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Advanced AOCS Design on the First Small GEO Telecom Satellite

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The advanced Attitude and Orbit Control System (AOCS) design of the Small GEO platform is now being adapted for the first commercial mission. The Small GEO telecommunications satellite is a new development to fill a niche in the telecom industry for small platforms weighing about 1.5 tonnes and targeting payloads of 300 kg and 3 kW. The first mission will launch into Geostationary Transfer Orbit (GTO). Small GEO is being developed by a Consortium led by OHB-System AG. The Swedish Space Corporation is a partner in the Consortium and supplies the AOCS and Electric Propulsion (EP) subsystems. The project is currently in Phase C and the first mission will fly in 2012. This article gives an overview of the AOCS development status.

The AOCS architecture is a three-axis stabilized system using reaction wheels for attitude control, star trackers for attitude determination, and EP for orbit control. The AOCS software is being developed using model-based design techniques and test driven development. Results from subsystem level testing of flight code will be presented.

The AOCS design is characterized by a number of advances in technology beyond traditional telecom satellite designs. Perhaps the largest deviation from a traditional design is complete reliance on EP for orbit control. Angular momentum management of the reaction wheels relies solely upon EP in the nominal modes. The EP is not used in the safe modes and therefore a cold gas system is included On-board. The cold gas system uses Xenon, the same fuel used by the EP. Another advance is the reliance upon APS-based star trackers. APS (Active Pixel Sensor) star trackers have a number of advantages over their CCD-based cousins in terms of robustness. The traditional fine sun sensor is simplified to a fault tolerant system of solar cells giving low, but more than adequate, accuracy. In addition, a GPS sensor will be flown on-board as an experiment.

I. Introduction

THE Small GEO Attitude and Orbit Control System (AOCS) will be the most advanced AOCS yet to fly on a telecommunications satellite. The design uses the latest sensors, actuators, and software development techniques. The design includes both full controllability by ground and advanced autonomy to maximize flexibility for the customer.

The Small GEO telecommunications satellite is a new development by a consortium led by OHB-System AG under overall management of the European Space Agency (ESA) to fill a niche in the telecom industry for small platforms weighing about 1.5 tonnes and targeting payloads of 300 kg and 3 kW. The platform design provides the capability for direct injection into geostationary orbit as well as injection into Geostationary Transfer Orbit (GTO), both with high mass efficiency. In GTO configuration, the platform will weigh approximately 2.5 tonnes. The platform is compatible at minimum with the following launchers: Ariane 5 (secondary passenger under Sylda), Soyuz GSC (from Guiana Space Centre), LandLaunch and Proton from Baikonour, as well as Atlas V. Potential future launchers like GSLV MkII, Falcon 9, and Angara 3A are considered as well.

The prime contractor for Small GEO is OHB-System AG (Germany). The Swedish Space Corporation (Sweden) is a partner in the consortium and supplies the AOCS and EP subsystems as well as the EP-based mission analysis. The other partners are RUAG AG (Switzerland) and LuxSpace Sàrl (Luxembourg).
The project for platform development, ESA's ARTES 11, is currently in the implementation phase (Phase C/D). Negotiations are complete for nearly all equipment. The first Structural and Thermal Models arrive in 2009 and the first Engineering Models will arrive in early 2010. The first release of the AOC Core software flight code is in October 2009. Launch is expected in 2012.
The contract for the first commercial mission was signed in November 2008 between ESA and HISPASAT S.A. The second mission is currently under development.

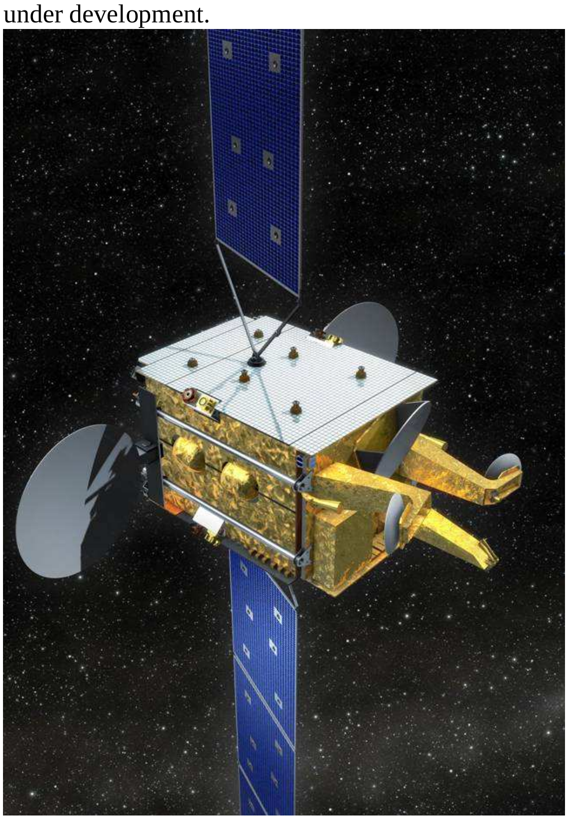

Figure 1: Hispasat AG1 (OHB-System)

II. Mission Overview

The first Small GEO mission will be the HISPASAT Advanced Generation satellite (Hispasat AG1). Its main payload of Ka-band transponders will serve Spain, Portugal, the Canary Islands, and America. An advanced Ku-band payload will also be carried. It will also carry a GNSS receiver as a demonstrator.
The satellite will be launched into a Geostationary Transfer Orbit. A bi-propellant system on-board will provide the injection into geosynchronous orbit. Final orbit transition and placement is however performed with the Electric Propulsion (EP). Electric propulsion is then used for all nominal station-keeping and momentum management for the entire lifetime of 15 years[1]. The mission phases are described below from an operator's viewpoint.

A. Separation from launcher

The solar array and antenna payload is stowed when SGEO is separated from the launcher into GTO as shown in the figure below.

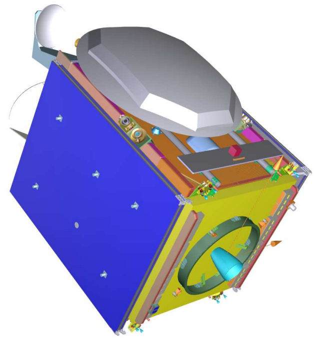

Figure 2: Stowed SGEO

The satellite is turned on automatically at separation. The software is booted up automatically and AOC Core initializes in Standby Mode. After at least one sample in Standby Mode and after valid measurements have been attained by the sun sensors and gyro, the AOC Core will automatically transition to SAM due to high separation rates.
Once in SAM, the cold gas system will be requested. When the cold gas system becomes valid, the AOC Core will reduce the angular momentum of the satellite in CGMD. When the angular momentum has been reduced to a sufficiently low level to be absorbed by the reaction wheel assembly, an autonomous transition to RWSA is performed. The reaction wheel assembly will be requested on at entrance to RWSA. The satellite will then rotate the zenith panel towards the sun under reaction wheel control. CGMD can be entered autonomously at any time if the total angular momentum exceeds the capabilities of the reaction wheel assembly. Once sun pointing has been achieved, it will be maintained autonomously with no rotation (default) around the sun vector limited by the bias drift of the gyro.
If the reaction wheel assembly is unhealthy and never becomes valid, CGSA will be entered and used for sun acquisition.
The following plots from the Scenario Test Report of the most recent AOC Core software intermediate release show the nominal sun acquisition sequence after launcher separation with a tip-off rate of 5°/s.

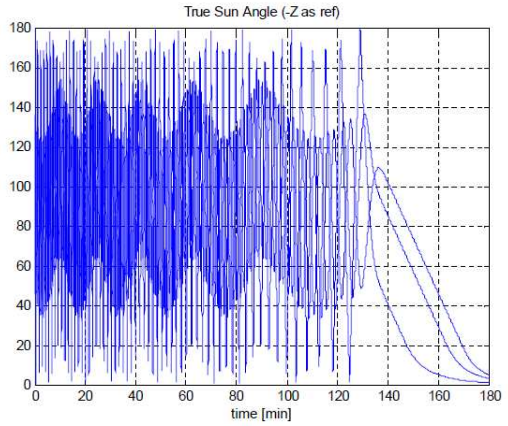

Figure 3: Scenario plot angle between sun vector and -Z_{SCB}

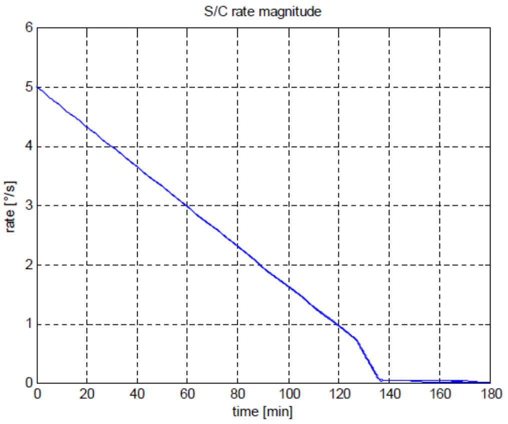

Figure 4: Scenario plot of spacecraft angular rate

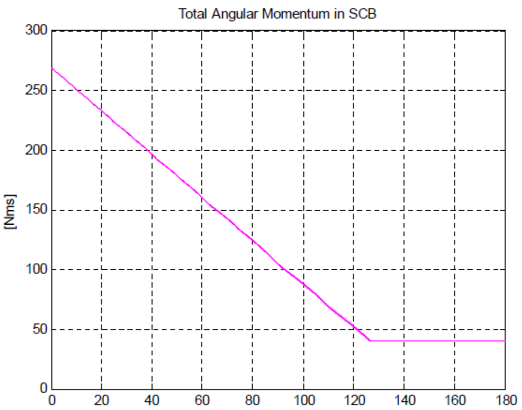

Figure 5: Scenario plot of total angular momentum

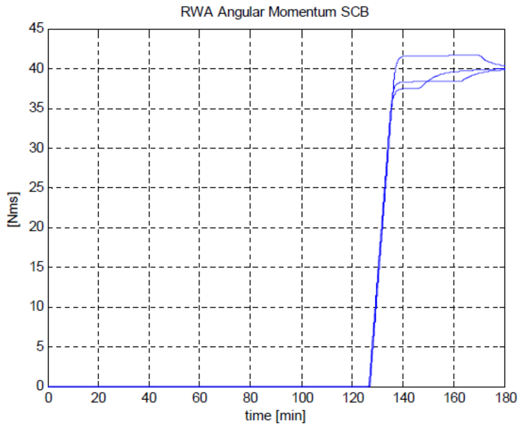

Figure 6: Scenario plot of reaction wheel angular momentum

B. Transfer to geosynchronous orbit

Once sun acquisition has been achieved, ground will deploy the first two solar array panels as shown in the figure below. This will be done in SBM to ensure that no control loops are active during solar array deployment. SBM is entered by telecommand from any mode.

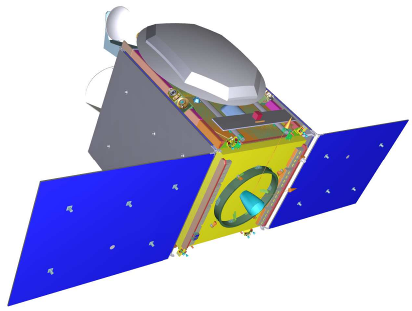

Figure 7: Partially deployed SGEO

After partial deployment of the solar array, SAM will be re-entered by telecommand. Once sun acquisition has been re-established, NM will be entered with the current attitude as reference. The star tracker must be valid before NM can be entered. Sensor and actuator check-out is best performed at this stage in NM.

With the current attitude as the initial condition, an attitude reference can be generated on ground and up-linked to the satellite as chebyshev polynomials. The reference will be sun pointing for most of the orbit. Slew maneuvers will be performed just before and just after the chemical propulsion firing around apogee. Ground based attitude reference generation accounts for operational limitations during slew maneuvers. The attitude reference during chemical propulsion firing will be either inertially constant pointing or orbit tangent pointing as subscribed by the mission analysis strategy. The On-board Orbit Propagator is by default not used during chemical propulsion firing.

C. Final orbit insertion

Before starting final orbit insertion, the solar array must be fully deployed in order to provide sufficient power to the EP. Once again, SBM should be entered by telecommand for solar array deployment to ensure that the control loops are not active. Afterwards, entrance to SAM should be commanded to re-establish sun pointing. With deployment complete, NM can be entered by telecommand with the current attitude as reference.

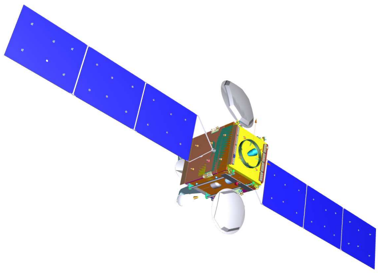

Figure 8: Fully deployed SGEO

The final orbit insertion is EP based and requires an attitude reference that rotates the satellite around the nadir vector while alternately firing the EP thrusters. EPCM is entered by telecommand. The EP command profile and attitude reference are generated on-board because the orbit insertion is time critical. This guidance is initiated by telecommand. The On-board Orbit Propagator is also initialized with the current orbit.

D. Nominal operations

Before start of nominal operations, the payload antennas must be deployed. As with the solar array, this is done in SBM to ensure that all control loops are disabled. SAM is commanded afterwards to re-establish sun pointing. The NM is entered with the current attitude as reference. At this point, the On-board Orbit Propagator should already be initialized by ground. Ground can choose to either up-link an attitude reference profile or to command Autonomous Earth Pointing guidance based upon the On-board Orbit Propagator. An EP command profile must be generated on ground and up-linked to the satellite.

The attitude reference profile and EP command profile are generated together on ground. The profiles have a duration of two weeks but are up-linked every week. If the attitude reference becomes invalid due to age etc., the Autonomous Earth Pointing based upon the On-board Orbit Propagator will be used. Alternatively, Autonomous Earth Pointing can be enabled as the nominal guidance function. The up-linked attitude reference profiles introduce an error due to their representation as Chebyshev polynomials. The primary sources of error in the propagator are initial orbit determination error and EP thruster uncertainties. A proper comparison between a full scale ground-based orbit propagator and the On-board Orbit Propagator is still pending. In comparing the two approaches, the operator can choose to command via the attitude reference profile for full controllability or via the on-board attitude guidance for operational flexibility.

The following plots from the Scenario Test Report of the most recent AOC Core software intermediate release show achieved attitude and orbit control during a nominal single orbit in Normal Mode.

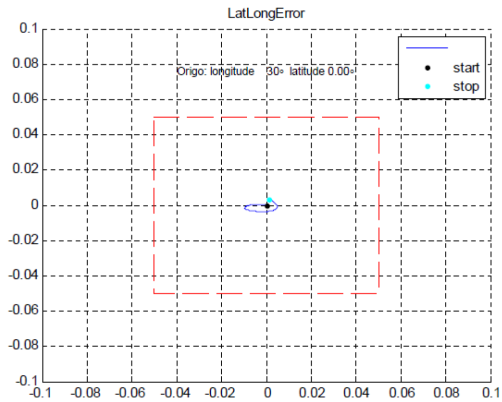

Figure 9: Scenario plot of error in latitude and longitude

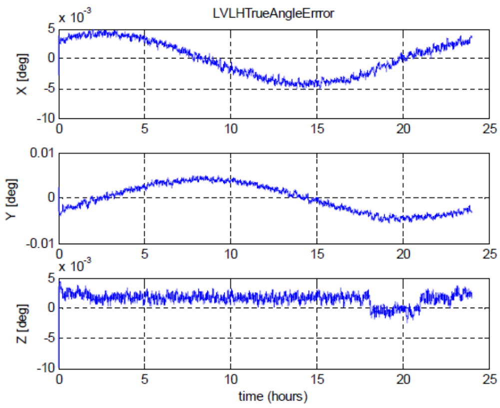

Figure 10: Scenario plot of angular error in SLO coordinate frame

E. Orbit relocation

Orbit relocation is similar to final orbit insertion. It is EP based and requires an attitude reference that rotates the satellite around the nadir vector while alternately firing the EP thrusters. EPCM is entered by telecommand. The EP command profile and attitude reference are generated on-board because the orbit insertion is time critical. This guidance is initiated by telecommand.

F. Graveyard orbit insertion

Graveyard orbit insertion is essentially the same as orbit relocation in terms of AOC Core usage.

III. Attitude and Orbit Control System Design

The AOCS consists of both software and hardware. The AOC Core is the AOCS software function integrated into the overall on-board software (OBSW). The AOC Core commands the following actuators:

- Reaction wheel assembly (RWA),
- Cold gas thrusters (XREA),
- EP thrusters (EP),
- Vernier thrusters (VREA),
- Liquid apogee engine (LAE),
- Solar array drive mechanism (SADM),

based upon inputs from the following sensors:

- sun presence sensors (SPS),
- gyros (GYR),
- star trackers (ST).

In addition, the AOC Core performs AOCS related Failure Detection and Isolation. Recovery is performed centrally in the overall OBSW.

The mode diagram for the AOC Core is shown below. Automatic transitions are black and identified with prefix “Ta”. Ground commanded transitions are grey and identified with prefix “Tc”.

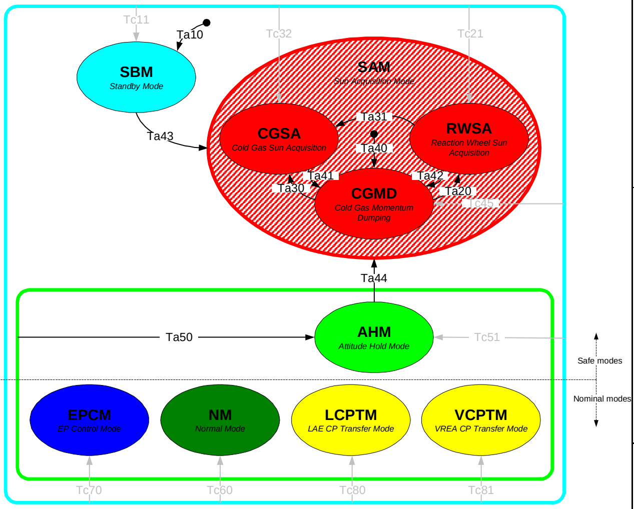

Figure 11: AOC Core modes and transitions

ID	Automatic transition from	Automatic transition to	Trigger
Ta10	AOC Core Initialization	SBM	At software initialization
Ta20	CGMD	RWSA	Low body rates and low reaction wheel angular momentum
Ta30	CGMD	CGSA	Unhealthy (excessively invalid) reaction wheels
Ta31	RWSA	CGSA	Unhealthy (excessively invalid) reaction wheels
Ta40	SAM Initialization	CGMD	SAM is initialized if either transition Ta43 or Ta44 triggers.
Ta41	CGSA	CGMD	High body rates or high reaction wheel angular momentum
Ta42	RWSA	CGMD	High body rates or high reaction wheel angular momentum
Ta43	SBM	SAM	Elapsed time since entrance to SBM exceeds TBD seconds or high body rates or high reaction wheel angular momentum
Ta44	AHM	SAM	Elapsed time since entrance to AHM exceeds TBD seconds or high body rates or high reaction wheel angular momentum
Ta50	AHM, EPCM, NM, LCPTM, VCPTM	AHM	System FDI trigger

Table a: automatic transitions

Ground commanded transitions are triggers sent from any mode to any mode for full commandability.

G. Sensors and Actuators

1. Star tracker

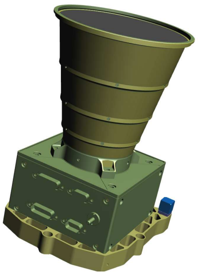

Figure 12: ASTRO APS (Jena-Optronik)

The most critical sensors for the AOCS are the two cold redundant star trackers. As indicated in Ref. 2, SGEO carries no earth sensor, but relies solely on its star sensors. The chosen star sensor is the Jena-Optronik ASTRO APS. It uses the most radiation hard CMOS Active Pixel Sensor on the market, the STAR1000. The Active Pixel Sensor (APS) detector is an alternative to a Charge-Coupled Device (CCD) detector. Each pixel has an individual readout, thereby avoiding "blooming" of charge over to neighboring pixels. This improves performance with bright objects in the field of view, e.g. the moon. The centroid calculations are done on the same integrated circuit resulting in tracking rates, e.g. 10 Hz, that are much faster than CCD-based star trackers. Built-in software algorithms automatically identify and compensate for anomalies such as white spots. A Peltier cooler is included. The star sensor is a single unit with camera head, electronics, and a shorter baffle than on CCD-based star trackers. The units will be mounted externally on the antenna deck.

2. Sun sensor

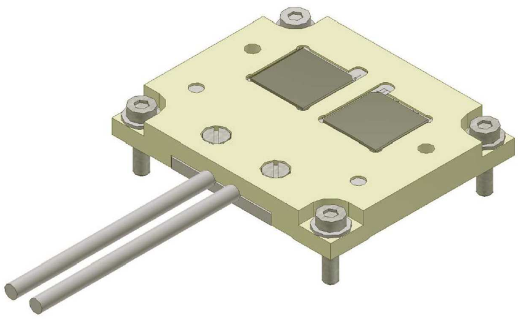

Figure 13: SGEO CoSS (Bradford Engineering)

The chosen sun sensor will be supplied by Bradford Engineering. It has redundant detection elements each with a near hemispherical field of view. The outputs approach a cosine function of the angle of incidence of the sunlight onto the detection elements. For this reason the redundant device is named Cosine Sun Sensor or CoSS. The proposed design maximally exploits the extensive heritage of dual chip devices used by TNO in its Sun Acquisition Sensor flying in a variety of space programs, including several geosynchronous telecommunication satellite platforms with lifetimes up to 15 years.

A suite of these sensors will be mounted on the satellite: two sensors on each panel, except for the antenna deck which will have three, i.e. thirteen sensors in total. For any given sun vector, at least one sensor on each lit panel will not be shaded by external equipment such as antennas, solar array wings, or towers. The result is an omni-directional sensor system for sun vector estimation and sun presence detection. Sun presence detection is unequivocal in geosynchronous orbit, but albedo is a concern in geosynchronous transfer orbit. The sun vector estimation algorithm in the AOC Core software uses the orthogonality of the cells and ground calibration from the supplier.

3. Gyrometer

At writing of this article, no gyro has yet been chosen. The gyro should support the Safe modes and replace the Star Tracker during drop-outs. There are three European candidates, all with their specific advantages and disadvantages. One of the primary trade-offs is performance versus reliability.

4. Reaction wheel assembly

Figure 14: RSI 68 (Rockwell Collins Deutschland)

The choice of reaction wheel was driven primarily by the need for large angular momentum capacity resulting from the EP thruster firing. Temperature and accommodation requirements were also significant. The chosen wheels provided by Rockwell Collins Deutschland have integrated electronics and a nominal capacity of 68 Nms. The four reaction wheels will be operated in hot redundancy in a pyramid configuration. In the case of one wheel failure, the minimum capacity of the momentum ellipsoid is 54 Nms. Speed control by angular momentum redistribution will minimize zero-crossings.

5. Chemical Propulsion Subsystem

The Chemical Propulsion Subsystem (CPPS) consists of eight (4+4) 10N Vernier Thrusters and one 400N Liquid Apogee Engine (LAE). The CPPS is only used for transfer from GTO to GEO, nominally with four LAE burns of varying length. The Vernier thrusters are used for attitude control before, during, and after the LAE burn. In case of LAE failure, the subsystem is dimensioned for transfer with Vernier thrusters only. Upon successful transfer, the subsystem can be passivated with two opposing vents.

6. Electric Propulsion Subsystem

The EP Subsystem (EPPS) consists of two branches of four EP thrusters and two branches of four xenon cold gas thrusters. The subsystem is described in detail in Ref. 3. A propellant supply assembly is

being developed to supply xenon with a flow rate of ~200mg/s at beginning of life to the cold gas thruster assembly or with a flow rate of ~5mg/s to the EP thruster assembly.
The cold gas thruster assembly is used only for rate damping and momentum management. Thus, the system is only used for generation of torques, i.e. no forces. The AOCS accounts for the flow limitations that arise from sharing a xenon supply system with the EP. At end of life, the flow rate can be reduced to one eighth of the initial flow rate.
The nominal EP branch is the HEMP-T with a thrust of 45mN. The redundant EP branch is the HET with a thrust of 75mN. An alternative thrust setting of 40mN for the HET is also being evaluated.
The eight EP thrusters are mounted in pairs on the East and West panels with thrust directions symmetrically ordered around the nadir direction. In nominal operations each thruster has thrust vector components in the directions orthogonal to the orbital plane and tangential to the satellite velocity vector in inertial space.
Several cant angles have been analyzed though the nominal cant angle between north and the thrust vectors is 46°.

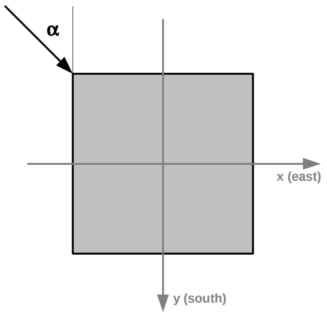

Figure 15: Thruster cant angle α

The configuration is chosen such that the lifetime centre of mass (CoM) movement is bounded by the eight thrust vectors with an additional margin to ensure that torques can always be generated for wheel unloading. Other considerations in choice of configuration are plume impingement and North/South efficiency due to cosine loss.
The eight EP thrusters are configured as follows.

Configuration	Branch	EPs	Comment
3D symmetry	HET	1,4,5,8	Symmetric in three dimensions
	HEMPT	2,3,6,7	

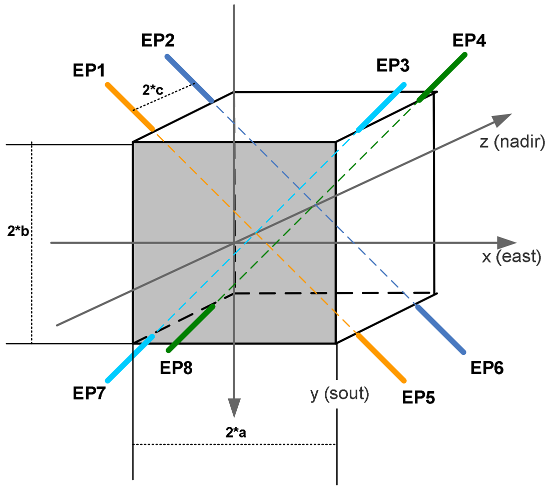

Figure 16: HAG1 EP thruster configuration

Force Direction	Thrusters	Comment
SE	EP1, EP2	South East
SW	EP3, EP4	South West
NE	EP7, EP8	North East
NW	EP5, EP6	North West

H. AOCS Software

At the start-up of the AOC Core development phase (Phase C), the AOCS team considered methods of improving the AOC Core Software quality with the aid of agile methods, e.g. Scrum. The Test-Driven Development (TDD) technique was particularly liked by the team.
Given the complexity of the AOC Core software and the fact that a well adapted AOC Core architecture was already in place, TDD in its purest form was hard to achieve.
Therefore, an attempt was made to integrate the concept of Scenarios derived from previous SSC satellites (PRISMA and Smart-1) with TDD resulting in Test-First Design (TFD) for the SGEO AOCS development.
The main purpose with SGEO AOCS Scenarios is to connect required high-level functionality (i.e. Level 2 User Requirements) early in the AOC Core development. The AOC Core software requirements (i.e. Level 3 Unit Requirements) would be implemented in conjunction with the Scenarios by adding Unit Tests (and the Matlab/Simulink software to implement, execute, and post-process them).

The product backlog items would then be the Scenarios primarily, and secondarily the AOC Core level-3 requirements. The burn-down of the latter would then be accomplished by the Unit Test and documented by the Unit Test Report.

The AOCS team agreed on the following demands for the SGEO AOCS Scenario Framework:

- The Scenario must be possible to run automatically (i.e. typically by starting a MATLAB script).
- The success of a Scenario must be able to be automatically checked.
- To ensure quality (the TDD way) all completed Scenarios should be successfully re-run at the conclusion of each Sprint, *this means that the Scenario Test Framework must be able to batch run scenarios.*

The result of a Scenario (batch) run should be presented in an auto-generated Scenario Test Report.

Later an additional requirement was added to connect the Scenarios to the upcoming AOCS Subsystem verification process: the Scenarios shall be linked to level-2 Requirements.

The nominal Scenarios are listed here. The failure Scenarios are still under development.

SEPARATION_GTO_NOMINAL
NORMAL_1DAY_GTO_NOMINAL_ECLIPSE0_EP1
NORMAL_1DAY_GTO_NOMINAL_ECLIPSE1_EP1
NORMAL_1WEEK_GTO_NOMINAL_ECLIPSE1_EP1
INCLINED_NORMAL_GTO_NOMINAL_ECLIPSE1_EP1
REPOS_GTO_GTO2GEO_ECLIPSE1
REPOS_GTO_GEO2GEO_ECLIPSE1
REPOS_GTO_GEO2GRAV_ECLIPSE1
LEOP_DELTAV0_NOMINAL
LEOP_DELTAV1_NOMINAL
LEOP_CHECKOUT_GTO_ST
LEOP_CHECKOUT_GTO_RWA
LEOP_CHECKOUT_GTO_XREA
LEOP_CHECKOUT_GTO_XREA_SAC
LEOP_CHECKOUT_GTO_NM

Intermediate software is now released every other sprint, i.e. every other month. The intermediate software has a complete architecture and is executable, but only contains the functionality needed for the completed Scenarios. A Scenario Test Report and Unit Test Report are included in every release. The number of completed Scenarios is the primary metric of code maturity. In this respect, it is the tests that drive the development. Plots from the most recent Scenario Test Report are presented in Sections II.A. and II.D. above.

IV. Mission Analysis

I. EP-based Station-Keeping

The objectives of the EP-based Station-Keeping are tabulated here.

7 day cycle
The baseline station-keeping strategy is a seven day cycle with 6 consequent days of pre-planned station-keeping maneuvers followed by one day devoted to orbit determination.
Reducing propellant consumption
Controlling the inclination drift is the most costly task of the station-keeping and including the control of eccentricity and drift into the inclination maneuvers is a major concern for saving propellant.
Minimizing thruster firings
EP thrusters have a limit as to how many times they may be fired. The baseline constraint is one firing per thruster and day.
Angular momentum
The angular momentum should be kept in an allowed range.
Tolerant to loss of one EP branch power supply It should be possible to continue the mission in spite of loosing one of two power processing units and hence half the number of thrusters.
Tolerant to station-keeping cycle discontinuation
In the event that the station-keeping cycle discontinues, the satellite should not reach the edge of the geo-slot in less than 48 hours.

Table b: Station-keeping objectives

For the EP configuration of SGEO, the Station-Keeping (SK) control for inclination, longitudinal drift, and orbit eccentricity is performed with combined N/S-E/W maneuvers.

Station-keeping for the EP platform needs to be performed almost daily because of the inherently low thrust force of an EP system. For SGEO, a SK cycle of 7 days has been defined. SK maneuvers are performed during 6 of these days while the 7th day is reserved for OD.

A SK EP profile is generated every 7th day and uploaded to the satellite for autonomous station keeping and angular momentum control for the next 7-day control cycle. For safety, the uploaded EP profile contains maneuvers for 14 days.

To generate the EP profile on-ground, the satellite orbit is propagated for a 14-day period taking into account natural disturbances. The orbital corrections needed per day are then determined from the propagated orbit and these are fed into the SK algorithm. The algorithm then optimizes the thruster profiles taking into account the thrust forces including plume impingement and angular momentum build-up. The EP profile generation scheme is illustrated with Figure 17.

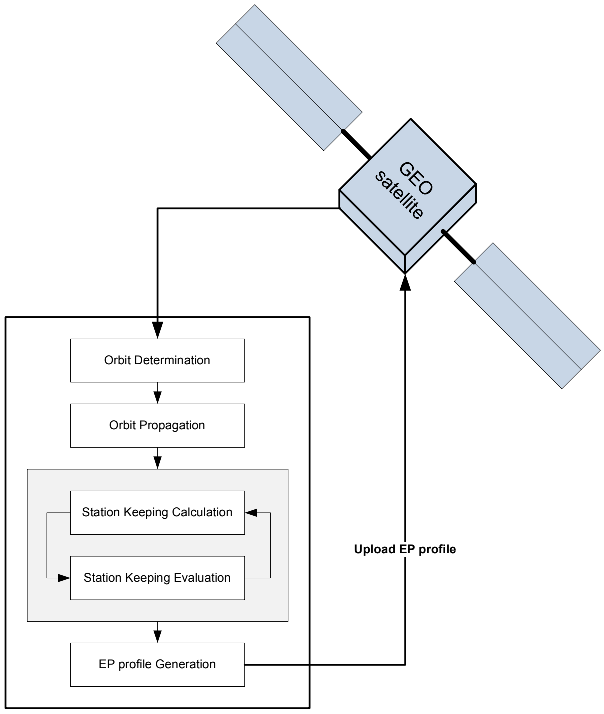

Figure 17: EP Profile Generation Scheme.

The most propellant consuming parts of the SK maneuvers are the inclination corrections and the angular momentum control. The thruster configuration and SK control scheme selected for the SGEO platform is able to efficiently control the inclination, mean longitude and orbit eccentricity while at the same time minimizing the angular momentum built-up.

The EP configuration allows for simultaneous correction of the orbit parameters in the same SK maneuvers. This is possible by taking advantage of the E/W components of the EP thrust used to control the inclination thus making it possible to also control the longitudinal drift and eccentricity.

The longitudinal drift is controlled by balancing the drift acceleration with the total E/W thrust. The component of the eccentricity that is perpendicular to the desired inclination change is controlled through the distribution of the E/W thrust components between the nodes.

Small corrections of the eccentricity component parallel to the desired inclination change are made entirely within the bounds of a nominal inclination maneuver. For larger corrections of this eccentricity component it is possible to apply a strategy where the thrusts are separated slightly at each node.

The SK algorithm optimizes the thruster times to control inclination and longitudinal drift together with out-of-plane angular momentum buildup. The in-plane angular momentum is handled by selecting the best thrust order leading to the optimized in-plane angular momentum control scheme described below. It needs however to be noted that it is not possible to control all angular momentum buildup within SK maneuvers. The residual momentum is handled by using a dedicated thruster pair if necessary.

Another important aspect of the SK algorithm used in SGEO is that it handles even the correct separation of thrusters such that no overlapping of thruster use occurs. The algorithm also allows for cool-down times between thrust arcs. This separation is achieved by first calculating the optimal thrust strategy, then separating the thrust arcs in order to calculate the optimal thrust strategy again. This is repeated in an iterative process until a solution is found.

1. *Optimized In-plane Angular Momentum Control*

The combined SK strategy used for SGEO also provides the possibility to minimize the angular momentum build-up occurring along the X_{ECI} direction. This brings two additional benefits. The strategy will counteract the angular momentum build-up due to the shift along the radial direction of the spacecraft's Center of Gravity (CoG) (Z_{SCB}). The momentum control takes place during SK maneuvers which inherently are located near the $\pm Y_{ECI}$ axes. As a result, the eclipse zones are always avoided since they will be located near $\pm X_{ECI}$.

The ability to counteract the radial shift in CoG can be seen if the angular momentum disturbance matrix of a N/S maneuver with four thrusters is analyzed. The situation is illustrated with Figure 18.

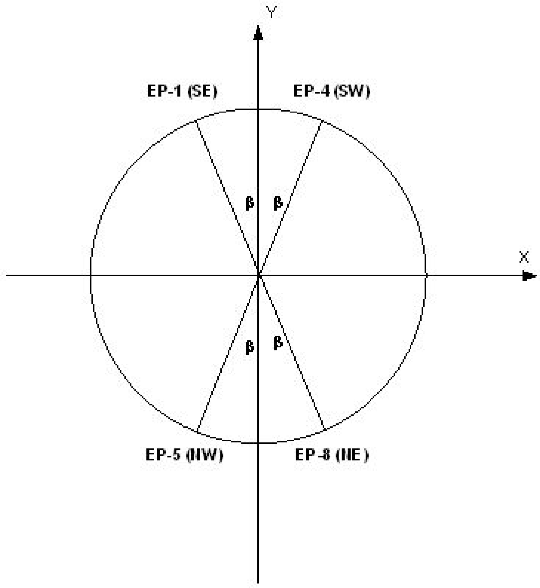

Figure 18: Thruster firing locations in X_{ECI} - Y_{ECI}.

Depending on the order of the thruster firings, it is possible to reverse the sign of the angular momentum build-up along X_{ECI} while keeping the sign of the build-up along Y_{ECI} unchanged. This can be seen in the disturbance vector

$$\begin{bmatrix} 4dtF(-\cos(\beta)p_z + \sin(\beta)a - \sin(\beta)b) \\ 4\cos(\beta)dtFp_x \\ 0 \end{bmatrix},$$

where p_x, p_y, p_z are the respective CoG displacements in SCB, *a,b,c* are the lengths defined in Figure 16, *F* is the thrust force magnitude and *dt* is the thrust time.

The dependency of p_z can be removed by selecting an appropriate angle β. If there is no displacement of the CoG along Z_{SCB}, the angular momentum in X_{ECI} is removed by altering the thruster firing positions.

The angular momentum management along Y_{ECI} can be separated into one dedicated maneuver close to +/- Y_{ECI} but slightly separated away from the node lines in order to interfere as little as possible with the N/S SK.

Note also that this strategy works only for the minimization of angular momentum build-up along X_{ECI} since, even if it is possible to find a firing strategy that minimizes the build-up along Y_{ECI}, this will lead to loss of control of the eccentricity.

The capabilities of the SGEO SK strategy have been demonstrated by the simulation of a full 15-year mission. The satellite is assumed to be positioned at a worst case drift position around 120 deg E longitude. The simulation makes use of the ground based SK strategy described above with the exception that the SK cycle is 14 days. The preconditions for the simulation are as follows:

- Longitude 120E
- EP : HET 40 mN , Isp 1330, cant angle 43 deg.
- Plume impingement: worst case
- Center of Gravity dislocation: worst case
- BoL mass: 1967 kg , EoL mass: 1797 kg (total propellant consumption 170 kg)

The results from the simulation are illustrated in Figure 19. The figure clearly shows that the spacecraft stays well within the geosynchronous control box while simultaneously controlling the angular momentum.

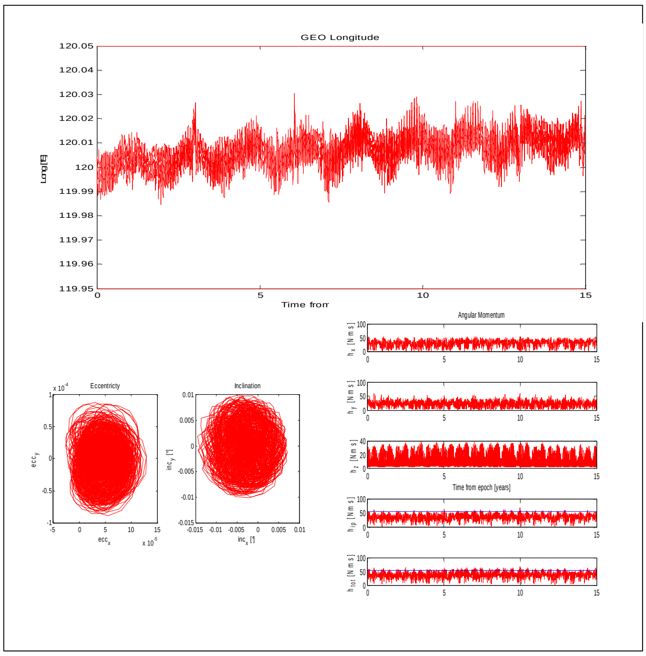

Figure 19: Simulation results from SGEO 15-year simulation with worst case drift and disturbance conditions.

J. EP-based Orbit Relocation

When leaving or entering a GEO-slot at a certain longitude it is important to stay clear of neighboring satellites. Typically, a repositioning maneuver is carried out at a drift rate of no less than 0.5 deg /day leading to a semi-major axes differing from that of stationary geosynchronous orbit by a little less than 40 km. A typical long requirement is a square box in longitude and latitude with a side of 0.2 degree or slightly less than 150 km. A box with a side in longitude and latitude of 150 km and a depth of 75 km is considered here.

In order to properly leave the GEO-slot and to start drifting in longitude requires the semi major axis of the orbit to be changed and this in turn means that a tangential thrust is needed. The requirement to leave the slot on the radial "face" of the box places a lower bound on the thrust needed.

The ΔV needed to reach a drift rate of 0.5 deg/day is constant at $\Delta V=D*V/3$ where D is the drift rate in parts of an orbit per revolution, D=0.5/360 and V is the velocity of the spacecraft in inertial space V = 2*pi/T*R_{GEO} where T is on sidereal day, 86164 s, and R_{GEO} the radius of the geostationary orbit 42164200 m. This gives a ΔV of 1.42 m/s.

Figure 20 shows the trajectories resulting from continuous thrusts with multiples of a 15 mN tangential thruster. From Figure 20 it can be seen that a manoeuvre starting at the centre of a GEO slot needs a constant thrust of between 30 and 45 mN in order to take the a mass of 1850 kg out of a 150 km box without violating the neighbouring slots.

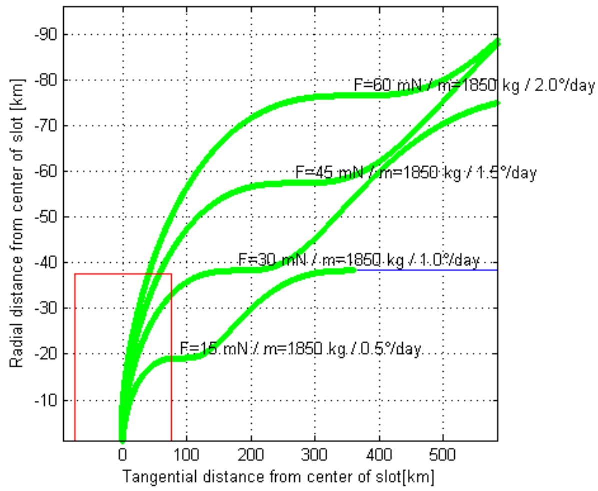

Figure 20: Leaving slot with multiple 15 mN thrusters. Longitude drift rates refer to the final drift after two days of thrusting.

A tangential thrust force of at least 38 mN in the orbit tangential direction has been calculated to be required to safely perform orbit relocation.

Since an increased uncertainty in Center of Mass (CoM) along Z translates into increased angular momentum build-up, it has been found necessary to incorporate a function within the AOCS which monitors the build-up of the angular momentum within the reaction wheel assembly, and modifies the resulting attitude slew profile and EP firing times to maintain the angular momentum within the specified limits.

The control function to be implemented within the AOC Core is divided into two consecutive steps:

A) A nominal calculation of the EP activation flags and slew rate profiles is performed assuming the CoM is perfectly known (i.e. no unexpected delta on the CoM position). The expected (ideal) result at the end of one pair of thruster firings is that one of the angular momentum components shall be zero.

B) After two firings (one thruster pair) the angular momentum build-up in ECI is checked. If the monitored angular momentum component is not "zero", new EP activation flags are defined in order to avoid additional build-up of this component.

For each thruster firing, the spacecraft is rotated 90° around Nadir whilst respecting the limited torque capabilities of the reaction wheels. During the station acquisition the spacecraft will experience a total of four full rotations and 16 individual EP thruster firings per orbit.

It is important to note that due to the symmetry of the orbit and the number of spacecraft rotations (2 over an half orbit) determined by the predefined EP activation flags and slew rate profiles, any drifts of the in-plane components will be simply cancelled out over an orbit.

Thus, the control function will focus on removing drift from the Z-axis which could lead to exceed the maximum magnitude of the angular momentum build-up in the Reaction Wheel Assembly (RWA).

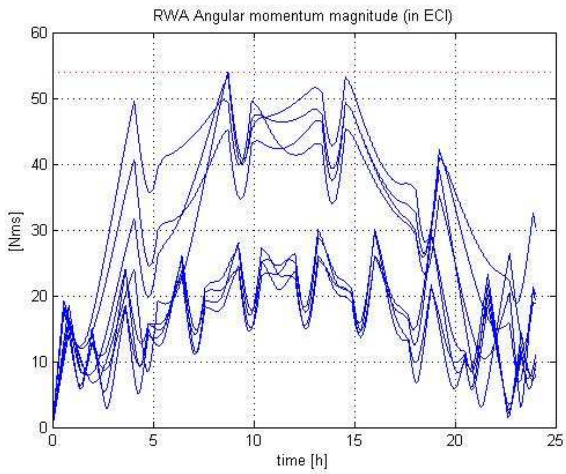

Figure 21: Set of 8 simulations (CoM Box Pessimistic Case) , 46° cant angle

Case 1 :: max RWA amom = 30.1 :: [dX,dY,dZ]=[-0.015,-0.022,-0.026]

Case 2 :: max RWA amom = 53.9 :: [dX,dY,dZ]=[-0.015,-0.022,0.066]

Case 3 :: max RWA amom = 30.1 :: [dX,dY,dZ]=[-0.015,0.022,-0.026]

Case 4 :: max RWA amom = 49.4 :: [dX,dY,dZ]=[-0.015,0.022,0.066]

Case 5 :: max RWA amom = 29.7 :: [dX,dY,dZ]=[0.015,-0.022,-0.026]

Case 6 :: max RWA amom = 53.4 :: [dX,dY,dZ]=[0.015,-0.022,0.066]

Case 7 :: max RWA amom = 29.2 :: [dX,dY,dZ]=[0.015,0.022,-0.026]

Case 8 :: max RWA amom = 49.7 :: [dX,dY,dZ]=[0.015,0.022,0.066]

As can be seen simulation #2 had the highest amom buildup (53.9 Nms), but it remains within the specified limit of 54 Nms.

Details of simulation #2 are shown in the following figures.

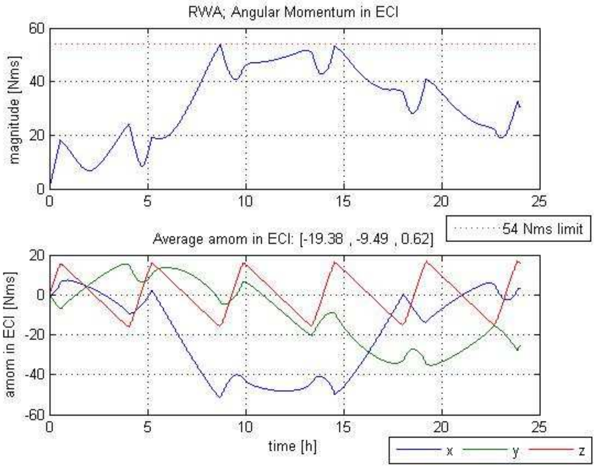

Figure 22: Simulation #2 , 46 deg cant angle, Pessimistic CoM box, Angular Momentum in ECI

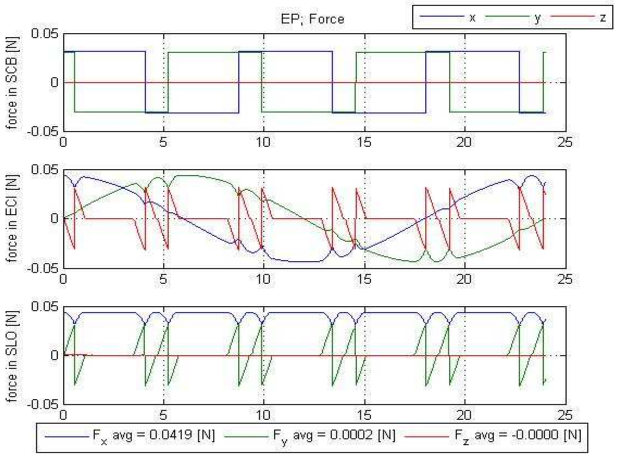

Figure 23: Simulation #2, 46 deg cant angle, Pessimistic CoM box, EP Thrust

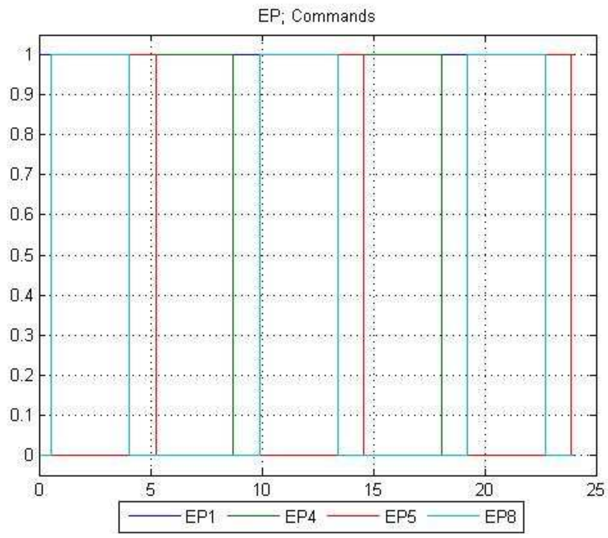

Figure 24: Simulation #2 , 46 deg cant angle, Pessimistic CoM box, EP Commands

<u>EP thruster firing times:</u>

- Thruster 1 and 5 : ≈ 1.161h
- Thruster 8 and 4 : ≈ 3.494h (=maximum firing time)

	Expected	**Obtained in worst case**
Maximum magnitude Angular Momentum build-up in the RWA	< 54 Nms	53.9 Nms
Minimum EP averaged force expected in SLO along the X-axis	> 38 mN	41.9 mN
Maximum thrusters firing time for each thrusters	< 4h	**3.494 h**

Table c: Summary of worst case results for cant angle 46°, pessimistic CoM box

K. On-board Orbit Propagator

An important component of the AOC Core software and, in fact, the SGEO platform, is the On-board Orbit Propagator (OOP). In a failure situation, Autonomous Earth Pointing guidance based upon the OOP must replace the nominal attitude profile for up to one week. The OOP must account for on-board knowledge of thrust behavior. Ground can optionally force Autonomous Earth Pointing guidance based upon the OOP instead of uploading an attitude profile.

In the chosen implementation, the orbit propagator is based on a 4^{th} order Runge-Kutta integration and the embedded software includes the following environmental disturbances:

- Earth gravity field (including tri-axiality): The WGS84 earth gravity model has a development order of 16 and a development degree of 16. Via a table of 2*153 WGS84 coefficients, the Cnm and Snm Tesseral Harmonics elements are computed. Then, through further steps of calculation, the different zonal harmonics elements, marked Jn (n=[1,16]), are obtained.
- Sun and Moon gravity field
- Solar pressure (in any configuration of the spacecraft which implies knowledge of the contributions from the solar array and antennas)
- Atmospheric drag (for use in Geostationary Transfer Orbit)

Each of these disturbances can be independently estimated with varying levels of complexity. This complexity should also be compared against the accuracy and account for the impact on the overall pointing error.

Some uncertainties must be considered as well:

- Thruster error and uncertainties are certainly the main contributor in orbit determination error. These errors include:
  - Plume impingement
  - Thrust level (about 1% of the nominal thrust level for EP, about 5% for CP)
  - Thruster pointing (worst case being about 0.1deg for EP and CP; it can be defined as function of transient at start-up of thrusters)
- Centre of gravity uncertainties
  - Its coordinate in the spacecraft body frame at a given time (millimetric)
  - Its drift over satellite life due to tanks depletion (millimetric to centimetric)
- Sensors uncertainties
  - Position, orientation of the sensors
  - Quality of the signals received, such as noise
  - Quality of the measurements to be used, such as drift and bias
- Modelling uncertainties (especially on natural disturbances)

Based on these assumptions, a first set of analysis is presented in order to quantify the propagated error (in ECI frame) that can be expected after 7days of propagations with EP based station keeping within the SGEO framework.

Therefore, the initial conditions are:

- 7days run with Station Keeping with 2009/09/24 as starting date (eclipse period)
- 30deg West longitude
- End of Life configuration (assuming see-below worst case)
  - Centre of Mass uncertainties: [-15;+22;+16]mm
  - Centre of Mass drift of +8mm over the Zscb axis
- EP thrusters: HET 40mN, Isp of 1330s
- GPS based time propagation
- **Position and Velocity <u>perfectly</u> initialised**
- **Plume impingement enabled**
- Sensors and modelling uncertainties enabled (but needs to be refined, these are for now rough uncertainties)
- **No solar pressure on antennas taken into account (dissymmetry provider)**

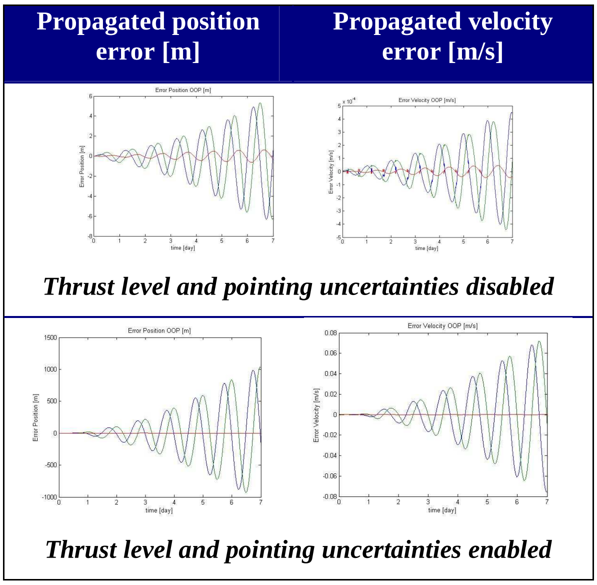

Figure 25: Preliminary estimate of the propagated error in ECI frame after 7days with/without thrust uncertainties

The first obvious conclusion drawn is that the model errors and uncertainties from EP as well as CP propulsion must be carefully identified and quantified.

Indeed thruster performance is one of the key issues here and their contribution to the position error after one maneuver cycle. One possible solution is to calibrate the thruster disturbances on-board based upon reaction wheel measurements.

The second main contributor to the error of propagation is the initial error on the position and velocity.

For a preliminary assessment, two-station ranging over 24 hours of free orbit is assumed. 1σ uncertainties of the orbital state parameters are obtained by taking the root square of the diagonal elements of the error covariance matrix.

	1-σ uncertainties
Radial position [m]	14.68
Tangential position [m]	860.37
Normal position [m]	28.66
Radial velocity [m/s]	3.30e-4
Tangential velocity [m/s]	1.68e-3
Normal velocity [m/s]	2.43e-3

Table d: 1σ uncertainties on position and velocity (initial orbit state)

Note that these uncertainties can easily be improved by a factor 4. Part of this improvement is due to the reduced systematic errors in range measurements and to the selection of stations which allow the best orbit visibility according to a given longitude. Thus, it is expected to reduce the propagated position error by a factor 4, as well.

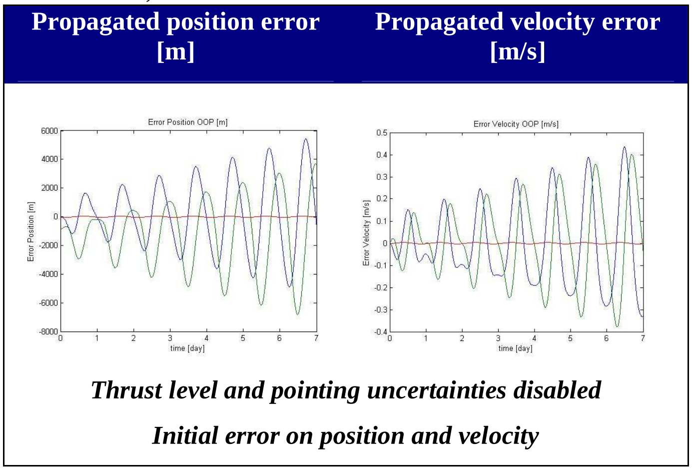

Figure 26: Preliminary estimate of the propagated error in ECI frame after 7days with initial orbit state error

And last but not least, two important issues are the execution time as well as the processor load.
A benchmarking has been done based upon a time period of 7 days with a CPU of 40 MHz. The results are acceptable.

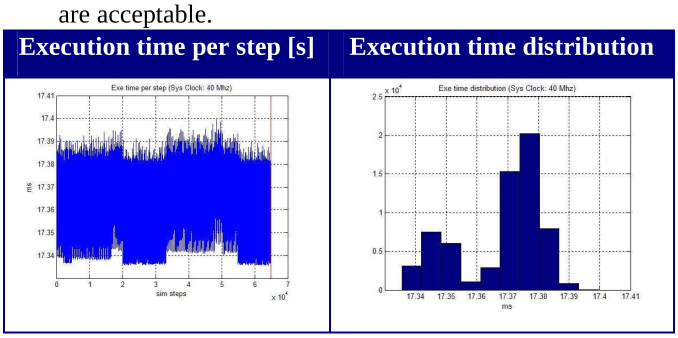

Figure 27: Preliminary benchmark on the On-board propagation execution time

L. On-board GNSS Receiver

The Hispasat AG1 will be the first commercial telecom satellite to carry a GNSS receiver into geosynchronous orbit. It will be a single chain demonstrator. The output will be sent to ground as telemetry for evaluation. The intent is to prove the feasibility of using GPS signals in geosynchronous orbit for orbit determination. Assuming a positive result, the GNSS receiver will be fully integrated into future SGEO platforms. Redundant receivers will be flown. Additional software will be added for data handling and failure detection, isolation, and recovery. The AOC Core software will be extended to use the receiver output for orbit determination and provide any necessary input, such as thrust disturbances, to the receiver[4]. Autonomous station-keeping guidance will also be implemented.
On-board usage of a GNSS receiver will give the telecommunications satellite operator a number of positive advantages:

- reduce the daily workload of ground stations during the 15 years mission lifetime,
- reduce the cost intensive GTO phase with several ground stations collaborating in a network in order to track the satellite,
- increase the mission safety through less man-machine interactions, i.e. higher satellite autonomy.

The SGEO platform will also be improved by gaining continuous access to on-board orbit determination.

- The pointing error will not increase over the week due to increasing orbit propagation error.
- The influence of EP thrust errors will be reduced. Robustness will increase since larger thrust errors will be measureable.
- It will no longer be necessary to prevent EP thruster firing for one day every week to support orbit determination by ground-based tracking.
- Improved position knowledge combined with on-board station-keeping guidance should increase precision and reduce fuel consumption.
- On-board station-keeping guidance will remove the need for ground-based EP command profile generation and up-link.

V. Conclusions

A number of advances in sensors, actuators, and software development techniques are used in the Small GEO Attitude and Orbit Control System (AOCS) design. Electric propulsion is used for all nominal station-keeping and momentum management. Attitude estimation is based upon star trackers only. These advances will contribute to making Small GEO a competitive platform in its niche of the telecommunications market.

VI. Notation

Three coordinate systems are used in the article, the Earth-Centered Inertial system (ECI), the Spacecraft Local Orbital (SLO), and the SpaceCraft Body frame (SCB). SLO is fixed at the satellite's center of mass with X_{SLO} parallel to the velocity vector and Z_{SLO} parallel to the radial vector. SLO is sometimes referred to as Local Vertical Local Horizontal (LVLH). SCB is fixed in the satellite's body and centered in the launcher plane interface with X_{SCB} normal to the East panel, Y_{SCB} normal to the South panel, and Z_{SCB} normal to the Earth deck as shown in Figure 28.

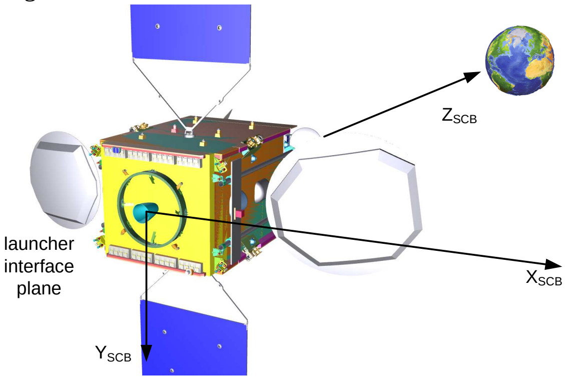

Figure 28: SpaceCraft Body frame (SCB)

VII. References

1. H. Lübberstedt, Th. Miesner, A. Winkler, P. Rathsman, J. Kugelberg, „Solely EP based Orbit Control System on Small GEO Satellite," *30th International Electric Propulsion Conference*, Florence, Italy, September 17-20, 2007.

2. Sten Berge, J. Kronander, E. Vinterhav, P. Rathsman, H. Lübberstedt, P. Zentgraf, „Advanced AOCS Design for the Small GEO Telecom Satellite," *31st Annual AAS Guidance and Control Conference*, Breckenridge, CO, USA, February 1-6, 2008.

3. Alain Demairé, Bjarne Andersson, J. Stanojev, P. Rathsman, „New Electric Propulsion missions at SSC: The use of SMART-1 heritage and new lessons learnt," *31st International Electric Propulsion Conference*, Ann Arbor, MI, USA, September 20-24, 2009.

4. Mittnacht, M., Gottzein, E., Hartrampf, M., Konrad, A., Krauss, P.A., Kühl, C, Marareskoul, D., Grechkoseev, A., and Islentjev, E., "Commercial Use of GNSS Signals in GEO", Automatic Control in Aerospace 2004, A. Nebylov Ed., Vol. 3, Elsevier Science Ltd, Oxford, UK, 2005.