Abstract

The possibility of electric thruster (ET) usage for spacecraft propulsion system essentially depends upon solving the problem of effects of running ET on spacecraft units serviceability. The problem of alkali metal ET effects on a spacecraft is considered based on results of theoretical and experimental investigations, including space experiments. It is established that metallic propellants condensation is of a "threshold" character, determined by the deposited film composition, particle energy, surface material and temperature. There is an alkali metal film vaporization effect, that may lead to partial or full removal of the deposited film.

Introduction

Development of advanced spacecrafts is characterized by the increase of spacecraft lifetime in space. Thus, long-duration effects of ET exhaust plumes on spacecraft construction elements and materials can lead to restriction of spacecraft possibilities or its failure.

The application of alkali metal for ET propellant calls for solving the problem which is common for all of them, that is protection from deleterious effects on the spacecraft and, mainly, from alkali metal condensation on spacecraft surfaces. This problem is connected with selection of the optimum propellant for different missions where ET is used. Lithium is the best propellant for ET as applied for far space missions. Caesium is preferable in ET intended for transport interorbital transfer flights. It is advantageous to use Na and K (as well as Cs) in plasma sources intended for research tasks carried out both in space and laboratories.

Spacecraft Own Ambient Atmosphere

During spacecraft service its outer surface is contaminated from spacecraft environment, the contamination sources may be both external (astronomical) and inherent to this spacecraft. The latter are the spacecraft own atmosphere and products of propulsion system operating. The spacecraft own atmosphere with non-operating propulsion system contains polymeric materials outgassing products, gases leaking from pressurized compartments, and wastes of life-support system for piloted spacecrafts. The operating propulsion system is an additional contamination source and its relative contribution may be sufficient.

Typical data for the ambient atmosphere of a spacecraft with a chemical propellant propulsion system at 250...300 km altitude orbits (Space Shuttle) are presented in [1].

Under certain conditions the spacecraft own atmosphere species may condense on the spacecraft outer surface, thus offering a certain danger for spacecraft operation. So, for example, the "Apollo-13" program failed partially due to loss of window transparence. Contamination of thermal control coatings, infrared sensors and other equipment was registered on spacecrafts and orbital stations "Skylab", "Nimbus-IV", "ATS-6" and others. Thermal control coatings, optical sensors, windows, solar cells, antennae are especially vulnerable surfaces of spacecrafts.

In geostationary orbit the spacecraft surface contamination measurement was conducted on the satellite SCATHA P78-2 [2]. Measurements during several years gave the average contamination mass 5.3 \( \mu \text{g/cm}^2 \) per 1000-day flight. The same masses of deposition films were observed on Soviet satellites, too.

The spacecraft propulsion system operation with ET essentially changes the spacecraft environment, composition and formation rate of surface contamination film. The first attempt to determine the effect of ET plume on solar cell performances in nature conditions was undertaken on SERT-II object [3]. The essential deterioration of cell performances was not observed. However, the mercury propellant deposition rate or the condensed film thickness were not measured in this experiment.

In 1975 year, on the Russian spacecraft of "Kosmos"-type the first measurement of film thickness and deposition rate for ET with potassium propellant was conducted [4]. The Hall-type ET was used with the following parameters: power 1...3 kW, propellant flow rate \( (1...5) \cdot 10^{-3} \text{ g/s} \), exhaust plume...
effluence velocity 15...20 km/s. The deposition sensor was positioned in close proximity to the thruster, in the backflow hemisphere. The deposition film thickness was 0.1 μ per 13 minutes of thruster operation. Measurements were conducted by the "quartz-balance"-type sensor.

During "Salut-7" orbital station operation the quantitative data on spacecraft own atmosphere were obtained for operating and non-operating engines [5]. The results of mass-spectrometer measurements of particle fluxes to the spacecraft are shown in Fig. 1. It is necessary to note that the distance between the operating engine and the mass-spectrometer was approximately 10 m, during experiments the solar cells were positioned so, that they "screened" the sensor from the operating engine to the utmost. However, the sufficient pressure change was revealed at the sensor location, that was unusual according to routine knowledge of gas plume expansion. The measurement results for charged particles fluxes of SPT thruster plume are another example of plume unusual expansion (see Fig. 2).

The presented examples show that, due to complex processes in the plumes of various-type engines, propellant particles and the incoming flow particles interaction processes and other factors of space flight, the spacecraft is surrounded by the engine propellant particles cloud, whose density depends upon engine type, spacecraft geometrical configuration, etc. In other words, no one may be sure, that any spacecraft surface point inaccessible for propellants of various engines, including ET.

Alkali Metal as ET Propellant

The prospects of alkali metal use as an ET propellant are explained by the following features of alkali metals: the low atom ionization potential in comparison with other substances; capability to generate adsorptional films on ET electrodes surfaces with the low output of electrons emission; capability for long-duration storage on spacecraft board in a solid or liquid state without need of high pressurization tank for the feed system; low-voltage charge ignition at low electrode temperatures and generation of diffusive-type arc-discharge at low-voltage discharge.

However, problems related to essential changes of conventional composition of the spacecraft own ambient atmosphere and alkali metal particles deposition on spacecraft surfaces arise with the ET operation on alkali metals (Cs, Li, K).

Alkali metals have high adsorption energy relative to some materials, therefore, in contrast to gaseous propellants, they may deposite on surfaces and form sufficiently thick coatings, that lead to the disturbances of optical, electroinsulating and heat properties of spacecraft surfaces. The important properties of alkali metals, that govern the features of their effect on the spacecraft, are the following: the high chemical activity relative to some substances, the high values ions-on-atoms charge-exchange section, that in the ET plasma plume lead to generation of ions, having small velocities in the main direction of ET plasma efflux.

The main processes, that lead to generation of alkali metal particle fluxes to spacecraft during alkali metal ET operation:
- particle collisions in the ET plasma plume, followed by the change of the particle velocity vector direction to backflow direction for some particles;
characteristics, which are provided metal flux to the solar cell element; solar cell structure change, particle impingement, various evaporator; a nozzle to accelerate and to direct the alkali heat evaporation of alkali metal film on the surface, film facility was created, which included: alkali metal evaporation systems) characteristics returning may be caused formation on the solar cell surface, the experimental out that the solar cell photoelements (and other elements, inducing decrease and, moreover, solar cell optical properties of thin films depend on both film impurities on the spacecraft surface, i.e. this film may cover a surface with the effect on which may lead to its operation failure. The to structure change of islet or its migration along the one of the main spacecraft construction elements, the have islet structure, which may changes for time due alkali metals effect on the spacecraft. Solar cells are the thin films with the thickness up to several Angstrom deposition on the spacecraft various elements; nonlinear with the rise of alkali metal film. The equipment operation; During the experiments it was established that the photoelements area is 5 cm². The measurement results are shown in Fig. 3.

Laboratory Investigation

Development and improvement of alkali metal ET are accompanied for a long time by experiments of alkali metals effect on the spacecraft. Solar cells are one of the main spacecraft construction elements, the effect on which may lead to its operation failure. The alkali metal film, formed on solar cells surface, impedes solar radiation passing to solar cell working elements, inducing decrease and, moreover, solar cell energy generation failure.

In order to investigate the process of alkali metal film formation on the solar cell surface, the experimental facility was created, which included: alkali metal evaporator; a nozzle to accelerate and to direct the alkali metal flux to the solar cell element; solar cell elements, which are provided by the volt-ampere characteristics (VAC) measurement means; stabilized source of solar radiation, i.e. sun imitator; deposition measurement means. Investigations conducted in the vacuum chamber with ~ 0.3 volume at vacuum pressure ~10⁻³ Pa. To provide "without-oil" vacuum, two turbomolecular pumps of TMN-200 type are used.

To determine the quantity of deposited alkali metal the quartz microbalance is used. The solar cell silicon-base photoelements with and without protective coatings are investigated; the photoelements temperature is maintained at 20 °C level during the experiment. The photoelement area is 5 cm². The measurement results are shown in Fig. 3.

During the experiments it was established that the solar cell element power decrease rate is essentially nonlinear with the rise of alkali metal film. The photoelement output power is P - P₀ (P₀ is the photoelement power in the absence of Cs deposited film) at film thickness δ = 0.7 µ, P ~ P₀ at δ = 1.4 µ, P - 0 at δ = 2.5 µ.

Then it was established that with the time going after deposition, the alkali metal film becomes to evaporate, i.e. the solar cell photoelement returns partially (or fully in some cases) its VAC. It's known, that, as a rule, the thin films with the thickness up to several Angström have islet structure, which may changes for time due to structure change of islet or its migration along the surface, i.e. this film may cover a surface with the essential nonuniformity (i.e. on microlevel). The optical properties of thin films depend on both film thickness, and film structure, i.e. islet size, form and crystalline structure. As a whole, it's necessary to point out that the solar cell photoelements (and other systems) characteristics returning may be caused by heat evaporation of alkali metal film on the surface, film structure change, particle impingement, various radiations acting on surface.
It was established too, that independently upon protection of tested solar cell elements, the deposited Cs film does not affect the element structure, i.e. the Cs chemical interactions with the silicon surface of photoelements and with the protection coatings were absent.

So, the main factor that leads to photoelement characteristics degradation at the heat velocities of deposited Cs, is the effect of light flux "shadening" by alkali metal film deposited on the elements surface.

Investigations of alkali metal deposition on elements of laboratory experimental facility, that included plasma sources, working on Cs and K, were conducted. To determine the quantity of deposited alkali metal, the optical sensor of deposition is used, which is based on determination of transparency of glass sample surface with deposition. It was established that at alkali metal flow rate $5 \times 10^{-5}$ kg/s and arc discharge power 1kW, the alkali metal particles backflow density is small; the sensor registered decrease of light flux through the sample by value equal to the sensor sensitivity threshold ($10^{-2}$ g/cm²) after the source continuous operation during ~1 hour for the sensor position at the source nozzle exit plane, at 0.4 m distance from the source. At the same time the sensor, being positioned in the plasma plume (even out of visible boundary of the plasma plume) registered considerable deposition of the alkali metal. As this took place, at the moment of source start the film deposition rate was essentially higher than in the source steady-state operation.

The effect of caesium film thickness on solar absorptance coefficient $\alpha_s$ was investigated for thermal control coatings of various types under the laboratory conditions. As seen in Fig. 4, even films of small thickness (0.01 $\mu$) are able to change considerably $\alpha_s$ herefore, to affect the heat regime of thermal control coating and the spacecraft in whole. The similar results were obtained for other performances of spacecraft surfaces (transparence of optical units, emission coefficient $E$, etc.). Space experiments results and ground investigations of alkali metal film deposition on various surfaces lead to the necessity for a more detailed consideration of thin metallic film formation process. It's obvious, that the presence of ET propellant particles near the surface is deficient for thin films formation, since side by side with the particles adsorption process, causing thin film formation, the reverse process - desorption exists, too. The metallic propellant condensation (or film formation) has a "threshold" character, determined by the depositing film composition, particles energy, surface type and temperature. The "threshold" character of thin film formation may be illustrated by experimental results (see Fig. 5). The logarithm of caesium particles flux density on thermal control coatings of various types is plotted on the ordinate, the surface temperature of thermal control coating samples is plotted on the abscissa. This figure curves relate to various types of thermal control coatings. Any of these curves divides the parameters region in two subregions: thin films don't form in the lower subregion, whereas they will form in the upper subregion. In other words, the "critical" value of caesium particles flux density exists for each type of thermal control coating temperature, and its excess guarantees the thin film formation, whereas the presence of fluxes lower than this value rules out the therefore, to affect the heat regime of thermal possibility of caesium film formation. Note, that these results refer to Cs vapour condensation, when

![Fig. 4](image-url)
the evaporator temperature did not exceed 100 oC. The Cs particles had the corresponding energy.

To assess the results, obtained at ground facility, it's necessary to determine the ground experiments adequacy extent to space experiments. So, it's necessary to take into account such effects, as the presence of chamber residual gas, the absence of cosmic particles fluxes and radiation fluxes on the spacecraft surface, the effect of vacuum chamber volume limitation (chamber walls) on coating formation, the presence of gravitation in ground conditions, and others.

**Space Experiments**

The usage of more and more perfect diagnostic equipment on spacecraft allow in succeeding years to analyse the spacecraft own atmosphere and to measure the particles fluxes to spacecraft surface, to investigate the spacecraft plasmic environment. However, the space experiments refered to the alkali metal ET effect on spacecraft, were absent, although such thrusters were being developed in laboratories [6].

In 1990, using the "Progress M-4" spacecraft an experiment was set up with the purpose for measuring the mass of alkali metal deposition on the spacecraft. In this experiment the plasma sources provided the plasma fluxes sufficiently greater than ET of similar power. The Cs (mass flow rate 0.05...0.12 g/s) and triple alloy Na-K-Cs (mass flow rate 0.06...0.5 g/s) were used as propellants. The power of plasma source with Cs propellant at 100...200 A current reaches 2 kW, with Na-K-Cs propellant at 600 A current value was 6 kW. An investigation was performed in the framework of orbital experiment with plasma plume injection into the ionosphere [7].

The experimental program envisaged such attitude of spacecraft, that the plasma plume effluence from the plasma source was opposite to space flight direction. So, it may be argued, that this experiment conditions allowed to get a maximum value of alkali metal deposition mass.

The plasma source operated on "Progress M-4" spacecraft during three sessions, when the spacecraft was over the ground measuring posts. At the first session the plasma source operated on Na-K-Cs propellant, at the second and third sessions two plasma sources operated on Cs propellant. The plasma sources run time, when the plume was generated, was 230 s at each sessions (Fig. 6).

To registrate the surface contamination, caused by plasma source operation, the deposition sensor of "quartz-balance"-type was used. Since these sensors have a wide dynamic range and high sensitivity ($\Delta m = 10^{-9}$ g·cm$^{-2}$), they are usually used in many space programs.

In the "Progress M-4" experiment the sensor was located at the distance of $r = 0.3$ m (the first session),

![Experiment Procedure](image)

**Fig. 6**
In the first session the construction elements were positioned between the plasma source plume and the deposition sensor, so the sensor was into the shadow from the direct flux of plasma source plume particles. In this session the telemetry information acquisition conditions allowed to receive the sensor data only after 90 minutes of spacecraft flight. The total result reflects two consecutive processes: 1) during 4 minutes, deposition of particles flux from the spacecraft own ambient atmosphere with the plasma source operation; 2) during 86 minutes, evaporation of alkali metal film, previously formed during 4 minutes.

The similar conditions of telemetry information acquisition were in the third session. Data, presented in Table, were received, too, after 90 minutes of spacecraft flight. As in the first session, the result presents the overall effect of deposition (4 min) and evaporation (86 min). In contrast to the first session, the alkali metal particles deposition might occur not only from the spacecraft own ambient atmosphere, but immediately from the plasma source plume too.

In the second session the telemetry information gave the data on alkali metal film deposition during all the period of plasma source operation (230 s) and film degradation for the next 870 s. The resulted thickness of the deposited film is presented in Table. The film thickness was 0.44 μm after evaporation during 870 s.

Using this data (see Table and Fig. 3), one may evaluate the consequence of plasma source operation with alkali metal propellant for the "Progress M-4" spacecraft. The Cs film deposition of thickness up to 0.7 μm on the solar cells panels does not decrease the solar cells serviceability, whereas at the film thickness more than 2.5 μm the power loss equals to 100%. On the "Progress M-4" spacecraft the solar cells were positioned at the distance of 6.5 m from the plasma source. In order to the Cs film thickness on the solar cells was 2.5 μm and they lost all the power, the plasma source had to be operated continuously for 140 hours in the second session and 4000 hours in the third one. The similar evaluations may be done, too, for other sensitive surfaces of the spacecraft.

<table>
<thead>
<tr>
<th>Session No.</th>
<th>Average current of plasma source, A</th>
<th>Propellant flow rate, g/s</th>
<th>Film thickness, μm</th>
<th>r, m</th>
<th>Plume extension angle, degrees</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>450</td>
<td>0.45</td>
<td>0.56</td>
<td>0.30</td>
<td>85</td>
</tr>
<tr>
<td>2</td>
<td>110</td>
<td>0.07</td>
<td>1.17</td>
<td>0.16</td>
<td>85</td>
</tr>
<tr>
<td>3</td>
<td>200</td>
<td>0.07</td>
<td>0.02</td>
<td>0.17</td>
<td>65</td>
</tr>
</tbody>
</table>

### Evaluation of Effects on Spacecraft

When the spacecraft is equipped with alkali metal electric thruster, the power of possible contamination of spacecraft various systems outer operational surfaces is the factor governing the choice of propellant and electric thruster operational regime, its optimal position on the spacecraft. To evaluate the contamination power, it's necessary to found the distribution of propellant particles flow rate density in the spacecraft environment and evaluate the quantity of deposited mass, accounting the location, orientation and temperature of the contaminated surface and using the balance of rates of alkali metal vapor deposition and its evaporation from the surface.

Let us consider the model, whose basic elements are described in Ref. 8, as a model of the ET or plasma sources jet peripheral zone, which is the main source of the effect on the spacecraft. According to this model, the plasma flow is divided into two zones: 1) the zone of the expanding stream of effectively accelerated particles (1-z)N, whose velocity is constant V (N is a propellant particales flow rate); 2) the zone of the expanding stream of weakly accelerated particles zN. In these streams, the plasma flow is supposed to be frozen, and its specific heat ratio γ = 5/3.

In spherical coordinate (R, θ) with the center, lying on the jet axis at the distance from the thruster exit, which is equal to its diameter, we have the following relation for the first zone particles concentration ratio:

\[
\pi (\theta) = \pi R^2 n (R, \theta) \frac{V_{\text{max}}}{N} = \frac{\nu}{\sec^4 \theta} \left( \frac{a'}{a^2 + \tan^2 \theta} \right)^\gamma + 1
\]

(1)

where \(\nu\) is a normalising coefficient, \(a' = \tan \theta_0, \theta_0\) is a characteristic angle of the jet divergence. The considered jet zone is limited with the angle \(\theta_s\), which can be determined from the equation:

\[
1 - z/N = 2\pi R^2 V_{\text{max}} \int_0^{\theta_s} n (R, \theta) \sin \theta d\theta.
\]

(2)

The flow in the second zone is supposed to be a flow from a source and it is modelled with the function:

\[
n V (R, \theta) = n (R, \theta_\star) V (R, \theta_\star) \exp \left[ -\beta (\theta - \theta_\star) \right].
\]

(3)

This function is selected on the base of the experimental data on both neutral gas and plasma expansion in vacuum, Ref. 8. The \(\beta\) value can be found from the relation:

\[
z \dot{N} = \int_{\theta_\star}^{\pi} 2\pi R^2 n V (R, \theta) \sin \theta d\theta = 2\pi R^2 n (R, \theta_\star) V_{\text{max}} \left( \beta \sin \theta_\star + \cos \theta_\star + \exp \left[ -\beta (\pi - \theta_\star) \right] \right) / (1 + \beta^2).
\]

(4)
Using the function \( \pi \) from the equation (1), the following expression for the particles flux density can be obtained:

\[
\eta(R, \theta) = \frac{N \pi(\theta_*)}{\pi R^2} \exp \left( -\beta (\theta - \theta_*) \right).
\] (5)

It is expedient to check the adequacy of the considered model of the plasma jet peripheral zone, using it for explanation of the test results.

Let us use the data of the orbital experiment conducted on the spacecraft "Progress M-4" with a plasma source operating on Cs, Ref. 7. The source parameters are as follows: the propellant rate \( m = 7 \times 10^{-5} \) kg/s, the outflow effective velocity \( V_{\text{max}} = (2.5 \pm 0.5) \times 10^{-3} \) m/s, the plasma source diameter \( d_a = 2.5 \times 10^{-2} \) m. The mass of Cs, deposited on the sensor of 0.3 cm² area during the plasma source operation (230 s) in the second session (Fig. 6), was \( 53 \times 10^{-6} \) g. To determine the required parameters, involved in the relations for the particles flux density, the results of plasma source ground test were used: in the points \( R = 0.5 \text{ m}, \theta = 0^\circ \) and \( R = 0.25 \text{ m}, \theta = 45^\circ \), the charged particles measured concentration \( n_e = 10^{17} \text{ m}^{-3} \); the portion of the effectively accelerated ion 80...90% corresponds to \( z = 0.1...0.2 \), Ref. 8.

Using these data, we find parameter \( a', \pi(\theta_*), \beta \) and \( \Theta_0 \) at \( z = 0.1 \) and 0.2 from the relations (1), (2) and (4). Then we calculate the Cs particles flux density at the point, where the sensor is situated, by the formula (5). For the second session \( nV = 1.7 \times 10^{19} \text{ m}^{-2} \text{s}^{-1} \) at \( z = 0.2 \), and the mass of deposited Cs was \( 23 \times 10^{-6} \) g and \( 46 \times 10^{-6} \) g, respectively. A good agreement between the calculated mass values of deposited particles and those measured during the flight experiment confirms the possibility of using the model considered for the evaluation of the ET effect on the spacecraft.

Let us consider as an example two variants of the lithium MPD thruster, which were tested in a laboratory: the MPD thruster of 450 kW power and 4000 s specific impulse, and the MPD thruster of 750 kW power and 6000 s specific impulse. There were of the same flow rate (0.3 g/s), angle of inclination of the nozzle-anode edge to the thruster axis (45°) and \( d_a \) (0.3 m). According to the estimation based on the results of the laboratory tests, the portion of neutral particles, leaving the thruster, was 2 to 10% of the total propellant flow rate. So two values \( z = 0.02 \) and \( z = 0.1 \) are considered. As a limiting angle \( \Theta_0 \), characterizing the flow of the central effectively accelerated part of the jet, the Prandtl-Mayer of the stream boundary at the nozzle corner point. The Mach number at the MPD thruster exit, which was necessary for the calculation of the angle, was determined by the temperature \( T_e = 8(9) \times 10^{3} \) K, measured at the anode exit. This temperature corresponded to \( M_0 = 7(10) \). The first values are indicated here for the first variant of \( z \), and the values in the brackets are given for the second variant.

The Li particle flux density distributions are presented in Fig. 7 and 8, they are calculated ignoring possible shielding of isolated portions of a spacecraft body and outer construction elements. The chosen range of coordinate change \((x,r)\) includes, in general, the envelope sizes of those spacecrafts, for which equipment by MPD thrusters is expedient.

![Fig. 7](image-url)

![Fig. 8](image-url)
The most probable region of Li MPD usage transplanetary flights (including flights to Mars). For this conditions different spacecraft surfaces have various orientation: the solar cells are constantly directed to the Sun, the thermal control system radiators are directed, as a rule, to the opposite side; the receiving - transmitting antennae are constantly directed to the Earth. The spacecraft surfaces, excluding solar cells panels and thermal control system radiators, are rotated relative to the Sun, at the geostationary orbit the spacecraft is in the Earth shadow only for 0...5 hours, whereas at the low Earth orbit the existence times under the Sun and in the Earth shadow are approximately equal (45 min).

At the shadow side of orbit the temperature of solar cells, heat insulation and other outer surfaces decreases down to 120 K, excepting thermal control radiators, whose temperature is 230...310 K (in some types - down to 190 K). Under Sun lighting conditions the surface temperatures are: 330...350 K for solar cells, 370...420 K for shield-vacuum heat insulation, 230...310 K for windows. The special equipment may have elements with cryogenic temperature of surface (down to 20 K).

Under the Mars flight conditions the lower level of spacecraft surface temperature changes insignificantly, whereas the maximal temperature values decrease (less than 1.5 times), since the Sun constant decreases. Near the Mars orbit it is half as many as at the Earth orbit and is equal to 0.7 kW/m²

The rate of alkali metal film thickness increase or decrease depends upon the difference between the evaporation rate from the surface at the given temperature and the deposition rate on the surface (see Fig.18 in Ref.6, for example).

If the Li particles flux density is equal or more than $3 \cdot 10^{15} \text{ m}^{-2} \text{s}^{-1}$ in the region of solar cells location (-1m $\leq r \leq$ 3m, 2m $\leq r \leq$ 6m), the estimated rate of Li evaporation from the surfaces at all ranges of temperatures is less than the deposition rate by a factor of 2 and more orders. Then, if not to provide additional protection, a film as thick as 1 $\mu$m may be deposited here $4.5 \cdot 10^4$ hours of operation.

So, the indicated estimations permit to maintain that the Li utilization requires the special measures for spacecraft surface protection (shields, heating and so on) and the Cs use in electric propulsion has no restrictions in practice, taking into account Cs evaporation rate is higher by a factor of few order in comparison with Li.

**Conclusion**

The theoretical analysis and data of experiments (carried out in ground laboratories and in flight conditions) permit to make the following conclusions.

The main factors, effecting on unwanted event - formation and growth of the alkali metal film on the spacecraft surface, are the particle flow magnitude per unit surface $nV$ and the surface temperature. The situation becomes more beneficial for the spacecraft with increasing the temperature $T$ and decreasing the $nV$.

Consequently, any measures, directed on spacecraft surface temperature increasing, namely, location of the most sensitive to the spacecraft construction elements alkali metal deposition in the "hot" areas, solar radiation utilization, supplementary special heating of the separate surfaces and so on are desirable.

The effects that are responsible for particle flow decreasing: arrangement of the electric propulsion in the areas spaced maximum widely from spacecraft surfaces sensitive to the alkali metal deposition; the measures on special management of the on/off modes of electric propulsion (for example, propellant flow rate reduction at the moments of starting and discharge voltage feed advance to the moment of beginning the propellant feed to the engine), electric propulsion optimum operational modes selection (duration and relative pulse duration of the electric propulsion on), utilization of the special protective screens and so on can be a great advantageous.

So, depending on concrete conditions, being occurred at the given point of the spacecraft (and the relation between $nV$ and $T$), both the growth and the reduction (degradation) of the thickness of just deposited film of alkali metals can be observed. It is well to bear in mind that the films of various thickness can exist and it will not disrupt the operationability of the spacecraft construction elements in practice. At last, it should be remembered that various spacecraft design elements can be responsible to the film deposition variously and different alkali metals are differ widely in ability of deposition (for example, Cs is far less favourably disposed towards the condensed film producing than the Li).

By and large we can say that unwanted event - alkali metal film deposition on the spacecraft surface cannot be an obstacle for utilization of the alkali metals as propellants in electric propulsion. Required precautions can be always taken to compensate these deleterious effects (it is particularly easily to make as to the Cs).

**References**

