3D Computer Simulation of Spacecraft Charging Effects

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1. Introduction

Charging of spacecrafts in near–earth orbits is determined by space environment in the orbit. Dominating physical processes are different in principle for various altitudes. So, the spacecraft charging model which is based on computation of external particle fluxes on the spacecraft surface should include *(a)* space environment data, *(b)* adequate description of physical processes of surface/environment interaction including secondary processes, *(c)* algorithm of the spacecraft surface model construction, and *(d)* numerical methods for solution of the problem.

In this work, we consider items *(b)* and *(c)* mainly as the most actual to be solved for simulation of spacecraft charging in various orbits. Our model of spacecraft charging in geosynchronous orbit (GEO) [1–3] was implemented in the COULOMB program package successfully used for modeling of GEO satellites [2, 3].

The principle difference between spacecraft charging in GEO and one in low-earth orbits (LEO) is determined by the fact that plasma of magnetosphere is hot and rare, and one of ionosphere has low temperature and high density. In this case, new method for computation of primary particle fluxes on the spacecraft surface was developed and implemented in the program package NPI_LEO.

To do the simulation of spacecraft charging in LEO, we need the procedure of mathematical description of real spacecraft surface. The procedure was developed earlier [3] and employed in the COULOMB package. In the NPI_LEO package, the method developed was generalized to provide a powerful tool for 3D modeling of spacecraft charging in LEO. Physical model of spacecraft charging was built after detailed analysis of principle physical peculiarities of the ionosphere plasma particle currents in the vicinity of spacecraft in LEO. Charging of real spacecrafts was investigated, and the results are presented in Sec. 4 below.

2. Physical model of charging in LEO

In real conditions, following processes determine the spacecraft charging in LEO:

- impact of ram ion current on the spacecraft surface, and originating of the wake region;
- impact of high energy auroral electrons (E_{aur}~10...30 keV) which is probable in polar region;
- the photoelectronic emission on the surface which depends on intensity of solar light and material of the surface;
- secondary electron emission which depends on the primary particle energy and material of a surface.

Fluxes of primary particles in LEO (i.e. ram ions and auroral electrons) are unidirectional ones, and the impact of the Solar light is unidirectional too. The spacecraft charging in this case dramatically depend on orientation of the spacecraft relatively to these directions. So, the aim of the physical model is to provide an adequate method for computation of particle currents on various spacecraft design elements.

As far as the currents of charged particles depend on the surface potential, the equilibrium can be reached at a value of the potential φ_s. The complete balance equation of currents in the case of spacecraft charging in LEO taking into account the impact of auroral electrons is the following:

$$j_i(\varphi_s) - j_e(\varphi_s) - j_{aur}(\varphi_s)(1-\delta) + j_{phot} = 0\,,$$

where j_i, j_e, j_{aur}, j_{phot} are ram ion, cold electron, auroral electron and photoemission current densities; δ is secondary electron emission coefficient depending, in general, on the impact auroral electron energy spectra.

For negatively charged surface (as it really is) electrons are repellent particles, and density of the current density on potential is Boltzmann type:

$$j_e(\varphi_s) = j_e^{(0)}\, \mathrm{e}^{\frac{e\varphi_s}{kT_e}}\,,$$

where $j_e^{(0)} = -e\sqrt{\frac{kT_e}{2\pi m_e}}N_e$,

e – elementary charge, k – Boltzmann constant, T_e and N_e – temperature and concentration of particles, m_e – electron mass.

The main problem for charging in LEO modelling is the description of ram ion current because the motion of positively charged particles near negatively charged surface has some peculiarities [4] and the general solution of the problem can be obtained using direct simulation of particle motion in the vicinity of spacecraft. However, the problem can be solved in the case when Debye length

$$D = \left(\frac{kT_e}{4\pi e^2 N_e}\right)^{\frac{1}{2}}$$

is short, $D \sim 1$ cm, and for objects having sizes $R \sim 1$ m $D << R$. This case,—the case of "large plasma probe "—was analyzed in [4] in details. It was shown that for high potentials the outer border of the charged layer near the surface is rather sharp, and introduction of the ram ion capture effective surface with appropriate radius R_c is valid:

$$R_c = R\,\mathrm{F}\left(\frac{e|\varphi_s|}{kT}\left(\frac{D}{R}\right)^{4/3}\right),$$

where F was tabulated in [4] for various argument values.

The physical meaning of the ram ion capture radius concept is that the charged object effectively attracts particles from distance $\sim R_c$ from the surface. The modelling of particles trajectories near the charged surfaces [5] show, that the captured particles have approximately straight trajectories when moving inside the charged layer to the spacecraft surface.

Influence of magnetic field on the particle motion in the vicinity of spacecraft is negligible in this case, and particle currents in NPI_LEO program package are computed without taking into account magnetic field effects.

Test computations based on the physical approximation above, and detailed comparison of the results obtained with ones presented in [6] have demonstrated good agreement of this new approach with previous ones.

Photoemission current j_{phot}, as primary estimations show, provides weak positive charging of the lightened elements of spacecraft when no auroral impact occurs. Moreover, if $j_{phot} > j_{aur}$, the auroral electrons may produce no effect in this case too. But for real spacecraft having complicated surface, the real charging picture should be investigated by detailed modeling. For LEO spacecrafts, computation of currents on every spacecraft design element and solution of the balance equation can be done in terms of the 3D model of the spacecraft surface only.

3. 3D model of spacecraft surface

To perform the calculations, the real spacecraft should be represented by 3D mathematical model describing the most important features of the spacecraft design and providing the information on the layout of the spacecraft design materials [3, 7]. The set of basic elements (primitives) is selected by the NPI_LEO package user for construction of the given spacecraft model. Fragments of one element made of different materials are considered as different primitives. For the geometrical model construction, the following primitives are used: plane, sphere, cylinder, cone, tore and their fragments.

The primitives are described by parametric equations in local coordinate system. Translation transformations and rotation by Eulerian angles are used to transit into the global coordinate system of the model.

Note that the 3D model construction is a heuristic procedure so we included a program of quick plotting of the model on computer display into the NPI_LEO package. The user may select various projection of the model for visualization or use standard VRML browser. If test computations for the model constructed show that the model is to be modified (e.g. some elements should be done in more details), the user can make it easy.

In course of the model discretization, each element is divided into parametric triangles. The geometrical centre, the square, the normal vector is determined for each triangle. The triangle geometrical centers and normal vectors are saved in the model database to be used in the subsequent calculations. Surface properties, in particular, surface conductivity and information on electric contacts with metal ground (first reported in [8]) are saved in the database too.

4. Simulation results

For spacecraft orbit altitudes of order of 1000 km, Debye length D is high enough, and the plasma electrostatic shielding is weak. In this case, the electric field of the charged spacecraft occupies rather wide area around the spacecraft. When the modeling is done, and the electrostatic potential distribution on the spacecraft surface is computed, the graphic tools of our package make it possible to plot the equipotential lines of electric field in space around the spacecraft. The graphic visualization of the computation results shows typical features of the spacecraft charging in the given environment.

Distribution of electric field near the spacecraft charged by ram ions and auroral electrons in ionosphere plasma is shown on fig.1. The spacecraft surface here is assumed to be dielectric, the case corresponds to 'the most dangerous' case when the charging produces the highest gradients of potential. As we see on fig.1, when ram ion flux direction and electron flux direction are non-parallel, and electron current density is high, the hard charging of the spacecraft arises. The equipotential line picture is very sophisticated in this case because, as it was noted above, potential gradients are more expressed on dielectric surfaces irradiated by the high-energy electrons.

Actually, real satellites have dielectric, as far as metal (i.e. conducting and connected with metal ground) design elements. When some elements connected with metal ground are subjected to the direct impact of auroral electrons, the potential of the ground here is inevitably high. In this case, high values of potential may arise on the elements which are not irradiated by auroral electrons directly.

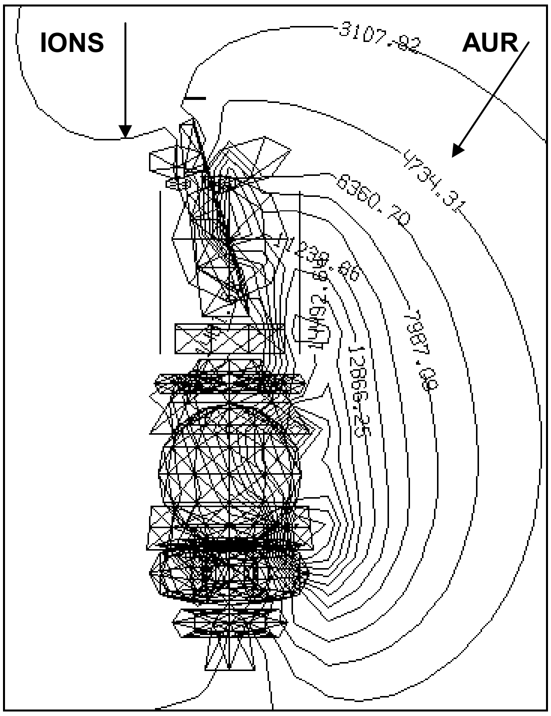

Fig.1. Electric field around the satellite charged by ram ions and auroral electrons

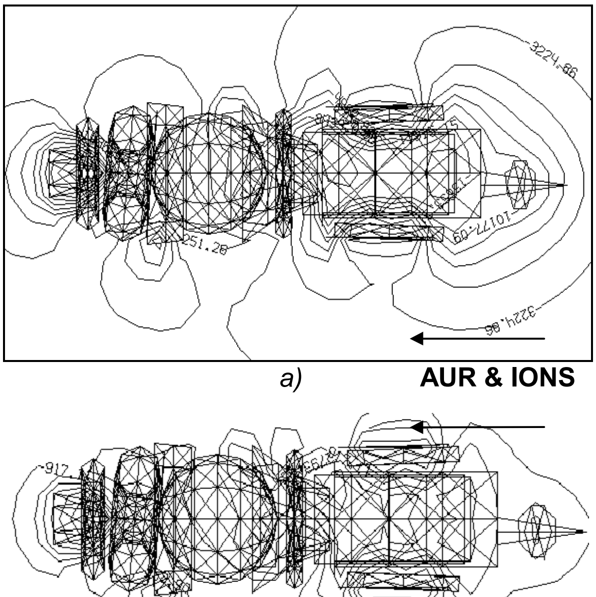

Fig.2. Electric field around the satellite charged by ram ions and auroral electrons in various altitudes: *a)* ~1500 km; *b)* ~500 km. Directions of the ram ion flux and auroral electron flux are the same in both cases

The example is shown on fig. 2a, 2b. We see here that equipotential line density is very high near the metal nozzle (left part of the figure), due to the fact that dielectric tore element near the nozzle is not irradiated by electrons directly either and has low potential. High gradients of potential arise in this case between the conducting cylinder and dielectric solar array panels. The reason for very high equipotential line density in the region is the same: high potential on the metal ground and low potential on the dielectric surfaces of the arrays which are not irradiated by high energy electrons directly.

Fig. 2b shows the distribution of electric field near the same satellite, as on fig. 2a, for high density ionosphere plasma. We see that configuration of lines closest to the spacecraft surface modifies minimally, but the region occupied by electric field is restricted due to the Debye plasma shielding.

The above method of simulation data visualization is inapplicable for LEO spacecrafts having orbit altitude ~300...500 km, i.e. where $D<10^{-2}$ m. In this case, equipotential lines are so close to the spacecraft surface that it is impossible to draw a distinctive picture with lines. The procedure of visualization of

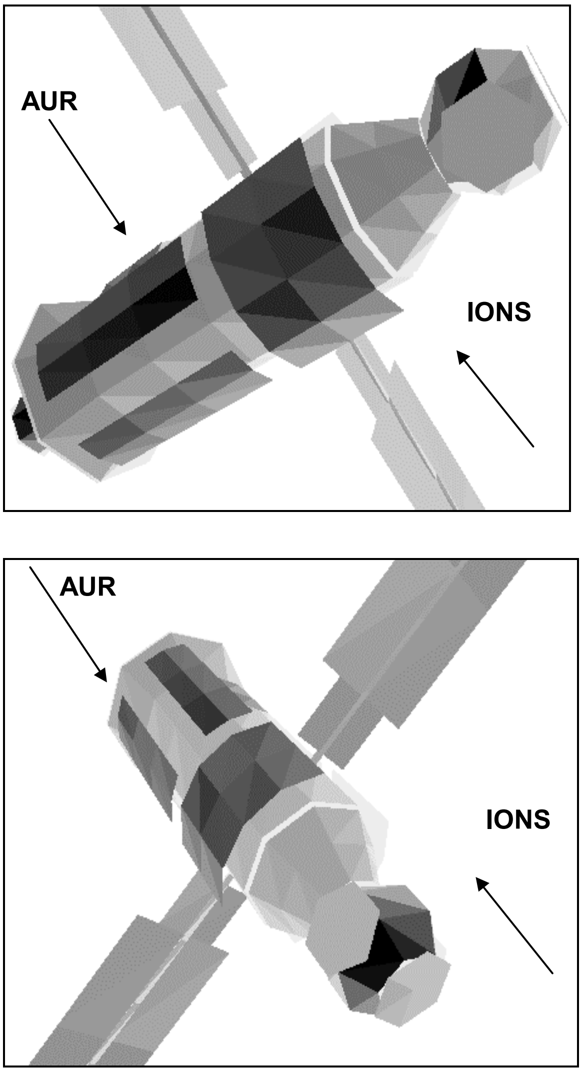

Fig. 3. VRML visualization of Zarya module surface charging for various direction of the ram ion flux and auroral electron flux

electric potential distribution on the spacecraft surface directly should be employed here. The best and the most universal way was found to develop VRML approach.

Spacecraft surface database is converted here into the standard VRML file format. Colors of the model triangles in colored VRML scenes, and grades of black in greyscale ones are set according to the charging simulation data. Black areas on fig. 3 correspond to the highest values of the surface potential.

The advantage of VRML visualization are following: the user can see the simulation results everywhere on the surface using easy VRML transformation; no special interface or graphic tools are necessary because standard (plug-in or external) VRML browser is used here; screenshots of VRML scenes can be printed in any desirable projection.

Fig. 3 are the screenshots constructed for model of the Russian module of International Space Station. In the model, cooler panels on the spacecraft are dielectric, so the distribution of electric potential along these panels exists that is clearly shown on the pictures.

The advantage of VRML visualization method of the charging simulation results is following: one can easily realize the peculiarities of potential distribution on the surface for the given environment conditions.

5. Conclusion

The 3D model of spacecraft charging simulation, main items of which are presented above, enables to compute the distribution of electric potential on the surface under the impact of ram ions in ionosphere plasma, Solar light and high-energy auroral electrons. Plasma density effects and, mainly, orientation effects are revealed and interpreted in terms of the model. Visualization of the computation data provides an opportunity to see the charging picture as a whole, as far as to analyze it in details. VRML technology developed was found to be very fruitful and gives no alternative in the case of short Debye length in LEO.

The NPI_LEO program package was developed to be used for similar simulation problems. Due to the fact, that physical processes governing charging in LEO had been analyzed deeply in terms of the 'large plasma probe theory', we need not to perform the straightforward particle trajectory modeling in our program. So, the computation resources for spacecraft charging analysis do not demand special computers or workstations. NPI_LEO program package is recommended as the engineering tool for real spacecraft charging analysis.

In future, the most actual problem to be solved is the elaboration of simulation result processing algorithms. At any case, visual analysis of computation data, especially done in the VRML variant, needs great time and attention and does not provide elimination of errors. Automatic intellectual algorithm aimed at solution of any definite problem (probable discharge points; modification of the spacecraft design; restoring the spacecraft environment conditions in terms of the information about surface potential, etc.) will provide the effective exploitation of 3D simulation results.

6. References

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