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PLENARY SESSION

A brief history and an overview of EDB Fakel

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This speech recounts the brief history of EDB Fakel, including its origin in a Naval theme, next following reorganization for the Soviet Mars exploration program and a beginning of Stationary plasma thrusters' application. An overview of developed and perspective products by EDB Fakel, namely «Stationary plasma thrusters», «PlaS-type plasma thrusters with a hollow anode», «Thermal catalytic thrusters» and «Integrated thruster units» is demonstrated. General present-day activities of EDB Fakel are set forth.

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Thrusters with closed drift of electrons: ideas and trends of development in the past and at present time

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Key words: plasma thruster, closed drift of electrons, design development ideas, electric and magnetic fields, processes and performance control, acceleration channel, magnetic induction distribution, accelerating layer, channel geometry, wall erosion, magnetic shielding, accelerating layer thickness and position.

In connection with the Fakel's 65- years anniversary it is reasonable to review some ideas determining the progress in the development of thrusters with closed electron drift (stationary plasma thrusters - SPT and thruster with anode layer-TAL). TAL was invented in 1961 by the scientist of the Kurchatov Institute of Atomic Energy (KIAE) Askold Zharinov [1,2] on base of his studies at the end of 1950-th demonstrated the possibility to accelerate ions in the plasma layer with ExB fields. By the beginning of the 1970-th a series of the effective enough double stage TAL laboratory (lab) models able to operate with powers till 100 kW and specific impulses till 8000s had been developed by team of the Tsentral Institute of the Machine Building (TsNIIMash) under Zharinov's leadership. These works had confirmed the possibility to effectively ionize the propellant particles and to accelerate obtained ions at least in the double stage ExB discharge.

In USA in 1962 there had been invented the ion accelerator having the main elements of the modern SPT [3], and during several years there were studied operation and characteristics of this type accelerator lab models. But in 1965-1966 these activities were stopped, probably, due to low thrust efficiency of these lab models in comparison with ion thrusters. During the same times in USSR the team of IAE scientists under leadership of Alexey Morozov had developed and studied the SPT and by 1967-1968 the thrust efficiency of the SPT lab models of the 1-3 kW power level were demonstrating an accelerator ("anode") thrust efficiency up to 30-35 % and specific impulses up to 1800s [1,2]. Obtained performance were better than that ones of the other plasma thrusters of the same power. This result was obtained due to the following ideas [1,2]:

- Morozov idea to create an extended ExB layer in plasma and to use the "focusing" magnetic field lines in the acceleration zone to focus ions during their acceleration;
- an idea of IAE team to use the noble gases as propellant what was simplifying testing of thrusters and future solution of the problems for the propellant storage and operating thrusters compatibility with the other spacecraft systems**.**

Further studies had shown that an idea to focus ion flows in the SPT discharge were not very successful but creation of the "focussing" magnetic field topology was connected with an increase of the magnetic field induction in direction from anode to the accelerating channel exit at least in the part of accelerating channel. And this feature ensured more stable SPT operation what was firstly theoretically analyzed by Morozov and confirmed experimentally by IAE team.

More over this idea allowed optimization of the ratio of the accelerated ion flows to the external and internal accelerating channel walls changing the magnetic field line geometry[1,2].

The next idea of Morozov was to realize the 1st SPT flight test. It was not a scientific one but was very useful for the progress of the SPT development in USSR [1,2]. This idea was supported by academicians L.Artsimovich, head of the IAE plasma research division, A. Alexandrov, IAE director, A. Iosiphian, director of the All-union Institute of Electromechanics (VNIIEM) and by Yu. Danilov, head of the developmental division in the State Committee on Atomic Technology. As result this flight test was prepared and successfully realized in 1972 onboard Meteor satellite of VNIIEM by teams of IAE, VNIIEM, Design bureau (DB) Zarya and its division in Kalinigrad (nowadays the experimental DB Fakel) [4]. During this test the 1st experimental SPT successfully operated in space and had confirmed the possibility to correct a satellite orbit. Therefore this test is considered as the birth of SPT as a space propulsion thruster and Morozov nowadays is called in Russia as a father of SPT. The mentioned test had activated the SPT studies in USSR and besides IAE and design bureau (DB) Fakel there were involved into these studies the Moscow Aviation Institute (MAI), Tsentral Institute of the Aviation Motor Building (TsIAM), Moscow Institute of Radioelectronics and Automatics (MIREA), Moscow State University(MSU) and others.

The next interesting Morozov's was the electromagnetic thrust vector control by creating the different magnetic field lines inclination relative to the thruster axes in the different azimuth sections of the accelerating channel. To realize such non uniformity the IAE scientists had developed a magnetic system with additional coils and narrow flat external magnetic poles. And it was shown that with such magnetic system it is possible to get the deviation of the SPT thrust vector up to 5-7 degrees from the thruster axes. But more important was that this model had demonstrated significantly higher thrust efficiency than models with extended in longitudinal direction magnetic poles used in SPT models earlier [1,2]. Further studies had shown that an increase of thrust efficiency is occurred due to the high rate of the magnetic induction grow from anode to the exit direction and due to positioning of the accelerating layer (AL) with the main part of the electric potential drop near the exit of the accelerating channel. But models with the narrow poles had low lifetime. Therefore MAI experts had studied dependence of the thickness and position of AL on the magnetic field lines geometry and magnetic induction distribution along the channel using control of the magnetic field characteristics to realize the following ideas:

- to create the magnetic field in the AC required to get high thrust efficiency of thruster with increased discharge chamber wall thicknesses in the AC parts bounding AL;
- to shift the AL and wall erosion zones in the exit direction beyond the magnetic poles plane by increasing the rate of the magnetic induction grow in the exit direction and shifting the magnetic induction maximum in the same direction to get high enough thrust efficiency and to increase the wall thicknesses beyond the magnetic poles plane and thruster lifetime.

And there had been invented solutions allowing realization of the mentioned ideas. As result by DB Fakel under support of MAI the 2nd generation thrusters of the SPT-50 and SPT-70 types had been developed with high enough performance level and increased by 3-5 times lifetime relative to the models with the "narrow" poles as well as the SPT-100 and SPT-140 thrusters with lifetime till ~10000 and higher. Nowadays the SPT-230 thruster was developed at Fakel using the same basic solutions. Performance and lifetime of these thrusters are sufficient to solve the most part of tasks for the motion control of spacecrafts with mass over 500 kg.

In the mean time the significant progress was achieved in the TAL development, namely:

- the series of the single stage TAL lab and engineering models with the competitive performance had been developed also using the magnetic systems creating the increasing magnetic induction from anode to the channel exit and one of these models was successfully tested in space onboard US satellite; - -
- there was developed the TAL models with accelerating layer fully shifted out of the thruster

design by shifting the magnetic induction distribution maximum far enough from the magnetic poles and positioning anode exit plane approximately in the "magnetic poles plane"[5].

It's worth to note that the last result had realized the dream of Zharinov to get the fully electrode-less or wall-less ion flow acceleration in the ExB plasma sheath. The rest problem for such thruster is protection of the magnetic poles from sputtering by the high energy radial ion flows appearing in the boundary regions of the main ion flow at the channel exit due to great electric potential difference in between these regions with high potential and magnetic poles having typically potential close to the cathode one. And traditional for TAL protection is made by the graphite guard rings.

Nowadays one of the challenging problems is the further SPT and TAL lifetime increase. For TAL as was mentioned above solution of this problem was already found. For SPT case nowadays there are studied the following ideas:

1. The so-called "magnetic shielding" [6], wich is realized in JPL and other USA groups, in the Fakel PlaS type thrusters [7] and Russian Keldysh Center and other labs and firms in the world. This shielding is achieved br the shift of AL in the exit direction by moderate shift of the magnetic induction distribution maximum beyond the magnetic pole plane and significant widening of the exit part of AL in the inter pole gap.

2. Shift of the accelerating channel exit part and its widening beyond the magnetic pole plane by 40-45 degrees at each wall and shift of the magnetic induction maximum area out of thruster design sufficient to position whole AL and wall erosion zones in the widened part of the AC to obtain magnetic shielding of the remaining wall parts to protect magnetic poles after long thruster operation[8].

These approaches are in elaboration now and it will be shown in the given paper that the physical bases for these approaches are the same.

ACKNOWLEDGMENTS

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TECHNICAL SESSION A

ION THRUSTERS

Electron emissions from NACES high performance cathode based on C12A7:e- *electride* material for in-space electric propulsion applications.

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The NACES (Narrow and controlled energy beam spectrum) electron source is based on the novel electride C12A7:e- material. This cathode is intended to provide an electron flow for primary plasma production and/or ion beam neutralization of plasma thrusters. This low work function ceramic material provides and effective thermionic electron emission currents at operation temperatures for below of those thoriated tungsten or lanthanum hexaboride currently used. Thus, it might improve the performance of hollow cathodes or other thermionic electron emitters currently employed in space as electron sources.

The physical properties and characteristics of C12A7:e- material will be reviewed [1-3] in the context of electric propulsion applications. The NACES cathode design incorporates this promising ceramic to produce a substantial electron flow for ion beam neutralization. Its electron emission performance has been studied in the laboratory by electrically biasing the cathode with respect to a metallic electrode [3]. The observed dependence of these current-voltage (IV) emission curves with the electride sample temperature will be discussed as well as future improvements in NACES cathode design in the perspective of its space applications.

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GAS-ELECTRIC ISOLATOR OPERATION AS PART OF A LOW-POWER RADIO-FREQUENCY ION THRUSTER WITH A WIDE RANGE OF THRUST CONTROL

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Radio-frequency ion thrusters (RIT) can become an alternative to stationary plasma thrusters (SPT) to solve a number of small satellite motion control tasks. The development of such thrusters with various power levels has been carried out at the RIAME MAI since 2010 [1]. A low-power radio-frequency ion thruster (LP RIT) with a wide range of thrust control is among them. It is intended for use on board the low-orbit small satellites for precise compensation of the aerodynamic drag force. The gas-electric isolator in the xenon feed line to the gas-discharge chamber (GDC) is one of the components of such thruster [2]. Its function is to separate electrical circuits under different potentials.

Thrust control in a wide range is provided by substantial changes in the propellant mass flow rate and emissive electrode voltage. For such conditions, a gas-electric isolator of special design was developed with a function of pressure dependence on flow rate varying within the limits sufficient to ensure the absence of gas breakdowns. Tests of such gas-electric isolator at the special experimental facility allowing for making tests in the presence of radio-frequency field and plasma in the GDC, verified that its properties correspond to the conditions of operation as part of LP RIT with a wide range of thrust control.

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OPTICAL PLASMA DIAGNOSTICS FOR RADIO-FREQUENCY ION THRUSTERS

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Key words: *ion thrusters, plasma diagnostics, optical diagnostics, plasma parameters.*

The development of a method for non-invasive measurements of plasma parameters inside the discharge vessel of a radio-frequency ion thruster (RIT) is the main objective of this work. With this method the characterization of a RIT will be possible without disturbing the thruster operation by inserting a probe. Furthermore, the establishment of such a method is supposed to contribute to shorter test cycles for thruster qualification since the thruster can directly be tested in flight configuration, as well as improving lifetime and performance.

To obtain a plasma's parameters from the line intensities of an optical emission spectrum (OES), it is usually necessary to consult complex theoretical models. Such models exist for e. g. argon [1] and xenon [2, 3]. For alternative propellants such as krypton or iodine which are of major interest [4, 5], new models need to be established. Especially for molecular propellants like iodine, these models will be much more complicated as more species in the plasma need to be considered and the required parameters are not well known to date. To overcome this challenge, the plasma parameters are measured with a Langmuir probe while the OES is recorded simultaneously. The setup is shown in figure 1. The measured spectra are correlated with the Langmuir plasma parameters. Several correlations could already be found for both xenon and krypton [6], an example is shown in figure 1. With these obtained correlations and an OES from a RIT, the plasma parameters inside the RIT's discharge vessel can be obtained.

The method will be applied to a setup in which the RF-coil is surrounding a glass vessel similar to a RIT as well as to a RIT-10 itself to measure the plasma parameters at different points of operation. The resulting plasma parameters will be compared to other methods, e. g. RPA measurements. Also, a simple theoretical approach is pursued for comparison, in which the OES is calculated from the electron energy distribution function (EEDF).

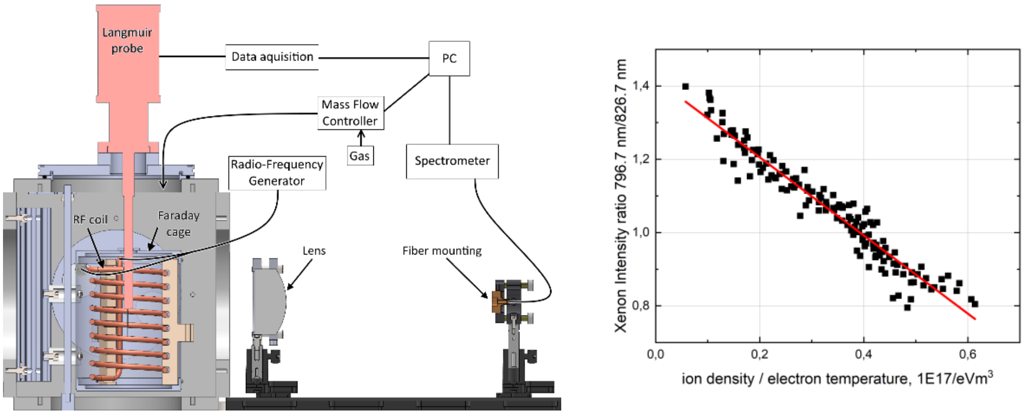

Figure 1: Correlation measurement setup (left) and an example of a correlation for xenon (right) [6]

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Compatibility of the electride C12A7:e- with alternative propellants and production of thin films

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Keywords: *electride, C12A7:e-, hollow cathode materials, alternative propellants, thin films*

A neutralisers performance is, among others, judged by the total electron current emitted from the insert and its lifetime. Since the extracted current mostly increases with operating temperature while the lifetime decreases, a mission-based compromise must be made. C12A7:e- is a crystalline anionic material (see Fig. 1), which allows electron emission even at low temperatures. Current research activities indicate that it might be operated already at room temperature [1] due to its low work function of 0.6 eV – 2.4 eV [2,3]. State of the art materials are operated at higher temperatures (1100°C for BaO and 1800°C for LaB6 [4]). A decreased operating temperature may increase the lifetime of the neutraliser system.

However, C12A7:e- has a significant lower thermal conductivity compared to LaB6 [5,6]. In this work thin films of the electride will be presented as an alternative approach to bulk materials. Deposited on a high conductive material, the thermal properties of the device may be improved while one still profits from the excellent emission properties of C12A7:e-. The setup and the fabrication technique will be discussed.

Another aspect discussed in this paper will be the compatibility of this novel material with various alternative propellants such as Iodine or dirty Xenon. For electric thrusters some alternatives to the expensive usage of Xenon exist already, but these propellants also need to be qualified for the corresponding neutralizer. Influences on the material's surface structure and the work function will be analysed.

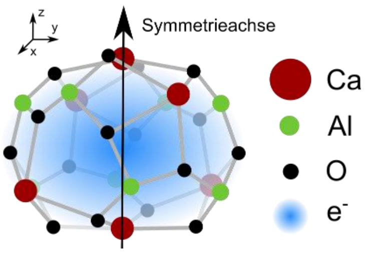

Figure 1: Crystal structure of C12A7:e-

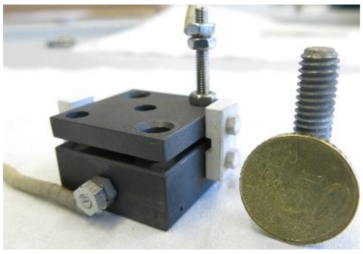

Figure 2: Setup for work function determination

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INVESTIGATION OF C12A7:E⁻ UNDER HARSH CONDITIONS IN RELATION TO HOLLOW CATHODE NEUTRALIZERS

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Key words: *C12A7:e⁻, hollow cathode neutralizer, Raman spectroscopy, thermionic emission*

The ceramic C12A7:e⁻ gains the interest of more research groups and companies; especially, in electric propulsion in connection with hollow cathodes. The remarkable properties of the material are its low work function, high thermionic emitting current, and the low reactivity with other substances, once it is exempt from its free oxygen anions in a reduction process.

As for other emitter materials in hollow cathodes, it is very important to know the effects of long operation phases on the performance of C12A7:e⁻. One major concern is the contact with residual oxygen molecules in the propellant and the test chambers.

In this paper, we investigate the effects of oxygen on the performance of the material in terms of emitted current and work function. Oxidation rates of the material at different temperatures and atmospheres are measured, while the level of oxidation, which corresponds to the electron concentration, is investigated via Raman spectroscopy. This work aims for correlating the electron concentration to a certain work function and extracted current.

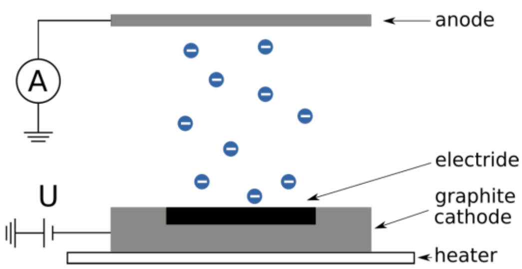

Figure 1: Schematic of the setup for measuring the work function

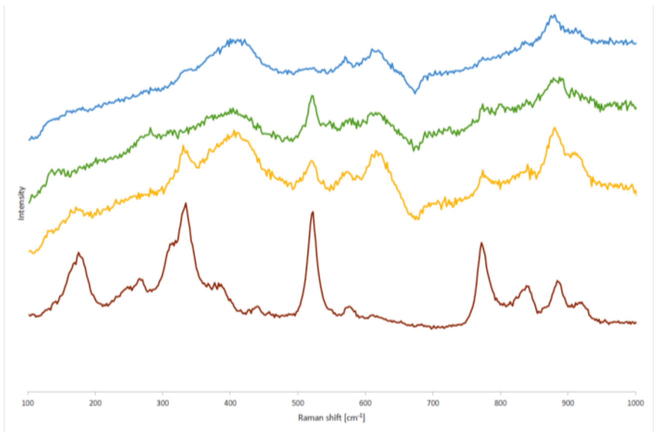

Figure 2: Raman spectra of C12A7 samples at different electron concentrations (bottom curve with the lowest concentration)

RESEARCH TESTS OF LOW-POWER RADIO-FREQUENCY ION THRUSTER WITH ELECTRODES OF ION-EXTRACTION SYSTEM MADE OF CARBON-CARBON COMPOSITE MATERIAL BASED ON NON-WOVEN CARBON FRAME IPRESSKON®

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To effectively solve the problems of controlling the orbital motion of small spacecraft in low orbits, it is necessary to equip such vehicles with corrective propulsion systems with high specific impulse and long lifetime. Ion thrusters (IT), in which the value of specific impulse reaches several tens of kilometers per second and the lifetime - several tens of thousands of hours [1], can be considered as thrusters for such propulsion systems. From the point of view of IT lifetime, the most critical of its elements is the accelerating electrode of the ion-extraction system (IES) that is destroyed under the influence of charge-exchange ions, which are born in the interelectrode gap and in the zone of ion beam neutralization [2]. The erosion rate of the accelerating electrode, ceteris paribus (the geometry of IES holes, the potentials of the electrodes, the charge-exchange ion current density distribution), depends on the coefficient of ion sputtering of the material. The carbon-based materials have the lowest ion sputtering coefficient.

Traditional carbon-based materials (isostatic graphite and pyrographite) have low values of tensile strength and modulus of elasticity, and this causes problems when using them as the material for the manufacture of IES electrodes. This problem is solved by using carbon-carbon composite materials (CCCM), however, CCCM based on the woven or thread reinforcing frames have a heterogeneous structure due to the size of the unit cell (0.5-1.0 mm) that prevents high-grade finish and leads to frequent interelectrode breakdowns.

JSC "Kompozit" has mastered the technology of production of the CCCM based on non-woven carbon frame Ipresskon® (CCCM-Ipresskon®) [3]. This constructive material, while retaining the main advantages of CCCM based on woven reinforcing frames, has the minimum possible for CCCM size of the structural cell (10-30 microns), which allows ensuring the required roughness of the surface for the IES electrodes made of CCCM-Ipresskon®, comparable to that of the metal electrodes. The low value of the surface roughness, as well as of the coefficient of linear thermal expansion, makes it possible to produce not only the accelerating, but also the thin emission electrode of IT IES from CCCM-Ipresskon®.

IES electrodes from CCCM-Ipresskon® were produced for a laboratory model of a low-power radio-frequency ion thruster (RIT) [4]. The laboratory model of RIT with electrodes from CCCM passed research tests with duration of about a thousand hours. The paper presents the main operating parameters of RIT, taken during the tests. The influence of deposition of the accelerating electrode sputtered material on the walls of the RIT discharge chamber on the ionization rate of the plasma-forming gas was investigated.

After the tests, the structure of the accelerating electrode of IES was examined using an electron microscope. The experimentally obtained erosion pattern of the electrode surface facing the ion beam neutralization zone was estimated using a profilometer. Such pattern was compared with the one calculated by the software package IOS-3D [5].

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NUMERICAL MATHEMATICAL MODEL OF RADIO-FREQUENCY DISCHARGE IN THE DISCHARGE CHAMBER OF RADIO-FREQUENCY ION SOURCE

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Key words: *radio-frequency discharge, mathematical modeling, radio-frequency ion source, free molecular flow, ionization.*

Today, one of the problems of current interest is the cleaning of near-earth space from space debris. One of the ways to solve this problem is to create a spacecraft that will transport space debris objects to disposal orbits. In this case, a promising method is to push space debris objects with an ion beam generated by an ion source located on board the spacecraft. The advantages of this method are: weak divergence of the generated beam, a high velocity of ion outflow (which saves the propellant), and a significant lifetime of ion sources [1]. A mathematical model of the discharge in the discharge chamber of a radio-frequency ion source is presented, which allows one both to calculate the integral parameters of the thruster and to obtain the distribution of local plasma parameters over the discharge chamber volume. The presented model differs by relatively low requirements for computing resources needed for the calculation. The mathematical model consists of four interconnected submodels providing calculation for the following: density of neutral atoms, density of ions, distribution of the root-mean-square values for the time-dependent component of the electromagnetic field and for eddy electron currents in the discharge, and the stationary component of the electric field in the chamber volume. The geometry for every submodel is considered as axisymmetric.

The density of neutral atoms is calculated for a free-molecule flow in the chamber [2]. The calculation is based on placing the ring sources of particles at the boundaries of the considered region. The boundaries of the region are set by a number of linear segments. Particle flows between segments are considered. Boundary conditions of three types are set at the region boundaries: an inlet, a wall, and an ion-extraction system (partially transparent wall). The particle flow from the inlet boundary is calculated on the basis of the known propellant mass flow rate. The result is a system of equations, the solution of which gives the flows of particles. The neutral particle density in the chamber volume is calculated using such flows. Besides, the ionization effect on density is taken into account.

The calculation of the ion density is based on modeling the ion motion in the stationary component of the electrostatic field in plasma [3]. First, the initial distribution of electrostatic field and the distribution of ion density are set. Then the ion trajectories are calculated, on the basis of which the ion density is recalculated. It is subsequently used to recalculate the electric potential distribution. The calculation is performed iteratively until it converges. Besides, the effect of the neutral atom density and the temperature of the electrons, the distributions of which are calculated by other submodels, on the ion origination is taken into account.

The non-stationary electric field is calculated taking into account the coil location, its current amplitude and frequency, and the eddy currents in plasma. Besides, the effect of the charged and neutral particle density on the collision frequency in the plasma is taken into account.

To verify the model, the calculated integral parameters are compared with the experimental values for the thruster with an ion beam diameter of 10 cm. Good agreement between the calculation and experiment is shown.

The resulting mathematical model can be used both for calculating integral parameters while designing radio-frequency ion sources, and for analyzing processes in a radio-frequency inductive discharge.

This study was supported by the grant of the Government of the Russian Federation allocated from the federal budget for state support of scientific research conducted under the guidance of leading scientists in Russian educational institutions of higher education, research institutions and state research centers of the Russian Federation (Contest 7, Resolution No.220 of the Government of the Russian Federation of April 9, 2010). Agreement No. 075-15-2019-1894 of December 3, 2019.

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NEW FEATURES IN APPLICATION OF HIGH-DENSE CARBON-CARBON COMPOSITE ON NON-WOVEN BASIS IN ION SOURCE'S ION EXTRACTION SYSTEMS

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Keywords: carbon-carbon composite, non-woven preform, oxidized polyacrylonitrile, structural homogeneity, complex-profile grids

Materials traditionally used for grids in an ion-extraction system (IES) of an ion thruster (IT), such as titanium- and molybdenum-based alloys have a number of serious disadvantages, including the main ones: relatively high values of ion sputtering coefficients characterizing erosive ablation; relatively high values of thermal expansion coefficient (CTE); the danger of intergranular corrosion.

Alternatively, carbon materials such as artificial graphites could be used for IES grids instead of metal alloys. Given their significantly lower ion sputtering coefficient and CTE, they are the most promising materials when used for the accelerating grid as a key structural component of the ion thruster in terms of lifetime [1].

However, the low strength of graphites due to the lack of reinforcement imposes a serious limitation on their use in the manufacture of particularly thin and complex-structures. In contrast, carbon-carbon composite materials (abbreviated as C/C-composites) do not have this disadvantage, given their reinforcing system and high mechanical properties. However, commercially available C/C-composites, either fabric- or core- or filament-based, are characterized by a relatively large structural cell (0.5-1.0 mm) and structural heterogeneity, thus making it impossible to ensure the required high surface finish of the grid.

That said, C/C-composites developed in JSC Kompozit based on oxidized non-woven polyacrylonitrile (PAN) have a somewhat different structure. We obtained a fine-meshed reinforcing carbon preform branded as Ipresskon® with a density of at least 0.5-0.6 $g{\cdot}cm^{-3}$ [2], as well as a pitch coke matrix-based high-dense C/C-composite (density ≈ 1.90 $g{\cdot}cm^{-3}$) with this preform. The extremely small size of its structural cell (10-30 μm) and low porosity (less than 1%) made it possible to obtain an accelerating grid for a complex-profile IES of IT with a low surface roughness (R_a less than 0.6 μm) using mechanical processing [3].

However, given its low CTE in the temperature range from +20 to 400°C (merely 0 to $1.0{\cdot}10^{-6}\ K^{-1}$) indicating its high dimensional stability in the wide temperature range, as well as its sufficiently high mechanical strength, C/C-Ipresskon® can also be used for a thin emission grid. The uniform microporous reinforcing system C/C-Ipresskon® allowed us to produce a thin emissive grid with a thickness of only 0.3 mm with a surface roughness of R_a = 0.16 μm (10th purity grade for metals). Combination of the resulting properties of C/C-Ipresskon® made it possible to achieve an intergrid gap of less than 1 mm in IT IESs having a diameter of up to 200 mm.

The results of analysis of the microstructures obtained are presented.

A new application of C/C-Ipresskon® as part of IES of a slightly divergent beam ion injector is described. Preliminary results of comparative tests between an ion injector with a complex-profile IES [4] and that one with a classical round-aperture IES are presented.

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Radio frequency ion source temperature field modeling and measurements

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Key words: *radio frequency ion source, heat fluxes, electrodes of ion extraction system, thermal modeling, thermal imaging measurements.*

Model of thermophysical processes occurring during the operation of radio-frequency ion source (RIS), which can be used for several applications including space debris objects contact-less remove by ion beam is proposed. The model uses analytical expressions for a number of components of heat flows coming out of the discharge plasma onto the surfaces of structural units that differ significantly from those used in known computational models [1-3].

The main differences of this computational thermal model relate to the distribution of radiant heat flows over the source surfaces.

An additional radiant flux being a result of recombination of ions and electrons on the surfaces bordering the discharge, namely the surfaces of gas-discharge chamber (GDC) and screen grid (SG), is introduced. In available models, it is assumed that the ionization energy eV_i is released and entirely absorbed by the wall at the point of the surface, near which recombination occurred. In reality, the photons originated as a result of recombination transfer the energy spatially with distribution close to isotropic. The surface roughness changes the isotropic indicatrix by reducing the radiation coming out at wide angles of photon escape. This radiant flux is partially absorbed by the surfaces of GDC, SG and accelerating grid (AG). The recombination radiation consists of the UV light quanta for more than 2/3. The share of the absorbed flux is defined taking into account the coefficients of UV radiation reflection R_{GDC} and R_{SG}.

An analytical solution for the densities of power fluxes p_{GDC}, p_{SG}, q_{GDC}, q_{SG} leaving the surfaces of GDC and SC and absorbed by them (fig. 1) is given in [4, 5] for a hemispherical GDC and with the assumption of radiation cosine indicatrix.

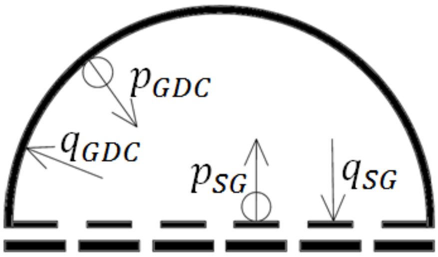

Fig. 1 – Recombination radiation fluxes in hemispherical GDC

Materials used were Al_2O_3 for GDC and Ti and C for SG. The values for R_{GDC} and R_{SG} are taken from [6, 7]. The calculated flux densities are shown in table 1. Relative power fluxes are introduced for convenience:

$$\bar{p}_{GDC} = \frac{p_{GDC}}{A},\ \bar{p}_{SG} = \frac{p_{SG}}{A},\ \bar{q}_{GDC} = \frac{q_{GDC}}{A},\ \bar{q}_{SG} = \frac{q_{SG}}{A},$$

where $A = j_B V_i$, j_B is the Bohm ion current density.

Numerical values in the first row of table 1 were calculated for the case of absence of UV radiation reflection that was implicitly accepted in models [1-3]. Even in this case, $\bar{q}_{GDC} = 0.875$ and $\bar{q}_{SG} = 1$, which means that GDC absorbs 12.5% less power comparing to value $j_B V_i$ that is used in the known models (ref. to column 3). If UV radiation is partially reflected, the power densities $\bar{q}_{GDC}$ and $\bar{q}_{SG}$ decrease, $\bar{q}_{SG}$ being 22% less with Titanium used for grids.

Table 1 – Relative power flux densities for recombitanion radiation

R_{GDC}	R_{SG}	$\bar{q}_{GDC}$	$\bar{q}_{GDC}^{(0)}$	$\bar{q}_{SG}$	$\bar{q}_{SG}^{(0)}$	$\bar{p}_{GDC}$	$\bar{p}_{SG}$
0	0	0.875	1	1	1	0.5	0.25
0.12 (Al_2O_3)	0.11(C)	0.848		0.993		0.616	0.311
0.12 (Al_2O_3)	0.3 (Ti)	0.898		0.786		0.622	0.418

The radiant flux with density $\bar{p}_{GDC}$ through the holes in SG reaches the AG changing its thermal balance, which is not taken into account in any known model.

Another difference of this model is related to the calculation methodology for the distribution of radiant flow caused by de-excitation of atoms and ions over the surfaces of GDC and both grids of the ion-extraction system (IES).The density of this power flow was calculated within an approximation of 0-dimensional plasma. It turned out that the flux density on the grids decreases from the center to edges more than twice, and on the GDC surface the flux density decreases from the apex to the bottom for about one and a half times. All known models use the uniform surface density distribution for such flux.

Besides, components of radiant heat fluxes in the IES unit cell were calculated using a better physically justified methodology. The shares of the radiant flux entering the cell, which are absorbed by IES screen and acceleration grids, as well as shares which come out of the source and return to GDC, were determined. The flux shares were calculated taking into account the UV reflection coefficients R_{GDC} and R_{SG}.

When using the proposed method, the value of the heat flux reaching the accelerating grid has increased most significantly (by 70%) compared to the heat flux calculated using the previously accepted equations. This circumstance should be taken into account in the computational thermal models of ion source with perforated electrodes.

Based on the developed physical-mathematical model, a computational thermal model was developed, which was verified by temperature measurements. Temperature calculations and measurements were made for RIS-10 source with an ion beam diameter of 8 cm in three different operating modes. The deviation of the numerical values of temperatures is within 5%, which allows making a conclusion that the computational thermal model is adequate.

This study was supported by the grant of the Government of the Russian Federation allocated from the federal budget for state support of scientific research conducted under the guidance of leading scientists in Russian educational institutions of higher education, research institutions and state research centers of the Russian Federation (Contest 7, Resolution No.220 of the Government of the Russian Federation of April 9, 2010). Agreement No. 075-15-2019-1894 of December 3, 2019.

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LASER-INDUCED FLUORESCENCE VELOCIMETRY ON A RADIOFREQUENCY-DRIVEN ION SOURCE

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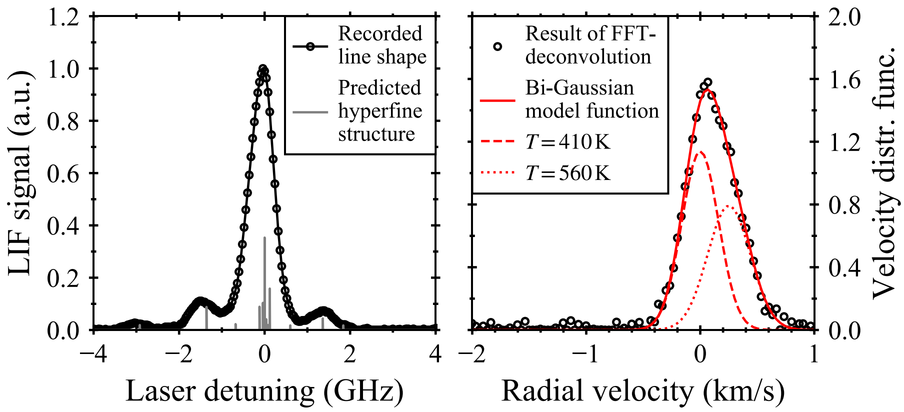
Key words: *plasma diagnostics, laser-induced fluorescence, velocity distribution function, radiofrequency ion source, radiofrequency ion thruster.*

Laser-induced fluorescence spectroscopy (LIF) is a non-intrusive plasma diagnostics technique allowing for species-sensitive measurement of local plasma parameters such as density, mean velocity, and velocity dispersion of ions and neutrals. Gaining knowledge of these quantities aims at improving the understanding of electric propulsion's underlying plasma physical mechanisms and is needed to provide a basis of data for thruster efficiency and lifetime simulations.

In LIF-based experiments, the absorption profile of a specific optical transition associated with the species under investigation is scanned by means of a laser; the fluorescence caused by subsequent deexcitation of atoms or ions within the probe volume is recorded from which the quantities of interest can be extracted. In particular, LIF velocimetry provides a way to determine velocity fields and velocity distribution functions (VDFs) from the Doppler-shifted and -broadened absorption line shapes mapped by the fluorescence signal.

As an extension to the Advanced Electric Propulsion Diagnostics (AEPD) platform developed at IOM [1], a LIF setup targeted at VDF measurement of Xe and Xe^+ has been realized recently. It comprises a tunable high-output-power diode laser system operating in the near infrared and fiber-coupled vacuum probes for both laser excitation and fluorescence detection offering various orientational options. Laser wavelength and output power are simultaneously measured; phase-sensitive detection is used for LIF-signal amplification.

So far, LIF velocimetry has been applied quite extensively for characterization of Hall-effect thrusters (e.g. [2−4]), whereas gridded ion thrusters have not yet been in the focus of such examinations. In this contribution, we present LIF velocimetry data obtained from measurements in the beam and within the discharge chamber of a radiofrequency-driven ion source. Furthermore, we plan on using our LIF setup to carry out measurements with a RIT10 thruster.



Left: Xe-LIF signal recorded in the center of the discharge chamber of a radiofrequency-driven ion source and hyperfine structure of the probed transition.

Right: Corresponding Xe-radial velocity distribution function extracted by means of deconvolution.

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TECHNICAL SESSION B

STATIONARY PLASMA THRUSTERS

iodine Fed Advanced Cusp Field Thruster - iFACT

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Since several years, the industrialisation of the satellite market has accelerated. In particular, the market for telecom satellites has changed significantly, because of several new players and numerous plans for satellite constellations. In addition, also the market for Earth observation satellites is in motion, due to the fact that the new market actors are not only focusing on telecommunication services, but also target earth observation missions.
In parallel to this development, electric propulsion, with its high mass efficiency, became an enabler to improve the cost efficiency of satellites, especially with respect to launch costs, because it allows the launch of more satellites per launch vehicle instead of one. So far, the savings have been potentially eaten up by the additional costs of the electric propulsion system, which are mainly influenced by the complexity of the required electronics (PPU) and the high cost of the xenon propellant.

In the scope of the European framework programme for research and technological development Horizon 2020, the iodine Fed Advanced Cusp Field Thruster programme investigates ways to enable:

- Iodine as disruptive propellant for electric thrusters,
- Maturation of the Advanced Cusp Field Thruster (ACFT) as disruptive thruster principle, in three different power classes,
- Calcium aluminate (C12A7) as disruptive, low-work function emitter material for cathodes,
- Significant reduction and simplification of the PPU

The programme strongly focuses on iodine as alternative propellant for electric thrusters. Iodine as propellant can lead to a significant simplification of the propellant feeding subsystem architecture, which enables a significant cost reduction. It will further allow decreasing the mass of the feeding subsystem because of its high storage density. In order to enable iodine as propellant several challenges have to be mastered such as its corrosive nature, potential feeding line clogging, or cathode compatibility. The programme aims to identify and solve each single challenge to increase the technology readiness towards a higher level.

We will present an overview of the iFACT programme, its current status including preliminary results and the key milestones of the programme.

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Electric Thruster Selection Criteria and Examples

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The paper presents a set of very well-known criteria to be taken into account when selecting an electric thruster concept for a propulsion system aboard mainly nano or small spacecrafts.
Unfortunately, those criteria are sometime forgotten by some thruster designers or "people who do differently" who propose hardware that can be simply highly non-competitive with respect to other concepts.
A list of the known possible concepts of Electric Propulsion will be presented. The compatibility of those thruster concepts with an unified propulsion system (sharing the same propellant tank) is highlighted: this can come from the solid state of the propellant or as well from the electric conductibility of the propellant (see figure below).
The criteria to be taken into account are described in the paper. Among them one can cite: the number of thrusters used in the system, the unified propulsion concept, the worst-case design, the redundancy philosophy, the system mass impact and the possibility of sharing (partly and/or timely) the electrical power with the EP thrusters.
In order to allow an objective selection among the thruster concepts, it is proposed in the paper to rely on the definition of Issp (the System specific impulse) introduced by Peter Erichsen in the 90s.
Issp number takes into account all parameters influencing the impulse-dependent part of the propulsion system mass, such as the mass of the propellant storage and electric power supply systems, and therefore it characterises the performance of propulsion systems particularly for the size of the system considered for nano or small spacecrafts much better than the Isp "Thruster-specific Impulse" alone which only includes the propellant mass.
Here, Issp has been adapted for covering the cases of unified or non-unified propulsion system as well as the worst-case design concepts with or with no redundancy.
Several examples of output of selection will be presented for relevant cases in order to allow thruster designers or "business angels" to make their own best selections before taking any investment decision into a particular concept.

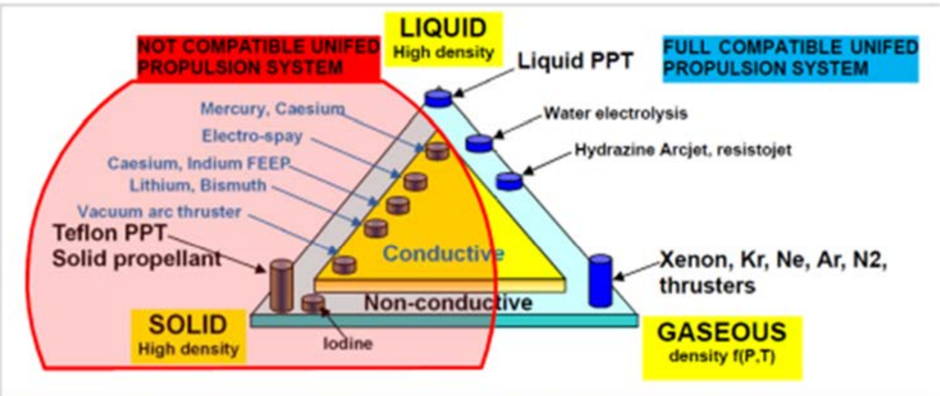

Figure 5: Electric propulsion propellant concepts with their phases, electric conductivity and compatibility with unified propulsion system concept
(a thruster concept not compatible of the unified propulsion system concept is defined as the impossibility for the thrusters to be fed by a common propellant tank)

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Abstract for RGCEP 2020

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Modeling of Cryopumps for EP Usage

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Telecommunication satellites and also planetary exploration spacecraft use more and more electric propulsion systems for maneuvering in space. The main advantage of these propulsion systems is their low fuel consumption and with that higher Δv capabilities.

Space systems and propulsion units have to fulfill extensive on-ground testing. As EP systems usually have low thrust and longer run times compared to chemical propulsion ground testing times are further extended. EP qualification has to be performed in dedicated vacuum chambers. The German Aerospace Center DLR operates the electric propulsion vacuum test facility STG-ET in Goettingen [1] [2]. EP testing requires powerful pumping systems for keeping pressure low enough. At STG-ET cryogenic pumps are used for these tasks, which use cryogenic pumps that freeze gases on cold surfaces inside the vacuum chamber (Figure 1). The detailed processes on cold pumping surfaces are very complex and depend on optical and mechanical properties of ice layers composed of several species. Furthermore, sputtering caused by exhaust plume ions modifies surface and deposits on a cold pump plate.

This paper presents some basics and numerical modeling of the cryogenic processes during cooldown, operation, and recycling of the pumps. This may help for optimizing the usage of cryopumps for EP facility operation.

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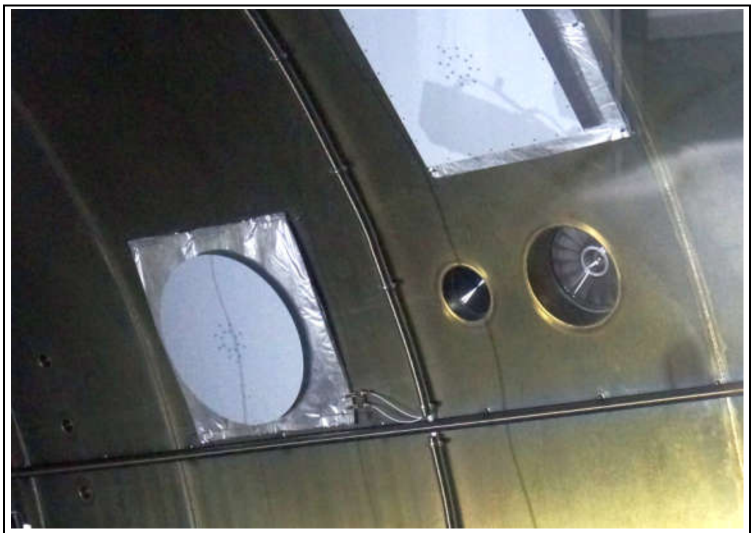

Figure 1: Cryopumps installed in the DLR Test Facility STG-ET

Optimization OF AN IODINE FED RF-NEUTRALIZER

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Key words: *Iodine , Rf-neutralizer.*

The halogen iodine might be a viable alternative propellant for both Hall thrusters [1] and gridded ion thrusters [2]. However, hollow cathodes may not be well suited for long times of operation with iodine due to its strong chemical reactivity at the typical operation temperatures of the emitting insert, which is often exceeding 1000 K. Instead of a hollow cathode, we develop a RF-neutralizer. The principle of the plasma generation in such neutralizers is essentially the same as in radio-frequency ion-thrusters. Other than the thruster, the neutralizer does not require a grid system for extraction, a simple orifice is sufficient for extracting electrons from the plasma. The emitted electron current is determined by the ion current reaching a negative biased metallic collector. A positive biased tungsten mesh may be used to mimic the ion beam of a thruster during neutralizer testing.

Measurement results of a first prototypical neutralizer were presented on the last IEPC-conference [3]. The performance results operating the rf-neutralizer with iodine were comparable to those with xenon, if sufficient small flows were applied. Generally, the neutralizer requires only extraordinary small flows of about $10\ \mu g\ s^{-1}$ at extraction currents up to $200\ mA$ operating in diffusion mode, yielding excellent gas utilization factors of up to 40. However, the extraction costs of about $300\ W\ A^{-1}$ were higher than optimal. One approach to enhance the power efficiency is to modify the neutralizer in such a way, that a plasma bridge is sustained between the collectors situated in the discharge and the anode downstream of the neutralizer`s orifice, similar to the operation of hollow cathodes. To realize this extraction mode, the neutralizer’s geometry will be modified. The investigations will include global modelling as well as further test activities, first in standalone tests of the neutralizer and consecutively coupled tests with an RIT.

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PROBE DEVELOPMENT FOR ELECTRIC PROPULSION DIAGNOSIS

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Key words: *Debye radius, Electric propulsion, hall engine, ion energy spectrum.*

Retarding potential analyzer is widely used to study the energy spectrum of electric propulsion plasma plumes. In a typical configuration the probe uses three grids with cell diameter equal to several Debye lengths for the plasma which surrounds the probe. The first grid is kept under floating potential and limits plasma interaction with internal grids. The second grid is kept under small negative potential for electrons repelling. A positive potential is applied to the third grid and only ions with sufficiently high energy can pass through it, making small positive current on the collector. As a result, the dependence of the collector current on the voltage on the third grid allows one to obtain the ion energy distribution function. Most probes require geometry adaptation to measure at a given density of plasma with the required accuracy. But when studying a plasma plume at different angles to the axis of the electric propulsion, the plasma density can differ by several orders of magnitude, which does not allow sufficiently accurate measurement of the required parameters.

This paper describes a new probe, which is capable of measuring the ion energy spectrum with less than 5% error when ion current density varies from 0.001 A/m^2 to 100 A/m^2. The probe uses two mutually aligned electrodes instead of grids and ion-retarding positive potential is applied directly to collector. Results of numerical modeling with various ion current densities for two hall engines KM-75 and KM-10 with different energy spectrums are presented. Also investigation of mutual electrode misalignment influence on probe work is described. The paper represents experimental verification of numerical modeling and comparison with a conventional three-grid probe. In the end, conclusion about applicability of new probe in ion energy spectrum investigation in the plume of various types of electric propulsion is made.

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ELECTROMAGNETIC COMPATIBILITY TEST FACILITY WITH REVERBERATION CHAMBER FOR ELECTRIC PROPULSION

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Key words: *EMC, reverberation chamber, radiated susceptibility, EMS, EMI*

Like all electronic devices, ion thrusters need to fulfill electromagnetic compatibility (EMC) requirements. As the use of those thrusters increases, Justus-Liebig-University in cooperation with University of Applied Sciences at Giessen (THM) started building an EMC test facility. Its first part will be a reverberation chamber (RC) made of a cylindrical vacuum vessel.

In Europe ECSS-E-ST-20-07 is applied which partly refers to other standards, meaning especially the American MIL-STD-461. The latter describes the use of a RC as alternative test procedure for radiated susceptibility (RS) measurements. Since thrusters for satellites can only operate in a space-like environment, the testing has to be performed inside a vacuum vessel.

One option is to use a metallic vacuum vessel and making it a RC. In such a configuration, the sidewalls of the vacuum vessel also serve as electromechanical shielding. Unlike anechoic chambers, which need absorber structures, a RC can be build using significantly less space. Furthermore, the generation of radiofrequency (RF) modes with any E-field strength, which are formed through multiple reflexions of the RF wave inside the RC, needs less power than generating equal fields in anechoic chambers. Beside these advantages, there are also several disadvantage. First, since its geometry determines the lowest frequency allowing a standing wave (mode) inside the chamber, there is a low cutoff frequency beneath the RC cannot be used. With a future length of about 2.6m, the RC will meet MIL-STD-461 over approximately 0.4 GHz. Second, multiple modes only allow for 4π integral emission and irradiation measurements.

We will present the progress in setting up this facility and show first results of simulations and experimental investigations.

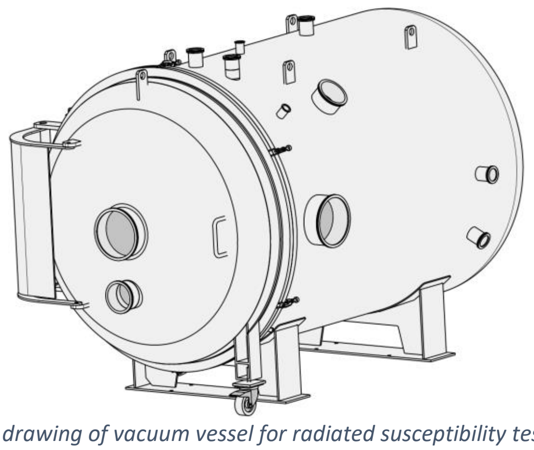

Figure 1: CAD drawing of vacuum vessel for radiated susceptibility test facility

EP Activities at IOM in Leipzig

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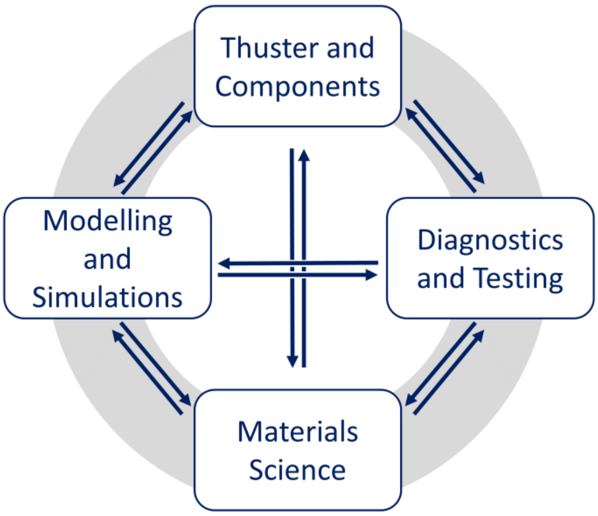
The development of modern EP technologies requires an advanced approach. It needs to combine several disciplines, namely sophisticated modelling and simulations, choosing and characterizing suitable materials, design and manufacturing of thruster and components, as well as development of diagnostic tools and thorough testing. All of these disciplines are strongly correlated.



For many years, the Leibniz Institute of Surface Engineering (IOM) is successfully following this approach in its electric propulsion activities that in addition profit from more than 40 years of experience in the development of broad beam ion sources.

In this contribution, present EP activities of the IOM are summarized. Selected examples of new or extended fields of work are:

- Thruster and components [1]: Development of an engineering model of a cathodeless RF plasma bridge neutralizer.
- Diagnostics and Testing [2,3]: Establishing laser based techniques for plume and discharge characterization, such as laser-induced fluorescence (LIF) or two-photon absorption laser induced fluorescence (TALIF), for characterizing the ion velocity distribution or neutral gas flow, respectively, including Kr as alternative propellant to Xe.
- Modelling and Simulations: Erosion modelling of gridded ion thrusters.
- Materials Science: Studying the erosion behaviour of EP-relevant materials upon interaction with alternative propellants.

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The study of the operational process specific features in very low-power plasma accelerators

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With reference of present and foreseeable market demand for small spacecraft propulsion systems EDB Fakel has initiated scientific & research works to create innovative very low-power PlaS-type plasma thrusters (VLPPT). The works' key point is a determination of minimum feasible dimension parameters of VLPPT's discharge and magnetic systems capable of maintaining a stable ionization and acceleration operational process along with an ultimate miniaturization of VLPPT systems' mass-dimension parameters. In order to accomplish the above said task EDB Fakel has developed two experimental laboratory models of very low-power plasma accelerators (VLPPA) based on two different miniaturized principal designs conventionally named PshiK-D and ZhuZha-D.

This paper presents general results of the research works conducted with VLPPA PshiK-D and ZhuZha-D aimed at an experimental search and optimization of brand new engineering solutions for PlaS-type VLPPTs which are developed at EDB Fakel. The results of the PshiK-D accelerator performance and lifetime research works are presented. Specific features of the ionization and acceleration operational process in a very small-dimension discharge chamber are defined. General directions of further activities on the PlaS-type VLPPTs development are specified.

Development of the low power SPT experimental models operating with Krypton and Xenon with increased design lifetime

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Key words: Stationary Plasma Thruster, development, experimental model, applications, propellant, performance, lifetime

Stationary Plasma Thrusters being one of the so-called Hall thrusters (HT) already many years successfully operate in space. They are regularly used for the near Earth and geostationary spacecraft final orbit transfer after their launching and for the further orbit corrections and their keeping as well as for realization of such missions as the SMART-1 spacecraft flight to the Moon. Last times their application is significantly extended. Particularly there is started the HT application onboard small satellites (SS) of the Starlink constellation with the planned number of SS over 10 000 and during 2 years already launched and operate over 1000 SS of this constellation[1]. And it was difficult to provide the required quantity of Xe typically used earlier in HT for the propulsion systems (PS) of such number of spacecrafts because Xe is very expensive substance and its production rate in the world is limited. Therefore the propulsion experts were searching for HT propellant alternative to Xe and the 1st new one practically used in HT is Kr representing as Xe an inert gas, having potential of ionization slightly exceeding that one of Xe. These Kr features ensures the possibility to get satisfactory HT performance, simplification of the problems of its storing and solution of the operating HT compatibility with the spacecraft structural elements. In the case of the Kr storing in the simple pressurized tanks increasing mass of tanks the "wet PS" delivery to the operation orbit could be cheaper than in Xe case due to one order of magnitude lower Kr cost, if the launch cost will be low enough. The last one could be significantly reduced using the launching of the SS packages consisting large number of satellites as the launching of the 60 Starlink SS per launch. Taking into account that creation of the multi satellites constellations is planned in Russia too and the great experience of Russian experts in the SPT development and application it is reasonable to develop the Russian low power SPT's operating with Kr able also to be competitive operating with Xe. One of the simple ways to solve this task is modification of the existing thrusters for operation with Kr and such modification was made at Fakel where the SPT-70M model was developed having satisfactory performance level operating with Xe and Kr [2]. But it is still not confirmed its claimed design lifetime 7000 hours. More over taking into account more intensive SPT structural element erosion rates of thruster operating with Kr under comparable operation modes the obtaining of the Kr SPT large lifetime becomes complicated. Therefore there was started development of the new low power SPT-50 and SPT-70 experimental models (EM) with operation power ranges 250-1000 W for the SPT-50 EM and 800-1350W for the SPT-70 EM using solutions promising increase of thruster lifetime. This development was made by RIAME

MAI as co-executor of the Bauman State Technological University within the frames of Applied Scientific Investigation funded by the Russian Ministry of Science and High Education where EDB Fakel was an Industrial Partner.

To obtain acceptable performance level the mentioned EM were developed for operation modes with the increased Kr mass flow densities till ~0,2 mg/s per 1cm^2 of the acceleration channel cross-section area in the interpole gap (and around 0,1 mg/s per 1cm^2 of the widened exit part of the acceleration channel) ensuring increased level of thrust efficiency [3]. Therefore the power density in these models is notably increased. To ensure the possibility to get large enough lifetime the SPT-50 EM and SPT-70 EM were designed with the widened acceleration channel exit part starting with the interpole gap and continuied beyond the magnetic poles plane, with the shifted magnetic layer with the maximum magnetic induction values positioned also beyond the mentioned pole plane [4]. These measures had allowed shift of the ionization and acceleration layer and erosion zones of the discharge channel walls beyond the magnetic pole plane and drastic reduction of the wall erosion rates (Figure 1) similar to obtained in thrusters with the “magnetic shielding” [5] and indicating the possibility to get increased lifetime of the developed engineering models.

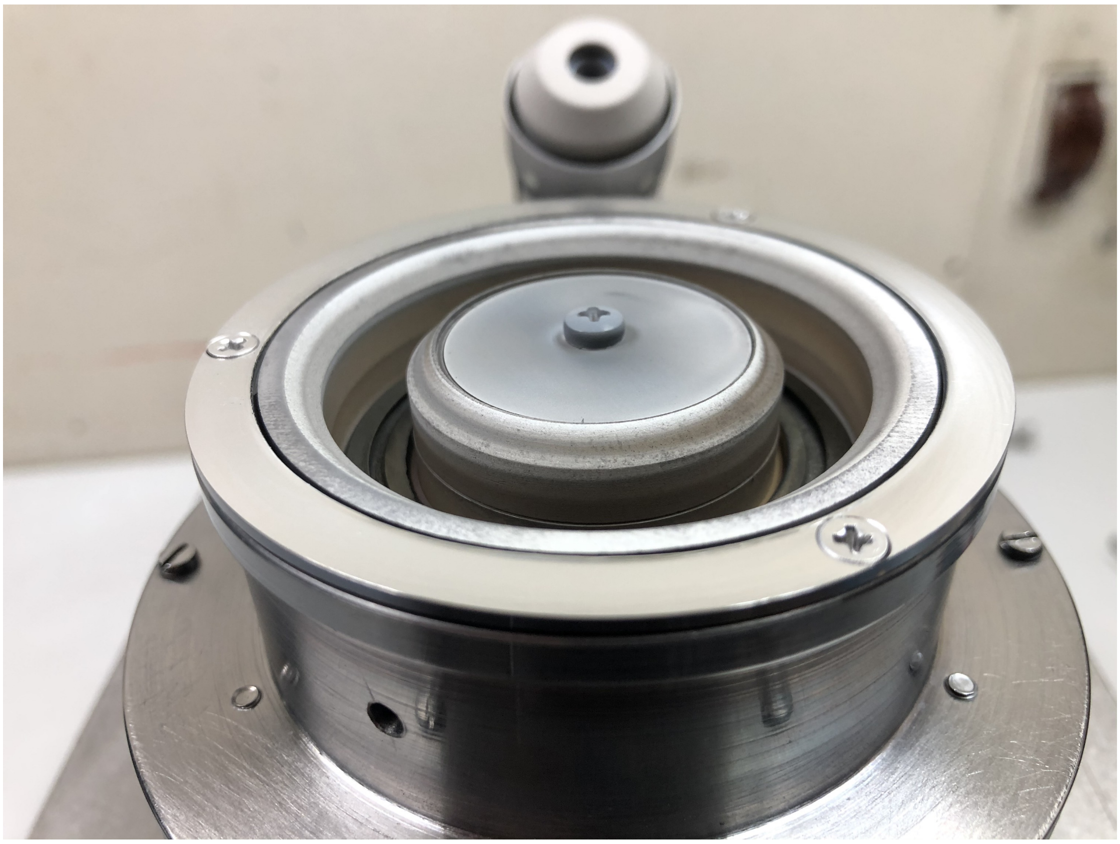

Figure 1- The SPT-50 EM after 20 hours of opertaion with Kr and power 1kW

Obtained output parameters of the SPT-50 EM operating with Xe and Kr are represented in the Table 1.

Table 1. Parameters of the SPT-50 EM operating with Xe and Kr for different operation modes

Operation mode	Discharge current, A with Xe/Kr	Discharge voltage, V	Thrust, мН with Xe/ Kr	Specific impulse, s with Xe/Kr
~250Вт	/1,0	250	/14,0	/1000
~600Вт	2,0/2,02	300	31,0/26,5	1550/1370
~800Вт	2,66/2,54	300	44,4/37,7	1664/1500
~ 1 кВт	3,33/3,31	300	55,7/48,5	1747/1660

Similar results were obtained for the SPT-70EM in the range of powers 800 - 1350 W. It is interesting that SPT -50 EM demonstrates increased specific impulses than existing thrusters SPT-50M and SPT-70M operating with Xe. As a whole obtained results confirm the possibility of the competitive lower power SPT development on base of the SPT-50 EM and SPT-70 EM for the small satellites motion control.

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SPECTRAL-TEMPORAL CHARACTERISTICS OF SELF EMISSION FROM TAL IN THE RADIO-FREQUENCY BAND FOR DIFFERENT PROPELLANTS

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Key words: *thruster with anode layer, electromagnetic compatibility, self emission.*

We shall consider procedure, hardware and software used for the experimental determination of spectral-temporal characteristics of self emission from the laboratory model of low-power thruster with anode layer (TAL) developed by the Bauman Moscow State Technical University. Measurements were made by RIAME MAI using vacuum facility equipped with a radio-transparent compartment and a shielded anechoic chamber for the discharge power of 600, 800 and 1000 W, vertical and horizontal polarizations and different propellants (krypton, xenon).

The conducted studies allowed us to identify a number of features of TAL radio emission in the spectral and temporal domains. Thus, in particular, it was noted that TAL has a broadband spectrum, recorded in the frequency band from 1 to 4 GHz. Vertical polarization dominated in the studied modes, and the level of TAL radio emission operating with krypton was 5-10 dB higher than in the case of its operation with xenon.

Mono-crystalline Lanthanum Hexaboride production peculiarities

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Emissive materials of rather different compositions and ways of their production have found a wide use in various fields of engineering – from consumer microwave devices to electron-beam welding cathodes in the industry. However, with regard to emissive materials for the most crucial products of the rocket and space technology with high-level requirements specified in terms of current density, work function and lifetime, Lanthanum Hexaboride LaB_6 prevails. LaB_6 thermoemitters have high emissive properties – a low work function, high density of emission current, high erosion resistance, emissive ability recovery after poisoning, etc.

The production of mono-crystalline Lanthanum Hexaboride may be divided into 2 stages: production of a poly-crystalline billet using the hot pressing method and production of a mono-crystal using the method of crucible-free zone melting.

In the process of elaboration of a mono-crystal production the phenomenon of a break of the feed crystal was detected, what is accompanied by a local explosion.

The paper objective is to investigate a cause of the disruption of a poly-crystalline billet melting process and exclude causes influencing the production of a mono-crystalline Lanthanum Hexaboride.

This paper addresses the main causes influencing the process of mono-crystalline Lanthanum Hexaboride production using the method of crucible-free zone melting. The structure and chemical composition of a billet from Lanthanum Hexaboride and a produced crystal have been investigated.

As a result, it was determined that a billet from poly-crystalline Lanthanum Hexaboride shall satisfy the requirements for the density, porosity and chemical composition.

THE NEW POSSIBILITIES TO INCREASE THE SPT-140D THRUSTER RELIABILITY

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Key words: stationary plasma thruster, thruster ignition, dual firing

An investigation of two simultaneously operating SPT-140D thrusters was carried out at EDB Fakel. The thrusters start capabilities both with a combined and independent power supply were investigated.

The thrusters were tested on the test stand KVU-165 at EDB Fakel facility. The SPT-140D thrusters were installed in the vacuum chamber in one plane at a distance of 700 mm between their centers. One unit was connected through the standard cable network. This unit was ignited following the nominal cyclogram. The 2nd unit was connected through a specifically developed cable network and was started in a manual mode. During single and dual firing, the pressure in the vacuum chamber was $2.2 \cdot 10^{-5}$ mbar and $4.4 \cdot 10^{-5}$ mbar (Xe), respectively. T

In total, several starts were performed with various cathode positions on one of the thrusters, as well as with a combined "minus" line of the power supplies, where one thruster firing from another thruster's cathode and two thrusters firing from one cathode was verified.

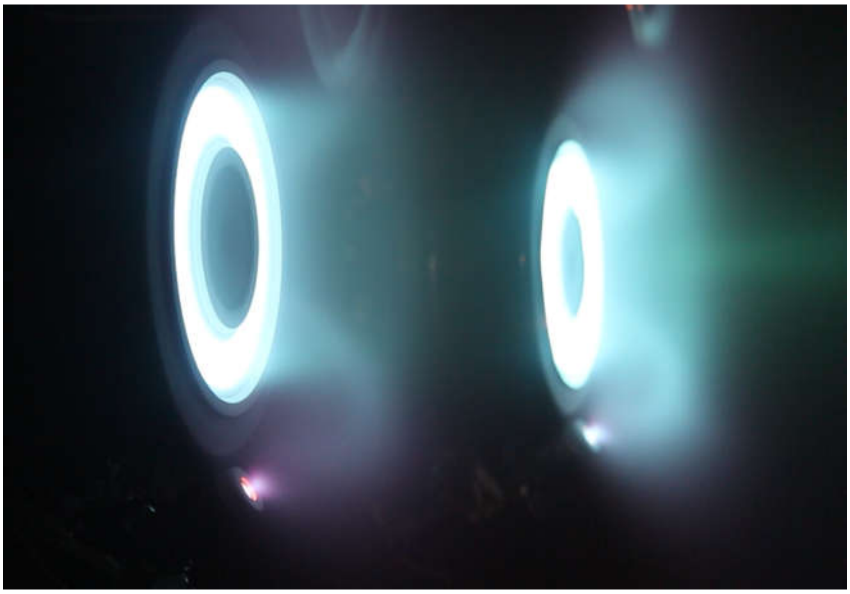

Figure 1 – Two SPT-140Ds in their simultaneous firing

All the starts were successful independently on the cathode position.

The starting parameters of each SPT-140D and start-up algorithm during a dual operation remained the same as for a single firing.

The possibility to start one thruster from another thruster's cathode was demonstrated. It was also proved that 2 simultaneously operating thrusters retain their operation which is usually observed when they are operated separately.

The possibility of a dual firing of 2 SPT-140Ds from one cathode and also the possibility of one thruster firing ignited from the neighboring thruster's cathode extends the capabilities of the thruster system and increases its reliability.

BENCH-TEST OF LOW POWER THRUSTER WITH ANODE LAYER ON KRYPTON

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Key words: plasma thrusters, thruster with anode layer, cathode neutralizer, power, thrust, specific impulse, krypton flow rate

Nowadays spacecrafts of different classes and missions are equipped with closed electron drift thrusters, for example a thruster with an anode layer and stationary plasma thrusters (SPT). These thrusters operating on xenon successfully perform station keeping and raise a spacecraft to the target earth orbit. At the same time, the emerging large constellations like OneWeb, StarLink, The Sphere [1, 2] resulted in sharp demand growth on EP systems creating the need to find a way to reduce its cost. One way to approach this problem is to use instead of xenon an alternative propellant that is cheaper; krypton is considered the most prospective. Considering that performances of both gases are close enough, one can use numerous design improvements already qualified on xenon. However, in order to ensure high efficiency and lifetime on krypton the design, in particular, the geometry of the accelerator channel [3] have to be optimized.

The thesis present result of the bench-test of the experimental model with anode layer manufactured in the Bauman Moscow State University for krypton. The unit was tested at the EDB Fakel's test facilities. The average diameter of the accelerator channel is over 13,5 mm and 6 mm respectively. The geometry of the acceleration channel had been optimized earlier for krypton. The thruster was tested together with flight cathode-neutralizers manufactured by the EDB Fakel. While testing the volt-ampere characteristics have been measured, as well as the thrust level within the range of 1,5-3,5 mg/s and discharge voltage 200-400 V. The thruster was firing at 0,4 kW up to 2 kW with thrust was within the range from 17 mN up to 85 mN. The testing revealed that the efficiency increases along with the krypton flow rate. At discharge voltage of 300V and 1,5 mg/s flow rate, the thrust efficiency was 28,5%; the increased flow rate 3,5 mg/s elevated the thrust efficiency up to 85 mN. The thrust efficiency in its turn reached 50% and the specific impulse exceeded 2400s.

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SPECTRAL-TEMPORAL CHARACTERISTICS OF SELF EMISSION FROM SPT-70 IN THE RADIO-FREQUENCY BAND FOR DIFFERENT PROPELLANTS

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Key words: *stationary plasma thruster, electromagnetic compatibility, self emission.*

We shall consider procedure, hardware and software used for the experimental determination of spectral-temporal characteristics of self emission from the laboratory model of stationary plasma thruster SPT-70 developed by RIAME MAI. Measurements were made using vacuum facility equipped with a radio-transparent compartment and a shielded anechoic chamber for different values of discharge power (600, 800 and 1000 W), vertical and horizontal polarizations and different propellants (krypton, xenon).

The conducted studies allowed us to identify a number of features of SPT-70 radio-frequency emission in the spectral and temporal domains. So, it is shown that SPT-70 has a broadband spectrum, recorded in the frequency band from 1 to 3 GHz. Horizontal polarization slightly dominated in the studied modes. When compared to the SPT operation with xenon, the maximum emission level excess in krypton operation was about 10 dB in the frequency range from 1 to 2 GHz for horizontal polarization.

Development and research of parameters and characteristics of modified SPT-70M operates on xenon and krypton

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EDB Fakel has manufactured and tested a second engineering model of SPT-70M (here and after – EM2) as part of the on-going modification of SPT-70 thruster which was used as part of the module TM-70 on such spacecraft as Plasma-A, Yamal-100 and it is operating on such spacecraft as KazSat-1, KazSat-2, Express-MD1, Express-MD2, Yamal-200. The development of EM2 was carried out taking into account the results of the previous study[5] of the lab and the first engineering models of SPT-70M

The second engineering model of SPT-70M has successfully passed a set of tests. The article shows the results of the parametric firing tests and thermovacuum tests of EM2, which was performed at EDB Fakel, and it shows the environmental tests' (mechanical test) result too. We studied parameters and characteristics of the thruster while powered by two propellant: xenon and krypton.

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Abstract of the article «Problems of measuring propellant flow rate on the test stands and their solution»

Rumyantsev A.V., Guskov K.V., Chubov P.N., Sinitsin A.P.

Discharge current, discharge power, thrust, specific impulse, reserve of a propellant are related to a propellant flow rate in a thruster, therefore a reliable definition of a propellant flow rate in a thruster is one of the most important tasks. The main limitations of flowmeters on the test stands are the following:

1) they are located outside a vacuum chamber;
2) they have no thermo-compensating system;

Therefore, testing one and the same thruster on different test stands leads to different results: thrust differences are in a range of (-1.1 %) ÷ (1.9 %), total flow rate differences are in a range of (– 4.4%) ÷ (15%). At measuring low cathode flow rates the differences are in a range of ±17%. A statistical processing of the XFC units test results~~,~~ showed that the differences make~~s~~ a range of (–10%) ÷ (8.3%). The error ranges of thrusters parameters definitions, averaged for 193 thrusters for F_T, G and I_{sp}, are equal to: (-3.5÷2.8) %; (-5.0÷5.0) %; and (-5.0÷7.5) %, respectively. The relation between I_{sp} and F_T, I_{sp} and G has a qualitatively correct character, but the dispersion in values is very large. A connection between F_T and G actually misses - it is described by a logarithmic function with a practically zero degree of reliability. The normalized density function of thrust $f(F_T)$ has a symmetric character, but f(G) is explicitly non-symmetrical with a noticeably expressed left-side asymmetry. For distribution of $f(I_{sp})$ a strongly pronounced right-hand asymmetry is characteristic, a maximum is significantly shifted towards lower values. All this testifies that the flow rate is measured on the test stands with an overestimation, and this leads to underestimated values of I_{sp}~ 1/G; i.e., a sufficient flow rate data reproducibility on the test stands is not achieved. Therefore, it is necessary to develop flowmeters for different flow rate ranges that could solve the problems of flow rate measuring on the test stands.

With this purpose a thermal flowmeter of a variable power based on CT1-18 thermistors is developed for measuring a flow rate and for thermal mode thermal stabilization at set temperature levels of gas flow T_f = 284÷304 K with environment temperatures 268÷323 K. [1,2]. The use of CT1-18 provided greater values of output signals and a high sensitivity that allowed to refuse from complex electronic schemes and place a flowmeter in the vacuum chamber before the XFC.

The data on the study of the flowmeter parameters with a diaphragm with an aperture of 1 mm is presented in the table. The maximum output signal is 95.5 V, the sensitivity is 27.27 V/(mg/s). None of the known flowmeters has such parameters.

G, mg/s	**1.66**	**2.32**	**3.31**	**4.31**	**5.30**	**6.62**	**8.28**	**9.94**	**13.25**	**16.56**
U(1), V	47.4	61.6	70.3	79.6	83.7	87.3	90.0	91.9	94.1	**95.5**
dU(1)/dG	**27.27**	17.58	9.70	6.36	4.48	4.15	1.75			

For measurement of a gas flow rate at environment temperature ~ (450÷500) K an infra-red flowmeter was designed. The chamber represents a cylinder with a spiral placed in it. The radiation of the spiral is displayed through 4 light guides located under 45° angle relative to the axis of the spiral. For exception of heating of spiral radiation receivers – photodiodes – curved copper tubes with a length of 150 mm, filled with quartz light guides with a diameter of 1.0 mm, are connected to the channels with light guides. At the other end of ~~a~~ the tube ФД-21 K type of photodiode is inserted. Diaphragms with openings of 0.5 or 1.0 mm can be inserted into the inlet and outlet channels of the gas flow. The data on the study of the output signal of the diaphragmed (d = 1 mm) flowmeter as a function of flow rate is shown in the table. The sensitivity in a flow rate range of up to 20 mg/s varies in the range of (4.5÷1.0) V/(mg/s).

G,mg/s	**0**	**2.16**	**4.33**	**6.49**	**8.66**	**10.8**	**13.0**	**15.2**	**17.3**	**19.5**	**21.4**
U(G), V	23.0	18.27	13.34	9.18	6.34	5.04	4.07	3.43	2.87	2.48	2.21

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Abstract of the article «Problems of measuring propellant flow rate on the test stands and their solution»

TECHNICAL SESSION C

MAGNETOPLASMODYNAMIC THRUSTERS

Assessment of Optimal Operational Regimes of Steady State Applied Field MPD Thrusters at IRS

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Future interplanetary cargo and manned missions can be performed with chemical propulsion systems at the expense of significant amounts of required propellant. This usually needs to be launched into the orbit. and leads in the long term to higher transfer costs. High power electric propulsion systems in the several 100 kWe and MWe range could enable interplanetary cargo orbit transfers at the expense of increased complexity by offering in a long term a very cost efficient and sustainable transportation via SEP/NEP space tugs and ISRU considerations.
Steady state applied-field magnetoplasmadynamic thrusters (AF-MPDT) feature relatively high specific impulse and high thrust densities. They are considered of being suitable for high power interplanetary applications in the power range of some MW. However, due to the relatively low TRL level, complexity, facility requirements and limited research the thrust efficiencies of steady state AF-MPD thrusters are typically between 30 and 55 %.[1-2] In the past 10 years at the Institute of Space Systems (IRS) a significant effort was made on experimental realization of high power AF-MPD thrusters, leading to very promising results, including continuous operation of well confined plume and stable operation at 0.4 T and 100 kW of arc power.[3] Several measurements have been performed, including tare forces, AF-coil characterization and thrust measurements with thrust efficiencies higher than 40 %.[3-4]
This paper presents an overview referring to the achievements at IRS in a summarizing way. To a significant extend this has been previously reported in a focused way e.g. in reference 6. These achievements concern:

- The operation at modes of significantly high voltages and hence mitigating the life time problem of the cathode. In addition, it shifts the duplet of current and voltage rather into a regime that is known and established for HET leading to the fact that PPU and hollow cathode systems as they are established for HETs can now be beneficial for AF-MPD thruster.
- The achievement of significant thrust efficiencies and this, in addition, accompanied by a significant throttability of one and the same thruster e.g. by variation of power, mass flow rates, magnetic flux, etc.
- The derivation of sensitivities related to the extent of electrode (anode) losses. Here, operational scenarios have been identified that mitigate the level of electrode losses while simultaneously creating a benefit for the jet power.

In all the data obtained ran through extensive verification and the used balance has been seignificantly improved in terms of design as well as in terms of thrust measurement processing.
Additionally, a larger hollow cathode with 24 mm diameter was tested, allowing this way a decrease of electrode ratio from 7.167 to 3.583, and variation to discharge and performance characteristics with main goal of optimized operation in range of 100 V. Moreover, for high voltage operation with anode voltage up to 450 V, a new LaB_6 hollow cathode has been designed and tested with SX3 anode up to discharge power of 15 kW in regulated voltage regime.[5] Finally, a conclusion on further development and perspective evaluation is given with respect to short and long term application of AF-MPD thrusters a main propulsion system.

A THRUST BALANCE FOR THE MINOTOR ECR THRUSTER

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Key words: *MINOTOR, Thrust-Balance, Thrust-Measurement.*

The MINOTOR project (MagnetIc NOzzle elecTron cylcOtron Resonance thruster), funded by the European Union in frame of H2020, has the objective to demonstrate the feasibility of the ECRA (electron cyclotron resonance acceleration) technology being disruptive for electric propulsion. The ECRA technology is based on the ionization of a propellant gas by resonant microwave heating of the plasma's electrons and subsequent acceleration of the plasma by a magnetic nozzle. In framework of this project the University of Giessen optimized a thrust balance in the thrust range from 0.1 to 120 mN.

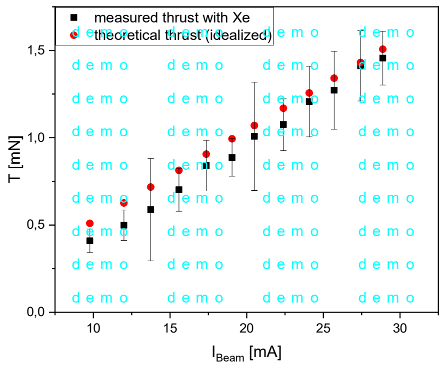

Fig.1: Left) ECRA thruster placed on the thrust balance and running at JLU's JUMBO facility. Right) Comparison of the measured thrust of the RIT 4 and the theoretical thrust

Thrust balances are needed in order to perform accurate thrust measurements of electric engines. Most thrust balance designss are typically based on a hanging, an inverse or a torsional pendulum [1]. The conventional hanging pendulum is the simplest configuration, but also is the least sensitive to external perturbations. The resolution increases with the length of the pendulum arm, so this setup is only useful for large test facilities and is primarily used for a high thrust to weight ratio, T / W. The thrust measurement with a torsional pendulum is independent of the mass of the thruster. This should yield a higher resolution than achievable with the other two configurations. However, due to the horizontal arrangement, larger thrusters will require stronger mounting structures in this configuration which in turn lowers the sensitivity again. Therefore, these pendulums are typically used to measure very small thrusts. Inverted pendulums are less stable, but more sensitive than hanging pendulums [2]. Thus, they allow thrust measurements over a wide range of T/W.

To characterize electrical thrusters with thrusts of a few milli-Newton, we propose modifications of a double armed inverted pendulum thrust balance. The balance concept formerly developed for German aerospace center (DLR) has been optimized by JLU in consultation with DLR Goettingen to achieve sub-mN resolution by adding an interferometer for optical measurements of the balance displacement, voice coil actuators driven by a PID-controller for compensating the thrust as well as a passive eddy current break for sensitivity adaptation. Furthermore, an absolute in-situ calibration procedure for precise thrust determination has been added.

The thrust balance's function is tested by determining the thrust of different thruster types e.g. of a simple cold gas thruster, a μHEMP thruster, a RIT 4, and an ECR-thruster. The results are compared with those measured with other thrust balances and a force probe in the thruster beam.

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INFLUENCE OF DISCHARGE CHANNEL PARAMETERS ON THE APPT PERFORMANCE

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The electrodynamic efficiency of APPT is largely governed by the ratio of the variable inductance to the initial one [1]. Thus, it is possible to vary the inductance of the discharge circuit, for example, by acting on the law of change of the variable component of the inductance depending on the movement of plasma blob along the discharge channel. For this purpose, the discharge channel is made expanding to the output, thereby increasing the inductance per unit length. Besides, in order to increase the inductance per unit length of the discharge channel, the electrodes behind the exit plane of propellant bars are made to have a triangular shape in plan and are inclined at a certain angle relative to the longitudinal axis of the thruster.

Additional increase in the inductance per unit length of the discharge channel can be obtained if the initial part of the discharge channel is made expanding also. In this case, the shape of the cross-section of propellant bars must correspond to the shape of the modified electrodes. For the sake of clarity of experiments comparison, the dimensions of bars of the new shape have been calculated in such a way that the working surface area has not changed.

The results of the experiments showed that the specific impulse increased by ~30% with the same discharge energy in the case of a trapezoidal discharge channel. That is, the proposed method of increasing the ratio of variable inductance to the constant one is a promising way to improve the thruster performance. The RF patent No. 2688049 was obtained for the APPT with the discharge channel of trapezoidal type.

The work has been fulfilled within the frames of implementation of the Federal Target Program "Research and Development in the priority fields of the science and technology sector growth in Russia for 2014-2020" (Agreement No. 05.607.21.0308. The unique identifier of agreement is: RFMEFI60719X0308.

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THE PROBLEM OF OPTIMAL DISCHARGE ENERGY IN ABLATIVE PULSED PLASMA THRUSTER

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At present, due to the progression of small spacecraft weighing up to 500 kg, there is a need for high-performance low-thrust electric propulsion systems (EPS) that would combine low mass and power consumption with a long lifetime and high total thrust pulse. One of the prospective trends is the development of EPS based on the ablative pulsed plasma thrusters (APPT).

For the time being, the RIAME MAI has developed a number of EPS based on APPT with the discharge energy from 8 J to 155 J [1, 2]. It is shown in [2] that the most promising way to improve the weight and size characteristics of APPT is to reduce the weight of the energy storage unit by optimizing its energy content and using power capacitors with increased specific capacitance. However, the question of the optimal discharge energy of APPT designed for various purposes remained open [3].

It is shown that if we use the total EPS mass at a given total pulse as the criterion of optimality, the APPT discharge energy will have an optimum depending on the specific capacitance of power capacitors, specific impulse of the thruster and the mass of the electronic units and other structural elements of the propulsion system. When calculating the optimal discharge energy, it is also necessary to take into account the lifetime limitations of electronic components in the number of discharge pulses.

It is concluded that while designing the APPT-based EPS the calculation of the optimal discharge energy allows significant reduction for the total mass of the propulsion system.

The works have been fulfilled within the frames of implementation of the Federal Targeted Program "Research and Development in the priority fields of the science and technology sector growth in Russia for 2014-2020" (Agreement No. 05.607.21.0308. The unique identifier of Agreement RFMEFI60719X0308.

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TECHNICAL SESSION D

POWER SOURCES AND THRUSTER-SPACECRAFT CONJUNCTION

MODERN TRENDS IN SPACECRAFT POWER AND PROPULSION SYSTEMS AND ELECTRIC THRUSTERS

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Key words: *Power and Propulsion System, Electric Propulsion, Thruster.*

Development tendencies of space power and propulsion systems (PPS) and their characteristics are considered for different types of spacecrafts (SC). PPS functions as well as PPS key components and main technologies are analyzed relating to different class of SC (or SC constellations) operated at different near-Earth orbits, high power level space tugs for Moon, Mars and deep space missions [1-8].

General design and architecture features of perspective PPS are described for different types of spacecrafts. Actual application area and future progress of modern electric propulsion (EP) based on different types of thrusters and issues of EP integration into SC power and propulsion systems are discussed [2, 3, 5].

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STUDIES OF JOINT OPERATION OF ELECTRIC PROPULSION AND POWER SUPPLY SYSTEM BASED ON CLOSED BRAYTON CYCLE

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Key words: *Power Supply System, Closed Brayton Cycle, Electric Propulsion.*

One of the actual trends in a development of spacecraft with electric propulsion is a significant increase of available onboard power and thrust level. Projects aimed on development and application of space vehicles based on powerful power and propulsion systems (PPS) with a power level from hundreds of kilowatts to tens of megawatts for near-Earth and deep space missions have been developed since the very beginning of space exploration Era [1, 2, 3, 4, 5]. To provide such power levels different types of spacecraft power systems are considered, including nuclear power supply system based on dynamic conversion of thermal energy into electricity in a closed Brayton cycle [2, 3].

During joint operation, electric propulsion and power supply system can have a mutual influence on each other's work. Insufficient consideration of this influence may result in unstable PPS operation. In this regard, it seems advisable to take an approach, which involves carrying out a sufficient amount of preliminary integration “cold” tests using functional simulators both electric propulsion and power system during experimental studies. Experimental data obtained during these tests, including possible mutual influence of components, allow to optimize control’s logic, procedure and sequence of the joint tests of the electric propulsion with power system, and to minimize technical risk of an incident or emergency. It is especially important when carrying out the experimental studies of high power level nuclear powered propulsion systems consisting of very complicated and expensive components, such as nuclear reactor.

Such a tests sequence was already implemented at the Keldysh Research Centre during experimental studies of joint operation of the electric propulsion and their power supply system based on the dynamic conversion of thermal power into electricity in the closed Brayton cycle with using gas resistance heater as a thermal simulator of a nuclear reactor. In the course of these experimental studies, for the first time in domestic practice, the stable joint operation of the key components of powerful space PPS – electric propulsion and power supply system based on closed Brayton cycle - was demonstrated both at stationary and typical transitional modes.

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TECHNICAL SESSION E

SPACECRAFT WITH ELECTRIC PROPULSIONS. FLIGHT DYNAMICS, PROPULSION CONTROL

APPROXIMATE METHOD FOR CALCULATING INTERORBITAL FLIGHTS OF A SPACECRAFT WITH LOW-THRUST ENGINES

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Ключевые слова: space transportation system, combined propulsion system, chemical upper stage, electrorocket transport module, interorbital flights

Currently, the most common orbit for placing communications satellites is geostationary (GEO). In most cases, traditional pulse schemes are used for transferring spacecraft (SC) from the reference orbit to the GEO, which involve several turns of the high-thrust main engine. However, in recent decades, researchers have increasingly shown interest in alternative flight plans for geostationary orbit, including those involving the use of a combination of high-and low-thrust engines on a spacecraft. The reason for this was the successful use of electric rocket engines (ERE) at the stages of re-entry into geostationary orbit of the spacecraft "AEHF-1", "Express-AM5", "SES-9".

A vast array of ballistic diagrams of flight of a SC on geostationary orbit using the ERE, there are three main:

1) a transient elliptical orbit is formed Using a launch vehicle (LV), with the radius of the apogee of the orbit being equal to, greater than, or less than, the radius of the geostationary orbit. The function of the upper stage SC performs its own electric propulsion system;

2) a low circular orbit is formed using the LV, then a transition orbit is implemented using the chemical upper stage (CUS). In the future, delivery is carried out by its own ERE, similar to option 1;

3) in the third variant, the LV forms the initial orbit, then a transition orbit is formed using the CUS. At the last stage of the flight, a reusable electric rocket transport module (ERTM) with a large energy capacity is activated, which makes a return flight to the initial orbit.

Spacecraft with electric rocket engines can have a very large length of active sections, so the analysis of the active section of the flight of vehicles with ERE is carried out on the basis of a complete mathematical model in the osculating elements, where the acceleration of the SC from the working ERE is considered as disturbing. When modeling the motion of a SC with an ERE, the usual system of differential equations in osculating elements is preferable, since it is possible to form the law of control of the thrust vector more effectively due to clear and simple equations of motion with traditional orbital elements. To avoid singularities, fixed final values of eccentricity and inclination that are different from zero and correspond to the required accuracy of the problem solution are set before integration. In addition to the jet acceleration created by the ERE, perturbations caused by the non-centrality of the Earth's gravitational field and perturbations created by the earth's upper atmosphere are considered as perturbing factors. The second zonal harmonic for the potential of the forces of attraction is used as the main one.

We formulate the problem of optimal change of the main elements of the orbit. As these elements, we take the large half-axis, inclination, and eccentricity of the orbit, and as an optimality criterion, the duration of the flight.

The solution of the variational problem can be carried out in accordance with the principle of maximum Pontryagin, provided that the ERTM works without shutting down, since in this case, the minimum total duration of the flight is provided.

Applying the maximum principle reduces the optimization problem to a boundary value problem for a system of ordinary differential equations. In the process of solving the boundary value problem, it is necessary to find the initial values of the conjugate variables. Finding initial values is usually performed using a modified Newton method [1], and it is necessary to set the initial approximation, which can be found based on the physical meaning of these values, or based on the results of solving simplified problems. The method used allows you to find the optimal solution, but it is associated with great computational difficulties and requires high qualification of the researcher.

In accordance with the principle of reciprocity in the theory of optimal control of a variational problem on the minimum duration of the dynamic maneuver with fixed boundary conditions is equivalent to the minimization of the generalized residuals by the variance of the terminal value component of the state vector at a fixed duration maneuver [3].

We introduce a terminal criterion in the form of a quadratic functional, which is the sum of the squares of residuals over the large half-axis, eccentricity, and inclination of the orbit, multiplied by their corresponding constant weight (undefined) coefficients.

The resulting control law has a fairly simple structure and allows you to calculate the dynamic maneuver without the procedure for solving the boundary value problem.

By selecting the values of the weight coefficients, it is possible to achieve simultaneous fulfillment of the final conditions for the large half-axis, eccentricity, and inclination of the orbit.

In order to verify the proposed approximate optimal calculation method, the simulation results were compared with the results of exact problem solving. At the first stage, the method was tested using the example of "circle-circle" flights. The results given in V. N. Lebedev's monograph [1] were used as a reference. The solution using a locally optimal control program gives slightly different values of the flight time relative to the results obtained using the maximum principle method. The difference is 1.2-1.6%. Deviations can be caused by varying degrees of accuracy of the boundary conditions of the problem.

At the second stage, the comparison was carried out on the example of flights of the "ellipse-circle" type. The source for comparison is the results given in V. G. Petukhov's monograph [2]. The final values of eccentricity and inclination for all cases are assumed to be zero. Analysis of the results showed that the difference in flight time between the solutions obtained using the proposed method and the exact results is quite small and is about 0.02-1%.

Below are the results of calculating the life of a heavy geostationary communications satellite equipped with an electric rocket propulsion system. The transition elliptical orbit is formed by the 3rd stage of the launch vehicle.

To determine the optimal parameters of the orbit in terms of the minimum flight time, we analyze the parameters of the orbit with a search of the apogee height on the segment ra = [40000; 80000] for i = 28º and ra = [70000; 110000] for i = 51.6 º. In the calculations, we assume the mass of the satellite for all transition orbits to be constant (m0=const). At the next stage of research, it is planned to conduct a refined analysis taking into account the dependence of the output mass of the SC on the apogee radius of the transition orbit.

The results of searching the heights of the apogee of the orbit for two inclinations revealed the optimal values. In the case of inclination i = 28º optimum apogee altitude is 60,000 km, i = 51.6 º this value increases up to 93000 km. Such a large difference in this parameter for different inclinations due to the fact that a significant part of the energy of flights to geostationary orbit is the change of inclination of the orbit. As a result of the flight, not the entire stock of the working body was spent. As a result, for the case with i = 28°, the remainder is 332.89 kg, and for i = 51.6°, it is 246.44 kg. The surplus of the working body can be spent on extending the active life of the SC, as well as for its transfer to the burial orbit.

Also, as a result of calculating two flights, the weight coefficients were selected according to the rule of simultaneous time when the elements of the orbit reached their final values.

Some difference in the time when the elements of the orbit reach the required value is due to the inaccuracy of the selection of weight coefficients. To reduce the difference, iterative selection of weight coefficients can be performed.

During the flight simulation of the SC, taking into account the impact of the Earth's atmosphere, it was found that its impact is insignificant, since the SC leaves the atmosphere after a few turns and is in it for a short period of time.

The simulation showed that the derivative of the functional does not change its sign, and the functional itself monotonically decreases to zero, minimizing residuals for the large half-axis, eccentricity, and inclination of the orbit.

The NEOS program was developed to automate the calculation of ballistic parameters for the flight of spacecraft with an electric rocket propulsion system between circular and elliptical non-planar orbits»

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MULTI-CRITERIA OPTIMIZATION OF INTERPLANETARY ELECTRIC PROPULSION MISSIONS

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Keywords: *interplanetary mission, electric propulsion, optimization criteria.*

Promising space projects are constrained by the high cost of putting spacecraft into orbit. This is especially true for missions to explore planets, asteroids, and interplanetary space. In [1, 2], a study of promising space projects that are "waiting" for their turn to be implemented while reducing the cost of putting payloads into working orbits. Figure 1 shows the expected trend of the annual traffic to a high geocentric orbit increasing and the removal cost decreasing based on the data from the work [2].

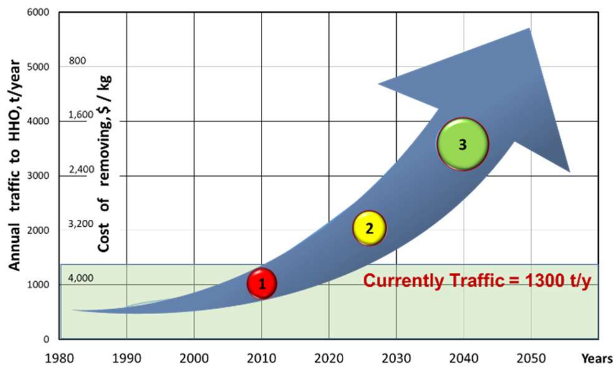

Fig. 1. Trend of the annual traffic and the removing cost

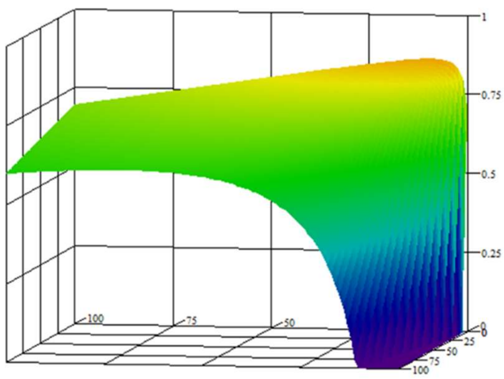

Fig. 2. The relative payload mass depending on the design parameters of the propulsion system

At a launch cost of 4,000 – 40,000 $ per kg projects related to telecommunication, navigation, broadcasting, remote sensing, scientific research, and other low-orbit near-Earth space programs are profitable. Modern cosmonautics reaches the first stage.

By reducing the launch cost to 2,800 $ per kg the second stage of space technologies development starts. The orbital business parks (production of materials, biotechnology), the large low-orbit groupings for global Internet access and Earth monitoring, Moon and Mars exploration and colonization, space control segment unmanned aerial vehicles become profitable.

When the launch cost become less than 1000 $ per kg, the third stage of development of space technologies will come. At this stage projects related to the orbital power generation to Earth and space objects, the resource extraction (from Moon and asteroids), the utilization of Earth and space debris, the space tourism (orbital flights, Lunar and Mars adventures) and other new markets are profitable.

Multi-criteria mission optimization is required to total mission efficiency increasing. The criteria are the specific payload mass, the removal and operation durations, the specific cost of payload putting into the working orbit, and the probability of successful mission completion. Numerous works on design and ballistic mission optimization using electric propulsion have considered these aspects separately and in various combinations but only few works are devoted to the complex problem.

We find it interesting to consider the relationship of these criteria with the design and ballistic mission parameters in some test cases: the spacecraft operation in the low Earth orbit; the spacecraft launching and maintaining to the GEO; the transport operations from the low-Earth orbit to the low-Moon orbit; the interplanetary flight Earth-Mars-Earth; the interplanetary flight Earth - asteroid belt.

For example, Figure 2 shows the relative payload mass calculated to the Earth-Mars flight to the different flight durations and the different propulsion systems [3]. Figure 3 presents the petal chart on the unidimensional criteria field of their 20 considered ballistic-design variants for Earth-Mars- Ceres mission [3]. Figure 4 presents the trajectory to the best ballistic-design variant.

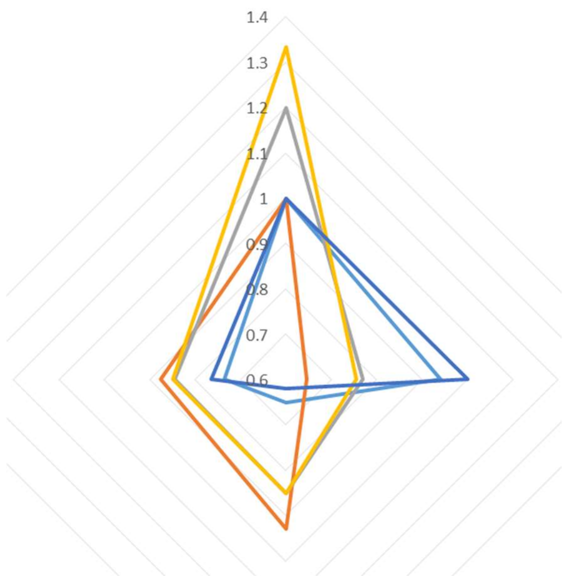

Fig. 3. The petal diagram on the criteria field (payload mass, mission duration, cost, and success probability) to the top five of Earth-Mars-Ceres

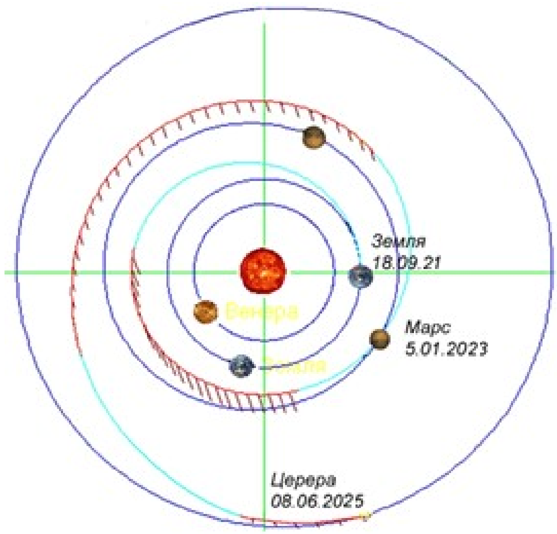

Fig. 4. The Earth-Mars-Ceres mission trajectory to the best ballistic-design variant

The developed methodology and software allow you to analyze the interplanetary missions and choose the best design and ballistic options to them.

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TRANSFER TO 433 EROS COMPARISON OF SPACECRAFT WITH NON-PERFECTLY REFLECTING SOLAR SAIL AND LOW THRUST ENGINE

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Keywords: *spacecraft, low thrust engine, propulsion system, solar sail, near-Earth object, near-Earth asteroid, motion simulation.*

This paper outlines the nonlinear flight simulation of a low thrust spacecraft and a solar sail spacecraft. The main purpose of the work is ballistic comparison of solar sail spacecraft flight and low thrust spacecraft flight. The mathematical motion model for such types spacecraft is formulated and described. The work considers the mathematical motion models within the heliocentric system of coordinates. Motion simulation process is the best way to assess the reasonableness of the mathematical model. On the basis of the formulated mathematical models the special software for nonlinear flight simulation is developed. Especially, the flight of the low thrust spacecraft from Earth's orbit to the potentially hazardous asteroid is simulated. The findings of the work consist of heliocentric trajectories, time dependency graphs of flight parameters, the values of flight duration and the value of propellant consumption.

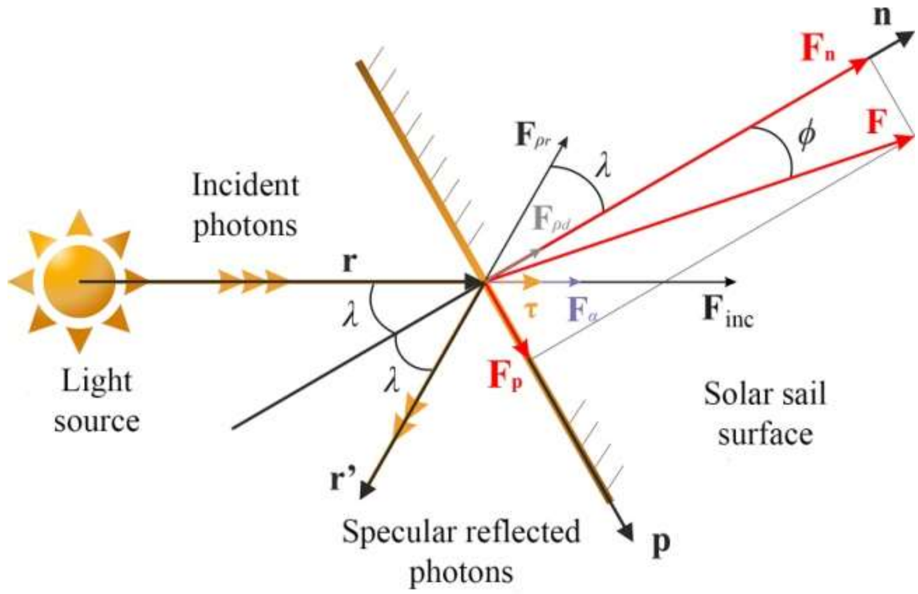
Research into interplanetary travel and circumsolar space provides an opportunity to answer many fundamental scientific questions and to use the achievements of astronautics for exploiting the virtually unlimited resources of the Solar system. Scientific research in space requires considerable expenditure and does not ensure fast returns [1, 2].

Nowadays scientists and society pay a huge attention to celestial bodies, such as asteroids. Asteroid research affords an opportunity to modern science to understand how the Solar system was formed. A promising way of cost reduction for such missions is the use of advanced physical principles of space travel such as movement by means of a solar sail. The possibility of saving a plenty of money has generated wide interest in the solar sailing technology. In recent years a considerable amount of work has been done in solar sailing. In the past five years great experience of the use of advanced physical principles of space travel such as movement by means of a solar sail has been obtained and described in [3]. The motion of a spacecraft on the Low Earth Orbit (LEO) is observed in [4]. Much less information is available about a flight of solar sail spacecraft to potentially hazardous asteroids and comparison with low thrust spacecraft flight. This paper considers solar sail spacecraft and low thrust spacecraft motion simulation. The aim of the present work is ballistic comparison of two different flights.

Interplanetary transfer is a flight between two planets that needs in performing of parabolic acceleration planetocentric maneuvers. Different interplanetary flight aims demand different sets of boundary conditions. Interplanetary transfer can be divided into three stages:

1. Earth escape.
2. Heliocentric flight from Earth orbit to an asteroid orbit.
3. Planetocentric flight.

Model of non-perfectly reflecting solar sail under study is presented in Fig. 1.



n is the solar sail normal, **r** is the radius vector, **p** is the vector in the solar sail plane directed to the **r**, **r'** is the vector directed along the motion of the reflected photons, **F** is the vector of full thrust force, $\mathbf{F_n}$ is the vector of thrust force directed along the **n** normal vector, $\mathbf{F_p}$ is the vector of thrust force directed along the **p** vector, $\mathbf{F_{inc}}$ is the vector of thrust force from incident photons, $\mathbf{F_\rho}$ is the vector of thrust force from specular reflected photons, $\mathbf{F_\alpha}$ is the vector of thrust force from absorbed photons, λ is the angle between the **r** radius vector and the **n** normal, ϕ is the angle of deviation of the **F** full thrust vector from the **n** normal

Fig. 1. Diagram demonstrating distribution of full thrust vector components with the solar sail under study

Low thrust spacecraft uses propulsion system and propellant to perform an interplanetary flight. Fig. 2 describes a low thrust spacecraft motion model.

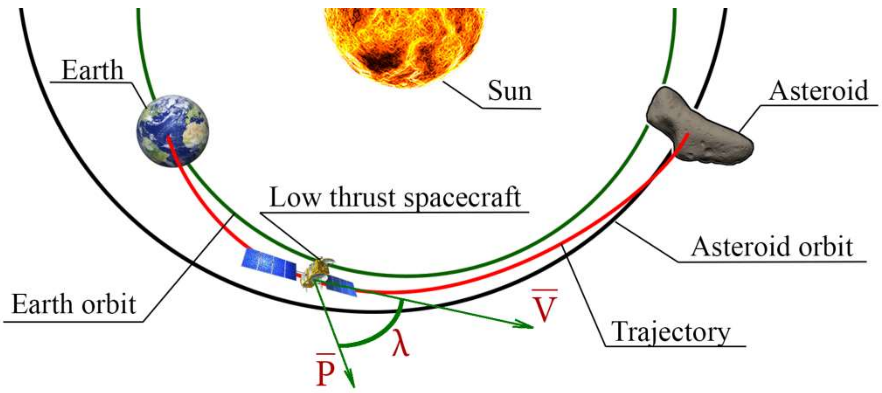


Fig 2. Low thrust spacecraft motion model

In comparison with low thrust spacecraft motion, the trajectory of a solar sail spacecraft has a multiturn form of motion. It takes 4 full revolutions around Sun to ascent to 433 Eros orbit. In contrast, low thrust spacecraft performs a half revolution around Sun to complete the mission. One more distinction between two trajectories is low thrust spacecraft motion was divided into two active parts and one passive part. During flight solar sail spacecraft had only active motion.

Flight duration of two missions is different too. It takes 2291 days and 159 days to enter the asteroid orbit for solar sail spacecraft and low thrust spacecraft respectively. The acceleration of low thrust spacecraft is greater than acceleration of a solar sail spacecraft, that is why spacecraft with propulsion system spends not much time to flight from Earth orbit to asteroid orbit.

The final difference between solar sail spacecraft and low thrust spacecraft motions is propellant consumption. In comparison with motion by means of solar sailing technology, it takes 376.87 kg propellant for low thrust spacecraft to ascent to asteroid orbit. There is no propellant consumption in solar sail motion.

In summary, a low thrust spacecraft is better for interplanetary flight in shortest time. But, if there are not any time constraints, solar sail spacecraft is the best way for interplanetary transfer.

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Spacecraft transfers between L_1 and L_2 libration points in the Earth-Moon system for a number of electric propulsion thrusters

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The spacecraft transfers ballistic optimization for the L_1-L_2 Earth-Moon missions was considered for the number of electric propulsions. The motion simulation was carried with the attraction of the Earth, the Moon and the Sun. The motion control was applied using the successful linearization method to estimate the derivatives and the gradient method to optimize the control laws. Electric thrusters with various thrust and specific impulse were analyzed. The optimal control programs for the L2-L1 and L1-L2 transfers and corresponding trajectories were obtained.

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DESIGN AND BALLISTIC ANALYSIS OF SPACECRAFT MISSIONS WITH AN ELECTRIC PROPULSION SYSTEM TO AN ASTEROID

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Keywords: *Nano-class spacecraft, electric propulsion, mathematical model, trajectory of motion, software package, ballistic characteristics, asteroid.*

In this paper, we consider objects with a non-spherical gravity field and small gravitational pull, for example, asteroids, planet's satellites, and comets. The authors consider motion control in the vicinity of an object with an irregular gravitational field. The mathematical model of the system takes into account the attraction of the Earth, the Sun, and the small body. The authors of the article developed a software package for modeling and visualization of trajectories and control programs that are used to support spacecraft near an asteroid, The paper analyzes the optimal transition between non-planar orbits of a nanosatellite with low-thrust engines. The acceleration from the engine thrust is constant in magnitude and directed along the binormal or transversal. Since the spacecraft has a single-engine (taking into account the characteristics of the nano-class spacecraft), the thrust can be directed either by the binormal or by the transversal [1].

The aim of work is developing a method for control spacecraft with low-thrust engines.

The specific impulse of the engine module is IsP1 = 1000 sec, IsP2 = 1500 sec and IsP3 = 2000 sec. The trust values are various for a particular altitude. The particular values were obtained by the direct search method [4].

The asteroid's gravitational field has a complex shape. When approaching the body, the disturbance generated by the irregularity of the shape has a strong effect. For missions planned to analyze the asteroid's surface or stay in orbit around the body, or even land on it, studying the body's gravitational field is important. For this reason, the authors developed a barycentric model of the gravitational potential-a model of single gravitational points for modeling the movement of the spacecraft. Knowing the deviation in orbital velocity due to the perturbation of the gravitational field, you can apply the necessary corrections to obtain the desired orbit [2].

The authors consider the issue of controlling the movement of a nanosatellite near objects with a non-spherical gravitational field. This publication examines the model for controlling the position of the spacecraft (fig.2). Spacecraft can maintain a constant position relative to the asteroid [3]. Thus, the proposed exploration mission will determine the current movement of the asteroid and develop a method for accurately predicting the trajectory of the asteroid in question. The mission allows us to solve the problem of determining the kinematic, physical and structural properties of a specific and at the same time typical asteroid. Figure 3 shows the layout of the bodies.

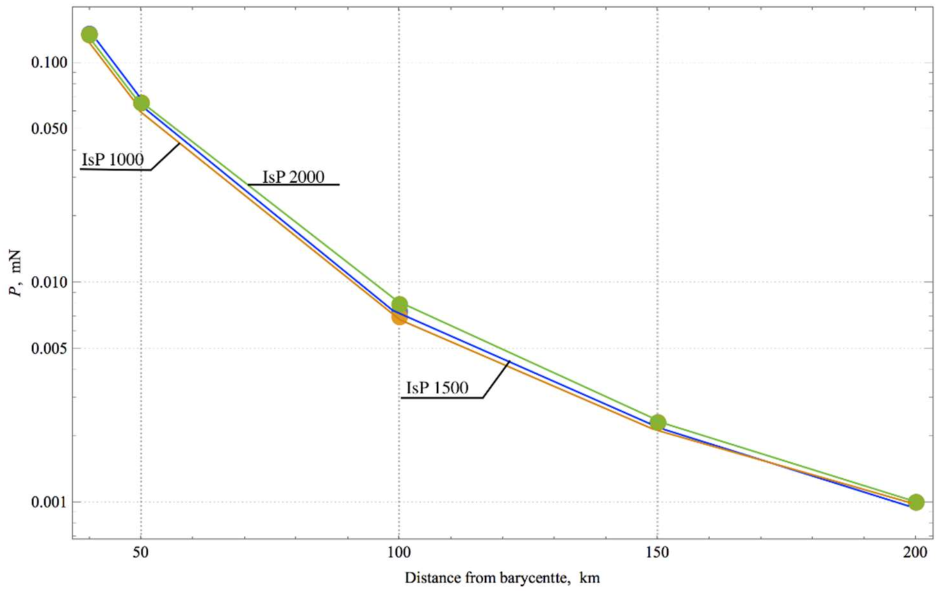

Figure 1 - Dependency of the thrust level to the barycentre distance.

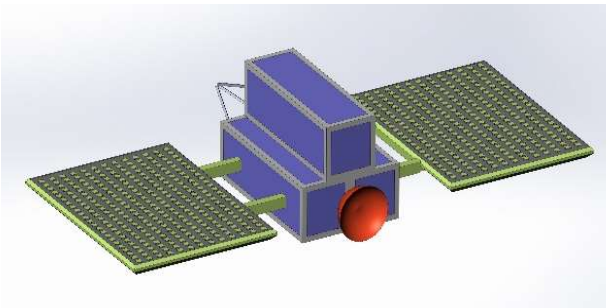

Figure 2 - Design appearance of the spacecraft

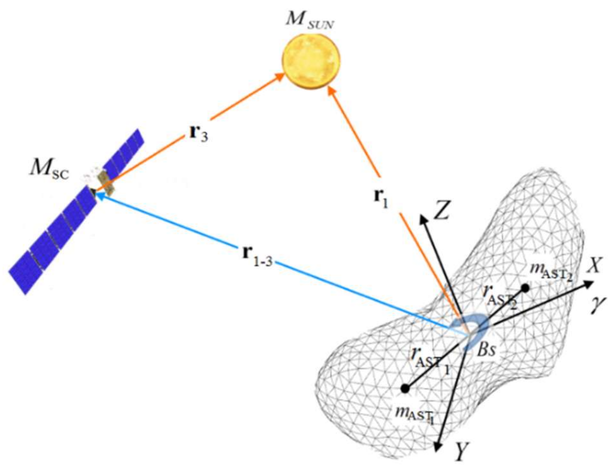

Figure 3 - Location of bodies in the system

This article discusses the use of low-thrust engines to stabilize near-asteroid orbits. Mathematical models of the asteroid gravitational field and controlled motion of a spacecraft with an electric motor in an uneven gravitational field are developed.

The obtained results are used to simulate the passive and controlled motion of a spacecraft as an n-body problem. It is established that passive motion is unstable. The authors proposed to use the known locally optimal control law for orbit stabilization. The levels of thrust and specific impulse and their dependence on the orbit height, initial velocity, and asteroid parameters were determined.

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SIMULATION OF SPACECRAFT MOTION TO THE EARTH QUASI-SATELLITE (469219) 2016 HO_3 USING LOW-THRUST PROPULSION

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Keywords: *quasi-satellite, electric propulsion, near-Earth asteroid.*

A quasi-satellite of a planet is an object that is in an orbital resonance of 1:1 with the planet. This allows the object to remain near the planet for many orbital periods. One of such objects is the asteroid (469219) 2016 HO_3, which is quasi-satellite of the Earth. Period of its revolution around the Sun coincides with the period of revolution of the Earth. It moves around the Earth along the complex path, its projection onto the ecliptic plane is shown in figure 1 [2].

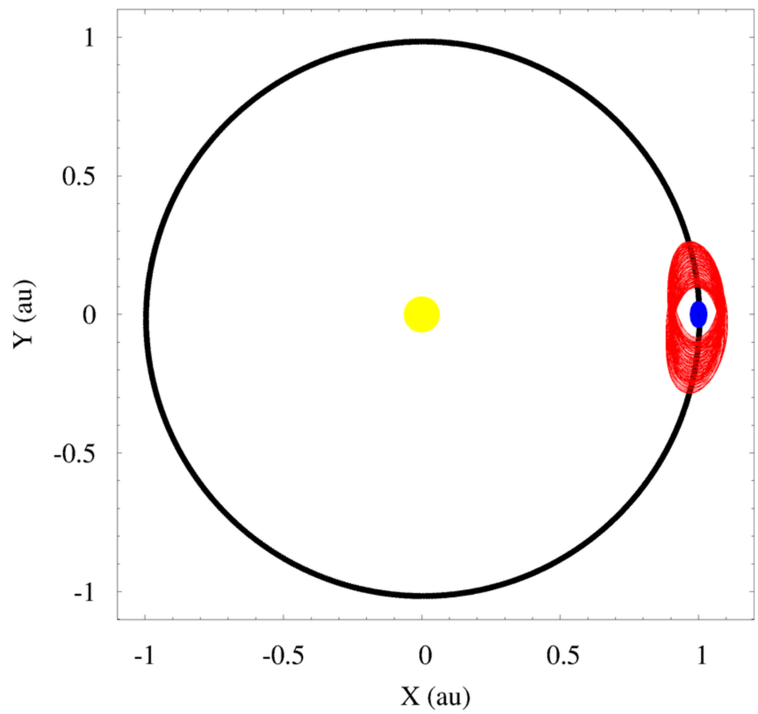

Fig 1. Projection of asteroid 2016 HO_3 orbit onto the ecliptic plane

Such objects. The distance from the asteroid 2016 HO_3 to the Earth is about 100 times the distance from the Earth to the Moon. In mid-2020, the asteroid the orbital flight phase, convenient for transfer mission. In this paper, we consider the flight from near-Earth orbit to this asteroid. Figure 2 shows the orbit of asteroid 2016 HO_3 in geocentric inertial reference frame from April 1, 2020 to April 1, 2021 and the trajectory of the first stage of flight to the asteroid. The asteroid trajectory is constructed using the JPS Horizons system.

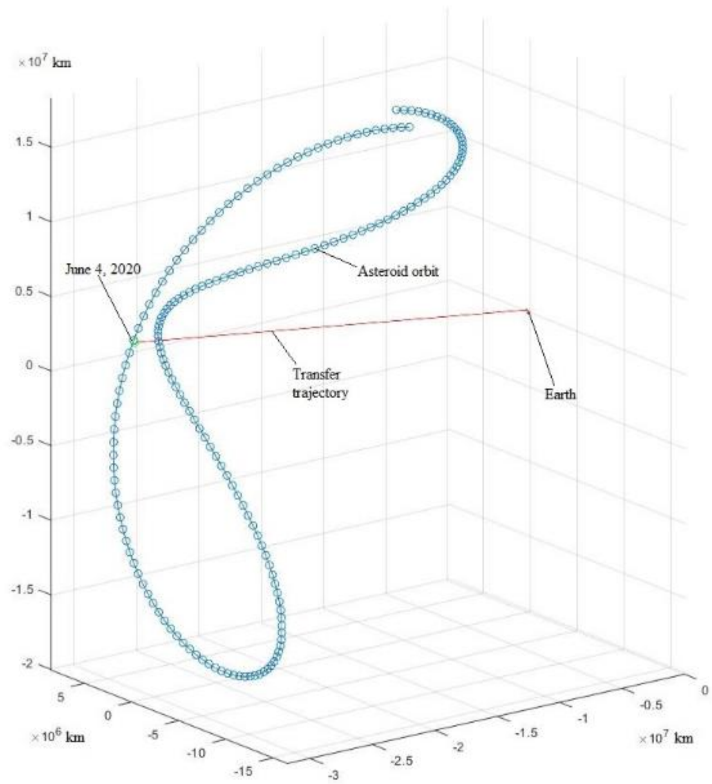

Fig 2. Transfer trajectory to the asteroid 2016 HO_3

The developed software allows to build a transfer trajectory taking into account external disturbances, select the optimal parameters of the parking orbit and the launch date.

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NARROWING THE AREA OF DEVIATION OF THE FINAL TRAJECTORY PARAMETERS USING THE ALGORITHM FOR REFINING THE THRUST OF ELECTRIC PROPULSION

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Keywords: *electric propulsion, systematic error, guidance, algorithm, interorbital flight, geostationary orbit.*

At present, an urgent problem is to increase the efficiency of transport operations for placing payloads into geostationary orbit.

One possible solution to this problem is the use of low-thrust electric rocket engines, whose high flow rate of the working fluid provides a significantly lower flow rate of the working fluid compared to a chemical fuel engine. However, low-thrust flights are quite long [1].

During the flight, various disturbing forces and moments act on the spacecraft, which will lead to significant deviations of the actual flight path from the nominal one. Therefore, it is necessary to periodically adjust the motion control program in order to ensure the specified accuracy of launching to the target orbit.

In General, the solution of the management problem is built in two stages:

1) creating an algorithm that allows you to translate the deviation vector of the final state into a certain region G, in which one or more components of the deviation vector of the final state satisfy the specified accuracy (for example, the deviation by inclination);
2) formation of control laws and algorithms for narrowing the area G to the area where all components of the final state deviation vector satisfy the specified accuracy.

This paper considers the solution of the management problem at the first stage.

The results of calculations show that during an interorbital flight using an electric propulsion system, the effects of various kinds of disturbances on the spacecraft, primarily due to inaccuracy in the implementation of the thrust value, lead to significant deviations of the final trajectory parameters. The experience of using electric jet engines in space shows that the deviation of the thrust value (systematic error) relative to the nominal value (form) is from-2.5% to +4.5% [2].

Since the spacecraft is affected by perturbations (perturbing accelerations, perturbations of the jet acceleration vector) that differ from the accepted a priori model, the mathematical model of motion should be periodically adjusted to the measured errors of the phase coordinate vector. Since at present sufficiently accurate models of perturbations of the Earth's gravitational field and lunar-solar perturbations have been developed, the amount of reactive acceleration is subject to refinement. The specified parameters are the amount of thrust of the electric propulsion system.

Passive sections are introduced on the flight path of a spacecraft with an electric propulsion system. On passive sites, the actual period of circulation is measured, on the basis of which the value of the thrust of the electric propulsion system is specified. After verifying the magnitude of the thrust is adjusted by the control program.

Thus, we have developed an algorithm for narrowing the area of deviation of the final trajectory parameters when flying to a geostationary orbit using low-thrust engines, which includes an algorithm for generating nominal programs for controlling the thrust vector and an algorithm for specifying the amount of thrust created by an electric propulsion system, based on measurements of the actual period of circulation.

The results of modeling the trajectory of a spacecraft with an electric propulsion system during a flight to a geostationary orbit showed that the algorithm for refining the thrust value and correcting the control program allows you to narrow the area of deviation of the final trajectory parameters, reducing the errors along the large half-axis by an order of magnitude and transferring the spacecraft to a near equatorial orbit.

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TECHNICAL SESSION F

ELECTRIC PROPULSION ON THE SMALL, MICRO- AND NANO- SATELLITES

Three-dimensional microlithography as an enabling technology for miniaturised electrospray thrusters

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Key words: *micropropulsion, miniaturisation, MEMS technology, electrospray*

The increasing number of very small satellites deployed or being prepared or considered for future deployment, such as picosatellites or CubeSats [1], has brought the subject of micropropulsion into the focus of development activities. Since conventional manufacturing technologies are approaching their limitations, novel technologies, especially the ones derived from microelectromechanical systems (MEMS) that in turn have evolved from semiconductor manufacturing technologies, have to be considered.

Any operating principle of a thruster that is worthy of consideration for micropropulsion has to scale favourably upon miniaturisation. One example of such a technology is the electrospray thruster [2], which uses an electric potential difference to accelerate droplets and/or ions from an ionic liquid from a cusp, the Taylor cone, forming at the liquid surface in a strong electric field. The smaller the geometric dimensions, the smaller the voltage needed for the formation of the Taylor cone. There is, however, one aspect of electrospray emitters that is problematic in miniaturisation: the hydraulic resistance the ionic liquid is experiencing between reservoir (tank) and emitter orifice must not be too low in order to achieve stable operation.

Initially, both internally wetted (capillary type) and externally wetted emitters were considered as technology candidates for electrospray thrusters, as well as a third type, employing porous materials shaped into needle emitters by mechanical micromachining. The porous needle emitters dominate in current developments, and porous emitters are also the leading technology with flight heritage as of today, while the development of internally wetted emitters is almost stalling due not least to the problem of insufficient fluidic resistance.

We propose to use the technology of three-dimensional lithography or two-photon lithography for the microfabrication of electrospray emitters and report on feasibility and design studies applying this technology to the three-dimensional patterning of the photostructurable polymer SU-8. SU-8 emitters had already been successfully manufactured in planar technology, and the electrical characterisation results of the emission properties of such emitters, among others by time-of-flight measurements, appear promising [3]. The increased freedom of design offered by 3D microlithography compared to classical planar (2D) photolithography are just beginning to be explored for the purpose of microemitter fabrication. We report on the state of the art and on recent progress in the fabrication of all-photopolymer electrospray emitters by 3D microlithography.

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DESIGNING A TRANSPORT MODULE WITH LOW-THRUST ELECTRIC PROPULSION ENGINES

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Key words: *space transport system, combined propulsion system, chemical thruster, electric propulsion transport module.*

Development of space transport modules, capable of carrying out a wide array of tasks of payload delivery to distant orbits, is one of the main trends in the field of space systems design. Analysis of effectiveness of combining of engines of high and low thrust for delivering spacecraft of various types into orbit is one of the prominent research trends. Implementation of electric propulsion transport modules (EPTM) allows to increase payloads and reliability of the orbit injection. This claim is supported by success of such missions as “AEHF-1”, “Artemis”, “Express-AM5” and others.

This paper considers the problem of orbit injection of “Meridian”, “Arctica-M”, “Glonass-K1”, and “Glonass-K2” spacecraft types with a “Soyuz-2” class rocket equipped with “Volga” upper stage and a transport module, fitted with an electric propulsion engine.

A hybrid orbit transfer scheme uses a chemical thruster to form an intermediary orbit on the stage of the orbit injection and an electric propulsion engine on the second stage to reach the target orbit. Any orbit, that is sufficiently different from the intermediary orbit in terms of semi-major axis, inclination and eccentricity could be considered as a target orbit. For example, this scheme is utilized for delivery of “Express” - type satellites into the geostationary orbit.

Parameters of initial and target orbits, payload mass characteristics, and transfer time serve as initial data for the solution of the design problem. It is assumed that the rocket carrier delivers the payload into an orbit with approximately 200 km altitude. Chemical boosters of the “Volga” upper stage facilitate the orbit transfer of the EPTM into an intermediary elliptical orbit. The final orbit injection is carried out with electric engines of the EPTM.

A problem of unification of EPTM parameters is formulated for a class of payloads in order to achieve minimal degree of suboptimality when compared to the “ideal” case, where every mission is fitted with a unique EPTM specifically designed from the ground up. The idea of a universal space transport system, equipped with a hybrid propulsion system, is presented in [1], where a method for determining its design parameters is given on the basis of simplified models of orbital movement and mass composition of the space transport system. The works [2] and [3] present an effective method of determining relative flight velocity with electric propulsion engines for various parameters of initial and target orbits.

EPTM mass is comprised from masses of structure, propulsion system, power supply system, working fluid storage system, and working fluid itself (xenon). The maximal orbital transfer time is used as one of the constraints.

The method for optimization of ballistic transfer parameters is given in [2]. It is shown, that in order to deliver payloads, corresponding to the spacecraft types, stated above, and 200x3500 km intermediate orbit, based on the iterative optimization procedure results, the EPTM should have the following design characteristics:

1) dry weight of the EPTM – 530 kg;
2) weight of the working fluid – 460 kg;
3) the propulsion system includes four SPD-140 electric engines;
4) overall power generation capacity (the solar array) – 15 kW;
5) solar array surface area – 63.3 m^2;

An EPTM with the aforementioned parameters would be able to form elliptical orbits with perigee from 500 to 25000 km and apogee up to 47000 km in time from 110 to 180 days, as well as circular orbits with altitudes up to 20000 km.

A design appearance of a universal EPTM is generated in the form of a digital model, based on the characteristics, obtained on the previous stage.

This digital model solves the following problems:

1) fitting of the EPTM under a nose fairing 4.11 m in diameter and 11.43 m long;
2) trying out different arrangements of the propulsion system;
3) determining internal layout of the system;
4) determining internal cable layout;
5) coupling of the EPTM with the “Volga” upper stage using a connecting adapter;

The payload connecting adapter (PCA) serves to establish structural and functional coupling of the payload and the EPTM. The “Glonass-K1” satellite is mounted on the PCA with the separation device. The PCA is a structure, consisting of a load-bearing frame and skin. It facilitates the transition from the hexagonal EPTM shape to the cylindrical frame that holds the payload separation device.

EPTM consists of two compartments: instrument and propulsion.

Instrument compartment contains instruments and antennas of various subsystems. Structurally, the instrument compartment is a hexagonal frame with honeycomb panels.

Propulsion compartment contains combined propulsion system, power supply system, and solar arrays of the transport module. It provides thrust and electric power to the transport module.

Structurally, the propulsion compartment is similar to the instrument compartment. The internal surfaces of the compartment are used to mount equipment and the external surfaces support the solar array.

The majority of propulsion compartment is occupied by the integrated propulsion system. It consists of the main electric propulsion system and the auxiliary propulsion system. The main electric propulsion system functionally serves as the primary means of thrust generation. The auxiliary electric propulsion system serves to create thrust, affecting the EPTM center of gravity, or providing attitude correction thrust impulses.

The electric propulsion system serves to inject the satellite into the target orbit and to remove the EPTM from the orbit after its mission has been accomplished. It includes four SPD-140 engines and the working fluid storage system.

The working fluid storage system serves to store and deliver the working fluid into the electric engines. It includes six storage tanks, piping, valves, heat-exchange units, and pressure reducers, in order to supply the electric engines with xenon gas in necessary state (temperature, pressure, flow).

The solar arrays serve to transform electromagnetic energy of the Sun into electric energy. It serves as the primary power source during the operation EPTM, including the initial orbit injection.

The thermal control system serves to establish and maintain permissible thermal conditions onboard of the EPTM during the flight. The thermal control system is designed in the passive configuration. Local heaters serve to provide heat to specific areas of the system. Heat pipes and heatsinks serve to remove and dissipate the unwanted heat into the outer space through dedicated radiation panels.

Screen vacuum thermal insulation minimizes heat exchange of the system with the outer space.

PTC Creo parametric was used to develop the design of the EPTM.

PTC Creo parametric is a family or suite of Computer-aided design (CAD) apps supporting product design for discrete manufacturers and is developed by PTC. The suite consists of apps, each delivering a distinct set of capabilities for a user role within product development.

Creo runs on Microsoft Windows and provides apps for 3D CAD parametric feature solid modeling, 3D direct modeling, 2D orthographic views, Finite Element Analysis and simulation, schematic design, technical illustrations, and viewing and visualization.

A model of the payload section of the nose fairing 4.11 m in diameter and 11.43 m long was developed with the payload model in order to test the sizing of the EPTM. This modeling has shown that the EPTM with the “Glonass-K” and “Volga” upper stage can be fitted into the payload area of the rocket-carrier.

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ANALYSIS OF THE EFFECTIVE USE OF THE SPD-50 ENGINE TO SUPPORT THE LOW ORBIT OF THE AIST-2 SMALL SPACECRAFT

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***Key words**: electric jet engine, small spacecraft, low orbit, aerodynamic drag*

Research has been conducted on the use of an electric jet engine to maintain the low orbit of a small earth observation spacecraft of the «AIST-2» class. From this series of satellites, the AIST-2D spacecraft is currently functioning, which was launched into orbit in 2016 with the following parameters: perigee height - 493.9 km; apogee height - 477.2 km; inclination - 94.2 deg [1]. This satellite does not have a jet propulsion system. The experience of operating the AIST-2D satellite shows that during the three years of operation, the period of rotation of the working orbit has decreased by 10 seconds (a decrease in the average altitude of about 10 km). This change in the parameters of the working orbit is not critical for observing the Earth with the required quality indicators. The satellite continues to function actively. It should be taken into account that the mission of the AIST-2D satellite is carried out during a period of low solar activity, in which the aerodynamic drag forces acting on the satellite are significantly less than in high solar activity.

In these studies, the space of the spacecraft's design parameters was used: the mass from 500 kg to 700 kg; the midsection area for oriented flight is about 2 m^2.

In the analysis, it was confirmed that the thrust force of the SPE-50 electric jet engine (Russian Federation, OKB Fakel) exceeds more than five times the aerodynamic drag force acting on the spacecraft under various atmospheric conditions in orbits with altitudes up to 400 km.

The analysis of possible orbit correction cyclograms over a long time interval was made. The cyclogram consists of correction cycles, which include sections of passive and active movement of the spacecraft. In passive areas, the electric jet engine does not burn, and the satellite carries out its main purpose with the help of the equipment installed on it. The passive section lasts until the deviation of the time of rotation around the Earth reaches the allowed value. The passive section is followed by the active section, where the electric jet engine turns on and creates a corrective acceleration. The active section continues until the parameters of the working orbit are restored. During the planned period of active existence, the satellite performs many correction cycles. [2]

The analysis showed that for circumcircular orbits with altitudes in the range from 400 km to 500 km, the active correction area for the allowed deviation of the orbital period by 3 seconds (equivalent to a deviation of 2.5 km for the average height of the circumcircular orbit) is no more than one day (0.4 – 0.7 days) for various atmospheric conditions. The passive part of the movement can last from 5 days to 220 days, depending on the altitude of the orbit and the current solar activity. For example, for low solar activity, in a circumcircular orbit with a height of about 450 km, the time of passive movement before a deviation of 3 seconds can range from 90 days to 120 days.

Based on the results obtained, the fuel mass was calculated for correction. The estimation of fuel reserves for the entire life of the satellite was carried out. The analysis showed that with different solar activity, the fuel consumption for maintaining the working orbit of a small SPACECRAFT with a height in the range of 400 km to 500 km for 5 years will be from 5 kg to 10 kg.

As a result, the conducted research shows the relevance of using an electric jet engine to correct the low orbit of the AIST-2 spacecraft. In orbits with an average altitude of about 450 km and above at low solar activity, maintaining an orbit for 3 years may not be necessary in an era of low solar activity (unless precision maintenance of the original orbit is required). In orbits with altitudes below 450 km, maintaining the orbit is necessary to ensure the long-term existence of the satellite (5 or more years).

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ANALYSIS OF THE EFFECT OF ROTATION OF SMALL SPACECRAFT SOLAR PANELS ON LOW ORBIT CORRECTION USING AN ELECTRIC JET ENGINE

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Key words*: electric jet engine, small spacecraft, low orbit, aerodynamic drag.*

he object of the study is a small spacecraft of the AIST-2 class. From this series of satellites, AIST-2D is currently functioning, which was launched into orbit in 2016 with the following parameters: perigee height - 493.9 km; apogee height - 477.2 km; inclination - 94.2 deg. The mass of the satellite is 531 kg, the dimensions of the body were about 1m × 1m × 2 m. the Area of solar cells is about 6 m2. There is no jet propulsion system for correction.

In the process of designing new satellites of the specified class, based on Board electrojet dvigatelei other equipment (for example, additional optical-electronic observation equipment) you may need to increase electric capacity of the power plant. Increasing electrical power can be achieved by increasing the area of solar cells and changing their position relative to the Sun, as well as using the rotation of panels relative to the Sun.

Changing the position of solar panels relative to the oncoming aerodynamic flow of the earth's residual atmosphere in low orbit results in a change in the midsection area and the satellite's ballistic coefficient. This change has an effect on the number of corrections of the orbital period, which is becoming more frequent due to the increase in the aerodynamic drag force. [1]

A study was conducted of changes in the time parameters of the low-orbit correction of the AIST-2 satellite due to changes in the position of solar panels relative to the aerodynamic flow vector. For calculations were used the space and ballistic design parameters: the mass from 500 kg to 700 kg; the area of the midsection of the hull about 2 m^2; of solar panels from 6 m^2 to 10 m^2; the angle between the normal of the solar panels relative to the vector of the aerodynamic flow from 0^0 to 90^0; low altitude near-circular orbit from 400 km to 500 km. To calculate the time parameters of the correction sections, the following characteristics of the electric jet engine were used: thrust force-20 mN; specific impulse-12500 m/s.

Studies have shown that in the space of design characteristics as a ballistic coefficient can be equal to values from 0.0033 m^2/kg to 0.0276 m^2/kg. With a minimum ballistic coefficient, maintaining the orbital period with an accuracy of 3 seconds using the thrust force of an electric jet engine is possible at all orbital heights from the specified range. At the average state of atmospheric density according to GOST P25645.166-2004 [2], the relative time of active sites in one correction cycle will be from 1% to 5 %, depending on the altitude of the orbit. Fuel consumption for maintaining an orbit for five years can vary from 3 kg (in an orbit with a height of 500 km) to 12 kg (in an orbit with a height of 400 km).

At the maximum ballistic coefficient, maintaining the period of rotation with an accuracy of 3 seconds using the specified thrust force of the electric jet engine is also possible at all orbital heights from the specified range. At the average state of atmospheric density, the relative time of active sites in one correction cycle will be from 6% to 29 %, depending on the altitude of the orbit. The fuel consumption for maintaining the orbit for five years can vary from 15 kg to 72 kg.

As a result, based on the results obtained, it is possible to select the required area of solar cells and the allowed rotation angles of their panels for the AIST-2 satellite class, depending on the height of the working orbit and the state of The earth's atmosphere for the planned period of the satellite's existence.

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ABLATIVE PULSED PLASMA THRUSTER FOR CONTROLLING ORBITAL MOTION OF SMALL SATELLITE OPERATING AS PART OF A CONSTELLATION OF EARTH REMOTE SENSING SATELLITES

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Exploration of Arctic requires creation of transport and logistics infrastructure. Severe natural and climatic conditions in that region require particular approach to information support for ship navigation, which cannot be provided without the use of space-based observation systems. The available Russian systems for Earth remote sensing (ERS) are unable to fully satisfy the need for space data in the Arctic region.

World experience in development of space ERS systems shows a tendency of growth of the main target characteristics - spatial and spectral resolution, as well as imaging efficiency. Increase in imaging efficiency is achieved due to transition from single small remote sensing satellites to their constellations.

To solve the problem of controlling orbital motion of small ERS satellite as of a part of constellation, it is necessary to equip such satellites with corrective propulsion systems (CPS). At the same time, such CPS must have rather low power consumption and long enough lifetime.

It is shown in [1,2] that if the power attributed to the orbital motion control of small ERS satellite is less than 150 W, it is desirable to equip it with CPS based on ablative pulsed plasma thruster (APPT).

In 2019-2020, the RIAME MAI designed, developed and tested the APPT model for a small ERS satellite. The paper presents the main results of research tests of such model. According to test results, the calculation of the main performance of electric propulsion system (EPS) based on the developed APPT was made.

The paper also presents the main results of mission analysis for a small ERS satellite that operates as a part of constellation and is equipped with EPS based on the developed APPT.

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FORMATION AND MAINTENANCE OF THE ORBITAL STRUCTURE OF MULTI-LEVEL SPACE CONSTELLATIONS FOR EARTH MONITORING USING ELECTRIC PROPULSION

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Keywords: *multi-level space system, Earth remote sensing, low-orbit small satellite, diffraction optical elements, electric propulsion system, orbit control.*

Existing space monitoring systems usually consist of Earth remote sensing spacecraft that operate in low orbits and provide consumers with high-resolution information about ground objects. However, the work of spacecraft in low earth orbit is characterized by such negative factors as a high rate of change of the parameters of the working orbit due to atmospheric effects and other perturbations, constraints on the bandwidth of the review, a short time of stay of the device in the zone of radio visibility of ground point to transmit the information. Continuous monitoring of specified areas of the Earth is provided by the spacecraft in geosynchronous, in particular in geostationary orbit. It is known that projects of information systems of this kind are actively developed in the interests of the US Department of defense (e.g. "MOIRE" project) [1].

At the same time, such a system is a large-sized structure, the deployment of which in space is associated with the solution of a set of fundamental problems in the field of space flight mechanics and control of position and orientation in orbit. At the moment, it is difficult to provide the space grouping with a sufficient number of spacecraft with a diffraction optical system. The use of low-orbit small spacecraft, for example, on the basis of the developed in Space-Rocket Center "Progress" together with Samara University unified platform "AIST-2" [2], will make it possible to create a constellation of the necessary number of Earth remote sensing spacecraft in a short time. Equipping spacecraft with an electric propulsion system will increase their active life time and flexibly manage the configuration of the orbital grouping in low orbits up to 500 km high.

Thus, the problem of rapid, flexible and effective monitoring of the Earth is solved by creating a multi-level constellation of spacecraft consisting of small maneuvering Earth remote sensing spacecraft equipped with electric propulsion systems and operating in low orbits, as well as spacecraft with a diffraction optical system operating in high-elliptical (geosynchronous) or geostationary orbits. Bringing high-orbit spacecrafts to the point of standing and then holding them is another task that can be solved using electric propulsion systems.

Thus, the new generation of Earth remote sensing systems should have the following characteristics (properties):

- availability of several levels of primary information providers (spacecraft);

- a complex structure characterized by distributed control and informational network interaction of spacecrafts with each other and with a ground control center;
- availability of various types of target equipment (optoelectronic, diffraction-optical, etc.) that allow monitoring the Earth's surface in real time;
- the ability to self-organize using effective methods of orbits' control based on high-precision and cost-effective propulsion systems, such as electric propulsion systems.

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PROPULSION SYSTEM FOR A SCIENTIFIC-EDUCATIONAL NANOSATELLITE

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This paper presents the results of propulsion system (PS) flight prototype development, meant for universities' scientific-educational nanosatellites orbit correction. One of the main requirements are safety measures for the personnel and the environment during testing, pre-flight preparations and in flight. The most common university nanosatellite standard – CubeSat 3U – was considered. The main stages of the development were described in [1-2].

Using waterethanol solution to prevent freezing as a working liquid allows meeting all the requirements. The proposed PS has high energy consumption to grant vaporizing the working liquid and its further heating. Such consumption can be supplied by power supply unit (PSU) with high peak power energy storage devices. They are charged for a relatively long time when there is extra power generated and discharged fast on correction pulse required. Super capacitors were used as such storage device in this study [1-3].

The tank has three parallel spiral springs to extrude the working body. This grants high reliability of the system.

The calculations performed showed that a single correction pulse to change velocity of a 4.5 kg nanosatellite for 0.1 m/s at 500K heated vapor temperature will require 0.4g of working liquid. The nozzle exhaust velocity will be 1500m/s.

The SamSat nanosatellite platform, developed in Samara University [4] allows (1-1.5)U space for a PS. It is used for working body tank with dimensions about 170×10^{-6} m^3. Considering minimal working liquid density 940 kg/m^3 its volume would be 160g. This mass allows 400 correction pulses. The delta-velocity budget is about 40 m/s which is enough for most of nanosatellite maneuvering tasks.

An autonomous control system with PSU was designed for the PS. It contains of three boards. One of them is a standard PC-104 board with MCU. It is set in the support systems stack. The other two boards are non-standard and were designed for high-current and noisy components. They are set up on the top and on a side of the tank. The boards contain valve driver, power keys and supercapacitor battery charger-balancer. The required amount of energy is stored in the battery of 8 supercapacitors. Each cell has capacity of 100 Farad and maximum voltage of 2.7V.

Measuring the thrust of the PS in vacuum was done using the designed setup. Fig.1 shows calibration data (nonlinearity about 0.1%). Figure 2 shows PS in the thrust measurement setup inside the thermal-vacuum chamber. Figure 3 shows the results of one of the tests.

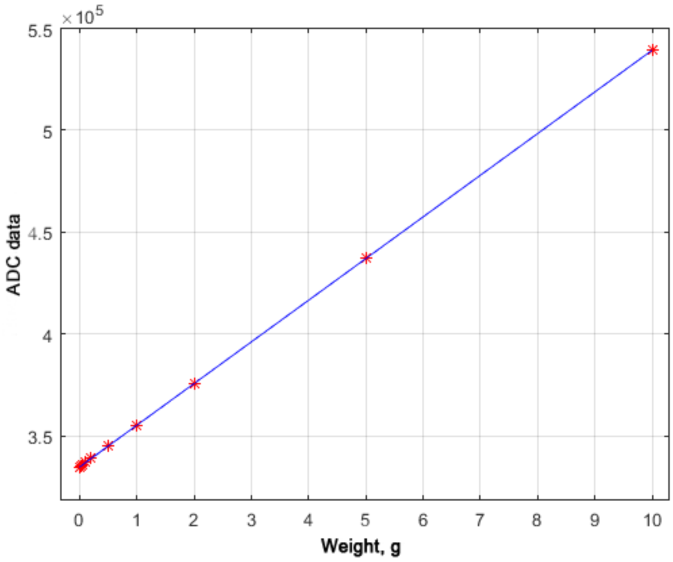

Fig.1. Calibration datafor thrust measuring setup

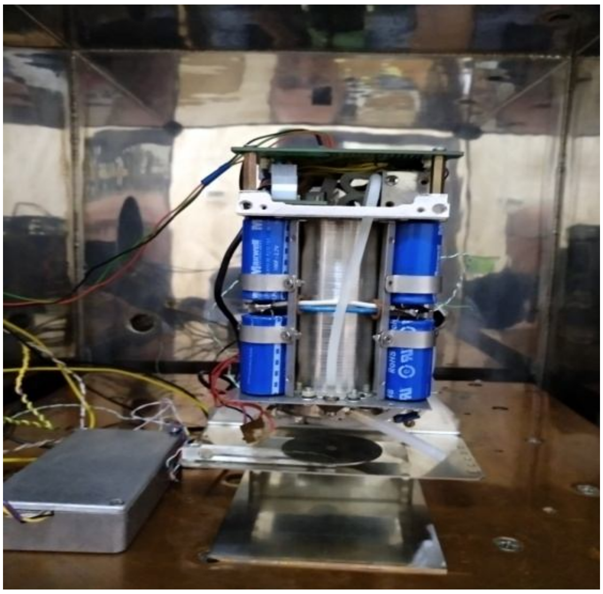
Fig.2. PSinthe thrustmeasuringsetup

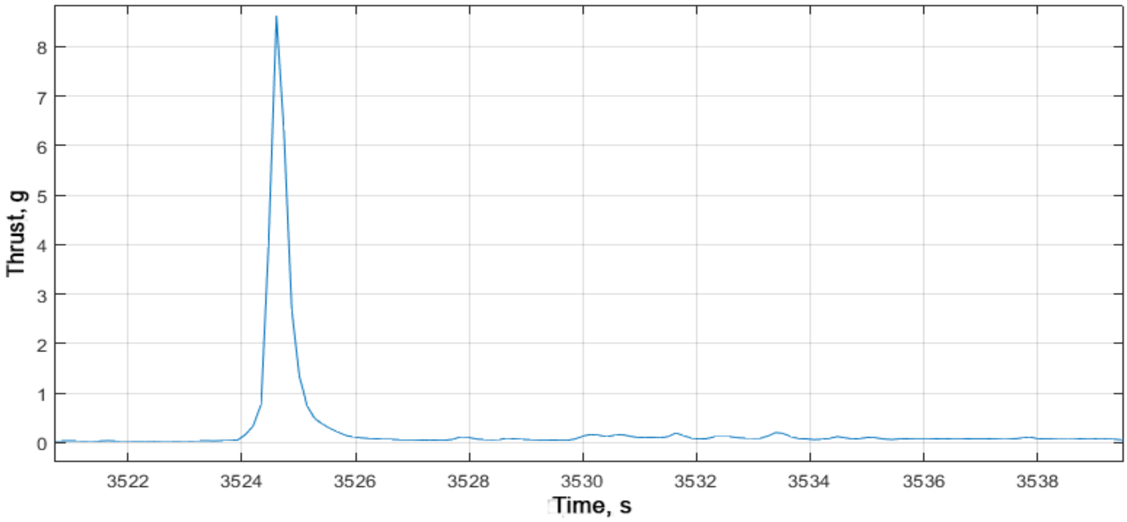

Fig.3. PSthrusttestresults

It can be seen that the developed PS fight prototype meets the requirements.

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Current state and perspective development trends of propulsion systems for small spacecraft of multi-satellite orbital constellations

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Currently, the world is actively developing the market of multi-satellite constellations of small spacecraft (smallsats). At the beginning of 2020, it was announced the creation of about twenty constellations of the smallsats weighing up to 500 kg. The total number of smallsats is several thousand.

Key aspects of such satellite constellations are permanent atmospheric drag compensation in low earth orbits and a precise positioning for each satellite in constellation.

The thrusters for these satellites have the following features:

- very low mass and sizes;
- high technological efficiency associated with large-scale production;
- high reliability in unlimited start/stop cycles;
- low thrust, high impulse, thrust vectoring control.

The paper covers the current state and perspective development trends of propulsion systems to ensure the functioning of the constellations of the smallsats, in particular: thermocatalytic thrusters (TCT), stationary plasma thrusters (SPT), pulsed plasma thrusters (PPT), ion thrusters featuring various ionization methods, colloid electric thrusters that use water as propellant. Their advantages and disadvantages are shown. Conclusions are made about the need for research on promising thrusters types and the development of miniaturization of existing structures, ensuring the possibility of mass production of propulsion systems for the constellations of the smallsats.

COLLOID PROPULSION SYSTEM FOR NANOSATELLITES

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In this paper a propulsion system based on a colloid microthruster with a slit emitter for nanosatellites is consiedered. The results of parametric analysis and testing of a prototype microthruster are presented. According to the test results, thrust was obtained at the level of 0.15 mN. Based on the numerical calculations, the developed design should provide a total pulse of 1000 N·s with the overall dimensions of the system not exceeding CubeSat 1U unit.

Colloid propulsion microthruster, slit emitter, extractor electrode, electrostatic thruster, nanosatellite.

In this paper a colloid propulsion system based on a colloid microthruster with a slit emitter made of a dielectric material is presented. The slit emitter provides a specific thrust of greater values compared to the needle emitters.

During the development of a colloid microthruster it was determined that the key parameters influencing the distribution of electrical field in the emitter/extractor system are emitter slit height, emitter-extractor distance and the aperture of extractor electrode. It was found that the desired characteristics could be achieved with the emitter slit height of 20 μm, emitter-extractor distance of 3 mm and extractor aperture height of 3 mm. That combination of parameters provided adequate electric field strength at the emitter slit opening and good gradient of electric field along the emitter/extractor gap with a good uniformity across the emitter slit.

The designed emitter consists of two flat elements made of dielectric material separated by thin spacer. The thickness of the spacer determines the width of the capillary slit formed by a gap between the edges of an emitter halves.

A prototype of a colloid microthruster was fabricated and tested. During the thrust tests the device was installed on a precision balance, the change in the readings of which was used to determine the thrust generated by the microthruster. According to the test results, thrust was obtained at the level of 0.15 mN (0,015 gf) which is enough for orbital manoeuvres nanosatellites of CubeSats 2U and 3U unit.

The designed propulsion system contains a colloid microthruster, propellant feed system, thermostabilization system, power supply units of emitter and extractor, a control system, charged-particle neutralizer and fastening components. Propellant is supplied into the microthruster by a thermo-throttle and a control valve.

In a charge neutralization system to compensate ionized particle current from the propulsion system it is possible to use a cathode with thermionic emission or an alternating electric field between the emitter and the extractor. In the first case, emitted electrons in the cathode-neutralizer compensate the positive charge of propellant particles at the thruster exhaust. In the latter case, the charge of emitted propellant particles alternating on a set frequency.

The overall dimensions of the developed colloid propulsion system is not exceeding CubeSat 1U unit while its power consumption during operation will not exceed 12 W.

According to the numerical calculations the developed design of a colloid microthruster should provide a total pulse of 1000 N·s.

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TECHNICAL SESSION G

ELECTRIC PROPULSION APPLICATION IN PLANETS AND ASTEROID RENDEZVOUS MISSION

MARS- plus EUROPA-INPPS Flagship Missions with High Power Electric Thrusters and Heavy Science Payload

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Abstract

The first non-human INPPS (International Nuclear Power and Propulsion System)flagship flight with orbits Earth-Mars-Earth-Jupiter/Europa (after 2025) is the most maximal space qualification test of INPPS flagship to carry out the second INPPS flagship flight to Mars with humans (in the 2030th). This high power space transportation tug is realistic because of 1) the successful finalization of the European-Russian DEMOCRITOS and MEGAHIT projects with their three concepts of space, ground and nuclear demonstrators for INPPS realization (reached in 2017), 2) the successful ground based test of the Russian nuclear reactor with 1MWel plus important heat dissipation solution via droplet radiators (confirmed in 2018) and plans by Russia for space qualification in the 2020th and 3) the perfect celestial constellation for a Earth-Mars/Phobos-Earth-Jupiter/Europa trajectory between 2026 and 2035.

Primary it will be described the status of INPPS flagship with its subsystems with focus on high power electric thrusters. As a function of Isp for INPPS flagship electric thrusters heavy science payload opportunities to Mars – up to 18 t – and to Europa moon – up to 11 t can be transported: like VaMEx (Valles Marineris Explorer) and FAIM/GRIPS (instrument package proposal to observe Mars atmospheric dynamics from hydroxyl airglow) and for Europa TRIPLE (an ice melting probe with integrated nanoAUV payload).

In addition the realizations of the ground and nuclear demonstrators and higher Earth orbit robotic assembly of INPPS will be sketched. Moreover the MW nuclear reactor as the power source for the flagship agrees with the 1992 UN principles relevant to the use of nuclear power sources (NPS) in outer space. Therefore this talk will sketch the legal and policy issues of INPPS flagship.

TECHNICAL SESSION H

ELECTRIC PROPULSION APPLICATION FOR UTILIZATION OF SPACE DEBRIS

CURRENT EXPERIMENTAL ACTIVITIES ON DEVELOPING AN ARCJET DEORBIT MODULE AT IRS

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Key words: *Arcjet, Additive Manufacturing, Mega Constellations, Deorbit System.*

Mega constellations are currently a major topic for mission analysis at both space agencies and industry. The goal is a shift from single large, heavy, and expansive geostationary Earth orbit satellites to several small low-cost satellites in low Earth orbit (LEO). With shorter life times and large numbers, lower qualification requirements and mass production could reduce costs significantly. With increasing numbers of satellites featuring flexible designs, advanced deorbit mechanisms become necessary to self-sustain the constellations and mitigate the risk for creating space debris.

This study presents the progress on developing a deorbit module based on a thermal arcjet system at the Institute of Space Systems (IRS). In the field of electric space propulsion, thermal arcjets combine high thrust-density with a reasonable weight-specific impulse. For a deorbit system, this allows a compact design to achieve fast and active deorbit of a satellite. Adequate deorbit strategies are considered crucial to self-sustain a constellation for a longer period of time [1]. Deorbiting decommissioned satellites fast could save significant amount of operation time, hence, reducing overall constellation costs on the one hand, but could also reduce the risk of collision during descent on the other.

During ESA's Cleanspace activity, IRS conducted an in-depth system study investigating several thermal arcjet systems for the purpose of end-of-life deorbiting [2]. These form the basis of the current development to achieve flexibility in design by the application of additive manufacturing (AM). The goal is a highly integrated nozzle design featuring optimized performance, via integrated cooling channels, regenerative gas generator, lightweight design, and optimized internal nozzle geometry for a chosen operational point. The suitability of AM tungsten, which is required for the arc environment's high temperatures, was already demonstrated during a test campaign at IRS [3].

This paper presents the experimental results of the thrust performance gained for characterizing the AM tungsten nozzle's regenerative cooling channels. Furthermore, lifetime limiting factors will be discussed based on the test campaign. Parallel to the experimental activity, a tool for designing optimized nozzle geometries for different propellants and power classes is being developed and first results and verification will be shown.

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LOW-THRUST ENGINES APPLYING IN PROBLEM OF INSERTION AND SPACE DEBRIS DISPOSING AT GEOSTATIONARY ORBIT

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The rocket-space system project, intended for space debris deorbiting from geostationary orbit, is studied.

In accordance with international requirements, fragments of space debris (FSD) should be transferred to disposal orbit – circular orbit, which is 235 km higher than GSO.

The most interesting case is applying of low-thrust for this purpose, since this type of engines provide orbital maneuvering with minimal waste of working fluid.

At the same time, the flight duration is a distinguishing characteristic for low-thrust transfer, especially when moving to GSO. Therefore, we use hybrid transfers that include maneuvering with chemical and low-thrust engines.

Main feature of our space-rocket system is using of already designed and applied systems. Major components of the space-rocket system are listed in Table 1.

Table 1 – Main components of space-rocket system

N	Space-rocket system components	Prototype	Purpose
1	rocket	“Soyuz-2.1b”	injection of SDC to parking orbit
2	booster	“Fregat”	transfer to geostationary transfer orbit
3	spacecraft debris collector (SDC)	based on satellite bus “Ekspress-2000”	transfer to GSO and cyclic disposal operation

We consider SDC’s trajectory plan that takes into account its transfer to GSO and following cyclic disposal operations between GSO and a disposal orbit:

1) rocket injects the booster with the SDC to parking orbit;
2) the booster, in impulse mode, launches the SDC to geostationary transfer orbit with maximum possible eccentricity;
3) the SDC moves to GSO by its low-thrust engines and rendezvous and dock a FSD;
4) the SDC carries out series of shuttle operations between GSO and a disposal orbit that includes two stages:
   4.1) the SDC with FSD on-board moves to disposal orbit, where the SDC unfixes the FSD;
   4.2) the SDC move back to GSO and rendezvous and dock to next FSD;
5) shuttle operations continue until the SDC has working fluid.

Next we consider main parts of trajectory plan.

Rocket inject booster with SDC to parking orbit with parameters – height 200 km, inclination 51,6 deg.

Next we consider step-by-step optimization of the trajectory plan.

The problem of hybrid transfers optimization consists of a SDC transferring to GSO with maximum possible mass of working fluid for shuttle operations.

We suppose that the booster “Fregat”, in impulse mode, launches a SDC to geostationary transfer orbit with maximum possible eccentricity. Next, the SDC moves to GSO with low-thrust by optimal trajectory.

It appeared, that optimal parameters of the geostationary transfer orbit are semi major axis 16 500 km, eccentricity 0,6 and inclination 51,6 deg.

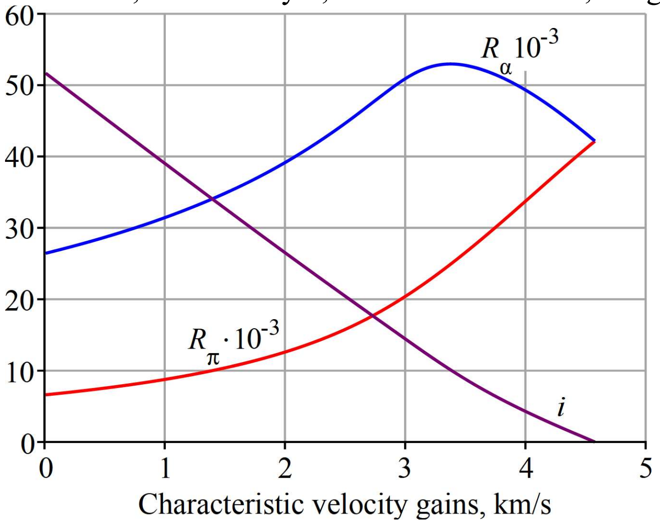


Figure 1 – Trajectory of SDC between geostationary transfer orbit and GSO

The trajectory of SDC between geostationary transfer orbit and GSO is plotted in fig. 1, where time history of perigee R_π, apogee radius R_α and inclination i is shown. We consider gain of characteristic velocity as time.

Total duration of transfer is 157 days.

Total characteristic velocity gains is 4,6 km/s.

Total working fluid consumption is 950 kg.

Based on 3400 kg initial mass of SDC (i.e. satellite bus “Ekspress-2000”) that includes 2150 kg of construction mass, we obtained that there are 300 kg of working fluid to shuttle operation.

For shuttle operation optimization we introduce two criterions. The first is maximization of disposed FSD number (equally to minimization of characteristic velocity gain) and the second is minimization of shuttle operation duration.

Shuttle operations consist of several transfers between GSO and a disposal orbit. Boundary conditions for one cycle of operation are constant (figure 2).

Trajectory at one cycle of shuttle operation consists of two stages. The first (forward) is increases orbit and keep its zero eccentricity. The second (backward) returns SDC to GSO into the standing point (toward the next FSD as in figure 2).

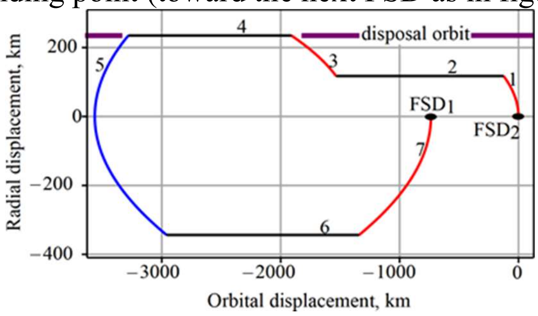


Figure 2 – Scheme of shuttle operation at one cycle

On figure 2 there are following designations:

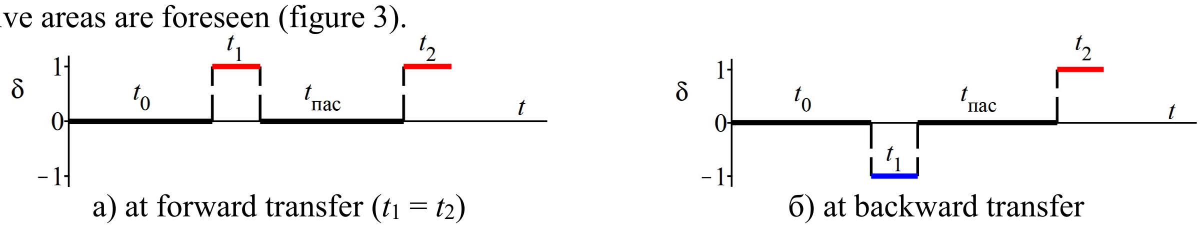
numbers 1 – 3 trajectory of forward transfer (GSO – disposal orbit);

numbers 4 – 7 trajectory of backward transfer (disposal orbit – GSO);

even numbers are passive areas (low-thrust engine is off);

odd numbers are active areas (low-thrust engine is on).

FSD motion control is carried out by reversing of transversal acceleration from thrust, also passive areas are foreseen (figure 3).



a) at forward transfer ($t_1 = t_2$)　　б) at backward transfer

Figure 3 – Control program

where δ is a sign of acceleration from thrust

A limitation of optimization problem is mass of SDC working fluid.

Here we have a specialty that for forward transfer SDC moves both itself and FSD, but for backward it pulls itself only. Thereby, acceleration from thrust value at different parts of shuttle operation will be different.

Also we neglect the decreasing of working fluid mass, since it small compared to mass of SDC. It means that the value of acceleration from thrust is determined by SDC construction mass and mass of FSD at forward transfer, and SDC construction mass only at backward transfer.

In this approximation, number of shuttle operation cycles and its duration are written as:

$$n = \frac{M_0' - M_c + \dfrac{V_{x_2}}{C} M_0'}{\dfrac{V_{x_1}}{C}\left(M_0' + M_{FSD}\right) + \dfrac{V_{x_2}}{C} M_0'}, \quad T = \left(t_{act_1} + t_{pas_1}\right) \cdot n + \left(t_{act_2} + t_{pas_2}\right) \cdot (n-1).$$

For mass of FSD 2000 kg value of acceleration from thrust at forward transfer is 0,22E–6 m/s^2 and 0,39 E–6 m/s^2 at backward, when value of thrust is 0,996 N.

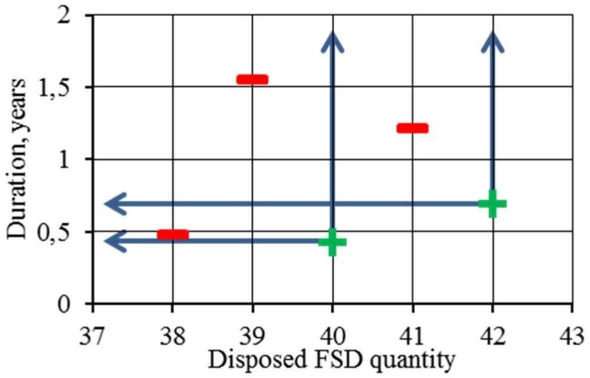


Figure 4 – Solutions of shuttle operation problem in frame «*n* – *T*»

We obtained five solutions of shuttle operation optimization problem, two of them are V. F. D. Pareto optimal (designation "+" at figure 4):

1) 42 disposed FSD (it is a maximum) during 252 days;
2) 40 disposed FSD during minimal duration 151 days.

Arrows on figure 4 show direction of criteria deterioration. Solutions above and left of arrow are not optimal (designation "–" at figure 4).

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