FIELD EMISSION ELECTRIC PROPULSION PLUME EFFECTS ON MICROSCOPE SATELLITE

L. Jolivet, B. Debray
CNES, 18 avenue Edouard Belin, 31401 Toulouse Cedex 4 (France)
Tel : (33).5.61.27.46.57 - Fax : (33).5.61.28.29.92 - e-mail : Laurent.Jolivet@cnes.fr

ABSTRACT

Microscope is a scientific mission planned to be launched in 2006. The major technologic innovation for the platform is the use of Field Emission Electric Propulsion (FEEP) that will be operated as propulsion subsystem for the first time. The issue of the interactions between the FEEP thrusters and the spacecraft is assessed in this paper. It appears that specificity of FEEP technology (high voltages, liquid metal propellant) prevents law-scale approach from more commonly used electric thrusters. FEEP plume effects require a dedicated study and little information is available at this time. As a preliminary work, a reflection is given on each typical electric propulsion plume effect with emphasize on the spacecraft electrostatic equilibrium.

GLOSSARY

FEEP	Field Emission Electric Propulsion
S/C	Spacecraft
CNES	Centre National d'Etudes Spatiales (French Space Agency)
ESA	European Space Agency
EP	Equivalence Principle
EPS	Electric Propulsion Subsystem
EPSA	Electric Propulsion Subsystem Assembly
BC	Baseline Configuration
RC	Redundant Configuration
EMC	Electro Magnetic Compatibility
LEO	Low Earth Orbit
SPT	Stationary Plasma Thruster
IRI	International Reference Ionosphere

1. Introduction

Microscope (MICROSatellite à traînée Compensée pour l'Observation du Principe d'Equivalence) is the fourth CNES project based on the Microsatellites line of products. The scientific objective of the mission is to test the Equivalence Principle (EP) with an accuracy of 10^{-15} about three orders of magnitude better than the present accuracy with on ground experiments. To reach this goal, it is necessary to compensate the primary perturbation forces upon the satellite. These forces include solar radiation pressure, atmospheric drag, outgassing from system components, and magnetic and gravity gradient torques. Compensation of these forces, named "drag compensation" in the following text, involves and will demonstrate the use of FEEP (Field Emission Electric Propulsion) thrusters. The thrusters have to be capable to operate during one year in a continuous mode. That will be the first time that the FEEP technology will be used in flight as a propulsion subsystem and consequently there is no feedback from previous flights concerning the plume effects on the spacecraft. It must be feared that damaging effects could appear before the end of the mission due to the intensive use of the thrusters. Therefore, CNES has started a study on the sputtering, contamination and electrical effects at the spacecraft level in order to estimate the impacts of such effects on Microscope over the mission life. Purpose of this paper is to show the preliminary results of this study.

2. Microscope satellite and FEEP Electric Propulsion Subsystem

The European Space Agency (ESA) is in charge of the supplying of the Electric Propulsion Subsystem (EPS), developed and built by ALTA S.R.L, to CNES. The Microscope EPS is described in details in another paper [1]. This chapter sums up only the main Microscope and EPS characteristics that can take place in plume effects.

Microscope is a microsatellite with an estimated mass of 170 kg. The spacecraft (S/C) orbit is a low (650 to 700 km altitude) polar heliosynchronous and circular orbit, with an ascending node at 6 or 18 hours (local time). The expected mission duration is one year. The Microscope EPS is composed with four identical Electric Propulsion Subsystem Assembly (EPSA) units (Figure 1). Two EPSA configurations are presently considered for implementation onboard Microscope:

- a 2 thrusters per EPSA configuration (called "Baseline Configuration"-BC)
- a 3 thrusters per EPSA configuration (called "Redundant Configuration"-RC)

For both configurations, there is one neutralizer per pod (plus a cold redundant one).

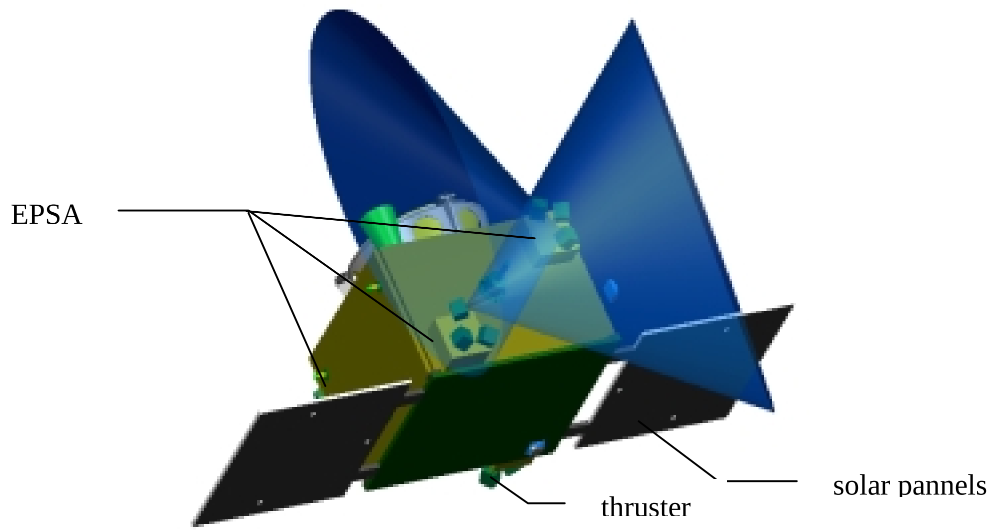

Figure 1: schematic view of Microscope. The FEEP EPS shown here is in Redundant Configuration (3 thrusters) and two plumes from thrusters have been drawn. 95% of the ions emitted by a thruster are ejected in the corresponding cone.

ALTA S.R.L is developing the FEEP technology selected in the frame of the Microscope project. To sum up the main characteristics, a metal propellant (cesium) is kept in a liquid form in a tank by a heater. Cesium flows through a slit by capillarity and is ionized and ejected by a strong electric field between an emitter (up to 5 kilovolts positive) and an accelerator plate (up to 7 kilovolts negative). A neutralizer in the vicinity of the thruster emits electrons in order to neutralize the ion beam (Figure 2).

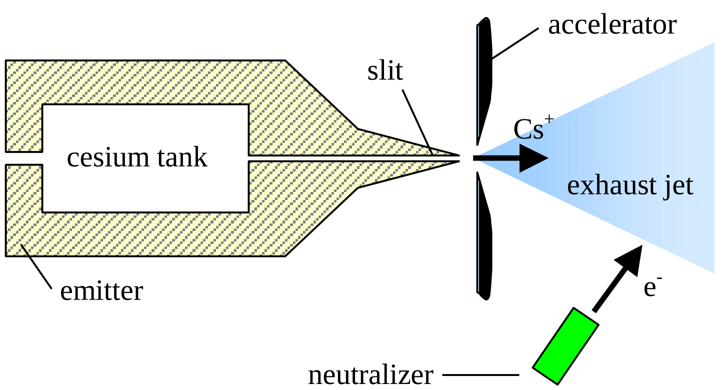

Figure 1: schematic view of an ALTA S.R.L FEEP thruster

Such technology suits very well for the drag compensation on Microscope [1]. Main characteristics of interest for the study of plume effects are summarized hereafter:

- The commanded thrust range from 1 µN to 150 µN for BC and RC configurations.
- Propellant is cesium. The propellant mass is not defined at the present time but must be lower than 40 g per thruster in the BC and 75 g per thruster in the RC.
- The geometry of the exhaust jet is specified by a cone, the exhaust jet cone, which is defined by the slit geometry itself and by 2 aperture angles:
  - in the plane defined by the slit and the center axis of the jet, the half angle of aperture is 15°,
  - in the plane perpendicular to the slit, the half angle of aperture is 40°.

  95% of jet ion beam current shall remain inside this exhaust jet cone.

3. Preliminary study of plume effects on Microscope

The common plume effects observed in electric propulsion are the perturbation of the S/C electrostatic background, the contamination and the sputtering of S/C sensitive systems, the creation of perturbing torques and Electro-Magnetic Compatibility (EMC) effects. When such effects are estimated as damaging for the mission, they can lead to a modification of the S/C layout. That is the reason why a study of the plume effects is useful even in the earliest phases of a project. As written previously, Microscope will be the first mission using FEEP technology as propulsion subsytem. The consequence is a total absence of in-flight data concerning FEEP plume effects. Microscope project is presently at the end of phase A and the main objective of this first study is not an accurate assessment of the plume effects but rather a preliminary understanding of the different plume effect phenomena linked to the FEEP technology specificity.

This preliminary study has been focused on the S/C electrostatic equilibrium, which is developed in the next chapter. Sputtering, contamination, torques and EMC effects are then discussed in a less detailed way at the end of the paper.

4. Electrostatic equilibrium of Microscope spacecraft

The S/C potential at steady state results from an electrostatic equilibrium between the different currents collected or emitted by the spacecraft. For a passive satellite, i.e. no current emitted by S/C equipments, the S/C potential is only due to the space environment surrounding the spacecraft and its interactions with materials from the S/C. These interactions are mainly the photoemission and the secondary electron emission. Photoemission is the emission of cold electrons due to the high-energy photons emitted by the sun and collected by the spacecraft. The secondary electron emission is the emission of cold electrons consecutively to the impact of ions or electrons of high energy from the surrounding plasma. The spacecraft is an electrically isolated body, hence the sum of currents is zero at the steady state.

$$\sum_{e,i} I_{e,i} = 0 \qquad (1)$$

The S/C potential is self-adjusted in order to verify this equation. The next figure (Figure 3) shows the different currents between the spacecraft and the space environment. A typical potential curve is reported on the figure. The region of potential drop between the space environment and the spacecraft is called the sheath.

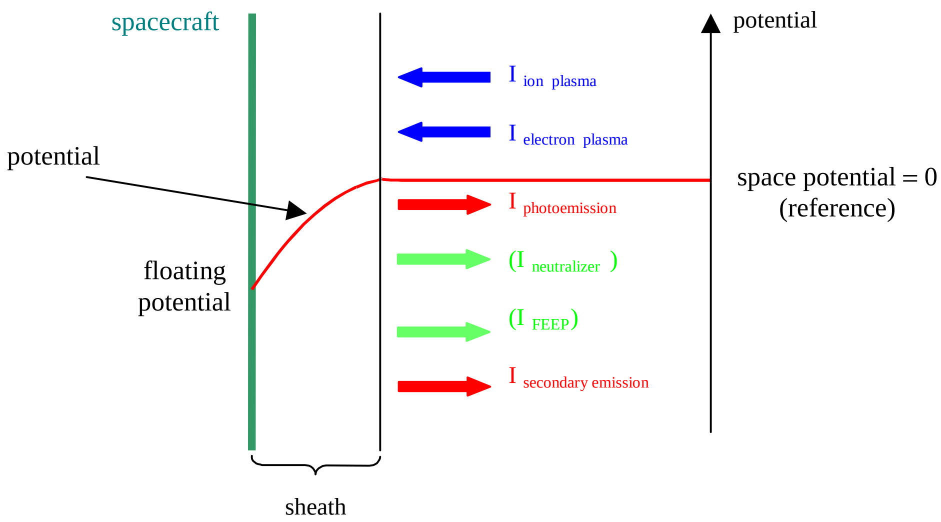

Figure 2: a typical potential curve between a spacecraft (left) and the space (right, reference potential = 0). At steady state, the S/C potential is self-adjusted to balance the currents (floating potential).

At high value about several thousands V.m^{-1}, the electric field resulting from the potential drop between spacecraft and space environment can initiate damaging electrical discharges. Sheath is usually thin, about a few centimeters in Low Earth Orbit (LEO), so the S/C potential must stay close to the surrounding plasma potential to avoid such hazardous and damaging phenomena. In LEO (400-1500 km), the ionospheric plasma density is in the range 10^4 to 10^6 cm^{-3} and the electron temperature is about 0.1-0.3 eV. In presence of such cold dense plasma, the S/C floating potential is very close to the space potential, about few volts [2] and therefore the Microscope orbit is safe concerning arcing phenomenon for a passive spacecraft. But the FEEP EPS onboard Microscope is a highly active equipment. FEEP thrusters are ion current sources that can strongly change the electrostatic equilibrium of the spacecraft. Electron current sources (neutralizers) will be implemented to counterbalance the emission of ions on the electrical standpoint. Electrons are produced in a neutralizer by a thermoionic cathode. Two functions can be distinguished for the neutralizers:

- the neutralization of the space charge in the vicinity of the thruster (local)
- the compensation of the total ion current emitted by the FEEP EPS (global)

Local function:

One potential objective at the local level is to guarantee proper neutralization of the space charge in the vicinity of the ion emitting region. This function is very important for thrusters providing higher thrusts like Hall Effect Thruster SPT-100 where the ion density at the thruster exit plane is about 10^{12} cm^{-3} and the ion drift energy is about 300 electronvolts. For such thrusters, without neutralization a strong positive space charge would appear due to the large amount of ions emitted and it would prevent further emission of ions. Functional properties of the thruster would be strongly altered. For FEEP thruster, situation is very different. The ion density is very low, about 10^9 cm^{-3} at the accelerator exit plane, and the ion drift energy is several thousands eV. The space charge is very low and high velocity ions will not be strongly perturbed. Considering the Child-Langmuir equation giving for a simple geometry the maximum current density $j_{\max}$ for a space charge limited emission:

$$j_{\max} = \frac{4}{9}\varepsilon_0\left(\frac{2e}{m}\right)^{\frac{1}{2}}\frac{V_0^{\frac{3}{2}}}{L^2} \qquad (2)$$

where ε_0 is the dielectric permitivity of vacuum, V_0 is the acceleration potential, L is the acceleration distance, e is the elementary charge and m is the particle mass. Cesium ion mass and Xenon ion mass are very close (m_{Cs} = 132.9 a.m.u, m_{Xe} = 131.3 a.m.u). Hence, for a given acceleration distance, FEEP cesium maximum current density is about two order higher than SPT-100 Xenon maximum current density due to the voltages applied. On the contrary of Hall Effect thrusters, FEEP thrusters work without neutralizers. Several tests have been performed in test bench without neutralizers and show good functional properties. It must be pointed out that an influence of the background chamber density in the neutralization process exists and has been reported in reference [3]. Effective space charge neutralization can improve the thruster functional properties but it is not the purpose of this paper to describe in details the topology of electric field near the thruster. More information can be found in reference [3]. It must also be noticed that typical ionospheric density will be sufficient to provide the electrons necessary to the neutralization of the plume only few centimeters away from the thruster.

Global function:

The objective is to counterbalance the global ion current emitted by the thrusters to keep the S/C potential constant. If a positive ion beam is ejected from an electrically isolated spacecraft, it will lead to a negative charge on the spacecraft. Even a small unbalance between the ion and electrons currents can result in rapid field build up around the spacecraft. The relation between the thrust F and the ionic current I emitted is given by the equation:

$$F = \psi I_i \sqrt{\frac{2V_e m_{Cs}^{+}}{e}} \qquad (3)$$

where Ψ is a plume geometry factor, I_i the ion current, V_e is the potential applied to the emitter, m_{cs} is the cesium ion mass and e is the elementary charge. A cesium FEEP thruster provides roughly 1 µN per 10 µA of emitted current. Hence, a thruster operating at its maximum thrust requires a current about 1.5 mA. Representing the spacecraft by a conductive sphere of one-meter radius, the capacitance C of this sphere will be of the order of 10^{-10} farads. A net unbalanced flow of 1 mA will result in a change in potential ($dV/dt=I/C$) of 10^7 volts/second! In fact, such potential drop will never be reached. The negative S/C charging is limited by the recollection of ions emitted when the spacecraft structure reaches the kinetic energy of ions. At negative value near $-V_e$, ions emitted will be recollected (consequently thrust is not produced). Emitter voltage for the FEEP system onboard Microscope is 5000 V. In case of no global neutralization, the thrust will be perturbed (mission problem) and the spacecraft potential can go down to this value which is very hazardous for the spacecraft (risk of electric arcs). Speed of charge build up means that electron current flow cannot be regulated by an electronic control system but must be assured with a self-regulated system.

In a nominal situation, electric risks linked to the FEEP use are avoided. From the plume standpoint, the main problem can come from a dysfunction in the global function of neutralization (neutralizer breakdown, late ignition of the neutralizer) leading to electrical hazards. The neutralization issue is very difficult to assess on earth because it is tightly linked with the in-flight conditions. The local neutralization is more or less present in a test bench with the background density [3] and the global neutralization is not taken into account due to the grounding of on-earth experiments. Even if neutralizer is not required on-earth, its major rule must be kept in mind for the elaboration of ignition procedures or qualification of the neutralizers. It seems major to fully qualify the neutralizers especially by including them in the thruster lifetest.

As written before, the main problem can arise with a neutralizer breakdown that leads to a strong and fast S/C potential drop. In fact, the space environment can attenuate this electrical effect with an enhanced collection of ionospheric ions by the spacecraft. In this preliminary study, a rough estimation of the space environment influence in the global neutralization of the FEEP beam has been made.

Evolution of the spacecraft potential without neutralizers

Accurate assessment of the electrostatic behavior for a given satellite requires a good knowledge of the ionospheric plasma, the satellite geometry, and the S/C material physical properties. Complex situations can only be handled through specific experiments or numerical simulations. For this preliminary study, some basic analytical laws have been used to calculate the budget of currents and therefore to estimate the floating potential of the spacecraft.

An appropriate equation for a rough assessment of the budget current is:

$$\frac{1}{4}n_e q_e S_{sat}\sqrt{\frac{8k_B T_e}{\pi m_e}}\exp\left(\frac{q_e V_{sat}}{k_B T_e}\right) - j_{photo-emission}S_{solar} - S_{cross}q_i n_i v_{sat}\frac{2|q_i V_{sat}|}{m_i v_{sat}^2} + I_{FEEP} = 0 \qquad (4)$$

where n_i and n_e are the ionospheric ion and electron densities, k_B is the Boltzmann constant, T_e is the ionospheric electron temperature, q_i and q_e are the ion and electron charges, m_i and m_e are the ion and electron masses, v_{sat} is the spacecraft velocity along its orbit (~ 7500 m.s^{-1}), V_{sat} is the S/C potential, $j_{photo-emission}$ is the electron current density emitted by photo-emission (~ 5 nA.cm^{-2}) and I_{FEEP} is the currend emitted by the FEEP thrusters. S_{sat}, S_{cross} and S_{solar} are S/C geometric factors. S_{sat} is the S/C total surface, S_{cross} is the S/C equivalent cross section (sphere approximation) and S_{solar} is the surface exposed to the sun. These factors are calculated from the Microscope preliminary geometry indicated in figure 4. The first term gives the electron collection; it can't moderate the FEEP ion emission because both electron collection and ion emission tend to lower the S/C potential. Due to the low electron temperature, its contribution in the current budget is negligible for S/C potential lower than few negative volts. The second term is the electron current created by photoemission, it is relatively low and is present whatever the EPS state is, off or on. Hence, it can't counterbalance the current emitted by the thrusters. The only mean to compensate the FEEP current is a large collection of the ions in the space environment. Due to the large negative potential induced by an unneutralized ion beam, the sheath extends itself over a large distance in order to collect the necessary ions to have a total current of zero. Therefore, it is appropriate to choose the equation of orbited motion limited theory [2] as third term for the ion collection.

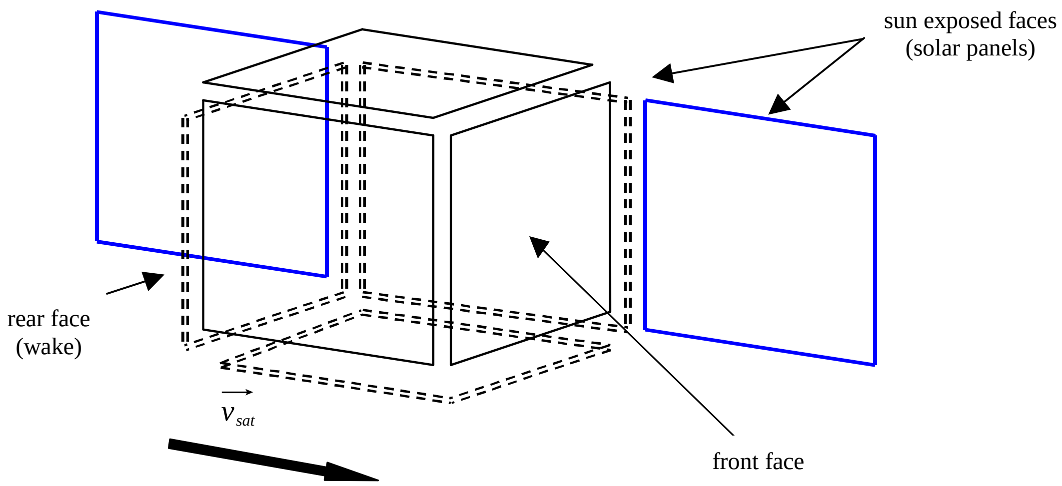

Figure 3: schematic view of the Microscope geometry. Due to the satellite spin, the four faces orthogonal to the solar panels are alternatively "front face".

Secondary emission has been neglected due to the low electron temperature for the orbit considered. It can have a sensitive effect on high latitudes where high energetic electrons of several keV exist but then the electron density is so low that the resulting secondary electron current will still be negligible in the current budget. Plasma characteristics have been obtained along the orbit by using the IRI Software. IRI (International Reference Ionosphere) is a code developed by the NASA. It gives the composition, the densities and the temperatures along the orbit depending on the solar activity. For a mission around 2007, data from 1996, which correspond to a minimum of solar activity due to the 11 years solar cycle, have been used for the computation.

Two situations coming from spacecraft attitude and orbit control studies have been considered. A nominal operation of the thrusters with a mean thrust of 10.5 µN by thruster for a total ionic current of 1.155 mA and the worst case with an EPSA breakdown leading to a maximum current at a given time of 5,86 mA. The S/C potential evolution along the latitude is shown for different times and for both cases in figure 5 and figure 6.

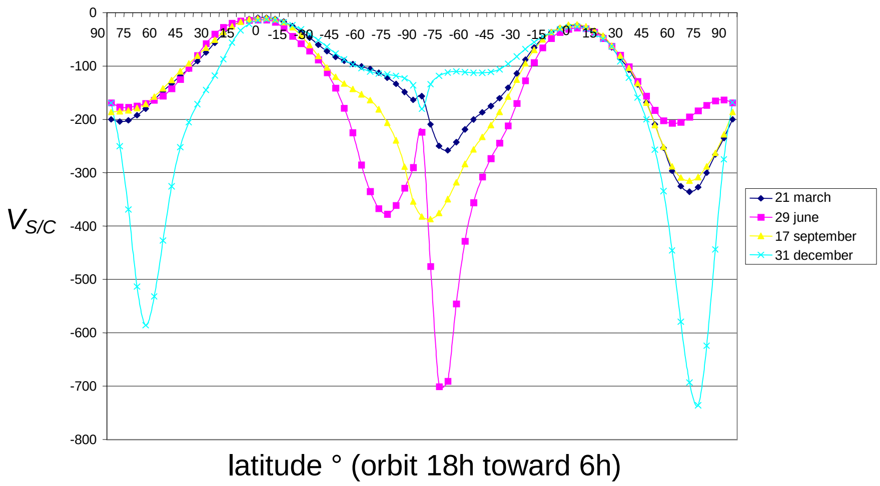

Figure 4: evolution of the S/C potential without neutralization for a nominal case (I_{FEEP} = 1.155 mA) along the latitude and for four different seasons.

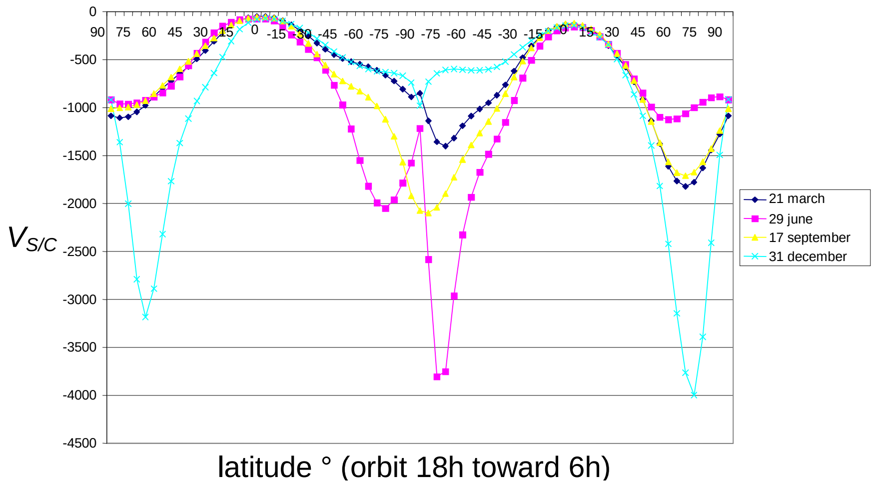

Figure 5: evolution of the S/C potential without neutralization for a worst case (I_{FEEP} = 5.86 mA) along the latitude and for four different seasons.

Results for very high latitudes have to be taken with care due to the difficulty for IRI to model the true conditions at high latitudes. Nevertheless, it appears from these preliminary results that it is possible for the S/C potential to reach several hundred negatives volts in a nominal situation and several kilovolts negatives for a worst case. Such values are very hazardous for the spacecraft and confirm the necessity for a global neutralization.

5. Erosion

Neutrals and ions impinging the spacecraft may have enough energy to extract material from exposed equipment and hence alter the functional properties of subsystems especially sensitive to erosion. The sputtering yield is specific to each material and depends on the incident particle energy and pitch angle. For the Microscope spacecraft, sputtering has two potential origins:

- the neutrals and the ions from the space environment
- the high velocity ions emitted by the thrusters

On the Microscope orbit, the energy of ions from the ionospheric plasma is very low (0.1-0.3 eV) and most of the S/C surface is not damaged by these low energy ions. The most sensitive part of the spacecraft is the front area where the particle flux due to the spacecraft velocity (7500 $m.s^{-1}$) creates sputtering. At the Microscope altitude, atomic oxygen is predominant and the resulting flux is about 10^{16} $particles.m^{-2}.s^{-1}$ [5]. With the S/C spin, the erosion is moderated over four surfaces. N.B: in case of neutralizer malfunction (see chapter 4), all faces will have an enhanced sputtering resulting from the global collection of ions necessary to counterbalance the FEEP ion current

Ions emitted by thrusters have a maximum energy of 12 keV at the acceleration electrode (+5 kV applied on the emitter and -7 kV on the accelerator plate). In the vicinity of the thruster, the electric field between the accelerator electrode and the space environment (reference potential = 0) decelerates ions. Hence, the maximum energy to take into account for S/C sputtering by direct ions is 5 keV. Such a value could lead to a high sputtering of the material but the erosion by cesium ions is difficult to assess accurately due to the lack of data at this energy level. In fact, the most sensitive piece to the sputtering consecutive to FEEP use is the accelerator electrode. It has been demonstrated elsewhere that the accelerator plate is less sensitive to the primary ions than to the charge exchange ions created in the vicinity of the thruster [5],[7]. This phenomon is taken into account in the thruster design and is not the purpose of a plume effect study. A thruster operating at its maximum capability ejects less than 10^{16} particles.s^{-1} and 95% of the ions are in the exhaust cone that doesn't intersect the spacecraft.

In conclusion, the major source of sputtering seems to result from the atomic oxygen flux. If in the next phases of the project a detailed study is required for erosion coming from the FEEP, a model representative of the plume outside the exhaust jet and sputtering yields for cesium, at different energies and angles for different materials, will be necessary.

6. Contamination

Contamination is the deposit of undesirable material on a S/C area. It can be due to spacecraft outgassing, space environment or active sources like thrusters. As regards plume effects, the contaminants come from two sources:

- direct contamination by the flow from the thruster. In that case the contaminants are the propellant and/or impurities from the thruster system,
- indirect contamination by the redeposit of sputtered material created by the plume impingement on a surface.

On a spacecraft, many systems are particularly sensitive to the contamination: solar arrays (power production), radiators (thermal control) and optical sensors (remote sensing and attitude determination & control). Contaminants can highly change their thermo-optics properties and hence generate dysfunctions like power loss, signal loss or temperature increase. For Microscope, the requirement concerning the direct contamination is that the mass of propellant coming back to the satellite shall be lower than 1% of the total mass extracted from the thruster. Supposing a propellant mass of 10 grams per thruster, the total propellant contaminant mass deposited on the spacecraft at the end of life shall be lower than 1.2 grams for a redundant configuration. More than to the ions ejected with high velocities from the thruster, this requirement is relevant to the neutral atoms evaporated from the liquid-metal surface inside the emitter slit as well as neutral microdroplets and neutral atoms extracted by field emission ([6],[7]). This contaminant emission is difficult to assess accurately. Hence, it is even more difficult to assess the real contamination on the spacecraft. At last, the final configuration is not fixed at this phase of the project. Therefore, the contaminant mass taken into account for this preliminary thought is the maximum contaminant propellant mass allowed i.e. 1.2 grams (on the basis of 10 grams propellant per thruster). There is no specific requirement for the impurities. For FEEP technology used onboard Microscope, these impurities are mainly due to the sputtering of the accelerator electrode by the charge exchange ions. These low energy ions are created by collisions between high velocity ions from the thruster and surrounding neutrals. They are sensitive to the high electric field in the vicinity of the accelerator plate and are accelerated toward it hence creating sputtering and contamination of the plate. A previous numerical study [6] has shown that the charge exchange ion current is in the order of about 0.01% of the current emission and is concentrated on the accelerator electrode. The S/C contamination by thruster impurities from the thruster is therefore negligible compared to the propellant contamination maximum level.

Indirect contamination has to be taken into account at system level with an accuracy model of the interactions between the plume and the spacecraft. A good knowledge of the plume, the S/C geometry and the interaction cesium/S/C materials is required. The first item has to be given by the FEEP supplier and the second item is given by the S/C layout. The last point is the more restricting because of the lack of data on sputtering by cesium (see previous chapter). Nevertheless, the S/C layout has been drawn in order to have no S/C area in the FEEP exhaust cones. This allows reducing the direct and indirect contamination.

To synthesize, a good assessment of the quantity of contaminants was not realistic at this level of the project. Nevertheless, it is necessary to have a rough idea for the S/C layout and an anticipation of potential hazard early in the project. On the basis of the previous arguments, it seems that the major contaminant will be pure cesium coming out directly from the thruster and that the maximum contaminant mass at the end of life of the spacecraft will be about 1.2 g, which is very low. However, there is no certitude concerning the respect of this 1% requirement with in-flight conditions.

Beside the quantity issue, a better understanding of the phenomenology of the contamination by cesium is essential. Cesium is a liquid metal and is not commonly used on spacecraft. Specifics issues have to be clarified: microdroplets behavior outside the thruster, risks linked to the metal specificity (shortcuts on sensitive parts), alteration of thermo-optic properties. Even a very thin layer of material can alter the solar array panel efficiency or perturb an optic instrument. These effects are difficult to assess theoretically. It can be interesting to obtain experimental data during thruster operational functioning in test bench e.g. solar array sample inside the vacuum chamber. One major difficulty is the handling of cesium. Cesium reacts explosively with water to form cesium hydroxide, which is the strongest base known. Cesium reacts in air with molecular oxygen that makes experiments complex (alteration of the sample properties at the vacuum chamber opening). Survivability of the thrusters to atomic oxygen influence in LEO has been reported in a dedicated paper [5]. It must be emphasized that the high vaporization pressure of Cesium will prevent durable contamination on hot parts of the spacecraft (e.g. solar array). More generally, the impact of metallic particles around or over the spacecraft has to be analyzed.

7. Torques and EMC effects

Ions coming out from an electric thruster and hitting the spacecraft contribute to create a torque. This torque could be harmful for high thrust electric thrusters where sometimes the plume creates a force on the solar panels. For Microscope, the layout is very compact and, as written before, there is no intersection of the exhaust plume with the spacecraft. Therefore, the torques could be neglected.

High-density plasma can interfere with signal emission from the spacecraft. This problem could arise for large electric propulsion system onboard of telecommunication satellites. For Microscope Mission, the thruster plasma densities are lower than the densities from plasma of the space environment only few centimeters away from the thruster. The EMC aspect due to the FEEP thruster is not a relevant issue concerning the plume effects (EMC inside the spacecraft due to the use of high voltages is another issue).

8. Conclusions

The aim of this study was to make a preliminary estimation of the FEEP plumes effects over the Microscope mission. Such a preliminary study in the earliest phase of a project allows configuring the spacecraft in order to reduce these effects. But for the FEEP propulsion, such work is not easy due to the lack of inflight data and dedicated experiments on earth. Moreover the specificity of FEEP system (high-energy ions, liquid metal propellant) makes harder the comparison with other electric propulsion system. Typical electric plume effects have been considered: electrical effects, sputtering, contamination, torques and EMC effects. The preliminary results have shown that the major issues are the electrical effects and the contamination. A major plume effect of FEEP system can result from a dysfunction of the neutralizer system leading to high negative potential on the spacecraft that can create arcing effects, thrust loss and accelerated sputtering. This study shows that the space environment doesn't allow a complete neutralization of the FEEP ion beam and that the neutralizer is necessary over the mission life. Contamination is very difficult to assess and must be studied with dedicated experiments. A more complete study will require specific tools (plume and interaction models) and cesium/material interaction data.

9. References

1. Marcuccio S., Nicolini D., Jolivet L., **"FEEP System for the Microscope Mission"**, 28th International Electric Propulsion Conference, Toulouse, France, 2003.
2. **"Space environment: prevention of risks related to spacecraft charging"**, Space Technology Course CNES/ONERA, Cépaduès Editions, 2002.

3. Tajmar M., Wang J., **“Three-Dimensional Numerical Simulation of Field-Emission Electric Propulsion Neutralization**”, Journal of Propulsion and Power, Vol. 16, No. 3, May-June 2000.
4. International Reference Ionosphere NASA NSSDC (National Space Science Data Centre) http://nssdc.gsfc.nasa.gov/space/model/models/iri.html
5. Ceccanti F., Marcuccio S., Andrenucci M., **“FEEP thruster Survivability in the LEO Atomic Oxygen Environment”**, 27th International Electric Propulsion Conference, Pasadena, US, 2001.
6. Tajmar M., Wang J., **“Three-Dimensional Numerical Simulation of Field-Emission Electric Propulsion Backflow Contamination”**, Journal of Spacecraft and Rockets, Vol. 38, No. 1, January-February 2001.
7. Tajmar M., Rüdenauer F., Fehringer M., **“Backflow Contamination of Indium Liquid Metal Ion emitters (LMIE) Numerical Simulations**”, 26th International Electric Propulsion Conference, Kitakyushu, Japan, 1999.

10. Acknowledgements

The authors are very grateful to O. Vandermarcq, Microscope Project Manager, for his help in preparing this paper and to Dr. J.F. Roussel (ONERA) who help us a lot through the great conversations we had.