In 2016, the Microscope satellite should be launched from Kourou by Soyouz and conclude 20 years of instrumental development in order to test the Equivalence Principle at levels of 10^{-15}. The instrument is composed of two differential accelerometers which compare the accelerations of two pairs of body in “free-fall” at 710 km altitude. As a founding principle of the Einstein’s theory of gravitation (General Relativity), the Equivalence Principle verification is a challenging target for most of the alternatives theory: quantum loops, string theory, dilatons, ...

This paper presents the current development status of the mission. It underlines also some specific results obtained from the qualification phase and the on-going development of the data process that should exhibit the mission performance objective.

1 Overview of the mission

The Equivalence Principle (EP) expressed by Einstein as a basis of its theory of General Relativity stipulates that all bodies, independently of their mass or intrinsic composition acquire the same acceleration in the same uniform gravity field. It was tested throughout the years by ground-based experiments with an increasing accuracy which reaches a few 10^{-13} by the Eötvös Group 1. The accuracy of this ground experiment is in particular limited by the local gravity gradients and its fluctuations, by the temperature and the magnetic field variations. The two papers in reference describe some clues for order of magnitude improvement in the future. The currently obtained best results are the following:

<table>
<thead>
<tr>
<th>Tests</th>
<th>Ref</th>
</tr>
</thead>
<tbody>
<tr>
<td>Torsion balance:</td>
<td></td>
</tr>
<tr>
<td>((\Delta a/a)_{\text{BeTi}}) = (0.3 \pm 1.8) \times 10^{-13}</td>
<td>1</td>
</tr>
<tr>
<td>((\Delta a/a)_{\text{BeAl}}) = (-0.7 \pm 1.3) \times 10^{-13}</td>
<td></td>
</tr>
<tr>
<td>Lunar Laser Ranging:</td>
<td></td>
</tr>
<tr>
<td>((\Delta a/a)_{\text{EarthMoon}}) = (-0.8 \pm 1.3) \times 10^{-13}</td>
<td>2</td>
</tr>
</tbody>
</table>
In papers 3,4, authors explain how these values constrain the alternative standard models as the dilaton string theory and underline the necessity to go further. This is the motivation of the space MICROSCOPE mission that should improve by two orders of magnitude the best already performed tests. Increasing the accuracy to the ranges predicted by some alternative theories to the General Relativity is therefore crucial to confirm or infirm the equivalence between inertial and gravitational masses at the heart of the metric theories.

Going to space helps to reduce the environment disturbances of the experience, linked to the local gravity fluctuations. By using a drag-free satellite, the instrument is orbiting along a quasi-geodesic trajectory. The drag-free system controls also the torques that helps to finely tune the attitude motion of the satellite against gravity or magnetic torques.

Placed at 710 km of altitude during a mission period of 2 years, the satellite will fly in inertial pointing on Sun synchronous, quasi-polar orbit most of the time as depicted in Figure 1. The Earth constitutes the gravitational source of the experiment. For one measuring axis in the orbital plane, the Earth's gravity source has a relative cyclic effect (at orbital frequency, i.e $1.68 \times 10^{-4}$ Hz). A rotation about the orbital plane axis can be applied to the satellite in order to increase the measurement frequency to $7.59 \times 10^{-4}$ Hz or $9.27 \times 10^{-4}$ Hz. These rotations give different conditions of operation with respect to potential systematic error sources: thermal behaviour, pointing stability, magnetic effects...

\[ \eta = \frac{a_1 - a_2}{\frac{1}{2} (a_1 + a_2)} = \frac{mg_1 - mg_2}{\frac{1}{2} (mg_1 + mg_2)} \]

Where $a_1$ and $a_2$ are the accelerations of the two tested bodies, $mg_1$ and $mg_2$ their respective gravitational masses, $mi1$ and $mi2$ their respective inertial masses. In the frame of General Relativity, $\eta$ is assumed null as a fundamental principle of the theory leading to the universality of free-fall in a uniform gravity field: i.e. $g = a$.

The two test bodies constitute the test-masses of two inertial sensors or accelerometers accurately centered to feel the same gravitational field. The mean output of the accelerometers gives the mean acceleration of the satellite. It is used by the servo-loop of the drag compensation control system.
to accelerate the satellite against the residual air drag or the radiation pressure with the help of gas micro-thrusters. As the test-mass are controlled along their 6 degrees of freedom, the accelerometers also deliver the angular accelerations that are combined with the star sensors to finely control the satellite angular motion and pointing.

The 320 kg satellite uses cold gas thrusters adapted from the developed technology for the ESA GAIA mission to fulfil the required thrust range and noise, $300\mu N \& 0.001\mu N Hz^{-1/2}$.

The payload is placed in a thermal case enabling mK stability at EP frequency (in inertial pointing or satellite rotating mode). It contains the sensor core of the accelerometers (including the inertial test-masses) and the Front-End Electronics with all reference voltages and pick-up measurements circuits. The digital electronics unit embarking the test-masses control law (ICUME on Figure 2) is placed on a wall of the satellite with other service equipment.

![Figure 2 - Schema of the opened satellite and picture taken during integration (credits CNES)](image)

## 2 The instrument description

The instrument comprises two sensor units (see Figure 3). In each sensor unit, two cylindrical test-masses are servo-controlled to remain concentric in order to be submitted to the same gravity field. By construction the concentricity of the two test-masses is achieved at $20\mu m$ accuracy and calibrated in orbit at $0.1\mu m$ in order to reject the gravity gradient effect at EP frequency.

The relative alignment of the two test-masses is also required to a fraction of 1milirad in order to minimize the projection of the angular motion or the gravity gradient from the radial axis: the measurement of the differential acceleration due to an possible violation is performed along the cylinder axis direction (Figure1). To achieve these accuracies of centering and alignment, all the parts have been machined to less than 3 to $5\mu m$ accuracy and controlled to $1\mu m$ accuracy. The integration of all parts has been realized with different controls of the partially assembled instrument with the use of a 3D machine control up to $1\mu m$ accuracy. At last, once integrated, the capacitive measurements of all electrodes surrounding the test-masses allow confirming the electrical geometry and assessing all the accuracies.

One sensor unit is composed of two test-masses with the same material (Pt-Rho10) for which no differential acceleration is expected unless noise and systematic errors: this is for demonstration of the experiment accuracy. The other sensor unit embarks a pair of test-masses made of Pt-Rho10 and Ti alloy which are dedicated to the EP violation detection. Both sensor units will be submitted to the same orbital conditions, calibration and data processes. The metrology of the parts associated to the measured characteristics of the flight model electronics have been considered to establish a budget evaluation of noise and systematic error sources: in the order of $10^{-12} m/s^2 Hz^{-1/2}$ for the acceleration noise measurement and a few $10^{-15} m/s^2$ for the sum of systematics perturbations due essentially to the thermal variations of either the sensor mechanics or the electronics unit (respectively passively limited to 1mK and 5mK at EP frequency) and also to the S/C pointing stability (< 10µ rad at one and three times EP frequencies).

When considering the different S/C pointings and in orbit environment, and with no correlation
between the major error sources, the detectability of any violation signal of EP should be $10^{-15}$. The payload was delivered to CNES in October 2014 and has been integrated in the satellite which will undergo the qualification environment tests in the summer of 2015: chocks, vibrations and thermal vacuum tests. The flight is being scheduled to April 2016.

Figure 3 - T-SAGE flight model payload (Twin Space Accelerometer for Gravitation Experiment): on the left, the two sensor units (SU) aligned with optical cube; on the middle, the two front-end high stable electronics, each one measuring one SU; on the right, the ICUME, the digital controller of the test-mass servo-loops and data interface with the S/C.

3 The data process

Beyond the development of the payload and the mission definition, ONERA and OCA (Observatoire de la Côte d’Azur, Nice, France) have been developing the Science Mission Center located in ONERA premises. This center is in charge of the definition of the mission scenario and its update during the two years' mission. Actually, every day of the in orbit flight, this Mission Centre analyses the data coming from the payload for the diagnostic of any anomaly. Every week, in addition to the payload data, mission data are collected to establish the validity of the passed scenario and the necessity to update the future one to be sent to CNES for implementation in the S/C scheduling. All this operational loop needs the development of software tools which survey the behavior of the major on board subsystems. The scenario will be also led by the Science Working Group of the mission in a monthly time loop to optimize the scientific return of the space experiment.

\[
\Gamma_{\text{mes, } dx} = \frac{1}{2} K_{\text{icx}} \cdot \eta \cdot g_{\text{sat}} + \frac{1}{2} \left[ \begin{array}{c} K_{\text{icx}} \eta_{\text{ex}} + \theta_{\text{ex}} \\ \eta_{\text{ey}} - \theta_{\text{ey}} \end{array} \right] \cdot \left[ \begin{array}{c} \Delta x \\ \Delta y \\ \Delta z \end{array} \right] + \left[ \begin{array}{c} K_{\text{idex}} \\ \eta_{\text{dy}} - \theta_{\text{dy}} \end{array} \right] \cdot (\gamma_{\text{resdf}} + \gamma_{\text{c}}) + 2 \cdot K_{\text{2exx}} \cdot (\Gamma_{\text{app, dx}} + b_{\text{idex}}) \cdot (\Gamma_{\text{resdf, x}} + C_{\text{x}} - b_{\text{ox}}) + K_{\text{idex}} \cdot \left( (\Gamma_{\text{resdf, x}} + C_{\text{x}} - b_{\text{ox}})^2 + (\Gamma_{\text{app, dx}} + b_{\text{idex}})^2 \right) + \text{noise}
\]

Table 2: $g_{\text{sat}}$ is the local gravity; $K_{\text{icx}}$ the common mode scale factor, $K_{\text{idex}}$ the differential mode scale factor; $\Theta$ stands for misalignment (‘c index’ with respect to S/C reference frame, ’d index’ test-mass relative misalignment); $K_2$ quadratic term of the scale factor, $b_e$ and $b_d$ are instrument bias components (e or d index precise common or differential).

*During eclipse phase, the payload is switched off. Some days are also lost during S/C maneuvers in order to avoid the star sensors to be dazzled by the Moon light. At the end, the useful duration is 500 days.*
Concerning the measurement equation, let us consider the two measured accelerations, each one by inertial sensor (linked to each mass), and subtracted to form a differential measurement. As the accurate measurement is realized along the cylinder axes (noted X here), only the X component of the differential acceleration is first considered at EP frequency. In the following equation, the measurement is expressed in the test-mass reference frame and shows the main disturbing sources, that may affect the signal of the EP violation.

During the mission, dedicated sessions are used to calibrate the instrument. Calibration sessions consist of amplifying the disturbing acceleration by applying larger reference signals. For instance \(K_{ldx}\) is estimated by biasing the S/C (common) acceleration along x with a sine signal of \(10^{-8} \text{ms}^{-2}\) at a few \(10^{-3}\)Hz, observing or not residue. The equation of measure is then corrected as follows:

\[
\Gamma_{mes, dx} = \frac{1}{2} K_{1cx} \cdot \eta \cdot g_{x, sat} + \frac{1}{2} \begin{bmatrix} K_{1cx} \\ \eta_{cx} + \theta_{dx} \\ \eta_{cy} - \theta_{dy} \end{bmatrix}^t \cdot [T - I \eta] \cdot \begin{bmatrix} \Delta_x \\ \Delta_y \\ \Delta_z \end{bmatrix}^t - \frac{1}{2} \begin{bmatrix} K_{1cx} \\ \eta_{cx} + \theta_{dx} \\ \eta_{cy} - \theta_{dy} \end{bmatrix}^t_{\text{computed}} \cdot \begin{bmatrix} \Delta_x \\ \Delta_y \\ \Delta_z \end{bmatrix}_{\text{calibrated}} \\
+ \begin{bmatrix} K_{dxx} \\ \eta_{dx} + \theta_{dx} \\ \eta_{dy} - \theta_{dy} \end{bmatrix}^t \cdot (\Gamma_{resd, x} + C) - \begin{bmatrix} K_{1dx} \\ \eta_{dx} + \theta_{dx} \\ \eta_{dy} - \theta_{dy} \end{bmatrix}^t_{\text{calibrated}} \cdot (\Gamma_{mes, x}) \\
+ 2 \cdot K_{2cxx} \cdot (\Gamma_{app, dx} + b_{1dx}) \cdot (\Gamma_{resd, x} + C_x - b_{0cx}) \\
+ K_{2dxx} \cdot ((\Gamma_{resd, x} + C_x - b_{0cx})^2 + (\Gamma_{app, dx} + b_{1dx})^2) \\
+ \text{Noise}
\]

In case of no interruption in the telemetry, the integration of the measurement signal can be performed over at least 20 orbits reducing the impact of the stochastic error (in particular, the instrument noise) and the required performance for all the subsystems (instrument, star sensors, S/C thermal environment, S/C self-gravity, GPS, ...) is sufficient to achieve the mission performance objective of \(10^{-15}\).

Unfortunately, the quality of data could be affected by the loss of telemetry or more frequently by micro-disturbances due to micro-meteorites that will saturate the acceleration measurement (range of \(2.5 \times 10^{-7} \text{m/s}^2\)). In this last case, the data will be affected during several seconds after the choc generated by the micro-meteorites and must be eliminated. Unfortunately, the measurement power spectrum is dramatically modified and out of the specifications of the processing, even with few holes due to telemetry defects or micrometeorites. Several processes have been studied to cope with these missing data. In\(^5\) it is shown that a simple ordinary least square (OLS) is not sufficient: the mission performance is reduced by a factor 10 with only 5 holes per orbit (at 4Hz frequency sampling, it corresponds to about 0.08% of loss).

To overpass this difficulty, the authors of the paper \(^5\) developed two methods\(^5,6\). The first one relies on fitting the signal PSD with a high order autoregressive model using the Burg’s algorithm adapted to missing data. The inversion of the system is performed with a process based on a Kalman filter algorithm that gives at the end the eventual EP parameter, but also the other unknown, the miss-centering of the test-mass and the noise parameters. This method is called KARMA (for Kalman-AR model analysis). The results of this method are satisfactory with respect to the specifications and can be used to reconstruct the data in the missing interval as shown in Figure 4. Another method \(^6\) derived from the Impainting method \(^7\) has been developed and gives equivalent results. This last method uses the fact that the mask of missing data is known and tries to recover the real temporal data with the minimum of coefficient.

4 Conclusion

The MICROSCOPE space mission will test the Equivalence Principle with a \(10^{-15}\) accuracy, i.e two orders of magnitude better than current experiments. The payload has been qualified and delivered
in October 2014 to the CNES satellite team. The payload is now integrated in the satellite that will be submitted to the environmental tests before the launch scheduled to early April 2016. A lot of effort has been paid to develop the Science Mission Center and in particular to cope with missing data or saturated data that should be eliminated. Two processing methods have been developed to allow processing the scientific data with these defects. They give good results in accordance to the required objectives.

Acknowledgements

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