MICROSCOPE satellite and its drag-free and attitude control system

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Abstract
This paper focuses on the description of the design and performance of the MICROSCOPE satellite and its drag-free and attitude control system. The satellite is derived from CNES’ Myriade platform family, albeit with significant upgrades dictated by the unprecedented MICROSCOPE’s mission requirements. The 300 kg drag-free microsatellite has completed its 2 years flight with higher-than-expected performances. Its passive thermal concept allowed for temperature variations smaller than 1 μK at the frequency of the equivalence principle test $f_{EP}$. The propulsion system provided a six-axis continuous and very low noise thrust from zero to some hundreds of microwatts. Finally, the performance of its DFACS (aimed at compensating the disturbing forces and torques applied to the satellite) is the finest ever achieved in low Earth orbit, with residual accelerations along the three axes lower than $10^{-12}$ m s$^{-2}$ at $f_{EP}$ over 8 days.

Keywords: general relativity, experimental gravitation, equivalence principle, space accelerometers, drag-free, microsatellite, cold gas propulsion

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1. Introduction

The weak equivalence principle (WEP) states that all bodies should fall at the same rate in a given gravitational field, independently of their mass or composition. MICROSCOPE is a CNES-ESA-ONERA-CNRS-OCA-DLR-ZARM fundamental physics mission dedicated to the test of the WEP in space. The MICROSCOPE satellite aimed at testing its validity at the $10^{-15}$ precision level by measuring the force required to maintain two test-masses (one made of titanium and the other of platinum alloys) exactly in the same orbit. The microsatellite was launched in 2016 into an altitude of 710 km dawn-dusk sun-synchronous orbit (6 PM local time at the ascending node). The scientific mission requires extremely accurate control of the linear and angular accelerations of the satellite. In science mission mode, the propulsion subsystem continuously overcomes the non-gravitational forces and torques (air drag, solar pressure, magnetic torques, etc) in such a way that the satellite follows the test-masses in their pure gravitational motion and keeps a well controlled rotation. The satellite is spun about the normal axis to the orbital plane in order to increase the modulation frequency of the Earth’s gravity field projection on the instrument measurement axis: the potential WEP signal is proportional to the Earth’s gravity field \[1\] and varies at a particular frequency, called $f_{EP}$.

The paper aims at describing the satellite developed for the MICROSCOPE mission in close synergy with the payload. Section 2 detailed the satellite main subsystems: they have constrained the mission scenario or contributed to the mission performance. The drag-free and attitude control system (DFACS) plays a particular role in the performance with the payload. Section 3 describes the DFACS and its needs. In the same way a particular focus was made on the propulsion subsystem in section 4 and on the precise orbit determination in section 5. The mission requirements in [1] have been distributed on these subsystems and are described in these sections which delivers also a summary of the observed in-flight performance. Finally, section 6 concludes the paper with a worked example of the DFACS performance in orbit which contributed to make this mission a success.

A list of notation and acronyms can be found in appendix in table A1.

2. Satellite description

2.1. Overview

The MICROSCOPE satellite [2, 3] has been developed within the framework of the Myriade micro-satellite product line. This line is dedicated to performing scientific or propaedeutic missions with reduced development schedule and costs targeting payload in the class of 60 kg/60 W. Because of MICROSCOPE’s challenging goals, the platform was specifically adapted in size but still with the line equipment. The architecture of the standard Myriade satellite is based on a platform with generic functional chains and on a mission-customized payload usually located on the top of the platform structure. In the case of the MICROSCOPE satellite, the design was constrained by (i) the need to control the acceleration of the satellite along its six degrees of freedom; (ii) the implementation of the payload as main attitude and orbit control system (AACS) sensor; (iii) the distinctiveness of payload interface and characteristics and (iv) the minimisation of the mass motions.

Since MICROSCOPE is more demanding in terms of performance and interface constraints, the Myriade generic platform has been adapted, and numerous elements of the microsatellite line were reused as ground support equipment, structural concepts and integration and
Figure 1. Upper panel: satellite reference frame with two disks illustrating the cylinder section of the two sensor units with the two units, SUREF and SUEP, containing the deorbitation wings. Lower panel: the bottom part of the satellite with the solar panels closed in launch position, the radiator baffle in the center with the two star-trackers on opposite sides.
Table 1. Main frequencies of interest for the different satellite pointing modes.

| Label | Frequency (mHz) | Period (sec) | Comment |
|-------|-----------------|--------------|---------|
| \( f_{\text{orb}} \) | 0.16818 | 5946.026 | Orbital frequency and EP frequency in inertial pointing |
| \( f_{\text{spin}2} = \frac{9}{2} f_{\text{orb}} \) | 0.75681 | 1321.339 | Spin rate frequency 2 (V2 mode) |
| \( f_{\text{spin}3} = \frac{35}{2} f_{\text{orb}} \) | 2.94315 | 339.773 | Spin rate frequency 3 (V3 mode) |
| \( f_{\text{EP}2} \) | 0.92499 | 1081.096 | EP frequency in V2 mode |
| \( f_{\text{EP}3} \) | 3.11133 | 321.407 | EP frequency in V3 mode |

The DFACS can operate in several satellite configurations: inertial pointing or rotating (i.e. spin) mode. In spin mode, the satellite rotates about the \( X_{\text{sat}} \) axis (figure 1) with a frequency rate of \( f_{\text{spin}} \), in the opposite direction with respect to orbital direction rotation for an addition of the two frequencies in order to increase the \( f_{\text{EP}} \). The gravity signal (i.e. the hypothetical EP signal) is then modulated at the frequency \( f_{\text{EP}} = f_{\text{orb}} + f_{\text{spin}} \), where \( f_{\text{orb}} \) is the orbital frequency. With different rotation velocities in order to vary the test conditions, the design allowed two spin frequencies: \( f_{\text{spin}1} = \frac{1}{2} f_{\text{orb}} \) and \( f_{\text{spin}2} = \frac{3}{4} f_{\text{orb}} \). After the commissioning phase, a third spin rate was derived from the first two spin configurations as the SpinMax mode with \( f_{\text{spin}3} = \frac{35}{2} f_{\text{orb}} \). Spin rate 1 was abandoned after the commissioning phase. Table 1 summarises the different satellite configurations and frequencies.

The satellite in flight configuration (i.e. with solar panel deployed) is around 1.36 m long (along \( X_{\text{sat}} \) axis), 2.78 m wide (along \( Y_{\text{sat}} \) axis) and 1.28 m deep (along \( Z_{\text{sat}} \) axis); the cross sections are of 1.52 m² normal to \( Y_{\text{sat}} \) and 1.12 m² normal to \( Y_{\text{sat}} \). Figure 1 gives an overview of the satellite.

2.1.1. Satellite and instrument reference frame. The satellite reference frame (see figure 1) is defined as follows:

- The \( X_{\text{sat}} \) is perpendicular to the launcher interface ring, while in orbit it corresponds to the spin axis.
- The \( Y_{\text{sat}} \) is defined by the direction of the two solar arrays (in stowed configuration).
- The \( Z_{\text{sat}} \) completes the triad and is aligned with the payload sensitive axis (cylinder axis).

Up to a micro-rotation, the instrument reference frame (\( X_{\text{inst}}, Y_{\text{inst}}, Z_{\text{inst}} \)) is defined as follows (see also figure 1):

\[
\begin{align*}
X_{\text{inst}} & \approx -Z_{\text{sat}} , \\
Y_{\text{inst}} & \approx +X_{\text{sat}} , \\
Z_{\text{inst}} & \approx -Y_{\text{sat}} .
\end{align*}
\] (1)
±1.4 mm. The in-flight data allowed to estimate the position of the CoG being at a distance of 0.4 mm from the sensor unit centers with a few tens of microns accuracy. There was no stringent requirement about the position of the CoG along $X_{\text{sat}}$ which is the rotation axis of the satellite. To minimise the parasitic forces and torques when spinning the satellite far from the center of mass, the inertia moments asymmetries have been measured and balanced to ±4 kg m² accuracy.

2.2. Structure

2.2.1. Payload module. The scientific instrument, called T-SAGE for ‘twin—spaced accelerometer for gravity experiment’, is accommodated inside the payload module (figure 2). It comprises two sensor units (SU) and two associated front-end electronics units (FEEU); see reference [4]. Each SU is a double electrostatic accelerometer composed of two cylindrical and concentric test-masses. One SU, called SUREF, comprises two test-masses of the same material for the check of the experiment, while the other SU, called SUEP, comprises two test-masses made of different materials for the equivalence principle test. The accelerometer measures the difference of accelerations [1] needed to be applied on the two test-bodies to keep identical motion while subject to the same gravity. The payload module is mounted on the panel opposite to the Sun in mission mode, for thermal stability reasons. It is located as close as possible to the center of the satellite in order to minimise the external perturbations. The satellite integration ensures that the test-mass centers of gravity are at less than 5 mm from the satellite center along $Y_{\text{sat}}$ and $Z_{\text{sat}}$.

The payload module was specified to provide a thermal stability of 1 mK at $f_{\text{EP}}$ at the SU interface, and of 10 mK at $f_{\text{EP}}$ at the FEEU interface [5]. The payload module is also designed to guarantee good mechanical alignment and stability with respect to the star sensor interface. Finally, the payload module is covered with a magnetic shield in order to limit the effect of magnetic field disturbances in the SU. It is designed as a two-stage system. The first stage accommodates both FEEU and their radiator: it is fixed to the platform through an interface ring by six titanium blades which guarantee the thermal decoupling from the rest of the satellite. The second stage accommodates the two SU and their magnetic shield: it is fixed to the first stage by six titanium blades which guarantee the thermal decoupling. Both stages are insulated by a multi layer insulator (MLI) to reduce radiative thermal exchange between the different stages and the platform.

2.2.2. Satellite structure. The MICROSCOPE structure has been designed and manufactured based on the same principle and material as the Myriade generic bus but with different dimensions and equipment accommodations (figure 3).

The compatibility of the structure with the auxiliary payload launching system, Soyounz ASAP-S, has been kept as design driver. The main structure is composed of six walls made with a honeycomb core and aluminum alloy skins, linked by aluminum alloy corners.

One wall accommodates the payload module, the star-tracker optical heads and the launcher interface structure and adaptor. Another one accommodates the IDEAS deorbiting system (see section 2.8). Two walls accommodate the cold gas propulsion system (CGPS—section 4). The remaining walls are dedicated to the classic Myriade platform functional chain (power, avionics, telemetry/telecommand, AOCS) and GNSS. The layout of the equipment has been optimised in order to balance the mass with respect to rotation axes and to minimise the inertia cross-products.

This modularity allowed us to optimise the acceptance and integration test (AIT) schedule by performing several integrations in parallel (payload module, CGPS, IDEAS and platform).
Figure 2. Payload module design.
2.3. Thermal control

From a thermal point of view, the satellite can be decomposed in several thermal cavities, each with its own independent thermal control: payload module stage, star-tracker optical heads, CGPS tanks, CGPS electronic module, satellite structure.

In order to minimise the power budget and thermal disturbances, the thermal control design is based mainly on passive principles using MLI and dedicated radiators. Heaters are used only in safe mode and transition mode to keep the temperature of equipment inside their operative or non-operative range; in science mission mode, heater activation is forbidden in order to protect the payload against electromagnetic perturbations. The platform radiators are the
Table 2. In-flight thermal variations observed at payload level.

| Location: satellite mode | Requirement (0-peak) | In flight results |
|--------------------------|----------------------|------------------|
| FEEU: inertial           | <10 mK at $f_{EP}$  | <4 mK            |
| FEEU: spin               | <3 mK at $f_{EP}$   | <0.4 mK for V2 mode |
|                          |                      | <0.8 mK for V3 mode |
| SU: inertial             | <1 mK at $f_{EP}$   | <0.5 mK          |
| SU: spin                 | <1 mK at $f_{EP}$   | <0.8 μK for V2 mode |
|                          |                      | <0.2 μK for V3 mode |

main sources of thermal disturbance to the instrument; for this reason, they are accommodated symmetrically and their surfaces have been defined in order to be as much as possible the same. However, the radiator for the FEEU thermal control is located at the launcher interface and cannot be balanced by a symmetrical one. In order to avoid any entrance at $f_{EP}$ from Earth’s albedo through this radiator into the payload module, the radiator is protected by a dedicated baffle. The performances of the thermal control are significantly better than the requirements; in-flight results are summarised in table 2, see references [5, 6] for more details.

2.4. Avionics and command/control

MICROSCOPE reuses the same on-board computer as the Myriade product line based on a T805 microprocessor and commercial off-the-shelf (COTS) components. It provides 1 GB of memory for housekeeping and science telemetry and up to 5 MIPS through an S-band link of 625 kbit/s.

The satellite modes and transitions are shown in figure 4:

- **MNLT** corresponds to launch mode: the satellite is off and only the separation detection circuit is powered.
- **MDGS** corresponds to solar array deployment mode: the satellite is switched on after the separation detection and the solar array is deployed after a countdown.
- **MACQ** corresponds to first acquisition mode: the satellite automatically points its solar arrays toward the Sun and spins slowly around the Sun direction to maximize the available power and stabilize the temperature.
- **MNOG** corresponds to a transition mode with a coarse pointing; this mode is used for commissioning all the equipment and as withdrawal mode during mission deterministic interruptions (eclipse season, moon transition, etc).
- **MNOF** is derived from the Myriade product line and makes use of CGPS as main actuator; it corresponds to a transition mode with a fine pointing that can also be used in the collision-avoiding operations.
- **MCAN** is a new mode that corresponds to the science mission mode; it is detailed in section 3.
- **MSV1** corresponds to a safe mode; its characteristics are the same as MACQ.

The MNLT, MDGS, MACQ and MNOG are generic Myriade modes which reuse the product line on-board software.
2.5. Power supply

The power supply chain reuses the same equipment as the Myriade product line. The solar array is composed of two panels; it uses the same cells arranged in a specific layout minimizing its magnetic momentum. The total surface is 0.84 m² and the maximum power is around 240 W.

The solar panels are stowed during launch; each panel is released after the separation using 3 pyro-nut mechanisms and deployed by one Carpenter joint [7]. Because the selected orbit and satellite attitude guarantee a good sunlight ratio, no solar array drive mechanism has been used.

The satellite battery is a standard Myriade product line equipment made with Li-ion cells; it provides a maximum energy of 390 Wh and a maximum capacity of 13.5 Ah.

The power conditioning and distribution unit (PCDU) ensures the regulation of the power generated by the solar array (up to 8 A). It distributes regulated buses (−15 V, +5 V, +12 V, +15 V and +20 V) and non-regulated buses (between 22 V and 37 V). The payload uses the non-regulated buses. The PCDU also provides the battery regulation, the generation of magneto-torque commands and the distribution of pyro commands (up to 12 power lines).

2.6. Attitude and orbit control system (AOCS)

The AOCS is used in low-level modes such as the launch and early orbit phase sequence, the transition modes and the safe mode. In the science mission mode, the AOCS is performed by the DFACS (see section 3).

The AOCS manages three modes:

- **MAS**: corresponds to the satellite safe mode; \( X_{\text{sat}} \)-axis is pointed toward the Sun with an accuracy of 20° and the satellite is spun about \( X_{\text{sat}} \) with an angular velocity of 0.25° s\(^{-1}\) (4.36 \( \times \) 10\(^{-3}\) rad s\(^{-1}\)).

- **MGT3**: corresponds to coarse pointing transition mode; \( X_{\text{sat}} \)-axis describes a cone of 10° aperture whose axis is perpendicular to the orbital plane and the satellite is spun about \( X_{\text{sat}} \) with an angular velocity equal to 3 times the orbital mean motion (i.e. orbital pulsation).

- **MSP**: corresponds to fine pointing transition mode; \( X_{\text{sat}} \)-axis is pointed perpendicular to the orbit with a quasi inertial attitude pointing.

The AOCS mainly uses the sensors and the actuators of the Myriade line equipment:
Table 3. Link between satellite modes and AOCS modes.

| Satellite mode | AOCS mode | Satellite control | Main active equipments |
|----------------|-----------|-------------------|------------------------|
| MNLT           | None      |                   |                        |
| MDGS           | None      |                   |                        |
| MACQ           | MAS       | Sun coarse pointing| SAS, MAG               |
| MSV1           | MGT3      | Slow rate spin around $X_{\text{sat}}$ | Reaction wheel, MTB |
| MNOG           | MSP       | $X_{\text{sat}}$ normal to orbit with 10° of tilt | MAG |
|                |           | Satellite spin around $X_{\text{sat}}$ | Reaction wheel, MTB |
|                |           | At 3 times orbital period |                        |
| MNOF           | MSP       | Three-axis control | Star-tracker, CGPS     |

- Three Sun acquisition sensors (SAS) with a hemispheric field of view make it possible to determine the Sun direction; two are accommodated along the same axis ($X_{\text{sat}}$-axis) with opposite direction, the third along a perpendicular direction ($+Z_{\text{sat}}$-axis).
- One magnetometer (MAG) performs the measurement of the three components of the magnetic field in the range of $\pm 60 \mu T$.
- One reaction wheel provides a momentum of 0.12 N m s and a maximum torque of 5 mN m.
- Three magneto torque bars (MTB) along three degrees of freedom, capable of generating a magnetic moment of 12 A m².
- A star-tracker assembly with two optical heads co-aligned and directed toward $-X_{\text{sat}}$ direction (anti-solar).

The star-tracker performances at $3\sigma$ are:
- RMS noise of 30 $\mu$rad around the line of sight and 250 $\mu$rad perpendicularly;
- Bias of 50 $\mu$rad around the line of sight and 200 $\mu$rad perpendicularly.

Table 3 summarises the correspondence between satellite modes, AOCS modes and equipment used.

2.7 Global navigation satellite system — GNSS

G-SPHERE-S [8] is a new spatial single-frequency GPS receiver manufactured by SYRLINKS, and is the outcome of a CNES R & D program aimed at the design of a low cost GNSS software receiver based on COTS components. MICROSCOPE was the first flight opportunity for this new GNSS. The receiver can also process Galileo signals as an experimental option, the baseline being the GPS signal.

The receiver software is highly configurable and the performance has already been improved using CNES orbit determination team analysis during ground tests and the commissioning phase. The MICROSCOPE satellite rotates around $X_{\text{sat}}$, which is also the cross-track axis, and the two antennas are placed on $X_{\text{sat}}$ opposite faces (see figure 1) in order to collect GPS signals transmitted to the receiver through an analog coupler. GNSS performances are described in more detail in section 5.

2.8 Innovative DEorbiting aerobrake system — IDEAS

Because of the low ballistic coefficient of the satellite, the time needed to de-orbit exceeds the limitation of 25 years enforced by French space law. This led to the development of a deorbiting system. This system has little impact on mass, volume and power. The selected
passive de-orbiting system called IDEAS (innovative DEorbiting aerobrake system) fulfills the requirements and has much less acceleration perturbation than chemical propulsion with tank sloshing. IDEAS is composed of two parts:

- Two identical wings, each including two sails of 4.54 m length and 0.39 m width and an inflatable boom ensuring its deployment,
- A vessel of 290 bars which inflates the sail masts with nitrogen.

Its operating principle is to deploy at the end of the mission braking surfaces (i.e. wings) that increase the atmospheric drag of the satellite, accelerating the natural reduction of the orbit. The geometry of the wings has been optimised in order to maximize the ratio between the deployed surface and the mass of de-orbiting system. In addition the design minimises, after their deployment, the difference between the minimum and the maximum drag surfaces with respect to the attitude of the satellite (figure 5). The drag area of the satellite increases from 2.09 m² to 5.44 m². In figure 5, the efficiency of the drag is shown through satellite tracking from Earth. The decrease rate of the orbit semi-major axis goes from $-0.28 \text{ m}/\text{day}$ to $-0.87 \text{ m}/\text{day}$ after the release identified by the slope discontinuity on figure 5. The change of the amplitude variations in this figure is due to a change of orbit determination data source.

3. DFACS design and needs

3.1 Drag-free and attitude control system (DFACS) motivations and challenges

The ‘drag-free’ mode stands for a satellite mode where it is accelerated to compensate the air drag and all other forces applied on the satellite (Sun radiation and Earth radiation mainly). There are basically two ways of performing ‘drag-free’ in space. In both cases the test-mass of an inertial sensor has to be free-falling and preserved from any forces except the gravity field. The first way consists in keeping the test-mass freely floating in a ‘cage’; the satellite (or the ‘cage’) follows the motion of the test-mass. The satellite propulsion system applies the needed thrusts to overcome the perturbations so that the test-mass remains free-falling and protected from external forces. The second way consists in measuring the acceleration of the test-mass with respect to the satellite and applying the needed thrusts to nullify the output of the accelerometer. As these outputs are determined by the forces applied to the satellite, these latter are totally compensated by the propulsion system. In principle, both methods are equivalent. In GPB [9, 10], LISA Pathfinder [11] or in LISA [12] missions, a combination
of these two methods is used with a very small electrostatic control of the test-mass as no acceleration measurement is needed. MICROSCOPE uses the second method like the GOCE mission [13], the first mission implementing ONERA’s space accelerometers in a drag-free system. It was also chosen because of the possibility to deliver acceleration measurement in a large range of $10^{-5}$ m s$^{-2}$. At last, it appeared more complex to guarantee the perfect free-falling of two concentric test-masses at the same time over long periods with no variation of the alignments and the centring by stray forces.

The DFACS specifications come from the measurement equation detailed in [1]. For the purpose of this paper, the difference of the applied acceleration to a perfect concentric pair of test-masses can be simply expressed by:

$$\vec{\Gamma}_d = \vec{\Gamma}_1 - \vec{\Gamma}_2 = \delta(2,1) \vec{g},$$

where $\vec{\Gamma}_i$ is the acceleration applied on the test-mass $i$ ($i = 1$ for inner test-mass and $i = 2$ for the outer test-mass); $\vec{g}$ is the Earth’s gravitation field; $\delta(2,1)$ is a good approximation [1, 14] of the Eötvös parameter of material 2 versus 1 defined by:

$$\delta(2,1) = \frac{m_2}{m_1} - \frac{m_1}{m_2}.$$

The actual instrument may present very small differences in the scale factor of the 2 test-masses (respectively $K_1$ and $K_2$). And thus, the measured signal $\vec{\Gamma}_{md}$ is expressed by $\vec{\Gamma}_{md} = K_1 \vec{\Gamma}_1 - K_2 \vec{\Gamma}_2$. The measurement equation is also expressed in terms of common-mode acceleration $\vec{\Gamma}_c$ (i.e. the mean applied acceleration to both concentric test-masses) and differential-mode acceleration $\vec{\Gamma}_d$ (the difference of applied accelerations):

$$\vec{\Gamma}_{md} = K_c \vec{\Gamma}_d + 2K_d \vec{\Gamma}_c,$$

where $K_c = \frac{1}{2}(K_1 + K_2)$ and $K_d = \frac{1}{2}(K_1 - K_2)$. This equations shows that, in order to relate $\vec{\Gamma}_{md}$ to $\vec{\Gamma}_d$, it is necessary to minimise the product $K_d \vec{\Gamma}_c$ by balancing the specifications on $K_d$ and on $\vec{\Gamma}_c$ depending directly on the performance of the drag-free. Reference [1] shows how this equation leads to the requirements on the scale factor matching $K_d$ to $1.5 \times 10^{-4}$, with the common-mode acceleration limited to $10^{-12}$ m s$^{-2}$ at the EP frequency about all axes by the DFACS. The DFACS control loop acts over 0.1 Hz bandwidth, the accelerometer one being about 1 Hz. Equation (2) can also be adapted when the two test-masses are not perfectly concentric and miscentred by $\vec{\Delta}$, specified to about 20 $\mu$m along all axes.

$$\vec{\Gamma}_d = \vec{\Gamma}_1 - \vec{\Gamma}_2 = \delta \vec{g} + ([\Omega] - [\Omega]) \vec{\Delta},$$

where $[\Omega]$ is the Earth gravity gradient tensor in the instrument’s frame and $[\Omega]$ is the gradient tensor of inertia which is linked to the attitude motion of the satellite defined by $[\Omega] = [[\Omega]] + [[\dot{\Omega}]]$ with $[[\Omega]]$ a tensor representation of the operator $(\vec{\Omega} \times)$ where $\vec{\Omega}$ is the satellite angular velocity vector and $[[\dot{\Omega}]]$ its time derivative. The ‘inertial’ part of the equation depends on the attitude control. With an error allocation of $2 \times 10^{-16}$ m s$^{-2}$ for each term, the specifications on the angular motion are:

- $[[\dot{\Omega}]]$ limited to $10^{-11}$ rad s$^{-2}$ at $f_{EP}$ both in inertial and rotating modes;
- $[[\Omega]]$ limited to $10^{-9}$ rad s$^{-1}$ at $f_{EP}$ in rotating mode.

It is worth noticing that a specification of $10^{-9}$ rad s$^{-1}$ at $f_{EP}$ (which is about 3 mHz in rotating mode) corresponds to an attitude stability of a fraction of $\mu$rad at $f_{EP}$. It is obtained
Figure 6. DFACS servo-loop. \( \mathbf{Q}_c \) is the quaternion control setpoint, \( \mathbf{F}_{CGPS} \) is the force and torque applied by the CGPS, \( \mathbf{\dot{\gamma}} \) is the acceleration applied to the payload.

thanks to an active attitude control and the use of T-SAGE angular axes for attitude estimation at \( f_{EP} \).

3.2. DFACS servo-loop description

The DFACS control loop (figure 6) uses the scientific instrument as the main sensor for delivering the linear and angular accelerations and the CGPS as the actuator. The DFACS can be controlled by any combination of the test-masses’ measured acceleration. Depending on the session, either the SUREF or the SUEP is switched on alone and serves as the DFACS reference sensor. Most of the time, the outer test-mass of reference sensor was chosen for the science sessions while few sessions have been performed with the mean value of two test-masses’ measurements. The linear acceleration is transmitted almost without processing to the drag-free control laws, the estimated DC offset being subtracted from the command to minimise the thrust consumption. The attitude estimation uses T-SAGE angular acceleration measurement and the star-tracker measurement quaternion. The Kalman formulation is very convenient for hybridizing these two measurements. The acceleration measurement is used in the Kalman state model for the short term, the star-tracker measurement is used for the Kalman filter updating and therefore for the long term.

Various sets of hybridization filters and controllers are available for different uses. The controller’s outputs (force and torque in satellite frame) are projected into the thruster axes. As many as 38 loops are closely involved: 6 times 4 loops for the suspension of the test-masses, 6 loops for the DFACS itself and 8 loops for the local regulation of the thrusters.

3.3. DFACS software

Figure 7 presents the software architecture of the DFACS. On top of this figure, the MSP mode, defined in section 2.6, performs a fine attitude control through the star-tracker measurement and CGPS torques. This mode is also used in collision-avoidance procedures but its main function is to be the gate for the drag-free mode MCA in which one or several test-masses of T-SAGE are used. The MCA mode is made of many tunings: the MCA3 (attitude only) and MCA6 (six-axis control) for instance have low-gain robust control and are used to estimate the angular
bias of the drag-free test-mass, to change the attitude guidance, etc. The MCAcp performs an automatic sequence of tests for the thrusters. The other high-gain tunings are dedicated to inertial sessions (MCAi), rotating session velocity 1 (MCAs1), etc.

As shown in table 1, several rates of satellite rotation were tested. Finally the selected EP science sessions were performed with spin rates 2 and 3, the latter called also SpinMax. Figures 8 and 9 refer to the SpinMax mode. Figure 8 shows the Bode diagram hybridization filter between star-tracker quaternions and the T-SAGE angular accelerations $\dot{\Omega}$. The hybridization frequency remains high in order to have reasonable convergence duration. The shape of the transfer is modeled using advanced techniques [15, 16]. With this filter, the attitude estimation is not corrupted by errors at $f_{EP}$ on the star-tracker.

Concerning the control of the linear acceleration, as air drag acts mainly at $f_{EP}$ with 30 $\mu$N intensity (i.e. $10^{-7}$ m s$^{-2}$ with the 300 kg satellite mass), a minimal rejection of 10$^5$ (i.e. 100 dB) is then needed to reach the required $10^{-12}$ m s$^{-2}$. The challenge becomes to drop the gain above $f_{EP}$ quickly enough, to keep stability margins. Figure 9 illustrates the drag-free controller and compares the original tuning to the one actually used in orbit in SpinMax. In flight the air-drag force was smaller than 3 $\mu$N, i.e. ten times weaker than the design hypothesis and a 90 dB rejection was sufficient; but on the other hand this gain had to be maintained at higher frequency due to the introduction of the SpinMax mode. We managed to have a net delay margin of 1 s at 0.075 Hz but this control is extremely tight.

### 3.4. Attitude guidance in science mode

The science measurements were carried out in either inertial sessions or rotating sessions. During the inertial sessions, the satellite attitude follows the one degree per day drift of the orbital plane: we may use the exact term of quasi inertial mode, but for simplicity in the paper we use the wording inertial mode in opposition to spin (or rotating) mode. The main axis of the accelerometer ($X_{inst} \approx Z_{sat}$) is in the mean orbital plane (figure 10).

On the right side of figure 10, the ‘$X_{sat}$ to Sun’ angle can be observed between the spin axis $X_{sat}$ and the Sun direction. The orbit is entirely sunlit except from May 8 to August 4 where eclipses happen around the southern pole. This angle plays an important role for thermal and
Figure 8. Bode diagram gain and phase in SpinMax configuration for the two transfer functions of the hybridization attitude controller. $\theta_{\text{est}}$ is the Kalman estimator filter output for the best estimate of attitude, $\theta_{\text{SST}}$ is the output of the star-tracker, $\theta_{\text{ACC}}$ comes from a double integration of the angular acceleration output of the accelerometer. The continuous line is the transfer function of the Kalman filter with respect to the star tracker, the dot line is the transfer function of the Kalman filter with respect to the accelerometer.

Micro-perturbation aspects. Science sessions were interrupted for the eclipse seasons because of non-optimal thermal stability. In addition, once per month, the satellite has to be slightly tilted to avoid the Moon’s glare to the star-trackers. During these few days, science sessions are also interrupted. Between two Moon phases, several EP sessions are played surrounded by calibration sessions. EP sessions were defined in the mission design with a typical duration of 8 days, actually 120 orbits. All operational constraints that led to build the mission scenario are detailed in [17]. In particular, the noise of the accelerometer higher in flight than expected in inertial pointing [14] led to define rotating sessions as the baseline for the EP test thanks to a better accelerometer performance in this configuration. In rotating mode, the differential acceleration measurement performance reaches about $10^{-11}$ m s$^{-2}$ Hz$^{-1/2}$. Inertial pointing was still used for calibration as it needs less performance.

As the potential EP violation signal is presumably proportional to the gravitational field [1], it is expected to be a sine at the modulation frequency of the gravitational field. So the attitude guidance defines precisely the EP test frequency.

The calibration sessions were dedicated to the accelerometer calibration. Based on an inertial pointing, they consist in performing two different types of stimuli:
Figure 9. Drag-free controller closed loop gain and phase: the continuous line represents the original tuning called old (adapted to spin rate 1) and the dotted line is the one called new obtained for SpinMax.

Figure 10. Inertial sessions (left), rotating sessions (middle) and side view (right).

- Linear: an additional signal (sinusoidal at $f_{cal} = 1.22848 \times 10^{-3} \text{ Hz}$) is set on DFACS accelerometer axis output, leading to a sinusoidal thrust command along the calibration axis.
- Angular stimuli: in the same way, the attitude set-point follows a 50 mrad amplitude sinusoidal profile at $f_{cal}$ and leads to a satellite periodic oscillation.
The objective of these stimuli is to generate a reference signal that can be measured by the accelerometer out of the DFACS loop.

4. Cold gas propulsion system—CGPS

The European Space Agency (ESA) provided the microthrusters and the CGPS command electronics. This contribution was derived from the cold gas propulsion system of GAIA and LISA Pathfinder European missions [18]. CNES was in charge of the implementation of the whole CGPS, including general design, integration and testing. The tanks and the high-pressure regulators were off-the-shelf components.

4.1. CGPS requirements

The specification on the CGPS has been established to fulfill the DFACS needs in terms of science performance [1] and for the satellite maneuver needs. The thrust range command of the CGPS was specified to 500 $\mu$N with a minimum thrust command of 1 $\mu$N and a resolution of 0.2 $\mu$N. In order to give stability margins on the DFACS servo-loop, the thrust’s time response has to be lower than 250 ms at 63% of the thrust command. The overshoot (difference between the executed and the commanded impulse) must be lower than 25% of the commanded impulse on 250 ms. In static conditions (constant thrust set point) the thrust noise must be less than 1 $\mu$N Hz$^{-1}$ above 1 Hz. The thruster specific impulse dispersion must be lower than 5% for every thrust set point. The acceleration measured by T-SAGE in variable thrust conditions (for a thrust set point variation of 2 $\mu$N) has to be lower than $10^{-9}$ m s$^{-2}$.

4.2. CGPS description

The CGPS is divided into two identical and independent subsystems called CGPSS (see figure 11), accommodated on the two opposite walls of the satellite ($-Z_{sat}$ and $+Z_{sat}$).

Each CGPSS stores, within 3 connected tanks of 9 L, a total of 8.25 kg of nitrogen at 345 bars decreasing to a minimum operational value of 10 bars at the end of the mission. The two systems are independent and commanded by the on-board computer on the basis of the DFACS output and with a particular survey of the consumption balance between the two walls.

From the tanks, a double-stage mechanical pressure regulator ensures a regulated 1 bar output pressure to the input of a 0.7 liter plenum. The role of the plenum is to absorb the internal leakage pressure during non-operating phases and to reduce crosstalk between thrusters. The thruster module, fed by the plenum, comprises 4 nominal micro-thrusters and 4 redundant ones. During the mission, the 4 redundant micro-thrusters were fortunately not necessary. Nevertheless, they were tested at the end of the mission for technological survey. For each micro-thruster, a measurement of the gas flow is available through a mass flow sensor (MFS). The flow is controlled continuously by the CGPSS electronics that modifies the nozzle cross section of the micro-thruster valve using a piezoceramic actuator.

The micro-thrusters are accommodated by pairs (the nominal and the redundant one) at the corners of the satellite walls. The position and the orientation of thrusts have been optimised in order to maximize in every direction the total force and torque acting on the satellite.
The CGPSS electronics control module ensures the interface with the on-board computer and runs the flow regulation at 50 Hz and delivers measurements at 4 Hz as the payload does: the algorithm includes a specific anti-hysteresis controller in order to improve time responses.

4.3. CGPS ground tests

4.3.1. Microperturbation. The requirement of a maximum induced acceleration of $10^{-9}$ m s$^{-2}$ has been verified with numerical simulations taking into account the dynamic response of T-SAGE and the main characteristics of the satellite. It has been converted into an equivalent
specification of a 1 kg mass which can move less than 1.2 nm. Two sources of microvibration were identified during the preliminary design phase:

- Moving masses inside mechanism (pressure regulator and micro-thruster). The moving mass requirement on micro-thruster being lower than the sensitive level of the microvibration table used for testing, its conformity could not be assessed directly by tests. The moving mass of the plunger and the amplitude of its motion were estimated using a dedicated experiment in order to provide a dynamic model. Using DFACS numerical simulations, the thrust command variations at 4 Hz were estimated to be lower than 2 μN, leading to a small displacement of the plunger and a maximum value of 0.3 kg nm compliant with the requirement.

- ‘Clank’ created by tank volume variation due to pressure evolution over time. A dynamic model of the pressure regulator was created using the information provided by its manufacturer; results were compliant with the requirement and no need for further tests was identified. Furthermore, when the tank pressure decreases due to gas consumption or temperature variation, the volume of the gas decreases and a sudden displacement of the tank liner and fiber may occur, creating acceleration spikes. A ground test was performed recording the tank external surface with two high speed cameras, in order to reproduce via a stereoscopic effect a 3D image of the tank while the tank’s internal pressure is decreasing. This set-up enables a measurement, with a few milliseconds’ resolution, of the displacement of the tank external surface and a characterization of the sudden variation due to pressure change. Some out-of-specification spikes were observed; however, their frequency was around a few occurrences per hour and their perturbation induced on payload was deemed acceptable.

4.3.2. Fluidics test and thrust regulation. A simplified engineering model of CGPSS composed of two micro-thrusters, a pressure regulator and an engineering model of the electronic control module was used early in project development to characterize the coupling between CGPS elements. Two major phenomena were observed during these tests.

First, strong oscillations appeared when a thrust step was commanded after a long idle period (i.e. thrust commanded to a fixed low thrust value); the problem was due to the control loop which automatically adjusted thrust hysteresis gain to a too-high value, too much increasing the thruster control loop sensitivity and generating oscillations. Once the algorithm was modified and control loop gain values were at limited level, this phenomenon disappeared without degradation of the response time.

Second, a ghost mass flow was observed even when a zero thrust was commanded. This phenomenon is due to the impact of the pressure regulator whose response time is far greater than that of the thruster (several minutes over 250 ms). The pressure regulator adjusts its working point according to the total gas flow requested by the thrusters. When the global mass flow changes faster than the response time of the pressure regulator, the output pressure never converges to a fixed value. Consequently, a ghost mass flow induced by pressure variation is detected by the MFS. This phenomenon is amplified by the dead volume between MFS and the thrust valve nozzle, which were misestimated during preliminary development.

The discrepancy between the real mass flow and its measured value by MFS is roughly 2% when thrusters are operating. This value has been judged acceptable for the mission thanks to the DFACS closed loop margins.

Nevertheless, the existence of a ghost mass flow changed the in-flight procedure for thruster offset calibration and drove to a modification of the parameter of the leakage
detection algorithm implemented on the on-board fault detection, isolation and recovery (FDIR) procedure.

Response time, overshoot and thruster noise requirements involve short-duration phenomena which are not observable in flight due to telemetry sample frequency limitation. For this reason, their validation was performed mainly on the ground as a result of improved observability.

Based on the analysis of several ground test sessions, response time requirements were respected for 97% of commanded steps and overshoot was respected for 90% of the time. The upper panel of figure 12 compares the commanded force with respect to the realized one.

The thrust noise was characterized on ground by two different means:
Table 4. Total cold gas consumption observed in orbit.

| Satellite configuration | Mean consumption in g/orbit | Total consumption in g |
|-------------------------|----------------------------|------------------------|
| Inertial                | 1.2                        | 2300                   |
| Calibration in inertial pointing | 5.0                        | 1400                   |
| Rotation V2             | 1.2                        | 1100                   |
| Rotation V3             | 6.6                        | 10700                  |
| Transitions             | 1.2                        | 700                    |

- In the low frequency band [0.001–0.1] Hz, the noise was directly measured on a micro-balance developed at Onera for the GAIA, LISA Pathfinder and MICROSCOPE missions [19];
- In the high frequency band [0.1–10] Hz, the noise was inferred by the MFS flow measurement recorded at 50 Hz frequency sampling with high sensitivity and good resolution.

The lower panel of figure 12 shows the high frequency noise recorded during satellite thermal vacuum test. Some dispersion between thrusters were observed during ground testing and credited to manufacturing dispersion that lead to a different non-linear response between thrusters (micro-thruster control algorithms use the same parameters for all the thrusters); nevertheless, these differences remain limited and do not affect DFACS. Reference [20] gives more information about CGPS design.

4.4. CGPS in-flight performances

In-flight performance characterizations were limited by T-SAGE’s rate acquisition (4 Hz maximum). However, the response time of the subsystem is confirmed in space by the excellent behavior of DFACS, especially in rotating mode which has very small delay margin.

Thrust range, thrust resolution and thruster linearity were directly verified during the in-flight commissioning by comparing the propulsion telemetry with the accelerometer measurements. The results fit with the expected values, except for the CGPS scale factor being 10% under the ground calibration. The error is attributed to an offset introduced by the mass flow ground equipment used to calibrate the MFS during thruster manufacturing. These differences between commanded thrust and performed thrust were indirectly confirmed by gas consumption, which was 10% less than expected. It was also assessed during tests dedicated to collision-avoidance maneuvers by tracking the satellite position at full thrust range.

Nevertheless, this scale factor does not affect the global behavior of DFACS because, in closed loop, the accelerometer drives its output to zero. The scale factor error is absorbed by the servo-loop gain margins. The thruster offsets have been calibrated monthly, showing a drift less than 1.4 μN/month on one thruster case, and ten times less for the others.

With a starting total gas capacity of $2 \times 8.25$ kg, the gas consumption depends on the type of session [1, 17] (table 4). The consumption is determined by the gyroscopic torque which varies with the square of satellite rotation frequency. Reference [21] gives more information about in-flight CGPS performances.
### Table 5. Orbit determination performance requirements on positioning error. In bold: driving errors.

| Frequency | Radial | Along-track | Cross-track |
|-----------|--------|-------------|-------------|
| DC        | 100 m  | 100 m       | 2 m         |
| $f_{EP}$  | 7 m    | 14 m        | 100 m       |
| $2f_{EP}$ | 100 m  | 100 m       | 2 m         |
| $3f_{EP}$ | 2 m    | 2 m         | 100 m       |

### 5. Precise orbit determination

#### 5.1. Scientific requirements

In order to estimate the equivalence principle violation parameter, it is necessary to accurately compute the gravity acceleration and the gravity gradient at each point of the orbit [22]. Therefore, a positioning performance of a few meters on the orbit determination is required at the EP test frequency $f_{EP}$ and its first harmonics. The requirements on the orbit determination (OD) concern positioning biases (DC errors), and positioning errors at $f_{EP}$, $2f_{EP}$ and $3f_{EP}$ frequencies (table 5). The accuracy requirement on the knowledge of the satellite position is driven by the need to correct the accelerometric measurements from the difference of position of the test-masses coupled to the Earth’s gravity gradients. The dominant part of the gravity gradient tensor has component of the form $\frac{\partial}{\partial r} \left( \frac{\mu}{r^2} \right)$, $\mu$ being the Earth’s mass and $r$ the orbit radius. Thus errors at the $f_{EP}$ frequency on these components computed in the instrument frame can emerge from positioning errors at various frequencies in the local orbital frame leading to the requirements of table 5 [1, 5].

The orbit determination is performed with the ZOOMIC automated processing chain, derived from the CNES ZOOM reference software [23], used for many other missions. A precise orbit was computed for each session of the MICROSCOPE mission within the day following the end of the session. Orbit and associated products (expertise report, error assessment) were then delivered weekly to the scientific mission center [17]. The requirements on OD have been established after several iterations and trade-offs by considering the mission performance needs [1] and the ZOOM performances.

However, due to the orbital dynamics, the satellite OD is mainly sensitive to the constant cross-track perturbations and the perturbations at the orbital frequency in the orbital plane, whatever the rotation frequency rate of the satellite. Consequently, in rotating mode, these errors will hardly impact the gravity acceleration or the gravity gradient at the frequency of the test, $f_{EP} = f_{orb} + f_{spin}$.

#### 5.2. GNSS receiver measurement and processing

The receiver delivers position, velocity and time (PVT) and L1 C/A code and carrier phase measurements for 9 channels, with a time to first fix below 90 s. The default data sampling time is 10 s, but a higher sampling time (2 s) is possible in technical sessions. The receiver clock drift is about 1 s/day and a posteriori corrected using GPS information even if not dimensioning.

At an altitude of 710 km, code and phase observables are affected by ionosphere delays, whose magnitude can reach several tens of meters.

The major advantage of using an ionosphere-free combination is to smartly reject ionosphere-affected measurements in the pre-processing, and then to keep the maximum information without being affected by very disturbing ionosphere effects. The combination used is
the semi-sum of code and phase data. The resulting noise is the code noise divided by 2, with the ionosphere effect canceled.

5.3. Orbit determination performance

The precise determination of MICROSCOPE orbit relies on the new spatial GNSS single-frequency receiver G-SPHERE-S [8]. The GPS ionosphere-free-based, PVT-based combined to the one-way Doppler-based OD [24] are computed for each scientific session, allowing cross-check analysis. GPS-based OD is the reference orbit. The performance is controlled through several indicators, such as estimated covariance, orbit overlapping analysis, magnitude of estimated parameters and final OD residuals [24]. The OD accuracy estimate for scientific sessions (120-orbit length) is given in table 6. Among the driving errors in bold in table 6, the requirement of 7 m at $f_{\text{EP}}$ (which is $f_{\text{orb}}$ in inertial mode) is the most challenging to achieve without GPS receiver.

6. DFACS performances: a worked out example

In this section, the behavior of the DFACS is illustrated by an example typical of a 120-orbit long session. We set our discussion on session 256, a SpinMax session beginning end of April 2017, with the DFACS controlled by the outer test-mass of SUEP instrument. This session is representative of all science sessions which have been repeatable over the 2 years of mission.

6.1. Orbital perturbations

In inertial and low spin rate sessions, the needed thrust is dominated by magnetic torques to be compensated, while in SpinMax and calibration sessions, drag and external forces dominate. In addition, the SpinMax gyroscopic torques due to the angular guidance about the accelerometer placed out of the center of gravity, have a major contribution to the compensation thrust. Finally the control is dominated by torques which have determined the consumption and thus the mission duration; the linear control requires very low thrust as illustrated in table 7. The main thrust components are due to the rotation of the satellite. The rotation is performed about the $X_{\text{sat}}$ at the center of the concerned SU and thus the small offcentring of the center of gravity out of the $X_{\text{sat}}$ axis has to be compensated (origin of the major thrust about $Z_{\text{axis}}$). This offcentring generates also disturbance, through the gravity gradient, that must be corrected at $2 f_{\text{EP}}$. At last, the moment of inertia of the satellite is not fully balanced and induces an additional DC component in the torque command. The DC linear thrust force is minimised thanks to the estimation of the accelerometer offset which is subtracted from the command force control in order to reduce the consumption.

In order to save propellant, the accelerometer estimated DC offsets are compensated in the loop, and the force at DC is the resulting compensation residual. The air drag acts mainly in the orbital plane ($Y_{\text{sat}}$, $Z_{\text{sat}}$) and at $f_{\text{EP}}$ it was measured lower than 3 $\mu$N. When the Sun is distant from $X_{\text{sat}}$ (by 25° here), the solar pressure produces a force of 5 $\mu$N about $Y$ and $Z$ axes at $f_{\text{spin}}$ frequency. The torques are dominated by gyroscopic effects (static, $Y$ and $Z$ axes) caused by non-diagonal inertia terms ($\Omega^{2}_{\text{spin}}$-dependent). The magnetic and gravity gradient torques act at

\begin{table}
\centering
\caption{Orbit determination estimate accuracy in drag-free mode.}
\begin{tabular}{|c|c|c|c|}
\hline
Component & Radial & Along-track & Cross-track \\
\hline
RMS OD accuracy & 10 cm & 30 cm & 15 cm \\
\hline
\end{tabular}
\end{table}
Table 7. Session 256, DFACS control force and torque in satellite reference frame.

| Force $\mu$N | Torque $\mu$N m |
|--------------|-----------------|
| $X_{sat}$    | $Y_{sat}$ | $Z_{sat}$ | $X_{sat}$ | $Y_{sat}$ | $Z_{sat}$ |
| DC           | -7.99  | 18.08   | -3.06  | -2.91  | 94.82   | 186.38  |
| $f_{orb}$    | 0.38   | 0.03    | 0.02   | 0.2    | 0.01    | 0.02    |
| $f_{spin}$   | 0.23   | 4.90    | 4.94   | 1.00   | 4.22    | 4.02    |
| $f_{EP}$     | 0.08   | 2.67    | 2.49   | 0.07   | 0.41    | 0.12    |
| $2f_{EP}$    | 0.19   | 1.21    | 0.28   | 39.50  | 1.20    | 1.05    |

Figure 13. Session 256, DFACS commands to propulsion over 1 orbit ($\mu$N).

It is clear that whatever the type of session, the propulsion system is mainly used to control the attitude and very little for compensating the satellite drag as illustrated by the table 7.

Once the command is projected from the reference satellite frame to the thruster frame, we observe on figure 13 that thrusters #1 placed on the $Z+$ satellite side (upper left curve) and #8 placed on the $Z-$ satellite side (lower right curve) remain idle for the entire 8 dayss session. In contrast, thrusters #3 and #6 are loaded close to 200 $\mu$N. Most of the EP sessions shows this 4 Hz typical plot, with a very smooth command and little randomness sent to thrusters in mission modes, ensuring a very low noise environment to the satellite.

6.2. Clank perturbations

The effect of clanks on the measurement is presented in [25] (where they are referred as ‘glitches’). We focus here on the satellite design and operation that lead to these clanks. Figure 14 shows an example of linear measurement given by the inner SUEP test-mass over one orbit. We observe spikes almost evenly spread on the orbit with a rate at particular frequencies such as $f_{orb}$, $f_{spin}$ and $f_{EP}$. A periodicity of $f_{spin}$ ($T_{spin} = 340$ sec) would suggest a solar origin and a periodicity of $f_{EP}$ ($T_{EP} = 321$ sec) would suggest the Earth albedo. Most spikes happen when $Y_{sat}$ panel is oriented toward the Earth, but not systematically.
This session 256 is one of the most perturbed compared to other SpinMax sessions at other dates. Because of these strong variations with time, spikes may probably have an internal origin [26]. It cannot be hyper velocity impacts coming from the outside, which have a dissymmetrical signature: the mean acceleration should be, in this case, not null which is not observed. In addition the frequency of these spikes is much more correlated with the lighting of the surfaces of the satellite by the Sun and the Earth, which confirms an internal origin. The spikes are produced by a mechanical force due to thermomechanical expansion of the satellite coatings with temperature variations. This force generates what we call ‘clanks’, ‘crackles’ or ‘glitches’ and were already observed in the CHAMP mission [27] and in the LISA Pathfinder mission [28]. Here, the clanks can come from two different origins:

- Internal clanks due to variation of temperature around the payload module covered by an MLI, rather at the beginning of the inter-moon period;
- Surface clanks due to fluctuation of Sun or Earth illumination of the MLI covering the satellite. The density of spikes could depend on the angular distance between \( X_{\text{sat}} \) (spin axis) and the Sun (see figure 10).
Figure 15. Session 256, innovation of the hybridization filter for 1 orbit in radians in the satellite reference frame.

The spikes are lower than $10^{-7}$ m s$^{-2}$, short and with damped sine shape (see figure 14). Because they are brief with a mean value near zero, their impact on DFACS performance is negligible. Knowing T-SAGE transfer function, we invert the spike of figure 14 and find that it roughly corresponds to a $100 \, \text{g} \times \mu\text{m}$ instantaneous displacement (for example a 2 g piece of MLI suddenly moving by 50 $\mu\text{m}$). This inversion is sufficiently precise for satellite analysis but not for science correction as shown in [22, 29] where a particular process was applied to cope with clanks which effect is actually seen differently by the two test-masses.

6.3. Attitude hybridization

Figure 15 presents the innovation of the hybridization filter for one orbit (i.e. the difference between the star-tracker measurement and the estimated attitude). We observe a clear periodicity at $f_{\text{spin}}$ with an amplitude of about 200 $\mu\text{rad}$ on $X_{\text{sat}}$ (spin axis and star-tracker line of sight) and 50 $\mu\text{rad}$ on cross axes. This innovation is interpreted as an error from the star-tracker, caused by the rotation of the star pattern in the rectangular field-of-view. Some days before the beginning of session 256 not in the plot, the amplitude is comparable but the error around $X_{\text{sat}}$ is mostly at $2f_{\text{spin}}$. This plot shows the interest of the hybridization filter which discards the star-tracker field of view errors; the star-tracker is used only at very low frequency. With the hybridization, the error at $f_{\text{EP}}$ is about 50 times smaller (3.1 $\mu\text{rad}$ at $f_{\text{EP}}$ about $X_{\text{sat}}$) than the error of the star-tracker alone. In the same way for the angular velocity, if the star-tracker measurement was used alone to control the satellite, we would have a stability of about $6.1 \times 10^{-8}$ rad s$^{-1}$, or 61 times above the requirement. The hybridisation filter is a key element for the performance of the DFACS and makes the requirement to be fulfilled as illustrated in the figure and in the following sections.

6.4. Drag-free performance

The drag-free performance is estimated through the accelerometer output used for the control loop. Figure 16 shows the typical residual acceleration observed along $X_{\text{inst}}$: $2 \times 10^{-13}$ m s$^{-2}$ at $f_{\text{EP}}$. Along $Z_{\text{inst}}$ the residual acceleration is lower than $2.2 \times 10^{-13}$ m s$^{-2}$ and lower than
Figure 16. Session 256, spectrum (FFT) of the accelerometer output used by the DFACS along $X_{\text{inst}}$ with the requirement line standing for the $f_{\text{EP}}$ needs.

0.1 \times 10^{-13} \text{ m s}^{-2} \text{ along } Y_{\text{inst}}. \text{ The performance is also looked at other frequencies in order to evaluate the effect of recombinations as specified in reference [1].}

The control gain quickly drops above $f_{\text{EP}}$ (3.11 mHz for the showed session). The $2f_{\text{EP}}$ peaks are interpreted as angular-to-linear coupling due to propulsion: the 39.5 $\mu$N m torque at $2f_{\text{EP}}$ is compensated by propulsion which inevitably causes a perturbation in force. This perturbation is rejected by drag-free, but with a limited gain. The first bump at $2 \times 10^{-2}$ Hz comes from a transmission of star-tracker stochastic noise, and the second one at $2 \times 10^{-1}$ Hz is intrinsically due to the test-mass suspension. Isolated spectral lines are observed around 1 Hz and 2 Hz, caused by aliasing of signal at higher frequencies than the 4 Hz frequency sampling. The numerous spikes between the mHz and a few 10 mHz are the spectral signature of the satellite clanks as described in reference [25].

6.5. Attitude control performance

Figure 17 shows the measured angular acceleration about the $Z_{\text{inst}}$-axis. The $f_{\text{spin}}$ frequency peak is due to the attitude guidance: the orbit being heliosynchronous, the spin axis $Y_{\text{inst}}$ (i.e. $\sim X_{\text{sat}}$) follows the orbital plane drift (0.98 deg/day, i.e. $\sim 0.2 \mu$rad s$^{-1}$) so that the angular acceleration is modulated at $f_{\text{spin}}$ to $3.7 \times 10^{-9}$ rad s$^{-2}$. If the spin axis was strictly inertial, the $X_{\text{inst}}$ science axis would leave the orbital plane by at least 4 deg at the end of an 8 days session: then this angle becomes too high to be considered as small-angle in the equations used to establish high-level requirement tree [1].

With 120 orbits, the integration time helps to have a good rejection of the stochastic noise to $5.64 \times 10^{-12}$ rad s$^{-2}$ at $f_{\text{EP}}$ (figure 17). The bump at 0.2 Hz is due to linear-to-angular coupling. The bunch around 1.5 Hz and the peak at 1 Hz are residual frequency aliasing from the sampling process at 4 Hz. Bunches were reduced by lowering the instrument servo-loop cut-off frequency, even if there was no evidence of aliasing of higher frequencies at $f_{\text{EP}}$. 
Figure 17. Session 256, spectrum (FFT) of the angular accelerometer output about $Z_{\text{inst}}$ axis, i.e. $\Psi_{\text{inst}}$. The upper inset represents a zoom of the spectrum curve around $f_{EP}$. The lower inset represents a schema of the test-mass with its reference frame.

Figure 18. Session 256, spectrum (FFT) of attitude restitution (residual with respect to the guidance profile) in the satellite frame about $X_{\text{sat}}$, $Y_{\text{sat}}$ and $Z_{\text{sat}}$ from top to bottom.

6.6. Attitude ancillary data performance

The scientific exploitation of every session needs the attitude ancillary data [17]. The CNES ORAMIC tool [30] provides this ‘attitude file’ composed of the precise attitude, but also of the angular rate and acceleration. The precise attitude is computed after the day following the end of the session. Attitude and associated products (expertise report, error assessment) are delivered to science mission center the following week. The inputs of ORAMIC are the quaternions from the two star-tracker camera heads and the angular accelerations from T-SAGE. A particular feature of the algorithms implemented in ORAMIC is that they focus on performance at $f_{EP}$
frequency and harmonics. They can be sub-optimal at frequencies of low interest. Each type of satellite mode and thus each type of session has its own algorithm. One star-tracker head or two are selected in association to the considered test-mass for a posteriori restitution. On one hand, T-SAGE’s angular accelerations go through a double integration and on the other hand

| Table 8. Session 256, output of ORAMIC software for DFACS performance. |
|-----------------------------------------------------------|
| $F_{ep}$ (mHz) | 3.11134443 |

| Session 256 DFACS performance |
|-----------------------------|
| $X_{sat}$ | $Y_{sat}$ | $Z_{sat}$ |
| Residual linear acceleration (m s$^{-2}$) |
| DC | $1.53 \times 10^{-12}$ | $9.14 \times 10^{-12}$ | $1.96 \times 10^{-12}$ |
| @ $f_{ep}$ | $1.12 \times 10^{-14}$ | $2.19 \times 10^{-13}$ | $2.04 \times 10^{-13}$ |
| @2 $f_{ep}$ | $7.31 \times 10^{-12}$ | $1.39 \times 10^{-11}$ | $6.34 \times 10^{-12}$ |
| @3 $f_{ep}$ | $1.60 \times 10^{-12}$ | $2.31 \times 10^{-12}$ | $5.67 \times 10^{-12}$ |
| Attitude error (rad) |
| DC | $3.48 \times 10^{-6}$ | $1.59 \times 10^{-6}$ | $1.88 \times 10^{-6}$ |
| @ $f_{ep}$ | $1.92 \times 10^{-8}$ | $3.17 \times 10^{-7}$ | $3.18 \times 10^{-7}$ |
| @2 $f_{ep}$ | $8.25 \times 10^{-6}$ | $2.35 \times 10^{-7}$ | $2.61 \times 10^{-7}$ |
| @3 $f_{ep}$ | $2.71 \times 10^{-8}$ | $3.16 \times 10^{-7}$ | $3.27 \times 10^{-7}$ |
| Angular rate error (rad s$^{-1}$) |
| DC | $1.61 \times 10^{-9}$ | $2.06 \times 10^{-9}$ | $3.00 \times 10^{-9}$ |
| @ $f_{ep}$ | $1.98 \times 10^{-10}$ | $2.84 \times 10^{-10}$ | $2.61 \times 10^{-10}$ |
| @2 $f_{ep}$ | $3.19 \times 10^{-7}$ | $2.27 \times 10^{-7}$ | $4.14 \times 10^{-9}$ |
| @3 $f_{ep}$ | $1.40 \times 10^{-9}$ | $2.16 \times 10^{-10}$ | $1.49 \times 10^{-10}$ |
| Angular acceleration error (rad s$^{-2}$) |
| DC | $6.28 \times 10^{-9}$ | $8.03 \times 10^{-11}$ | $1.17 \times 10^{-10}$ |
| @ $f_{ep}$ | $7.48 \times 10^{-13}$ | $5.64 \times 10^{-12}$ | $5.21 \times 10^{-12}$ |
| @2 $f_{ep}$ | $1.28 \times 10^{-8}$ | $3.05 \times 10^{-10}$ | $5.21 \times 10^{-10}$ |
| @3 $f_{ep}$ | $8.23 \times 10^{-11}$ | $9.65 \times 10^{-12}$ | $1.34 \times 10^{-11}$ |
| Attitude associated uncertainly (rad) |
| DC | $1.42 \times 10^{-5}$ | $9.96 \times 10^{-6}$ | $3.59 \times 10^{-5}$ |
| @ $f_{ep}$ | $1.76 \times 10^{-6}$ | $6.90 \times 10^{-7}$ | $1.11 \times 10^{-6}$ |
| @2 $f_{ep}$ | $2.25 \times 10^{-6}$ | $6.41 \times 10^{-8}$ | $9.00 \times 10^{-8}$ |
| @3 $f_{ep}$ | $1.69 \times 10^{-6}$ | $6.91 \times 10^{-7}$ | $1.10 \times 10^{-6}$ |
| Angular rate associated uncertainty (rad s$^{-1}$) |
| DC | $2.09 \times 10^{-9}$ | $2.60 \times 10^{-10}$ | $5.16 \times 10^{-10}$ |
| @ $f_{ep}$ | $1.92 \times 10^{-9}$ | $2.03 \times 10^{-9}$ | $3.85 \times 10^{-11}$ |
| @2 $f_{ep}$ | $4.09 \times 10^{-9}$ | $5.15 \times 10^{-10}$ | $8.43 \times 10^{-10}$ |
| @3 $f_{ep}$ | $5.67 \times 10^{-10}$ | $2.00 \times 10^{-9}$ | $3.75 \times 10^{-11}$ |
| Angular acceleration associated uncertainty (rad s$^{-2}$) |
| DC | $8.17 \times 10^{-11}$ | $1.02 \times 10^{-11}$ | $2.02 \times 10^{-11}$ |
| @ $f_{ep}$ | $3.91 \times 10^{-11}$ | $3.95 \times 10^{-11}$ | $7.04 \times 10^{-13}$ |
| @2 $f_{ep}$ | $1.72 \times 10^{-10}$ | $2.46 \times 10^{-11}$ | $4.63 \times 10^{-11}$ |
| @3 $f_{ep}$ | $3.84 \times 10^{-11}$ | $4.19 \times 10^{-11}$ | $3.48 \times 10^{-13}$ |
star-tracker quaternions follow a double derivation process. Both information are combined and defines what we call hybridisation. T-SAGE outputs are kept for most of the spectrum, the star-tracker ones are used at low frequency (< 0.42 mHz in the example of session 256) and at particular frequencies like $f_{\text{spin}}$ in orbit plane (unobservable) or at $2f_{\text{EP}}$ (gravity gradient coming from the residuals non-sphericity of the test-mass).

Figure 18 shows the spectrum of the estimated precise attitude (in $\mu$rad with respect to the guidance profile) for session 256. In the spectrum plot, the sudden changes of magnitude are due to a modification of the weighting between the star tracker and the accelerometers outputs in the frequency domain.

### 6.7 Summary of DFACS performances for session 256

Table 8 presents a summary of DFACS performances over the 120-orbit SpinMax session 256, obtained by the ORAMIC tools described in section 6.6. They comply with the requirements with good margins. The uncertainties are calculated using redundancy of information on angular axes. It can be considered a good session even if the density of spikes is high.

The gas consumption is 392.9 g for $+Z_{\text{sat}}$ (Zp) panel and 392.6 g for $-Z_{\text{sat}}$ (Zm) panel (about 3.3 g/orbit/panel), mainly used to compensate gyroscopic torques.

### 7. Conclusion

The MICROSCOPE satellite flew from April 2016 to October 2018. The performance of the satellite and of the AOCS subsystems were much better than expected. The satellite has been developed to fulfill unprecedented requirements at the level of a micro-satellite line. This is the first time for a drag-free system to achieve a pico-g environment ($10^{-12}$ level in linear and angular acceleration SI units) in low Earth orbit. The flexibility to modify in ‘real time’ the flight configuration helped the science team to use this satellite as a physics laboratory in space. This largely contributed to the success of this physics experiment.

Tools have been developed by CNES to deliver a precise orbit and attitude restitution to the Science Mission Center based at ONERA. These data have been used to take accurately into account the effects of the Earth’s gravity and of the rotation of the satellite in the science data process. For each scientific session (calibration or EP test), the performance of the DFACS has been also delivered to establish the systematic error budget in each session. To establish the performance in terms of extraction of the Eötvös parameter, it is needed to consider instrument scale factor matching and misalignment for each session. Indeed the performance of the satellite is seen in common mode by the instrument. In reference [1, 5], the mission performance is thus detailed considering all differential defects. In particular in reference [5] the budget of systematic error was performed for each session and shows that the DFACS error contribution is less than a few $10^{-16}$ m s$^{-2}$ to the differential acceleration measurement to be compared to the $7.9 \times 10^{-15}$ m s$^{-2}$ mission objective (i.e. $10^{-15}$ on the Eötvös parameter).

At last, this experiment analysis is the first step to start envisaging future missions with better performances [31].

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Table A1. Acronyms and abbreviations.

| Item    | Signification                                      |
|---------|----------------------------------------------------|
| AOCS    | Attitude and orbit control system                  |
| BCU     | Payload module                                     |
| CGPS    | Cold gas propulsion system                         |
| CGPSS   | Cold gas propulsion sub-system                     |
| CoG     | Center of gravity                                  |
| COTS    | Commercial off-the-shelf                           |
| DC      | Direct continuous                                  |
| DFACS   | Drag-free and attitude control system              |
| ECM     | Electronic control module of the CGPSS             |
| EP      | Equivalence principle                              |
| FDIR    | Fault detection, isolation and recovery            |
| FEEU    | Front end electronic unit of T-SAGE                |
| FFT     | Fast Fourier transform                             |
| GNSS    | Global navigation satellite system                 |
| GPS     | Global position system                             |
| ICUME   | Interface control unit mechanical ensemble of T-SAGE|
| IDEAS   | Innovative DEorbiting aerobrake system             |
| LVC     | French acronym for central flight software (logiciel de vol central) |
| MAG     | Magnetometer                                       |
| MFS     | Mass flow sensor of the CGPS                       |
| MT      | Micro-thurster                                     |
| MTB     | Magneto torque bars                                |
| MIL     | Multi layer insulator                              |
| OD      | Orbit determination                               |
| PCDU    | Power conditioning and distribution unit           |
| RMS     | Root mean square                                   |
| SAS     | Sun acquisition sensors                            |
| SU      | Sensor unit                                        |
| SUEP    | Sensor unit for the equivalent principle test       |
| SUREF   | Sensor unit for reference                          |
| TM/TC   | Telemetry/telecommand                              |
| T-SAGE  | Twin space accelerometer for gravity experiment    |
| WEP     | Weak equivalence principle                         |

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Data availability statement

The data that support the findings of this study are available upon reasonable request from the authors.
Appendix A. List of acronyms, abbreviations and notation

See table A1.

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