Microscope: A Microsatellite for Equivalence Principle Measurement in Space

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ABSTRACT

The MICROSCOPE mission is developed in the frame of the CNES Myriade micro satellite family. The project is currently ending its Phase B, the Preliminary Design Review has been held in March 2011 and the launch is planned in 2015.

The scientific objective of the mission consists in a test of the Equivalence Principle (EP) between gravitational mass and inertial mass with a relative accuracy of 10-15; the payload is composed of a set of two 6-axis differential accelerometers developed by ONERA.

To achieve this goal, a drag free control of the satellite has to be achieved in order to limit the non-gravitational accelerations on the payload below 3.10^{-10} ms-2*Hz-1/2.

This paper begins with a introduction of the mission and the payload, explaining how mission requirements and payload I/F strongly constrain the design of spacecraft (drag free, microperturbation and stability).

The functional chains of the satellite are presented in detail with an emphasis on mechanical and thermal architecture, Acceleration and Attitude Control System (AACS) and Cold Gas Propulsion System (CGPS).

It is shown how the design of the satellite is optimized, melting new advanced technology (Payload, AACS, CGPS) and low cost, well proven methods and equipment of Myriade family.

MICROSCOPE MISSION OVERVIEW

MICROSCOPE, an acronym for ”MICROSatellite with drag free Control for the Observation of the Principle of Equivalence”, is a space mission in universe's science proposed by proposed by ONERA and CERGA Institutes, and approved in 1999 by CNES Scientific Program Committee.

This mission is developed in the frame of the scientific missions exploiting the MYRIADE microsatellite product line.

Equivalent Principle

Mass could be defined in two different ways :

as a inertial mass m_i , i.e. the term of proportionality between the external force acting on a body and its acceleration :

$$\vec{F} = m_i \cdot \vec{\gamma} \tag{1}$$

where F = external force; γ = acceleration; and m_i = inertial mass.

as a gravitational mass m_g , i.e. the “charge” of gravitational force (as the electrical charge for Coulomb force) :

$$F_g = G \cdot \frac{m_g \cdot M_g}{r^2} \tag{2}$$

where F_g = gravity force; G = universal constant of gravitation, r = the distance between the two body; M_g the mass of the body generating the gravitation field and m_g = the gravitational mass.

The Equivalence Principle (EP) postulates a perfect proportionality between the inertial mass and the gravitational mass of a body, whatever its chemical composition.

For two different bodies of inertial masses m_{1i} , m_{2i} and gravitational masses m_{1g} , m_{2g} we should then have :

$$\frac{m_{1g}}{m_{1i}} = \frac{m_{2g}}{m_{2i}} \tag{3}$$

Albert Einstein adopts this principle as a fundamental assumption for the theory of general relativity :

" *The ratio of the masses of two bodies is defined in two ways which differ from each other fundamentally,…, as the reciprocal ratio of the accelerations which the same motive force imparts to them (inert mass),…, as the ratio of the forces which act upon them in the same gravitational field (gravitational mass). The equality of these two masses, so differently defined, is a fact which is confirmed by experiments…*"

(Einstein, The Meaning of Relativity, 1921)

The violation of the Equivalence Principle would lead to evidence of a new interaction that is predicted by many current quantum theories of gravity.

Equivalent Principle has been always verified by experiment.

Table 1: Equivalence Principle experimental verification [1]

Researcher	Year	Method	Accuracy
G. Filipono	500 ?	Free fall	Low
S. Stevino	1585	Free fall	5×10^{-2}
G. Galilei	1590 ?	Free fall	2×10^{-2}
I. Newton	1686	Pendulum	10^{-3}
F. Bessel	1832	Pendulum	2×10^{-5}
Southerns	1910	Pendulum	5×10^{-6}
L. Eötvös	1922	Torsion balance	5×10^{-9}
Renner	1935	Torsion balance	2×10^{-9}
Dicke	1964	Torsion balance	3×10^{-11}
Shapiro	1976	Lunar Laser Ranging	3×10^{-12}
Baeßler	1999	Torsion balance	5×10^{-13}
Aldelberger	2008	Torsion balance	3×10^{-14}
Microscope	2015 ?	Earth orbit	10^{-15}

Microscope aims to test equivalent principle with an accuracy of 10^{-15}.

A confirmation of EP with an improvement of accuracy will help theoretical physicians to eliminate some theories of fundamental forces unification giving a prediction of EP violation not in accordance with the experiment.

A violation of the EP will be a major event in the history of physic defining the limit of the validity of the General Relativity theory.

Mission description

The principle of the mission[2] is to put two masses of different nature having the CoG at the same position in orbit around the Earth and to measure their trajectory when submitted only the gravity force.

If EP is verified (Figure 1) its trajectories will be identical for the same initial condition (position and velocity).

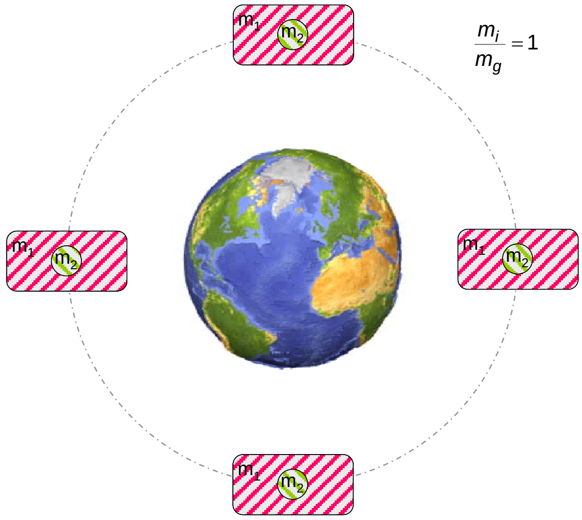

Figure 1: EP verification in space

If EP is violated (Figure 2) its trajectories will be different for the same initial condition.

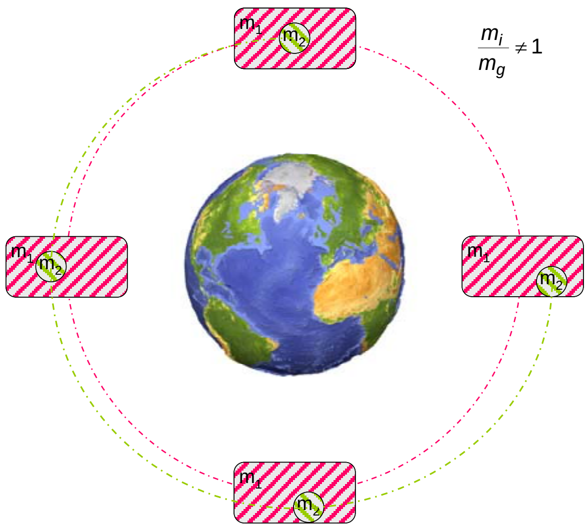

Figure 2: EP violation in space

In fact instead of measuring the trajectory of the masses we measure the force necessary to keep the masses at the same location during the orbit in order to deduce the accelerations γ_1 and γ_2.

If we suppose the EP violated we will have :

$$\vec{\Gamma}_d = \frac{1}{2} \cdot \left(\vec{\gamma}_1 - \vec{\gamma}_2\right) = \frac{1}{2} \cdot \left(\frac{m_{1g}}{m_{1i}} - \frac{m_{2g}}{m_{2i}}\right) \cdot \vec{g} \qquad (4)$$

$$= \frac{1}{2} \cdot \delta \cdot \vec{g}$$

where Γ_d = differential acceleration between two masses; δ = term of violation of the EP which we would compare to zero with an accuracy better than 10^{-15}.

This accuracy goal in LEO gravity field drives to the need to measure a differential acceleration smaller than 10^{-15} m/s^2. This value, which is not compatible with a measurement in continuous mode, could be achieved, using signal processing methods, if the gravity acceleration is modulated at the frequency f_{EP} with respect to the sensitivity direction of the accelerometers.

This modulation could be easy performed in space with a satellite in a circular orbit :

When the attitude satellite remains inertial, the modulation frequency of gravity signal is equal to the orbital frequency f_{or}.

When the satellite is rotating at the frequency f_{sp} around the direction perpendicular to the orbital plane, the modulation frequency of gravity signal is equal to the signed sum of the spin frequency and the orbital frequency.

The frequency f_{ep} which the EP is observed is :

$f_{ep} = f_{or} + f_{sp}$

Practically, the difference of location of Center of Gravity (CoG) of the masses creates a differential acceleration which is impossible to distinguish from a PE violation.

By consequence equation (4) is modified as follows :

$$\Gamma_d \approx M_c \cdot \left(\delta \cdot g + (Tg - Ti) \cdot \Delta - 2 \cdot Tc \cdot \dot{\Delta} - \ddot{\Delta}\right) + B_d + M_d \cdot \Gamma_c + \Gamma_{d^2} + \Gamma_n \qquad (5)$$

Where M_c = common modes default matrix, Tg = Gravity gradient Tensor, Ti = Inertia Tensor (composed by angular velocity and accelerations), Δ = difference of location of CoG of the two masses, Tc = Coriolis Tensor, B_d = differential bias, M_d = differential modes default matrix, Γ_c = Common acceleration of the masses, $\Gamma_d{}^2$ = second order errors, Γ_n = Instrument noise.

M_c , B_d , M_d , Δ and $\Gamma_d{}^2$ values shall be characterized during flight by calibration.

Ti and Γ_c , which is the results of all the non gravity forces acting on the satellite, shall be minimized by the satellite acceleration and attitude control system AACS.

Tg is estimated by orbit determination.

Noise level effect Γ_n could be minimized by filtering the signal over a long period of time T_i , the value of Ti depends mainly from the noise level of the differential accelerometer and the and frequency of the modulation of the gravity signal g.

The spin frequency value is chosen following the frequency response; for Microscope mission 2 different value have been chosen :

- $f_{sp1} = (0.5+\pi)\cdot f_{or}$
- $f_{sp2} = (1.5+\pi)\cdot f_{or}$

Taking into account the noise level expected for the ultra-precise accelerometers of the Microscope payload (10^{-12} ms^{-2}Hz$^{-1/2}$), the aimed accuracy of the test can be achieved with a duration of 120 orbits for inertial pointing attitude measurement and 20 orbits for spin mode attitude measurement.

At the end the mission consists to place, in a low earth orbit with a very small eccentricity, a differential accelerometer inside a spacecraft.

The satellite shall provide a very stable attitude and shall compensate any non-gravitational force acting on the sensor.

The EP measurement is performed several times with different values of f_{ep} in order to evaluate the influence of signal processing method on the results.

Dedicated calibration sessions shall be performed in order to correct the error of the instrument and achieve the accuracy goal of 10^{-15}.

The total duration of the mission is around 9 months including Platform in orbit commissioning but excluding unavailability periods.

Payload description

SAGE (Space Accelerometer for Gravity Experiment) is the differential accelerometer designed and developed by ONERA for Microscope mission.

SAGE is composed by a Sensor Unit (SU), a Front End Electronic Unit (FEEU) and an I/F Control Unit (ICU).

SU (Figure 3) is composed of two concentric electrostatic inertial sensors (i.e. two concentric cylindrical masses embedded in a cage composed of cylinders in gold-coated silica carrying the capacitive electrodes) protected by an invar envelop and maintained in ultra vacuum.

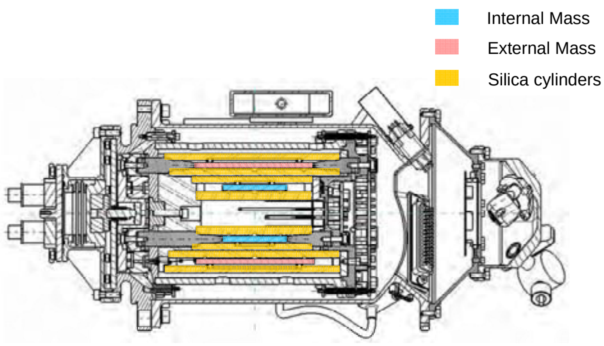

Figure 3 : Internal view of SU

A locking device supports the masses during launch phase; when released the masses are maintained in electrostatic levitation during all the mission.

Measurements of the electrostatic forces and torques, which result from the six servo-loops necessary to maintain each mass motionless, provides the 12 outputs of each SU (six for each mass).

The axial axis of the cylinder constitutes the sensitive axis of the sensor.

The different materials chosen for the EP test are PtRh10 platinum alloy for the inner proof-mass and TA6V titanium alloy for the outer one.

The SU is completed by FEEU which deals with the capacitive detection and ensures the AC/DC conversion, and by ICU which includes the servo-controller and the interfaces with satellite subsystems.

In order to confirm that the experimental protocol achieves the expected accuracy, a second differential accelerometer SAGE is installed aboard the MICROSCOPE satellite; both of its proof-masses are made of the same material (platinum alloy) and the extraction of δ is expected to give zero.

Both SUs are mounted on a common plate called SUMI assuring the alignment of sensitive axis; both ICU are integrated in a common equipment called ICUME.

The combination of the two differential accelerometers (SAGE-EP and SAGE-REF) constituting the scientific instrument is called T-SAGE for Twin-SAGE.

This instrument (Figure 4) constitutes the Payload of Microscope mission ; it is under development at ONERA DMPH Châtillon (France),.

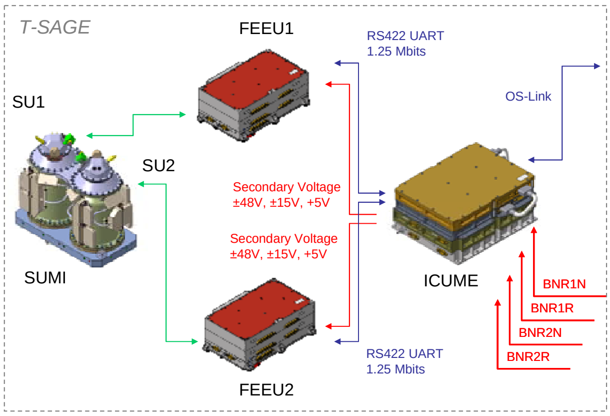

Figure 4 : Microscope Payload constitution

Mission requirements

Orbit altitude is a comprise between several factors :

Low orbits maximize the gravimetric signal and reduce the reentry time after the end of the mission.

High orbits reduce the atmospheric drag and parasitic effects of the Earth on the spacecraft (thermal perturbation and Star Tracker Field of View clearance).

The eccentricity shall be very low along all the mission duration.

Sun Synchronous Dawn/Dusk Orbits are preferred for power budget and thermal stability point of view.

Orbital parameter of Microscope are :

- Altitude : 720 km
- Local Time Ascending Node : 6h or 18h00
- Excentricity : 0.005

Other missions requirement are induced by the estimated performances of T-SAGE (scale factor and defaults).

The restitution of the gravity gradient (*Tg* of equation 5) needs :

- a knowledge of the accelerometer position with respect to the Earth center less than 7 m at f_{ep}.
- a knowledge of the spacecraft attitude less than 1 mrad at f_{ep}.,
- an angular stability less than 7 µrad on 3 axis at f_{ep}.

Common acceleration of the spacecraft (Γ_c of equation 5) shall be less than 10^{-12} m/s² at f_{ep}.

The attitude of the spacecraft shall be very stable :

- Angular velocity stability shall be less than 10^{-9} rad/s for rotating EP measurement sessions.
- Angular acceleration shall be less than 10^{-11} rad/s² for all the EP measurement sessions (inertial or rotating).

No moving mass is allowed inside the satellite.

Thermal environment of T-SAGE shall be very stable :

- Thermal stability of SU shall be less than 2 mK at f_{ep} (peak to peak).
- Thermal stability of FEEUs shall be less than 20 mK at f_{ep} (peak to peak).

Mission induced constraints on Spacecraft

The "unusual" Microscope mission requirements have several consequences on the satellite design.

The satellite shall protect the payload from all non-gravitational forces perturbing the EP measurement ; an active control of accelerations and attitude of satellite is necessary to reach the required background level of acceleration.

This specific function of Microscope is called Acceleration and Attitude Control System (AACS). It needs a very sensitive 6 axis sensor and very fine actuators.

In order to avoid specific new development and because of its high sensibility, T-SAGE is used as main sensor of the AACS control loop.

Detected perturbations are counteracted by the Cold Gas Propulsion System (CGPS); It is made of 8 Microthrusters (MT) using gaseous Nitrogen as propellant to produce the few µN of thrust required by AACS.

CGPS is developed by CNES with a major ESA contribution ; its design is mainly based on GAIA MPS Microthrusters developed by TAS-Italy Company.

Very stringent thermal stability requirement of the payload highly constrains the spacecraft design and its thermal control.

Thermal control of the payload shall be separated from the satellite one, both should avoid any active control method which could generate perturbation on T-SAGE.

In order to minimize thermal perturbation on T-SAGE, the payload shall be accommodated inside the spacecraft whereas it is commonly outside in the other Myriade family microsatellites.

Centering and Inertia of the satellite are also constrained by the mission.

The center of gravity of the satellite shall be as close as possible to the center of gravity of the 4 proof-masses during all the mission ; spin axis shall correspond to the main inertia axis of satellite, cross inertia and the difference between the other main inertia shall be minimized in order to reduce the gravity gradient torque on satellite.

The satellite shall guarantee an ultra-stable dynamic environment to the instrument. The requirement on the attitude rate stability in rotating mode around f_{ep} is equivalent to a requirement in attitude stability of the instrument better than 0.166 µrad at a very low frequency. This order of magnitude is far below the thermoelastic deformation between the instrument plate and the measurement axes of every kind of attitude sensors that can be laid out.

The mission needs to reduce any kind of internal micro-perturbations in the satellite; this term gathers three categories of perturbations :

- mechanical : direct forces applied on the prof-masses as the reaction to any displacement inside the spacecraft.
- gravitational : change of the induced gravity gradient when the satellite shape changes.
- magnetic : resulting from the electrical activity in the satellite circuits and equipment.

During eclipse season (3 monthes a year) the transition of satellite into the Earth shadow could generate several phenomena able to perturb the measurement (sudden solar pressure variation, MLI thermoelastic clank, non regulated bus voltage variation, etc..°); for this reason the EP test should be performed only during the period of the year without eclipse.

At the end the MICROSCOPE satellite plays the role of a space laboratory dedicated to a very complex experiment of physics, and for this it shall be able to point the instrument on 3 axis, it shall protect it against non-gravitational forces, and it shall ensure an ultra-stable thermal environment, in particular around the frequency f_{ep}.

SATELLITE DESCRIPTION

Myriade microsatellite family overview

Microscope mission is developed in the frame of the scientific missions exploiting the MYRIADE micro-satellite product line.

The development of the MYRIADE product line has started in 1999 under the lead of CNES with the goal to offer to scientists a versatile tool for testing small payload instruments in the range 60kg – 60W, for low duration missions (typically 2 years), with short development schedule and reduced costs.

The design of MYRIADE micro-satellite is a compromise between high performance, efficiency, robustness and cost. The architecture of the satellite is based on a platform with generic functional chains (AOCS, Energy, Communication, Computer, Structure, Thermal Control), and on a decoupled payload located on the upper part of the platform structure.

The search for very low cost equipment units leads to select a majority of commercial components which sustained a ground qualification to space environment radiations.

The bus (Figure 5) is build around a quasi cubic structure (600x600x550 mm), made with aluminium skins and honey-comb aluminium core.

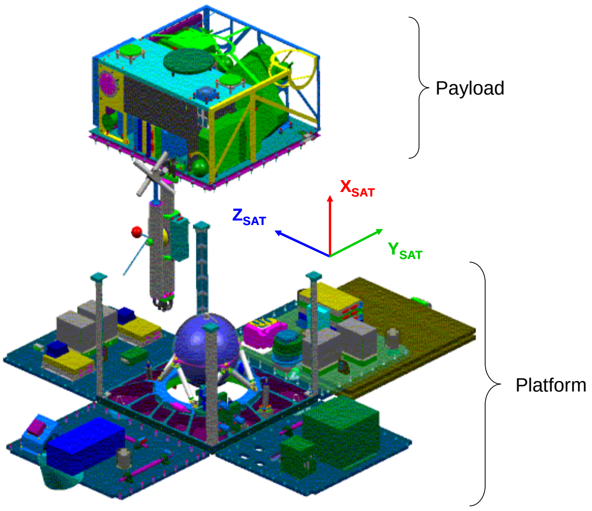

Figure 5 : Myriade product line standard bus

Myriade reference frame is defined as follow :

- X axis : is perpendicular to I/F separation plane and directed from the launcher toward the spacecraft
- Y axis is perpendicular to Solar Generator in stowed configuration and directed from the spacecraft toward the Solar Generator.
- Z axis is defined as X ∧ Y

The -X panel includes the launcher adapter and the propulsion subsystem with its hydrazine tank. The PF equipments are mounted on the ±Y, ±Z walls. The payload is located on the +X panel outside the bus.

Table 2 gathers MYRIADE main characteristics.

Table 2: Myriade PF characteristic and performances

Subsystem	Equipments	Performances
Structure	Alu structure, honeycomb panels	600x600x800 mm. Mass < 150 kg (with payload)
Power	SG (*TAS*) with AsGa cells (*Spectrolab*) Battery (*AEA)* PCDU (*ETCA)* SADM (*OERLIKON)*	2 panels, 0.9 m². Power 200 W max. Battery 14 Ah Voltage 22 to 37 V
AOCS	SAS *(Astrium)* Magnetometer (*IAI)* SST *(DTU)* Gyros *(Litef)* MTG *(IAI)* Reaction wheels : 0.12 Nms *(Teldix)* Propulsion : 4 x 1 N thrusters, hydrazine system Isp 210 s *(EADS Gmbh)*	1 axis, 3 axis pointing capability A priori pointing : < 0.02° (1σ) each axis Pointing stability : < 0.02°/s ΔV available : 80 m/s for 120 kg satellite
Localization /Orbit determination	Performed by Control Center *Option:* GPS	By Doppler measurements : Position at 3 σ : ± 350 m Localisation by GPS : < ± 1 m
On board data management and Command / Control (Payload has its own computer)	OBC with µprocessor T 805 *(CNES design manufacturing by STEEL)* Flight software : *(CSSI)*	5 Mips, 1 Gb memory (EDAC) Datation : ± 15 ms/UT (at 700 km altitude)
Communications S band	RX /TX : *(TES)* 2 antennas *(MBDA)* opposite sides, omnidirectional coverage	Error Bit Rate : 10^{-10} Error Bit Rate : 10^{-10} 10 or 400 kb/s to 600 kb/s – cold redundant 20 kb/s – hot redundant
Payload data downlink with X band *(Option)*	X band emitter *(TAS)* Antenna (*Chelton*)	18 Mb/s to 80 Mb/

The power subsystem is based on a single wing solar generator SG with 2 rigid panels with ultra triple-junction AsGa cells; the solar generator could be rotated by a Solar Array Drive Mechanism (option).

A Power Conditioning and Distribution Unit (PCDU) is in charge of :

- launcher separation detection
- battery (Li-Ion type) regulation.
- power distribution to equipment and payload
- thrusters and magnetotorquers commands
- pyro lines distribution

AOCS design is rather classic. It uses solar sensor, 3 axis magnetometer, high accuracy star sensor, gyros and GPS (option) as sensors, and magnetotorquers, reaction wheels as actuators. Hydrazine propulsion (4 x 1 N thrusters) (option) is only used for orbit control.

Four AOCS modes are used : acquisition/safe mode, transition mode, normal mode and orbit control mode.

On board data management and control/ command perform the following main functions :

- satellite configuration management
- mission plan management
- storage of house keeping and payload data, and transmission to S band station
- implementation of AOCS

The architecture is centralized: processing is achieved by one single On Board Computer with direct links with the other PF equipment units and the payload.

Thermal control is based on use of passive systems (paints, MLI, SSM coatings, …) and SW controlled heaters.

The first mission using Myriade platform has been the scientific and technological satellite DEMETER dedicated to the study of ionosphere perturbations and earthquakes.

DEMETER has been launched in June 2004 and has been retired from service en December 2010 overtaking his expected life time.

At may 2011 11 Myriade microsatellites have been launched cumulating more than 35 years of in orbit life ; 8 additional satellites are in development or ready to be launched.

A new generation of Myriade microsatellite called "Myriade Evolutions" is under study; with the target of a spacecraft up to 250 Kg and 250 W with a lifetime of 5 years.

First Myriade Evolutions satellite will be MERLIN, a mission made in cooperation with DLR dedicated to CH_4 monitoring in atmosphere.

Microscope functional chain

Considering the payload characteristics and the specific I/F requirements it was early realized that the Myriade generic platform taken as a whole could not be compatible with Microscope mission.

Nevertheless, the choice was made to take benefit of Myriade development in term of existing equipment units, structural concept, Ground Supporting Equipments and integration and validation principle and, in order to minimize modifications, development risks and additional costs.

MICROSCOPE adopts the same reference frame of Myriade product line.

The existing Myriade structure is not compatible with Payload volume and accommodation constraints so Microscope structure is a specific development based on the same principle and material of the generic bus but with a different volume and equipment accommodation.

Power functional chain is almost the same than the Myriade one; only SG as been modified because the existing geometry (a single wing of two panels) is not compatible with the centering and inertia constraints.

New SG is composed by two wings of one panel each, it will use the same UTJ AsGa cells of Myriade. The total surface of SG is 0.9 m2 and the total EOL power is 200 W.

All other equipment units of power functional chain are the same than Myriade product line.

In MICROSCOPE, the Attitude and Acceleration Control System (AACS) replaces the Myriade generic Attitude and Orbit Control System (AOCS) because the the real-time acceleration control of satellite is performed instead of the orbit control. However the two systems are very similar :

AACS architecture is based on five modes (Figure 6) :

- MLT is the launch mode, all the equipments are off.
- MAS is the acquisition / safe mode ; it corresponds to the Myriade product line standard acquisition / safe mode.
- MGT2 is the coarse transition mode, it corresponds to the Myriade product line standard transition mode except for the guidance orders which impose a conical attitude with a pro-grade spin rate at $2 \cdot f_{or}$ around the instrument spin axis.

- MSP is the fine transition mode, it is similar to Myriade product line standard normal mode except for the use of CGPS as actuator.
- MCA correspond to the main mission mode allowing the EP measurement, it is a Microscope specific new mode composed by several sub-modes allowing the transition between each mission session (calibration, inertial pointing and rotating session).

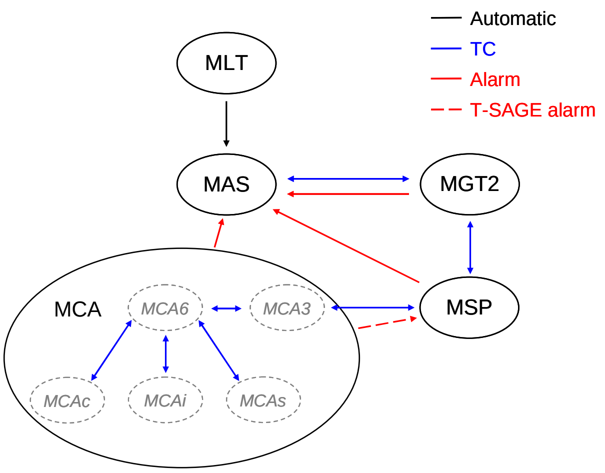

Figure 6 : AACS modes

For MAS and MGT2 modes, Microscope AACS uses the same equipment units than Myriade product line i.e. SAS and magnetometer as sensors and magnetorquers and reaction wheel as actuator (only 1 reaction wheel on X axis is present). No gyro or hydrazine propulsion is present.

For MSP mode, AACS uses the SST as sensor and CGPS as actuator.

For MCA mode, AACS uses the SST and T-SAGE as sensor and CGPS as actuator.

Communication functional chain is the S-band communication of Myriade product line working at 625 kbits/s. No X-band is present, payload data are downlinked through the S-band.

Data Management chain uses the existing Myriade product line On Board Computer.

Microscope flight software is adapted from the existing Myriade product line software. Main modifications concern AACS software for MSP and MCA mode. Drivers low level software, communication control software and AACS software for MAS and MGT2 modes have not been changed. The recurrence rate with existing software should be around 60-70 %.

FDIR logic is slightly different from standard Myriade one; during MCA mode (EP measurement mode) the MSP mode is used as an intermediate in case of a minor platform alarm of an anomaly due to the instrument (communication breakdown for example).

Withdrawal to MSP mode instead to MAS mode avoids to switching off the payload without generating any risk for the satellite (T-SAGE is not used as sensor in MSP mode) ; the switching off and on of the payload may change the errors of the instrument and force to perform a new calibration and finally replay all the sessions associated to this part of the mission.

Specific new subsystems of Microscope are :

- the GNSS subsystem, based on two GPS antenna and a new equipment, a low cost software Galileo/GPS receiver under development of TES. It is a technological passenger which will be used for on ground orbit determination in addition to Doppler ranging.
- the IDEAS desorbitation subsystem (Figure 7). In order to respect the time for re-entry into the atmosphere specified by the French space law, Microscope adopts a passive deorbitation subsystem based on two sails deployed at the end of the mission by two Gossamer arms. In this way the Surface/Mass ratio is increased and the altitude of the orbit is naturally reduced by the atmospheric drag. The inflating of the arms is ensured by gaseous Nitrogen stored at high pressure in adedicated titanium vessel. Passive solution has been preferred to active (solid propulsion) because of its low development cost and its adaptability to Microscope existing design.

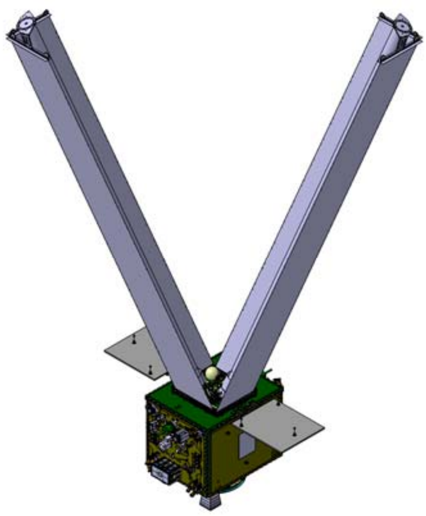

Figure 7 : Microscope end of life configuration (IDEAS deployed)

Mechanical and thermal architecture

MICROSCOPE thermo-mechanical architecture is mainly driven by SU accommodation constraints, satellite MCI budget goals and T-SAGE thermal stability requirement.

SU are accommodated in the platform near the center of gravity of satellite (the center of gravity shall be located between the two SU less than 3 mm from the spin axis).

The structure supporting the SU and the FEEU is called payload assembly subsystem (PAS), it includes also the thermal control hardware. PAS (Figure 8) shall :

- allow the centering of proof masses with respect to the satellite spin axis.
- offer a very high mechanical stability during all the mission.
- offer a thermal stability to SU better than 2 mK at f_{ep} (peak to peak value).
- offer a thermal stability to FEEU better than 20 mK at f_{ep} (peak to peak value)
- protect the SU from magnetic perturbations.
- guarantee the thermal control of FEEU ([10:45]°C operating range and taking into account their 6 W power dissipation each)

PAS is mounted on the –X panel (anti-solar panel) which offers a high thermal stability, it represents a thermal cavity insulated from the rest of satellite with its autonomous thermal control.

PAS (Figure 8) is composed by a two stage structure :

- fist stage supports the FEEU and its radiator, it is mechanically linked and thermally insulated from –X panel by six titanium alloy blades.
- second stage supports the SU and the magnetic shielding, it is mechanically linked and thermally insulated from the first stage by six titanium alloy blades.

Each stage of the PAS is individually covered by MLI (except the FEEU radiator) and conductively decoupled from the rest of the satellite; FEEU is protected from external IR Earth fluxes by a specific thermal baffle.

The two stage structure gives a progressive insulation to PAS and simplifies the thermal stabilization of T-SAGE.

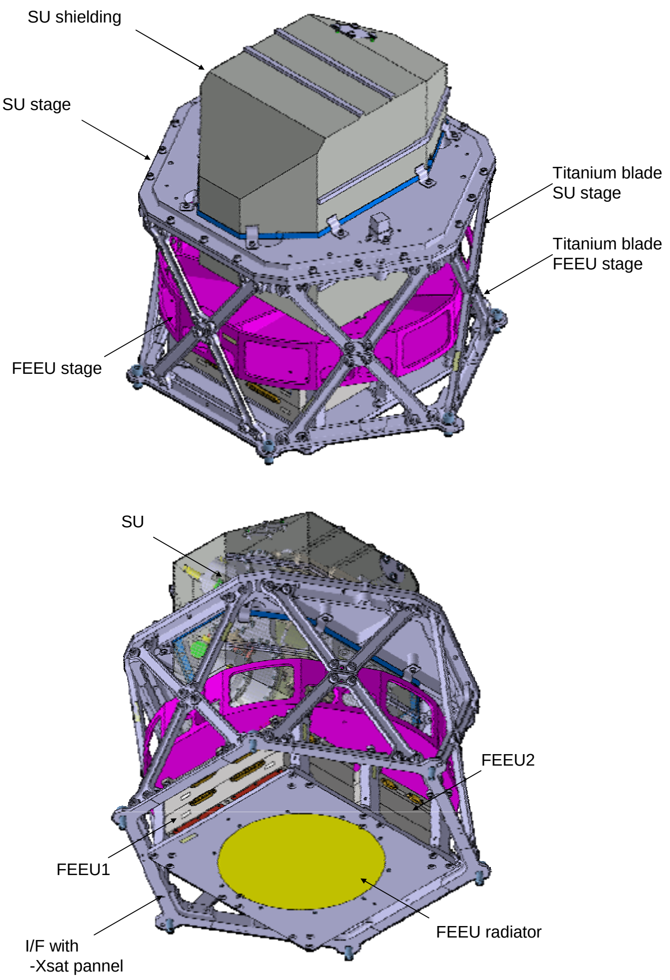

Figure 8 : PAS design

MICROSCOPE structural concept is directly derived from the Myriade product line platform. The structure is composed of six rectangular sandwich panels made of aluminium skin with honeycomb aluminium core.

Lateral panels are assembled by four L-spar support structures, and can be opened independently during integration; Z panels are dedicated to Cold Gas Propulsion System and Y panels accommodate the rest of the equipments except Star Tracker Optical Heads which are located on –X wall as close as possible of the PAS in order to allow a good natural alignment stability of Star Tracker measurement axes with respect to the instrument spin axis.

External layout (Figure 9) is mainly constrained by antennas accommodation and AACS equipment I/F requirement in order to minimize the modifications with respect to AOCS standard non-mission modes.

For centering and symmetry reasons, Solar Generator is separated in two identical wings of one panel each, mounted on Y panels and directed toward +X. During launch wings are folded; the release is guarantee by 3 pyrolock mechanisms and the deployment by 2 Carpenter blades, these component are already used on Myriade product line standard Solar Generator. Because of the good energy budget of Dawn/Dusk orbit, Solar Array Driving mechanisms are not necessary.

Y walls accommodate also the platform radiators.

Cold Gas Propulsion System thrusters are mounted on the Z walls, the location and the thrust direction are optimized in order to maximize the control capacity of AACS.

+X panel accommodates the desorbitation subsystem and – X panel the launcher I/F cylinder which integrates the launcher I/F adapter, protects the FEEU radiator thermal baffle and avoids the overtaking of launcher separation plane by any part of the satellite.

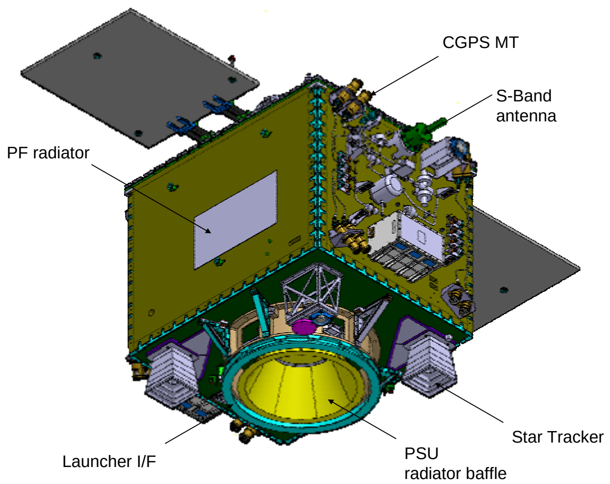

Figure 9 : Microscope external Layout

Equipment internal accommodation (Figure 10) has been optimized in order to balance the mass and platform radiator size (by balancing dissipated power) between –Y and +Y panels. This layout simplifies the centering and minimizes the thermal perturbation coming from the platform to the PAS, because the external IR Earth fluxes absorbed by radiators at f_{ep} are equilibrated in modulus and in phase opposition.

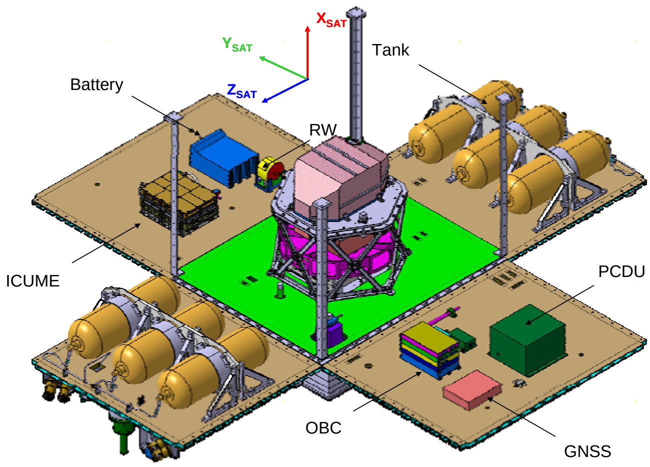

Figure 10 : Microscope internal Layout

Microscope satellite is designed to be compatible with the following launcher vehicles (in alphabetic order): DNEPR, Falcon, PSLV in DLA configuration; Rockot, Soyuz in ASAP-S configuration and VEGA in VESPA configuration.

Acceleration and Attitude Control System

The Acceleration Control Mode (MCA) is the nominal mode in which the mission is completed. The mode is completely new in terms of objectives and equipment.

MCA mode uses T-SAGE and Star Tracker as sensor and CGPS as actuator. The payload measurements provide linear and angular accelerations whereas the star tracker provides the angular positions.

The MCA mode allows to control the estimated accelerations and attitude in a specific point of the satellite called "Drag free point".

The drag free point is user-defined and it could be changed from one session to an another (normally, it is chosen at the reference point of the proof masses of one SU).

The controlled accelerations are directly provided by T-SAGE to AACS as the result of the combination of the accelerations measured by the proof masses corrected by the scale factor and the choice of drag free point.

Estimate attitude is the result of the hybridization between the attitude measurements provided by the Star Tracker and the angular accelerations measured by the SU.

Acceleration and attitude control laws define the total forces F_{com} and torques T_{com} to be applied on the satellite to compensate external perturbation; Thrusters selection logic transforms the commanded forces and torques into 8 thrust orders, one for each CGPS thruster, minimizing for each setting point (F_{com}, T_{com}) the total gas consumption.

The MCA control loop is shown in Figure 11.

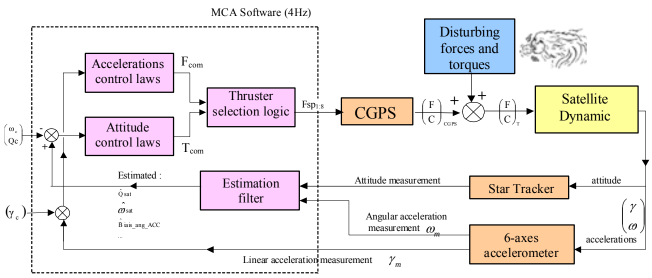

Figure 11 : MCA control loop

MCA mode is composed by several submodes which differ only by working points and the values of the controllers, while the equipment units used and the control loops remain the same.

Main AACS requirements are :

- Linear acceleration control shall be less than 10^{-12} m/s² at f_{ep}.
- Angular velocity stability shall be less than 10^{-9} rad/s for rotating EP measurement sessions.
- Angular acceleration shall be less than 10^{-11} rad/s² for all the EP measurement sessions (inertial or rotating).

For inertial session, angular acceleration requirement becomes a requirement on attitude stability of 8.95 µrad at f_{ep}.

For rotating sessions angular velocity stability requirements become a requirement on attitude stability of 0.17 µrad at f_{ep}.

The attitude and angular acceleration estimation is an important issue in the AACS performance achievement because of the very high stability required at f_{ep} frequency.

The expected performances of the Myriade product line star tracker are 11 µrad at f_{ep} (internal error) and the bias due to the thermoelastic deformations between Star Tracker line of sight and SU reference direction is estimated at 11 µrad at f_{ep}.

The attitude measurement provided by the Star Tracker could not be directly used by the SCAA but it shall be mixed with the attitude issued from the accelerometers angular accelerations (double integration) in order to obtain an attitude estimation compliant to the angular accuracy at f_{ep}.

The required gains of the hybridization filter for the SST measurement are of -15 dB at f_{ep} for inertial sessions and -50 dB at f_{ep} for rotating sessions.

For inertial session, a classical Kalman filter with a high pass filter for T-Sage attitude and a low pass filter for Star Tracker measurements is used (Figure 12).

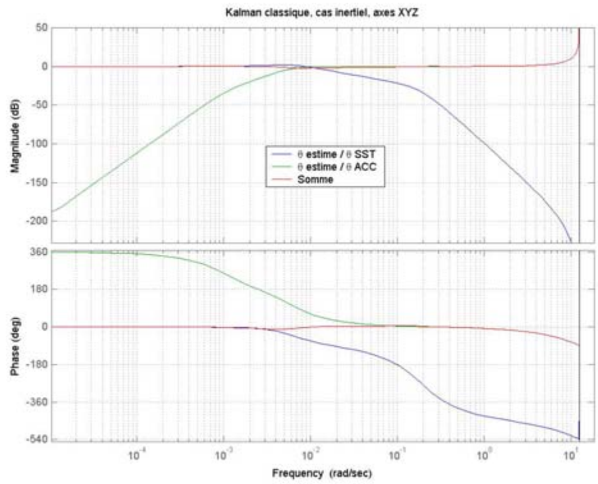

Figure 12 : Hybridization filter for inertial session

For rotating sessions, the high rejection required at f_{ep} for Star Tracker measurements prevents to use the classical Kalman filter and drives to a specific hybridization filter called "Sinus-Cut" with a deep tight local attenuation around f_{ep}.(Figure 13).

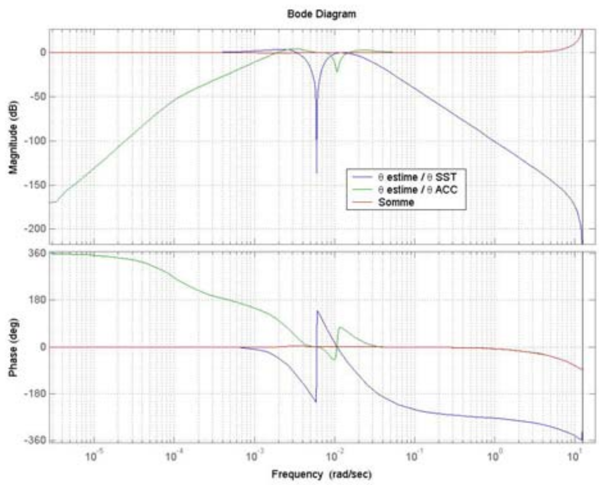

Figure 13 : Hybridization filter for rotating session

Transfer function between the hybrid attitude θ_{HYB} and star tracker attitude θ_{SST} is shown in Figure 14.

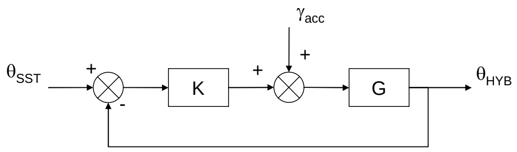

Figure 14: Transfer function of rotating sessions hybridization filter

Order of K and G filters is set differently axis by axis :

- For X axis we used a 2nd order filter for G and a 6th order filter for K.
- For Y and Z axes we used a 4th order filter for G and a 24th order filter for K.

At the end, "Sinus cut" hybridization filter is similar to a Kalman filter with gains made by transfer functions.

AACS controllers shall be designed to ensure sufficient rejection of sinusoidal perturbation (force and torque) at f_{ep}.

Maximum external perturbation (forces and torques input of controllers) has been estimated to 25 µN and 50 µN.m (aerodynamic drag and magnetic torque).

In order to respect the requirement of linear and angular acceleration for a 250 kg and 50 Kg*m² spacecraft, the gain of controller shall be lower than -100 dB at f_{ep} for linear acceleration controller and lower than -75 dB at f_{ep} for angular acceleration controller.

Figure 15 shows the linear acceleration Bode diagrams (green plots correspond to inertial session controller and blue plots to rotating session controller), gain is -103 dB the delay margin greater than 2 s.

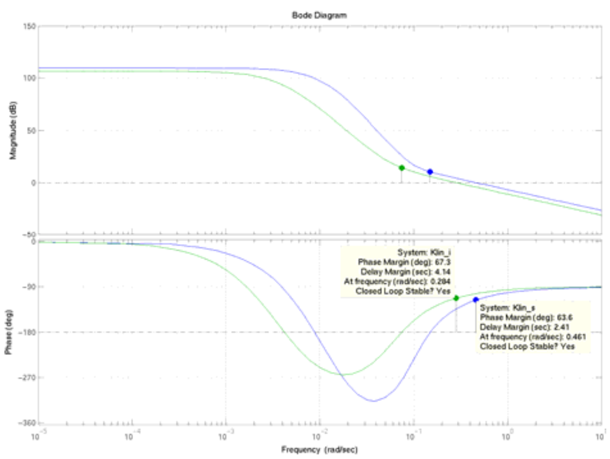

Figure 15 : Linear controller Bode diagram

AACS definition induces several requirements on its sensor and actuator i.e. on T-SAGE, Star Tracker and CPGS.

Concerning T-SAGE the angular performance requirements are more severe than the scientific needs because of its use in the hybridization filter.

The generic Myriade Star Tracker performance specification has been complemented with Microscope's specific requirements (harmonics errors at f_{ep}).

CGPS definition is highly dependent from AACS; thrust range, thrust resolution, thruster time response and thrust noise level requirements are specific to Microscope mission and drive to develop a new propulsion system based on cold gas.

Cold Gas Propulsion System

Cold Gas Propulsion Subsystem (CGPS) is composed by two identical and independent sub-systems called CGPSS which are accommodated on –Z and +Z panels.

Each CGPSS (Figure 16) is composed by 4 modules :

- Gas Distribution Module (GDM) stores and maintains the gas at its operational range (pressure and temperatures).
- Pressure Regulation Module (PRM), provides the gas distribution to the thrusters, it contains all the equipment units necessary to ensure the pressure regulation of the CGPS.
- Thrust Regulation Module (TRM) contains 4 nominal and 4 redundant Micro-Thrusters (MT).
- Electronics Control Module (ECM) contains the electronics items necessary to provide the power supply to all the CGPSS modules, control the TRM thrust, and ensure the avionic I/F with the PF OBC.

Each GDM is composed by 3 Arde D5048 Carbon Overlapped Pressurised Vessels filled with 8.25 kg of gaseous Nitrogen stored at the maximum pressure of 345 bars.

PRM is composed by existing off the shelf equipment, only the Pressure Regulator needs to be delta qualified to withstand 345 bars maximum pressure.

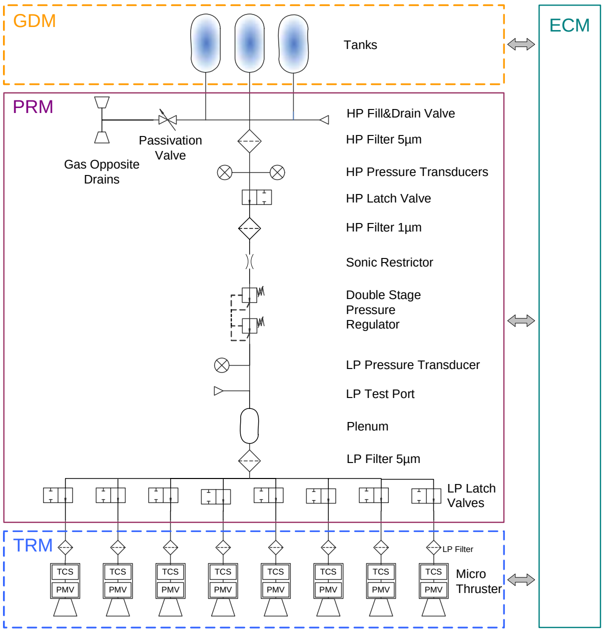

Figure 16 : CGPPS overview

TRM and ECM are provided by ESA.

Candidate MT are the Cold Gas micro-thrusters developed by Thales Alenia Space Italy in the frame of GAIA project (launch date 2013), their qualification has been achieved in 2011. This MTs operate in close loop using a miniaturized Mass Flow Sensor as thrust measurement and piezoelectrics actuator to modify the nozzle section and modulate the gas flow.

Candidate ECM is a new Microscope development based on the existing equivalent electronic module of GAIA ; MT driver boards and DC/DC boards are the same than GAIA, CPU card and I/O card have been designed specifically for Microscope.

In order to improve the time responses a specific new algorithm including an anti-hysteric controller has been study by TAS-I, it should be implemented in Microscope ECM.

However two other technologies developed in Europe are considered for MT and its control electronic : a Bradford concept controlled through a pressure sensor, and the MEMS technology developed by Nanospace. Their performances (Isp, noise, thrust range and resolution, etc..) should be compliant to Microscope main requirements and an ITT has been issued by ESA to select one of them.

Due to accommodation constraint and the short time duration of the mission (1 year), redundancy have been limited only to non flight proven equipment units (i.e. the MTs).

CGPS shall be operated in continuous way, because the noise generated by pulsed thrusters is not compliant with noise specification and would saturate T-SAGE range.

Main CGPS requirement are :

- Thrust range between [0:300] µN.
- Thrust resolution less than 0.2 µN.
- Thrust axial noise less than 3.22 µN rms in [0.001:10] Hz bandwidth.
- Thrust linearity less than 5%.
- Time response of 250 ms at 1σ in total thrust performance range.

SATELLITE PERFORMANCES AND BUDGET

Microscope is bigger and heavier than Myriade product line previous satellite.

The satellite dimensions with SG in folded configuration are : 1.380 m × 1.040 m × 1.580 m (X × Y × Z).

Maximum mass including N_2 is around 290 Kg.

Center of Gravity location is compliant with centring requirements but specific balancing masses shall be added.

Inertia values are slightly non compliant with respect to requirement; but the acceptability of these value has been verified by AACS analysis.

The satellite first lateral and longitudinal modes in launch configuration are respectively 26 Hz and 68 Hz, first frequency in free free configuration with Solar Generator deployed is 5.5 Hz.

In addition to the usual mechanical analyses, thermo-elastic analysis of the structure includes the study of the relative rotations between the scientific accelerometers and the star trackers interfaces at f_{ep} and at $3f_{ep}$ frequencies. The stability of the relative rotation in inertial pointing is compliant with the requirement (11 µrad at f_{ep} for 12µrad specified).

Mechanical model was also used for the satellite self-gravity analyses. These analyses include the computation of the spacecraft self-gravity and the variations of the gradient of gravity generated by thermal deformations, at f_{ep} frequency. Two methods were used for the mass discretization, one replaced the NASTRAN elements by concentrated masses at their centre, and the other used the NASTRAN mass matrix and the grids locations. Both methods showed equivalent results : the gravity gradient generated by the satellite at proof-masses centers is well within the required value (10^{-11} s^{-2} at f_{ep}).

Payload thermal needs gave birth to 26 specific thermal requirements which have been verified by thermal analysis. All the satellite values are compliant, the minimum margin has been reached for thermal stability at f_{ep} of FEEU in inertial mode with a result of 17.8 mK for a specification of 20 mK (peak to peak).

AACS specifications have been translated in 46 requirements. AACS performances have been verified using a numerical simulator integrating the behaviour of T-SAGE. Some non compliances appeared especially for angular acceleration and attitude stability during inertial sessions; their acceptability has been verified at mission performances level.

Gas consumption has been estimated by simulations taking into account :

- the mission profile (sessions sequence including transition between sessions)
- Variation of the external forces and torque following satellite orbital position and attitude.
- AACS control loop.
- Measured Isp vs Thrust profile of the Cold Gas Thruster (nano-balance measurement).
- Mass centering and inertia evolution following gas consumption.

Several simulations have been performed to estimate influence on gas consumption of some parameter (Isp profile, satellite magnetic moment, solar activity, etc…).

Typical gas budget (parameters equivalent to 1 σ value) gives a gas consumption of 37 % of expendable gas.

Worst case gas budget (addition of gas consumption difference issued from sensitivity analysis) gives a gas consumption of 96 % of expendable gas.

Maximum power budget is around 135 W in mission mode and 140 W in coarse transition mode, the minimum margins are 35 W for mission mode and 10 W for servitude mode in eclipse season.

Maximum Depth of Discharge (DoD) of the battery is reached during safe mode transition in eclipse season, the value is compliant with Myrade product line battery performances.

Due to specific new AACS algorithms, On Board Software behaviour have been evaluated running a prototype of flight software including AACS algorithms on an Engineering Model of the On Board Computer. Running time and CPU load are compliant with the targets (minimum margin of 35%).

Desorbitation time has been computed according to the French Space Law, the mean reentry time is :

- 71 years without IDEAS
- 28 years with IDEAS

Accuracy of EP measurement is estimated by the following equation :

$$EP_{accuracy} = \frac{\sqrt{D^2 + \frac{N^2}{T}}}{g(h)} \qquad (6)$$

Where D is the errors budget at f_{ep} of deterministic contributor expressed in m/s², N is the errors budget around f_{ep} of random contributors expressed in m/s²/√Hz, T is the length of the measurement session expressed in s and $g(h)$ is the gravity acceleration at the altitude h.

Error budget is built taking into account all the contributors (instruments, AACS, magnetism, local gravity, orbit, dating and thermal control).

Expected performances are :

$EP_{accuracy} = 0.81\times10^{-15}$ at 1 σ for inertial sessions

$EP_{accuracy} = 0.56\times10^{-15}$ at 1 σ for rotating sessions

For inertial sessions, major deterministic contributor is thermal control and major random contributor is T-SAGE itself; for rotating sessions, contribution to deterministic error are fairly shared and major random contributor is still T-SAGE.

DEVELOPPEMENT STATUS AND SCHEDULE

The decision of MICROSCOPE project has been made by CNES in 2004. Microscope Preliminary Design Review has been successfully held in March 2011.

The long time spent between these dates is justified by a major change in propulsion system (initial propulsion system using field effect electric thrusters has been replaced by cold gas thrusters) and the consequently rebuilt of the satellite.

During this time, the development of payload kept on through several Engineering Models. The qualification model of SU has been manufactured (Figure 17); qualification campaign started in December 2010 and it is scheduled to end before the end of 2011.

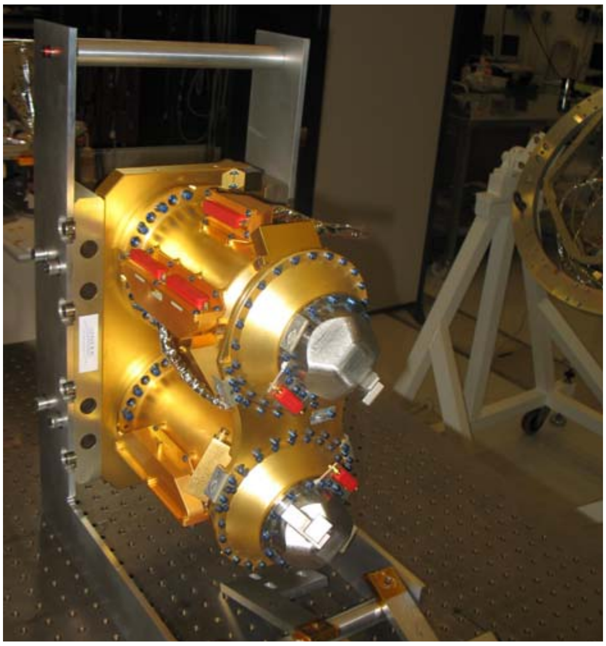

Figure 17 : SU qualification model

Results of first free falls done in ZARM free fall tower in Bremen (Germany) are compliant to predictions (Figure 18).

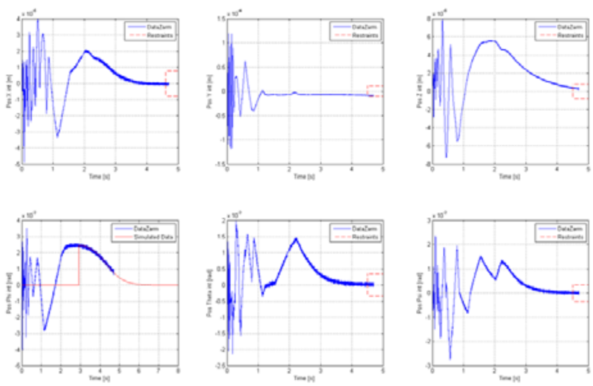

Figure 18 : SU free fall results

Manufacturing of flight models of FEEU and ICUME is running.

Due to high influence of thermal performances on the mission accuracy, a thermal qualification model of PSU (Figure 19) has been built and tested in 2009 with the SU Qualification Model and FEEU Engineering Models. Results of thermal balance test have been used to update thermal model and refine thermal predictions.

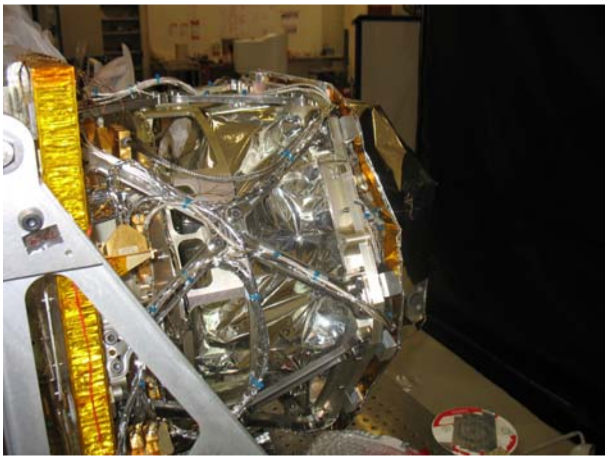

Figure 19 : PSU thermal qualification model

Major performances of the CGPS have been assessed by GAIA project; additional requirement will be verified in C phase (new thruster control loop algorithm).

CNES decision for C/D phases of the program is expected in October 2011, for a launch scheduled in 2nd semester of 2015.

CNES leads the development and will integrate the satellite with industrial support, ONERA builds and tests the instrument.

CONCLUSION

MICROSCOPE represents a challenging mission for several reasons :

The accuracy of the mission is unusual and obliges to rethink completely our way to work, many phenomena, like MLI clanks or thermo-elastic induced gravity gradient, which could be neglected in the frame of more classical missions could became a showstopper if not taken carefully into account from the beginning of satellite design.

The order of magnitude of many parameters is often far from the capability of an experimental characterization on ground and it could be assessed only by simulation.

As the satellite cannot be end to end tested on ground before launch, the operation plan shall be very flexible to identify and tune any discrepancy with respect to expected behaviour.

The platform and the payload strongly interact, and their performances are strictly dependent; the success of the mission shall be guaranteed only by a cooperative iterative work between CNES satellite team and ONERA payload team.

The frame of the project does not allow many changes with respect to Myriade family product line, new developments have been reduced to strictly necessary and reutilization of existing equipment preferred.

The satellite presented in this paper constitutes an excellent compromise between well proven robust technologies and narrowly specialized developments.

Microscope will constitute the first Equivalent Principle test made in space. Whatever the results on EP violation or not it will help scientific community to go forward in the understanding of fundamental physic laws.

Acknowledgments

Authors would like to thank all the persons working in the MICROSCOPE teams at CNES, ONERA, OCA and ESA.

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