Figure 7: EP Boom kinematic

CONCLUSION

The OHB SmallGEO family of satellites are designed for small and efficient geostationary missions with a launch mass of up to 3.5 tons. The Electra Satellite platform increases the payload capacity to 800 kg mass and 10 kW power by means of full electric propulsion.

Electra mission analysis have developed optimized orbit raising trajectories with a total transfer time in the range of 4 to 5 month for a typical GTO launch. Station keeping follows a combined strategy for N/S and E/W around the orbital nodes following the basic principles proven on Hispasat 36W-1. Based on the mission requirements an electric propulsion configuration consisting of two redundant EP thruster pairs on two articulated EP booms have been derived.

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