CNES(2001)DPI/E2U ESA-SCI(2001)7

Announcement of Opportunity

Science Contributions for

MICROSCOPE

(MICROSatellite à traînée Compensée pour
l'Observation du Principe d'Equivalence)

a CNES/ESA Collaborative Mission
to test the Equivalence Principle

21 December 2001

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1 Introduction and mission concept

1.1. Introduction

This announcement of opportunity solicits proposals from European scientists to participate in the MICROSCOPE mission, in the frame of a CNES/ESA co-operation, in view of:

- testing the Equivalence Principle (EP) between inertial mass and gravitational mass at a high level of precision, with a significant gain of 2 or 3 orders of magnitude compared to what is presently achieved on ground,
- demonstrating technological benefits concerning drag-free satellite control for a suite of future fundamental physics and astronomy missions,
- making optimum use of spacecraft resources by providing additional hardware which may benefit the mission in a wider context. Such contributions will be referred to hereafter as “stand alone instruments”.

MICROSCOPE is an approved CNES project, with a launch foreseen in 2005. It is the fourth CNES scientific project based on the MYRIADE microsatellite family. The name MICROSCOPE stands for MICROSatellite à traînée Compensée pour l’Observation du Principe d’Equivalence (Drag-Free Microsatellite for the Test of the Equivalence Principle). MICROSCOPE was selected by CNES in December 1999 on the basis of the proposed and accepted configuration for the payload, with Pierre Touboul (ONERA) as PI and Gilles Metris (OCA/CERGA) as co-PI. In 2000 and 2001, a mission analysis was performed and both the payload and the satellite definitions were concluded. They shall be reviewed in June 2002.

The project was proposed to collaboration with ESA, by responding to the ESA flexible missions F2/F3 call for proposals in January 2000. The involvement of ESA in the MICROSCOPE project was approved by the ESA Science Programme Committee on 11-12 October 2000 and the final ESA/CNES agreement was signed in June 2001. In this collaboration ESA will provide the Field Emission Electric Propulsion (FEEP) proportional micro-thrusters.

In return, the project is open to European co-operation on the basis of the already defined technical and scientific specifications. The «Core Programme» of the MICROSCOPE mission, which is also the basis of the definition of the prime payload and of the global mission, addresses the test of the Equivalence Principle. It is under the responsibility of the MICROSCOPE PI. Additional scientists from the ESA Member States who intend to make hardware, software or other scientific contributions to the «Core Programme» of the MICROSCOPE mission are jointly invited by CNES and ESA to send a proposal to this AO. This invitation also extends to the contribution of “stand alone instruments” as defined above.

Furthermore, specific operations of the drag-free satellite or of the payload could be undertaken after the success of the prime scientific experiment if the impact on the mission is acceptable. Scientists willing to propose such ideas are also invited to respond to the AO. The selected proposals will result in an «Extended Programme».

All the complementary scientific contributions must be compatible with the present design of the satellite and of the payload. They may be either contribution to the prime payload or additional stand-alone instruments.

The final proposals are due by 22 February 2002 and will be subject to an ESA/CNES joint selection.

1.2. Mission concept

The MICROSCOPE experiment is a Galileo free fall test with two masses composed of two different materials. These masses fly in orbit around the Earth at an altitude of about 700 km and are submitted to exactly the same gravitational field. A deviation from their respective orbital motion would hint to the violation of the Equivalence Principle EP.

The experiment shall be carried out by forcing the two test masses on exactly the same orbital motion by applying electrostatic fields between the freely floating masses and a rigid instrument housing that is attached to the structure of the spacecraft. Any asymmetries in the forces necessary to keep the masses positioned would be detected via accurate measurement of the applied electrostatic voltages. Asymmetries that would not be accountable for by instrumental errors would give evidence to the violation of the Equivalence Principle.

The gain in precision of two to three orders of magnitude compared to state of the art experiments will be realised by using a drag-free satellite, where non-gravitational forces applied on the satellite are compensated by the actuation of electrical thrusters. Such a regime substantially reduces the effects of gravity gradient fluctuations. Furthermore, long observation times of the free fall mass motion in steady conditions enable signal integration over days and thus the rejection of stochastic disturbances. The rotation of the observational frame with respect to the gravity field orientation, by spinning the satellite (spin mode) also helps in the discrimination of the eventual EP violation signal. Several rotation frequencies will be considered, the EP test will thus be performed at orbital frequency and at various other frequencies (see Figure 1).

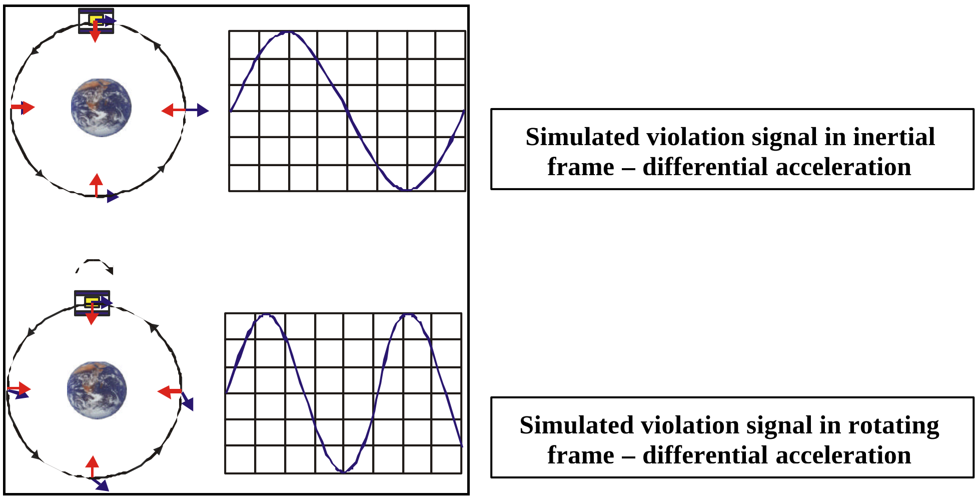

Figure 1: MICROSCOPE experiment: Shielded by the satellite, the two masses, made of different materials, fall around the Earth. They are submitted to the same gravity field and controlled along the same orbit. The potential EP violation signal in the inertial and rotating frame is shown.

The satellite payload is composed of two identical accelerometers, identical, except for their test mass materials. Each accelerometer is differential and includes two concentric cylindrical and coaxial test masses. These masses are made of the same material for the first accelerometer, which is dedicated to assess the accuracy of the instrument, and consist of different materials in the second accelerometer. In order to suppress systematic errors, the experiment relies on the double comparison of the output of the two pairs of electrostatic accelerometers. Their inertial masses are the test masses. The selection of the mass materials is a compromise between instrument accuracy requirements and theoretical interests.

The attitude as well as the atmospheric and thermal drag of the satellite will be actively controlled by a specific drag compensation and attitude control system in such a way that the satellite follows the two test masses in their gravitational motion. The major part of the one-year's mission duration will be dedicated to instrument calibration and to the determination of the instrument sensitivity to external environment disturbances like gravity gradients.

The satellite drag compensation involves Field Emission Electric Propulsion (FEEP). In addition to the measurement mode, when the satellite shields the instrument from Earth's and Solar radiation pressure and from atmospheric drag, this system allows fine calibration of the instrument by generating well known cinematic accelerations in all six degrees of freedom.

The MICROSCOPE mission exploits the MYRIADE microsatellite family developed by CNES. The satellite mass must be less than 120 kg with dimensions of about 800 x 600 x 600 mm^3 in order to be compatible with a low cost launcher such as the ASAP auxiliary payload of ARIANE 5 or small dedicated launchers.

The MICROSCOPE mission specifications require a modification of the MYRIADE satellite baseline in respect to fine motion, attitude control and thermal design. A Sun synchronous circular orbit has been selected with an altitude around 700 km (eccentricity lower than 10^{-2}) and with 6h00 or 18h00 local time at the ascending node. This enables to keep the satellite in a steady orientation with respect to the Sun which optimises solar panel power conversion, thermal stability and the thermo-elastic behaviour of the satellite structure.

2 Scientific objectives

The primary scientific objective of the mission is the test of the Equivalence Principle (EP) with an accuracy of 10^{-15}, which is about three orders of magnitude better than what can be achieved with present on-ground experiments. All recent laboratory experiments exploit the torsion pendulum technique and have to combat environmental instabilities, in particular the Earth gravity gradient fluctuations (see references 1-3). Earth-Moon laser ranging data have also been used lately to test the EP but the material composition of the two celestial bodies is not known sufficiently enough (ref. 4) to interpret the results.

Confirming the equivalence between inertial mass and gravitational mass to one part in 10^{15} would provide an important verification of the relativistic theory of gravitation and other metric theories, which postulate this principle. A confirmation should also further stimulate the interest in more accurate experimental or observational data on the Post Newtonian coefficients (ref. 5,6). On the other hand, the violation of the Equivalence Principle, which is an exact symmetry for General Relativity, would open the way to the demonstration of a new force, the existence of which is predicted by many quantum theories of gravity (ref. 7-9).

Approaches to quantum gravity, like for instance the Superstring theory, are intensively pursued at the moment. The existence of an additional massless scalar field superimposed to gravity would violate the EP. Such proposals need to be challenged by experimental evidence like EP test results.

The confirmation of the Equivalence Principle with an accuracy better than can be obtained now would pose strong constraints on unification theories. MICROSCOPE would then be the first attempt in space to search for direct evidence of new gravitational phenomena before the realisation of even more ambitious missions (ref. 10-12).

The proposed space experiment will, if not revolutionise the current theoretical approach to gravitation, nevertheless be an important and first step towards further EP tests, which then will demand cryogenic instruments.

MICROSCOPE will also be the first mission in Europe to test the technologies necessary for the development and the realisation of a drag-free satellite: accelerometers, electric thrusters and the necessary command laws and software. The study, realisation and in-orbit testing of a drag-free satellite are essential for the preparation of future Fundamental Physics missions like STEP (EP test) and LISA (observation of Gravity Waves). Also astronomy missions for the search of extra-solar terrestrial planets like DARWIN or projects in Earth Observation to precisely determine the gravity field of the Earth will employ drag free technology.

3 Mission implementation

3.1 Mission schedule

The basic development schedule for the satellite and the payload is the following:

Preliminary Design Review (PDR) for the Prime Instrument	18 Dec 2001
Interface Document for Stand Alone Instruments	May 2002
Satellite Phase A completion	June 2002
Engineering Model delivery (Prime Instrument)	Sept 2002*
PDR of Stand Alone Instruments	Oct 2002
Software User Requirement Document	March 2003
Critical Design Review (CDR) of Prime Instrument	March 2003
Satellite Phase B completion (PDR)	March 2003
Engineering Model delivery (Stand Alone Instruments)	May 2003
Satellite Phase C completion (CDR)	March 2004
Flight software delivery	March 2004
Flight hardware delivery	mid 2004
Launch	mid 2005

* This deadline does not apply to the potential provision of the test masses, for which an engineering model would be required only by November 2002.

3.2 The Payload

3.2.1 Electrostatic differential accelerometers

The core or prime MICROSCOPE instrument is based on ultra-sensitive, tri-axial electrostatic differential accelerometers, which have been developed and space qualified by ONERA for the CHAMP and GRACE missions. These accelerometers employ electrostatic levitation and minute motion and attitude sensing of solid test masses in a precisely manufactured and thermally controlled instrument cage.

The MICROSCOPE instrument consists of two differential accelerometers, which are identical except for their test mass materials. Each of the two instruments is composed of two concentric electrostatic accelerometers with quasi-cylindrical test masses, which have spherical matrices of inertia to reduce the effects of local gravity gradient fluctuations. Their sizes have been selected such to further limit the effects of different high order gravitational multipoles. All around the test masses, pairs of electrodes are engraved in the accelerometer core, made of gold-coated fused silica, for capacitive sensing of their position and attitude (see Fig. 2).

Eight quadrant electrodes sense the radial translations and rotations, two cylindrical electrodes at the ends of the test-mass detect their motion in the axial direction. The rotation of the mass about the axial direction is measured through dedicated flat areas on the mass and dedicated external electrodes.

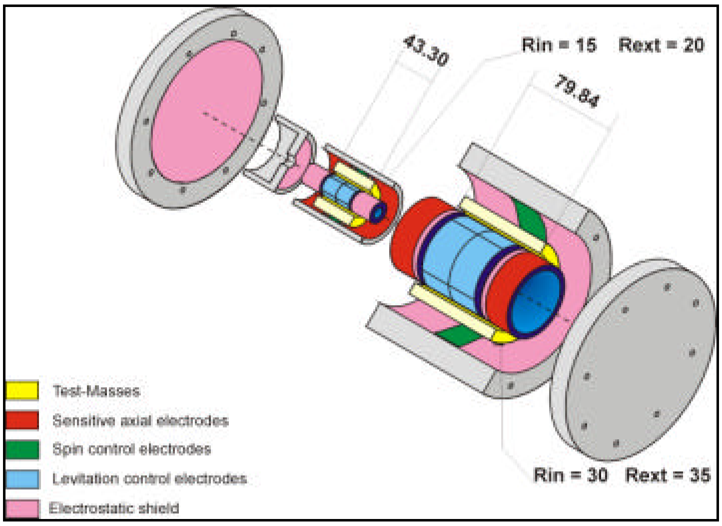

Figure 2: Differential accelerometer sketch

The same electrodes are used to generate electrostatic fields for the servo control of the masses to maintain them motionless with respect to the silica instrument frame. The resultant electrostatic force is derived from the accurate measurement of the applied voltages on pairs of electrodes (see Fig. 3). The mean force applied on both masses of the same instrument is then nullified by the satellite drag free control system. This system commands FEEP thrusters to move the satellite and thus the instrument silica frame to exactly follow the masses. A remaining differential force, which cannot be accounted for by instrumental errors, would then hint to a violation of the Equivalence Principle.

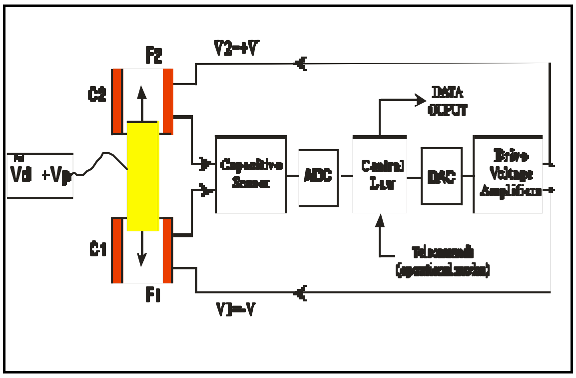

Figure 3: Capacitive sensing configuration. The capacitances C1 and C2 are maintained equal by applying the voltages V1 and V2 to move the mass at the centre.

The cylindrical test masses are centred during the instrument integration with an accuracy of 10µm. The relative position of the two masses will be evaluated in orbit through ground data processing. An accuracy of 0.1µm can be expected by exploiting the differential effect of the Earth gravity gradient field on the masses. Their relative position can be modified by offsetting the electrostatic servo-loops. Verification of the instrument sensitivity to this parameter will be performed during the calibration phase together with tests on the rejection rate of the Earth gravity gradient signal.

In the useful difference signal of the experiment, the common acceleration of the masses is rejected, in particular the residual cinematic acceleration of the satellite that is not compensated by the gravity field, for example the effect of surface forces. This is true when the sensitivity and the orientation of the two accelerometers are matched. Matching will be performed during the calibration phase by moving the satellite at well known frequency along the three axes with the help of the FEEP propulsion system. A matching of 100 parts per million is expected to be coherent with the $3x10^{-10}$ $ms^{-2}/Hz^{1/2}$ level of the expected residual satellite drag.

3.2.2 Instrument configuration on board the satellite

Both differential accelerometers are integrated into tight vacuum housings that also provide thermal insulation and magnetic shielding. These housings are mounted near the satellite centre of mass in order to reduce torque requirements on the propulsion systems to maintain the satellite rotation.
The sensitive axes of the accelerometer are oriented in the orbital plane along the spacecraft X-axis. The centres of the test masses are placed on the rotating axis of the satellite, normal to the orbital plane. The total mass of the payload will be about 31 kg (see Table 1).

Accelerometer	**Mass (kg)**	**Volume (mm^3)**
Mechanics		
Acc1 (Pt-Pt) Acc2 (Pt-Ti) Interface	9 8.5 2.5	$320 \times \varnothing 180$ $320 \times \varnothing 180$ $20 \times 210 \times 350$
Electronics		
analog & digital	2×4.5	$2 \times (240 \times 160 \times 150)$
Harness mechanics to electronics	2×1	
Total	**31**	

Table 1: Instrument mass and volume

The power consumption of the instrument has been estimated according to the different envisaged phases of operation. Each accelerometer requires about 16W secondary power. 40 W primary power is needed from the non-regulated 28V satellite bus when both instruments are switched-on and the DC/DC converter efficiency is considered.

The resolution of the instrument is evaluated from the noise of the electronics circuits as measured in the laboratory, from disturbances in the mass motion sources as modelled after experimental investigations and from the sensitivity to environmental influences. The performance of the instrument is optimised at frequencies around 10^{-3} Hz, which will also be the measurement frequency f_{EP} in the case of a rotating satellite.

3.3 The MICROSCOPE satellite

3.3.1 The CNES MYRIADE family

MYRIADE is a microsatellite program of the French Space Agency CNES which is devoted to scientific and technology missions. MICROSCOPE will be the fourth mission to be carried out in the frame of the MYRIADE program, which was started with the development of the DEMETER microsatellite.

The MYRIADE program includes:

- a common spacecraft bus/structure,
- a multi-mission ground system,
- engineering development processes,
- facilities and procedures for test and integration,
- a system validation bench,
- data system basis,
- qualified equipment,
- structures for documentation, project and quality management,
- on-board functional chains.

Ground segment:
The ground segment for the CNES micro-satellite family is shown in the following Figure 4:

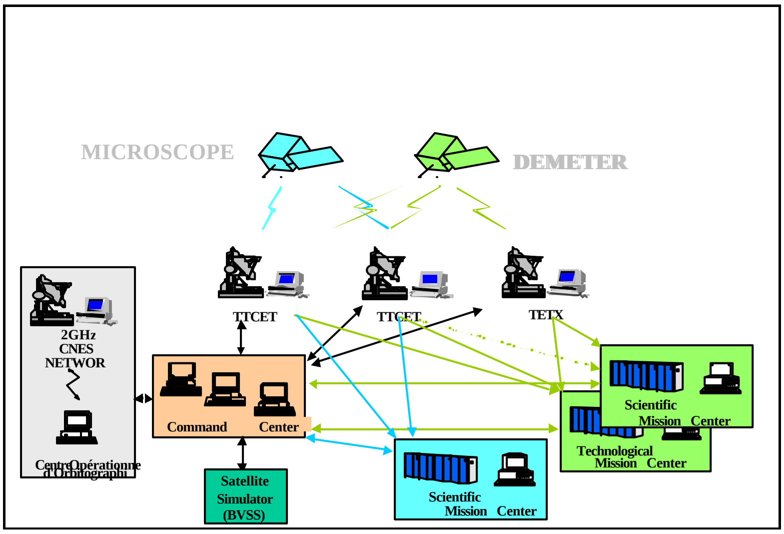

Figure 4: CNES microsatellite system architecture

Main Satellite Functional chains:
The satellite functional chains consist of:

- the structure (aluminum, aluminum honeycomb) with its interface to the launcher on the +X side of the satellite,
- the thermal control (using passive means and heaters),
- the electrical power generation, regulation and distribution function. It consists of steerable GaAs solar arrays, a battery, a power conditioning and distribution unit (PCDU) and two buses for power distribution, which are not voltage regulated.
- the command/control chain, with the on-board computer (OBC) and its on board memory (1 Gbits),

- the telemetry/telecommand TM/TC chain with its two emitter/receiver in hot redundancy,
- the hydrazine propulsion subsystem,
- the 3-axis attitude control system (ACS) consisting of
    - the sensors: 3 solar sensors, one 3 axis magnetometer, one star sensor
    - the actuators: 3 magnetotorquers, 3 reaction wheels.

The satellite TM/TC chain insures the management of the satellite configuration, on board data storage (housekeeping and scientific), data transmission to ground, the management of the received telecommands and the command of the ACS (Attitude Control System).

Telemetry and payload data are transmitted in S-band to the ground via a redundant on-board transmitter (cold redundancy). TM rate is 400 kb/s maximum (CCSDS format with Reed Salomon). Commands are transmitted in S-band from the ground to both on-board receivers (hot redundancy) at a maximum rate of 20 kb/s (format CCSDS).

Power Conditioning and Distribution Unit PCDU:
The PCDU is in charge of:
- the launcher separation detection and the connection of the main non regulated bus to the battery,
- the power distribution to the satellite subsystems and the payload,
- supplying powering the pyrotechnical devices.

The voltage levels on the power bus are between 22 and 38 V. The current is limited to 0.6 A for a single line.

On Board Computer (OBC):
The OBC is the central computing node for the control of the platform. It includes the following functions:
- startup/management of the OBC itself,
- failure detection and reconfiguration. This feature detects internal hardware errors caused by upsets or latchups and performs autonomous error recovery actions. A unique and direct emergency telecommand permits a hardware reset of the OBC. Multiple watchdogs protect against software errors.
- telecommand decoding: the TC format follows the CCSDS standard,
- telemetry coding: The CCSDS standard is used. 3 different rates can be selected by the flight software. A Reed-Solomon coder is included into the TM formatter with a fixed interleaving value of 5.
- distance measurement: The use of a dedicated direct telecommand permits to measure the delay time from the ground station to the spacecraft, and thus to synchronise the onboard time to UTC time.
- onboard time housekeeping: Based on a OCXO 10MHz frequency reference, an onboard hardware free-running counter permits to maintain a date and clock reference with a high accuracy (10^{-7} /day) between two consecutive ground station passes.
- storage, loading and execution of the flight software: Software storage is done into flash EPROM memory.

- storage, protection and download of data: an internal memory of up to 1Gbits is used for code execution and data storage under software control; maximum data rate for the acquisition of the prime payload is 1 Mb/s using the OSlink I/F.
- electrical interfacing: all platform equipment is connected to the OBC computer through intelligent I/O modules allowing to change the equipment interface (if compatible with each I/O channel capability) without flight software modification.

3.3.2 MICROSCOPE satellite layout

MICROSCOPE is a very specific mission that requires large adaptations to some of the functional chains developed in the frame of MYRIADE. The main drivers come from the mission requirements to minimize the effect of atmospheric drag and magnetic torque on the satellite. This system called the “Attitude Control and Drag Compensation Subsystem” controls the six degrees of freedom of the spacecraft. It is based on input from one of the accelerometers of the payload to the FEEP micropropulsion system.

The instrument itself is required to be maintained in a highly stable thermal environment around the EP measurement frequency f_{EP}. The operating temperature range of the experiment extends from +20°C to +40°C. It is expected to implement an active thermal control at the payload interface to keep the I/F temperature stable within 0.1K around f_{EP}.

The spin mode of the science mission necessitates the accelerometers to be installed in the center of gravity of the satellite. This also needs to be close to the geometric center of the spacecraft. It is anticipated that the distance between the satellite center of gravity and the accelerometer axis will be less than a few millimeters in order to reduce the ACS propulsion needs for torque. As far as possible, the instrument will be symmetric in respect to the geometric center of the spacecraft. The following Figure 5 shows the present configuration of the MICROSCOPE satellite.

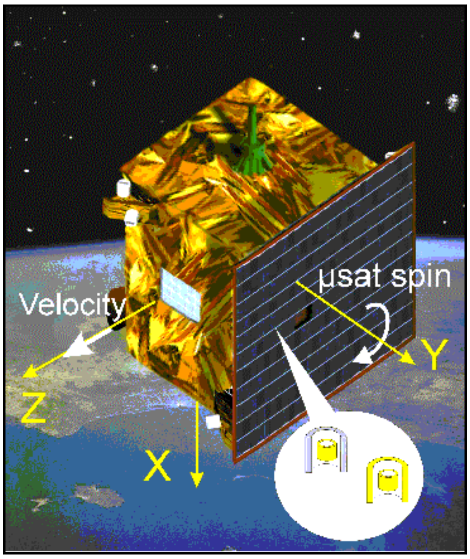

Figure 5: Artist view of the MICROSCOPE satellite and accelerometer test-masses in the orbital frame.

The Spacecraft Y axis is the spin axis. Its structure is a simple parallelepiped. The MYRIADE design is kept except for the dimensions, which are increased to 800 x

700 x 600 mm^3. The structural design provides high rigidity with eigenfrequencies higher than 45 Hz in the lateral and 100 Hz in the longitudinal direction for a total satellite mass of 120 kg.

The payload accelerometers are mounted in the center of the parallelepiped, their associated electronics is attached to the lateral panels. The accelerometer sensitive axes are oriented parallel to the orbital plane (XZ plane) and the axis joining the accelerometers centers is parallel to the Y-axis.

The requirements on alignment are:	accelerometer axis/Y_{CDG} :	$< 10^{-2}$ rd
	accelerometer centers/Y_{CDG}:	< 1 mm

The +Y panel is designed to accommodate both the launcher adapter interface and the instrument interface.

The solar array is mounted on the -Y side. It will deliver about 120 W (mean power available for the satellite all along the baseline Sun Synchronous Orbit). Two antennae (both are used for emitting and receiving) will be located on the satellite +X and -X sides.

The residual satellite magnetic moment has to be minimized in order to reduce perturbing torques induced by the interactions with the earth magnetic field and to avoid accelerometer perturbations. The residual magnetic moment of the spacecraft shall be less than 1 $A.m^2$, with variations less than 0.02 $A.m^2$ at f_{EP} and at 0.3 m distance from the accelerometers.

3.3.3 Attitude Control and drag compensation System (ACDCS)

A hybridization of the standard MYRIADE orbit control system and the drag free control system will be implemented to provide the following ACDCS modes:

Acquisition and Safe mode

This is to provide a safe mode where the solar array is sun pointing. The spacecraft enters this mode automatically after separation from the launcher, in this mode the satellite is also spinned down. The necessary torques are applied via 3 magneto-torquers. Attitude stabilization will be obtained either with a moderate spin around the Y-axis or one momentum wheel. 3 Sun sensors provide the Sun direction in the Spacecraft frame.

Star sensor waiting mode

The main purpose of the star sensor waiting mode is to provide a capacity of reorientation of the Spacecraft. This is used to rotate the satellite from a Sun pointing attitude to an attitude where the Y-axis is perpendicular to the orbital plane. It enables to stop the satellite spin or the momentum wheel and to start the attitude control (3-axis) using the FEEP thrusters.

Drag free acquisition mode

The Drag free acquisition mode provides a level of dynamics disturbances that is compatible with the switch-on of the payload. The angular acceleration measurements are first introduced in the 3-axis controller. A full 6-axis control follows which is based on the angular and linear measurements provided by the accelerometers.

Finally, the level of dynamics disturbances that is reached allows switching the payload into the fine measurement mode.

Mission and calibration mode

This mode allows the 6-axis control of the spacecraft with the performance required by the mission. The instrument is operated in its fine mode. The spacecraft attitude can be controlled inertially fixed or with a constant spin at well known frequency ($3x10^{-3}$ rad/s) around the Y-axis. In both cases, this axis is controlled perpendicular to orbital plane.

An attitude estimator performs the hybridisation of the star sensor and payload accelerometer angular acceleration measurements, whereas 3-axis common-mode acceleration measurements from the accelerometer are directly used as drag-free reference.

MICROSCOPE FEEPs

The performances for the control in translation and rotation of the MICROSCOPE satellite requires the use of thrusts that can be commanded in the 1 to 100 μN range. The Field Emission Electric Propulsion (FEEP) technology developed in Europe by the European Space Agency has been selected for MICROSCOPE.

Four identical pods will be placed external to the body satellite body on four of the corners. Two configurations are being studied; in the baseline 2 thrusters are mounted on each pod while in the redundant configuration there are 3 thrusters on each.

The directions of the thrusts are optimized to minimize the average thrust (i.e. the power consumption).

The main characteristics for the FEEP system are:

thrust range:	1-100 μN (1 thruster)
total impulse:	3750 Ns (1 thruster, 1 year mission, including margin)
thruster bandwidth:	5 Hz
thrust vector direction knowledge:	3 °
thrust vector stability:	3 °
noise on thrust magnitude:	$0.1\ \mu N/Hz^{1/2}$ at frequency ≥ 0.1 Hz
number of bits of the command:	12
sampling frequency of the command:	10 Hz
thrust scale factor uncertainty:	10 %
hysteresis:	5 %

Requirements for the ACDCS

	Specified parameter	Axe	Requirement	Unit
1	Instrument alignment w.r.t. the orbital frame	$\theta_x, \theta_y, \theta_z$	10^{-3}	rd
2	Instrument alignment stability	$\theta_x, \theta_y, \theta_z$	10^{-3}	rd $Hz^{-1/2}$ at fep
3	Maximum angular velocity (without the spin velocity)	$\Omega_x, \Omega_y, \Omega_z$	10^{-6}	rd.s-1
4	Angular velocity (stability)	$\Omega_x, \Omega_y, \Omega_z$	10^{-6}	$rd.s^{-1}.Hz^{-1/2}$ at fep
5	Maximum angular acceleration (with Drag-free)	$\Omega'_x, \Omega'_y, \Omega'_z$	10^{-5}	$rd.s^{-2}$
6	Angular acceleration (stability)	$\Omega'_x, \Omega'_y, \Omega'_z$	10^{-8}	$rd.s^{-2}.Hz^{-1/2}$ at fep

7	Maximum linear acceleration (with Drag-free)	Γx, Γy, Γz	10^{-7}, 5.10^{-6}, 5.10^{-6}	$m.s^{-2}$
8	Drag-free control residual noise	Γx, Γy, Γz	3.10^{-10}	$m.s^{-2}.Hz^{-1/2}$ at fep

3.3.4 Satellite Mass Budgets

Preliminary layout studies shown that the internal volume (more than 250 l) is sufficient to accommodate the functional chains equipment (chemical propulsion is replaced by electrical propulsion) and the scientific payload. The present mass satellite budget is:

structure:	43 kg	
thermal control:	2 kg	
power:	20 kg	
OBC and TM/TC:	6.5 kg	
ACS: (1 wheel):	3.5 kg	
propulsion:	20 kg	(4 FEEP pods with electronics)
accelerometers:	17.5 kg	(payload)
acc. Electronics:	11. kg	(payload)
secondary structures:	2.5 kg	(payload)
total satellite:	123 kg	(without 3 kg of shock damping system)

3.4 Launch vehicle

MICROSCOPE shall be launched with a DNEPR rocket provided by KOSMOTRAS.

3.5 Scientific Mission Centre

The scientific mission centre will be located at ONERA and will be in charge of:

- sending payload telecommands through the CNES Control Centre
- pre-processing and archiving the scientific and housekeeping data,
- partial processing of science and housekeeping data for quasi real time overview of the experiment
- managing the exchanges between ONERA and CERGA for the fine analysis

The total data flow rate will be 480 Mbits per day. The satellite memory of 1 Gbits capacity and the rate of the TM/TC link to the ground station of 500 Mbits/day is compatible with the payload needs.

3.6 Mission and science operations

The mission duration of one year is subdivided into 6 phases.

After the heliosynchronous orbit injection, the satellite is placed into a sun pointing, safe orbit. Then the “stars sensor waiting mode” is acquired using the nominal equipment of the microsatellite platform: sun sensor and star tracker, magneto torquers, possibly one reaction wheel and the FEEPS.

In the second phase, the two differential accelerometers are switched on, one by one and their operations are verified. In this phase also the electrical propulsion is calibrated.

In the third stage the drag-free and the fine attitude control is switched on and verified.

In the fourth phase, the accelerometers are switched into their highest sensitivity mode. The instrument and the ACDCS are accurately characterised: residual acceleration levels, stability of rotation axis and frequency, coupling between axes, instrument sensitivity to environment and gravity gradients.

After all calibrations, the EP experiment is realised with the first differential accelerometer in inertial and spinning attitudes, and with two angular phases along the orbit (defined at the equator passage). In order to verify that no severe drifts have occurred between the beginning and the end of the experiment, the previous phase of calibration is performed again.

The EP experiment is then performed with the second differential accelerometer with a new calibration at the end. According to the integration periods required for the filtering of the data, the minimum duration is estimated to last 6 months.

In case the prime objective of the mission can be achieved before the end of the nominal mission lifetime or if the spacecraft health and other necessary resources allow an extension of the mission, additional experiments can be performed. Examples for such investigations are given below.

- the FEEP's neutralisation efficiency,
- the Ground Laser tracking of the microsatellite,
- the drag-free system operation with a control versus the test mass relative position to the satellite, like in the LISA future space mission dedicated to the observation of gravity waves, instead of a control versus the acceleration provided by the instrument like in MICROSCOPE baseline operation,
- the gravity gradiometer operation and calibration by exploiting two single accelerometers of different housing, then not concentric, like in the GOCE mission,
- altitude decrease, aerology analysis and atmosphere entry methodology

3.7 Error and data analysis

The resolution of the instrument has been preliminarily evaluated:
- from the noise of the electronics circuits, as measured in the laboratory,
- from all foreseen disturbances introduced to test mass motion, as modelled after experimental investigations,
- from the instrument sensitivity to the environment.

A 10^{-12} $ms^{-2}/Hz^{1/2}$ resolution is reached at frequencies around 10^{-3} Hz, i.e. several orbital frequencies corresponding to the instrument frame angular rate with respect to the Earth pointing frame.

The accelerometer error budget is graphically displayed in Figure 6. At low frequencies, the thermal instabilities ΔT induce radiation pressure and radiometer acceleration fluctuations due to the residual gas at pressure P for the latter:

$$\Gamma_{radiometer} \approx \frac{1}{2m} PS \frac{\Delta T}{T}$$

where m is the mass of the test-mass and S is the area considered in the direction of the thermal gradient.

At higher frequencies, the position sensing resolution affects the resolution with a square frequency law.

$$\Gamma_{posnoise} = x_{noise}\left(\omega^2 + \omega_p^2\right)$$

The selected configuration leads to a computed passive stiffness (different from the active servo-loop one) between the mass and the instrument frame of less than 5×10^{-3} N/m (ω_p<0.1rad/s) and effects can thus be neglected at lower frequencies.

The thermal noise of the mass motion is derived from the damping factor estimated from dedicated laboratory experiments and is mainly due to the thin 5 µm wire, which is used for the charge control of the mass:

$$\Gamma_{wire} = \frac{1}{m}\sqrt{4k_b T H_{wire}}$$

where H_{wire} represents the gold wire damping.

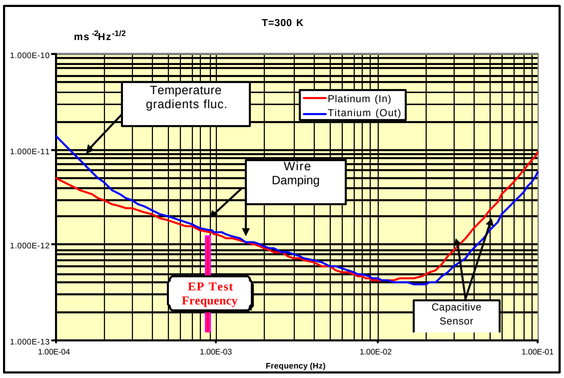

Figure 6: Accelerometers error budget

The performance of the mission has been deduced from the instrument accuracy and from the effects of all foreseen disturbances on the satellite. The instrument output will be integrated over 10^5 seconds taking into account the frequency and the phase of the eventual EP violating signal. The EP test is performed along the X-axis. Following the preliminary error analysis the obtained result is compatible with the mission objective.

4 Programmatics, Project and Science management

4.1 Status of spacecraft development

The requirements for the drag free control system have been consolidated through detailed analysis and simulations. The acquisition and transition modes have been simulated.
The requirements for the FEEP thrusters have been established and the provider has been selected. The development will begin before end of 2001.
The satellite thermal studies focused on the temperature control of the payload interface. An active concept is expected to be implemented which will be assessed in mid-2002 through a thermal test campaign.
The design of the structure has been focused on the payload mechanical interface and launcher adapter. A preliminary accommodation of the equipment has been drawn.

4.2 Status of payload development

The payload is mainly composed of two elements linked by harnesses of power or of data: the mechanical core with the test-masses enclosed in a vacuum housing (called SU - Sensor Unit) and the electronics for the control of the mechanical core and of the data formatting (called ICU - Interface Control Unit).

The payload is entering in its phase B development.

The electronics architecture philosophy and functions has been studied in 2001 and gave rise to an ONERA Invitation to Tender issued in August 2001 for the study, the production and the delivery of one EM and two FM of the ICU. The company has been selected in December 2001 with CNES consultancy and the contract has been issued based on the ONERA ITT statement of work. The delivery of the ICU Engineering Model is foreseen in the autumn of 2002.

The Sensor Unit design is being completed to start the production of the EM in the first quarter of 2002. Two SU EM shall be produced: one dedicated to ground tests with light silica test-masses and one with heavy test-masses (Inermet material with density of 17.6 close to the FM heaviest test-masses of density 20) dedicated to free-fall and vibrations tests.

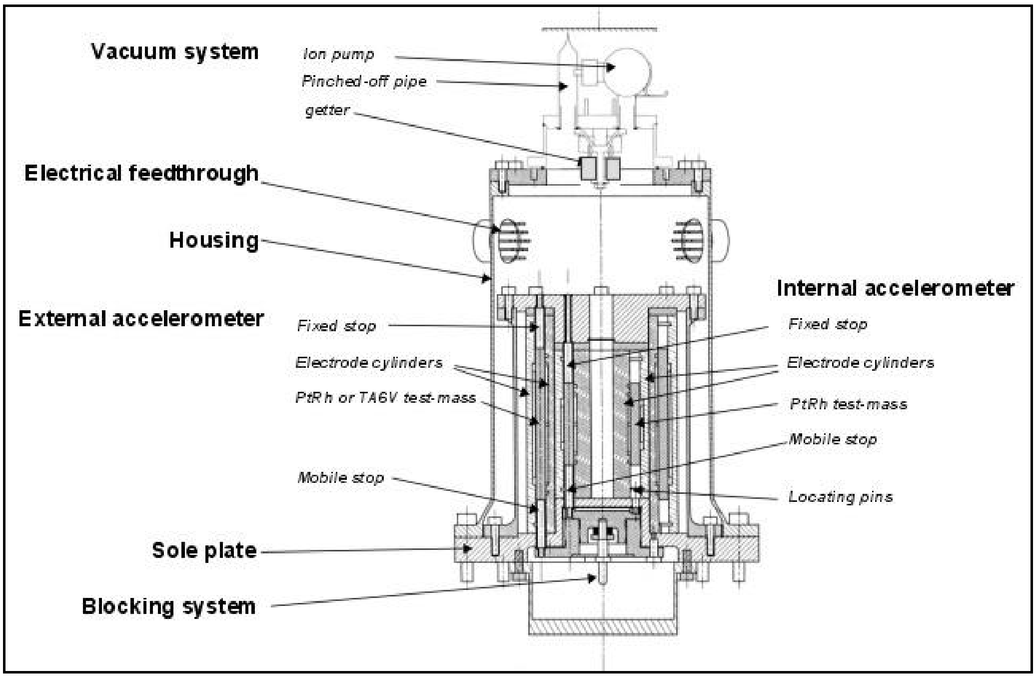

Figure 7: Sensor Unit design

As a baseline, the FM test-masses are made of Platinum (90%)-Rhodium(10%) and Titanium alloy and should be provided in autumn 2002. A specification document for the production of the test-masses with high accuracy has been established in 2001 on the base of a shape optimisation with respect to gravitational disturbance sources and on the base of manufacturing criteria.

The silica cylinders supporting the electrode groves and surrounding the test-masses have been designed and should be produced by a selected industrial subcontractor from January 2002 to April 2002. These silica cylinders are in the FM configuration and should not change from the EM to the FM.

The Sole plate, being the reference plane for the mounting process, is made of Invar and should be produced very accurately by a selected industrial subcontractor from January 2002 to April 2002.

The blocking mechanism system which ensures safely the motion damping of the test-masses during the launch phase is being designed. The interface has been selected but the "motor" type is under selection and should be either a stepper motor, either a pyrotechnic system or a "soft pyrotechnic" system under study in CNES. The final selection is planned for the first quarter of 2002.

4.3 Status of the Science Mission Centre

The Mission Centre will be set up at ONERA in its Châtillon centre. A WEB data service is being implemented by ONERA where the current status of development will be made available to the public. This web site is located at **http://microscope.onera.fr** and will be available after validation by ONERA at the end of December 2001.

A restricted area with payload documentation will be included in January 2002.

This address shall be used as Mission Centre access during the flight phase.

4.4 Project structure

4.4.1 Points of contact

Programme Manager:
Sylvie Léon, CNES, DPI/E2U, 2 place Maurice Quentin 75001 Paris (tel 33-144767802, email: sylvie.leon@cnes.fr)

Project Manager:
Olivier Vandermarcq, CNES, DSO/ED/MS, 18 avenue Edouard Belin 31401 Toulouse Cedex 4, (tel 33-561281466, email: olivier.vandermarcq@cnes.fr)

Principal Investigator PI:
Pierre Touboul, ONERA, DMPH, 29, avenue de la division Leclerc, BP 72, 92322 Chatillon Cedex (tel 33-146734801, email: touboul@onera.fr)

4.4.2 General Organisation

MICROSCOPE is part of the scientific program of CNES concerning Fundamental Physics. Figure 8 summarises the organisation of the MICROSCOPE project. CNES as the lead Agency is in charge of the microsatellite development, the Ground Control Centre in Toulouse and all pre-launch activities.

ESA is responsible for the procurement of the FEEP thrusters.

ONERA is responsible for the payload delivery, the scientific mission centre and for the scientific return of the mission. Mission and data analysis is performed in close collaboration with OCA/CERGA.

The ZARM Institute of Bremen, Germany is contributing to the project through DLR funding. This participation covers more than one hundred free-fall tests in microgravity at the Bremen fall tower making use of an improved double capsule. Tests of the Engineering Models and of the Flight Models are planned at Bremen.

Further French Institutes contributing to the preparation and the analysis of the mission are:

- GRGS (OMP Toulouse and OCA Nice) for the aspects concerning the Earth gravity gradients and orbitography,
- Ecole Normale Supérieure, Institut des Hautes Etudes Scientifiques Gif sur Yvette, Institut d'Astrophysique de Paris, Laboratoire Kastler Brossel, Observatoire de Paris Meudon for the experiment data analysis and the theoretical exploitation,
- more generally the GDR Gravitation et Experience, GREX.

Other scientific proposals for co-operation, from ESA Member States, can now be envisaged in the frame of the present joint CNES/ESA Announcement of Opportunity.

Suitable proposals will be selected by a joint CNES/ESA Peer Review Committee. The successful proposers will have the status of Co-Investigators .The existing MICROSCOPE Science Working Team, SWT, which includes the already selected PI from ONERA, the Co-PI from CERGA and the Co-I from ZARM, shall be enlarged to include scientists selected from ESA Member States. The PI shall chair the SWT.

A Steering Group is composed of CNES, ESA and ONERA representatives. It remains informed on the project development status and arbitrates and directs the project in case of conflicts, which go beyond the approved resources concerning budget, planning or manpower.

4.4.3 MICROSCOPE project organisation

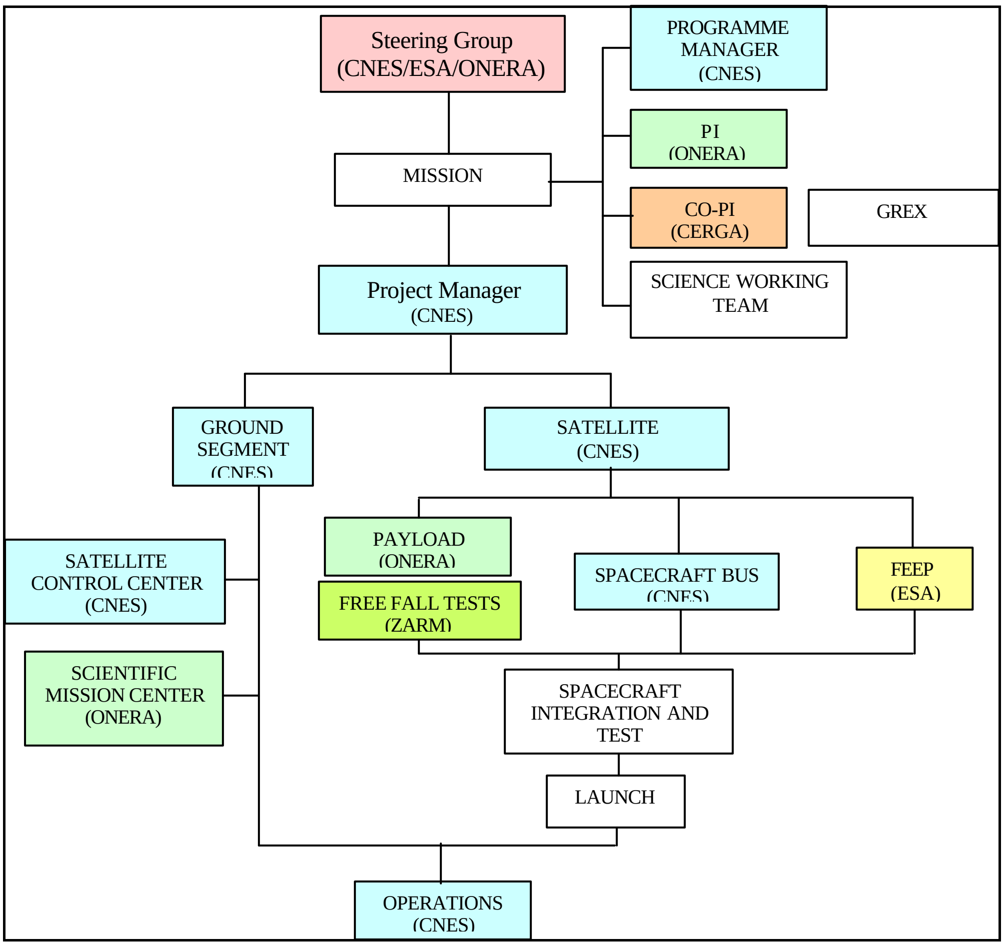

Figure 8: organisation of MICROSCOPE project

4.5 Model philosophy

4.5.1 Satellite Level

A protoflight philosophy will be applied to MICROSCOPE at the spacecraft level.

4.5.2 Payload level

At payload level, the usual standard shall apply, which is to develop one qualification and one flight model. Alternatively, an engineering model EM and a protoflight model for qualification and flight can be built. This depends on the status of development and qualification that has been reached by the equipment in respect to the environmental and test requirements applicable to MICROSCOPE.The EM shall be representative to the FM with respect to mass, volume, power and interfaces.

4.6 Hardware and software development schedules

4.6.1 Contribution to Prime Payload

Delivery of Engineering Model:	September 2002*
Delivery of Flight hardware:	September 2003

* This deadline does not apply to the potential provision of the test masses, for which an engineering model would be required only by November 2002.

4.6.2 Stand alone instruments

In addition to contributions to the prime payload, proposals for additional hardware that may benefit the mission in a wider context are solicited. Such contributions have been termed "stand alone instruments". Spacecraft resources for these instruments must not exceed 1 kg, 0.5 litre and 3 W in total. Their development schedule is the following:

Preliminary Design Review:	November 2002
Delivery of Engineering Model:	May 2003
Delivery of Flight Model*:	Mid 2004.

Flight Model*:
must be understood as "qualified, accepted and documented flight hardware".

4.6.3 Software

Delivery of the User Requirement Document:	March 2003
Delivery of flight software*:	March 2004

Flight software*:
must be understood as "validated software".

4.7 Environmental test plans

The equipment shall be accepted after the qualification has been demonstrated through tests as specified in chapter 7.
However, a full qualification test may not be required if it can be demonstrated by analysis and/or by previous tests campaigns achieved on the same equipment, that the qualification levels required for MICROSCOPE have been covered with success.

Acceptance/qualification tests shall include the following:

- functional tests
- EMC test
- Thermal- vacuum test
- Vibration test (sine + random)
- Shock test

For contribution to the prime instrument, the procedure for testing shall be agreed upon with the MICROSCOPE PI after successful selection.
For the stand-alone instrument, test performance is under the responsibility of the proposer.

4.8 Data rights and data archive

The scientific mission centre at ONERA will be in charge of archiving all scientific and housekeeping data.

A data analysis team under the responsibility of the PI shall perform the scientific data analysis. CoIs may be invited to join the data analysis team if they wish to do so and if they bring in special expertise.

Data rights for results from stand-alone instruments are with the responsible investigator and subject to the same rules as for the prime instrument as laid down in the following paragraph.

The full in-flight calibration of the instrument is foreseen to be completed not later than one year after the launch of MICROSCOPE. The scientific data from the mission will become publicly available not later than one year after the completion of this full in-flight calibration.

4.9 Outreach and public relations

CNES will have the overall responsibility for planning and carrying out outreach and public relations activities related to the MICROSCOPE mission in co-ordination with ESA.

During the development phase of the mission, ONERA will set up a web home page for MICROSCOPE as an information tool for the scientific and technology community and the general public. After launch, a more elaborated home page will include the latest news on the mission as well as preliminary scientific and technological results.

The MICROSCOPE CoIs shall actively contribute to outreach and public relations activities. They shall initiate and identify opportunities for publishing project related progress reports and technical/scientific results. Material suitable for release shall also be made available upon CNES and ESA request in order to support their respective campaigns.

5 Guidelines for proposals

5.1 Constraints for proposals:

A joint CNES/ESA Peer Review Committee co-chaired by the PI and an ESA-nominated scientist will select suitable proposals. Bearing in mind that in view of the advanced status of the MICROSCOPE Project, major changes or major additions at the payload or at the satellite level may be impossible to implement.

The risk that a failure of a proposed instrument propagates to a functional chain of the satellite shall be assessed and minimised.

5.2 Identified prime targets for proposal submissions

In spite of the advanced status of the MICROSCOPE project, there is room for participation and, in particular, the following areas have been identified for contributions. This list, however, shall by no means preclude proposals targeting other contributions as solicited in this AO.

- proof mass procurement, verification and validation,
- software contributions to drag free control,
- data and error analysis,
- additional experiments during or after the core mission making use of the MICROSCOPE infrastructure,
- stand-alone instruments: spacecraft resources of 1kg, 3 W and about 0.5 l will be made available for additional instruments that may benefit the mission in a wider context.

5.3 Proposal content

Proposals shall respond to the MICROSCOPE science objectives and programme constraints described in this AO and shall provide all requested information to permit a complete evaluation against the criteria listed in section 5.5. In particular, the proposals shall address areas identified under section 5.2 and demonstrate the schedule compatibility of the proposed contributions with the MICROSCOPE programme and provide clear interface data to monitor the assessment and preliminary definition of detailed interfaces.

Each proposal submission package will be composed of four parts, with all pages written in English and numbered.

Part I: Scientific and Technical Plan
Part II: Management Plan
Part III: Technical Interface Document
Part IV: Funding Proposal

5.3.1 Part I: Scientific and Technical Plan

Part I of the AO proposal shall not exceed twenty single-spaced type-written A4 pages, including illustrations, without reduction, executive summary and table of contents. The proposal shall not contain appendices. It shall be self explanatory and shall contain all information needed for the evaluation of the proposal in the areas of the science goal, instrument performance, science operations and all related support. The proposal shall adhere to the following table of contents:

Cover page
Executive Summary (1 page)
Table of Contents
Scientific Objectives
Proposed Investigation in Context of Other Missions
Technical description of Instrument, including its maturity
Data Reduction and Scientific Analysis Plans
Test and Calibration Plans
System Level Assembly, Integration and Verification
Flight Operations
Description of Instrument Team Qualifications and Experience
Organisation and management structure of the Instrument Team

Each of these topics is detailed hereafter.

Cover Page The Cover Page shall include the title of the proposal, the name, address, telephone and fax numbers and e-mail address of the Co - Investigator (CoI) and key team members as far as applicable.

Executive Summary The Executive Summary shall include the title of the proposal, the names and institutions of the investigators and summary information. The following aspects shall be addressed within a 1-page limit:

Objectives of the proposal
Performance of the instrument proposed to fulfil its anticipated goals
Summary of required spacecraft resources
Instrument operations and scientific analysis
Maturity of the proposed instrument
Compatibility with the MICROSCOPE schedule
Management scheme
Funding status
Departures from constraints stated in this AO (including the Annexes)
List of requirements in case this proposal depends on other proposals or sub-systems on the MICROSCOPE spacecraft

It is anticipated that the proposals will remain within the technical and programmatic constraints of the MICROSCOPE programme as described in this AO. If proposed options violate any of these constraints, a clear statement of each violation together with justifications shall be included in the Executive Summary. Further details of each violation shall be provided by the proposer in the appropriate sections.

Table of Contents As per the list in this chapter.

Scientific Objectives The section on Scientific Objectives shall provide a description of the proposed scientific investigation and clearly state the general scientific capabilities of the proposed instrument. The capabilities of the proposed instrument should be explained. The baseline performance envelope of the instrument and performance evaluation criteria should be explicitly stated and summarised in tabular form. Expected results should be outlined and discussed, as far as possible, in both qualitative and quantitative terms. If a proposal contains one or more options in the design, which lead to violations of technical or programmatic constraints, the scientific justification shall be given in this section.

Proposed Investigation in Context of Other Missions The capabilities of the proposed instrument should be compared, if relevant, to similar instruments flown or to be flown on-board other spacecraft.

Technical Description The section on Technical Description shall give a comprehensive and detailed technical description of the proposed instrument including instrument design, current status and ability of the proposed technologies in the baseline design and risk associated in any technology development. The heritage of the proposed instrument design (space qualified and experienced hardware) shall be stated. Detailed information on the interfaces between the instrument and the spacecraft should be given in the Technical Interface Document, to be submitted as Part III of the proposal.

Test and Calibration Plans This section describes all test and calibration (ground/pre-launch, in-orbit) plans, procedures and equipment deemed necessary to verify and calibrate the proposed instrument in order to achieve its scientific goals. The ground test and check-out equipment to be supplied for the instrument level testing shall be described in the Technical Interface Document.

System Level Assembly, Integration and Verification This section (AIV) should state how the proposal complies with the project provided pre-launch AIV programme at spacecraft system level. The ground test and checkout equipment to be supplied by the PI to support the system level test programme shall be described in Part III of the proposal.

Flight Operations This section describes the operational concept of the proposed contribution, and identifies specific requirements for flight operations support.

Description of Team Qualifications and Experience This section should provide bibliographical information of the Co-I, the Experiment Manager plus key technical personnel. Extensive bibliographies are not required although 5 key publications or key activities of particular relevance to the proposal may be listed.

Organisation and Management Structure of the Team This section should show a functional organisational chart and how the Co-I together with the Experiment Manager establish an efficient and effective management scheme for all aspects of his/her proposal, subdivided into the following topics:

- Development
- Science Operations

5.3.2 Part II: Management Plan

The Management Plan to be implemented by the Co-I for all aspects of the proposed investigation should be described in this section. The responsibilities of all Team members (Co-I, Experiment Manager) and their institutions should be detailed and an organigramme (organisational chart) provided. This section shall include a Development Plan demonstrating how the MICROSCOPE Programme Schedule shall be met. A schedule for the development of the contribution shall be given in sufficient detail for the project engineers to be able to monitor progress against reports to be provided by the Co-I. The Management Plan shall further contain the following detailed information:

- description and justification of responsibilities of all team member,
- name key personnel with responsibilities and related experience (Experiment Manager, lead engineers, planning and reporting, product assurance),
- description and justification of proposed method of acquisition (by the Investigator's institution, contracts with Industry),
- description and justification for the proposed development plan and schedule, identification of development status, critical problems and risk assessment,
- description and justification of the test programme with all facilities envisaged and their planned location,
- planned support from the Co-I to the overall programme activities,
- statement of compliance with the management, reporting and electronic information exchange requirements.

5.3.3 Part III: Technical Interface Document

The purpose of the Technical Interface Document (TID) is to document the Co-I response to the technical and programmatic requirements as described in this AO. After selection, the TID will be used as a basis to establish an Experiment Interface Document (EID) for each contribution, which will be maintained and updated at regular intervals. The EID becomes a contractual document between CNES/ESA and the selected Co-I. The Technical Interface Document shall contain as a minimum the following chapters:

System Requirements
Interface Requirements
Ground Support Equipment
Development and Verification
Testing and Operations
Product Assurance
Programme, Schedule and Management

Further information not covered within the standard format may be added at the discretion of the proposer. Although there may be some duplication of information to be provided, the purpose of Part III is mainly to provide factual information on all aspects of the proposal, whereas discussion, justification and risk assessment, etc. is to be provided in Part I.

5.3.4 Part IV: Funding Proposal

The Co-I shall include separate sections for his/her own resource provision and funding status as well as for the resource provision and funding status of each of the proposed team members with the details of estimated resources for each activity subdivided into the following topics:

- Development
- Science Operations

with justification of:

- Internal manpower resources
- Other internal institute resources
- External contracts led by the Co-I
- Total funding requirements

The provisions being requirements on national sources for funds and manpower shall be stated and justified. The authorities for these resources shall be identified and the current status of the application stated. If the funding is not firm, the procedure to be followed to obtain the funding shall be stated. Please, note that a firm funding commitment is required by 23 May 2002 (TBC). The authorities responsible for providing the funding will be required to signify compliance by signature of the Instrument Implementation Agreement (IIA).

Letter of Endorsement

Proposers are responsible for submission of a copy of the proposal directly to the their national funding authorities. A Letter of Endorsement from the national funding authorities is due on 22 February 2002 together with the proposal.

5.4 Proposal submission and schedule

Proposals shall be available in hard copy and electronic form. CNES/ESA also request the provision of an electronic version, but the evaluation of the proposals will be solely based on the timely submitted hard-copy version. Hard-copy proposals shall be submitted to CNES/ESA in accordance with the table below. All four parts shall be bound together**. The proposals must be received at CNES and ESA-HQ by 22 February 2002.** The proposer shall notify (by fax or e-mail) the addressee in the table below:

- when the proposal has been mailed, stating the actual mailing date,
- the details on the electronic form of the proposal, for example an FTP server address.

For the hard-copy version ESA-HQ will confirm in writing that the proposal has been received. The addressee in the table should be contacted in case the confirmation is not received within one week after submission. If confidentiality is required for Part IV of the proposal (Funding Proposal) this should be clearly stated, and this part may be mailed separately.

Five hard copies and an electronic version of the proposal shall be sent to both Sergio Volonte and Sylvie Leon.

Sergio Volonte ESA-HQ (SCI-CA) 8-10, rue Mario Nikis F-75738 Paris Cedex 15 France Tel.: +33-1-5369-7103 Fax: +33-1-5369-7236 Sergio.Volonte@esa.int	Sylvie Leon CNES, DPI/E2U 2, Place Quentin 75001 Paris France Tel.: +33 144 767 802 Fax: +33 144 767 859 sylvie.leon@cnes.fr

One hard copy and an electronic version of the proposal shall be sent also to Michael Fehringer and Pierre Touboul.

Michael Fehringer ESTEC (SCI-SH) P.O. Box 299 NL-2200 AG Noordwijk The Netherlands Tel.: +31 71 565 4117 Fax: +31 71 565 4697 Michael.Fehringer@esa.int	Pierre Touboul ONERA, DMPH 29, avenue de la division Leclerc BP 72 F-92322 Chatillion Cedex, France Tel.: +33 146 734 801 Fax: +33 146 734 148 touboul@onera.fr

The schedule for submission, evaluation and selection of proposals is as follows:

Issue of Announcement of Opportunity:	20 December 2001
Response due:	22 February 2002
Evaluation of proposals:	25 February - 1 March 2002
Peer Review Committee meeting:	15 March 2002
FPAG recommendation:	April 2002
SSAC recommendation:	2 – 3 May 2002
SPC endorsement:	23 – 24 May 2002

5.5 Selection criteria and procedure

The selection criteria for the proposed contributions will reflect the significance of the proposals for achieving the primary goal of the MICROSCOPE mission, which is the test of the Equivalence Principle to one part in 10^{15}. They will further help assess contributions, which target on drag free technologies and proposals for “stand alone instruments” as defined in chapter 4.6.2. The evaluation criteria will specifically include:

- scientific, technical and programmatic compatibility with the MICROSCOPE mission concept and schedule,

- value of the scientific/technological investigation and the technology employed for the MICROSCOPE mission proper and/or for the development of drag free technologies,
- in case of a “stand alone instrument”, its value to the mission in a wider context,
- value of the scientific/technological investigation and the technology employed for future missions,
- originality with regards to current and planned missions,
- ability of proposed instrument/investigation to satisfy its scientific objectives,
- demonstrated technological feasibility, readiness and development status of the proposed instrument/investigation,
- innovation of method or technology used for the investigation,
- reliability and space qualification of proposed instrumentation,
- competence and experience of the team in all relevant areas,
- adequacy of proposed management scheme to ensure a timely execution of the proposal within the MICROSCOPE schedule,
- adequacy of funding, human resources and institutional support for the proposed investigation.

CNES and ESA will jointly appoint the Peer Review Committee that will evaluate the proposals. It will consist of six members, three from each side and will be co-chaired by the MICROSCOPE PI and an ESA nominated scientist.

5.6 Points of contact

If necessary, requests for further information and clarification before the submission of the proposals should be addressed to:
Pierre Touboul and Michael Fehringer (see section 5.4 for details of addresses).

6 Abbreviations and references:

Abbreviation	Definition
ACS	Attitude Control System
ACDCS	Attitude Control and Drag Compensation Subsystem
CCSDS	Consultative Committee for Space Data Systems
EP	Equivalence Principle
EPS	Electric Propulsion Subsystem
FEEP	Field Emission Electric Propulsion
FDIR	Failure Detection, Identification and Recovery
I/F	Interface
OBC	On Board Computer
OCXO	On board frequency reference
PCDU	Power Conditioning and Distribution Unit – (spacecraft)
SSM	Second Surface Mirror
SL	Satellite
SC	Spacecraft
URD	User Requirements Document
UTC	Co-ordinated Universal Time
TBC	To Be Confirmed
TBD	To Be Defined

References

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XI) Worden P. & al, Class. Quantum Grav. 13 (1996) A155-A158

XII) Touboul P., Rodrigues M., Willemenot E. and Bernard A., Electrostatic accelerometers for the Equivalence Principle test in space, Class. Quantum Grav. 13 (1996)

7 Details of technical requirements

7.1 Lifetime

A minimum of 2 years lifetime is required considering a one-year mission.

7.2 Mass, power and size for stand alone instruments

maximum mass:	1 kg
maximum volume:	0.5 l
maximum power consumption (at non-regulated line):	3 W

7.3 Environmental test requirements

The levels specified below apply to the instrument mounting platform.

7.3.1 Radiation

A one-year mission with low polar sun synchronous orbit (650 to 700 km altitude) has to be taken into account.

7.3.2 EMC

Usual requirements that apply on space projects can be considered at this stage.
The continuous residual magnetic momentum of the equipment shall be less than $0.1Am^2$.

7.3.3 Mechanical environment

Resonance modes

The resonance modes of instruments shall be above 400 Hz.

Sine

The applicable requirement for the sine vibrations is the following (qualification level): 20 g for f < 100Hz

Random

The applicable requirements for the random vibrations are the following:

X axis - qualification level	
Frequency (Hz)	PSD (g^2/Hz)
20-100	+ 3db/oct
100-400	ESA PSS level + 3db
400-2000	- 3 db/oct

Y and Z axis - qualification level	
Frequency (Hz)	PSD (g^2/Hz)
20-100	+ 3db/oct
100-400	ESA PSS level + 3db
400-2000	- 4 db/oct

ESA PSS level: PSD (g^2/Hz) = 0.05 * (M+20)/(M+1), where M is the mass of the equipment in kg.

Duration: 60 seconds for each axis.

Shock
The applicable requirements for the shocks are the following:

Frequency (Hz)	Level (g)
20	100 (TBC)
20-1000	linear
1000-3000	1000 (TBC)

7.3.4 Thermal I/F

In the normal operations, the global thermal control of the equipment will be passive.

Conductive interface

Temperature
It is guaranteed that the temperature at the conductive interface on spacecraft side is (TBC):

Spacecraft Mode	**T min(C°)**	**T max (C°)**
On-ground storage	-20	+40
Launch and early operation	0	+40
Normal operation	0	+40
Safe mode	-20	+40

Radiative interface

Internal interface : TBD

External interface:
For an equipment exposed to the space environment (vacuum, Sun, Earth) and to the external sides of the spacecraft. The following characteristics for the external materials will be taken into account:

Material	**a**	**e**	**Tmin**	**Tmax**
MLI	TBD	TBD	-150	+150
Radiator SSM	0.1/0.2	0.8	-40	+50

7.3.5 Electrical I/F

Each equipment is interfaced with the Power Conditioning and Distribution Unit (PCDU) of the S/C for the power and with the On Board computer (OBC) of the S/C for the TM/TC communications.
It is preferred to have two separated connectors, one for the power line and one for the data exchange line.

7.3.6 PCDU I/F

The power is interfaced with the Power Conditionning and Distribution Unit (PCDU) bus of the S/C. The I/F is described in the technical specification of the PCDU (can be delivered on request).

Mass grounding

A bounding stud (M3) is requested preferably close to base plate of the equipment.

7.3.7 OBC I/F

The equipment can be electrically interfaced with the On Board Computer (OBC) of the S/C (direct point to point link).

The I/F is RS422 type, with asynchronous link with full-duplex octets:

- one pair TX+, TX- transmits octets from OBC to equipment.
- one pair RX+, RX- transmits octets from equipment to OBC.

The equipment is always the slave. It sends data only when requested by OBC. The exchange data rate is 9470 bauds. The data from equipment shall be contained in 64 octets (if necessary OBC can regulate higher rates). Each data byte is sent using 1 start bit, 8 data bits, no parity bit, 1 stop bit. If needed, an additional checksum byte can be used to protect for bit error during data exchange.

When OBC is off-line, the impedance seen by the equipment is high.

The equipment shall be insulated using optocoupleurs ; the available current from OBC is limited to a maximum of 4 mA (at 4 volts) on the Tx+/- line.

The idle status of each TX line corresponds to: TX=1
TX+ and RX+ are at 5 V and TX- and RX- are at 0 V