30th June 2013

SAR Formation Flying

Annex 8. Orbit Control Analysis

Document Version: V01_00

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Revision History

Version No.	Date	Author	Description of Change
V01_00	30th June 2013	ASU	Initial Issue

1 Table of Contents

2 Executive Summary

This report presents the design drivers, trade-offs, analysis and baseline solution for the Garada AOCS.

The key drivers in the design of the AOCS are: the large, variable and asymmetric inertias, which in turn drive the acquisition and safe mode strategies due to the potentially large gravity gradient disturbance torques which can be generated; the provision of a manoeuvre (slew) capability to enable sideways pointing; and normal mode performance which in turn drives the architecture. A view to exploiting heritage has been maintained in order to attempt to reduce costs.

The key trade-offs and baseline solutions are:

- A modified Bdot magnetic control strategy for acquisition and safe mode, which works with the gravity gradient disturbances;
- A simple star tracker only normal mode solution (i.e. no gyro required) with reaction wheels for actuation;
- A star tracker optical head configuration to allow continuous visibility.

The baseline hardware solution for Garada consists of:

- Star Tracker – a multiple optical head configuration which provides continuous coverage even in the event of a single failure to provide absolute attitude information during the nominal mission;
- GNSS – provides orbital position to the on-board navigation and guidance functions;
- Coarse Sun Sensors – provides a measure of the sun direction for use in sun acquisition, attitude anomaly detection and eclipse detection.
- Magnetometer – provides measurement of the geomagnetic field as inputs to the magnetic control law.
- Magnetorquer – provides simple and robust actuation using the geomagnetic field in acquisition and safe modes, and allows momentum management of the reaction wheels during the nominal mission;
- Reaction Wheels – a set of 4 wheels (allowing for a single failure) provides the nominal control actuation during the nominal mission.

3 Introduction

3.1 Purpose & Scope

This document presents the design for the Garada Attitude and Orbit Control Subsystem (AOCS).

4 Applicable and Reference Documents

AD01 Garada AOCS Requirements Specification

AD02 Garada AOCS Assumptions

5 Abbreviations

AOCS	Attitude and Orbit Control Subsystem
APE	Absolute Performance (Pointing) Error
APS	Active Pixel Sensor
ASM	Acquisition and Safe Mode
ASU	Astrium Ltd
CCD	Charge-Coupled Device
CESS	Coarse Earth Sun Sensor
CoM	Centre of Mass
CoP	Centre of Pressure
CSS	Coarse Sun Sensor
EU	Electronic Unit
FDIR	Failure Detection, Isolation and Recovery
LTAN	Local Time of Ascending Node
MTM	Magnetometer
MTQ	Magnetorquer
NM	Normal Mode
OCM	Orbit Control Mode
OH	Optical Head
PUS	Packet Utilisation Service
RW	Reaction Wheels
SEU	Single Event Upset
SI	International System of Units
STR	Star Tracker
TBC	To Be Confirmed
TBD	To Be Determined

6 Drivers

The main objective of the AOCS design has been to define a viable AOCS architecture meeting the platform requirements while identifying ways to maximize re-use of current AOCS elements in order to save cost.

Two key areas of the Garada spacecraft are pointing performance and manoeuvre capability, i.e. between two different observing attitudes.

In addition, due to the difference between the launch and operational configurations, the deployment strategy, which has to cope with widely varying inertias, is another key area.

Some of the key considerations are indicated in the next section.

6.1 Key Design Considerations

- Acquisition and Safe Mode strategy:
  - Thruster-based or magnetic-based acquisition?
  - Sun pointing or earth pointing?
  - Can the same strategy be used for initial acquisition (off-launcher) and as a contingency mode?
- Normal Mode architecture:
  - Gyroless (star tracker only) or gyros + star tracker?
  - Sensor configuration?
  - Actuator configuration?
  - Controller design?
- Large, and varying, inertias:
  - The inertias are very large which affects:
    - Control design and performance
    - Acquisition and contingency mode strategies
  - The inertias vary between stowed and deployed
    - Can common elements between stowed and deployed be used?
- Disturbance environment:
  - Which disturbances dominate and thus drive the design?
- Manoeuvre capability:
  - Slews between nadir and sideways looking (plus for delta-V attitudes).

6.2 Disturbance Environment

The ability to predict the behaviour and control the attitude of the spacecraft requires and understanding of how environmental disturbance torques will influence the attitude of the spacecraft. The magnitude of these disturbance torques will drive the definition of the AOCS system. In particular the size of the disturbance torques experienced can influence the attitude acquisition strategy, actuator sizing, normal mode control design and tuning and the achievable performance in the normal mode.

The following sections provide an estimate of the expected disturbance torques for two altitudes 580 km and 630 km.

6.2.1 Atmospheric Drag

The atmospheric drag on the spacecraft is strongly influenced by the density of the atmosphere which in turn varies significantly with orbit altitude and solar activity. For this assessment the atmospheric densities (from JB-2006) and shown in Table 6-1 were used, as illustrated in Figure 6-1.

H (km)	Low activity	Moderate Activity	High activity (long term)	High activity (short term)
560	1.96E-14	3.58E-13	1.52E-12	2.66E-12
580	1.47E-14	2.71E-13	1.22E-12	2.18E-12
600	1.14E-14	2.06E-13	9.82E-13	1.79E-12
620	9.10E-15	1.57E-13	7.93E-13	1.48E-12
640	7.41E-15	1.20E-13	6.43E-13	1.23E-12

Table 6-1: Altitude profiles of total atmospheric density (JB-2006)

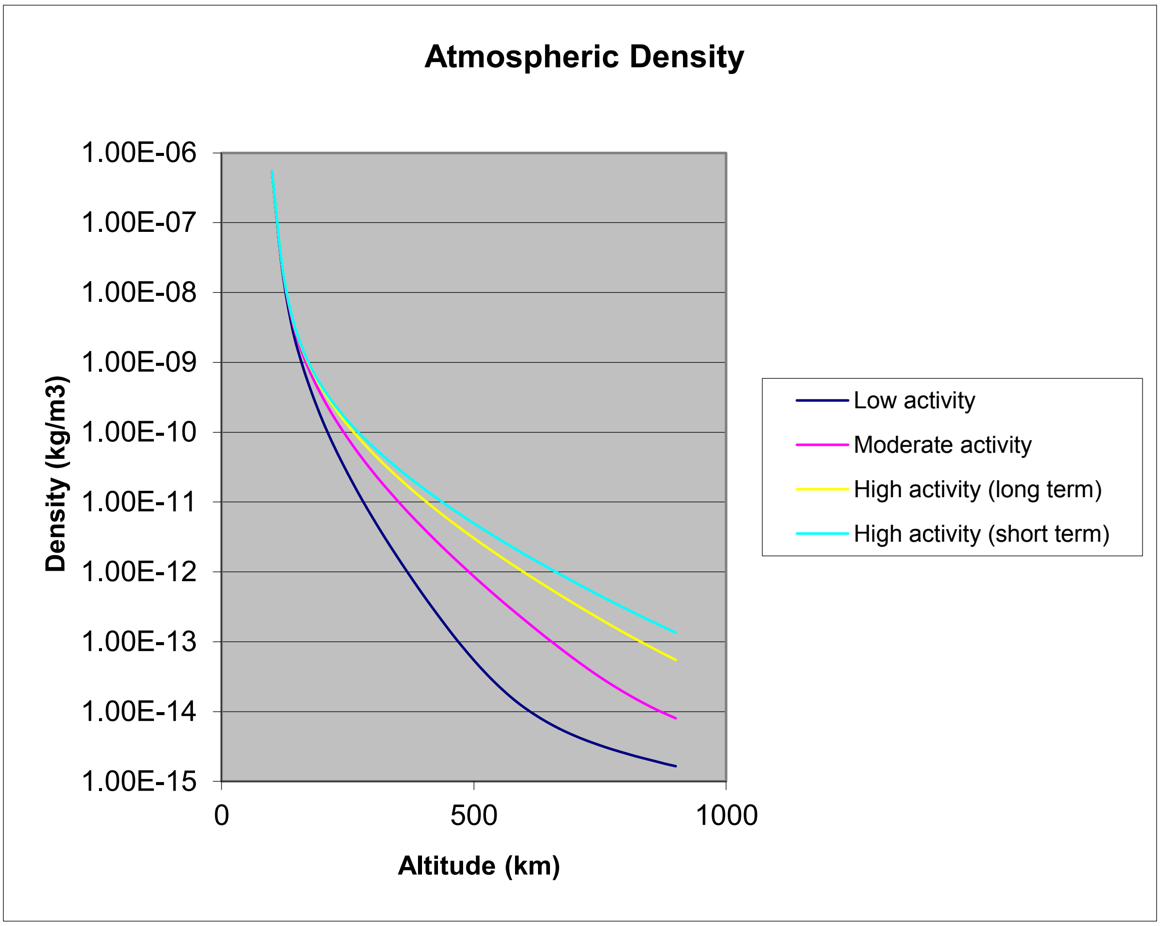

Figure 6-1: Altitude profiles of atmospheric density (illustrated)

The spacecraft geometry was considered to determine the spacecraft surfaces exposed to the atmosphere in the direction of travel, i.e. along the x-axis.

In nominal attitude, the spacecraft cross-section along the velocity vector is approximately 6 m^2. Given an offset between the Centre of Pressure (CoP) and Centre of Mass (CoM) of 0.2 m and a drag coefficient C_d of 2.2, this generates drag forces, F, and torques, T, for the various atmospheres as shown in Table 6-2 using:

$$F = \frac{1}{2} C_d \rho A V^2,$$

$$T = r \times F.$$

Where ρ is the atmospheric density, A is the area, V is the spacecraft velocity and r is the CoP offset.

	H (km)	Low activity	Moderate activity	High activity (long term)	High activity (short term)	
Force	580	0.01	0.10	0.47	0.84	mN
	630	0.00	0.05	0.25	0.47	mN
Torque	580	0.00	0.02	0.09	0.17	mNm
	630	0.00	0.01	0.05	0.09	mNm

Table 6-2: Drag Forces and Torques (generated perpendicular to the x-axis)

6.2.2 Solar Radiation Pressure

Solar radiation pressure on the spacecraft is strongly dependent on the spacecraft geometry and optical surface properties of the surface being illuminated. The radiation force is given generally by:

$$F = P\left[(1 - C_s)\hat{S} + 2\left(C_s \cos\theta + \frac{1}{3} C_d\right)\hat{H}\right] \cos\theta \, A.$$

Where P is the mean momentum flux given by $P = \frac{F_e}{c}$ (F_e is the solar constant and c is the speed of light), $\hat{S}$ is the unit vector from the spacecraft to the sun, θ is the angle between $\hat{S}$ and $\hat{H}$, the unit normal to the surface area being considered. The coefficients C_s and C_d are the fraction of incident radiation that is specularly and diffusely reflected respectively. Together with a further absorption coefficient C_a they sum to unity, i.e. $C_s + C_d + C_a = 1$.

In nominal attitude, the forces and torques have been calculated for an offset between the CoP and CoM of 2.0 m along the x-axis, 0.1 m along the y-axis and 0.2 m along the z-axis which generates drag forces, F, and torques, T, for the winter solstice when the solar flux is close to maximum, and are shown in Table 6-3.

		SRP (winter solstice)	
Force	**x**	0.16	mN
	y	0.00	mN
	z	0.06	mN
Torque	**x**	0.13	mNm
	y	0.17	mNm
	z	0.31	mNm

Table 6-3: Solar Radiation Pressure Forces and Torques

6.2.3 Gravity Gradient

The gravity gradient torque is influenced mainly by the spacecraft inertias and the orbit altitude. The torque is given by the expression:

$$C_{GG} = \frac{3\mu}{r_0^3}\vec{z}_0 \times I_G\vec{z}_0.$$

Where $\mu = GM_{earth}$, I_G is the inertia matrix of the spacecraft, r_0 is the distance from the spacecraft to the centre of the earth and z_0 is the unit vector to nadir.

The torques in the nominal, zero-roll, attitude are fairly small, as one axis of the spacecraft lies close to the nadir vector, i.e. a stable gravity gradient configuration. However, applying a 20° roll changes this and different gravity gradient torques are produced. Both scenarios are shown in Table 6-3.

		580km		630km		
		roll 0°	**roll 20°**	**roll 0°**	**roll 20°**	
Torque	**x**	>0.01	2.60	>0.01	2.70	mNm
	y	>0.01	>0.01	>0.01	>0.01	mNm
	z	>0.01	>0.01	>0.01	>0.01	mNm

Table 6-4: Gravity gradient nominal torques

In the extreme case of a tumbling spacecraft, say following an anomaly and safe mode transition, the gravity gradient torques can become transiently large. The magnitude of the maximum torques that can be developed are of the order as shown in Table 6-5 and .

		580km	630km	
		max	**max**	
Torque	**x**	1	1	mNm
	y	98	100	mNm
	z	97	99	mNm

Table 6-5: Maximum gravity gradient torques in stowed case

		580km	630km	
		max	**max**	
Torque	**x**	4	4	mNm
	y	101	103	mNm
	z	97	99	mNm

Table 6-6: Maximum gravity gradient torques in deployed case

These potentially large gravity gradient torques do not have a bearing on the nominal mode design or configuration, however has significant bearing on the strategy to employ for initial acquisition and safe mode.

6.2.4 Magnetic

Magnetic disturbance torques, T, acting on the spacecraft are described by the following equation:

$$T = M \times B$$

Where M is the magnetic moment of the spacecraft and B is the magnetic field of the earth, influenced primarily by the orbit altitude. For a residual magnetic dipole of 10 Am^2 on each axis the torque generated are shown in

		580km		630km		
		roll 0°	roll 20°	roll 0°	roll 20°	
Torque	**x**	0.50	0.30	0.50	0.40	mNm
	y	0.60	0.50	0.40	0.40	mNm
	z	0.30	0.30	0.40	0.40	mNm

Table 6-7: Magnetic moment nominal torques

The maximum disturbance torque about any axis is therefore of the order of 8 mNm and 7 mNm for 580 km and 630 km orbits respectively.

6.2.5 Disturbance Summary

It can be seen that the gravity gradient torques are by far and away the dominant disturbance and this will be reflected in the sizing of the actuators and in the strategy for initial acquisition and safe modes.

7 Trade-Offs

Given the nature of the Garada mission, a number of AOCS-level trade-offs can be performed. Section 5.1 indicated the main design issues and many of these for the trade-offs discussed below.

7.1 Acquisition strategy

The main choice for initial acquisition is between thruster-based and magnetic-based. The magnetic-based solution uses only simple equipment and is robust, but has a longer duration acquisition and no guarantee of sun intrusion if sensitive instruments are on the spacecraft. A thruster-based solution requires an attitude control propulsion system, with added cost and complexity but acquires faster and adds robustness to sun intrusion. A summary of the trade-off issues is shown in Table 7-1.

Thruster-based	Magnetic-based
Requires attitude control propulsion ○ Cost and complexity ○ Difficulty of accommodation	Only requires simple and robust equipment ○ MTQ, MTM (+CSS/CESS)
Expends propellant (requires additional propellant beyond that needed for delta-V.)	Expends no consumables
Shorter duration acquisitions ○ More robust to instrument sun intrusions ○	Longer duration acquisitions ○ Less robust to instrument sun intrusion ○ Power may be limited
Large torques available ○ Less susceptible to disturbances	Low control torques Susceptible to disturbances (e.g. gravity gradient)

Table 7-1: Summary of acquisition strategy trade-offs

The ideal solution is a magnetic-based acquisition. There are no instrument exclusions to consider, time is not a driver in acquisition and it is the most cost, mass and risk effective solution.

However the magnetic solution is only appropriate if it operates effectively in the disturbance environment. Garada can generate large gravity gradient disturbance torques which would dominate the magnetic control. The standard magnetic control employed on many missions is Bdot where the spacecraft uses the measured geomagnetic field derivative to damp the spacecraft rates and subsequently achieve a stable attitude rotating at twice the orbital period, i.e. following the magnetic field.

In the case of Garada, this would not work well as the gravity gradient would dominate. Fortunately, there is a modification to the Bdot, as investigated on TerraSar-L which works with the gravity gradient to achieve a stable attitude rotating at once the orbital period. This modification allows the selection of a magnetic-based acquisition.

7.2 Safe Mode Strategy

Given that a magnetic-based acquisition has been selected, it follows that the contingency mode could use the same strategy. There are no additional driving constraints for Safe Mode – the conditions are the same although the inertias may be those deployed but the modified Bdot is tolerant to a large range of inertia and thus is also suitable for Safe Mode.

7.3 Normal Mode Sensors

The AOCS can use a gyro or fly a gyroless option. The simplest configuration is gyroless, using the star trackers, plus estimation, for attitude determination. The gyro option aims at enhancing the on-ground stability restitution performances (where gyro data used on ground) and improving relative pointing error. Also in case of star tracker (STR) unavailability, the gyro data are also used in the on-board AOCS loop to propagate the satellite attitude. Thus in the event of including a gyro, a hybrid gyro-stellar attitude estimation function is also selected.

Although the inclusion of a gyro improves the pointing stability, this is usually only required for very high-precision missions. In the case of Garada, there are no stringent requirements on stability (RPE – Relative Pointing Error) that necessitates the inclusion of a gyro.

The star tracker configuration, given the attitude domain of the Garada mission can be defined to avoid outages due to blinding, and thus guarantee at least one optical head is available during observations even in the event of a failure. Indeed modern Active Pixel Sensor STR are proving to be very robust to outages.

The gyroless solution is therefore nominal baseline, and adding a gyro will increases cost – through from the procurement of the hardware until the final integration and testing.

7.4 Normal Mode Actuators

The normal mode actuators can be chosen between reaction wheels (RW) and control moment gyros (CMG). The CMG option is required for high agility mission. For lower agility needs the reaction wheel option is selected.

The Garada mission has no performance requirements on agility, and indeed has a highly constrained attitude domain. During observations, the spacecraft is always close to nadir pointing.

The AOCS uses a bang-bang acceleration guidance law to minimise slew duration. The sizing includes a tranquilization period of a few seconds in order to damp rate immediately after slew and start image acquisition with good performance.

Thus, without any drivers, and also considering the increase in cost and complexity of CMGs, the RW are the baseline. In order to cope with a single wheel failure, a set of four wheels is required.

7.5 Star Tracker Configuration

The STR configuration is discussed in Section 8.1. In summary STR configurations exists for any missions, with the main variable being the number of STR Optical Heads (OH). In any case of STR OH configuration, two Electronic Units (prime and redundant) in case of failure.

7.6 Baseline

In summary, the baseline from the trade-offs is:

- Acquisition and Safe Mode:
    - Modified Bdot (TerraSAR-L)
        - Works with gravity gradient
        - No propellant usage
        - Simple and robust equipment
        - Serves as Safe Mode also
- Normal Mode Sensors
    - Gyroless
- Normal Mode Actuators
    - Reaction wheels (set of 4)
- STR Configuration
    - Prime and redundant for sun-synchronous dawn-dusk orbit
    - Three OH for drifing orbit
    - Prime and redundant EU

8 Design

8.1 Star Tracker Configuration

The star tracker configuration is designed to allow availability of at least 1 optical head at all times even in the event of a failure. The orientation of the star tracker optical heads can easily be arranged to avoid earth blinding (note that star trackers are not used in initial acquisition or safe modes). As the roll of the spacecraft is restricted to less than 20°, any star tracker boresight elevated above the nominal horizon by 20° plus the earth exclusion angle (30°) will never nominally become earth blinded. This yields an angle above the horizon of 50°, which is an angle from the spacecraft +z-axis of 116°. This is illustrated in

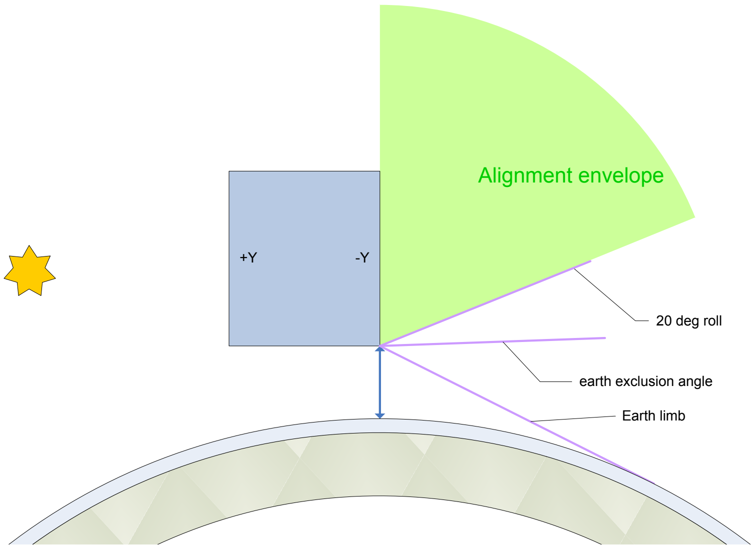

Figure 8-1: Star tracker alignment to avoid earth blinding.

For a sun-synchronous daw-dusk (LTAN = 06:00), the star tracker configuration is straightforward. In nominal operations, one side of the spacecraft generally points towards the sun and the opposite side away from the sun. This gives a large envelope for aligning the star trackers that avoids both sun and earth blinding. In this particular case, as blinding is avoided, only two optical heads are required – prime and redundant. Two possible configurations are illustrated in Figure 8-2.

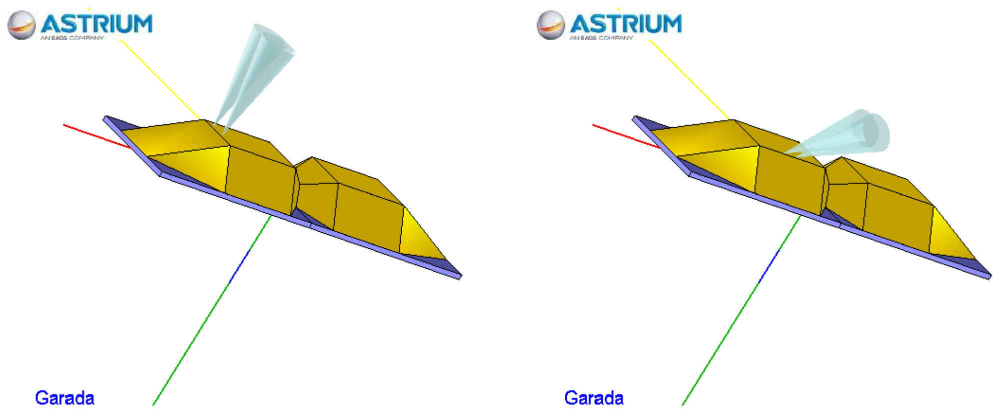

Figure 8-2: Star Tracker configurations for sun-synchronous orbit.

Should the orbit be a 42° inclination, then a different configuration is required. Due to the drifting orbit, there is a wide range of solar aspect angles. In fact the sun vector can lie anywhere in the –z spacecraft hemisphere (plus a portion of the +z hemisphere) which means that a star tracker will become sun blinded at some time. However by using 3 optical heads, this allows a failure plus a single sun blinding whilst leaving an available optical head. To ensure that this is the case, the separation between the optical heads needs to be greater than the sun exclusion angle (40°).

Given the two constraints of optical head elevation and separation, there is an envelope available in which to mount the optical heads. Two potential options are shown below. The first aligns the star trackers in the same plane with a separation of between 60° (to avoid simultaneous sun blinding) and 64° (to avoid earth blinding). The second aligns the star trackers in a triangular configuration with a 60° separation between the optical heads. These are illustrated in

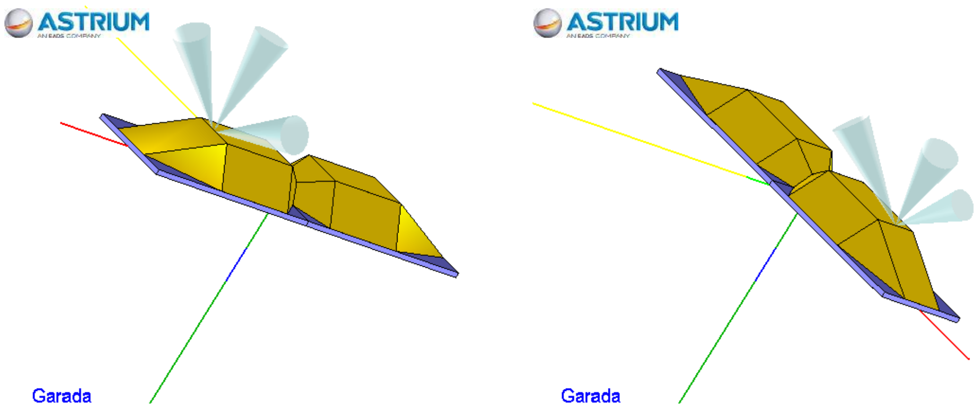

Figure 8-3: Star tracker configuration for low-inclination orbit.

Whilst the sun-synchronous solution would not be appropriate for a low-inclination orbit, the opposite is not the case. The three-head solution could be used for any orbit, and in the case of sun-synchronous where blinding is avoided, two optical heads would always be available which increases the accuracy of the star tracker measurements.

8.2 Reaction Wheel Configuration

There are no hard requirements on manoeuvrability, however the assumption has been made to achieve a 20° slew about the x-axis in 120s, which is an artificial driver but is useful for enveloping analysis. This has to be achievable using only 3 out of 4 wheels to account for a failure.

With a uniform tetrahedral configuration (i.e. equal torque in each axis), when all four wheels are acting with 200 mNm, a torque of 350 mNm can be achieved on each axis. This would result in a 20° slew about the x-axis in 140 s. In the case of a wheel failure, the torque drops to 230 mNm with a corresponding slew time of 170 s.

Modifying the 3-wheel case to provide sufficient torque about the x-axis results in a case where the wheel alpha angle is 30° and beta angle 27°. (Note that there are a range of solutions which meet the criteria, this being one case. Thus there is flexibility to take into account other requirements or drivers.)

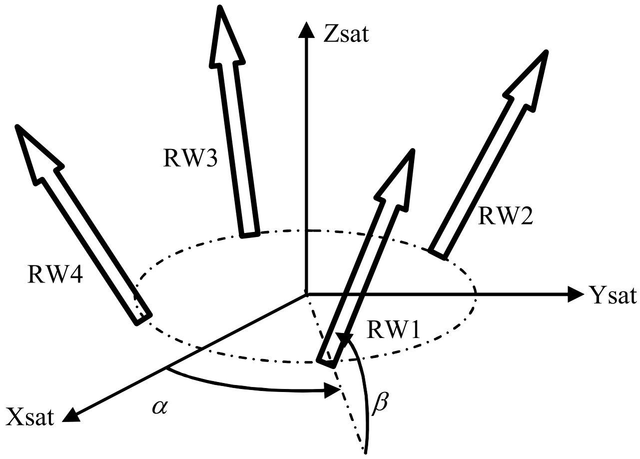

Figure 8-4: Tetrahedral wheel configuration

8.3 Sun Sensor Configuration

Although not used in nominal operations, a sun sensor can be used for attitude anomaly detection and to optimise the magnetically-controlled attitude to improve power generation on the arrays.

For a sun-synchronous daw-dusk (LTAN = 06:00), the sun sensor configuration is straightforward. The sun sensor (internally redundant) is placed on the side of the spacecraft the always sees sunlight (approximately in the same direction as the arrays). This ensures continual sun visibility outside of eclipses.

Should the orbit be a 42° inclination, then a different configuration is required. In order to cover the range of solar aspect angles multiple sun sensors can be fitted to cover the desired fields (note that sun sensor fields of view are available up to ±90°).

An alternative to multiple sun sensors is a Coarse Earth Sun Sensor (CESS). The sensor consists of six active CESS heads arranged orthogonally around the spacecraft. Each head has a hemi-spherical field-of-view and six thermistors mounted behind two different materials, three behind black OSR and three behind mirror (one pair is redundant). Using the temperature difference between the two materials the Sun and albedo flux can be derived, and using the measurements from the six heads it is possible to determine the state vectors.

The ideal configuration of the CESS is with six heads mounted orthogonally, as illustrated in Figure 8-5. In addition, the sensor heads must have a clear field of view with no spacecraft appendages impinging upon them. The fields of view of the sensor are required to overlap in order to ensure the correct operation of the system. If there is insufficient overlap, then there will be blind spots in the system. To date only systems with 90° overlap have been used. Given the monolithic external shape of the proposed Garada platform, ideal CESS head accommodation should be achievable.

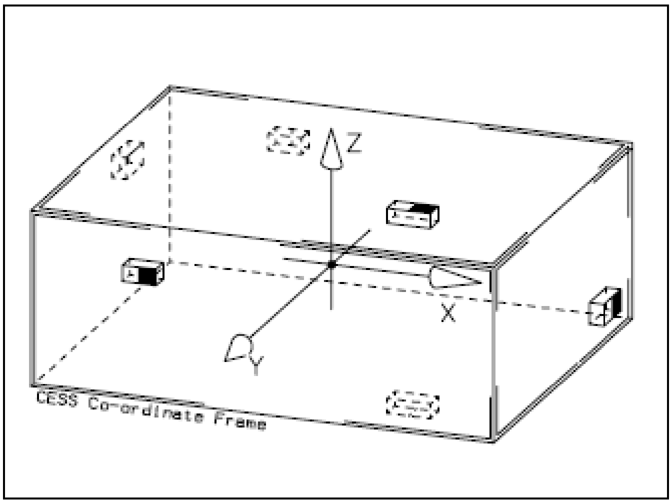

Figure 8-5: CESS configuration

8.4 Mode Architecture

8.4.1 Initial Acquisition and Safe Mode

Initial Acquisition & Safe Mode is a magnetic-based control mode. It is a variation of the b-dot controller employed on a range of Astrium spacecraft.

Due to spacecraft configuration, the proposed solution, for acquisition, is based on a traditional "Bdot Law" with gravity gradient stabilization. The aim of this mode is to acquire an attitude where the gravity gradient, which is the main perturbation, is null. This allows minimizing the perturbation and fulfilling the performances requirements

The modifications arise from the large gravity gradient disturbances generated from a spacecraft with the the Garada configuration, i.e. a large disparity between the inertias. In the standard b-dot, the spacecraft uses magnetorquers to remove the rates (measured by magnetometers) from the spacecraft and ultimately end up in a stable rotation at twice the orbit period aligning the spacecraft y-axis with the orbit normal. A small momentum bias is applied to the y-axis of the spacecraft to ensure that this condition is stable.

However, because of the large gravity gradient of a "long, thin" spacecraft, the gravity gradient disturbance fights against the twice orbital period rate. Hence the magnetic control is modified to rotate only at orbital rate meaning one axis of the spacecraft naturally ends up earth pointing. This is then a safe stable attitude which can be used both pre- and post- deployment.

This mode, as it is simple and robust is also used as the Safe Mode using the redundant equipment, or in the case of the reaction wheels, managing any failures.

8.4.2 Normal Mode

Normal Mode is a classical gyroless-wheel mode, with GPS for navigation and guidance generation and MTQ for wheel momentum management. An onboard magnetic field model provides the input to the momentum management. This mode can be used pre- and post-deployment.as well as in between individual deployment segments.

The baseline configuration and control (via magnetic off-loading) of the reaction wheels in NM has been optimized in order to minimize zero crossings and magnetorquers demands. The baseline configuration ensures no zero crossings while in steady state maintenance of operational attitudes (right & left looking) and with either all 4 wheels or 3 wheels (following a failure) active.

8.4.3 Orbit Control Mode

Orbit Control Mode is similar to Normal Mode. In the classic case, the reaction wheels are held at constant speed whilst the thrusters are used to deliver the delta-V manoeuvre and provide the attitude control. However in the case of Garada, there is no need for attitude control thrusters and therefore the option is for Orbit Control Mode to perform the delta-V burns open loop. This requires a single (a pair for redundancy) of delta-V thrusters.

Thus when the delta-V thrusters operate open-loop with the AOCS providing attitude control with the reaction wheels, Orbit Control Mode and Normal Mode look the same with the possible exception of controller tuning and FDIR monitors (the hardware is the same). Thus these two modes can be rationalised into one single mode with submode tuning.

There may be implications on when orbit manoeuvres can be performed, i.e. in the fully deployed state, and there may be constraints on the rate at which delta-V that can be delivered (disturbances will be absorbed by the wheels, constraining the integrated disturbances by the size of the wheels and the rate of magnetic momentum management.) However, this should be able to be managed operationally.

Using no propellant for attitude control during delta-Vs will also reduce the amount of propellant needed, or allow more propellant for orbit correction.

8.5 Hardware Architecture

The Garada AOCS hardware consists of:

- Star Tracker
- GNSS
- Coarse Sun Sensors
- Magnetometer
- Magnetorquer
- Reaction Wheels (4)

The hardware architecture is shown in Figure 8-6.

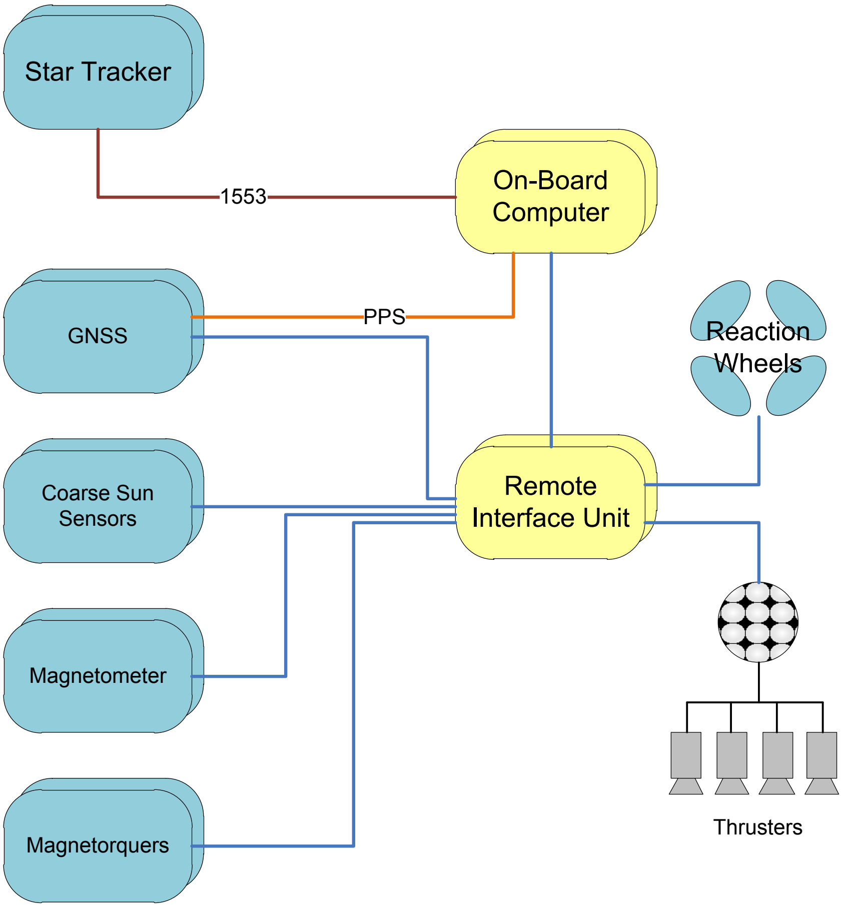

Figure 8-6: Hardware architecture

8.5.1 Star Tracker

The star tracker provides 3-axis attitude data in form of a quaternion. It is able to determine this data from a "lost in space" situation within a few seconds, and then provide a continuous tracking of the attitude. As a baseline, a star tracker with Active Pixel Sensor (APS) technology is adopted. This is the current state-of-the-art technology for modern star trackers, with a number of options available.

APS-based star trackers have excellent radiation hardness and operational robustness, being more tolerant to Single Event Upsets (SEUs) and flare events than the classical CCD (Charge-Coupled Device) star trackers. They deliver high accuracy and high update frequency even in presence of significant angular rates, and provide robustness towards moon, or bright objects, blinding. The star tracker is not nominally intended to look at the sun, but exposure does not cause permanent damage. The use of multi-head star trackers, or running multiple individual star trackers mitigates the effect of blinding.

Use of multiple observations, from skewed optical heads also provides improved 3-axis attitude performance. Since one optical head provides good performance around the axis perpendicular to its line of sight and reduced one around that line, it is necessary to use at least two optical heads with different line of sights for fine pointing performance. However in the case of Garada, the pointing requirements can be met with a single optical head, and thus the star tracker configuration is driven by the orbit characteristics and the avoidance of blinding. For this reason, the preferred candidate is a three optical head Sodern Hydra. The configuration consists of 2 Electronic Units (EU) and 3 Optical Heads (OH) including baffle, all cross-strapped. The star tracker heads are oriented such that they are not blinded by the earth, even when sideways looking and that only one can be blinded by the sun (to avoid simultaneous blinding this angle must be larger than twice the sun exclusion angle) at any time. Even in the event of a failure, this still leaves a free optical heed.

Alternative candidates for APS star trackers are Selex Galileo AA-STR and Jena Astro.

- Sodern Hydra
  - Provides absolute attitude knowledge, plus angular rates
  - APS technology
  - Heritage – baseline for Astroterra, Seosat, S5p etc.
  - Good robustness to blinding, flares and high kinematics
  - High frequency update
  - Arcsec accuracy
  - Prime and redundant baseline – multiple OH as an option
  - 1 OH = 1.25 kg, 1 EU = 1.75 kg, 10 W, c. (c. k€750-1000)

Figure 8-7: Sodern Hydra

8.5.2 GPS

The GPS receiver provides the position of the satellite, the velocity and a time reference that is used to synchronize the on-board software. The GPS receiver is generally used in cold redundancy. The expected accuracy for a single frequency GPS receiver is about 10 m (3D, rms), 1 cm/s (3D, rms), and 1 µs. An improvement by a factor of two is expected from the dual frequency GPS receivers.

For the medium accuracy required on Garada, a single frequency GPS is sufficient. As the position errors dominate the velocity errors, only the effect of these errors on the pointing budget is assessed. The typical error 10 m error is assumed to be equally split between all axes, resulting in a position error of 5.77 m about each axis.

Potential GPS suppliers are Astrium, Thales and Ruag and of course opportunities for Australian industry.

8.5.3 Sun Sensor

A Sun sensor arrangement of sufficient FoV in hot redundant triplex is attractive on several fronts: inherited design from other projects, provides a means of measuring the sun direction fo optimising the attitude during acquisition, detecting eclipses on-board and fulfilling FDIR functionality. The alternative, attractive especially for a fixed, sun-synchronous dawn-dusk orbit (i.e. the spacecraft solar aspect angle is constrained), is a single internally redundant sun sensor.

On Garada there is no driving requirement to have a particularly high accuracy Sun sensor, and so a coarse Sun sensor of the type shown in Figure 4-153 and Figure 4-154 below, which have been used in numerous missions, is deemed to be sufficient. Such Sun sensors are available from Bradford/TNO (i.e. the Coarse Sun Sensor, CSS) and Astrium (i.e. the BASS).

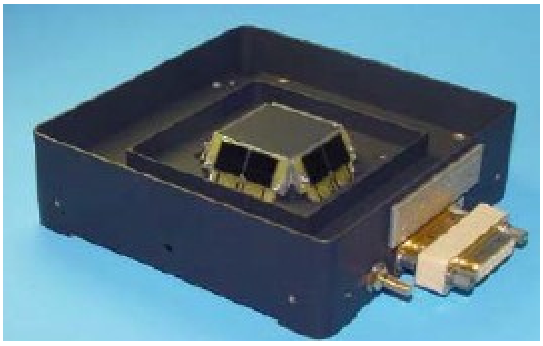

Figure 4-153 Bradford/TNO CSS

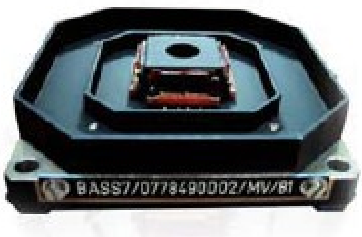

Figure 4-154 Example of Astrium BASS

8.5.4 Reaction Wheels

A cluster of four reaction wheels are used in Normal Mode to provide fine three-axis control and during other modes to provide momentum bias. The wheel sizing is selected to provide the optimum between control and momentum management – in the case of Garada this leads to large wheels because of the large inertias. The drivers behind the selection of the reaction wheels are:

- Momentum storage: environmental torques lead to a gradual accumulation of angular momentum. Large momentum capacity wheels can absorb more momentum before momentum needs to be offloaded, or in the case of Garada when momentum offloading is inefficient because of the current configuration of the earth magnetic field.

- Torque demands: manoeuvres require a torque profile to slew the spacecraft. Generally, the larger the available torque, the quicker the slews, Although not a hard driver, the large inertias suggest large torque wheels. In addition, for high slew rates, large momentum capacity is also required (as momentum is transferred between spacecraft and wheels during slew).

The reaction wheel configuration is a tetrahedral set of 4. All 4 wheels are operated in hot redundancy, with the capability to manage the mission in the event of a single wheel failure. With 4 reaction wheels the AOCS is fully robust to one wheel failure. Momentum management using offloading via the magnetorquers and the earth magnetic field ensures the wheels are maintained within their optimal operating envelope, avoiding zero crossings and excessive speeds.

Off-the-shelf wheels are available and the prime candidate are the Bradford W18. These wheels have significant heritage, flying on many European missions, provide > 200 mNm of torque and up to 40 Nms of momentum storage. Alternative wheels are available form Rockwell Collins Dynamics or from US supplier Honeywell.

- Bradford Engineering W18
    - Provides fine attitude control
    - Heritage – Extensive use on many missions
    - Range of characteristics available
    - Four wheel flight set
    - Internally redundant electronics
    - 1 RWA = 4.9 kg, 1 WDE = 8.8 kg, ~ 15 W, c. k€1000-1500

Figure 8-8 Bradford Engineering W18 reaction wheels and electronics

8.5.5 Magnetometer

The magnetometer is used during initial acquisition and safe mode to measure the earth magnetic field and thus calculate the optimum dipole direction for the magnetorquers. A magnetometer measures the magnetic field in three orthogonal axes. The magnetometer axis alignment accuracy is better than 1deg. The measurements accuracy better than 50 nT.

By way of example previous Astrium missions have used magnetometers provided by a number of different suppliers: ZARM, Lusopace, Tamam and Billingsley.

8.5.6 Magnetorquers

It is proposed to employ three 400 Am2 magnetorquers, with each providing torque capability about a distinct spacecraft axis. Such magnetorquers have been used on Aeolus: three orthogonally mounted MT400-2-L magnetorquers manufactured by Zarm Technik GmbH. Each magnetorquer consists of two simultaneous wound coils around a high permeability core material. Both coils are interfaced to the unit's interface connectors without any cross coupling connections. The winding consists of a double Polyimide insulated high temperature copper wire. The coils are held by coil isolator plates mounted at the ends of the core. A cutaway view of a torquer is shown in Figure 4-168 below and reveals the inherently simple design and limited amount of components.

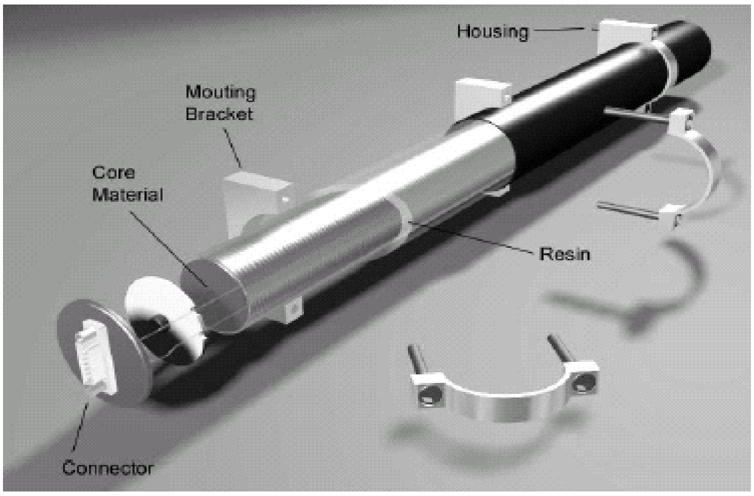

Figure 4-168 Magnetorquer Cut-Away View

8.6 Normal Mode

The purpose of Normal Mode is to provide:

- Normal SAR operation in nadir pointing attitude (with optional steering law),
- Sideways pointing SAR operation with roll of up to ±20°,
- Slew the spacecraft between the observational attitudes,
- Optionally slew the spacecraft to place spacecraft into the requisite attitude to perform delta V manoeuvres.

The Normal Mode controller has to ensure high pointing performance to meet the requirements. The nominal Norma Mode attitude is shown in Figure 8-9.

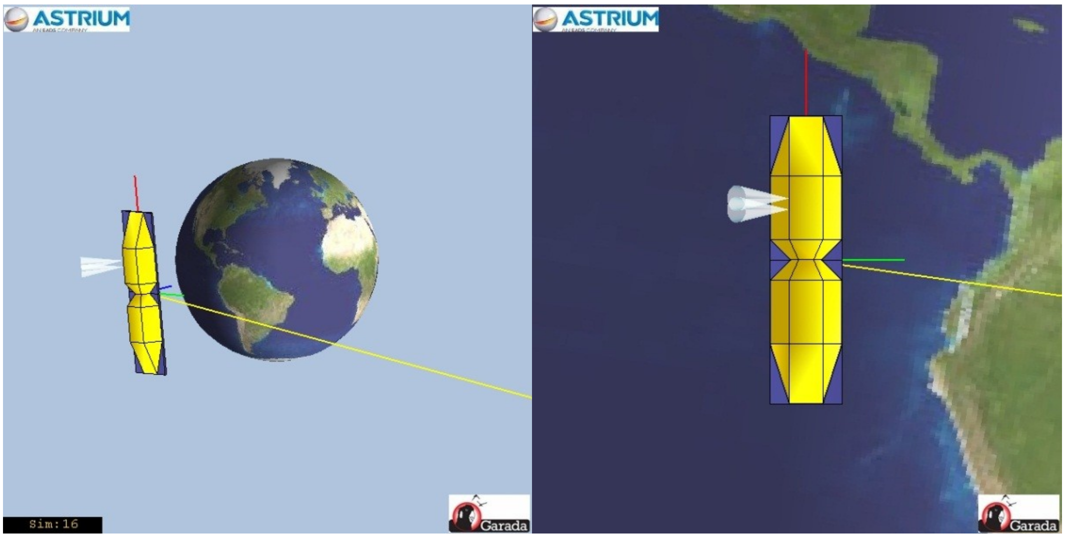

Figure 8-9: Spacecraft in nominal attitude

8.6.1 Controller Design

With the selected solution, i.e. star tracker only, typically a phase advance would be employed to give the requisite gain and phase margins with adequate performance. In the presence of significant flexible modes a more complex controller, for example with elliptical filters, may be employed however as Garada is a rigid structure with benign manoeuvring demands, a simple controller should be all that is necessary. A double phase advance may be employed to give more effective increase in phase at the necessary bandwidth, but again as Garada is not particularly challenging for AOCS controller design despite the large inertias and so the controller design is therefore not the critical aspect of the mission.

The classical design is well suited for the Garada mission where a single design tuning for NM science observation, NM slews and Delta-V manoeuvres can be achieved with comfortable margins in stability and performance. However, as shown in the mode structure, slews and delta-Vs are treated within submodes and so different tuning, for example if the delta-V disturbance torques were excessively large, can be realised.

8.6.2 RW Momentum Management

As the spacecraft is controlled by means of a set of reaction wheels, external torques will cause the momentum stored in the RWs to vary, both in a secular and oscillatory fashion. Unless the wheels are suitably offloaded (i.e. have their momentum managed via the application of other external torques) they risk becoming saturated (i.e. exceeding the nominal maximum momentum capacity).

Hence a reaction wheel offloading function shall be implemented to constrain the wheel momentum levels to lie within predefined ranges whilst the spacecraft is operating in Normal Mode. This is achieved by adopting a scheme similar to other missions whereby the momentum of each reaction wheel is controlled to a predetermined set point via the magnetorquers.

The momentum management of the reaction wheels is proposed to be performed continuously using the magnetorquers (MTQ). The MTQ interacts with the earth magnetic field to produce an off-loading torque. Determination of the magnetic field in the body axis is performed using an on-board model rather than the magnetometer which allows continuous use of the MTQ (i.e. not having to switch off to take a MTM reading).

The offloading torque is delivered in the plane orthogonal to the magnetic field as the magnetic field direction is uncontrollable (T=MxB where T is the offloading torque, B is the earth magnetic field and M is the MTQ magnetic moment). The magnetic control law is a closed-loop proportional gain control on the excess momenta vector for the active wheels (measured relative to the set-point):

$$T = -k\Delta H$$

Where T is the torque demand, k is the control gain and $\Delta\mathrm{H}$ is the momentum error.

Considered as a closed-loop proportional gain controller, k is effectively a time constant for offloading the excess momenta. The larger the gain, the faster the offloading. However, if the gain is too high the MTQs are continuously saturated, which reduces the performance of momentum management and increases the power demands.

In reality, as the disturbances are relatively low and the MTQ rather large, the final value of the gain does not need to be finely tuned and a relatively coarse tuning is sufficient. (The optimal value of MTQ k cannot be tuned analytically due to the non-static nature of the geomagnetic field. Instead, an iterative tuning approach must be taken in which the value of the parameter is gradually varied and the effect on wheel momentum management is assessed.)

The end result of this scheme is that the MTQ imparts a slowly varying continuous disturbance torque (which can be fed-forward to mitigate an attitude effect on the spacecraft) onto the spacecraft that is beneficial for managing the momentum of the wheels. The magnetorquers are driven independently from the attitude measurements, responding to errors in wheel momenta only (i.e. the magnetorquers do not respond to sensed attitude errors, only the reaction wheels do that). Thus the continuous management of the wheel momenta via the magnetorquers has negligible impact on the attitude behaviour of the spacecraft.

8.7 Acquisition and Safe Modes

The acquisition modes are designed for the inertia configuration of Garada. The control algorithm is the Astrium patented Bdot Law with gravity gradient Stabilization. The aim of this mode is to acquire an attitude where the gravity gradient, which is the main perturbation, is null The control sequencing is:

The Bdot Law with Gravity Gradient Stabilization performs and achieve the following objectives:

- Short term: loss of kinetic energy through a magnetic braking effect -> rate damping
- Long Term: alignment of the internal angular momentum (disposed along the spacecraft pitch axis) with the orbit normal and s/c rotating at the orbital frequency around the orbit normal -> Nadir attitude

The Astrium standard Bdot has been originally developed as an Astrium product. It is based upon a simple control law which consists in measuring the Earth magnetic field with a magnetometer, and performing with magneto-torquers the following command: M=-k Bdot(derivative expressed in a spacecraft frame), where M is the magnetic moment demand on the magnetorquers, k is a control gain and Bdot is the measurement of the geomagnetic field derivative in spacecraft frame.

Furthermore, in order to set the spacecraft spin axis, an internal angular momentum H is produced using the reaction wheels. From initial launcher separation (or in case of failure), the main stages of this mode are:

- Initially, when the spacecraft rotation rate is large, the magnetic control torque behaves as a damper. The spacecraft kinetic energy is dissipated and the angular rate reduced.
- Then, the Bdot law induces the (re)alignment of the internal angular momentum (placed along the spacecraft pitch axis) with the magnetic field rotation rate, i.e. with the orbit normal. At the end of this dynamic phase (large angles, angular momentum re-orientation and Bdot damping), the spacecraft pitch axis is oriented towards the orbit normal.
- In converged situation, when the spacecraft rotation rate is small, the control torque tends to make the satellite follow the magnetic field. The magnetic field having a bi-orbital component in the orbit plane, the satellite rotates at twice the orbital rate.

The mode is fully autonomous (no position knowledge required), and is not limited in time (no propellant consumption), provided that solar arrays are adequately oriented. Furthermore, only the most reliable units are used (magnetometers, magneto-torquers and reaction wheels). In particular, STR and GPS are not used.

In a sun-synchronous dawn-dusk orbit, this strategy is particularly well suited as the sun direction and the orbit normal are approximately colinear. This alignment ensures a constant illumination of the solar array, as shown in Figure 8-10.

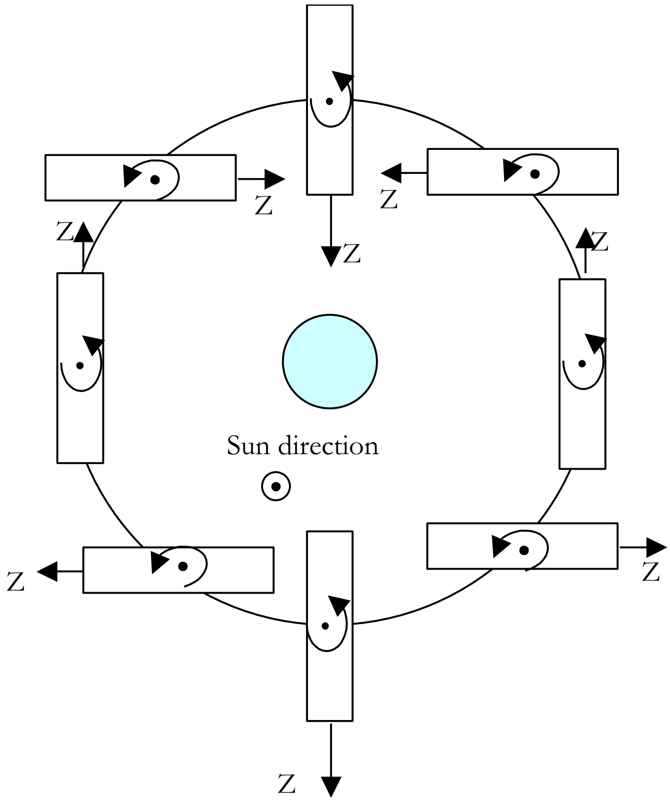

Figure 8-10: Bdot configuration

The converged situation of this mode is a spacecraft rotating at twice the orbital rate around the orbit normal. In such an attitude, and due to the huge Garada inertia difference, the gravity gradient disturbance torque would dominate the control torque. Thus the traditional "Bdot law" cannot be used as is. Therefore, an alternative concept has to be defined.

Instead of trying to control all disturbance torques and in particular the large gravity gradient torque, this new concept uses the gravity gradient for passive stabilisation. The classical "Bdot law" is then modified so that the spacecraft rotates at one time the orbital pulsation, and the gravity gradient passively stabilises the satellite along its lowest inertia axis as shown in Figure 8-11.

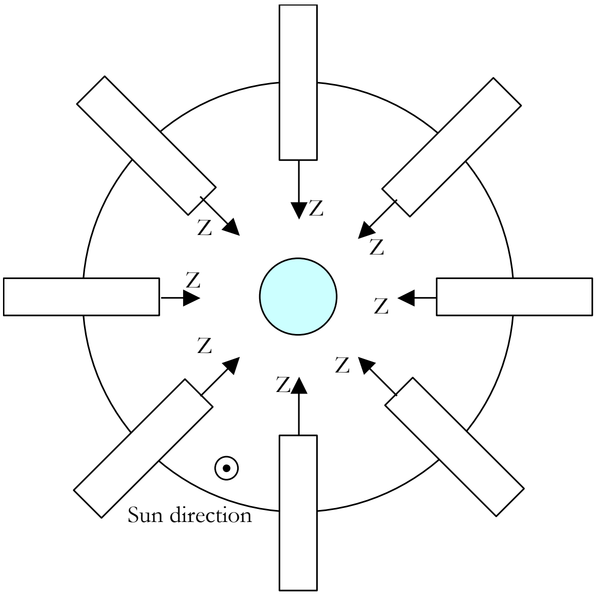

Figure 8-11: Modified Bdot configuration

To define a new magnetic control with orbital rotation rate, the previous Bdot control law is biassed, to ensure a reference rotation rate at orbital rate. This is realised by subtracting the cross product of the reference rate (orbital rate) and the measured geomagnetic field vector.

Thus the spacecraft will now rotate about the pitch axis, aligned with the orbit normal, at a mean rate of the orbital rate.

8.8 FDIR

The Garada FDIR design ensures the spacecraft is tolerant to all credible single point failures (SPF) and ensures the critical spacecraft survivability under all conditions. It uses two fundamental recovery strategies, switch to redundant equipment but continue the nominal mode (Redundancy Management FDIR level) or switch to Safe Mode (System Safety level) with scheduled operations stopped and the spacecraft in a stable safe state until ground intervenes.

The FDIR is designed around a hierarchical architecture (illustrated in Figure 8-12):

- At level 5, ground executes contingency recovery procedures (e.g. recover from Safe Mode).
- At level 4, OBC Reconfiguration Modules oversee the health and function of the OBC and flight software by monitoring hardware alarm inputs and performing OBC reconfigurations and restart in response.
- At level 3, the flight software monitors the health of the OBC and its own run time health through use of traps, alive flags and similar techniques and requests an OBC reconfiguration by the Reconfiguration Modules by setting a software alarm.
- At level 2, the OBC SW monitors telemetry in the Global Data Pool to identify failures of units and subsystems and performs the necessary recovery actions. These will either be;
    - Fail Safe - on-board switch to redundancy plus a backup mode, with termination of scheduled activities.
    - Fail Operational - on-board switch to redundancy, with continuation of the scheduled mode and activities.

- At level 1, the units internally detect and react to failures (e.g. LCL trip, inputs majority voting). If a level 1 action impacts the unit function then a level 2 monitor is used to detect this.

The monitoring and recovery actions utilise the following Packet Utilisation Standard (PUS) services:

- PUS 12: On board monitoring, status and limit check monitoring of parameters stored in the Global Data Pool
- PUS 5: Event reporting, to report nominal operations as well as anomalies, depending on the severity on-board action is required
- PUS 19: Event & Actions, event detection triggering autonomous action execution
- PUS 18: OBCP, On-Board Control procedures for FDIR.

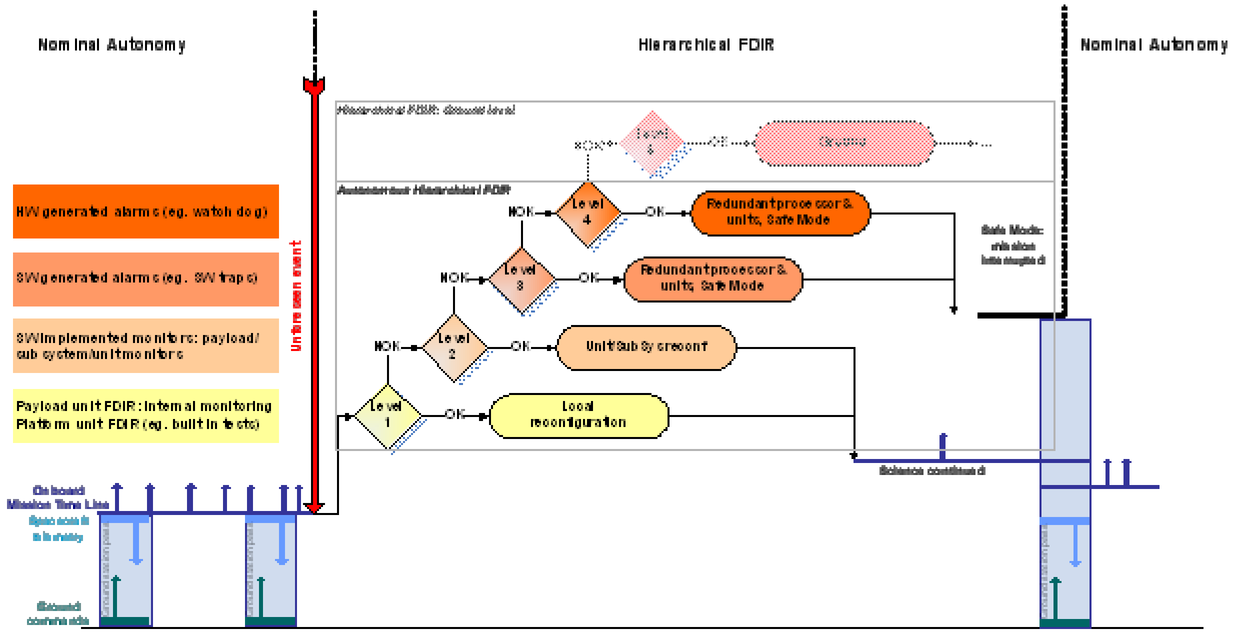

Figure 8-12: FDIR Strategy

9 Performance Assessment

9.1 Normal Mode

Simulations have been performed to assess the AOCS performance in Normal Mode. Data from a typical run is shown in Figure 9-1 and Figure 9-2.

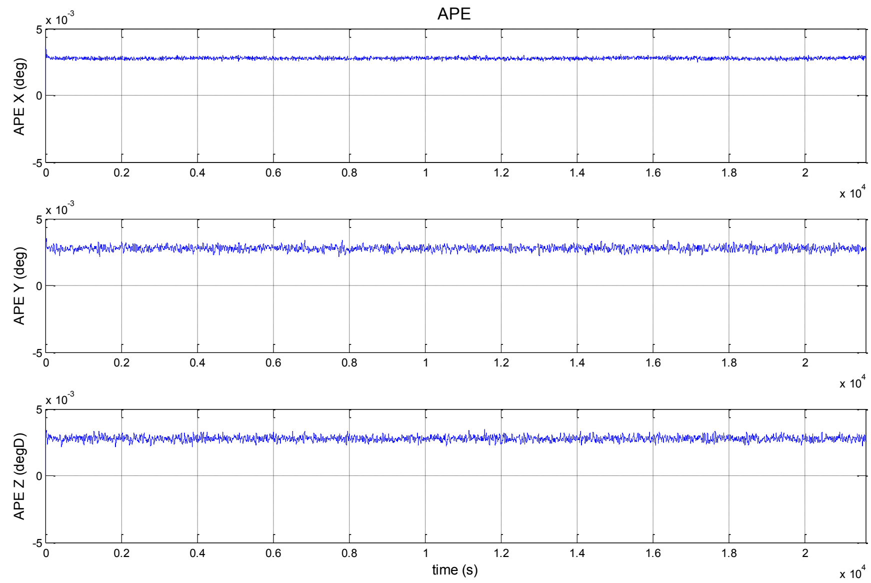

Figure 9-1: APE

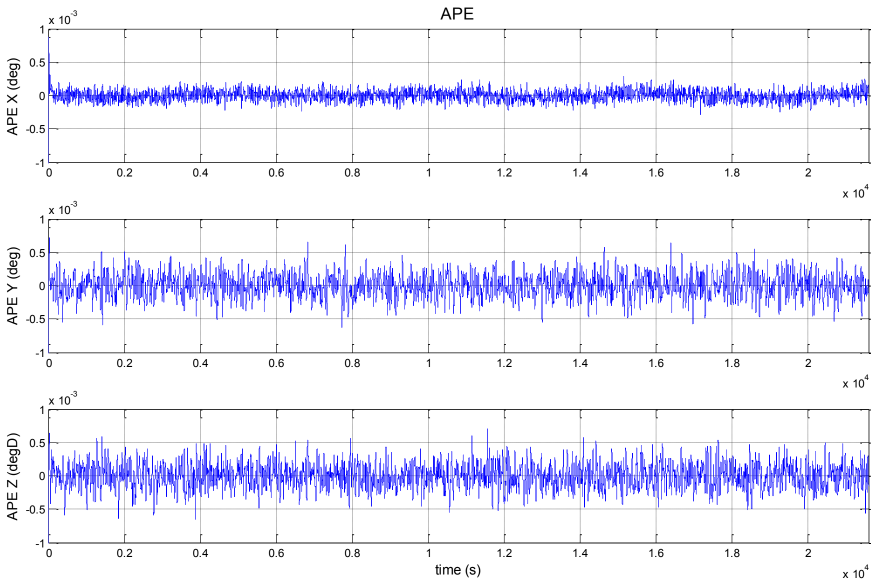

Figure 9-2: APE with bias removed

Note that the performance on the three axes are different. This is due in a large part to the different performance of the star tracker between the transverse and longitudinal axes, plus the inertia configuration and controller tuning.

Also not that performance presented here is at AOCS level. System level (e.g. payload misalignment, structure thermo-elastic) have not been included. These elements are usually introduced at system level budgets (although it is possible to simulate these details with a detailed high-fidelity simulator).

Quantitative results on the performance during Normal Mode are shown in Table 9-1.

	APE (° 95%)
X	0.0029
Y	0.0031
Z	0.0032

Table 9-1: APE results

The good performance of the spacecraft is largely due to the high-quality star tracker.

9.2 Initial Acquisition

Simulations have been performed to assess the performance during initial acquisition. A typical acquisition sequence is shown in Figure 9-3 and Figure 9-4.

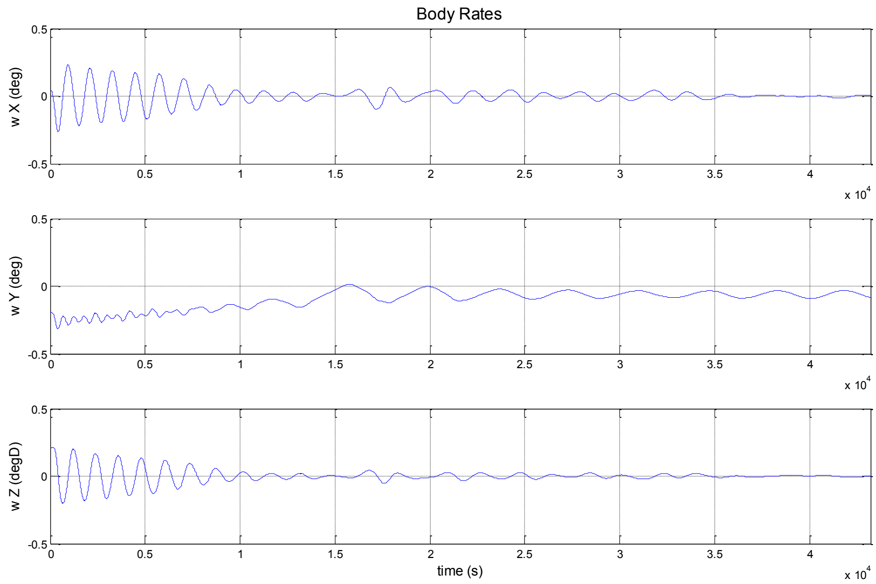

Figure 9-3: Body rates during acquisition.

Note how the body rates on the x and z axes converge towards zero, whilst the rate around the y axis converges to a mean of approximately -0.001°/s, equivalent to orbital rate, as expected. The convergence time for this particular simulation was 5 hours or a little over 3 orbits.

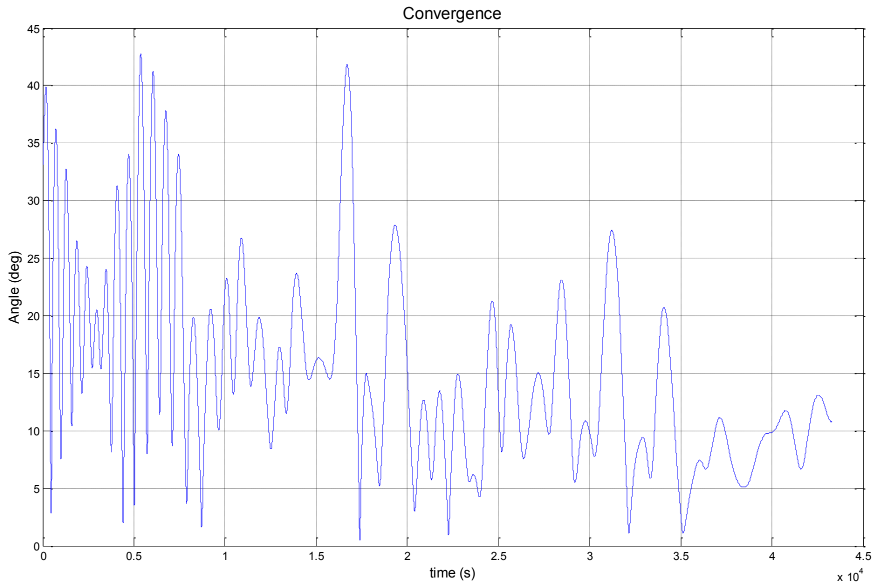

Figure 9-4: Convergence angle (between y axis and anti-orbit normal)

Note how the angle between the spacecraft y axis and the anti-orbit normal reduces towards zero, but during the course of this simulation never goes beyond 45°. This is important because even at a relatively large angle to the sun, the array will begin to generate power, and even at a sun angle of 31° (sun declination plus orbit inclination) the power availability is > 85%.

10 Conclusion

This report has presented the design drivers, trade-offs, analysis and baseline solution for the Garada AOCS. A solution has been defined and presented that meets the Garada performance and functional requirements, whilst maintaining heritage to reduce cost. Due to the use of a standard high-accuracy multi-head star tracker, the nominal mission is robust and accurate. Simple and robust equipment can be used for acquisition and safe modes, which in turn reduces cost and risk – moreover a strategy can be utilised with heritage from previous activities.

The key baseline solutions are:

- A modified Bdot magnetic control strategy for acquisition and safe mode;
- A simple star tracker only normal mode solution with reaction wheels for actuation;
- A star tracker optical head configuration to allow continuous visibility.

The baseline hardware solution for Garada consists of:

- Star Tracker – a multiple optical head configuration providing continuous coverage; even in the event of a single failure to provide absolute attitude information during the nominal mission;
- GNSS – provides orbital position to the on-board navigation and guidance functions;
- Coarse Sun Sensors – provides a measure of the sun direction for use in sun acquisition, attitude anomaly detection and eclipse detection.
- Magnetometer – provides measurement of the geomagnetic field as inputs to the magnetic control law.
- Magnetorquer – provides simple and robust actuation using the geomagnetic field in acquisition and safe modes, and allows momentum management of the reaction wheels during the nominal mission;
- Reaction Wheels – a set of 4 wheels (allowing for a single failure) provides the nominal control actuation during the nominal mission.