

1 Report

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Abstract

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1.1 Introduction

The spacecraft in space will always be affected by its environment resulting various impacts depending on the orbit (location), types of material as well as the environment condition that changes over time (Hastings and Garrett, 1996). The most common phenomena on the spacecraft is what so called charging. The level of charging depends on the energy of particles interacting with the spacecraft. At the lower energy, the form of interaction of charged particles with the spacecraft only affect the surface part called surface charging. However, the higher energy the worse affects might be occurred on the spacecraft and in this case the charged particles can penetrate deep inside the spacecraft component resulting in so called internal charging (Fennell et al., 2001).

Charging on the spacecraft can be simulated numerically. Numerous codes have been developed well to explain the behaviour of particles around the spacecraft as well as its interaction. Nevertheless, it still remains many questions since the numerical simulation is only an approximation of real condition. However, the numerical approach has given a good solution to various applications such as the spaceflight mission. The reliability of spacecraft has been proved well before launch into space. One of these effort is simulating the environment where the satellite will be placed during its mission. Many parameters has been taking into account with respect to its effect on the spacecraft. The results of the simulation can be significant point for decision maker for the spaceflight mission. This is one of reasons why this simulation is becoming primary interest for this study.

In this study, we attempt to simulate a spacecraft named Norsat-1 which will be placed in Low Earth Orbit (LEO) environment around 600 km altitude and polar inclination around 98.8° . Initially this satellite will be launched in early 2016 (<https://directory.eoportal.org/web/eoportal/satellite-missions/n/norsat-1>). This satellite has been equipped with two probes. It is interesting since this satellite will pass over

auroral region more frequently. In this region the satellite will be exposed not only to rapid variation of thermal component of the ionosphere (Hastings, 1995), but also to high energy of particles from the solar wind.

Since we use the EMSES (Electro Magnetic Spacecraft Environment Simulator) code (Miyake et al., 2013) in the simulation, it is important to point out that only the effects of background plasma as well as the photo electrons from the sunlight are taking into account. The simulation has been done into several cases and each case has been grouped into two, i.e. plasma flows with and without the photoemission effects on the spacecraft as well as the probes. All results will be presented in detail in the specific section in this report.

1.2 Theory

1.3 Numerical Methods

1.3.1 Numerical methods

To solve the problem numerically we use the EMSES code. EMSES uses the standard PIC method for plasma simulations. In the code we are able to define a spacecraft body, and the code then calculates the potential on that body using the capacitance matrix method. Although EMSES has the capability to do a full electromagnetic calculation, we have opted to use the poisson's equation solver for electrostatic problems. In the EMSES system we can define sunlit surfaces based upon an angle, and a current density. Sunlit surfaces will then emit electrons based upon a energy distribution. For a complete description of EMSES' capabilities see (Nakashima et al., 2009) Parameters are chosen to simulate the sun at the earth, but with an enhanced flux to emphasize the effect in question.

1.3.2 Theoretical calculations

$$I_i = \begin{cases} A|q|n_\infty\sqrt{\frac{8kT_i}{\pi m_i}}(without\ flow) \\ \frac{1}{6}A|q|n_\infty V + \frac{5}{6}A|q|n_\infty\sqrt{\frac{8kT_i}{\pi m_i}}(with\ flow) \end{cases}$$

$$I_e = -A|q|n_\infty\sqrt{\frac{8kT_e}{\pi m_e}}exp\left(\frac{|q|\Phi_d}{kT_e}\right)$$

$$I_{ph} = \frac{1}{6}AJ_s$$
(1.1)

Submitting these equations to $I_i + I_e + I_{ph} = 0$ and solving for Φ_d , we can get

$$\Phi_d = \frac{kT_e}{|q|} \ln\left(\sqrt{\frac{T_i}{T_e}}\sqrt{\frac{m_e}{m_i}} + \frac{J_s}{6n_\infty|q|}\sqrt{\frac{\pi m_e}{8kT_e}}\right) + \Phi_0$$

$$\Phi_d = \frac{kT_e}{|q|} \ln\left(\frac{5}{6}\sqrt{\frac{T_i}{T_e}}\sqrt{\frac{m_e}{m_i}} + \frac{V}{6}\sqrt{\frac{\pi m_e}{8kT_e}} + \frac{J_s}{6n_\infty|q|}\sqrt{\frac{\pi m_e}{8kT_e}}\right)$$

$$\Phi_0 = kT_{ph}, \quad V = speed\ of\ flow$$
(1.2)

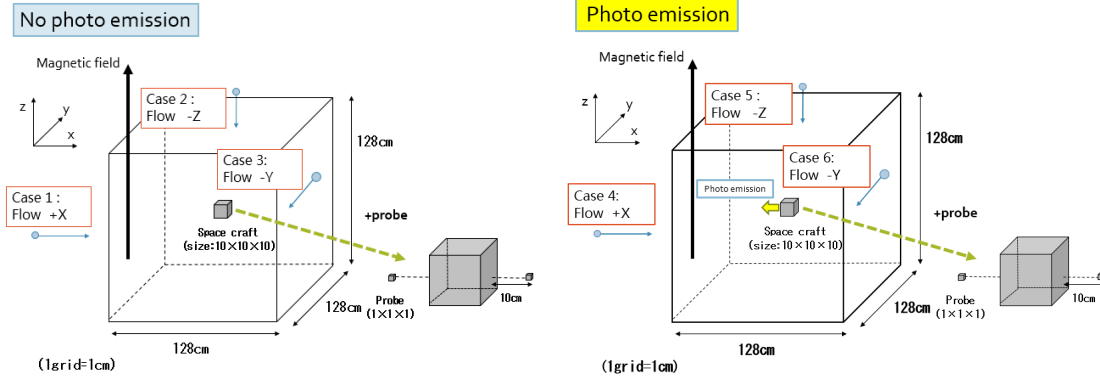


Figure 1.1: Research about spacecraft and surroundings without P-E and without. Left figure is no P-E simulation cases. Right figure is P-E simulation cases.

The theory is based on thin-sheath approximation, so if SC size is much larger than Debye length of the photoelectron, we can use it.

Submitting these equations to $I_i + I_e + I_ph$ and solving for ϕ_d ,

$$\begin{aligned}\Phi_d &= \frac{kT_e}{|q|} \ln \left(\sqrt{\frac{T_i}{T_e}} \sqrt{\frac{m_e}{m_i}} + \frac{J_s}{6n_\infty |q|} \sqrt{\frac{\pi m_e}{8kT_e}} \right) + \Phi_0 \\ \Phi_d &= \frac{kT_e}{|q|} \ln \left(\frac{5}{6} \sqrt{\frac{T_i}{T_e}} \sqrt{\frac{m_e}{m_i}} + \frac{V}{6} \sqrt{\frac{\pi m_e}{8kT_e}} + \frac{J_s}{6n_\infty |q|} \sqrt{\frac{\pi m_e}{8kT_e}} \right) \\ \Phi_0 &= kT_{ph}, \quad V = \text{speed of flow}\end{aligned}\tag{1.3}$$

1.3.3 Test case setup

We wish to simulate the effects of Photon emitted electrons in different test cases, and have thus set up the following 6 cases:

Case	Plasme flow	Photon emission
1:	$41600 \vec{e}_x$ m/s	0
2:	$-41600 \vec{e}_z$ m/s	0
3:	$-41600 \vec{e}_y$ m/s	0
4:	$41600 \vec{e}_x$ m/s	$-10^{-3} A/m^3 \vec{e}_x$
5:	$-41600 \vec{e}_z$ m/s	$-10^{-3} A/m^3 \vec{e}_x$
6:	$-41600 \vec{e}_y$ m/s	$-10^{-3} A/m^3 \vec{e}_x$

So test case 1-4, 2-5, and 3-6 are the “same” cases except that we run the simulation with and without photon emission to compare the cases two and two. We define 3 geometric objects, the spacecraft itself and two probes. In all cases the B field is in the \vec{e}_z direction.

Stepsize	Timestep	Density	BMagnitude
1.0cm	5E – 10s	1.0E5kg/m ³	50E – 6 T

Table 1.1: Input parameters in EMSES

1.4 Results

1.4.1 Induced electric current

The plasma is flowing in in relation to the coordinate system in the simulations. Due to this an induced electrical field, ε , will appear. The induced electrical field will neutralize the Lorentz force. Combined with the electrostatic approximation we can obtain the ε

$$\vec{\varepsilon} = v_D \times \vec{B} \quad (1.4)$$

This will cause a potential gradient perpendicular to the plasma flow and the magnetic field, using the electrostatic approximation we obtain the magnitude of the gradient.

$$\int E dx = -\phi \quad (1.5)$$

$$\phi = - \int \vec{v}_d \times \vec{B} \approx - \int (41600\text{m/s} \cdot 50E - 6\text{T}) dx \quad (1.6)$$

$$|\nabla\phi| = 2.08\text{m}^{-1} \quad (1.7)$$

Figure ?? shows the measured potential at case 6.

1.4.2 Photoemmission paths

The electrons emitted from the spacecraft due to the photoelectric effect, have a kinetic energy corresponding to a Maxwellian distribution with a temperature of $T_{ph} = 3.8481 \cdot 10^4\text{K}$. Figure ?? illustrates the trajectories of the emitted electrons in simulation 6. As the probes are situated 10cm to the sides of the spacecraft on the x -axis, the probes may be hit by the photo-emitted electrons. In the following section, ??, we show the number of electrons hitting the probes.

1.4.3 Accumulated Photoelectrons on Langmuir Probes

In the simulations both electrons and photoelectrons are absorbed by the probes. Table ?? shows the current of both regular electrons, as well as photoelectrons, interacting with the probes. The photoelectrons are emitted on the left side of the spacecraft and we see a larger current of photoelectrons here. The current caused by the photoemission is 10 – 100 smaller than the current from the electrons in the plasma.

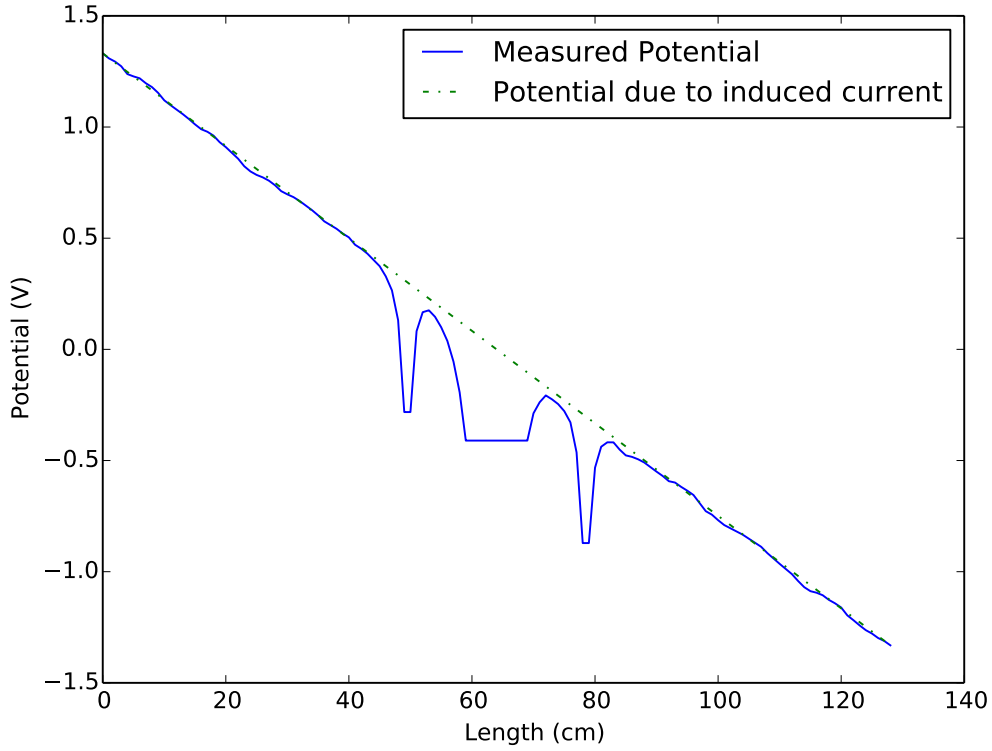


Figure 1.2: The blue line is the potential along direction x for simulation 6. In this case the potential gradient is along the x -axis. The dotted green line is the potential caused by the induced electrical field. This should be accounted for if we want to find the potential at the spacecraft and the probes.



Figure 1.3: The trajectories of the electrons emitted by the photoelectric effect in simulation 6. The possible paths of the photoemitted electrons coincide with the volume occupied by the langmuir probes. The photoemitted electrons are strongly affected by the magnetic field \vec{B} , and follows a gyrating path guided by \vec{B} . The photoemitted electrons are in all the studied cases emitted from the spacecraft in $-x$ direction, and the paths are similar. The langmuir probes are situated 10cm to each side of the spacecraft along the x -direction. (NOTE, should have axis labels, and domain length.)

	probe to +x	probe to -x
case 1	2.207E-07	2.851E-07
case 2	2.794E-07	2.807E-07
case 3	2.786E-07	2.772E-07
case 4	2.117E-07	2.609E-07
case 5	2.754E-07	2.616E-07
case 6	2.777E-07	2.512E-07

	probe to +x	probe to -x
case 4	2.3324E-09	3.49699E-08
case 5	1.9598E-09	3.51036E-08
case 6	1.3824E-09	3.66922E-08

Figure 1.4: The left table shows the electron current hitting the inserted probes at for the simulated cases 1 – 6. On the table to the right the photoelectrons hitting the probes are shown. The photoelectron current is smaller than the electron current from the plasma, varying from 10 – 100 times smaller.

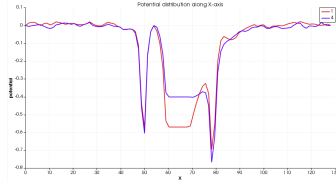


Figure 1.5: Potential of satellite and surroundings in the x-direction for case 1 and case 4.

1.4.4 Potential difference with P-E and no P-E

Case 1 vs case 4

Here we have the emitted electrons in the negative flow direction. As expected this leads to a drop in potential in the left probe which is facing the plasma flow. The potential drop over the probe is 3.8%. The right probe is now the wake where we have a drop in the ion density. This yields a large drop in potential compared to the left probe, but it also has a larger than the left probe when comparing case 1 and 4. This might be because the potential drop on the left side redirects more ions from the right side. The potential drop comparing the two cases is 10%. The potential rise over the satellite is 28%.

Case 2 vs case 5

With the flow of emitted electrons in the negative x direction we would expect a rise in electron density around the left probe. This can be seen in figure ?? where we see a 5.4% drop in the potential of this probe compared to case 5 with no emitted electrons. On the right probe we have a small rise in the potential of 3.3%. With no emitted electrons on this side of the satellite the rise in potential can be explained by looking at the increase in ion density as seen in figure ?. In the satellite we have a potential rise of 28%. So the change in potential in the probes are small compared to the change in the satellite.

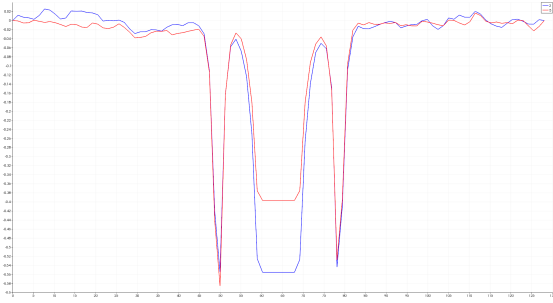


Figure 1.6: Figure show potential of satellite and surroundings in the x-direction for case 2 and 5.

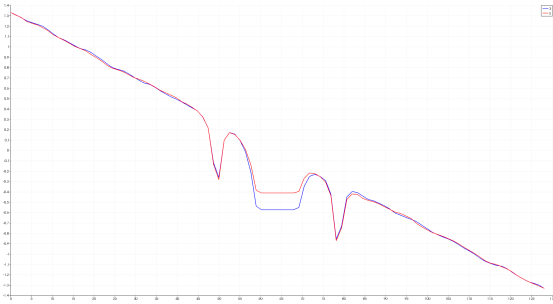


Figure 1.7: Figure show potential of satellite and surroundings in the x-direction for case 3 and 6.

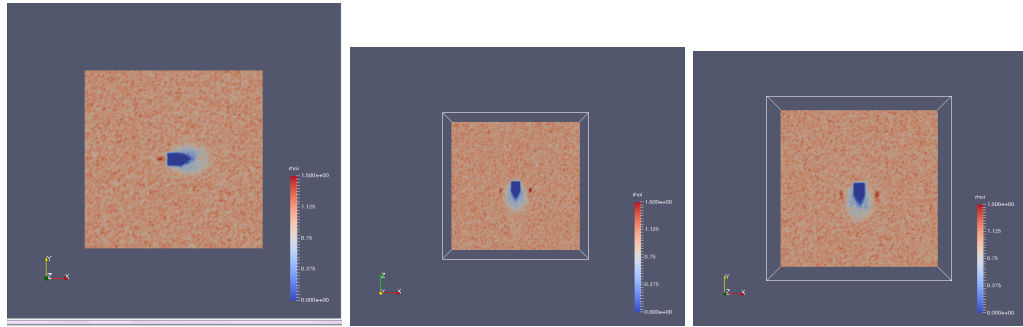


Figure 1.8: Ion density of spacecraft and surroundings without P-E Figure on the left displays case 1. Middle figure displays case 2. Rightmost figure displays case 3.

Case 3 vs case 6

A rather large drop in potential of 12% on both probes. Potential rise of 26% over the satellite.

1.4.5 Wake plots

The spacecraft is moving relative to the plasma flow, this causes a wake to be formed in the vicinity of the craft. Figures ?? illustrates the flow in case 1, 2 and 3.

1.5 Discussion

1.6 Conclusions

In the simulated cases with exaggerated

- Proposal for further studies (Probably see if photoemmission is relevant in tenuous plasma (MEO CASE, magnetospheric tail lobes))

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