MICROSCOPE mission phase B propulsion system activities at spacecraft level: requirements consolidation and assessment of neutralization and contamination effects

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Abstract: MICROSCOPE is CNES fundamental physics drag-free mission to test the Equivalence Principle in space with unprecedented precision. The project is in advanced Phase B stage, for a launch currently scheduled in 2011. In the nominal configuration, the drag-free attitude and acceleration control system is based on linear slit caesium FEEP actuators. Many engineering activities related to MICROSCOPE propulsion system have been performed at spacecraft level since the beginning of Phase B, directly by CNES or by ONERA in the frame of CNES R&T activities preparing future formation flying missions. The main results and conclusions are presented in this paper.

Nomenclature

\[ EPS(A) = \text{electric propulsion (sub-)system (assembly)} \]
\[ FEEP = \text{field-emission electric propulsion} \]
\[ FEP = \text{Equivalence Principle frequency} \]
\[ SCAA = \text{attitude and acceleration control system} \]

I. Introduction

MICROSCOPE mission (French acronym for MICROSatellite à trainée Compensée pour l’Observation du Principe d’Equivalence), funded under CNES Myriade microsatellite program, intends to verify the Equivalence Principle with a relative accuracy of \(10^{-15}\) (two orders of magnitude better than what has ever been achieved) by comparing inertial and gravitational mass of two bodies of different composition and density. A violation of the Equivalence Principle would give the evidence of a new interaction that is predicted by some quantum theories of gravity, and would confirm the interest of more accurate determination of post-newtonian coefficients. The test involves placing test masses of different materials on precisely the same orbit and maintaining the masses on this orbit by electrostatic forces. A difference in the forces required, due to a difference in the effect of gravity on the masses, will indicate an EP violation. The mission scientific payload is a high sensitivity electrostatic differential accelerometer developed by ONERA composed of two inertial sensors with test-masses in Platinum-Rhodium and Titanium alloy. A second differential accelerometer with both test-masses of the same material is used as reference. The instrument performance is strongly dependent on the noise environment. The altitude of Microscope low quasi-circular heliosynchronous dusk-down scientific orbit has been selected at 790km as a compromise between the conflicting needs of gravitational signal maximization and perturbing forces minimization.

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In order to compensate environmental external perturbations (residual atmospheric drag, solar pressure, magnetic torques etc.), during EP measurements the satellite is operated in drag-free mode. In this mode the S/C attitude can be inertial (the frequency of excitation of the equivalence principle is the orbital frequency, \(1.7 \cdot 10^{-5}\) Hz and the scientific measurement is performed over long periods, up to 120 consecutive orbits) or the sensitive axis of the instrument is rotated around the orbital pulsation vector \(f_{ep}\) (and \(f_{ep}\) can be selected at different values). The drag-free control system (called SCAA), entirely new with respect to Myriade family AOCS, is developed by CNES and is also used for instrument calibration. In the frame of a cooperation agreement between agencies, the micro-actuators of the SCAA are procured by the European Space Agency. The technology selected in 2000 as baseline for MICROSCOPE is field-emission electric propulsion (FEEP) using cesium propellant and slit-geometry emitters. This technology is under development by Galileo Avionica and Alta Space in the frame of ESA program Lisa-Pathfinder.

CNES propulsion activities in phase B have been focused on the consolidation of performance and interface requirements for the thrusters and on the assessment of propulsion effects at S/C level (mainly neutralization and contamination issues) in collaboration with ONERA DESP. Two feasibility studies of alternative propulsion technologies implementation have also been performed but will not be presented in this paper.

II. Consolidation of performance and interface requirements of the baseline EPS

The S/C baseline PDR layout is presented in Fig. 1. The retained propulsion system configuration makes use of 12 thrusters in partial hot redundancy, grouped in 4 identical blocks (called EPSA) located on the S/C Z walls, each including 3 FEEP actuators with dedicated tank and opening lid, 2 neutralizers, a PPCU and a support structure.

The main technical specifications can be summarized as follows:

<table>
<thead>
<tr>
<th>Specification</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Thrust range (per thruster)</td>
<td>1 to 150 (\mu)N</td>
</tr>
<tr>
<td>Total Impulse (per thruster)</td>
<td>3100 Ns</td>
</tr>
<tr>
<td>Thrust resolution</td>
<td>0.05 (\mu)N</td>
</tr>
<tr>
<td>Response time</td>
<td>(&lt;160)ms</td>
</tr>
<tr>
<td>Thrust noise – Log (PSD)</td>
<td>(&lt;0.01 \mu)N²/Hz</td>
</tr>
<tr>
<td></td>
<td>(&lt;0.01\cdot(0.1/f) \mu)N²/Hz</td>
</tr>
<tr>
<td>Beam divergence</td>
<td>(&lt;15°/40°)</td>
</tr>
<tr>
<td>Thrust knowledge</td>
<td>10%</td>
</tr>
<tr>
<td>Max power consumption</td>
<td>25.3 W</td>
</tr>
</tbody>
</table>

The FEEP thrusters are used in MSP (Stellar Propulsive Mode) and in MCA (Accelerations Control Mode), while MGT2 (Coarse Transition Mode) uses magnetotorquer bars actuation and kinetic wheel stiffness to align the satellite reference frame to the local geomagnetic field.

In MSP the thrusters are used (instead of classical reaction wheels) to perform 3-axis stabilization with a pointing accuracy of 0.04 degree on each axis. The attitude measurement is provided by the star tracker, derived to
get the angular velocity and filtered by a low-pass filter to eliminate high frequency noise and avoid FEEP high frequency excitation. The FEEP control law includes a large angle nonlinear algorithm (speed bias law) and a small angle linear algorithm. This mode is robust to the loss of one EPSA and allows to perform accelerometers switch on and in-flight test in Full Range Mode.

In MCA the thrusters are used also to provide linear acceleration control with very high accuracy \((10^{-12} \text{ m/s}^2)\) residual acceleration at FEP. The satellite structure and the orbit parameters have been carefully studied and chosen to achieve that goal (no eclipses during the 9-month-full-performance mission to avoid sudden solar pressure variations, no structural flexible mode under 4.5 Hz, no mobile elements). The main residual disturbance force to compensate is due to the air drag and should not exceed 25µN, while the perturbing torque can be up to 45µNm. All external perturbations have been modeled: drag (including the effect of the satellite speed and the atmospheric wind, taking into account stochastic air density variations extrapolated from CHAMP), solar pressure, Earth albedo, magnetic effects and gravity gradient. The total resultant perturbing force has a fixed component and a periodic component at orbital frequency due to the aerodynamics (the density is higher at the equator than at high latitudes), while the perturbing torque has a component at bi-orbital frequency (as the magnetic field in inertial pointing). For validation and simulation purpose, a phase B Matlab®/Simulink® simulator of MCA control loop has been developed with ESCAPE software application. The drag-free controller is similar to the one used in MSP, with the linear control forces in addition to the control torques for attitude control. A remarkable feature of Microscope drag-free system is the thruster selection logic algorithm. This iterative algorithm has been developed to find the \(T\) vector (with dimension 12, containing the commands to be sent to each thruster) that satisfies the equation \(F = M x T\) (where \(F\) is the vector with the three linear forces and the three torques to be applied on the base of the sensor measurements to null the external perturbations, and \(M\) is the 6x12 influence matrix whose coefficients are based uniquely on geometrical parameters) while minimizing the electrical power consumption criterion \(\min \sum_{i=1}^{N} T_i^2\).

The computation requires some 30 ms for 20 iterations carried out on a T805 processor and the presence on board of 5 tables of less than 250 parameters each (2 koctets for each table), and the result is quite precise (very low estimated induced error). The thrusters dynamics has also been taken into account in the simulator with the specified noise level of 0.1 µN/(Hz)^{1/2}, which is justified also by the fact that above the AACS control bandwidth (0.1 Hz) the propulsion noise is transmitted to the platform and has to be compliant with the proof masses control range.

The first positioning of the thruster blocs was aimed at enhancing X axis torque capability to enable kinetic wheel spinning up and down. The position of each thruster is the result of an iterative process that starts from the theoretical optimal configuration ensuring minimum torque control authority in case of single pod failure while minimizing the average (over the EPS) mean (over one orbit) thrust in case of single thruster failure in each EPSA. The minimum control capacity represents the minimum of the norm of forces and torques vectors applied by the thrusters in any direction, with one thruster at least at is maximal capacity, and is determined by Monte Carlo iterations. The results show that the minimum control capacity is 128µN with 4 EPSAs available, but it decreases to 64 µN (which is the lowest limit allowed to cover mission needs) when only 3 EPSAs are available. With the given constraints, the optimal configuration corresponds to have the thrusters of the same EPSA in a flat stellar geometry on the plane perpendicular to the CoG direction.

Figure 2: Visualization of exhaust jets wrt spacecraft surfaces
The final angles have been defined after CATIA visualization and processing (fig. 2) to take into account the spacecraft geometrical constraints, which means basically to avoid direct impingement of the exhaust jets (specified as divergence elliptical cones with angles of +/-15° and +/-40° including 95% of the emitted ion current) on solar panels and antennas.

An example of simulated thrust profiles over several orbits in inertial mode is provided in fig. 3 (100% corresponds to the maximum specified thrust of 150µN). It can be noticed that all thrusters are used, although the solicitation level is not homogeneous (in case of failure of one thruster or in spinning mode the repartition would be different); the difference between the average EPSA utilization rate is lower. The average thrust level in inertial scientific mode is rather low (22.5µN). In spinning mode it is higher (33µN). Peak values do not exceed 90µN in these modes. The maximum required thrust levels (up to 150µN) are needed in degraded scenarios (one thruster off), during transitions between modes (in particular from MGT2 to MSP and from MCA inertial to MCA spinning) and during instrument calibration phase (MCA calibration). The failure mode with 1 thruster off in each EPSA would actually require higher thrusts than the maximum specified to compensate the perturbations in spinning mode, which could not be supported.

For realistic thermal load calculations, the maximum utilization rate of a single EPSA in degraded mode (1 EPSA off) has been estimated at 175% over 1000s (the maximum with all thrusters at maximum thrust being 300%), while the peak value over 100s is 200%. The peak value over the whole EPS is 476% and the guaranteed commanded maximum is 717% (but the possibility of a 1200% command has to be taken into account for neutralization current budgets).

The estimation of the total impulse to be specified (3100 Ns per thruster) takes into account the thrust levels needed during 2 months of instrument calibration followed by 10 months of scientific operations (5 in inertial mode and 5 in spinning mode) in the worst admitted scenario (only 2 thrusters out of 3 operating in each EPSA) plus some margin for the command noise and the selection algorithm noise.
Instantaneous thrust discontinuities have been limited in magnitude and duration to avoid saturation of the accelerometers. This applies in particular to arcing phenomena translating into thrust interruptions, that should not exceed 20ms in duration and appear with rate lower than 1-2 per day not to produce scientific data loss.

The response time is a key-performance for the drag-free actuators, since an excessive delay would destabilize the control-loop. The 160ms expected for 10µN steps, added to the unavoidable delays due to filters, control algorithms and transmission links, leave very little margin.

III. Assessment of spacecraft electric charge compensation phenomena

The electrical interface between the EPS and the spacecraft has been studied to take into account phenomena like effective neutralization, thrust fluctuations related to SC potential variations and microperturbations due to current flows through the satellite structure. From the ground logic point of view, a common ground for all the EPSAs and the S/C structure (fig. 5) has been preferred to alternative options (isolated EPSAs or cross-strapped floating EPS with independent neutralizer for the S/C) for simplicity reasons (no need of galvanic or optical insulation between OBC and PPCU) but also to assure effective sharing of neutralization function between neutralizers (hot redundancy and increased flexibility/optimization) and to have the same common point as reference to estimate the thrust produced by the engines. In fact, the produced thrust is linked to the S/C floating potential $V_{sc}$ by the relationship $F = K \cdot (I_e - I_a) \cdot \sqrt{V_e - V_{sc}}$, where $I_e$ and $I_a$ are the emitter and accelerator currents, $V_e$ the emitter voltage.

Figure 4: Preliminary simulations of Microscope floating potential variations

Figure 5: Microscope EPS Ground Architecture
$Z_{gs} = 10k\Omega$, Zepsa1=Zepsa2=Zepsa3=Zepsa4=100m$\Omega$

Figure 6: Expected Current Loops on SC structure due to EPS utilization
Preliminary simulations have been performed with ONERA/DESP support to assess the expected variations of spacecraft potential as a function of total FEEP emitted current, number of active thrusters and latitude of the orbit (fig. 4). These simulations are based on experimental characterizations of the neutralizer engineering models produced by TAS and are extremely sensitive to variations in the expected emission laws (function of cathode temperature, anode voltage, plasma density and satellite wake effects). On Microscope LEO orbit, the ionosphere plasma is relatively dense (between $10^4$ and $10^6$ cm$^{-3}$ with a worst case of 2500 cm$^{-3}$ according to 1996 IRI model) and cold (0.1-0.3 eV) depending on latitude and time. The Debye length varies between 0.2-4cm (low latitude) and 4-9cm (high latitude), but can go beyond 40cm in polar stormy conditions. Appropriate (though simplified) laws are used to estimate the currents collected by the S/C (planar probe for small $\lambda_d$, OML spherical model otherwise), photoemission, secondary electron emission and effects of solar array (surface: $3.26m^2$) including maximum voltage on the interconnections (33.6V). The external MLI coating of the satellite (surface: $5.7m^2$) has been studied in both conductive (PCBE on aluminized kapton) and not conductive case (Mapatox). The results show that a maximum total neutralizer current of 11.2A is needed to balance the current budget of the satellite. When 4 neutralizers are used, little fluctuations of the spacecraft potential are seen even at maximum expected thrust levels (89µN per thruster): some -30V (that can actually reach -200V in worst case scenarios with extremely depleted auroral regions combined with strong solar wind and the satellite in eclipse phase). Actually, two simultaneously active neutralizers are sufficient to guarantee fluctuations of amplitude lower than 150V, but in case of failure of one of them the spacecraft could rapidly charge up to 1kV at high latitudes. A possible strategy to minimize power consumption could be to have only 2 neutralizers on in MCAi, while 4 would be required in MCAs and MCAc but could not be used during eclipses.

From the thrust performance point of view, potential variations of up to 200V are acceptable since the emitter current is not affected, and the effect of the Vsc variation on the total thrust magnitude does not exceed 1.8% of the commanded value (scale factor effect), which is within the steady-state requirement specified as:

$$F_c = k_0 + (1 + k_1) F_c^\infty + k_2 F_c^\infty + \text{Re}(Fc) + \text{noise},$$

with $k_0<2\mu N$, $k_1<0.05$, $k_2<222*10^{-6}\mu N^{-1}$, Re<1%Fc

The variable component due to Vsc fluctuating behaviour over the orbit has to be taken into account in the total thrust noise budget.

A potential identified drawback of the selected electrical configuration consists in the possible perturbations due to Laplace forces $\vec{F} = i*\vec{d} \times \vec{B}$ generated by current loops within the structure of a single EPSA or within the satellite structure (if the neutralizer of an EPSA is used also to compensate the charge emitted by the thrusters of other EPSAs) interacting with the Earth magnetic field (fig. 6). In the worst case (two neutralizers belonging to the same EPSA used to compensate the charges of the whole satellite) the estimations give a low frequency force of about 0.6µN that should be easily compensated by the SCAA. Other types of microperturbations have also been assessed and confirmed as negligible. For instance, the Lorentz force $\vec{F} = Q*\vec{V} \times \vec{B}$ due to the S/C charge and velocity in Earth magnetic field depends on $V_{SC}$ because $Q = C_{SC} * V_{SC}$, where $C_{SC}$ is the equivalent spacecraft capacity estimated at some 75pF. The worst case effect should not exceed 30 nN.

![Figure 7: Neutralizer characterization in different plasma environments (Pw = 5.5W; Va = 200V; theta = 0°)](image-url)
In order to consolidate the obtained results, detailed simulations will be performed by ONERA/DESP taking into account realistic neutralizer behaviour, and possibly also wake effects and full spacecraft dynamics (representative geometry, ion/electron source plume models, orbit detailed physical characteristics).

A key parameter in all potential calculations is the actual I-V emission law of the neutralizer in representative conditions. In fact, emitted current space charge saturation effects that could be masked by the utilization (during ground characterizations) of extractive polarized targets, might prevent the neutralizer from fulfilling its task, leading to mission perturbations or even ESD destructive events. The only way to get confidence in the selected neutralization device capabilities is to test it in representative plasma conditions and directly assess the favourable contribution of Microscope orbit plasma (mainly made of positive ions). This has been done in ONERA/DESP facility JONAS. Some preliminary results are presented in fig. 7. The plasma densities have been regulated at 5 different levels (from 0 to $2 \times 10^5$ cm$^{-3}$). The plasma has clearly an enhancing effect on the extraction efficiency, thus reducing the cathode (spacecraft) potential needed to reach the target neutralization current. There is no identified saturation, all plotted lines converging at a value (around 11mA) where the plasma environment has no effect (all produced electrons are emitted).

### IV. Assessment of plume contamination risk and effects

Cesium FEEP thrusters have never been tested in space, so no flight experience is available concerning the risk of propellant backflow on the spacecraft and the effects of such a contamination on the satellite sensitive exposed parts. To address this issue, two type of system level activities are necessary and have been performed in collaboration with ONERA/DESP (CNES R&T program): numerical modeling of the spacecraft in its plasma environment to estimate the amount of propellant coming back and localize the most exposed regions, and experimental verification of contamination effects on thermal coatings and solar cells.

#### A. Numerical modeling of propellant backflow

The numerical modeling has been performed with the code SPIS (Spacecraft-Plasma Interaction Software), developed by ONERA with ESA and CNES support. The GMSH model of MICROSCOPE is presented in fig. 8. The twelve small flat cylinders disconnected from the spacecraft represent the exit planes of the thrusters (isolated to allow a more accurate orientation of the slits). The first step was introducing active plasma source models (ions and electrons) in SPIS. The FEEP model, quite complex because not axially symmetric, was based on input plume data provided by Alta Space (typical energies in the 10keV range). Since the spacecraft mesh was planned to start at centimeter scale close to the thruster exists, the approach consisted in getting plume data on that boundary at around one centimeter from the FEEP slit, and modeling only the physics downstream. The accelerating potential gradient was considered to be completely contained within the very near field region (excluded from system scale simulations). This allowed to restrict the boundary data to ion and neutral fluxes, while the potential was assumed to be close to zero (plasma potential).

![Figure 8: MICROSCOPE GMSH geometrical mesh model used in SPIS simulations](image-url)
In the FEEP plume, both electrostatic (i.e., collective interactions of electrons and ions from the beam and the ambient, expressed through Poisson equation) and resonant charge exchange reactions (CEX) where taken into account. In CEX reactions fast Cs+ ions are transformed into fast Cs atom when they pick up an electron from a slow neutral Cs, which in turn becomes a slow Cs+ ion; since it is basically an electron jump from Cs to Cs+, a CEX reaction involves no momentum transfer and has large cross sections when resonant (it is the case for homogeneous ions and neutrals). As for the neutral flux estimation, an arbitrary ionization rate (mass efficiency) of 80% was assumed (current specification is 70%, declared target 90%, no measurement is available yet). Droplet emission has not been taken into account because the physics involved is very different (drift under flow pressure, charging, evaporation, and fast ion sputtering effects) and is considered a challenging target for future simulations. These CEX interactions are typical of electric thruster plumes, whatever their type, since they always emit fast ions and slow neutrals. In case of xenon ions and higher electron temperature (Hall thrusters) the major concern is the energy these initially slow ions can get when they are ejected from the plume by the potential gradient resulting from barometric electron distribution. It may be larger that ion sputtering threshold for SC coatings nearby, in which case their large fluxes can yield significant material erosion. For FEEP thrusters the flux of slow CEX Cs+ is drastically smaller but the ions might directly contaminate coatings (cesium being an extremely reactive alkali metal capable of efficiently degrade polymers through chemical processes). The fast Cs+ density is plotted in fig. 9 on a test case and on the S/C (iso-contour surfaces at n=10^{11} m^{-3} and n=10^{12} m^{-3}, yellow and orange respectively).

In the simulations, each FEEP is assumed to emit 300 µA of Cs+, which corresponds to an average thrust level of about 30µN. For the ambient plasma, a typical density value of 10^{11} m^{-3} drifting O+ ions at 7500 m/s orbital speed was taken. Ion and electron temperatures were set to 0.1 eV. From a numerical point of view, all ions (O+, drifting and slow CEX Cs+) were simulated by PIC method (Particle-In-Cell, Monte Carlo). The electrons were considered Maxwellian (barometric law). This allows using the implicit solver of non linear Poisson equation existing in SPIS. It makes the simulation stable even in case of Debye length shorter than the cell size (which is often the case in LEO type environment). A Lambertian emission law is used to describe the distribution of neutrals.

The CEX volume reaction rate (fig. 10a) has a quadratic dependence in thruster efflux, with a linear dependence both in the neutral and fast ion densities. The CEX slow ions then follow their own dynamics due to the local electric field: they are ejected rather radially out of the plume and fill the whole computation space, even behind obstacles (fig. 10b). Both ambient ions and electrons are strongly depleted in the wake (fig. 11), due to the ion velocity in the spacecraft frame and to the electrons own space charge, which induces a slightly negative potential.

The fluxes of CEX Cs+ are finally presented in fig. 12 (left side: solar cells direction view). The integration over the whole mission (1 year) taking into account corrective factors representative of the expected thrust profiles (presented in fig. 3) gives an estimation of the total cumulated deposit due to CEX ions: 10 Å/yr range close to FEEP nozzle, close to 1 Å/yr on other locations of the same spacecraft face, and below 1 Å/yr on solar arrays.

Another simple mechanism of contamination is direct impingement of neutral Cs from the plume (the Lambertian distribution representing them is broader than the ion beam ejection cone, so not every S/C surface is avoided). This direct neutral flux is also presented in fig. 12 (right side). Applying the appropriate corrective factors we obtain values in the range 0.2 to 200 Å/yr for 10^{-6} to 10^{-3} A/m² (i.e. green to red in the color scale). Neutral cesium deposit is thus of the same order of magnitude as the one from CEX Cs+, even a little larger locally.

Figure 9: Fast FEEP Cs+ density (test case on the left, iso-contour surface on MICROSCOPE on the right)
Figure 10: Volume CEX reaction rate and CEX slow ion density (iso-surfaces with $n = 10^7$, $10^8$ and $10^9$ m$^{-3}$).

Figure 11: Density of ambient species (O$^+$ on the left, electrons on the right).

Figure 12: Fluxes of CEX Cs$^+$ [Å/h] (left) and direct fluxes of neutral Cs [log(A/m$^2$)] (right).
B. Experimental characterization of contamination effects

The experimental direct characterization of caesium contamination effects on various samples has been undertaken in ONERA/DESP facilities. The objectives of this activity are the following:

- in-situ measurement of surface resistance variations of thermal coatings
- measurement of variations in thermo-optical properties (absorptivity and emissivity) of thermal coatings (both conductive and isolating samples) due to permanent chemical reactions with Cs (measurements to be performed after re-evaporation of the contamination)
- measurement of variations in the performances (generated power) of solar cell samples in both cold and warm conditions
- measurement of variations in the optical transmissivity of cover glass samples
- in-situ measurement of variations in thermo-optical properties of thermal coatings due to the presence of a contaminant layer (before re-evaporation) with calorimetric indirect methods

The first three tasks have been completed, work on the last two is in progress.

1. Variation of thermo-optical properties of thermal coatings

Several test campaigns have been performed on various lots of samples. The first results, quite difficult to exploit, were obtained by measuring (in situ) the change in conductivity of various materials (including Betacloth and SSM) with a deposition at ambient temperature of up to 1000Å (with large uncertainty) of Cs (fig. 13a). The samples were subsequently analyzed at CNES with a scanning electron microscope combined with a thin-window energy-dispersive X-ray detector, and by FTIR (Fourier transform infrared spectroscopy). Some results obtained with SSM are presented in fig. 13b (note the appearance of a band of absorption at 1230 cm\(^{-1}\) in replacement of the C-F link band at 1290 cm\(^{-1}\)) and 14 (oxygen and caesium have the same surface distribution). The most critical identified problem in the test procedure was the control of the thickness of the contaminant layer to be deposited. Starting from the second campaign, the tests have been performed in the COPHOS facility, where the samples can be controlled in temperature. The cesium ampoule supply is broken under nitrogen atmosphere, than transferred into the facility oven. The samples are surrounded by a Cs trap (cooled at -170°C) that avoids contaminant uncontrolled diffusion. Many characterization tests have addressed the control of the oven cesium efflux, that appears to be quite irregular because of the partial oxidation of the surface layer (many cycles are needed before it stabilizes). The re-evaporation rates are more predictable, but they exceed the theoretical estimations (based on extrapolated data).

![Figure 13: Surface resistance variation measurements (left) and SSM FTIR (right)](image_url)

![Figure 14: SEM/EDX cartography of SSM contaminated sample (global surface and single species identification: from left to right, Fluor, Oxygen and Caesium)](image_url)
This is obviously positive because Cs residence time on the exposed surface will be reduced: even at negative temperatures, the re-evaporation rate is not negligible (fig. 15a).

Within the COPHOS set-up, a first set of four samples (MAPATOX, PCBE on Nomex, Scotch Kapton 3M92 and MYLAR, fig. 16) were cooled down at -50°C, contaminated with about 2000Å of Cs, than warmed up to 80°C to re-evaporate the contaminant, transferred under vacuum to a spectrometer UV-Vis-NIR to measure the absorptivity (an exemple of result is shown in fig. 15b), than analyzed with a Gier and Dunkle under nitrogen to determine the emissivity. Only little modifications of the emissivity are observed. The absorptivity is increased in the case of polymers (Mylar and Kapton). After extended air exposure, all materials get back to their initial properties (probably the changes are due to surface reduction rather than chemical reactions).

A second set of four samples was then tested in similar thermal conditions: silver SSM with and without ITO, white paint SG121 FD, and RSR on Aluminium. This time about 2500 Å of Cs was deposited. The results are presented in the bar charts of fig. 17: the emissivity does not change appreciably; the absorptivity increases, especially for the white paint and the SSM (the ITO coating seems to have protective properties).

The thermo-optical effect of the resident contaminant layer will be addressed by next experimental campaign.
2. Variation of photovoltaic cells performances

A test campaign has been dedicated to active samples (solar cells) in-situ characterization. The samples are three individual junctions (top, middle and bottom) of a UTJ cell type GaInP2/GaAs/Ge (by Spectrolab), plus two silicon cells used in sun sensors. An external flasher (40W, Xe lamp) is used to produce light impulses and illuminate the samples through a quartz window (heated at 50°C to avoid Cs deposition). The short-circuit current is acquired by an automatic system synchronized with the flash trigger (acquisition time: 5ms; sampling frequency: 0.05ms; trigger frequency: 0.05 Hz). The test set-up is shown in fig. 18. An external cell is used as reference to prevent undetected drifts in the flash behaviour. For every cell the test protocol (fig. 19) consists in an initial moderate contamination in warm conditions (75°C, which is the expected operating temperature of the cells on Microscope), followed by a temperature decrease down to -100°C to guarantee thickness layer build-up with the same moderate incoming flux, then slow re-evaporation (at -20°C, corresponding to an evaporation velocity of about 0.5Å/min) and finally strong contamination at temperature close to operational conditions (50°C).

A huge quantity of data has been collected during the test campaign. The interpretation of the results has been sometimes difficult because of problems with the stability of the flasher and the unavoidable incertitude on the exact quantity of deposited caesium. Some results are presented in fig. 20.

![Figure 18: Photovoltaic cell test set-up schematic and picture](image1)

![Figure 19: Solar cell contamination test protocol](image2)
The conclusions are summarized in the tables of fig. 21: the performances of a UTJ cell under a realistic flux of Cs (<10Å) in nominal operating conditions (75°C) are unaffected; a deposit cumulated in cold conditions (during eclipses) can produce significant power losses, but the effect is almost entirely reversible (a margin of 2% should be sufficient). The sun sensor cells are not very sensitive to Cs (no permanent effect).

![Figure 20: Photovoltaic Cells results (examples): Bottom junction at 75°C and Top junction at -100°C](image)

![Figure 21: Tables with synthesis of the photovoltaic cells contamination test](image)

### V. Conclusion

In the frame of MICROSCOPE phase B activities, important efforts have been dedicated by CNES (with the support of ONERA/DESP) in order to consolidate the baseline propulsion system requirements and to assess neutralization and contamination effects.

The obtained results, summarized in this paper, are noteworthy and might find useful applications also in future drag-free or formation flying programs.

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