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MICROSCOPE Thermal Control Design and First In-Orbit Thermal Control Performance Results

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MICROSCOPE (« MICRO-Satellite à traînée Compensée pour l'Observation du Principe d'Equivalence », Drag-Free Microsatellite for the Test of the Equivalence Principle) is a microsatellite of the CNES Myriade series which includes Demeter, Parasol, Picard, Spirale, Essaim and Elisa missions (13 other satellites). MICROSCOPE was successfully launched on April 25th, 2016, from Kourou, French Guiana, on the Soyuz launcher. The orbit of MICROSCOPE is a Sun-synchronous orbit at an altitude of 710 km. The main scientific objective is testing the Equivalence Principle (EP) with a 100 times better accuracy than realized with experiments on Earth. The MICROSCOPE space mission aims at testing the universality of free fall with accuracy better than 10−15. The MICROSCOPE experiment requires very stringent thermal stability (in a low Earth orbit environment). The stability shall be better than 1 mK at f_EP (EP frequency) for the differential micro-accelerometers and 10 mK at f_EP for the electronic units. In addition, in order to reduce any perturbation, thermal control needs to be exclusively passive. Active heaters are not permitted during the payload science operations. Moreover the required thermal stability of the payload highly constrains the spacecraft design and the mission. The payload is uncommonly accommodated at the center of the spacecraft, whereas MYRIADE microsatellites usually have their payload on an external wall. While using mainly recurrent Myriade platform equipment, the propulsion module is quite specific: it is defined to provide a very steady environment to the experiment and a fine control of its attitude and of its drag-free motion along the orbit. This paper presents an overview of the thermal design of MICROSCOPE and the preliminary in-orbit results. In-Orbit Thermal Performance shows that the thermal control system has successfully met the thermal stability requirements. Thermal stability measured in-orbit in inertial mode is 4mK at electronic unit interface (to be compared to a requirement of 10 mK) and 0.5 mK at accelerometer interface (to be compared to a requirement of 1mK).

Nomenclature

AACS	=	Acceleration and Attitude Control System
CGPS	=	Cold Gas Propulsion System
CoG	=	Centre of Gravity
DOD	=	Depth of Discharge
EP	=	Equivalence Principle
ESA	=	European Space Agency
f_EP	=	Frequency of Equivalence Principle
f_EPi	=	Frequency of Equivalence Principle in inertial mode
f_EPs	=	Frequency of Equivalence Principle in spinning mode
f_ORB	=	Orbital frequency
f_SPIN	=	Frequency of spin mode
FFEU	=	Front End Electronic Unit

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ICU	=	I/F Control Unit
LEO	=	Low Earth Orbit
LEOP	=	Launch and Early Phase Operations
MLI	=	Multi-Layer Insulation
MQTh	=	Qualification Thermal Model ("Modèle de qualification thermique")
MT	=	Micro Thrusters
OBC	=	On Board Computer
ONERA	=	French National Aerospace Research Center ("Office National d'Etudes et Recherches Aérospatiales")
PCDU	=	Power Conditioning and Distribution Unit
P/F	=	Platform
S/C	=	Spacecraft
SAGE	=	Space Accelerometer for Gravity Experiment
SSM	=	Second Surface Mirror
SU	=	Sensor Unit
TCS	=	Thermal Control Subsystem
TRP	=	Temperature Reference Point
T-SAGE	=	Twin Space Accelerometer for Gravity Experiment

I. Introduction

MICROSCOPE, a microsatellite of CNES Myriade series, was successfully launched on April 25th, 2016, from Kourou, French Guiana, on the SOYUZ launcher.

In the overall organization, CNES is the lead agency in charge of the microsatellite (system, test and integration) and the S/C operations. ESA is responsible for the procurement of the Cold Gas Propulsion System (CGPS). Mission and data analysis functions are performed in close collaboration with ONERA and OCA (Observatoire de la Côte d'Azur).

The purpose of MICROSCOPE mission is to conduct a fundamental physics experiment testing the Equivalence Principle (EP) which postulates that a perfect proportionality exists between the inertial mass and the gravitational mass of a body, whatever its chemical composition. Resulting from the Equivalence Principle is the 'Universality of Free Fall' which states that all objects fall with exactly the same acceleration in the same gravitational field.

The Equivalence Principle is a major part of Albert Einstein's Theory of Relativity and experiments on Earth have so far returned the best accuracy value of EP at 10^{-13}. The main scientific objective of MICROSCOPE is to improve the value by two orders of magnitude (i.e. 10^{-15}).

In MICROSCOPE experiment, the Earth is the gravitational source about which free fall motion of two masses (named prof-masses), composed of different materials, is observed and controlled taking care that both masses are submitted exactly to the same gravitational field. The controlled electrostatic field that forces the masses to remain on the same orbit is accurately measured. A defect of symmetry gives rise to evidence of an EP violation.

MICROSCOPE uses ultra-sensitive accelerometers, micro thrusters, and drag-free flight in order to obtain its measurements. The satellite plays the role of a space laboratory devoted to a very complex experiment of physics, and for this, it must point the instrument on 3 axis, protect it against non-gravitational forces, and ensure an ultra-stable thermal environment, in particular around the frequency f_EP which is the frequency of excitation of the equivalence principle.

II. Spacecraft Overview

MICROSCOPE mission is developed in the frame of scientific missions exploiting the CNES MYRIADE micro-satellite product line. Therefore MICROSCOPE structural concept is directly derived from the Myriade product line platform.

The structure is composed of six rectangular sandwich panels made of aluminum skin with honeycomb aluminum core. However MICROSCOPE is bigger and heavier than the standard Myriade satellites:

- The satellite mass at liftoff is 303 kg.
- The satellite is cube-shaped (1.4 x 1 x 1.5 meters) when in its launch configuration.

The design of Myriade satellite is a compromise between high performance, efficiency, robustness and cost. The architecture of the satellite is based on a platform with generic functional chains (energy, communication, computer, structure…), and on a decoupled payload commonly located on the upper part of the platform structure.

MICROSCOPE mission specifications (i.e. CoG centering and thermal stability) require a modification of the MYRIADE satellite design and the Payload Assembly Subsystem is uncommonly accommodated at the center of the spacecraft (see Figure 1). This accommodation permits to have the center of gravity of satellite close to the differential micro-accelerometers and to have an ultra-stable thermal environment for the Payload. So MICROSCOPE thermo-mechanical architecture is mainly driven by payload constraints.

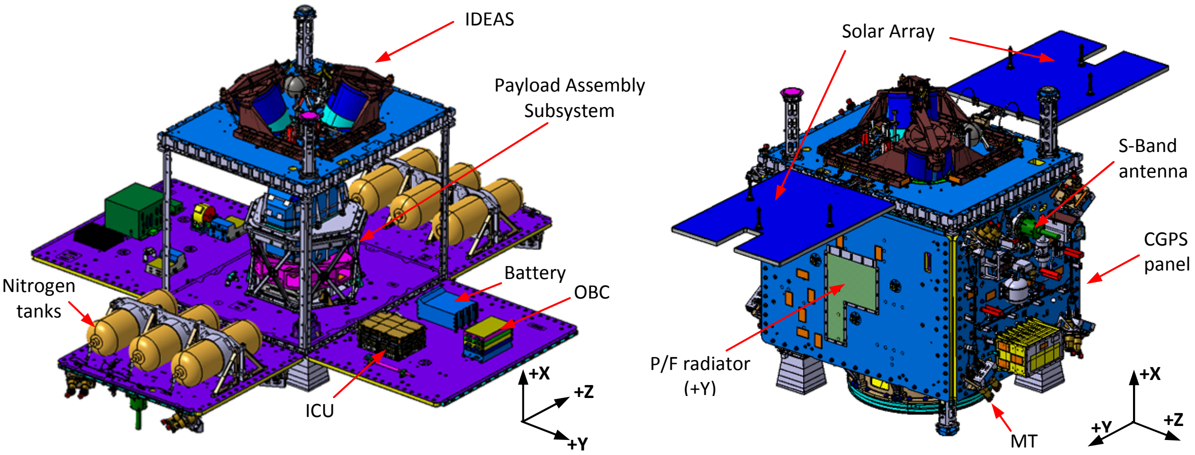

Figure 1. MICROSCOPE satellite internal layout (left) and external layout (right). MLI is not shown.

Equipment internal accommodation has been optimized in order to balance the mass and the platform radiator size (by balancing dissipated power) between –Y and +Y panels. This layout minimizes the thermal perturbation coming from the platform to the Payload Assembly Subsystem. In fact, if the S/C radiators (on –Y and +Y panels) are similar in size, the Earth IR fluxes absorbed at f_ORB (the orbital frequency) are in phase opposition and balanced.

The Payload Assembly Subsystem is the structure supporting T-SAGE (Twin Space Accelerometer for Gravity Experiment). T-SAGE is the combination of the two differential accelerometers designed and developed by ONERA for MICROSCOPE mission. Each SAGE is composed by a Sensor Unit (SU), a Front End Electronic Unit (FEEU) and an I/F Control Unit (ICU). ICU is installed on a panel of the platform (+Y panel).

The satellite shall protect T-SAGE from all non-gravitational forces perturbing the EP measurement. Thus an active control of accelerations and attitudes of satellite is necessary. This specific function of MICROSCOPE is named Acceleration and Attitude Control System (AACS). It needs a very sensitive 6 axis sensor and very fine actuators. Since there is no better instrument than an inertial sensor to measure the external forces, this function uses SAGE (Space Accelerometer for Gravitation Experimentation) as main sensor of the AACS control loop. On the other side of the process, a very fine actuator is needed to counteract the perturbations.

MICROSCOPE uses the CGPS (Cold Gas Propulsion System) to provide a very steady environment to the experiment and a fine control of its attitude and of its drag-free motion along the orbit. The CGPS is made of eight micro thrusters (MT), distributed in pairs on four opposite corners of the satellite and using gaseous nitrogen as a propellant to produce the few micronewtons of thrust required by the AACS. Six tanks are symmetrically accommodated in order to avoid self-gravity disturbances on the masses. CGPS has to counteract in all directions all non-gravitational forces acting on the satellite, like residual atmospheric drag, electromagnetic forces and pressure exerted by the Sun (radiation pressure).

Since the few micronewtons of thrust of CGPS are not sufficient for the end of life desorbitation of the satellite, a specific new sub-system of desorbitation was developed for MICROSCOPE. This desorbitation subsystem (IDEAS) is accommodated on +X panel. It is based on two sails deployed at the end of the mission in order to increase the surface/mass ratio.

III. Mission

A Sun synchronous circular orbit has been selected with 18h00 local time at the ascending node. This enables to keep the satellite in a steady orientation with respect to the Sun which optimizes solar panel power conversion, thermal stability and the thermoelastic behavior of the satellite structure.

Orbit altitude (710 km) is a compromise between several factors. Low orbits maximize the gravimetric signal and reduce the reentry time after the end of mission. High orbits reduce atmospheric drag and parasitic effects of the Earth on the spacecraft.

The mission needs to reduce any kind of internal micro-perturbations in the satellite, this term gathers different categories of perturbations: mechanical (i.e. direct forces applied on the prof-masses as the reaction to any displacement inside the spacecraft), gravitational (i.e. change of the induced gravity gradient when the satellite shape changes) and magnetic (i.e. resulting from the electrical activity in the satellite circuits and equipment).

Some of these perturbations may be much stronger during the eclipse season (three months a year) at the transition of the satellite into Earth’s shadow, inducing several phenomena: sudden solar pressure variation, MLI thermo elastic clank, non-regulated bus voltage variation, etc. For these reasons, the EP test should be performed only during the period of the year without eclipses.

The EP test is performed with satellite in quasi-inertial mode and in spinning mode (with a rotation around the X-axis). When the satellite remains in quasi-inertial attitude with the Sun in the direction normal to the orbital plane, the f_EP is equal to the orbital frequency f_ORB (i.e. f_EP = f_ORB). When the satellite is in spinning mode at the frequency f_SPIN (the satellite is rotating around the direction perpendicular to the orbital plane), the f_EP is equal to the sum of the spin frequency and the orbital frequency (i.e. f_EP = f_ORB + f_SPIN).

The objective is to perform the EP measurement several times with different values of f_EP in order to evaluate the influence of signal processing method on the results.

The quasi-inertial pointing attitude is the most constraining concerning the thermal stability requirements. In fact, the frequency of variation of major thermal perturbations (i.e. Earth IR heat flux on S/C radiators and albedo heat flux on payload radiator) is the orbital frequency f_ORB and so f_EP.

The spinning mode has the advantage to reduce thermal perturbations at f_EP frequency, larger than f_ORB.
All the contributors that have an influence on the accuracy of EP measurement (instruments, AACS, magnetism, local gravity, orbit, thermal stability...) are taken into account in an “error budget”. It will be noticed that, for quasi-inertial sessions, the major deterministic contributor in the error budget is the thermal stability.

The mission duration is 2 years, due to the limited embarked quantity of propulsion gas.

IV. MICROSCOPE Thermal Design

A. Spacecraft Thermal Design

The thermal control subsystem (TCS) is responsible for maintaining MICROSCOPE equipment units within their specified temperature limits and for meeting the thermal stability requirements during scientific mission phases.

MICROSCOPE has a very challenging thermal design including a tight temperature stability requirement for the instrument: the required stability at f_EP in inertial mode is 1mK at SU interface, 10 mK at the FEEU interface.

This very stringent thermal stability requirement of payload highly constrains the spacecraft design and its thermal control. Moreover active heaters are not permitted during the payload science operations (neither on spacecraft nor on payload) in order to avoid any interference with the payload measurements. Consequently thermal control on the satellite purely relies on passive methods, transporting excess heat from the internal electronic components to external radiators installed on the satellite side panels (+/-Y).

The thermal control subsystem employs passive thermal control techniques such as black painted interior to maximize internal radiative heat transfer, SSM radiator area on external surfaces of panels to reject heat dissipated on board, external multi-layer insulation (MLI) to minimize heat losses from other external surfaces as well as internal MLI to minimize radiative heat transfer (especially for the Payload Assembly Subsystem). The outer layer of external MLI is 50µm Kapton with Mapatox-K (MAP), a protective coating against atomic oxygen.

Maximum power budget is around 94W in mission mode (i.e. the mode allowing the EP measurement). The power dissipated by the equipment accommodated on the S/C panels is about 80W. The power dissipated by the equipment accommodated in the Payload Assembly Subsystem is about 14 W (7W for each electronic unit).

The active thermal control is made on 21 software controlled heater lines (2 lines are dedicated to the payload and 19 lines are dedicated to the platform). These lines are used during survival mode or payload non-operational phases for maintaining the equipment within their design temperature ranges.

B. Payload Assembly Subsystem Thermal Design

The thermal stability of the accelerometers and electronic units is a key and driving thermal requirement.

For this reason, a specific Payload Assembly Subsystem has been defined to achieve the temperature stability requirements of SU and FEEU.

The Payload Block weighs approximately 50 kg. Its diameter envelope is 52 cm and its height 55 cm.

The Payload Assembly Subsystem is an optimized set from the mechanical and thermal point of view, to guarantee to the accelerometers (SU) and to their electronic units (FEEU), at the same time a good structural behavior during the vibrations of the launch and a great thermal stability necessary to scientific measurements.

In order to guarantee the 2 required levels of thermal stability (1 mK for the SUs and 10 mK for the FEEUs at f_EP), the Payload Assembly Subsystem must be insulated from the platform and the external environment, which constitute thermal sources of disturbances. It is radiatively insulated from the platform environment by a system of MLI blankets.

The Payload Assembly Subsystem is installed on the anti-Sun panel of the spacecraft (-X) to be able to use the high natural external flux stability of this S/C side along the 18-hour local time orbit, thus avoiding thermal cycling from Earth's radiative heat flux.

The structure of the Payload Assembly Subsystem is composed of 2 stages of insulation (see Figure 2):

- The first stage of insulation supports the electronic units, thermally linked to the outer radiator, and insulated itself from the satellite structure by six titanium alloy bars.
- The second stage of insulation supports the two accelerometers and the magnetic shielding, necessary to protect the SU from magnetic perturbations. It is mechanically linked and thermally insulated from the first stage by six titanium alloy bars.

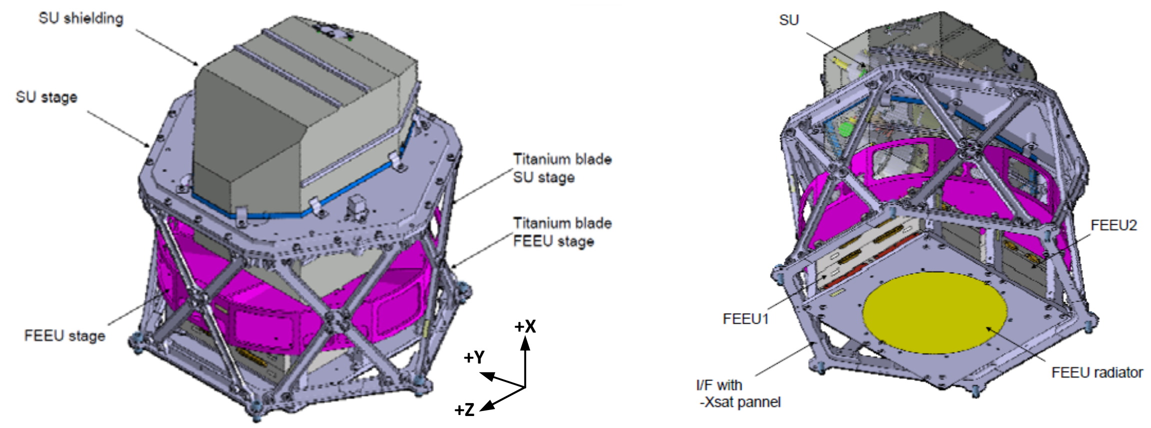

Figure 2. MICROSCOPE Payload Assembly Subsystem design.

The FEEU radiator is anti-Sun pointed (–X) and it permits the evacuation of the dissipated heat. The dissipated heat is about 7W for each electronic unit and it is negligible for the accelerometers. This radiator allows maintaining the electronic units within their operating temperature range (+10°C; +45ºC). The FEEU radiator is thermally coupled to the instruments by aluminum bars. The size of the FEEU radiator has been minimized in order to reduce the thermal perturbations coming from Earth. Consequently during operational phases the temperature of the FEEU and SU are maintained towards the top of their allowable temperature range. In worst hot case the temperature at FEEU TRP is +45°C (including 8°C margin).

An aluminum white painted baffle is installed around the FEEU radiator to limit the inputs of variable heat fluxes coming from the Earth, which would penalize the thermal stability of the FEEU. In fact, without this baffle, the varying surface temperature of the Earth's surface over the course of an orbit would cause too large temperature disturbance. The FEEU baffle is conductively decoupled from the FEEU radiator. Its diameter at the base is 25 cm, its height 20 cm and its opening half-angle is 25°.

Heaters have been sized to ensure that the minimum temperature limits are respected for the coldest conditions (survival mode). These heaters are mounted on the internal side of the radiator.

V. Thermal Control Development Philosophy

At payload level, a thermal qualification model (MQTh) of the Payload Assembly Subsystem was developed in the first phase of the project (2009) in order to prove the feasibility of a completely passive thermal control (i.e. without active regulation which would disturb scientific measurements) to reach the required performances of thermal stability, particularly ambitious.

The thermal qualification model needed to be representative in terms of thermal inertia, geometry, harness, MLI and radiative couplings. The Payload Assembly Subsystem was insulated with MLI and installed in a "mock-up satellite" in order to reproduce the flight configuration.

The thermal testing performed with the MQTh consisted in reproducing the thermal disturbances (representative of those awaited in flight) at the interfaces of the Payload Assembly Subsystem. The thermal disturbances were generated by powering test heaters (installed on the mock-up satellite and on the payload radiator) at the orbital frequency f_ORB. The effects of these thermal flight-like disturbances on the temperature stability of the electronic units and of the accelerometers were measured in different configurations.

The thermal testing has shown that the Payload Assembly Subsystem can meet the thermal stability requirement (1mK at SU interface, 10 mK at the FEEU interface). The feasibility of a completely passive thermal control was therefore successfully verified.

At the spacecraft level, a protoflight philosophy has been applied to MICROSCOPE, so the qualification of the thermal control was performed during satellite thermal vacuum testing (see Figure 3). MICROSCOPE completed thermal vacuum testing in November 2015.

Figure 3. MICROSCOPE thermal vacuum testing configuration.

The thermal balance/thermal vacuum test campaign has been highly successful. All S/C and payload functions were verified in cold and hot temperature levels. A good correlation of the thermal model has been achieved.

This correlated thermal model has been used to generate flight predictions for operating (hot and cold) and non-operating cases. The predictions for critical mission phases (like payload operational phase with passive thermal control in worst hot case) were successfully performed.

The flight predictions have shown that the thermal control system is able to maintain the satellite and payload within their design temperature ranges during the whole lifetime.

VI. Thermal Control Performance During LEOP

MICROSCOPE was successfully launched on April 25th, 2016, from Kourou on a SOYUZ launcher (flight VS14). As scheduled, MICROSCOPE was released on its orbit about 4 hours after liftoff. It must be noted that MICROSCOPE (the central auxiliary Payload) was not powered on the launcher.

This initial phase, from the liftoff to the separation, was carried out without any problem:

- The automatic deployment of the solar arrays and equipment units activation occurred successfully: the lower DOD of the battery was at 7.5% (comparing to maximum acceptable value of 80%).
- Thanks to the high inertia of the satellite, the separation (after 4 hours from liftoff) occurred at very favorable levels of temperature (about +15°C on P/F panels) in accordance with the thermal analysis.
- The convergence of the Attitude Control System (and so the orientation of the solar array panels to the Sun) was achieved more quickly than foreseen. The first heating line (the battery heating line) was activated few hours after the convergence.

Thermal Control performances during LEOP, by comparison between telemetries and flight predictions, are fully satisfactory: 34 of the 40 thermal sensors showed a discrepancy lower than 5°C (see Table 1). The higher discrepancy is limited to 6.8°C on Star Tracker (SST) TRP.

	Thermal sensors	**Temperature limits**		**Telemetry (°C)**	**Thermal Analysis Predictions (°C)**	**Discrepancy (Telemetry - Analysis)**
		Tmin (°C)	**Tmax (°C)**			
P/F	TRP_BR_PDU	-9.0	45.0	6.5	8.6	-2.1
	T_BOOST_PCU	-9.0	45.0	9.2	9.1	0.1
	TRP_BAT_1	4.0	40.0	16	16.4	-0.4
	TRP_BAT_2	4.0	40.0	16	16.4	-0.4
	TRP_OBC	-20.0	45.0	4.6	6.4	-1.8
	TRP_ELEC_SST	-25 (OFF)	+55 (OFF)	5.7	6.4	-0.7
	TRP_SSTYP1	-65 (OFF)	+75 (OFF)	-41	-34.7	**-6.3**
	TRP_SSTYP2	-65 (OFF)	+75 (OFF)	-39	-34.7	-4.3
	TRP_SSTYM1	-65 (OFF)	+75 (OFF)	-37.8	-31.0	**-6.8**
	TRP_SSTYM2	-65 (OFF)	+75 (OFF)	-37.8	-31.0	**-6.8**
	TRP_LANCEUR	-60.0	60.0	-20	-17.3	-2.7
	TRP_GNSS	-35 (OFF)	+55 (OFF)	6.6	5.9	0.7
	TIF_ROUE	-11 (ON)	+45 (ON)	2.8	3.9	-1.1
	TIF_RXTX1	-35 (ON)	+45 (ON)	1.5	0.0	1.5
	TIF_RXTX2	-35 (ON)	+45 (ON)	1.4	0.9	0.5
	TRP_RESIDEAS	-35.0	55.0	17.1	13.4	3.7
	TRP_BTCU	-35 (OFF)	+55 (OFF)	13.2	11.4	1.8
	TRP_MAG	-20 (ON)	+55 (ON)	10.5	10.1	0.4
CGPS	TRP1_ECMZP	-34 (OFF)	+55 (OFF)	-17.2	-17.4	0.2
	TRP2_ECMZP	-34 (OFF)	+55 (OFF)	-16.6	-17.4	0.8
	TRP_TRM_1_ZP	-25 (OFF)	+55 (OFF)	8.6	3.6	**5.0**
	TRP_TRM_2_ZP	-25 (OFF)	+55 (OFF)	10.5	5.1	**5.4**
	TRP_TRM_3_ZP	-25 (OFF)	+55 (OFF)	-2.8	-2.3	-0.5
	TRP_TRM_4_ZP	-25 (OFF)	+55 (OFF)	-3.6	-1.5	-2.1
	TRP1_ECMZM	-34 (OFF)	+55 (OFF)	-17.9	-18.4	0.5
	TRP2_ECMZM	-34 (OFF)	+55 (OFF)	-17.6	-18.4	0.8
	TRP_TRM_1_ZM	-25 (OFF)	+55 (OFF)	8.5	3.0	**5.5**
	TRP_TRM_2_ZM	-25 (OFF)	+55 (OFF)	6.9	2.4	**4.5**
	TRP_TRM_3_ZM	-25 (OFF)	+55 (OFF)	-4.6	-2.7	-1.9
	TRP_TRM_4_ZM	-25 (OFF)	+55 (OFF)	-3.3	-3.8	0.5
PAYLOAD	TRP_ICUME1	-28 (OFF)	+55 (OFF)	-2.9	-3.6	0.7
	TRP_ICUME2	-28 (OFF)	+55 (OFF)	-3.3	-3.6	0.3
	TRP_FEEUEP	-25 (OFF)	+55 (OFF)	16	16.0	0.0
	TRP_FEEUREF	-25 (OFF)	+55 (OFF)	16	16.0	0.0
	TRP_PLAFEEU	-25 (OFF)	+55 (OFF)	17	16.2	-0.8
	TIF_YM_BCU	-40.0	40.0	-4	-3.3	0.7
	TIF_YP_BCU	-40.0	40.0	-5.6	-4.6	1.0
	T_RADFEEU	-40.0	40.0	10.7	11.8	1.1

Table 1. LEOP temperatures comparison between telemetry and flight predictions.

As expected, only 2 of the 21 heater circuits were activated: the battery heating line and the FEEU heating line. Power consumption discrepancy between flight predictions and in orbit measurement was limited to 3% both for battery line and FEEU line (see Table 2).

The LEOP phase (2 days of operations) was carried out successfully. At the end of the LEOP, the thermal control system was performing as expected. All units were within their temperature limits and the majority was very close to their predicted temperatures and heater powers.

	Power consumption (LEOP)		
Heating line	**Telemetry**	**Thermal Analysis Predictions**	**Discrepancy**
Battery line (H_BATT_N)	3.6W	3.7W	3%
FEEU line (H_FEEU)	15 W	14.6 W	3%

Table 2. LEOP power consumption of heating lines.

VII. Thermal Control Performance During Commissioning Phase

During the in-orbit commissioning phase the satellite was checked-out, calibrated and its performances verified. This phase was essential to ensure that the satellite, the payload, the drag-free controller (i.e. CGPS) and the ground segment perform as expected.

The commissioning was performed from April to mid November 2016. A 3 month period of pause (from June to the end of August) was scheduled because of eclipses (i.e. unstable thermal conditions) and because of some anomalies that occurred on star trackers, on CGPS electronic unit and on payload electronic unit. These anomalies were treated and solved without consequences on the mission.

By mid-November 2016 (more than six months after launch), the satellite has completed its in-orbit checkout and its commissioning phase.

MICROSCOPE satellite readiness status was achieved, allowing start of nominal scientific EP experiments.

Thermal Control performances during commissioning phase were fully satisfactory (see Table 3). Almost all thermal sensors showed a discrepancy between telemetries and flight predictions that not exceeded 5°C. As scheduled in commissioning operation plan, every nominal heating line was activated one at a time. All the heaters were operated for three minutes and the heater power measured. All heaters were within 10% of expected power.

The temperature requirements and the thermal control performances for platform, CGPS and payload were successfully met.

	Discrepancy (Telemetry - Analysis) (only discrepancy > 5°C are reported)			Synthesis
Satellite mode	**P/F Thermal sensors**	**CGPS Thermal sensors**	**Payload Thermal sensors**	
MNOG mode (9th Mai 2016)	/	/	/	**100% of thermal sensors shown discrepancy < 5°C**
MCAN mode (i.e. scientific mode) 15th June 2016	TRP_RESIDEAS → delta-T= +5.4°C (IDEAS TRP)	TRP_TRM_2_ZP → delta-T= +5.8°C TRP_TRM_1_ZM→ delta-T= +5.2°C TRP_TRM_2_ZM→ delta-T= +5.5°C (3 sensors at MT TRP)	/	**90% of thermal sensors shown discrepancy < 5°C**
MCAN mode (i.e. scientific mode) 29th August 2016	TRP_RESIDEAS → delta-T= +5.6°C (IDEAS TRP)	TRP_TRM_1_ZP → delta-T= +6.5°C TRP_TRM_2_ZP → delta-T= +6.8°C TRP_TRM_1_ZM→ delta-T= +7.0°C TRP_TRM_2_ZM→ delta-T= +5.5°C (3 sensors at MT TRP)	/	**87% of thermal sensors shown discrepancy < 5°C**

Table 3. Commissioning temperatures comparison between telemetry and flight predictions.

VIII. In-Orbit Thermal Stability Performance

As stated previously, thermal stability is a critical parameter for MICROSCOPE mission due to the high thermal sensitivity of the accelerometers and to their electronic units. As such, thermal stability is a major design driver for the satellite and for the mission.

Thermal stability measured in orbit in inertial mode et in spinning mode during commissioning phase is shown in Table 4 and Figure 4 for the electronic units and in Table 5 for the accelerometer.

As it can be observed, thermal stability in orbit is better than requirements:

- 4 mK at FEEU interface to be compared to a required stability of 10 mK (in inertial mode).
- 0.5 mK at SU interface to be compared to a required stability of 1 mK (in inertial mode).

The milli-Kelvin passive thermal control met successfully its requirements.

	Thermal stability at FEEU TRP	
	Requirement	**In flight results**
inertial mode	10 mK @ f_EPi	4 mK @ f_EPi
spinning mode	3 mK @ f_EPs	0.3 mK @ f_EPs

Table 4. Thermal stability at FEEU TRP.

	Thermal stability at SU TRP	
	Requirement	**In flight results**
inertial mode	1 mK @ f_EPi	0.5 mK @ f_EPi
spinning mode	1 mK @ f_EPs	0.1 mK @ f_EPs

Table 5. Thermal stability at SU TRP.

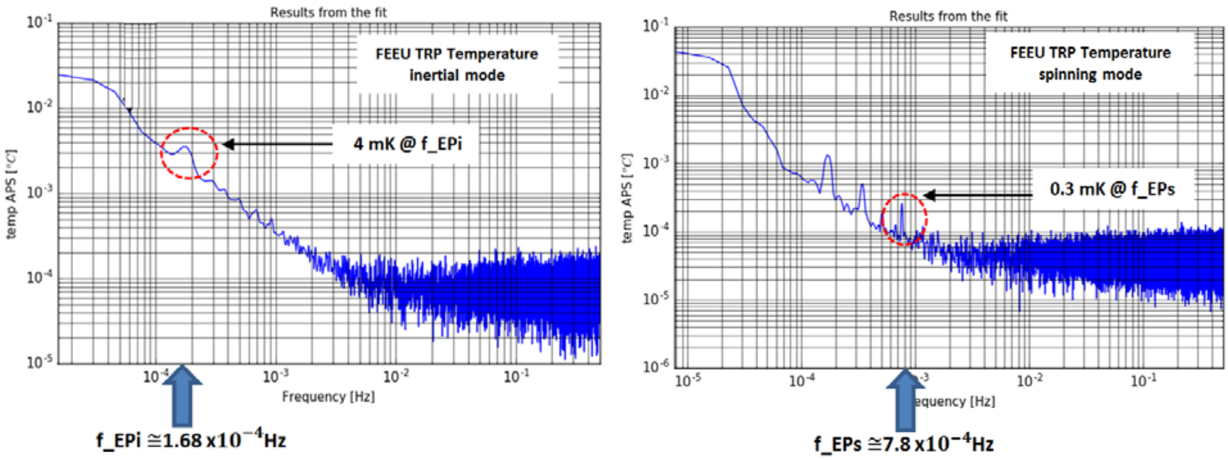

Figure 4. MICROSCOPE thermal stability at FEEU TRP.

IX. Conclusion

This paper outlines the in-orbit thermal control system performance results of MICROSCOPE, from LEOP to the end of Commissioning phase. Thermal Control performances are fully satisfactory. All units were within their temperature limits and the majority was very close to their predicted temperatures and heater powers.

The MICROSCOPE passive thermal control system has fully met its design requirements, providing a thermally stable environment for the sensitive payload. Thermal stability measured in-orbit in inertial mode is 4mK at electronic unit interface (to be compared to a requirement of 10 mK) and 0.5 mK at accelerometer interface (to be compared to a requirement of 1mK)

These results clearly illustrate that milli-Kelvin passive thermal control in low Earth orbit is possible.

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