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"Microscope : A scientific Microsatellite development"

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ABSTRACT

MICROSCOPE is a CNES-ESA-ONERA-CNRS-OCA-DLT-ZARM scientific mission developed in the frame of Myriade Microsatellite family. The scientific objective consists to test the Equivalence Principle between gravitational mass and inertial mass with an relative accuracy of 10^{-15} ; i.e. one hundred times better than the one obtained today on Earth .

Satellite has been launched the 25th of April 2016 for a 2 years in orbit lifetime.

This paper begins with a introduction of the scientific goals, a presentation of the mission and the payload definition, explaining how the S/C has evolved in time in order to fulfil the stringent mission requirement always keeping in line with microsatellite development approach. The main part of the paper is focused on the description of actual spacecraft design with a presentation of all the functional chains, their performances and the ground validation process. Most innovative elements are the AACS (Attitude and Acceleration Control System) running simultaneously 38 control loops in order to keep the payload in drag-free condition during scientific sessions and (CGPS) Cold Gas Propulsion System generating and modulating a continuous thrust in the range from 1 to 300 µN with an accuracy of 0.1 µN. A special attention is given to micro perturbation control plan and satellite validation logic due to high sensitivity of PL and the impossibility to perform full representative test on ground.

It is shown how the design of the satellite is optimized, melting new advanced technology and low cost, well proven methods coming from Myriade family. The paper will end with a presentation of first in-flight results especially the commissioning phase.

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 ONERA and CERGA Institutes, satellite has been launched the 25th of April 2016 by Soyouz with VS14 flight.

This mission has been 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 term of proportionality of gravitational 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 :

$$\frac{m_{1g}}{m_{1i}} = \frac{m_{2g}}{m_{2i}} \qquad (3)$$

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.

Mission description

The principle of the mission 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 its trajectories will be identical for the same initial condition (position and velocity); If EP is violated its trajectories will be different for the same initial condition.

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 :

$$\Gamma_d \approx M_c \cdot \left(\delta \cdot g + (Tg - Ti) \cdot \Delta - 2 \cdot Tc \cdot \dot{\Delta} - \ddot{\Delta} \right) + B_d + M_d \cdot \Gamma_c + \Gamma_{d2} + \Gamma_n \qquad (5)$$

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 :

- $f_{sp1} = (7/2) \cdot f_{or}$
- $f_{sp2} = (9/2) \cdot f_{or}$

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 2 years months including Platform in orbit commissioning.

During eclipse season (3 months 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 will be performed only during the period of the year without eclipse.

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).

Sun Synchronous Dawn/Dusk Orbits (Figure 1) have been chosen based on power budget and thermal stability.

Orbital parameter of Microscope are :

- Altitude : 707 km
- Local Time Ascending Node : 18h00
- Excentricity : 0.005

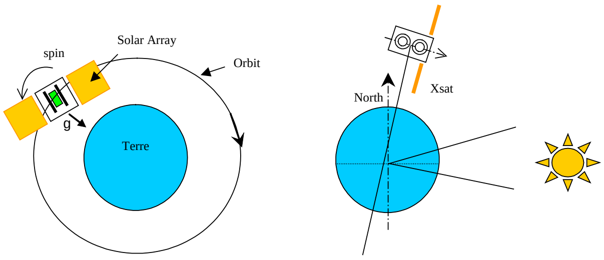

Figure 1: Microscope satellite orbit

Other mission requirements are induced by the performances of payload (scale factor and defaults).

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

- Knowledge of the accelerometer position with respect to the Earth center less than 7 m at f_{ep}.
- Knowledge of the spacecraft attitude less than 1 mrad at f_{ep}.,
- 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}.

SATELLITE DESIGN

Design constraint

Microscope satellite design is driven mainly by mission and payload requirement and by constraints issued from Myriade platform.

Mission and payload impose to satellite:

- to control the accelerations of the satellite around all six degree of freedom
- to use payload measurement in AOCS control loop.
- to respect instrument I/F requirements

Myriade constraints impose to the satellite :

- the compatibility with Arianespace auxiliary passenger opportunity and I/F (i.e ASAP-S for Soyouz ans VESPA for VEGA).
- to reuse all the Myriade functional chains not directed involved in performance
- to maintain Myriade validation process and tools

Six axis control needs the generation of long continuous very small thrust (few µN) with an high accuracy, low noise and short response time.

These characteristics are not achievable with classic Myriade chemical propulsion system and a new developpement was necessary.

Payload sensitiveness highly constrains the design of the satellite :

- The center of Gravity of the satellite shall be very close (less than 3 mm) with respect to spin axis of satellite which is aligned with the centers of gravity of proof masses.
- No moving mass is allowed inside the satellite; that means that any liquid is forbidden.
- Thermal environment of instrument shall be very stable :

  Thermal stability of proof shall be less than 2 mK at f_{ep} (peak to peak).
- Thermal stability of Front end electronics units (FEEU) shall be less than 20 mK at f_{ep} (peak to peak).
- 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).
- All kind of external perturbation (magnetic, gravitational, etc) shall be identified and minimized.

SATELLITE DESCRIPTION

Satellite mode

Figure 2 shows the satellite mode and the transition between them

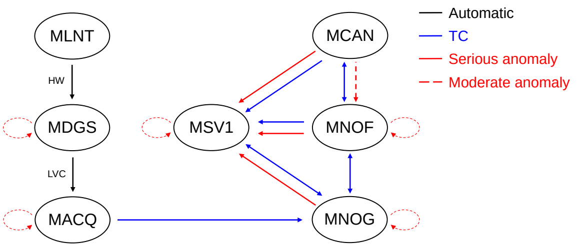

Figure 2: Microscope satellite mode

MLNT is « launch mode » : it correspond to the configuration from the mating with the launch vehicle until spacthe ecraft separation; satellite is OFF and only the circuit of separation detection is powered.

MDGS is "Solar array deployment mode" : transition from MLNT is automatically performed by an hardware mechanism, during this mode satellite is switched ON and the solar array is deployed after a countdown. This mode has been introduced in order to avoid an anticipated deployment in case of false separation detection during the launch. Duration of countdown is mission dependent.

MACQ is "Sun acquisition mode" : after solar array deployment the satellite direct its X axis toward the sun in order to maximize the available power, once the sun direction acquired , satellite is put in slow spin around X axis in order to homogenized temperatures.

MNOG is "Rough pointing mode" : this mode is a transition mode from safe mode to mission mode : used for commissioning all the satellite equipment including the payload and cold gas propulsion system. This mode is also used as safe mode for long and deterministic mission interruption as in eclipse season.

MNOF is "fine pointing mode": the attitude of satellite is fined controlled using micro-propulsion system and star tracker. This mode is also used to perform collision avoiding operation.

MCAN is the "mission mode": payload data are used by AOCS in addition to micro-propulsion and star tracker, to control the satellite around its 6 dof ; all the scientific sessions are performed in this mode modifying guidance and AOCS control law parameters according to session characteristics.

Table 1 resume satellite mode and their communality whit respect to Myriade family.

Table 1: Satellite mode

Sat Mode	AOCS Mode	Myriade Legacy
MLNT	MLT	Yes
MDGS	MLT	Yes
MACQ	MAS	Yes
MNOG	MGT3	Yes
MNOF	MSP	Partial
MCAN	MCA	None
MSV1	MAS	Yes

Payload description

Payload is composed by a double differential high precision accelerometer called T-SAGE (Twin Space Accelerometer for Gravity Experiment); one SAGE accelerometer (SAGE-EP) is used to EP measurement the other one SAGE-REF is used as a reference measurement.

Each SAGE (Figure 3) is composed by a Sensor Unit (SU), a Front End Electronic Unit (FEEU) and an I/F Control Unit (ICU).

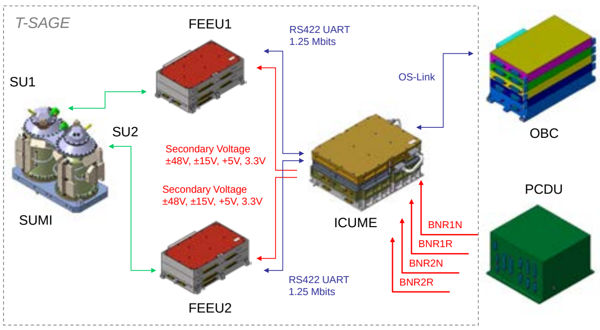

Figure 3: T-SAGE design

SU (Figure 4) 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.

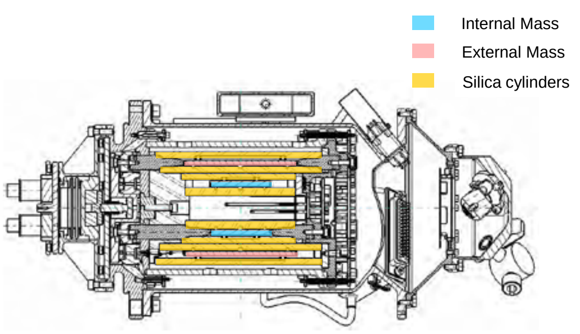

Figure 4: SU overview

The different materials chosen for the SU-EP are PtRh10 platinum alloy for the inner proof-mass and TA6V titanium alloy for the outer one; SU-REF have both masses made in PtRh10 platinum alloy.

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

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.

A locking device supports the masses during launch phase; when released the masses are free to fall.

The position of each mass around its 6 degrees of freedom is finely measured using capacitive detectors, this information is used by payload control loop to maintain the mass motionless applying electrostatic forces and torques. In this way, for each SU, the center of gravity of booth masses is maintained strictly in the same position and masses are submitted to the same gravitational force.

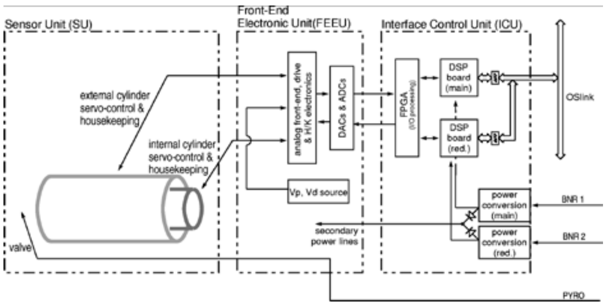

Figure 5: SAGE electrical design

T-SAGE has been developed by ONERA DMPH Châtillon (France)

Mechanical and thermal architecture

Satellite is built around Payload module (BCU)

BCU accommodate two SU and two FEEU in a two stage architecture :

- First stage accommodates booth FEEU and its radiator it is fixed to P/F structure by titanium alloy blades which guarantee thermal insulation
  Second stage accommodates booth SU and its magnetic shielding; it is fixed to 1st stage by titanium blades which improve the thermal insulation from the rest of the spacecraft. Dumpers have been introduced inside this

stage in order to reduce mechanical vibration level at SU I/F.

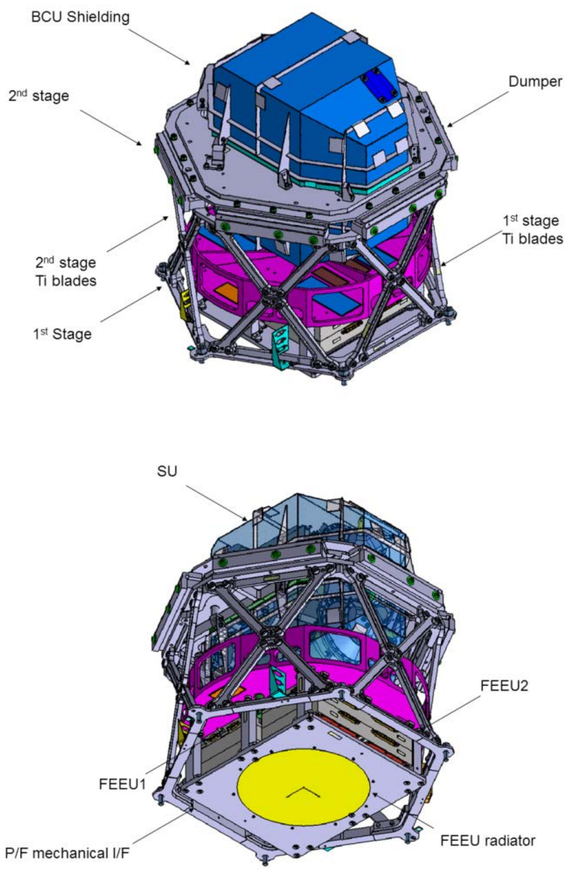

Figure 6: BCU design

Each stage of the BCU is individually covered by MLI (except the FEEU radiator) and conductively decoupled from the rest of the satellite; FEEU radiator 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 the instrument.

BCU is accommodated in the center of satellite in order to respect the constraint of proximity between SU and the center of gravity of satellite.

BCU 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.

MICROSCOPE structural concept is directly derived from the standard Myriade 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 integrated independently; Z panels are dedicated to Cold Gas Propulsion System and Y panels accommodate the rest of the equipment except Star Tracker Optical Heads which are located on –X wall as close as possible of the BCU in order to allow a good natural alignment stability of Star Tracker measurement axes with respect to the instrument spin axis.

External layout (Figure 7) is mainly constrained by fairing allocated volume, antennas accommodation and AACS equipment I/F requirement.

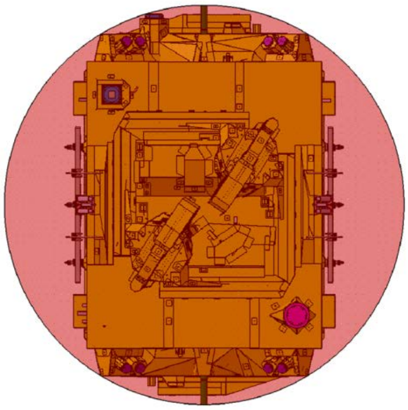

Figure 7: Microscope wrt fairing allocated volume

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 after the deployment.

Y walls accommodate also the platform radiators.

Cold Gas Propulsion System thrusters are mounted on the Z walls, the location of the thruster (MT) and their thrust directions have been optimized in order to maximize the control capacity of AACS.

+X panel accommodates the desorbitation subsystem IDEAS and – X panel the launcher I/F adapter.

Equipment internal accommodation (Figure 8) 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 fep are equilibrated in modulus and in phase opposition.

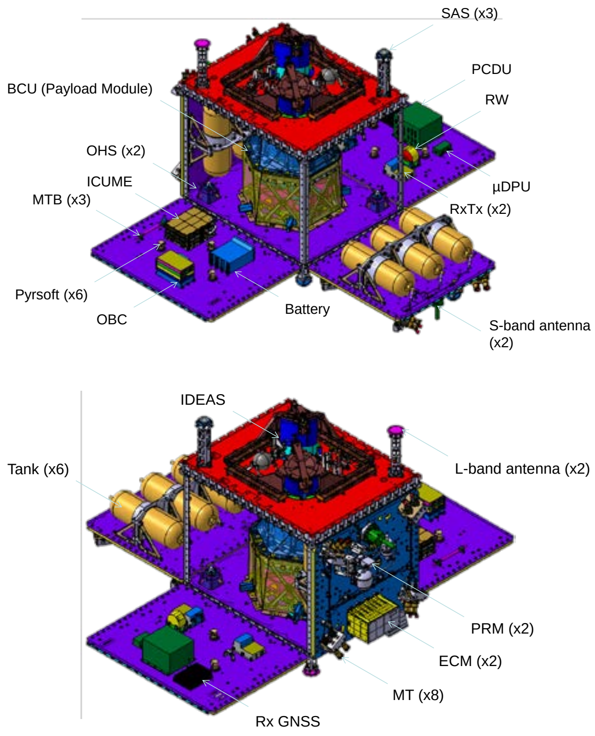

Figure 8: Equipment accommodation

Thermal control is based on a passive design in order to minimize heater power budget.

Active thermal control is used in non-mission modes to keep equipment in its operative or non-operative ranges.

During mission mode heater activation is forbidden in order to minimize electromagnetic perturbation during scientific session.

Power and avionic subsystem architecture

Power and avionic subsystem are the same of all the prievious Myriade Microsatellite.

Avionic subsystem is based on standard Myriade OBC; specific Microscope new equipment have been developed taking into account its specific I/F (UART).

Communication subsystem is based on standard Myriade S-band receiver/emitter equipment; receivers operate in hot redundancy, emitter in cold redundancy.

According to TM budget, neither memory mass storage or X-band emitter are necessary, scientific TM is transmitted by P/F S-band subsystem.

Power system is based on standard Myriade equipment.

Power Conditioning and Distribution Unit (PCDU) is in charge of launcher separation detection, battery regulation, power distribution to equipment and payload, magneto-torques commands and pyro lines distribution (up to 12 lines)

PCDU non Regulated Bus voltage between 22 and 37V and Solar Array maximum current is 8 A

Solar Array is composed by two wings of one panel each, it based on the same UTJ (Ultra Triple Junction) AsGa cells of Myriade. The total surface of SA is 0.84 m2 and the total EOL power is around 190 W

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.

Battery is the standard Myriade Li-Ion battery with a capacity of 13.5 Ah.

Cold Gas Propulsion System architecture

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 9) 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.
- MT module contains 4 nominal and 4 redundant Micro-Thrusters (MT).
- Electronics Control Module (ECM) provide power supply to all the CGPSS modules, control the TRM thrust, and ensure the avionic I/F with the PF OBC.

GDM and PRM are composed by existing off the shelf equipment;, MT of Microscope are the identical to Microthruster used in GAIA and Lisa-Pathfinder (LPF) mission; ECM are a new development merging existing GAIA boards (MT driver and monitor) with specific Microscope elements (Control and power).

The core of the system is the MT; each MT is composed by a miniaturized Mass Flow Sensor (MFS) which provide an immediate measurement of the flow of gas going through the thruster and by a Proportional Valve (PV) which allow to vary in continuous way the flow using piezoceramic actuator.

A control algorithm running on ECM at 50 Hz adjust for every MT PV opening according to the mass flow measured by MFS in order the obtain the commanded thrust; algorithm include a specific anti-hysteric controller in order to improve time responses with respect to GAIA/LPF performances.

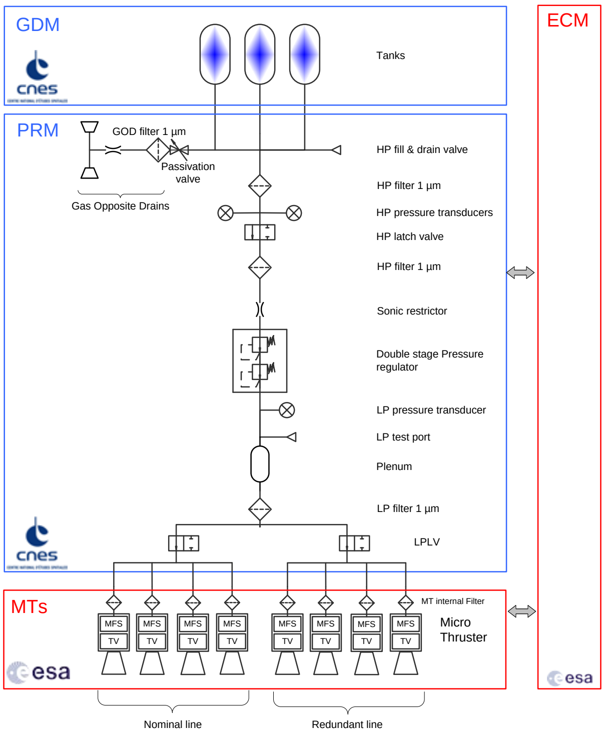

Figure 9: CGPSS overview

Main CGPSS characteristics are :

- MEOP : 345 bars
- Mass of gas : 8.25 Kg of N_2
- MT Thrust range between [0:500] µN.
- MT Thrust resolution 0.1 µN.
- MT Thrust axial noise less than 3.22 µN rms in [0.001:10] Hz bandwidth.
- MT Thrust non-linearity less than 5%.
- MT Time response of 250 ms (at 1σ).

TRM and ECM are provided by ESA.

Acceleration and Attitude Control System

Microscope mission does not need an orbit control, however a fine control of satellite acceleration is necessary to suppress all the non-gravitational forces acting on payload.

For this reason Attitude and Orbit Control System (AOCS) is replaced by Attitude and Acceleration Control System (AACS).

AOCS architecture turn on five mode which correspond to different mission phases; low level mode as MAS and MGT3 use standard Myriade equipement like sun sensor (SAS), Magnetometer (MAG), magnetotorquebar (MTB) and reaction wheel (RW); high level mode use a mix of Myriade equipment like star-tracker (STR) and new specific development (CGPS and T-SAGE).

Table 2 summarizes main AACS mode characteristics.

Table 2: AACS mode

AACS Mode	Mode Controle	AACS equipement
MLT	None	None
MAS	Sun coarse pointing Slow rate spin around X	SAS, MAG MTB, RW
MGT3	X normal to orbit with 10° of tilt S/C spinned around X at 3 time orbital period	MAG MTB, RW
MSP	3-axis attitude control	STR CGPS
MCA	6-axis control	STR, T-SAGE CGPS

The Acceleration Control Mode (MCA) is nominal mission mode.

MCA mode allows controlling the estimated accelerations and attitude in a specific point of the satellite called “Drag free point”; 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).

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

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 Fcom and torques Tcom 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 (Fcom, Tcom) the total gas consumption.

Scientific session (inertial, rotating, calibration, etc..) differs each other only by guidance, drag free working points and the values of MCA controllers.

The MCA control loop is shown in Figure 10.

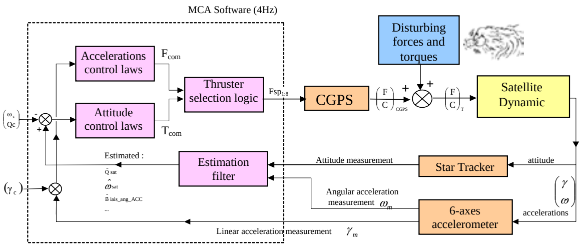

Figure 10: MCA control loop

At the end MCA mode involves 38 control loops working at 3 embedded levels :

- 1st level is made by T-SAGE control loop : 24 control loops (4 masses 6 dof each) working at 1027 Hz
- 2nd level is made by MCA control loop : 6 control loops (6 dof of S/L) working at 4 Hz
- 3th level is made by MT control loop : 8 control loops (one each MT) working at 50 Hz

Microscope specific development

Microscope take on board two specific subsystems :

- the GNSS subsystem, based on two GPS antenna and a new low cost software Galileo/GPS receiver. GNSS receiver is used for on ground orbit determination in addition to Doppler ranging.
- the IDEAS desorbitation subsystem (Figure 11). Microscope adopts a passive deorbitation subsystem based on two wings, composed each by a sail and a Gossamer arm, and a inflating system using gaseous Nitrogen stored at high pressure (290 bars) in a dedicated titanium vessel. At the end of mission the Surface/Mass ratio is increased deploying the sails (); as a result the atmospheric drag raise and the natural reduction of orbit's altitude is accelerated. After deployment inertia of satellite vessel. Passive solution has been preferred to active (solid propulsion) because of its low development cost and its adaptability to Microscope existing design.

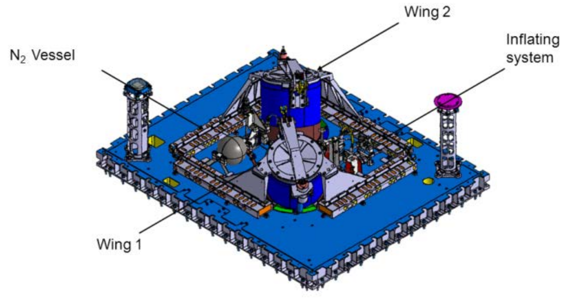

Figure 11: IDEAS design

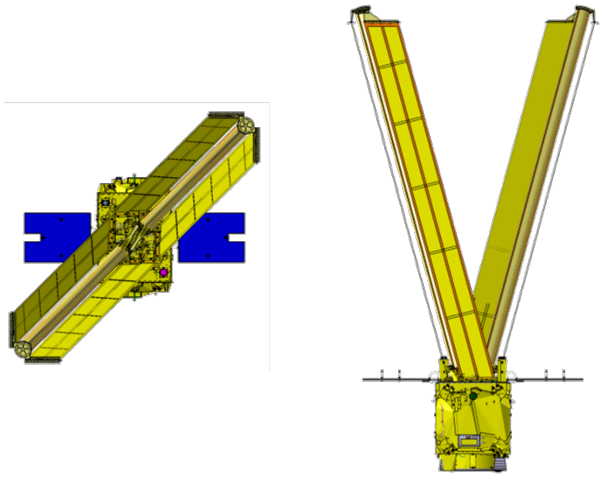

Figure 12: Microscope after IDEAS deployement

SATELLITE PERFORMANCES AND BUDGET

Satellite dimensions with SG in folded configuration are : 1.375 m × 1.050 m × 1.500 m (X × Y × Z).

Satellite mass at launch was 302 Kg (including 16.5 Kg of gaseous N_2). Center of Gravity location with respect spin axis is less than 1 mm (specific balancing masses have been added).

Thermal stability of payload is compliant to requirement :

- Thermal stability of SU is estimated better than 1 mK at fep.
- Thermal stability of FEEU is estimated better than 8 mK at fep.

Power budget in mission mode is 104 W (maximum power budget is 114 W in coarse mode).

Gas consumption has been estimated with a Monte Carlo approach: Minimal mission scenario could be done in 100% of cases; extended mission scenario could be done in 97% of cases.

SATELLITE DEVELOPMENT

Microscope mission was approved by CNES Scientific Program Committee in 1999.

Since from beginning most critical development was identified in payload and propulsion system; payload development was led by ONERA which previously provided similar instrument in the frame of GRACE and GOCE missions; concerning propulsion a cooperation with ESA, which faced off with same need for Lisa-Pathfinder mission, was established, ESA will deliver micropropulsion system as its contribution to Microscope mission, a trade off was made on propulsion system and Cesium Field Emission Electric Propulsion (FEEP) was selected as thruster for booth mission.

Mission feasibility studies were held from 2000 until 2003; as output satellite design gave a mass of 120 kg and a power budget of 80 W.

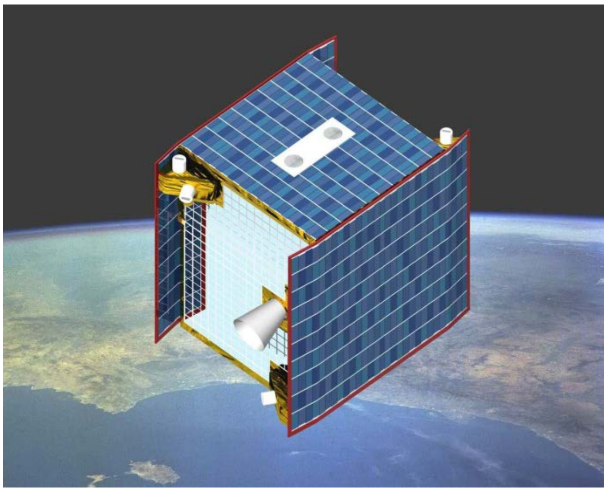

Figure 13: Microscope satellite phase A design

Preliminary design phase was held from 2003 to 2006.

Satellite budged at the time of preliminary design review (Figure 14) gave a mass of 190 Kg and a power budget in mission mode of 187 W

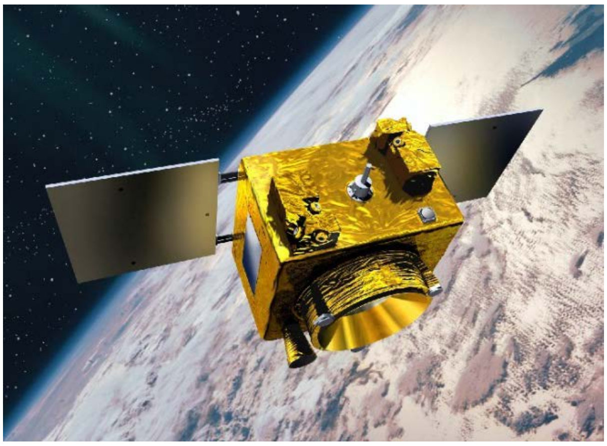

Figure 14: Microscope satellite phase B design

Due to de difficulty encountered on FEEP development CNES decide in 2009 to give up with FEEP and to study the implementation of cold gas propulsion system on the satellite.

Additional feasibility studies and preliminary design phase was held between 2009 and 2011; ESA support CNES decision and continued the cooperation providing cold gas thruster and its electronic control developped in the frame of GAIA.

Critical design phase was held from 2012 to 2014, at the time of critical design review satellite mass was estimated at 317 Kg and power budget 125 W

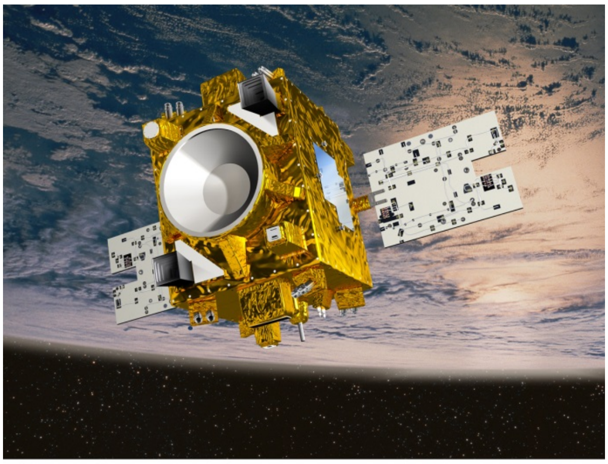

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Figure 15: Microscope satellite phase C design

Integration and validation phase last from May 2014 until February 2016.

Launch campaing took place from March to April 2016

Microperturbations

Microperturbations are defined as any kind of dynamic and sporadic phenomenon which could create a temporary perturbation on payload measurement.

The choice made at the beginning of the project concerning the orbit (Dawn/Dusk sun synchronous) and the decision to forbid active mechanism (reaction wheel, etc..) during mission mode allowed to reduce the occurrence of micro-perturbation.

In addition a micro-perturbation control plan has been established around several themes :

- Design rules : harness design to reduce Laplace electromagnetic forces, heater design including magnetic compensation, avoiding of liquids and ball-and-socket joint.
- Analysis and dimensioning : mainly related to gravity field variation and micro-gliding at junction due to thermoelastic, and the estimation of RF forces generated by the antenna during emission
- Tests : test have been performed on MLI to measure the micropertubation produced by the exposition of the external layer to sun illumination, the microperturbation generated by the CGPS tank during depressurization has been characterized, MT internal moving masses corresponding to a change in thrust set point have been also measured.
- Use of the mission simulator helped to estimate the effect on mission availability due to the impact of micrometeorites and microdebris.

Validation logic

Due to high sensitiveness of payload a complete and exhaustive validation of satellite on ground is not possible.

At payload level mainly two kind of test are possible:

- SU Free fall with mass unlocked using ZARM freefall tower facility in Bremen, which, giving few seconds of “almost weightless” conditions, allow to verify the beginning convergence of mass control.
- SAGE full chain test with masses locked, simulated data are superposed to SU measurement acquired by ICU using so-called “secondary entrance” in order to allow the verification of the integrity of payload and to check the control loop behavior.

A specific EGSE called Suzon, including an engineering model of ICU and numerical model for FEEU and SU, has been built for validation purpose, it has been coupled to satellite test bench in order to allows the software validation and to run scientific session to prepare mission exploitation.

Satellite went through « classical » validation process: reference test of all functional chain was performed at beginning and at the end of qualification sequence.

Qualification started in September 2015 and ended in December 2015, qualification test include EMC test, electrical compatibility test, thermal vacuum and Thermal balance test, physical measurement test, mechanical qualification tests (sine and acoustic), Solar Array development test and RF compatibility test.

Main milestones were thermal vacuum test and End-to-end AACS test.

Goal of thermal vacuum test was triple : verify the representativeness of thermal model, perform and extensive validation of all functional chains at extremes temperatures, characterize the CGPS working in representative condition (under vacuum and at several temperatures).

Thermal vacuum (Figure 16) consisted in 31 different phases each on corresponding to a specific validation goal and its duration was 21 days due to high thermal inertia of BCU and CGPS.

Figure 16: Satellite thermal vacuum test

End2End AACS test aimed to simulate on satellite a scientific session, sky seen by STR has been simulated using a specific test bench based on a tablet; payload measurement have been produced using secondary entrance, on flight central software was set in MCAN mode in order to verify the capability of AACS to work in representative condition with real hardware.

IN FLIGHT RESULTS

Microscope has been launched the 25th of April 2016 from Guyana Space Center by Soyouz on VS14 flight as an auxiliary passenger of Sentinel-1B.

Injection accuracy was very good and all orbital requirements have been respected by the launcher.

In flight commissioning is scheduled from end of April until beginning of September and it will performed in two phases :

- First phase from end of April until end of June aims to validate the behavior of all equipment, to go through all satellite modes and to characterize the each mission session.
- Second phase from end of July until beginning of September aims to validate the optimization introduced in payload and AACS setting following the exploitation of the first phase.

As preliminary evaluation: all equipment provides the expected performances; temperatures are on average within 2°C thermal model prediction; energy and gas consumption budgets showed important margin with respect analysis.

First drag free has been performed the 9th of June at 7h00 UTC; Figure 17 show the commanded forces and torques and Figure 18 show the acceleration (linear and angular) at drag-free point

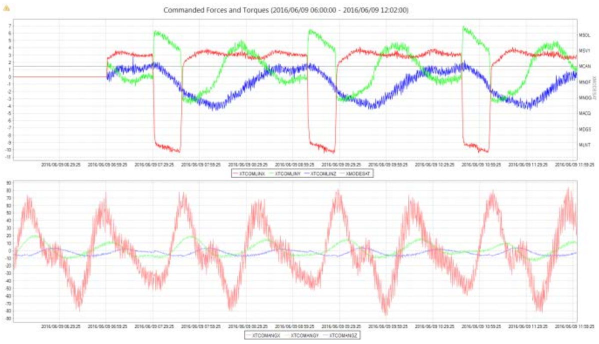

Figure 17: Satellite commanded forces and torques

Major non gravitational force to be compensated is solar radiation pressure; major torque is generated by the gradient of gravity.

AACS compensate also atmospheric drag, however generated force is limited due to orbit altitude and low solar activities

Note that during this drag-free session, AACS was settled in coarse mode so for mission session residual accelerations should be lower.

Figure 18: Residual acceleration at drag-free point

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.

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 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.

Even if satellite development lasted for around 16 years, schedule was very thigh especially during integration and validation phase because of the commitment to fit with Sentinel-1B launch date; difficulties meet on propulsion and payload jeopardized the development plan and leaded to several redesign of the spacecraft.

The satellite represents an excellent compromise between well proven robust technologies and narrowly specialized developments; excellence of design has been confirmed by first months of in flight commissioning.

Microscope constitutes first high accuracy 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.

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