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ADVANCED AOCS DESIGN FOR THE SMALL GEO TELECOM SATELLITE

Sten Berge,[*] Jon Kronander,[*] Emil Vinterhav,[*] Peter Rathsman,[*] Hendrik Lübberstedt[†] and Peter Zentgraf[‡]

This article gives an overview of the Small GEO Attitude and Orbit Control System (AOCS) design, how it meets the needs of the Small GEO target commercial missions, and the advances in the design that make it unique.

The Small GEO telecommunications satellite is a new development managed by 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. Four target commercial missions have been developed for Small GEO, each with their specific requirements on the AOCS. The prime contractor for Small GEO is OHB-System AG. The Swedish Space Corporation is a partner in the consortium and supplies the AOCS and EP subsystems.

The AOCS architecture is a three-axis stabilized system using reaction wheels for attitude control, star trackers for attitude determination, and Electric Propulsion (EP) for orbit control. The mode architecture and full sensor and actuator configurations are 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 potential 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 with 4pi steradian coverage giving low, but more than adequate, accuracy. In addition, future improvements of the design will be proposed, such as the addition of a GPS sensor in the onboard control loop.

INTRODUCTION

This Small GEO Attitude and Orbit Control System (AOCS) design has a number of advances beyond the traditional telecommunications satellite designs. Modern sensors, actuators, and software development techniques are used to improve the mission capacity 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

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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. The other partners are Oerlikon Space (Switzerland) and LuxSpace (Luxembourg).

The project is currently in the development phase (Phase B). The implementation phase (Phase C/D) will begin in the second quarter of 2008. Invitations to Tender have been released for AOCS equipment and proposals are expected in the first quarter of 2008. Negotiations for equipment to the first satellite and long term agreements for upcoming satellites will be pursued promptly after receipt of proposals. Requirements for the AOCS subsystem, the AOC Core software, and the AOCS units (Star Tracker, sun sensor, Gyrometer, and Reaction Wheels) are developed in the current phase. AOCS analysis is also performed and the most important functions in the AOC Core software are prototyped.

MISSION OVERVIEW

Figure 1 TV broadcast reference mission (OHB-System)

Four reference missions with associated payloads have been developed for Small GEO, each with their specific requirements on the AOCS (Table 1). The AOCS will accommodate these four reference missions and, in particular, the first payload mission. A proposal for the first payload mission was just recently selected by ESA. Negotiations are still ongoing.

	Reference missions			
	SMM – Ka-band mission	Hybrid P-/X-/Ka-band ComSat	Data relay satellite	TV broadcast mission
Required system half cone pointing error	±0.1° (design goal: ±0.07°)	±0.1° (design goal: ±0.06°)	±0.1°	±0.1°
Antenna symmetry	symmetric	2m dish on West panel, no dish on East panel	unequal areas	symmetric
Antenna shading of sun sensors	possible on earth deck	possible on earth deck	unlikely	unlikely

Table 1 Reference mission impact on AOCS

The Scalable Multimedia Mission (SMM) with its multiple spot-beam concept is the most demanding in terms of system pointing requirements. The Hybrid P-/X-/Ka-band communications satellite mission requires a single antenna which results in a large asymmetric torque when the antenna is active. The payloads can potentially shade the sun sensor system as well. The most common geosynchronous payload, a TV broadcast mission, is shown in Figure 1.

THE ATTITUDE AND ORBIT CONTROL SYSTEM DESIGN

The Small GEO AOCS has been designed to be both flexible and advanced. It accommodates general payloads, several launchers, and two initial orbits. An overview of the AOCS design is given below along with a discussion of the key requirements.

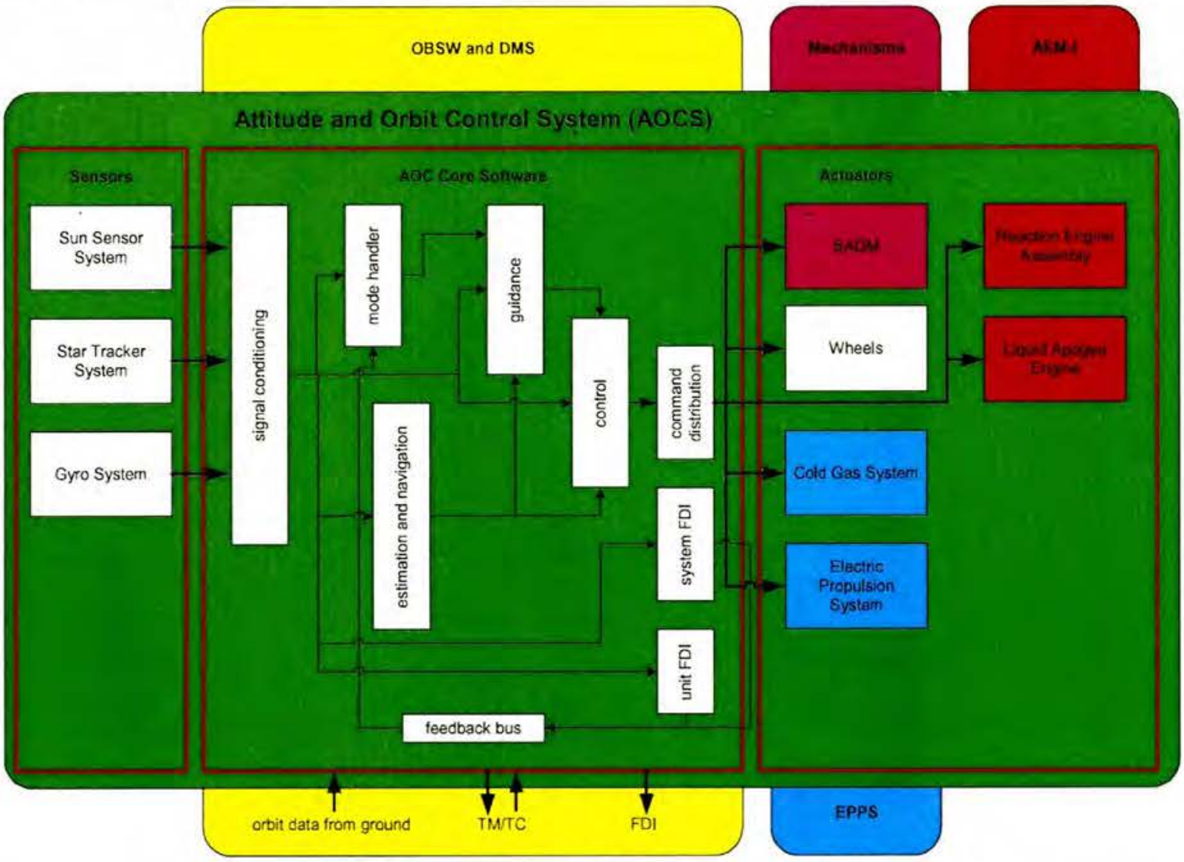

Figure 2 AOCS architecture

The overall AOCS architecture is pictured in Figure 2. It interfaces directly with four other subsystems: the On-Board SoftWare (OBSW), the Solar Array Drive Mechanism (SADM), the Integrated Apogee Engine Module (AEM-I), and the Electric Propulsion Subsystem (EPPS).

Requirements and design drivers

To accommodate a launch into GTO requires an additional bi-propellant system with one Liquid Apogee Engine, 4 nominal Vernier thrusters, and 4 redundant Vernier thrusters. For the case of direct injection into geosynchronous orbit (GEO), these thrusters and the fuel, oxidizer, and pressurant tanks are simply left out of the structure. The impact on the AOCS is the additional functionality to manage the firing of the bi-propellant thrusters. The bi-propellant system is passivated once geosynchronous orbit is reached. The electric propulsion is then used for all orbit control.

The use of electric propulsion for station-keeping is a cornerstone of the Small GEO design and profoundly affects all subsystems onboard. During station-keeping, the EP pulse times are long and often, up to 45 minutes per thruster and using four thrusters per day. This invalidates traditional free-drift orbit determination. The long pulses also induce momentum build-up in the wheels. The EP thrusters are mounted symmetrically about the center of mass so that build-up can be removed by sequential firing of thrusters. However, the reaction wheels must still accommodate up to 30 Nms during a single thruster firing.

The absolute pointing requirement for the system is 0.1° with a design goal of 0.06°. The largest error contributions come from the structure and the orbit propagation. The orbit propagation error has two error sources: the error due to imperfect calibration of the electric propulsion, and the error in the original orbit determination. A driving requirement is to not have to command the satellite more often than once per week with a goal of commanding only once every two weeks. Thus, an electric propulsion command sequence must be calculated every week based upon the most recent orbit determination and the calculated orbit propagation. In fact, the orbit propagation is an integral part of the command sequence optimization. At the end of the week, the on-board position knowledge is based upon a forward propagation of a week old orbit determination with imperfectly calibrated electric propulsion firings on a daily basis. This error in position knowledge impacts directly the pointing performance.

The requirements for off-nadir pointing and antenna mapping are not a concern for the star tracker based attitude control as compared to an earth sensor based system with a limited field of view. There is however an impact on the orbit control because the EP thrusters are fixed in the spacecraft body frame. Off-nadir pointing will result in EP thrusting with thrust components in the nadir and zenith directions that must be accounted for in the mission analysis.

In order to guarantee continuous service availability, the star trackers can be replaced for up to 10 minutes with a gyrometer without degradation in pointing performance. This translates into a requirement for a medium class gyrometer or better.

In order to maximize the efficiency of orbit injection and orbit transitions, e.g. to graveyard orbit, pointing reference profiles are uplinked from ground to accommodate non-trivial attitude references for firing of the Liquid Apogee Engine and the Electric Propulsion Thrusters along the orbit tangential. EP thrust arcs of several hours can be accommodated if the angular momentum build-up in the reaction wheels is respected. Switching EP thrusters will off-load angular momentum, but this often requires a spacecraft slew to maintain the thrust vector along, e.g. the orbit tangent.

Sensors and actuators

The star tracker is the primary sensor in the nominal modes and has therefore high reliability requirements rather than high accuracy requirements. Star tracker accuracy is inherently high. Fundamental requirements are:

- Solve "lost in space" autonomously with a few seconds observation,
- Attitude measurement at > 1 Hz,
- Very low outage times over lifetime,
- Attitude acquisition from lost in space with S/C rates a few °/sec,
- End of Life Accuracy better 0.03° all axes.

The purpose of the Gyrometer is to provide the AOCS with inertial rate measurements during all mission phases and modes throughout the lifetime of the spacecraft. In addition, the Gyrometer will be used for inertial attitude propagation during temporary Star Tracker drop-outs without compromising the pointing requirements. Reasonable expectations are:

- Bias drift ≤ 0.01 °/hour,
- High reliability for continuous use.

The bias drift requirement can be made less stringent in direct consequence to the worst case expected outage time of the star tracker.

Figure 3 A sun sensor unit consisting of a nominal and redundant cell on a substrate

The purpose of the sun sensors is to provide the AOCS with sufficient input to coarsely estimate the sun vector in the spacecraft reference frame during all mission phases and modes throughout the lifetime of the spacecraft. In addition, the sun sensors should have no blind spots. Thus, confusion between albedo, straylight, and sun is not possible. Sun presence detection is unequivocal. sun sensor failure detection is also straightforward for this configuration. Cold redundancy is inherent in the design of each sun sensor unit which shall consist of a cell pair (Figure 3), one nominal cell and one redundant cell. Cell pairs are mounted on brackets.

The number of required cells is at least 6 pairs. Only six pairs or one pair per panel would be required on a simple box structure. A higher number of cells may be required to account for shading, e.g. of the solar array wing or the antennas. At least one cell pair on each of the three simultaneously lit spacecraft panels must be visible for every possible sun vector in the spacecraft body coordinate system. A possible mounting scheme is shown in Figure 4.

The sun presence system must be able to autonomously and unequivocally determine if the sun is visible to the spacecraft, i.e. not eclipse. The required measurement accuracy is low given the large solar array and the low power requirements in Safe mode when the sun sensors will be used to point the spacecraft. Thus, desirable characteristics are:

- Omni-directional coverage for sun presence system,
- Eclipse detection,
- Sun vector resolution error half angle $\leq 15°$.

Available sun sensors currently on the market tend to fall into one of three categories: fine sun sensors, pyramid-based sun acquisition sensors, and simple cells. The first two categories often require additional simple cells strategically placed on the spacecraft body to provide omni-directional sun presence detection. Given the pointing accuracy needs for this application, a system based entirely upon simple cells is possible.

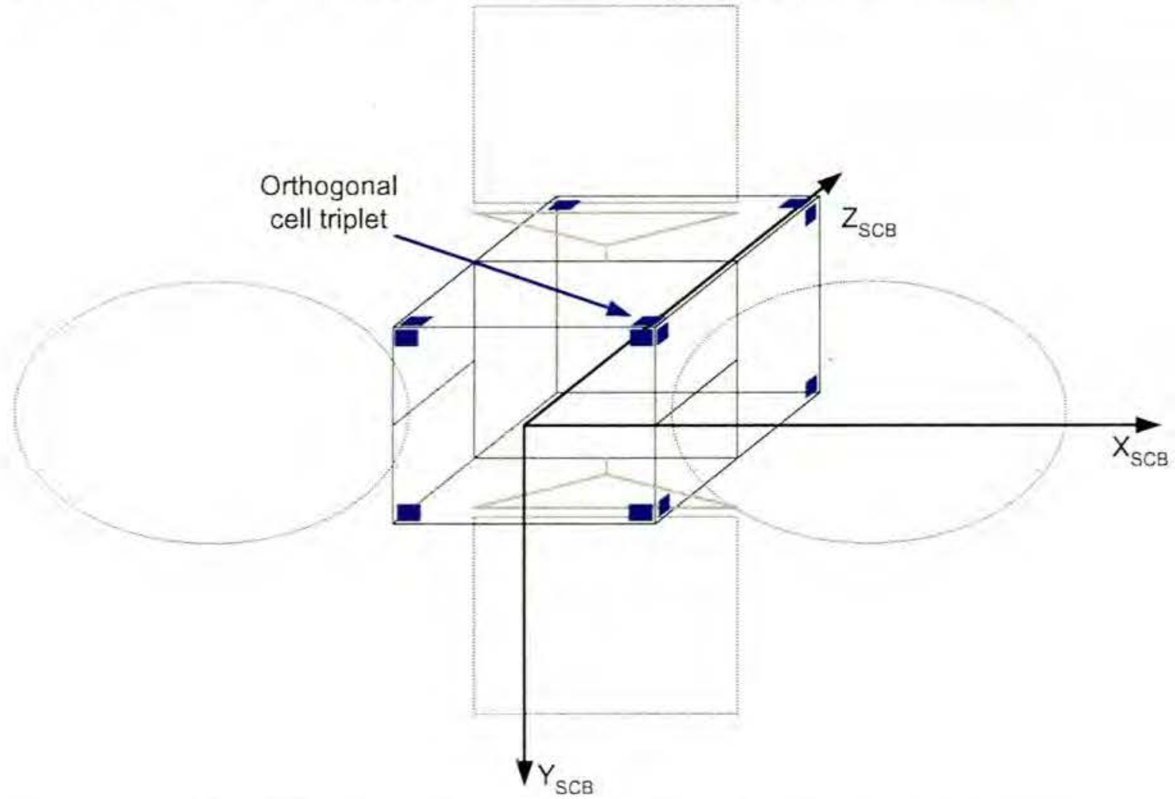

Figure 4 Possible mounting of sun presence cell pairs on spacecraft

There is an intention to fly a GPS receiver as an on-board flight demonstration to qualify its use in geosynchronous orbit. It will not be used on-board, but data will be downlinked to support orbit determination on ground. In future satellites, an on-board GPS receiver may provide continuous autonomous orbit determination. This will drastically reduce the pointing error due to orbit determination and propagation.

The four reaction wheels will nominally be operated in hot redundancy in a pyramid configuration. All four wheels can then be spun up to a set point such that the nominal speed variation never approaches zero. The total spacecraft angular momentum will still be zero. The failure case with one wheel deactivated will also be supported. This will require zero crossings of the three remaining wheels. The primary needs of the reaction wheels are:

- Momentum capacity > 33 Nms,
- Torque > 31 mNm,
- Zero-crossings qualification > 11000 (2 per orbit for 15 years).

A one-axis Solar Array Drive Mechanism (SADM) is needed for each solar array wing to support sun pointing of the solar array while simultaneously earth pointing the payload. For re-orbiting manoeuvres, such as transition to graveyard orbit, a single EP thruster can be pointed along the orbit tangent while the spacecraft is rolled and the SADM actuated to maximize solar power generation.

The cold gas thruster assembly is used only for rate damping and momentum management in the Safe modes. Thus, the system is only used for generation of torques, i.e. no forces. A 50 mN Xenon cold gas system has been baselined feeding from the same Xenon tank used by the Electric Propulsion. The AOCS accounts for the flow limitations that arise from sharing a Xenon supply system with the EP.

Eight EP thrusters have been baselined. They are fixed on the spacecraft body symmetric about the Center of Mass. The thrust vectors are ideally perpendicular to the nadir vector. Mission analysis has shown that all current European EP developments could be used, from Hall Effect Thrusters to Gridded Ion Engines.

AOCS mode architecture

The mode architecture is shown in Figure 5. The entrance to the state machine at start up is via the black dot. The grey arrows represent commanded mode transitions while the black arrows represent autonomous transitions. The direction of the possible transition is indicated by the arrow head, e.g. a line with arrow heads on both ends indicates that the transition is possible in both directions.

The Safe modes: Standby Mode (SBM), Sun Acquisition Mode (SAM), and Attitude Hold Mode (AHM) are shown in the upper part of the diagram. This mode group is entered autonomously at start-up or after a failure. The Sun Acquisition Mode can use either reaction wheels or cold gas as primary actuator. If healthy, reaction wheels are the preferred actuator. The Nominal modes: Normal Mode (NM), Electric Propulsion Control Mode (EPCM), and Chemical Propulsion Transfer Mode (CPTM) are shown in the lower part of the diagram. This mode group is used for nominal operations and geosynchronous orbit insertion.

The autonomous switching criteria between the modes for every allowed transition are indicated in Table 2. A grey cell indicates that an autonomous transition is not possible. An "autonomous switch to Safe" can be attributed to numerous failure cases which make it impossible for the AOCS to retain the higher mode {NM, EPCM, CPTM}, e.g. extended loss of star tracker.

		from					
		SBM	SAM	AHM	NM	EPCM	CPTM
to	SBM						
	SAM	Large angular momentum OR Loss of sun pointing		Large angular momentum OR Loss of sun pointing			
	AHM				Autonomous switch to Safe	Autonomous switch to Safe	Autonomous switch to Safe
	NM					Autonomous end of Momentum Management or Station-keeping	
	EPCM				Autonomous initiation of Momentum Management		
	CPTM						

Table 2 Description of autonomous transitions between modes

Each mode will configure a set of AOCS functions. The AOCS functions include several control laws depending on the reference, e.g. sun vector or Chebyshev polynomial, and the actuator, e.g. reaction wheel, electric propulsion, cold gas, or Vernier thruster. In addition, there are several reaction wheel momentum management functions. Guidance functions are the uplinked quaternion reference, in the form of Chebyshev polynomials, or an autonomous earth pointing guidance based upon the on-board orbit propagation.

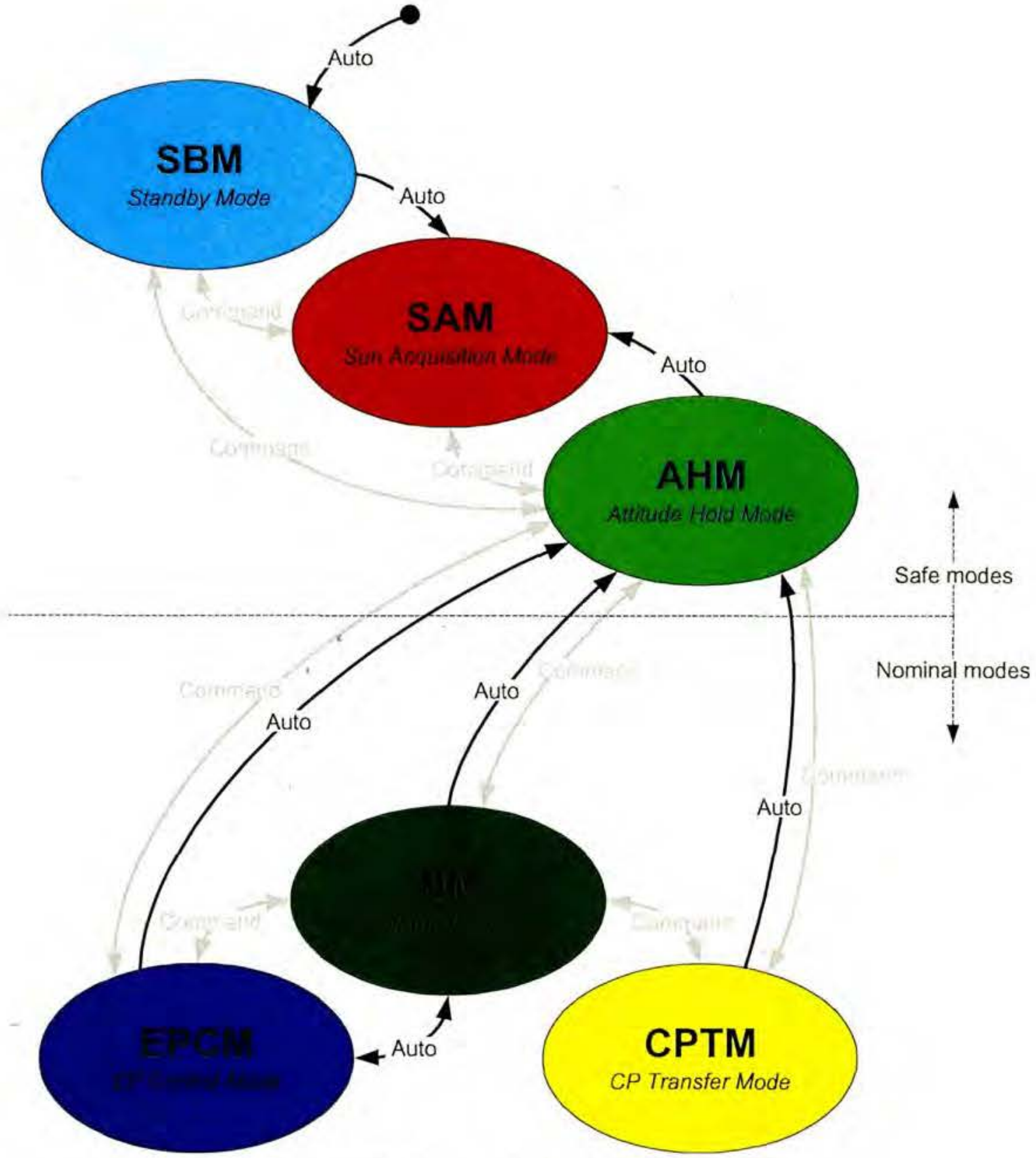

Figure 5 AOCS mode architecture diagram

ADVANCES IN THE ATTITUDE AND ORBIT CONTROL SYSTEM

The most significant advances are discussed separately below.

Software design concept

The AOCS software is an input/output state-space system developed in MATLAB/Simulink®. C-code will be generated from the models using Real Time Workshop (RTW). The software function, called a core, is then compiled and linked into the rest of the on-board software, the Data Management System. This concept has been demonstrated in previous satellites built by Swedish Space Corporation such as ESA's SMART-1[2] and PRISMA[3]. The AOC Core includes guidance, navigation, and control functionality including an AOCS mode handler, signal conditioning, command distribution, and AOCS related failure detection, isolation, and some subsystem level recovery. A simulator core is developed in parallel with the AOC Core for early closed loop testing. The simulator core contains dynamics, environment, sensors, and actuators. The closed loop with the simulator core will be used throughout the software lifecycle for pre-verification and debugging. Integration tests of the AOC Core are run on an emulator of the target processor before delivery to the On-board Software. The entire On-board Software is then tested on the target processor as shown in Figure 6. Experience has shown that software development with MATLAB is very efficient with high quality.

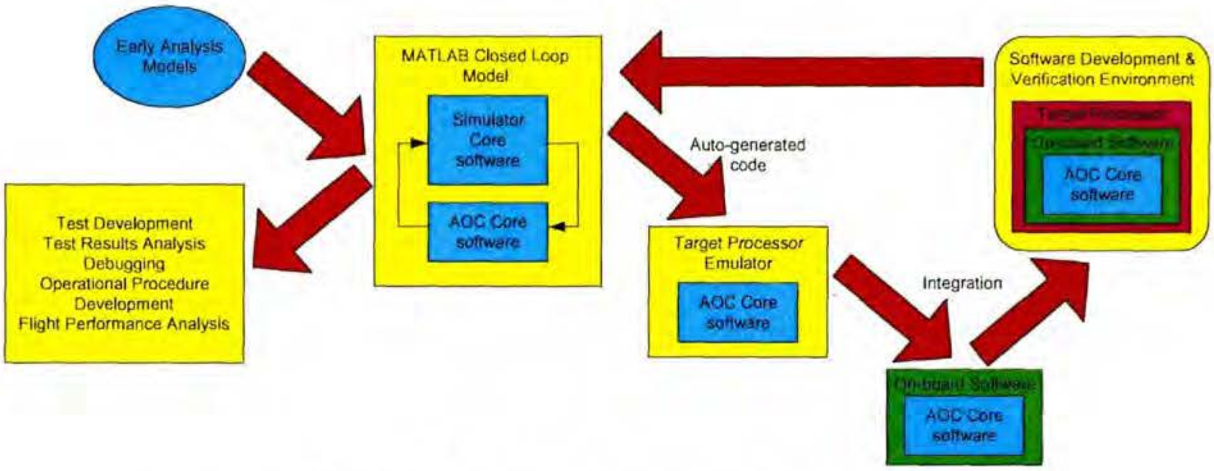

Figure 6 AOCS software development chain

Station-keeping with Electric Propulsion

Electric Propulsion (EP) has a very clear advantage over traditional chemical propulsion systems thanks to its high specific impulse[1]. A number of different EP configurations have already flown in geosynchronous orbit. The chosen thruster configuration is pictured below in Figure 7. Note that each thruster off-points from the center of mass in all three axes. The off-pointing defines an envelope in which the center of mass will move over the lifetime of the spacecraft. The spacecraft has been constructed so that the center of mass movement is as small as possible and with small deviations in location for both the direct injection and GTO spacecraft configurations.

The electric propulsion thruster configuration presented here has been analysed for applicability to Small GEO. Use of EP presents a number of key challenges for the mission analysis and AOCS, such as: combined manoeuvres, off-nadir pointing, EP firing sequence, angular momentum build-up limitations, eccentricity control, and orbit relocation, e.g. transfer to graveyard orbit.

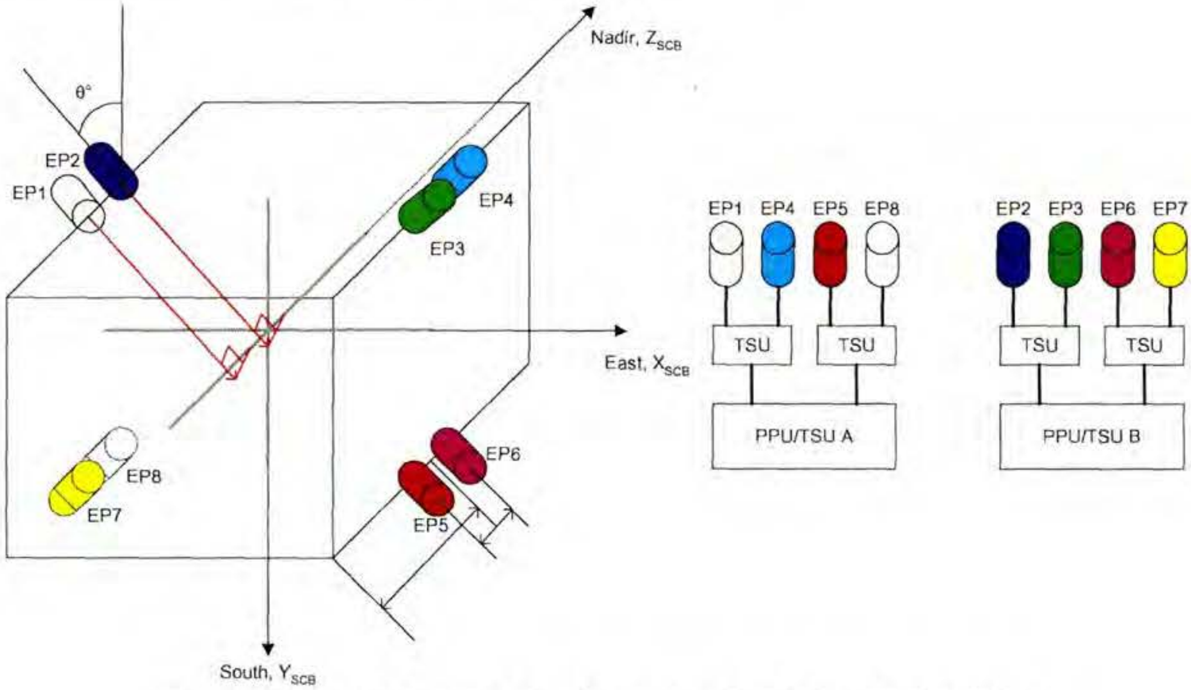

Figure 7 Redundancy concept for Electric Propulsion thruster configuration

A single branch shares one Power Processing Unit (PPU), e.g. EP1, EP4, EP5, EP8. North-South station-keeping at a single node will result in an angular momentum increase which can be removed at the other node. The mission analysis is based upon use of a single branch only. For half an orbit, the angular momentum is stored in the wheels. For example, a nominal South manoeuvre will result in an angular momentum build-up around Y of a few Nms. At the opposite node, the North manoeuvre removes this build-up.

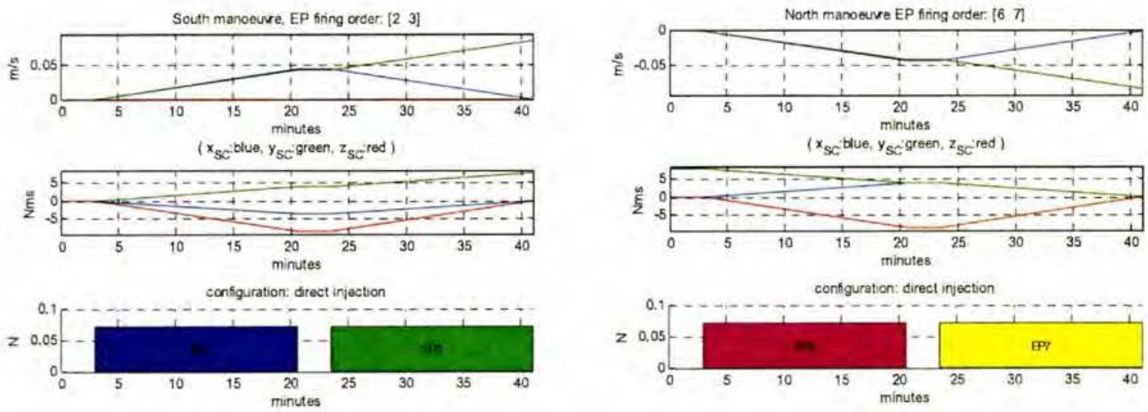

Figure 8 Ideal South and North manoeuvres

As displayed above in Figure 8, an equal division of one day's North-South station-keeping over the two nodes will completely remove all angular momentum build-up. This is no longer true in a combined manoeuvre where an East/West component is present at every node. In general, this component will always be in the same direction for a specific slot due to the longitudinally varying acceleration of the earth's tri-axiality. There will be a resultant in-plane angular momentum build-up as shown in the plot below of the simulation of a nominal combined manoeuvre strategy. There is no resultant out-of-plane angular momentum build-up because the North and South components at each node are equal. This principle is used for out-of-plane momentum management.

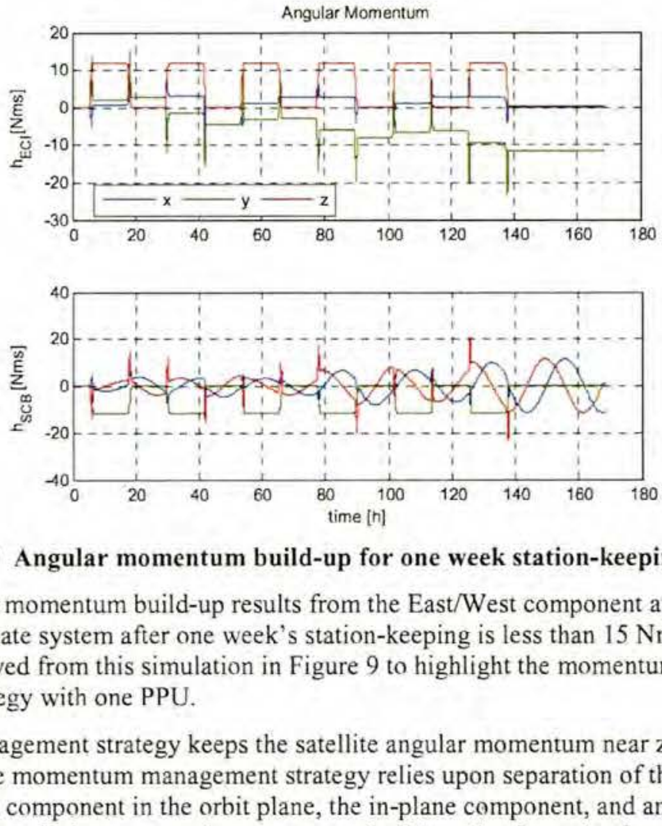

Figure 9 Angular momentum build-up for one week station-keeping strategy

The in-plane angular momentum build-up results from the East/West component at every node. The build-up in the spacecraft coordinate system after one week's station-keeping is less than 15 Nms in Figure 9. All torque disturbances were removed from this simulation in Figure 9 to highlight the momentum build-up due entirely to the station-keeping strategy with one PPU.

The momentum management strategy keeps the satellite angular momentum near zero or another set point specified by ground. The momentum management strategy relies upon separation of the inertial angular momentum vector into a component in the orbit plane, the in-plane component, and an out-of-plane component. Environmental disturbances induce an angular momentum build-up that is predominantly out-of-plane. The predominant in-plane angular momentum build-up is a side-effect of the single PPU station keeping profile needed for East/West station-keeping.

Off-loading of the in-plane component relies upon the fact that there are two thruster pairs per redundant side that can generate force free torque around the Nadir/Zenith axis. Every PPU has one EP pair that generates Nadir torque and one pair that generates Zenith torque. Twice per orbit, the in-plane component of the inertial angular momentum is aligned with the Nadir/Zenith axis. Once per week, two EP thrusters should be fired to off-load the in-plane component. The firing times are similar to those for North-South station-keeping.

Off-loading of the out-of-plane component relies upon the fact that the EP configuration generates opposing torques for North and South correction manoeuvres. Dividing the inclination ΔV equally between the nodal firings will result in inertial angular moments that null each other out. However, a net angular momentum build-up in either the North or South directions can be generated by redistributing the ΔVs between the nodes. This should have no effect on the total inclination ΔV for the orbit. The effect on the other orbital elements is still being analysed.

The weekly momentum management strategy requires no more than one additional pulse from each EP thruster per week. In combination with the nominal EP station-keeping profile with manoeuvres at every node 6 days per week, the total number of pulses per thruster per week is six for the two EP thrusters that only perform station-keeping and out-of-plane momentum management and seven for the other two EP thrusters that fire the additional pulses for in-plane momentum management. No EP firings are performed on the seventh day so as to not disturb the orbit determination. Momentum management can be done by ground profile command or autonomously or both.

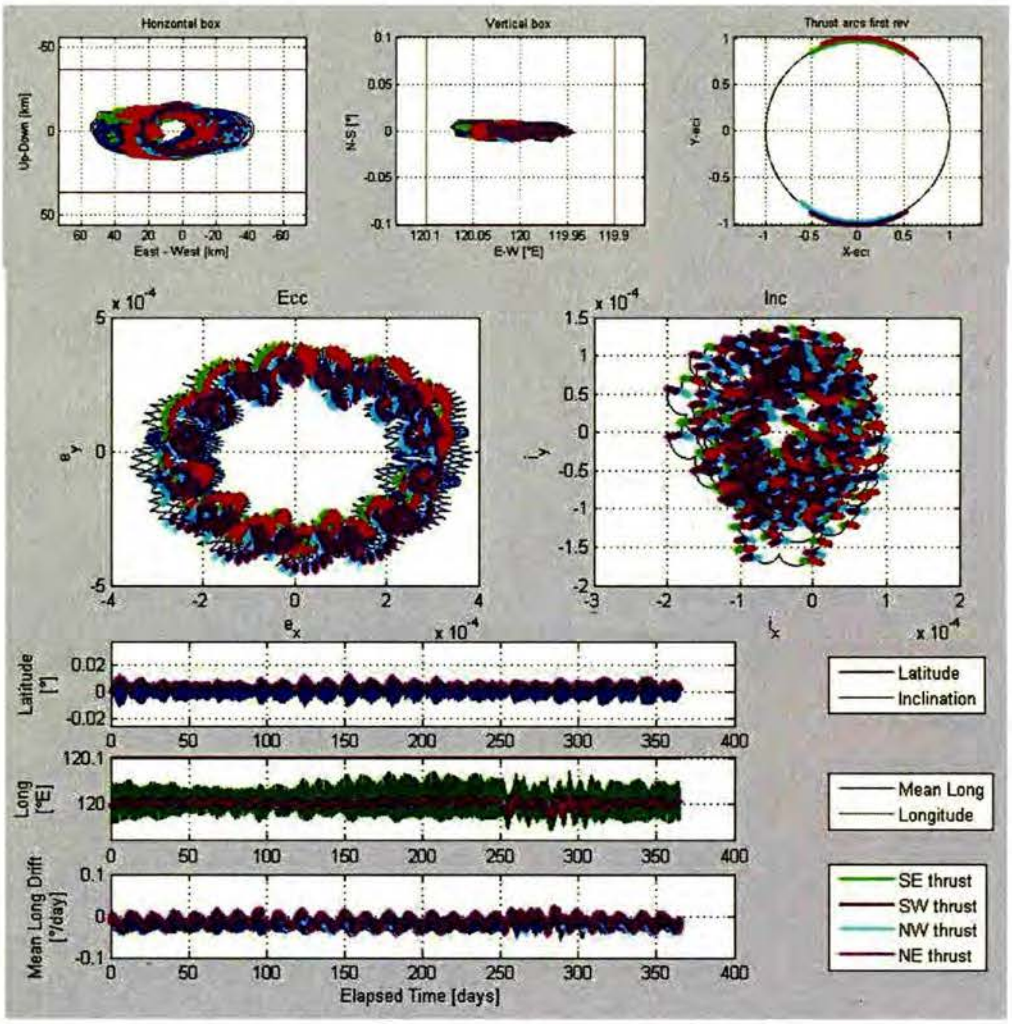

Figure 10 One year of station keeping with electric propulsion

Out-of-plane momentum management is spread evenly over the six days with manoeuvres. This corresponds to redistribution of the inclination ΔV components at each node to generate a few Nms every manoeuvre. The East/West component must of course remain undisturbed.

In-plane momentum management is performed once per week at one of the two conjunctions of the in-plane angular momentum vector and the spacecraft's Zenith/Nadir axis. At the chosen conjunction, two EP thrusters are fired to generate torque in the correct direction. If one conjunction is in eclipse, the nominal momentum build-up can be off-loaded at the other conjunction.

The EP command profile controls the orbital elements while simultaneously generating a residual out-of-plane torque over the orbit. The sign and magnitude of this torque can be chosen to off-load the out-of-plane component of the momentum build-up within the framework of the nominal station-keeping EP command profile. Thus, only the in-plane momentum build-up will require additional EP thrust activity.

A one year non-linear simulation with automated optimization of electric propulsion thrust including momentum management was performed as proof-of-concept. The result is shown in Figure 10. In practice, two week thrust command and attitude reference profiles will be generated every week on ground and up-linked to the satellite. This analysis demonstrates the capability to plan combined manoeuvres with only the nominal branch of the electric propulsion thruster configuration to simultaneously control East/West drift, eccentricity, and inclination. Even the nominal out-of-plane momentum management is included in the thrust sequence optimization. The fuel consumption and accumulated impulse for this simulation is shown in Figure 11.

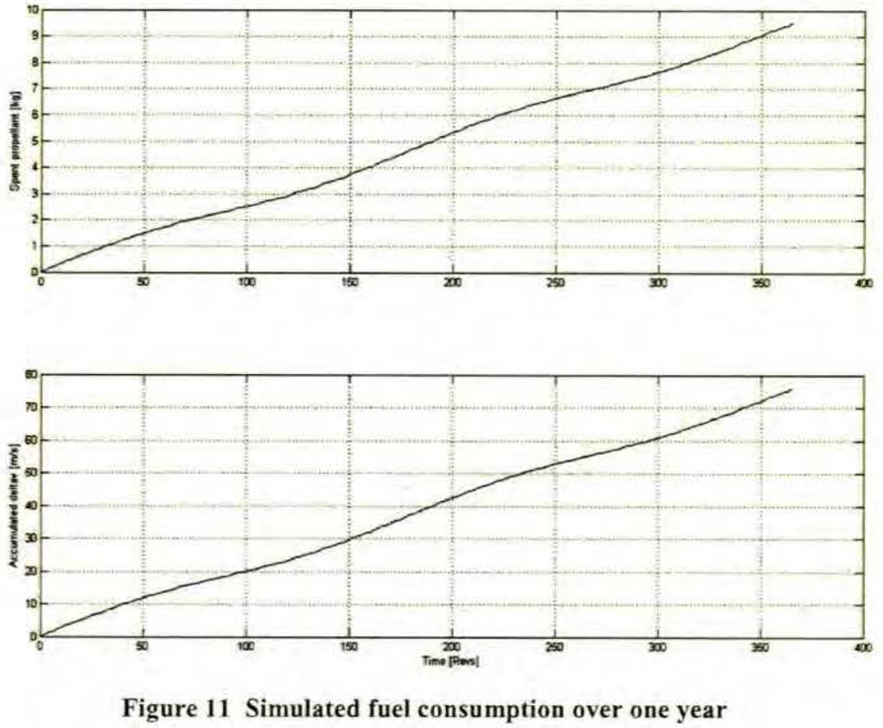

Figure 11 Simulated fuel consumption over one year

Star Tracker based attitude control

Pointing of Small GEO will be based upon star trackers with no support from an earth sensor. This places implicit requirements on orbit position knowledge. Orbit determination and propagation are discussed in a separate section below. Otherwise, there are a number of advantages in using star tracker based attitude estimation. Most importantly, absolute attitude can be measured for any spacecraft attitude limited only by possible blinding of the star tracker. This is the primary advantage when compared to an earth sensor and its limited field of view. The baseline for Small GEO is two hot redundant star trackers. This will be especially useful during transfer with electric propulsion when the attitude will vary considerably. Earth sensor based AOCS designs require a high performance gyrometer to accommodate attitudes that are not Earth pointing, e.g. while firing the liquid apogee engine. Propagation of attitude with a gyrometer is time limited due to drift. The star trackers will give absolute measurements at greater than 1 Hz. Another important improvement with star tracker based attitude measurement is independence from eclipse. Earth sensor based systems require extra attention during the eclipse season. Star tracker based attitude estimation results in a very flexible platform for the operator.

Another advance is the potential reliance upon APS-based star trackers. 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. The potential APS detectors have been designed for space application with radiation-hard design techniques making them tolerant to gamma dose levels up to 10 MRad(Si). The first APS-based star tracker will fly on ESA's PROBA-2 in 2008.

As discussed in the requirements section above, a system decision was taken to fly a gyrometer on-board that could replace the star trackers for 10 minutes without degradation in pointing performance. The gyrometer will function primarily as a failure detection and recovery sensor used to replace the star tracker during unexpected outages and as primary sensor in the Safe modes. It is also valuable as a complementary sensor when only one star tracker is available. Even during the firing of the Liquid Apogee Engine, the star trackers will be the primary attitude sensor. Preliminary analysis shows that the largest accelerations that could cause perceived jitter of stars is the firing of the Vernier thrusters. The thrust noise from the Liquid Apogee Engine appears to be small in comparison. An APS-based star tracker with a high tracking rate should be able to provide attitude measurements in the presence of these thruster accelerations. The first flight will confirm the performance of the star tracker during the firings of the Liquid Apogee Engine.

Fault tolerance

Small GEO has high reliability requirements. The satellite must be able to sustain a single hardware or software failure or operator error. Even short outages on the order of minutes carry heavy economic penalties. These requirements can affect the AOCS design differently, e.g. reliability calculations versus functional redundancy. Reliability for the star trackers, or any electronic unit, is improved with cold redundancy. However, hot redundancy is still required for the star trackers due to potential lunar blinding. This particular decision may be revisited since APS-based star trackers, if procured, have improved tolerance to moon in the field of view.

Autonomous Failure Detection, Isolation, and Recovery (FDIR) is built into the Small GEO software. The FDIR monitors and reconfigures the spacecraft as necessary. All autonomy can, of course, be disabled. A portion of the FDIR application software is integrated into the control loop, checking sensor data for validity before it is used by the AOC Core software. Likewise, the AOC Core software carries system detection and unit detection and isolation algorithms whose outputs go to the FDIR monitoring and reconfiguration algorithms. For certain system detection, the AOC Core will autonomously recover as indicated in Table 2. This FDIR framework allows for early software design before full information on units is available. The structure, including recovery actions and interfaces to application software components like the AOC Core, can be fixed while error detection algorithms can be added later as the project progresses. The interfaces to the AOC Core software, including the FDIR related signals, are shown below in Figure 12.

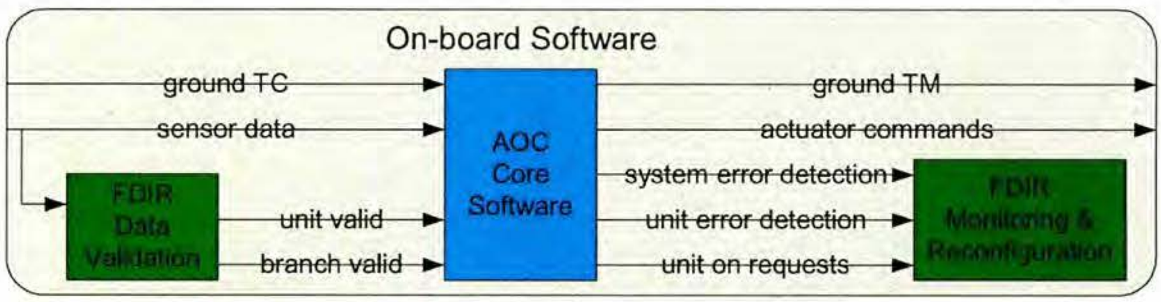

Figure 12 FDIR framework in relation to AOCS software

Orbit determination and propagation

Several orbit determination techniques have been analysed for Small GEO. Two-station ranging is the preferred baseline. Use of GPS for on-board orbit determination is not qualified for geosynchronous orbit and therefore not currently an option. As stated in an earlier section, the current plan is to fly a GPS receiver as a demonstrator on Small GEO.

The orbit determination and propagation are drivers for the pointing budget. Without an earth sensor, Small GEO is dependent upon knowledge of its position in space to point to the earth. Another driver is up-linking of commands to the satellite not more often than once a week. Thus, the orbit must be propagated forward one week. At the end of the week, the position error has two primary contributors: the error in the initial orbit determination and the error in the calibration of the EP thrust vectors. This position error is required to be less than 8 km (3σ) for Small GEO. Note that this accuracy or better is expected for co-located satellites.

Current analysis is confident that the position knowledge requirement is easily met with on-board GPS-based orbit determination, even without EP thrust calibration. On-orbit calibration procedures using for instance wheel measurements to estimate the EP thrust level, may further reduce the parasitic forces. The on-board orbit propagation integrates actual thrust commands in real time, thereby accounting for missed or modified thrusts.

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. The design accommodates both direct injection and GTO launches. These advances will contribute to making Small GEO a competitive platform in its niche of the telecommunications market.

NOTATION

Two coordinate systems are used in the article, the Earth-Centered Inertial system (ECI) and the SpaceCraft Body frame (SCB) which 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 13.

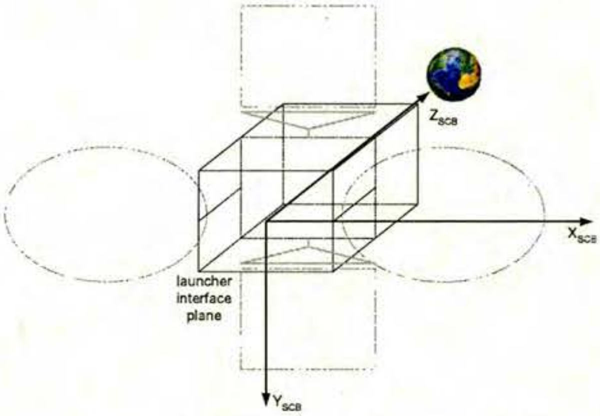

Figure 13 SpaceCraft Body frame (SCB)

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