SIMULATIONS OF PLASMA THRUSTER AND LARGE ANTENNA EFFECTS ON THE ELECTROSTATIC BEHAVIOR OF SPACECRAFT IN GEO

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ABSTRACT

This paper presents simulations of plume and spacecraft interactions in the frame of electrical orbit raising to geosynchronous orbit, including spacecraft charging, erosion and contamination. Three-dimensional plume modelling is used to estimate the thickness of material erosion on solar arrays. Two configurations of the electric propulsion are assumed: either four low-power or one high power thrusters instead. The neutralization of the plume is ensured by a simplified electron model following a Maxwell-Boltzmann assumption. A basic model is used to assess the impact of solar array bus voltage on plume current collection and spacecraft potential. All results suggest that covering materials will be strongly impacted by charged and neutral particles causing their erosion and redeposition. Finally, spacecraft charging under severe geosynchronous environments is studied in a configuration of large deployed mesh antenna reflectors. Their influence is shown to be dependent on the spacecraft attitude as so forth on the magnetic local time.

1. INTRODUCTION

The recent interest for the electrical orbit raising (EOR) of geostationary (GEO) telecom spacecraft offers a solution to embark more payload at reduced costs. However, some concerns arise from the necessity to spend a significantly longer time within the radiation belts and from the prolonged exposition to thruster plume. Such a spacecraft, once arrived at GEO, may not be as beginning of life (BOL) in some extent. The goal of this paper is to study the impact of Hall effect thrusters on solar array erosion and contamination during EOR and to assess the charging risk when combined with large wire antenna reflectors deployed at GEO.

Several scenarios of EOR to GEO can be used by spacecraft operators. They may impact spacecraft surface material contamination by propulsion products, especially at solar arrays level. In a super synchronous transfer orbit (SSTO), the spacecraft is injected along an elliptical trajectory with the perigee in LEO at a few hundreds of kilometres of altitude and an apogee above 60 000 km. The electric propulsion is then used to increase both the apogee and perigee. This is achieved with a thrust force in the same direction as the spacecraft velocity vector. During that period of about 1 month and a half (typically the case of Intelsat 115 West B and ABS 3A), the solar panels are rotating continuously at a rate of 2π per orbit period (~20 hours) to face the Sun. They are thus regularly oriented towards the thrusters direction. In a second phase, the apogee is decreased while the perigee is augmented. This is achieved in 5 to 7 months with a thrust assumed to be constantly directed along the small ellipse axis. Assuming also the thrusters have a fixed position relative to spacecraft body, we expect thus the solar arrays to rotate at a rate of 2π per year.

In this paper, we focus on the comparison of using either four low-power thrusters or either larger one high-power system in terms of ions backflow to solar panels. In a second section, we present an estimation of the impact of large deployable mesh antenna reflectors after an EOR phase.

2. EOR SPACECRAFT CHARGING AND EROSION

The modelled spacecraft is totally conductive, covered by black kapton (body and rear side of solar panels), by Indium Tin Oxide (ITO) (solar arrays and optical solar reflectors) and by conductive black paint (body antenna reflector). The body is 4.5 m × 2.5 m × 2.5 m. Four solar panels of 3.6 m × 5.0 m each rotate along the Y axis.

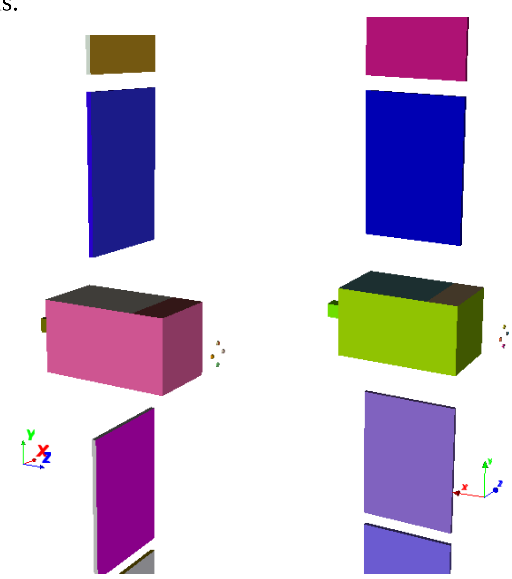

Figure 1. Spacecraft geometry including electric thruster and solar array rotation during the EOR phase

Two solar arrays positions are represented in Fig. 1, with solar arrays oriented towards the thrusters and perpendicular to this direction. During the EOR phase, we expect the large antenna reflectors not to be deployed in order not to degrade them with the plume. The spacecraft is immersed in a quiet environment represented by a plasma density of 10^9 m^{-3} and temperature of 3.0 eV. Two options have been assumed for Hall effect electric thrusters: A) four lower-power thrusters (SPT100 type) and; B) one high-power thruster (SPT140 type). They are located at 1.15 m from the spacecraft body (booms not modelled) and emit a total Xenon ion current of 20 A with an ionizing efficiency of 95 %. Their drift energy is 160 eV. The unionized neutrals perform charge exchange (CEX) reaction with fast ions and generate low energy ions (temperature of a fraction of eV). The dynamics and erosion by Xenon ions (fast and CEX) is computed self-consistently with the potential using the Spacecraft Plasma Interaction Software (SPIS) [1]. Plume neutralization by cathode electrons is of prime importance for assessing plasma potential and backflow of CEX ions to spacecraft surfaces. Full particle-in-cell (PIC) 2D simulations of fast Xenon ion plasma showed that the plume expansion is composed of three regions [2]: thruster exit where electron cooling may be described by a polytropic law; dense plume with isothermal expansion (yet anisotropic) due to the potential well of the plume and; expansion transition to ambient plasma possibly defined by another but yet undefined polytropic law. 2D hybrid simulations showed also that a non-negligible amount of low energy ions (about 10%) is generated in the plume by electron-neutral ionizing collisions [3]. These ions dynamics needs to be solved with a PIC approach to accurately model the backflow [3]. The hybrid model used in this paper is composed of a PIC model for ions and a Maxwell-Boltzmann distribution for electrons (ambient and from the neutralizer) with a temperature of 3 eV and a density of $10^9 m^{-}3$ at the reference potential of 0 Volts. It addresses partially the physics since we assume an isothermal plume expansion. The transition with the ambient environment is achieved smoothly by the high density of electrons at zero volts, hence avoiding unrealistic large negative potentials given by the Maxwell-Boltzmann distribution when the plasma density decreases. This model is believed to provide a good order of magnitude of the electric field around the thruster and spacecraft, which is an important parameter for CEX ion trajectories computations. The potential of spacecraft surfaces influence the energy of impacting CEX ions and their ability to erode materials. A spacecraft immersed in a dense and low energy plasma is known to be active as it collects a large amount of current on solar arrays interconnects. We assume two situations: negative interconnects with almost no impact on current collection (but a lot on erosion), and positive interconnects which can possibly collect a large amount of current and change spacecraft voltage. In this latter case, we model the solar cells interconnects by a 0.3 m × 0.3 m patch with a bias voltage of +50 V. Finally, the SPIS model of erosion includes emission of neutral products following a PIC approach for their dynamics in the volume.

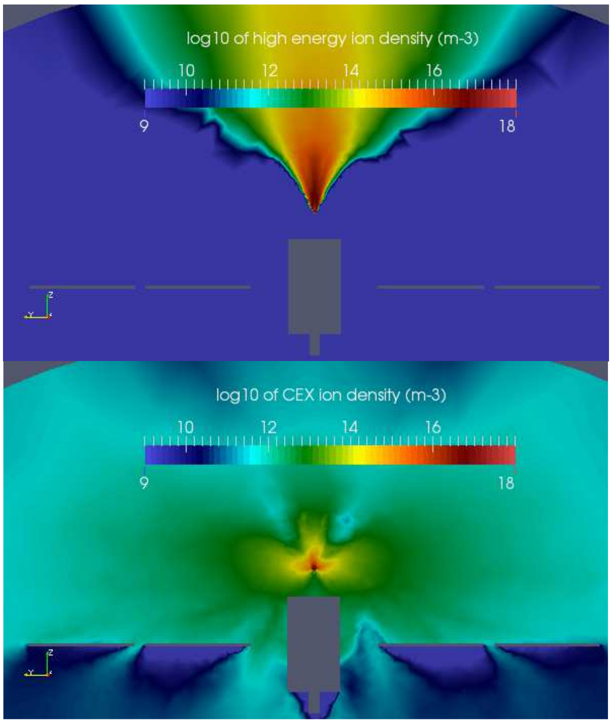

Figure 2. High energy (top) and low energy (bottom) xenon ions density for one high power thruster (option B) and negative interconnects

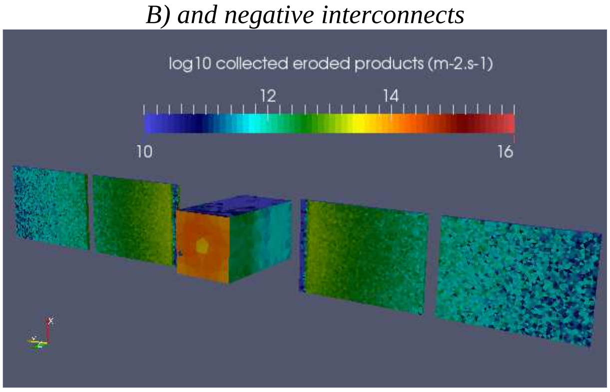

Figure 3. Contamination fluxes for one high power thruster (option B) and negative interconnects

Fig. 2 presents the plume expansion in the case of one high power thruster, negative solar cells interconnects, and solar arrays oriented in the same direction as the thruster. Fast xenon ions have a maximal density of 10^{18} m^{-3} and expand with an angle of ±30° with respect to the thruster centreline. CEX ions are generated close to the thruster exit, where both ion and neutral xenon densities are maximal. The potential inside the plume is +50 V with respect to the ambient plasma, as given by the Poisson equation solution (quasi-neutrality in this case) and Maxwell-Boltzmann electron distribution. This high potential ejects CEX ions out of the plume with two preferred orientations: toward the centreline and at ±70°. The thruster potential of -0.5 V significantly deflects CEX ions. A large amount of them hit its surface (producing the largest source of eroded products) and a non-negligible fraction is accelerated as a backflow towards the spacecraft body and solar panels.

The erosion of the thruster surface is a source of contamination for other surfaces, as shown in Fig. 3. The deposit rate on solar arrays is about 1 nm within 120 days, 70 days and 7 days, for option A, option B and option A with positive interconnect, respectively. In the latter case, the spacecraft surfaces float at -25 Volts and increase the CEX impact energy from 50 eV up to 75 eV with respect to the negative interconnects. As erosion strongly depends on energy in the 10 to 100 eV range, the solar array bus voltage is of prime importance. It is also observed that using a single high-power system generates twice CEX as the four low-power thrusters because the density of neutrals and xenon ions is larger at thruster exit (same current on a smaller area). The solar array erosion rate by CEX is about 1 nm within 7, 3 and 2 days respectively for the same conditions as above. It signifies that it is efficient at removing contaminant, which is good, but also technical covering layers, which is of course to be taken into account at spacecraft design level. Globally, under the assumptions made in this paper, one can conclude that ITO could be totally removed from solar array surfaces after EOR.

3. EFFECT OF DEPOYABLE WIRE ANTENNA REFLECTORS AT GEO

In this section, we assume that only the dielectric cover glass remains on solar arrays after EOR and that 25 % of the ITO area is removed from the optical solar reflectors. Two wire antenna reflectors 5 m in diameter made of gold are deployed, see Fig. 4. The wire diameter is 25 µm and the mesh is composed of 30 openings per inch, hence with a transparency of 94 %. The model used in SPIS for such reflectors consists of thin surfaces intercepting 6% of the particles (charged, neutrals, photons). We simulate a nominal situation with the thrusters off and immerse the spacecraft in the ECSS GEO worst case environment for surface charging risks [4].

Using wire antenna reflector has a clear impact on spacecraft charging because they offer much less metallic surfaces to the sun irradiation than plain reflectors. Photoemission is strongly reduced with respect to plain reflectors and the spacecraft potential can decrease down to hundreds to thousands of volts negative within a few tens of seconds, see Fig. 5.

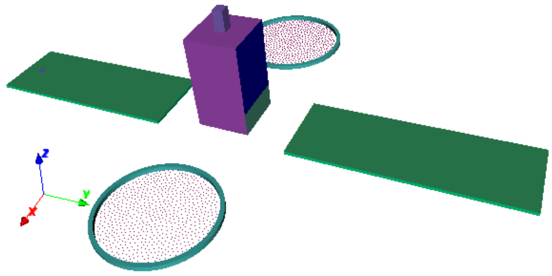

Figure 4 - Spacecraft geometry including deployed wire antenna reflectors at GEO.

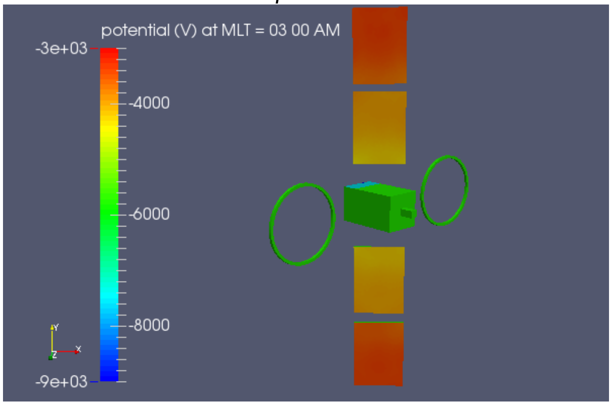

Figure 5 – Spacecraft charging at GEO

4. CONCLUSION

The SPIS numerical models permit to simulate the environment and the plasma dynamics at the full scale of a 3D spacecraft. This paper simulated the use of a single high-power Hall effect thrusters and four low-power thrusters in terms of fast and charge exchange ion density, energy and contribution to solar panel erosion. Under the assumptions made in this paper, thin layers of covering conductors will be skipped out from solar panel surfaces, leading to well-known risky situations under GEO worst case environments. It was shown that the solar array bus voltage needs to be taken into account for accurate estimation of spacecraft voltage and erosion issues. Future efforts will focus on the improvement of interconnects, the description of the electrons from the neutralizer (both in the field close to the thruster exit and in the far field), plume chemistry... These necessary upgrades will permit to assess many risks during EOR (current leakage, ESDs, erosion, contamination…) and at GEO.

5. REFERENCES

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4. ECSS Space engineering - Spacecraft charging," ECSS-E-ST-20-06C, ESA-ECSS, 2008.