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THE SENTINEL-2 SATELLITE ATTITUDE CONTROL SYSTEM CHALLENGES AND SOLUTIONS

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ABSTRACT

Sentinel-2 is a European polar satellite system built on a constellation of two similar satellites for the provision of operational land services based on an optical Earth observation payload. Sentinel-2 is being built by Airbus Defence and Space in the framework of the European Space Agency's Global Monitoring for Environment and Security program, now termed Copernicus, and is currently in an advanced phase D status.

Sentinel-2 operates in a sun-synchronous orbit with 786 km mean altitude and Earth-oriented attitude in all operational modes. While the required pointing performance is moderate, the crucial design driver for the attitude control system is the attitude knowledge required for precise geo-location of the images taken by the Multi Spectral Instrument payload.

The first two satellites are currently undergoing completion at Airbus Defence and Space in Friedrichshafen, Germany, with Sentinel-2A scheduled for launch in Q2 2015 and Sentinel-2B roughly one year later.

After on overview of the Sentinel-2 mission and satellite design, the attitude control system architecture is presented. The preeminent features of the attitude control system are the Earth-pointing safe mode and the high accuracy on-board attitude and position estimation.

High-accuracy attitude estimation is provided by a constant-gain Kalman filter using the measurements of the multi-head high performance star tracker and a fibre-optical four axes gyro. Specific design solutions are employed to minimise thermal distortion. A dual-frequency GPS receiver provides the orbit state vector.

The Earth-pointing safe mode ensures easy transition back to normal mode and stable thermal conditions for platform and payload. The main sensor is the Airbus Defence and Space developed Coarse Earth and Sun Sensor, actuation is performed by magnetic torquers and thrusters. Floquet theory has proven to be a valuable tool for the tuning of the controller with proper consideration of the periodic nature of the magnetic actuation.

Elastomer wheel isolators and solar array drive micro stepping are employed to avoid image distortion caused by micro-vibrations. A detailed solar array drive model, whose parameters have been identified from test data, is used for performance analysis.

For the implementation of the attitude control algorithms software, a model-based approach has been successfully employed, which allows continuous simulation and testing from the start of the development until pre-validation of the algorithms before specification for coding. Airbus Defence and Space has recently developed a complete software development process based on automatic code generation directly from the algorithms model as natural extension of the model-based development applied for Sentinel-2.

1. MISSION OVERVIEW

Since the early 1970's, regular acquisition of the Earth's land surfaces has been performed via the US Landsat 1-7 series of satellites, very recently complemented by the launch of the Landsat Data Continuity Mission in February 2013, now termed Landsat 8. European competence has been established in 1986 with the first launch of the French series of SPOT satellites under leadership of the French Space Agency CNES. While the first Landsat satellites were using a scanning technique, the SPOT satellites were among the first missions built upon the push broom technology.

In continuation of these very important precursor missions Europe is now establishing a fully operational service of the so-called Sentinel missions, of which Sentinel-2 constitutes the optical land observation mission. Sentinel-2 is also based on the push broom technology. The key features of Sentinel-2 and its precursor missions are summarized in the Table 1 and Fig. 1.

The first two Sentinel-2 satellites are undergoing completion in the facilities at Airbus Defence and Space, scheduled for launch of Sentinel-2A in Q2 2015 and Sentinel-2B roughly one year later. Sentinel-2 will feature a major breakthrough in the area of optical land observation since it will for the first time enable continuous and systematic acquisition of all land surfaces world-wide with the Multi Spectral Instrument (MSI), thus providing the basis for a truly operational service.

The Multi Spectral Instrument acquires images in 13 spectral channels from visible and near infrared to short wave infrared with a swath of 290 km on ground and a spatial resolution from 10 to 60 m. The data ensure continuity to the existing data sets produced by the series of Landsat and SPOT satellites, and will further provide detailed spectral information to enable derivation of biophysical or geophysical products.

Table 1: *Key performance parameters of Landsat, SPOT and Sentinel-2 (MS = multispectral, PAN = panchromatic).*

Performance	Landsat	SPOT	Sentinel-2
Number of satellites	8+	5	2+
Swath	185 km	2 x 60 km	290 km
Spatial resolution	MS: 30m, 100m PAN: 15 m	MS: 10m,20m PAN: down to 1.5m	MS: 10m,20m,60m
Spectral bands	Up to 7 MS + 1 PAN VIS/NIR/SWIR/TIR	4 MS+ PAN VIS/NIR/SWIR	13 MS VIS/NIR/SWIR

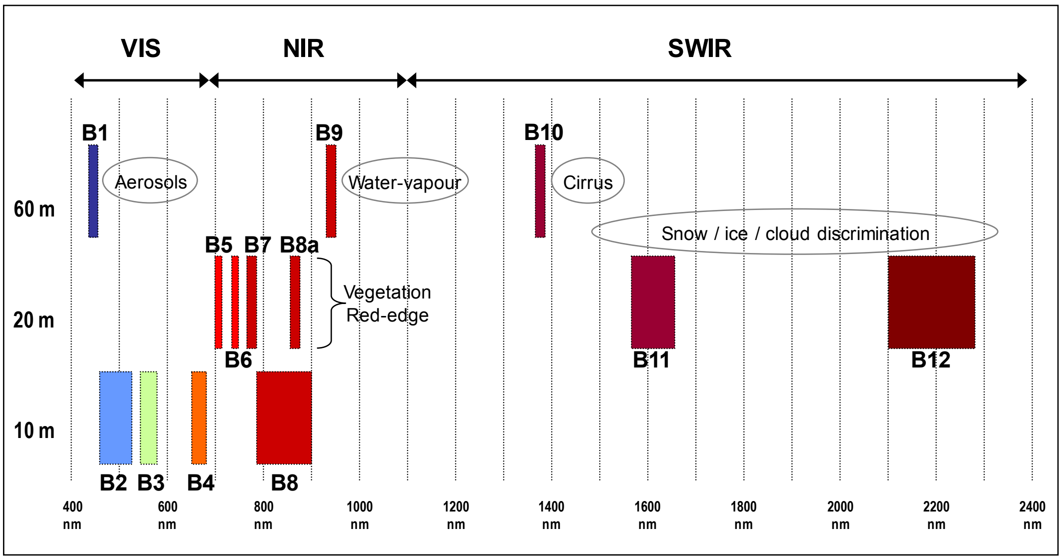


Fig. 1: *Sentinel-2 provides 13 spectral bands from visible (VIS) and near infra-red (NIR) to short wave infra-red (SWIR) at different special ground resolutions from 10 to 60 m.*

Flying in the same orbital plane with 180° phase separation, the constellation of two satellites designed for an in-orbit nominal operational lifetime of 7 years each, will acquire all land surfaces in only 5 days at the equator. In order to support emergency operations, the satellites can further be operated in an extended observation mode allowing imaging of any point on Earth even on a daily basis.

Excellent geometric image quality performances are achieved with geo-location better than 16 m, thanks to an innovative instrument design in conjunction with a high-performance satellite attitude and orbit control subsystem centred around a dual frequency GPS receiver, high-performance multi head star trackers and a fibre-optical gyro.

To cope with the high data volume on-board, data are compressed using a state of the art wavelet compression scheme. Thanks to a powerful mission data handling system built around a newly developed very large solid-state mass memory based on flash technology on-board compression losses will be kept to a minimum.

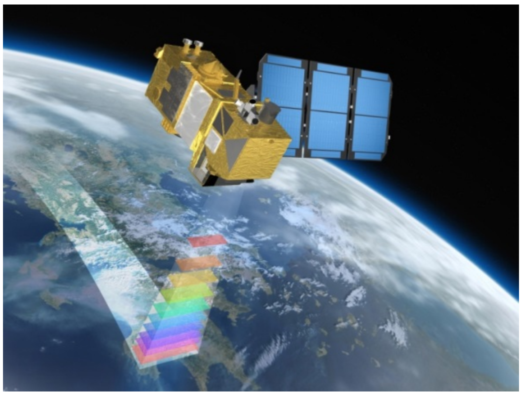
The Sentinel-2 satellite design features a highly flexible operational concept, allowing downlink of all mission data to a nominal X-band core ground stations network. In addition, users could receive mission data sets at selected X-band local user ground stations or through an Optical Communication Payload (OCP) via an inter-orbit optical link to a geostationary relay satellite at Ka-band user ground stations. Different priority schemes can be selected in flight to allow transmission of critical image data with the shortest possible latency.

The system is designed for high system autonomy allowing for pre-programming of the operational schedule for 15 days in advance without interference from ground. Apart from the nominal and extended imaging modes, the satellites also feature a calibration mode to support regular in-orbit radiometric calibration of the instrument.

Overall, the Sentinel-2 satellites are designed to provide in-orbit availability for the instrument data greater than 97%, which fulfils the requirements of a fully operational system for multi spectral Earth observation.

Table 2: *The Sentinel-2 mission provides multi spectral imagery with a global coverage of land surfaces from -56° to +84° degree and 5 days revisit time.*

Mission	
Mission Lifetime	15 years
Number of Satellites	2
Nominal In-Orbit Lifetime	7.25 years with consumables for additional 5 years
Nominal Orbit	Sun synchronous 786 km mean altitude, 10:30 local time of descending node
Land Coverage	-56° to +84° latitude
Global Revisit Time	< 5 days
Global Accessibility	< 2 days, < 1 day above 45° latitude
High Quality Mission Products	Level 0, 1a, 1b, 1c , and higher Levels
Mission Phases	LEOP, commissioning, operational, de-orbiting



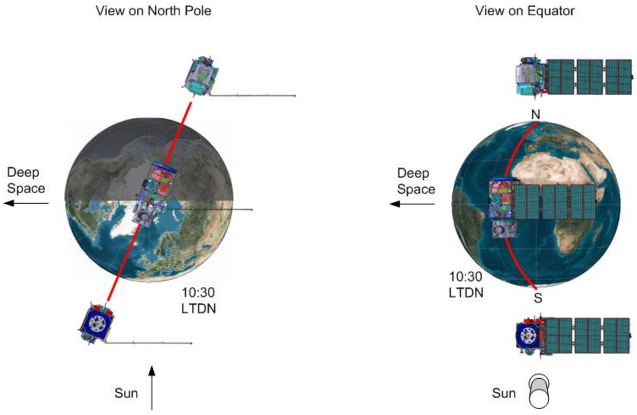


Fig. 2: *Left: Artist's view of Sentinel-2, Right: Two identical satellites with 180° phase separation are placed in a sun-synchronous orbit with 786 km mean altitude and 10:30 local time of the descending node.*

2. DRIVING REQUIREMENTS

The key objective of the Sentinel-2 mission is to image continuously and systematically world-wide all land areas between -56° (Patagonia) and +84° (Greenland) latitude with a twin satellite constellation. This observation scenario requirement determines the overall system architecture together with all the boundary constraints from the beginning of the mission at launch, injection, initial acquisition and system stabilization, platform and payloads commissioning through the routine operations phase and ending with the disposal phase, when the satellite approaches its end of life. In addition the accommodation of an Optical Communication Payload as a pre-operational communication experiment to a geostationary relay satellite is a further key element.

The driving system requirements impacting the attitude control system are

- Multi Spectral Instrument Line of Sight absolute pointing error ≤ 2 km at 99.7 % confidence level
- Geo-location accuracy of data ≤ 20 m at 95.5 % confidence level without the need of any ground control points

Derived from the system requirements the attitude control system on-board performance requirements are defined:

- Absolute pointing error ≤ 1200 µrad per axis at 99.7 % confidence level
- Absolute attitude knowledge error ≤ 10 µrad per axis at 95.5 % confidence level
- Absolute rate error ≤ 20 µrad/s per axis at 99.7 % confidence level
- Absolute horizontal position knowledge error ≤ 12 m at 99.7 % confidence level

The attitude control system has to provide the required performance in the presence of disturbances caused by the rotating solar array and the Optical Communication Payload. Micro-vibrations caused by the attitude control system shall be minimized to avoid detrimental impact on the image quality.

The agility requirement for Sentinel-2 are moderate with slew rates < 0.5°/s. The nominal attitude is geocentric Earth pointing with yaw steering to compensate for the Earth rotation and avoid image distortion. In addition an extended observation mode shall be supported, which introduces a roll angle of about 20° to allow imaging of neighbouring ground tracks. Further slews around the yaw axis are needed to point the thruster for orbit control manoeuvres.

3. SATELLITE DESIGN

The Sentinel-2 satellite comprises a compact aluminium structure carrying all the platform equipment, the Multi Spectral Instrument and the Optical Communication Payload.

The Sentinel 2 Multi Spectral Instrument (MSI) is a filter based push broom imager. It provides imagery in 13 spectral channels with spatial resolutions ranging from 10 to 60 m. The instrument features an optical telescope providing a wide field of view to achieve the required swath width of 290 Km. An oblong pupil equivalent to 15 cm diameter has been selected to achieve a compact design and optimized optical performance.

The Sentinel-2 Optical Communication Payload (OCP) is a secondary payload for the evaluation of high-speed data communications by means of a laser link using a geostationary satellite as relay. It is foreseen to use the OCP pre-operationally in complement to the baseline X-band payload data handling subsystem. The OCP consists of a frame unit structure and a moveable coarse pointing assembly.

Electrical power is provided by one deployable 7.2 m² solar array with triple junction Gallium-Arsenide solar cells providing 2300 W at begin of life. The power is distributed via a 28 V unregulated bus with maximum power point tracking. The solar array is tilted 22.5° for optimum sun incidence and rotated by a twist capsule solar array drive following the sun with orbital rotation rate of 0.06°/s on the sunlit side of the orbit. In eclipse the solar array is rotated back with increased rotation rate of 0.2°/s to re-start forward rotation at the end of the eclipse.

The data handling system consist of the on-board computer (OBC) with a redundant MIL-STD-1553B bus interface and the remote interface unit (RIU) which provides the non MIL-STD-1553B interfaces for the equipment.

A S-band communication system is used for telecommand, telemetry and ranging. Payload data is transmitted by the X-band payload data transmission system.

The mono-propellant chemical propulsions system consists of two branches with four tilted thrusters each which provide thrust for orbit manoeuvers and three dimensional torque for attitude control. The tank, the pipework and the thrusters are assembled into a self-standing propulsion module, which allows independent development and testing and provides for easy integration into the satellite.

Table 3: *Key characteristics of the Sentinel-2 satellite.*

Key Characteristics	
Launcher	Vega, backup Rockot
Satellite launch mass	1225 kg
Satellite moments of inertia	700 / 1100 / 1300 kgm²
Satellite Dimensions (Stowed)	3390 mm x 1630 mm x 2350 mm
Fuel	130 kg Hydrazine monopropellant
Power	2300 W begin of life, 7.2 m² solar array, 1250 W typical consumption

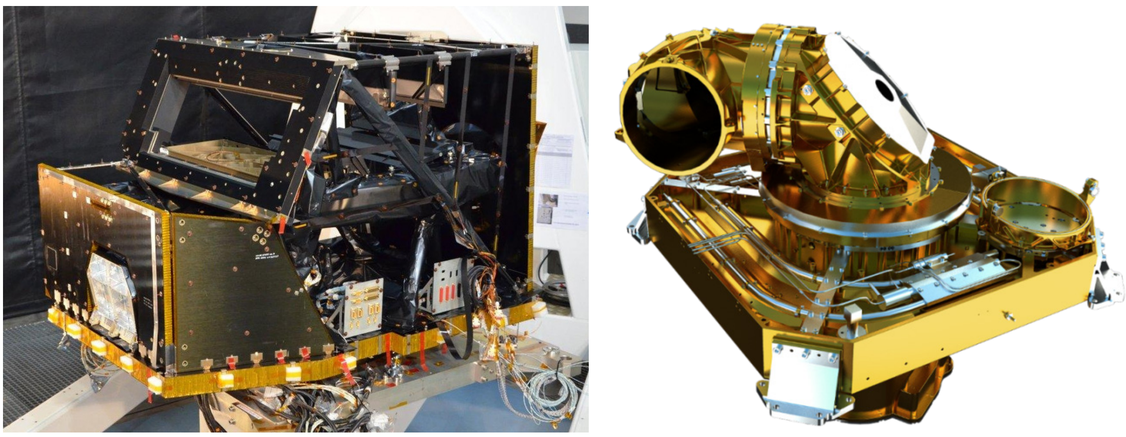


Fig. 3: *Left: The Multi Spectral Instrument (MLI removed), Right: The Optical Communication Payload (Tesat).*

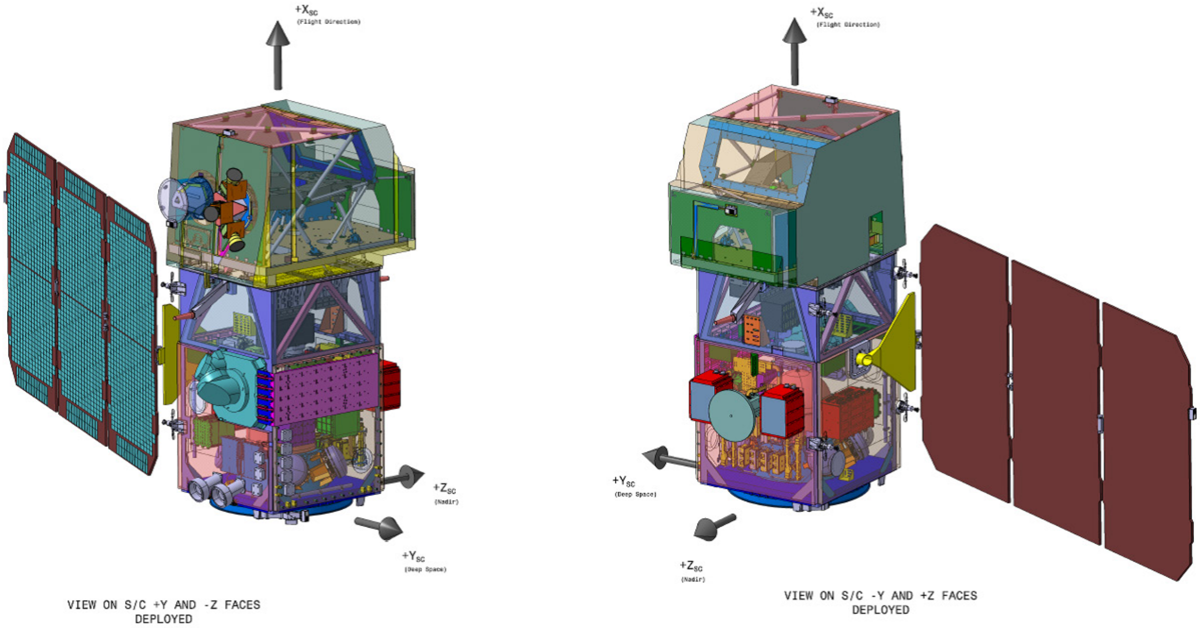


Fig. 4: *Sentinel-2 in flight configuration.*

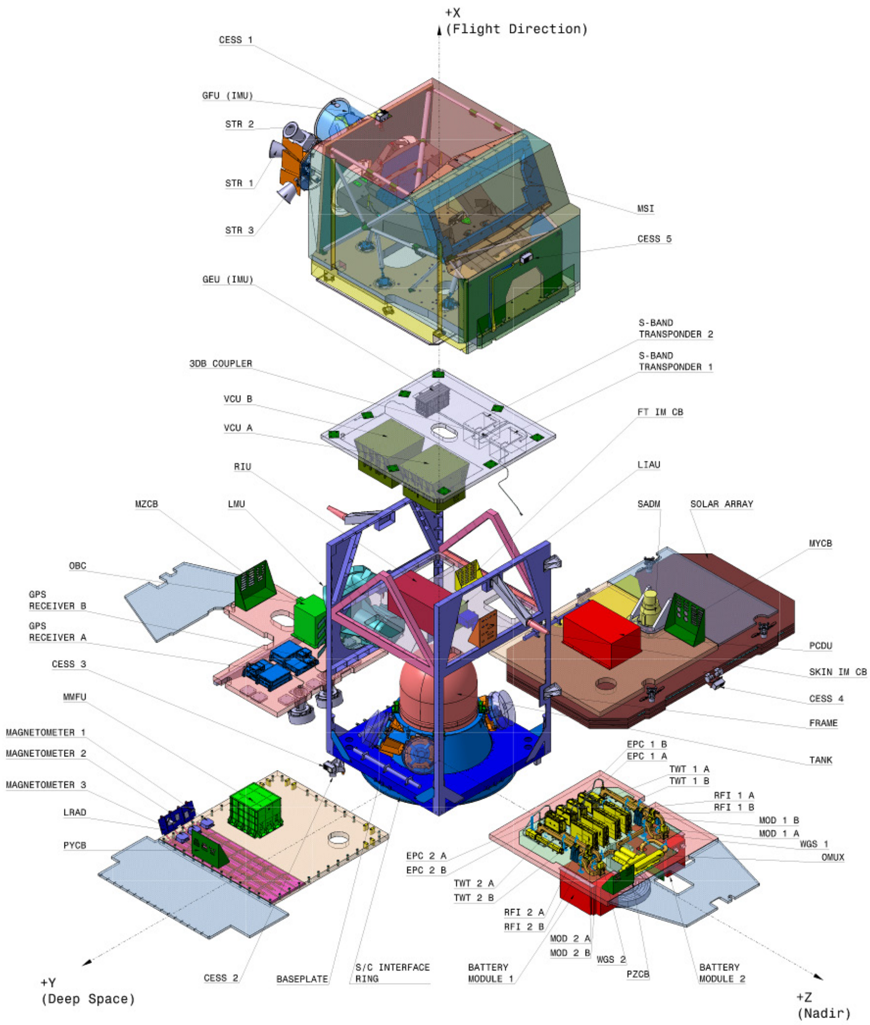


Fig. 5: *Sentinel-2 exploded view.*

4. AOCS DESIGN

The attitude control system is composed of the control algorithms residing in the on-board software executed on the OBC and a suite of sensors and actuators to meet the required control and state estimation performances. The design is one failure tolerant to maintain full performance after possible loss of one equipment unit.

In particular challenging are the attitude and position knowledge as well as the rate error requirements, which are essential to meet the image quality needs.

A. OPERATIONAL MODES

The Sentinel-2 attitude control system provides four operational modes and a number of sub-modes to perform autonomous attitude control (Fig. 6).

INITIAL ACQUISITION MODE AND SAFE MODE

The Initial Acquisition Mode (IAM) is entered after separation from the launcher. Its purpose is to damp out the satellite rates and acquire a coarse Earth-pointing attitude from any initial condition to ensure safe thermal and power conditions for the satellite.

The Safe Mode (SFM) is entered in case of a failure. Functionality and performance is identical to the Initial Acquisition Mode, however for independency from nominal modes redundant equipment is used.

When the mode is entered the satellite rates measured by the gyro are damped (rate damping sub-mode RD). As soon as the residual rates are sufficiently small the satellite acquires an Earth-pointing attitude using the Earth vector measured by the coarse Earth and Sun sensor in the Earth Acquisition sub-mode (EA). Following Earth acquisition the satellite is rotated around the yaw axis into the proper flight direction (Yaw Acquisition sub-mode YA) and finally reaches the Steady State sub-mode (SS).

The Deployment sub-mode (DEP) is entered only once for the deployment of the solar array after separation from the launcher and initial rate damping.

The actuators used are the thrusters for fast rate damping and acquisition supported by magnetic torquers to minimise fuel consumption in steady state. Magnetometers are used to measure the magnetic field needed to compute the proper magnetic moment commands for the magnetic torquer.

NORMAL MODE

The Normal Mode (NOM) is the mode for the nominal operation of the instrument. When entered from IAM or SFM the first sub-mode is the Attitude Hold Mode (AH), a robust mode which moves the satellite from the coarse attitude during ASM with attitude errors up to 30° into a much more accurate Earth pointing attitude with pointing errors in the millirad range before switching to the Fine Pointing sub-mode (FP) for observation.

To allow the instrument to look sideward, the satellite can be rotated about 20° around the roll axis (Extended Fine Pointing, EFP) using the Slew (SL) and Back Slew (BSL) sub-modes.

The sensors used in NOM for attitude estimation are the three high performance star trackers and the high performance gyro, the main actuators are the four reaction wheels. The magnetic torquers are used to dump the angular momentum stored in the reaction wheels and avoid saturation of the wheels.

ORBIT CONTROL MODE

The Orbit Control Mode is used to perform orbit correction manoeuvres. The first sub-mode is the Slew sub-mode (SL) to rotate the satellite around the yaw axis and orient the thruster in the right direction. This is followed by a stabilisation period (sub-mode STAB) before the thrusters are fired (Delta-V sub-mode DV). Attitude control during the delta-v manoeuvre is performed by thruster off-modulation. Finally the satellite is rotated back in its nominal flight direction (Back Slew sub-mode BSL).

With exception of the OCM DV sub-mode the sensors and actuators used are similar to the Normal Mode.

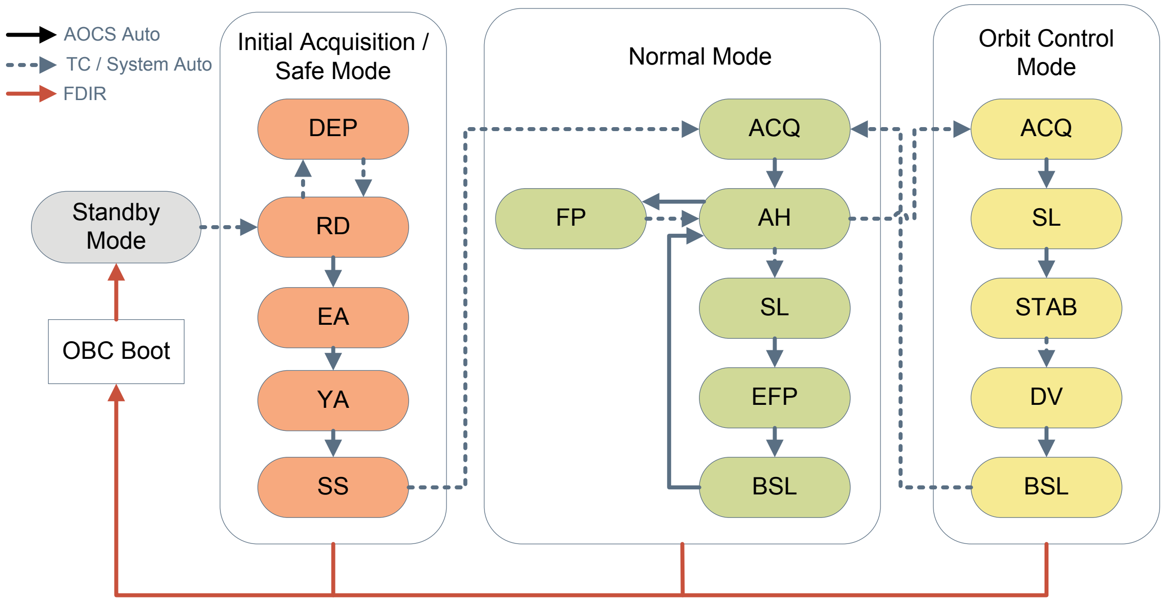


Fig. 6: *The attitude control system provides four operational modes and a number of sub-modes.*

Table 4: *Sentinel-2 AOCS hardware to mode matrix.*

	IAM	SFM	NOM	OCM
MAG	X	X		
CESS	X	X		
MIMU		X		
IMU	X		X	X
STR			X	X
GPSR			X	X
MTQ	X[1]	X[1]	X[2]	X[2]
THR	X[3]	X[3]		X[4]
RW			X	X
SADM	X[5]	X[5]	X	X

1 *Magnetic torquer used to provide attitude control torque*
2 *Magnetic torquer used for magnetic momentum control*
3 *Thruster used in on-modulation to provide attitude control torque*
4 *Thruster used in off-modulation to provide linear force and attitude control torque*
5 *Solar array drive not used in rate damping sub-mode*

B. EQUIPMENT AND ACCOMMODATION

The following figures and tables present an overview of the equipment used by the attitude control system. The sensors and actuators are connected either to the on-board computer by the redundant MIL-STD-1553B or to the remote interface unit when analogue lines are used.

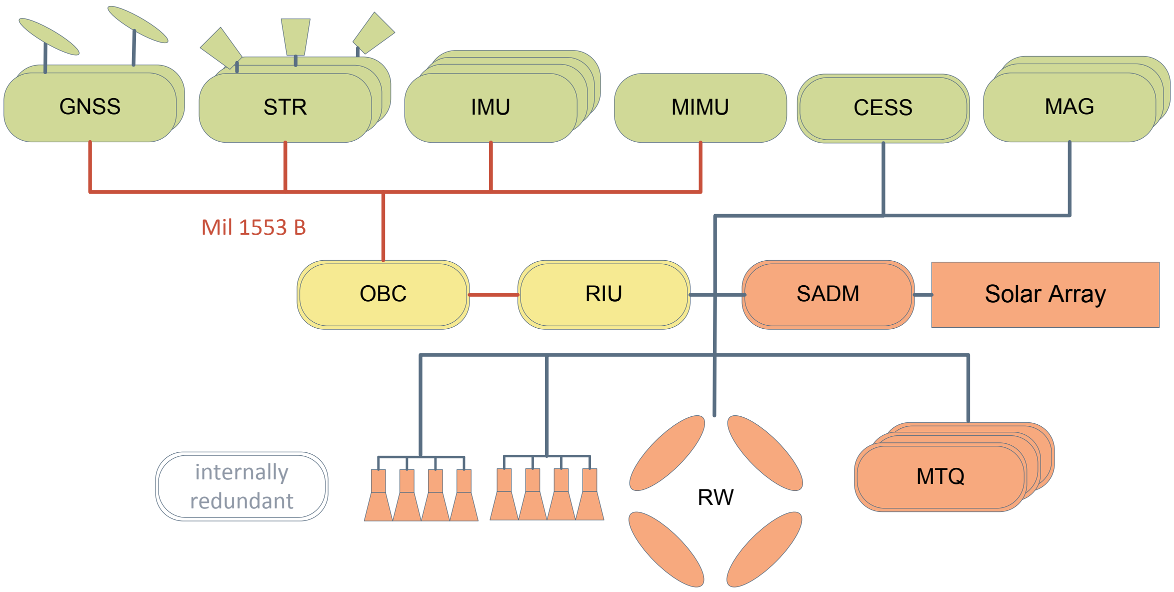


Fig. 7: The AOCS related equipment is either connected via MIL-STD-1553 bus or via the remote interface unit for analogue signals.

Table 5: *Sentinel-2 AOCS related equipment.*

Unit	Type	Supplier	Unit Name
OBC	On board computer, ERC32 based	Ruag, S	
RIU	Remote interface module, front end for OBC	Patria, SF	
MAG	3-axis fluxgate magnetometer	ZARM Technik AG, D	FGM-A-75
CESS	Thermo-optical coarse earth- and sun sensor	Astrium, D	
MIMU	3 axis ring laser gyro	Honeywell, USA	MIMU
GPSR	2 band GPS receiver	RUAG, A	
STR	Active pixel sensor star tracker	Jena Optronik, D	Astro APS
IMU	High performance fibre optical gyro	Astrium, F	ASTRIX 200
MTQ	140 Am^2 magnetic torquer	ZARM Technik AG, D	MT140-2
RW	18 Nms reaction wheel	Honeywell, USA MOOG Bradford	HR12 (S2A) W180 (baseline for S2B)
THR	1 N monopropellant thruster	EADS ST, D	CHT1N-6
SADM	Stepper motor plus gear box, twist capsule, potentiometer angle sensor	RUAG, CH	SEPTA-34-C

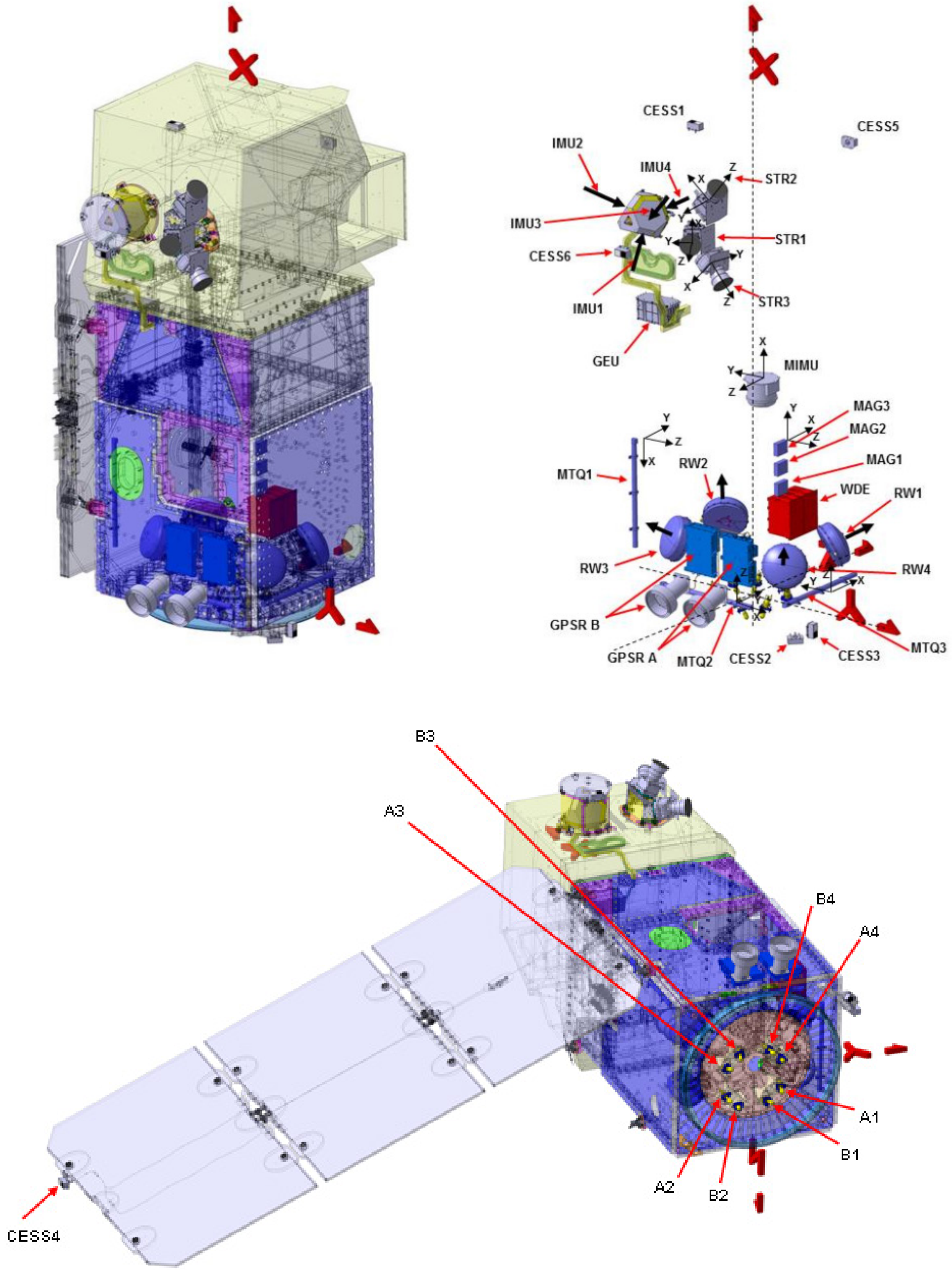


Fig. 8: *AOCS equipment accommodation on the satellite.*

The key sensors for normal mode control and especially the high accuracy attitude knowledge are the four channel Inertial Measurement Unit with three out of four channels redundancy and the three high performance APS star trackers with large angle separation and two out of three redundancy. Orbit position and velocity knowledge used for the attitude reference is provided by two dual frequency GPS receivers in cold redundancy.

The attitude measurement accuracy on-board requires fusion of at least two star tracker measurements with the measurements of the inertial measurement unit. The attitude estimation based on a Kalman filter provides precision attitude knowledge for the geo-location of the MSI image products, whereas direct star tracker measurements propagated by the inertial measurement unit to the current time are used for the attitude control.

Thermal stability with respect to the MSI line of sight is accomplished by direct attachment to the MSI instrument on a common support structure. The star trackers viewing directions are optimized with respect to maximization of the line of sight separation and avoidance of Sun and Earth in the field of view during normal observation. During orbit control manoeuvres transient sun blinding of star trackers might occur; however, the attitude estimation concept is robust towards the transient blinding of one or even two star trackers .

A specific feature of the star tracker is the accommodation of dedicated radiators to each star tracker housing, in order to control the APS chip temperature. Use of the radiators avoids a heat flow into the support structure through the star tracker interface, which would lead to detrimental thermal distortion.

The three magnetic torquers in orthogonal configuration with two cold redundant coils each provide torque for continuous reaction wheel desaturation and for attitude control in Initial Acquisition / Safe Mode steady state.

Three hot redundant three-axis magnetometers measure the magnetic field vector needed for the commanding of the magnetic torquers in Initial Acquisition and Safe Mode. In Normal Mode when the orbit position is known from GPS receiver measurements or the on-board orbit propagator a magnetic field model is used.

The internally triple redundant coarse Earth and Sun sensor is the main sensor for the Earth-pointing Initial Acquisition and Safe Mode. The six sensor heads are accommodated on the satellite (two on the platform, three on the Multi Spectral Instrument and one on the tip of the solar array) in an orthogonal configuration with full spherical view. They provide the coarse Earth and Sun vector by measuring the temperatures of two plates per head with different thermo-optical properties (mirror and black).

The three-axis rate measurement unit is used in Safe Mode for rate measurement independent of the inertial measurement unit used in all other modes.

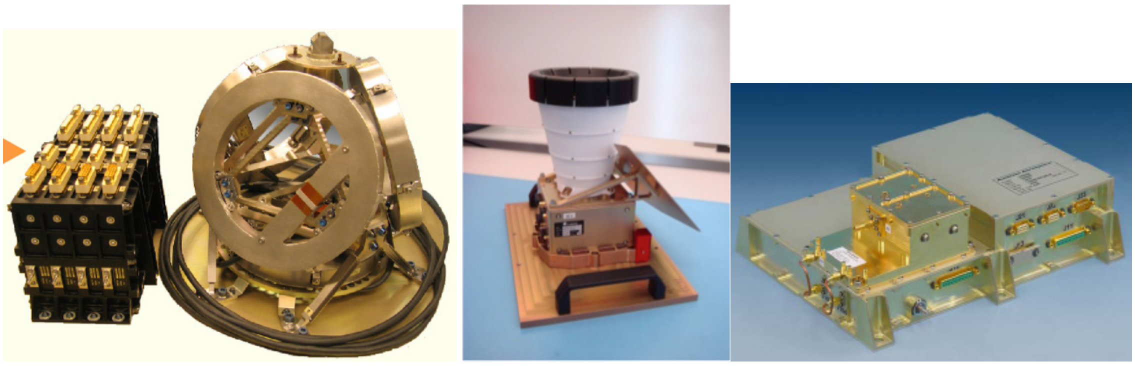


Fig. 9: *Astrix-200 inertial measurement unit, Jena-Optronik Astro APS star tracker (Jena Optronik) and RUAG GPS receiver (RUAG).*

Fig. 10: *Left: ZARM 140-2 magnetic torquer (ZARM), Right: ZARM FGM-A-75 magnetometer (ZARM).*

Fig. 11: *Left: Honeywell MIMU rate measurement unit (Honeywell), Right: One measurement head of the coarse Earth and Sun sensor.*

Four reaction wheels with three out of four redundancy are used as primary actuators in Normal Mode. Nominally all four wheels are in operation and the additional degree of freedom is used for null space control to avoid zero crossings. In case of one wheel failed full performance is achieved with three wheels, however, zero crossings can no longer be avoided. Wheel momentum is controlled through a continuously operating off-loading loop with magnetic torquers. The wheels are mounted on Airbus Defence and Space developed elastomer dampers (Viton type) to minimise the transmission of micro-vibrations to the instrument and avoid detrimental impact on the image quality.

The two cold redundant branches with four 1N monopropellant thrusters each together with the Hydrazine blow down tank are integrate into the propulsion module. As a self-standing assembly the propulsion module can be integrated and tested individually before being integrated into the main structure through the launcher interface ring. The thrusters are tilted with respect to the satellite x-axis to provide full three dimensional torque for attitude control in Initial Acquisition and Safe Mode using pulse width modulation as well as during orbit manoeuvers using off-modulation.

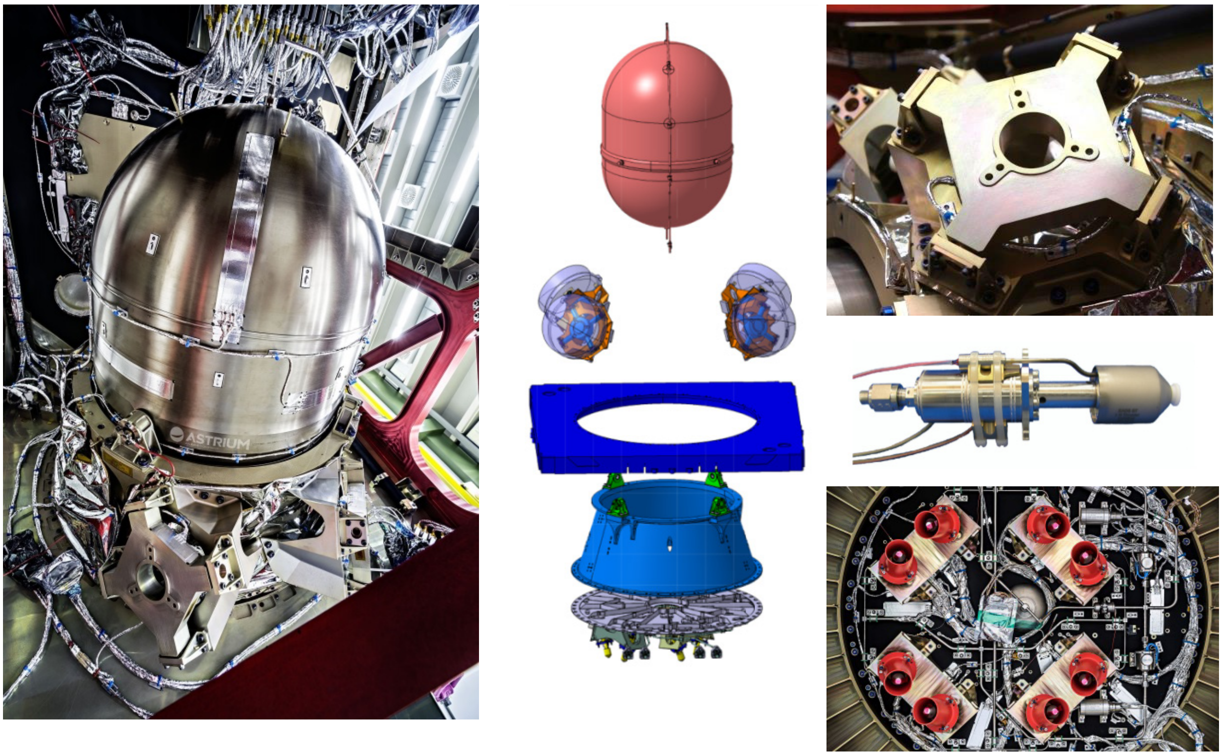


Fig. 12: *Left: The propulsion module with reaction wheel mounts, Right: The elastomer wheel isolator, the Airbus CHT1N-6 thruster and the tilted thruster configuration with red protection covers.*

The internally redundant solar array drive rotates the solar array in all modes. On the sunlit side of the orbit the rotation rate is equal to the orbital rate (0.06°/s) for Sun-tracking. In eclipse a fast rewind (0.2°/s) is performed to be ready for Sun-tracking again when leaving eclipse. Micro stepping is used for Sun-tracking to minimize vibrations. Since no imaging is performed in eclipse the disturbances created by the change of the rotation direction and the fast rewind have no impact on the mission. The twist capsule design of the solar array drive avoids the need to transfer the signals of the coarse Earth and Sun sensor head mounted on the tip of the solar array for undisturbed view over slip rings.

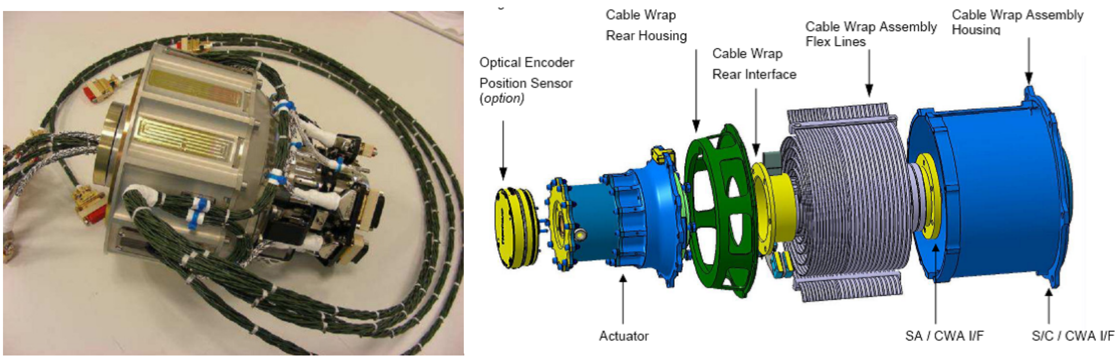


***Fig. 13:** RUAG SEPTA-34-C twist capsule solar array drive (RUAG).*

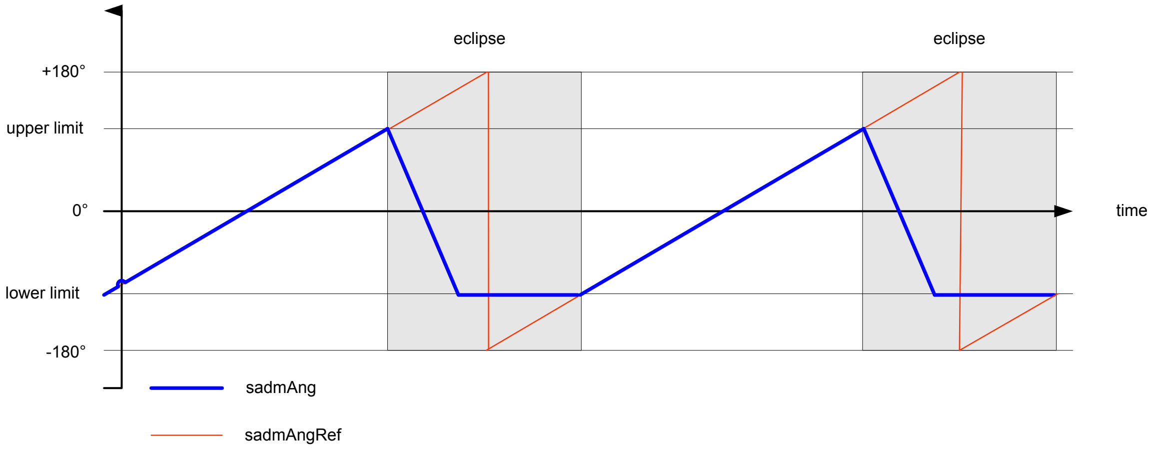


***Fig. 14:** Solar array drive angle evolution, sun tracking with 0.06°/s during sunlit and fast rewind in eclipse.*

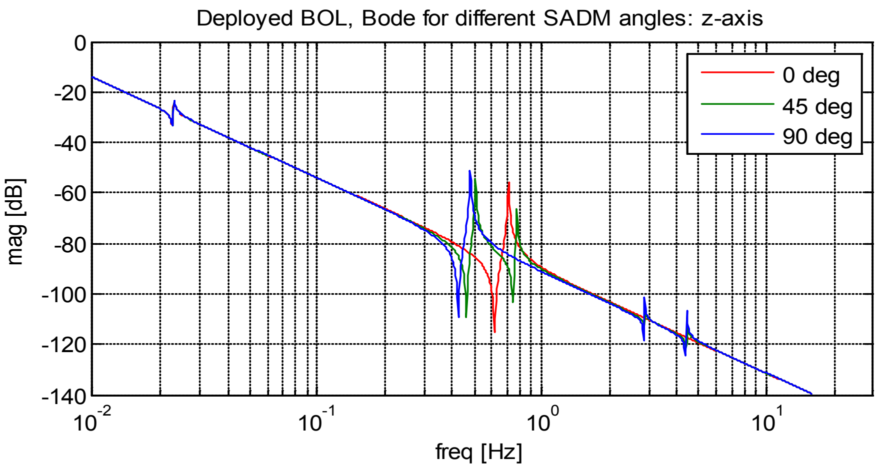


***Fig. 15:** Bode diagram showing the first sloshing mode at ~ 0.02 Hz and the solar array flexible modes (first mode ~ 0.4 Hz) for different solar array drive angles.*

C. SOFTWARE ARCHITECTURE

The AOCS algorithms implementing the control laws form the core of the attitude control application. The interface between the data handling part and the attitude control algorithms is the data pool, a shared memory area where the data is exchanged.

Therefore the attitude control algorithms have a clear interface with no direct access to hardware or operating system functions which allows independent development and testing of the attitude control algorithms software with straight forward integration into the overall system.

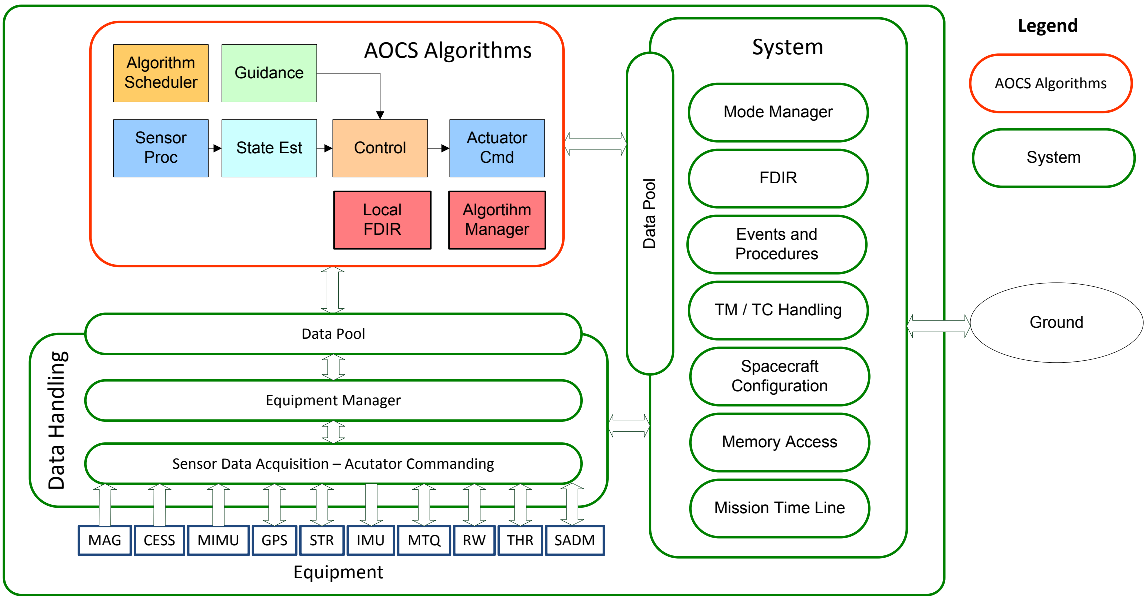


Fig. 16: The software architecture provides a clear interface for independent algorithms development and testing.

D. AOCS ALGORITHMS

The AOCS algorithms implement sensor processing and management, state estimation, attitude guidance, the controllers, and actuator commanding and management as well as algorithm mode management and local failure detection, isolation and recovery, for example consistency checks, data replacement and selection from hot redundant units.

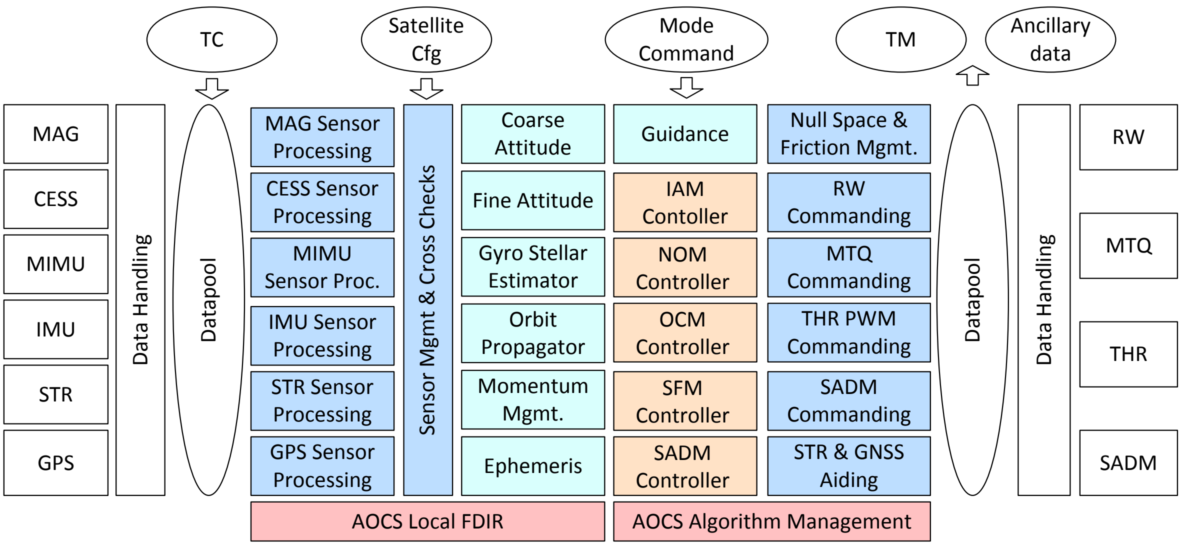


Fig. 17: AOCS algorithm modules.

INITIAL ACQUISITION AND SAFE MODE

The Initial Acquisition and Safe Mode uses the Sun and Earth vectors from the coarse Earth and Sun sensor, the satellite angular rate vector from the inertial measurement unit (in IAM) respectively the miniature inertial measurement unit (in SFM) and the magnetic field vector measured by the magnetometers. Consistency checks are applied on all sensor data. In the Coarse Attitude module roll and pitch angles are determined from the measured Earth vector. The yaw angle is determined from the orbital rate, which is estimated as the difference of the inertial rate measured by IMU or MIMU and the rate relative to the local vertical local horizontal frame computed by pseudo differentiation of the Earth vector.

A pure rate controller is used for the rate-damping phase. After acquisition of the final Earth pointing attitude via Earth Acquisition and Yaw Acquisition sub-modes, steady state control is performed with PD control for roll and pitch angles and proportional rate control the around the yaw axis.

The computed control torque is distributed to the magnetic torquers and the thrusters. Magnetic torquer commanding uses the magnetic field vector provided by the magnetometers to determine the magnetic dipole moment needed. Thruster commanding uses standard pulse width modulation. During rate damping phase the thrusters are naturally the main actuators while in steady state thruster actuation shall be minimised. Therefore an attitude and rate dependent dead-band is applied for thruster commanding which together with proper controller tuning allows attitude control in steady state with only few or even no thruster pulses.

NORMAL MODE

Normal Mode control is based on star tracker attitude measurement and rate measurements by the inertial measurement unit. In the Fine Attitude module least square fusion of the Star Tracker quaternions is performed and the attitude is propagated to the current time.

The reference attitude is computed from the GPS measured satellite position and velocity. The on-board orbit propagator propagates the GPS measurement to the current time and ensures availability of the orbit state vector in case of GPS outage.

A PID controller with roll-off filter is used to compute the attitude control torques which are commanded to the reaction wheel array. Reaction wheel array null-space control avoids wheel zero crossings. A wheel friction estimator is used for friction compensation. The Momentum Management module computes the wheel de-saturation torque, which is continuously applied by the magnetic torquers.

For attitude knowledge the star tracker and inertial measurement unit measurements are fed into a Gyro Stellar Estimator, which provides estimation using a constant gain Kalman filter.

ORBIT CONTROL MODE

The Orbit Control Mode is similar to Normal Mode except for the Delta-V sub-mode when four thrusters are fired simultaneously for orbit control. A PID controller is used to compute the attitude control torques needed to compensate the disturbances mainly caused by thruster mismatch or misalignment. The control torques are then applied by off-modulation of the tilted thruster configuration.

SOLAR ARRAY DRIVE CONTROL

The solar array drive is operated in all modes with the solar array tracking the sun on the sunlit part of the orbit and fast rewind in eclipse. In Normal Mode and Orbit Control Mode the reference solar array drive rotation angle is derived from the computed Sun direction. In Initial Acquisition and in Safe Mode the eclipse transition detected by the coarse Earth and Sun sensor is used to switch the solar array drive rotation.

5. EXPECTED PERFORMANCE

During the AOCS design and performance verification campaign extensive simulations with parameter variation have been performed on a high fidelity numerical simulator. In the following figures some typical performance results are displayed to indicate the expected attitude control accuracy.

A. INITIAL ACQUISITION AND SAFE MODE

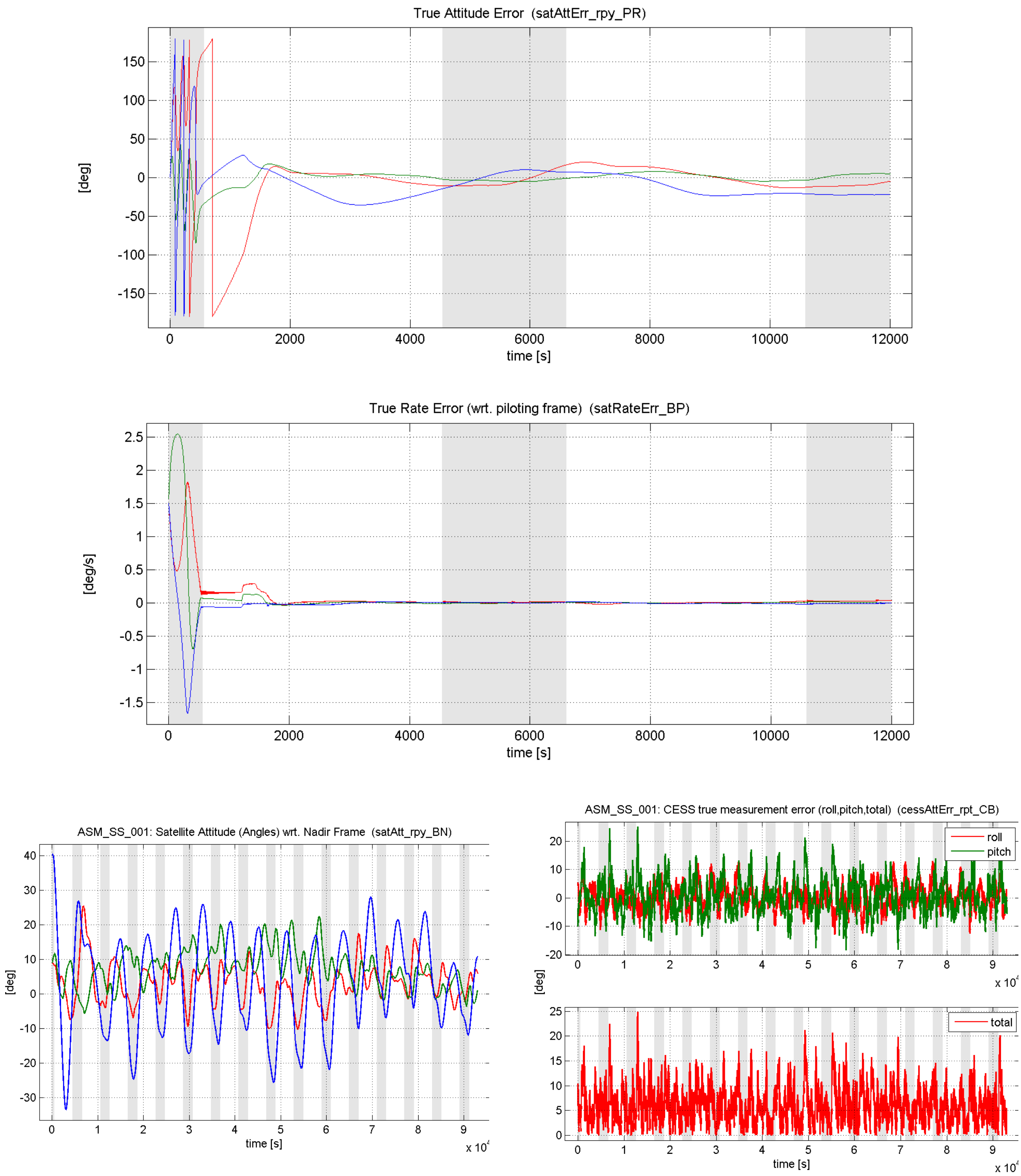
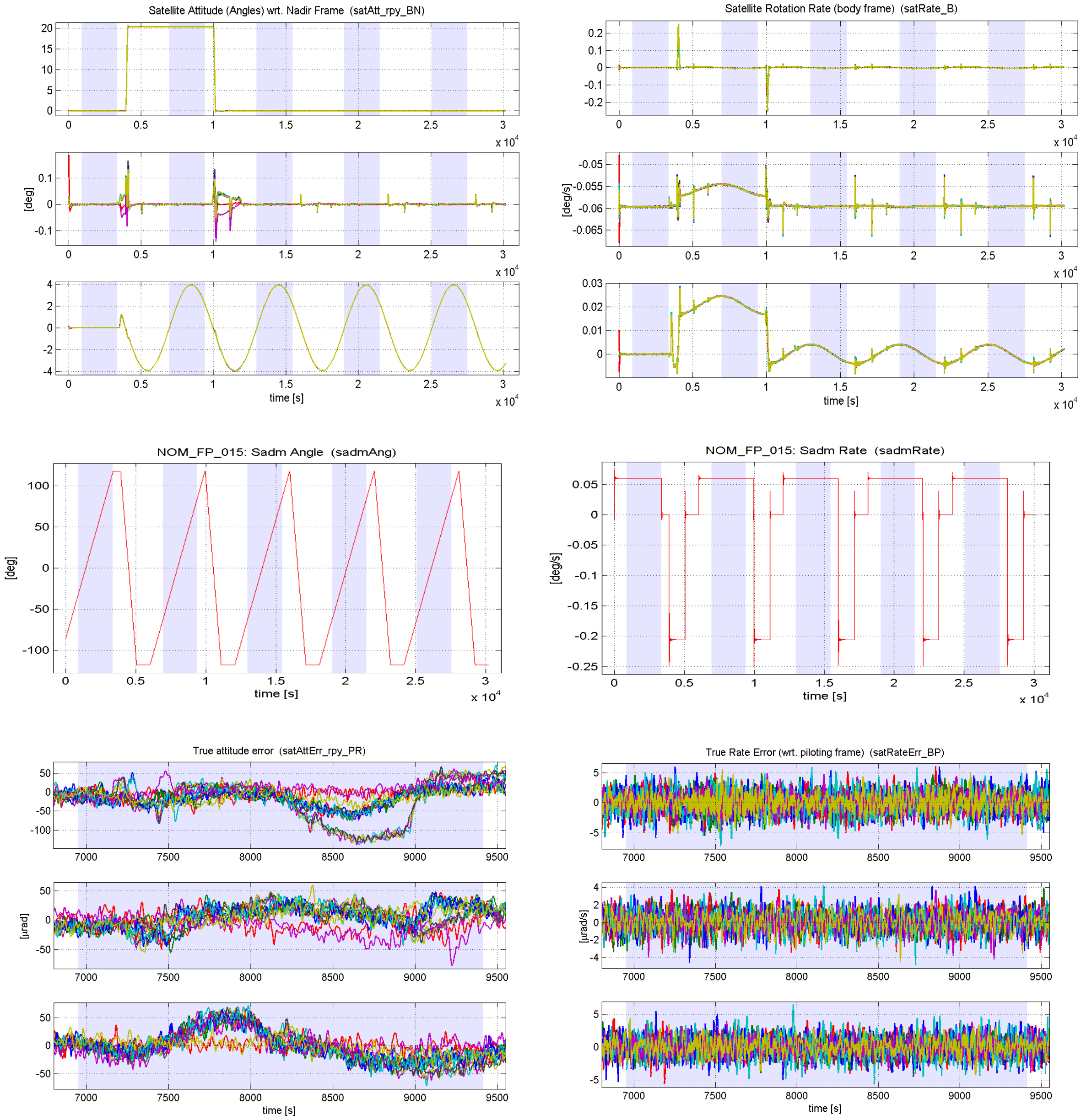


Fig. 18: *Top and middle row: True attitude and rate error for initial acquisition from 1.5°/s initial rate around each satellite axis. Nominal Earth pointing attitude is acquired within less than 2000 sec. Bottom row: Steady state Initial Acquisition and Safe Mode pointing performance over one day (left) . The coarse pointing accuracy is driven by the coarse Earth and Sun sensor measurement error (right). Grey background indicates eclipses.*

B. NORMAL MODE



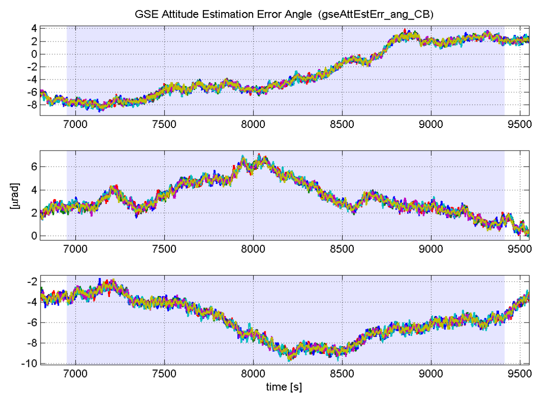


Fig. 19: *Top row: Satellite attitude w.r.t. Nadir frame and inertial rate. Between 4500 sec and 10000 sec the AOCS is switched to Extended Fine Pointing with 20° roll angle. The yaw angle oscillates between ±4° for yaw steering. The spikes in attitude and rate are caused by the rotation changes of the solar array drive. Grey background indicates the imaging phases.*

Middle row: Solar array drive angle and rate.

Bottom row: Zoom of true attitude and rate error for one imaging phase. Left: Gyro Stellar Estimator estimation error over one measurement phase.

6. SPECIFIC SOLUTIONS

A. EARTH POINTING SAFE MODE

The Sentinel-2 attitude control system implements an Initial Acquisition and Safe Mode with Earth-pointing steady state attitude offering

- Fast and easy transition to normal mode without the need for large angle slews;
- Stable thermal conditions for platform and payload
- Stable power conditions
- Stable equipment field of view conditions allowing undisturbed star tracker and GPS receiver operations before transition to Normal Mode.
- No need of a priory state information at mode entry
- Full spherical view of the coarse Earth and Sun sensor - no search operations necessary

The final Earth pointing attitude is based on Earth vector and inertial rate measurements. Roll and pitch angles are deduced from the Earth vector, the yaw angle is estimated from the orbital rate which can be computed as the difference of the inertial rate and the change of the Earth vector in the satellite fixed reference frame.

For the initial rate damping and acquisition phases the tilted thruster configuration provides sufficient torque around all spacecraft axes to cope with high initial rates. However, in the final steady state magnetic actuation by the magnetic torques is preferred and thruster usage shall be minimized to safe fuel and avoid thruster operation in short pulse mode. Therefore a dead-band considering attitude and rate is applied on the thruster commanding to enable the thrusters only in case of insufficient control provided by the magnetic torquers.

But how to design the magnetic PD control loop to optimally exploit the magnetic torquers capability and minimise the use of the thrusters? In the following we investigate the question whether the estimated yaw angle shall be fed back for control (gain for yaw angle feedback $K_{pz} \neq 0$) or not ($K_{pz} = 0$). To do this, we analyse the stability of the control loop with magnetic torquer actuation only.

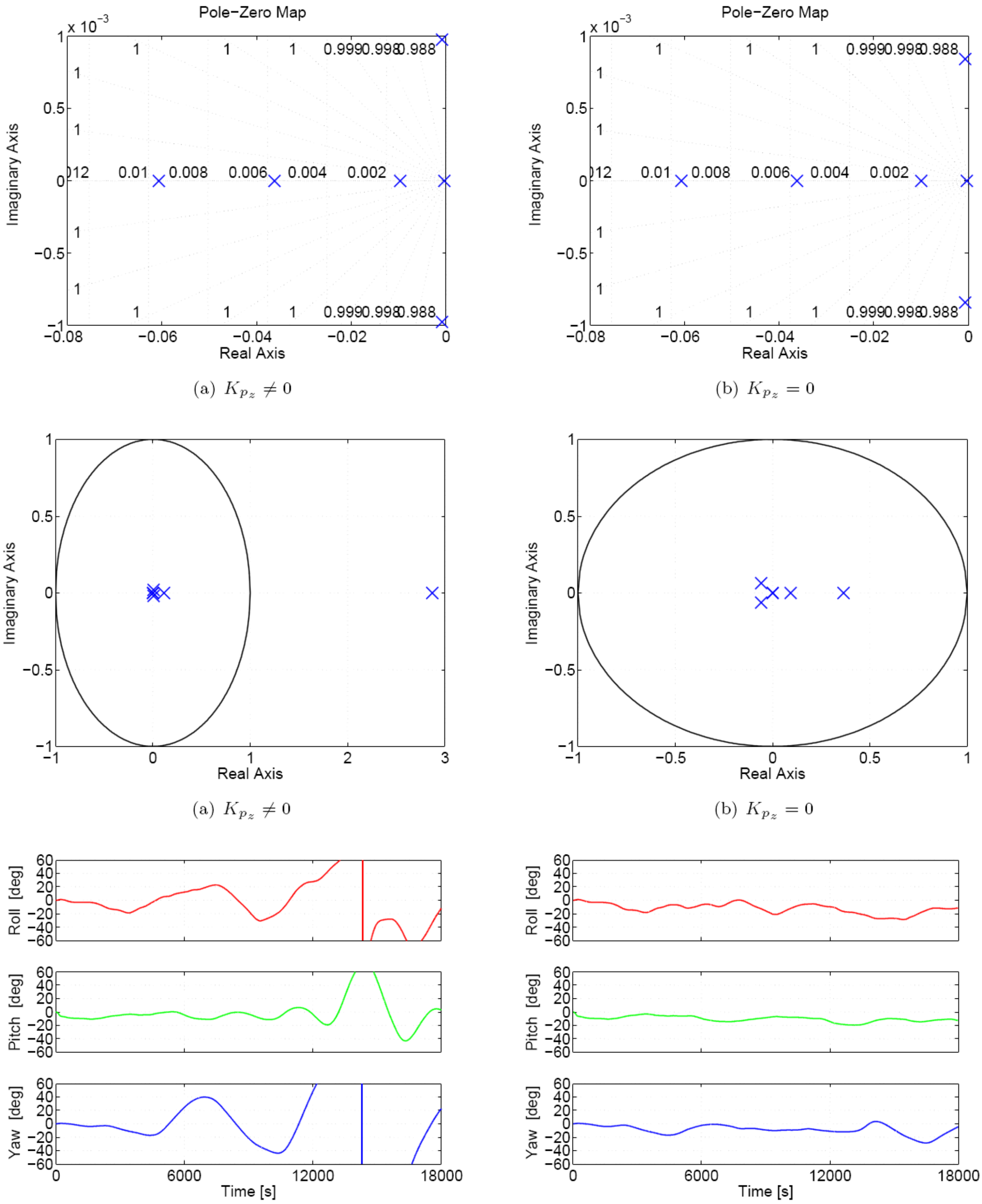
The simplest approach is to consider this control loop as a linear time invariant system. As shown in Fig. 20, top row, the control loop seems to be stable for both alternatives. However, in numerical simulations on a high fidelity simulator the system seems to become unstable with yaw angle feedback (Fig. 20, bottom row left).

The control loop using magnetic torquers is by nature not a time-invariant but a periodic system with periods of ~100 min (one orbit) and ~24 h (one Earth rotation) due to the interaction of the magnetic torques with the Earth magnetic field. Disregarding the time variance of the system can lead to the false conclusion of a stable system, while the periodic system is in fact unstable.

For proper evaluation of the stability the Floquet theory for periodic systems (see for example [7]) is employed. Fig. 20, middle row, presents the eigenvalues of the corresponding monodromy matrix. The control loop with yaw angle feedback has one eigenvalue outside the unit circle indicating instability of the periodic system in line with the result of the numerical simulation.

In the actual attitude control system with magnetic torquers and thrusters such instability of the magnetic torquer control loop only will not lead to a loss of satellite attitude. However, thruster usage and fuel consumption will increase due to the thrusters taking over once the attitude error exceeds the thruster actuation dead-band.

For the complete detailed discussion please refer to [5].



(a) $K_{p_z} \neq 0$ (b) $K_{p_z} = 0$

(a) $K_{p_z} \neq 0$ (b) $K_{p_z} = 0$

(a) $K_{p_z} \neq 0$ (b) $K_{p_z} = 0$

Fig. 20: *Stability analysis of the periodic control loop for magnetic torque actuation [5]. Left column: with yaw angle feedback, right column without yaw angle feedback. Top row: Poles and zeros of the linearized time invariant system implying a stable system, Middle row: Eigenvalues of the monodromy matrix for the periodic system, the case with yaw angle feedback is now identified as unstable. Bottom row: Numerical simulation confirms the result.*

B. ATTITUDE KNOWLEDGE

A clear driver for the attitude control system is the required absolute attitude knowledge error ≤ 10 μrad per axis at 95.5 % confidence level which can only be achieved by proper fusion of star tracker attitude and inertial measurement unit rate measurement data.

To fulfil this requirement a series of design solutions have been implemented:

- Accommodation of three high accuracy APS star trackers (Jena Optronik Astro APS, total noise level < 10 μrad 1 σ) on a thermally stable structure mounted on the Multi Spectral Instrument to reduce thermally induced misalignment between the star trackers and the instrument line of sight. Each star tracker is equipped with a dedicated radiator to avoid conductive heat flow through the star tracker interface into the mounting structure causing thermal distortion. The angular separation between the star trackers is maximized to allow nominal performance using any two out of the three units (minimal separation ~60°);
- Accommodation of the measurement head of a high accuracy inertial measurement unit (Airbus Astrix-200, angular random walk < 0.00015 $°/h^{1/2}$) on the Multi Spectral Instrument;
- In flight identification and calibration of the star tracker mutual alignments and the alignment with respect to the instrument line of sight;
- Implementation of a Gyro Stellar Estimator with a constant gain Kalman filter in the AOCS algorithms which fuses 10 Hz IMU and 2 Hz STR measurement data.

The Gyro Stellar Estimator is used for attitude knowledge only. For attitude control least square fusion of two out of the three star trackers is employed, which however is obviously not sufficient to meet the required attitude knowledge.

A trade-off of different solutions for the accommodation of the inertial measurement unit measurement head and the design of the Gyro Stellar Estimator has been performed, which is documented in [6]:

- Measurement head inside the spacecraft vs. mounted on the Multi-Spectral Instrument;
- Need for state augmentation of the Gyro Stellar Estimator to estimate time varying misalignment between inertial measurement unit and star trackers;
- Estimation of time-correlated star tracker noise terms vs. covariance tuning.

Finally, the measurement head of the inertial measurement unit has been placed on the Multi Spectral Instrument and a six state Kalman filter with covariance tuning to account for time correlated noise is used.

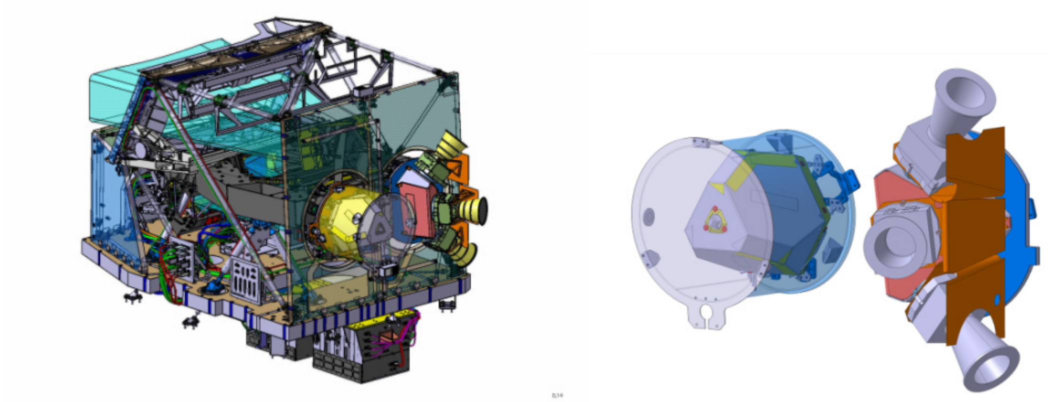


Fig. 21: *The inertial measurement unit sensor head and the star trackers mounted on the Multi Spectral Instrument to minimise thermal distortion.*

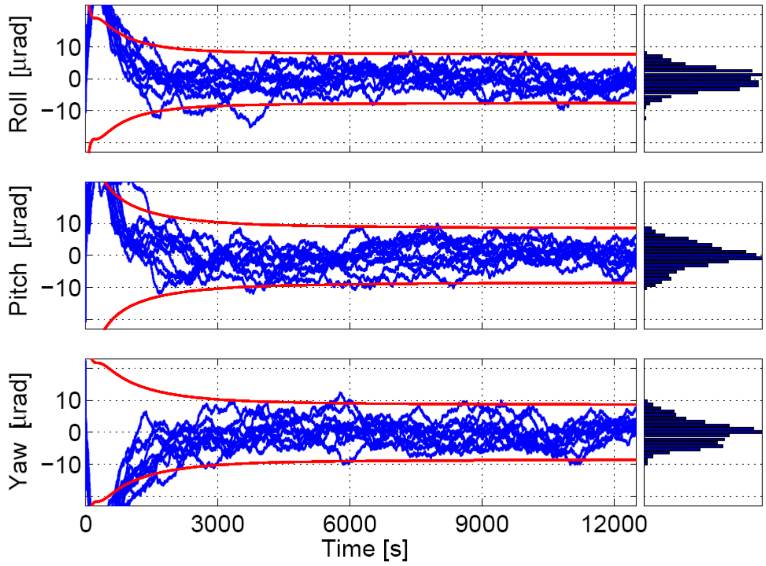


Fig. 22: *Simulated attitude estimation performance provided by the Gyro Stellar Estimator [6].*

C. SADM DISTURBANCE IDENTIFICATION

A detailed solar array drive simulation model has been developed to assess the solar array drive impact on the satellite pointing considering all relevant parts and properties including gear stiffness, backlash, friction, damping and detent torque.

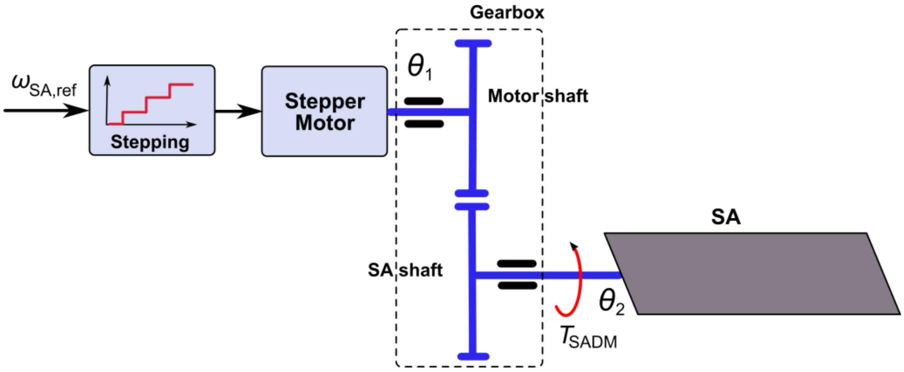


Fig. 23: *Schematics of the solar array drive simulation model.*

Proper parameterization of the model proved to be a non-trivial task, and therefore an identification of the model parameters using a hardware test setup was performed.

For the parameter identification, a qualification model of the RUAG SEPTA-34-C solar array drive has been mounted on a Kistler measurement table equipped with force sensors. An additional torque sensor is used to measure the axial torque exported by the SADM. A mass attached to the SADM represents the rotational inertia of the solar array.

The measured axial torque has been used to identify the model parameters with the help of the Matlab System Identification Toolbox. Fig. 25 presents a comparison of the measured (blue) and the simulated axial torque in time and frequency domain showing a very good match for the final identified parameterization (red), in particular for the location of the resonance peaks. The green line shows the axial torque for the preliminary model parameterization deduced from engineering data before parameter identification.

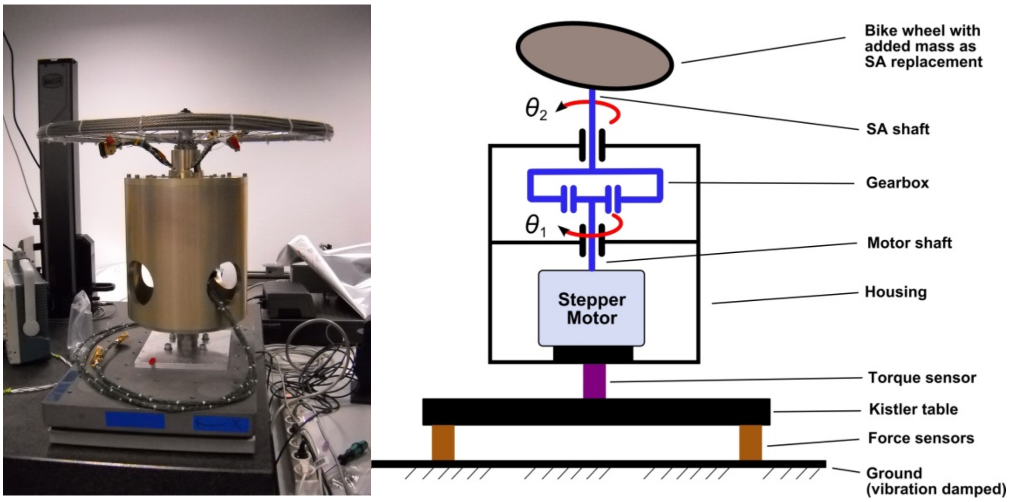


Fig. 24: *Test set-up for solar array drive parameter identification.*

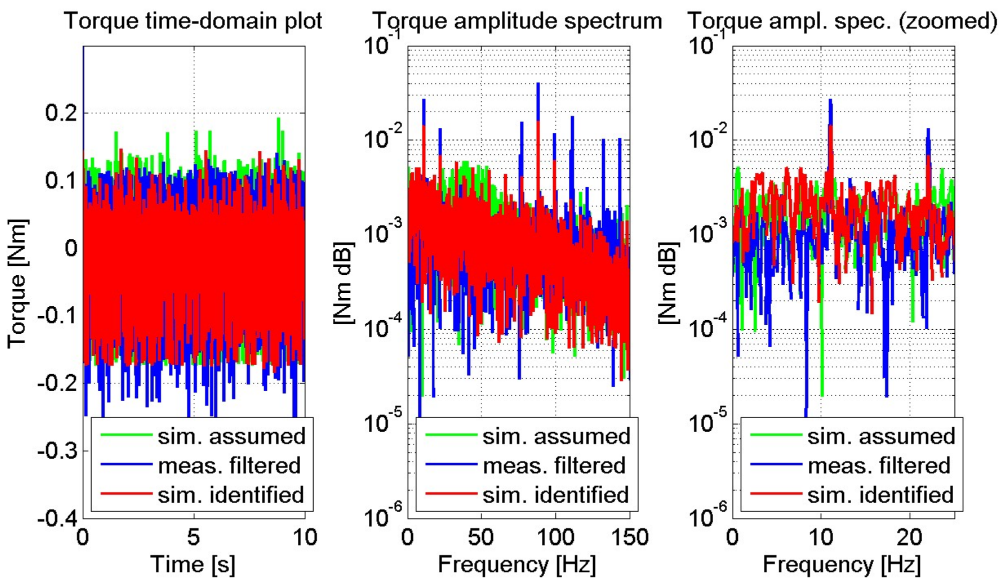


Fig. 25: *Comparison of measured and simulated axial solar array drive torque for model validation in time and frequency domain. Blue: measured torque, green: simulated torque with preliminary model parameters, red: simulated torque with identified model parameters.*

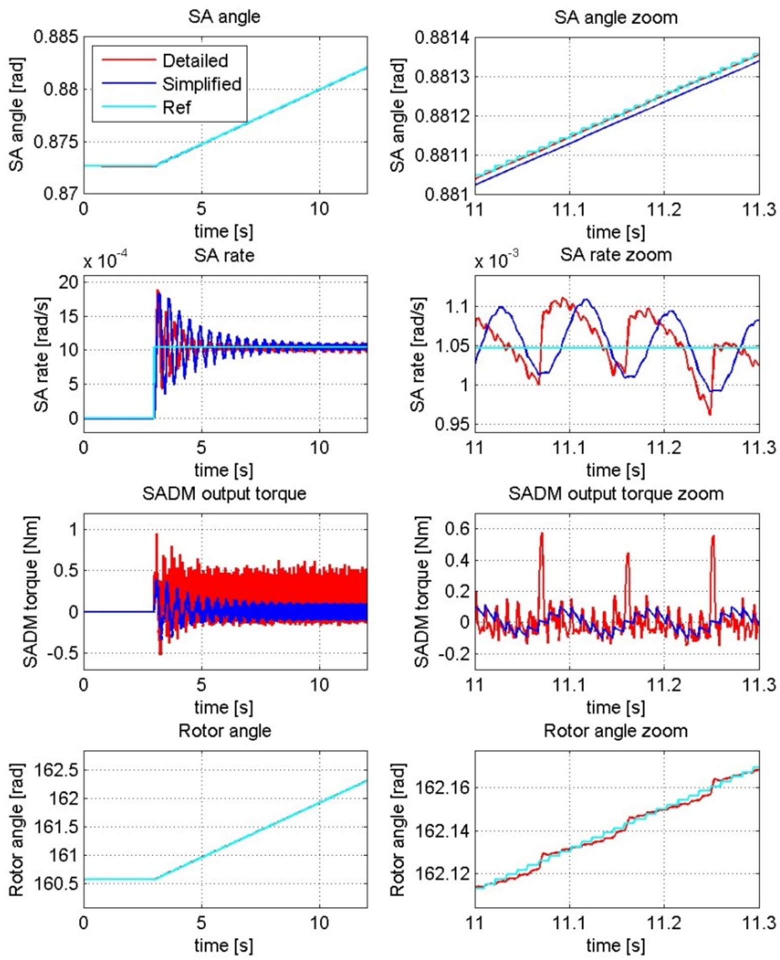


Fig. 26: *Time domain solar array drive simulation results for micro-stepping, 1 kHz simulation frequency.*

D. END-OF-LIFE DISPOSAL

Since uncontrolled re-entry is sufficient, end-of-life disposal for the Sentinel-2 satellite was found to be uncritical with limited impact on the satellite design. As outcome of a trade study, the following scenario was defined for disposal operations with minimum impact on the satellite's power and thermal state.

1. Controlled descent from nominal 786 km altitude orbit to a circular 653 km altitude disposal orbit to leave the highly populated 800 km region quickly. Due to the low thruster capacity at end of life the descent is performed by 116 delta-v manoeuvres of 10 min each. The total delta-v is 73 m/s consuming 47 kg of fuel. One manoeuvre per orbit is performed;
2. Satellite passivation and un-controlled re-entry within 25 years.

The disposal operations can be executed using standard orbit control manoeuvre flight procedures. Although aerodynamic drag increases by a factor of three, the AOCS is still able to safely control the satellite attitude.

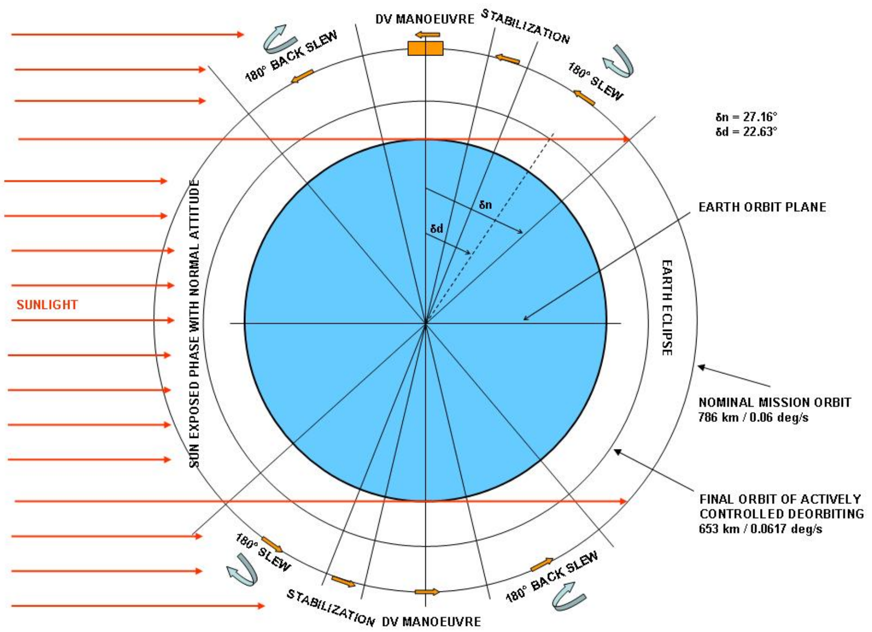


Fig. 27: *For end-of-life disposal the satellite orbit is lowered to a 653 km circular orbit followed by satellite passivation and uncontrolled re-entry within 25 years.*

7. IMPLEMENTATION

The AOCS algorithms, driving the AOCS functionality and performance, have been developed and implemented using a model based development approach. The clear data pool interface enables independent development and pre-validation of the AOCS algorithms before specification for software coding.

In the first step a model of the AOCS algorithms is implemented into a high fidelity Matlab / Simulink functional simulator. The algorithm implementation is accompanied by continuous testing starting at unit level and evolving up to full algorithm open and closed loop tests to ensure correct algorithm behaviour. The algorithm model, documented in the Control Algorithm Specification, together with unit and open loop test vectors then serves as specification for the algorithm software coding.

After software coding and successful open loop testing the algorithm software code is re-integrated in the functional simulator for AOCS design and performance verification. Selected test cases are later used as reference test cases for software and hardware in the loop test benches.

In the case of Sentinel-2 the algorithm software has been coded manually by CRITICAL Software corporation of Portugal as subcontractor. However, Airbus Defence and Space has recently developed a complete software development process based on automatic code generation directly from the algorithms model as natural extension of the model based development presented here. Besides the obvious savings due to the removal of the coding process, software verification activities and metrics like code coverage are moved to the algorithm model thus leading to a faster and more robust development schedule with minimum interdependencies.

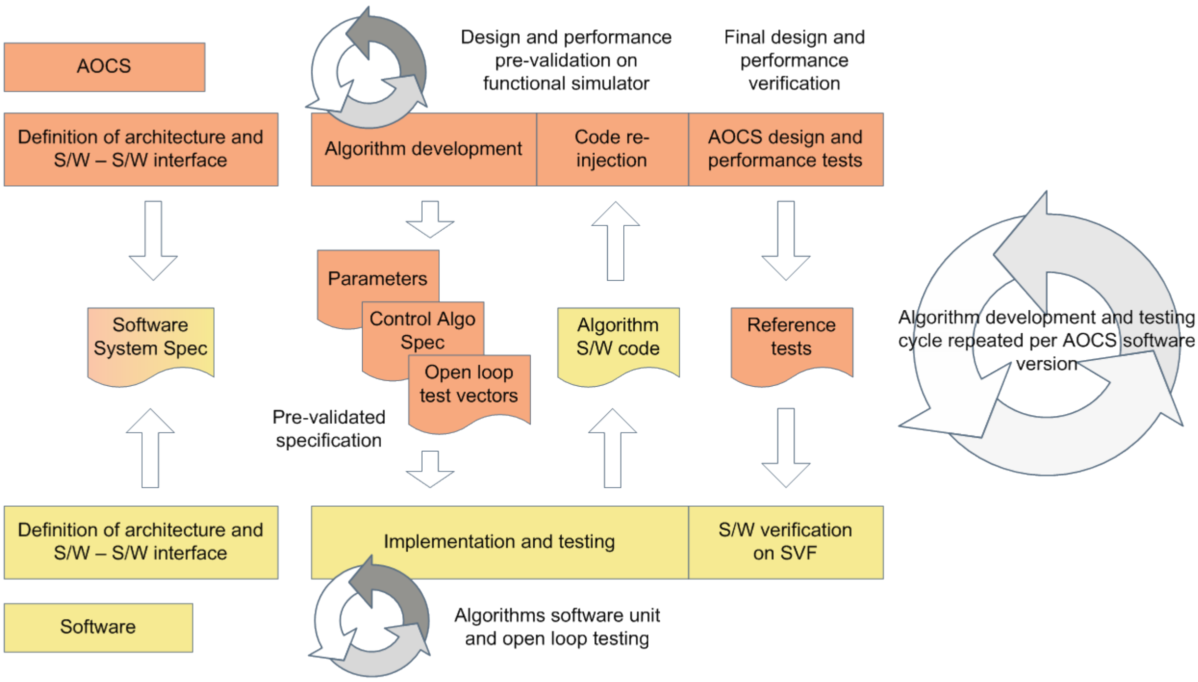


Fig. 28: *The model based algorithm software development process allows independent development and pre-validation of the AOCS algorithms before specification for software coding,*

8. CONCLUSION AND OUTLOOK

The Sentinel-2 attitude control system is well suited to match the requirements for pointing and attitude knowledge. Specific solutions include the Earth-pointing safe mode and the high-accuracy attitude estimation provided by a sensor set mounted on the Multi Spectral Instrument and a Gyro Stellar Estimator. The reaction wheels are mounted on elastomer isolators to minimise disturbances caused by micro vibrations. A model-based design approach is used for algorithms development, which can be extended to automatic code generation.

Integration of the first two Sentinel-2 satellites is currently well under way at Airbus Defence and Space in Friedrichshafen, Germany. Integration and functional testing of Sentinel-2A will be completed by summer 2014 to be ready for the environmental test campaign in the second half of 2014. Launch of Sentinel-2A is scheduled for Q2 2015 with Sentinel-2B following roughly one year later.

Airbus Defence and Space is currently preparing a proposal for two additional flight units to replenish the operational constellation over a 20 years' timeframe. This Copernicus program extension will serve a large user community on the basis of long-term, high-revisit, and high-resolution multi-spectral observations of the Earth's land surfaces.

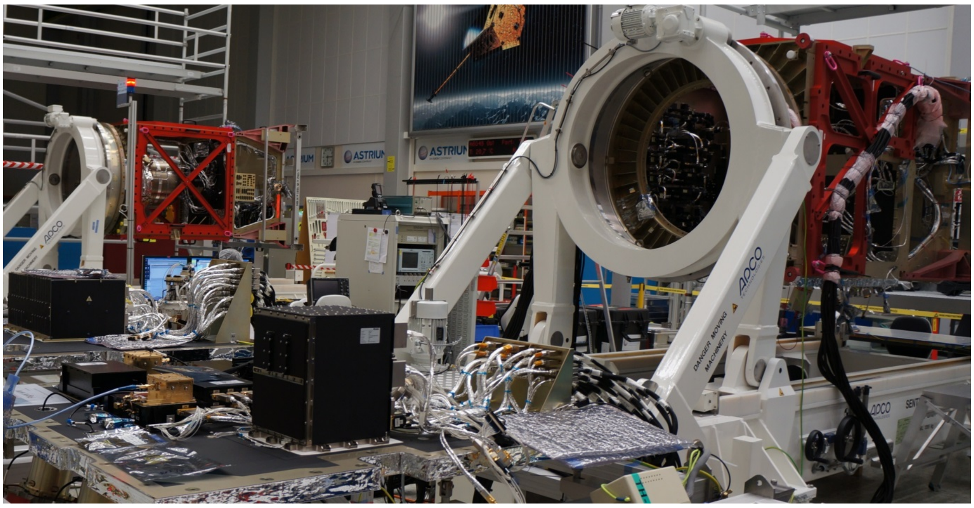


Fig. 29: *Sentinel-2A and Sentinel-2B during integration in the clean room.*

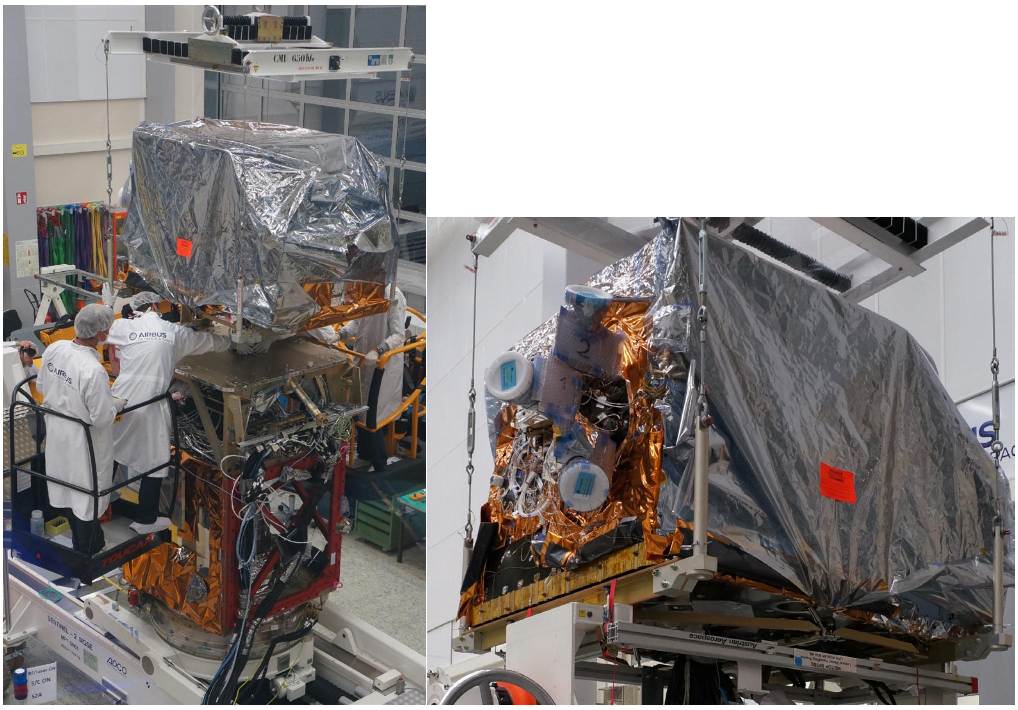


Fig. 30: *Left: Mating of platform and MSI, Right: Star trackers mounted on the MSI*

9. REFERENCES

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10. DISCLAIMER

The work described in this paper was carried out under a programme of and funded by the European Space Agency and the European Union.

The view expressed herein can in no way be taken to reflect the official opinion of the European Space Agency and or the European Union.

11. ANNEX: SENTINEL-2 FACT SHEET

sentinel-2

→ THE OPERATIONAL COPERNICUS OPTICAL HIGH RESOLUTION LAND MISSION

Last update August 2013

MISSION OBJECTIVES

European wide-swath high-resolution twin satellites super-spectral imaging mission designed for data continuity & enhancement of Landsat and SPOT-type missions, for COPERNICUS operational land and security services. These applications include:

- land cover, usage and change-detection-maps
- geophysical variable maps (leaf chlorophyll content, leaf water content, leaf area index, etc.)
- risk mapping
- fast images for disaster relief

MISSION PROFILE

- Launch: 2014 › Launcher: Vega or Rockot
- 7 years lifetime (consumables for 12 years)
- Sun Synchronous Orbit at 786 km mean altitude
- Mean Local Time at Descending Node: 10:30
- Global revisit time: 5 days with 2 satellites flying concurrently (3 days at 45° latitude)
- Twin satellites on the same orbit, 180° apart from each other
- Land coverage: -56° to + 83° latitude
- Maximum imaging time per orbit: 40 minutes
- Nominal nadir pointing, extended viewing capabilities
- Geo-Location: 20 m (2σ) without Ground Control Points
- Calibration: radiometric calibration on-board
- Security: TC authentication
- Operational configuration comprises 2 satellites

SATELLITE PLATFORM

- 3 axis stabilized earth pointing
- Star tracker, inertial measurement unit and 2-band GPS receiver for precise attitude and position knowledge
- Rate measurement unit, coarse earth sun sensor, magnetometer and magnetic torquers, thrusters, wheels
- Propellant: 117 kg Hydrazine (N_2H_4)
- Onboard position knowledge: <20 m (3σ)
- Onboard attitude knowledge: <10 µrad (2σ)
- Launch mass: 1200 kg
- Satellite dimensions (Stowed): 3.4 m x 1.8 m x 2.35 m
- Electical power: › Solar Array: 7.2 m², 1700 W (EOL), GaAs Triple Junction Cells › Battery Capacity: 87Ah (EOL)
  - Satellite power consumption: 1.4 kW (nominal mode)
- Payload data storage capacity:
  - 2 Gbits (End-of-Life) TM/TC storage capacity;
  - 2.4 Tbit (EOL) mission data storage capacity
- Communication links: › X-Band Science Data: effective 520 Mbps (8 PSK); › Optical Communication Payload for mission data retrieval through EDRS; › S-Band TT&C: 64 kbps up (SPL/PM), 128 kbps (SPL/PM) / 2048 kbps (OQPSK) down
- Thermal control: passive with Deep Space Radiator. Thermistor controlled Heater Circuits
- Reliability: > 0.7 › Availability: 97%

SATELLITE PAYLOAD

MSI (Multi Spectral Instrument)

- Imaging principle: filter based push broom imager
- Telescope design: Three mirror anastigmatic telescope with Silicon Carbide mirrors and structure, and dichroic beam splitter to separate VNIR and SWIR spectral channels
- Focal plane arrays: Si CMOS VNIR detectors, HgCdTe SWIR detectors, passively cooled (190 K)
- Electronics: front end, video and compression electronics, including state-of-the-art wavelet-based data compression
- Combination of on-board absolute calibration with a solar diffuser covering the full FoV, dark calibration over ocean at night, and vicarious calibration over ground targets
- 13 spectral bands: 443 nm– 2190 nm (including 3 bands for atmospheric corrections)
- Spectral resolution: 15 nm– 180 nm
- Spatial resolution: 10 m, 20 m and 60 m
- Swath: 290 km
- Radiometric resolution/accuracy: 12 bit / < 5%

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European Space Agency