Electric propulsion for small satellite-inspector

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Abstract: **This paper presents the results of development (on first stage) of small satellite-inspector equipped with electric propulsion power module and the results of its parametrical and dynamical optimization. The method of calculation of characteristic speed expenses as a main performance for maneuver realization is developed. Laws of optimum control of jet acceleration in the form of dependence on orbit elements deviations in a task of relative movement control of two satellites, one of which is satellite-inspector are received by means of Pontryagin maximum principle. Modeling of relative movement of two satellites, one of which is the satellite-inspector equipped with electric propulsion module was carried out. The algorithm of synthesis of satellite-inspector's principal design parameters, containing the iterative scheme of determination of electric propulsion power module parameters, was developed and realized. The external view of small satellite-inspector with mass of 750 kg equipped with the electric propulsion power module which included 8 engines "SPT-100" (production of FSUE EDB Fakel) was obtained on base of this algorithm.**

Nomenclature

V_X	=	characteristic speed
μ	=	relative payload of satellite-inspector
a_0	=	initial jet acceleration
$\overline{p}'$	=	vector of design parameters of the electric propulsion power module
c	=	exhaust velocity of a working fluid
Z	=	range of dynamic operations

I. Introduction

The present stage of space technologies development is characterized by the increasing growth in the numbers of spacecrafts which are built and lofted to orbit. The ground space control systems, as the practice of space flights shows, cannot provide sufficient information about the purpose, technical characteristics, and actual usage, of launched spacecraft. There are new objectives, which for the most part can be most successfully achieved by space-based means. These objectives include: approximating a spacecraft to determine its type; following the spacecraft to analyze its operability and technical condition; control of general space environment in a given sector. The vehicle which is designed to perform these tasks is known as satellite-inspector.

Thus, orbital inspection is a complex of orbital operations carried out by means basing in space and directed on obtaining information about space objects by rapprochement with inspected object for its technical diagnosing.

Space objects concerning which inspection can be carried out, are the functioning and failed spacecrafts and stations, systems of long disclosure (like "solar sail", multilink antennas and scanners), large-size objects of space junk.

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Now we known some successfully realized projects of satellite-inspectors. They are small satellites of series «XSS», system «Orbital Express», spacecrafts «SBSS» and “DART” (USA), satellite «BX-1» (China), project «PRISMA» (Sweden, ESA). It should be mentioned that the analyzed satellites-analogs belong as a rule to the class of small satellites with weight up to 1000 kg and are equipped with the chemical rocket engines working at unicomponent fuel (hydrazine). At the same time, satellites, which are carrying out inspection with small thrust, created by the electric propulsion power modules (EPPM) are of the most practical interest.

II. Dynamic maneuvers of satellite-inspector and peculiarities of their implementation by means of electric propulsion

In accordance with the purpose of the satellite-inspector, it is able to perform the following classes of dynamic operations[1]:

1) rendezvous with space object (the task of bringing the satellite-inspector from the initial state X_0 to the final variety X_K);

2) accompanying of a space object during a specified time interval (problem of control of the relative motion of two spacecrafts, one of which is satellite-inspector, another one is satellite-target);

3) holding in the field of space X_K during the specified time.

Modern stage of space flights requires increased efficiency of created and developing space technique. One possible way of solving this problem is to use for space maneuvers and transport operations prospective propulsion systems with high performances. Such systems include electric propulsion (EP) of small thrust operating on the principle of acceleration of the working fluid in an electrostatic or electromagnetic fields.

Modern electric propulsion still have such thrust that the acceleration given to low orbit spacecraft is of the same value as the disturbance of the environment. In this case, the models describing pulse orbit correction engine of big thrust at apogee-perihelia points are not applicable. Orbital maneuvers become long in case of small thrust, including a continuous multiturn, which significantly complicates the problem of satellite's control with electric propulsion.

A. Problem of design-ballistic optimization

The problem of calculation of satellite-inspector's trajectories equipped with electric propulsion is to obtain approximate solutions, allowing you to select the structure and parameters of the control law, to determine the energy of the maneuver and the EPPM design parameters.

As a particular criterion of optimality in the problem of design-ballistic optimization we consider a relative payload of satellite-inspector[2]:

$$\mu = \frac{M_{PL}}{M_0} = 1 - \alpha_{PU} \cdot \frac{a_0 c}{2\eta} - \gamma_{PS} \cdot a_0 - \frac{a_0 T_m}{c}(1 + \gamma_{SFS}), \tag{1}$$

where M_{PL} – mass of payload, M_0 - initial mass of the satellite-inspector after separation from the rocket, α_{PU} - specific mass of the power plant; a_0 – initial jet acceleration, c - exhaust velocity of a working fluid, η - coefficient of efficiency of the engine, γ_{PS} – mass of propulsion system, T_m – total motor time, γ_{SFS} – mass of system of feed and storage of working fluid.

The vector of design parameters of the EPPM $\overline{p}'$ includes: N_{PU} - power of the power plant (W), S_{SP} - size of solar panels (m^2), M_{WF} - mass of working fluid (kg), n – quantity of engines in EPPM, P_Σ - total thrust of all engines (N), a_0 – initial jet acceleration (m/s^2), c - exhaust velocity of a working fluid (km/s), R – engines resource (h). The specified set of parameters can be narrowed, as some of design parameters are interrelated.

The problem of joint optimization of controls *y(t)*, trajectories *x(t)* of dynamic operations *z* and EPPM design parameters p' usually divide into two independent:

1) dynamic – finding the optimal control programs and obtaining dynamic characteristic of the maneuver, which is characteristic speed V_X. Then the optimal control is defined as:

$$\overline{y}_{opt}(t, p') = \arg\min_{\overline{y} \in Y} V_{XK}(\overline{u}, \overline{p}', \overline{x}_0, \overline{x}_\kappa). \tag{2}$$

2) parametrical – finding of optimum design parameters of the EPPM:

$$\overline{p}'_{opt} = \arg\max_{\overline{p}' \in P} \mu[V_{XK}(\overline{y}_{opt}, \overline{p}', \overline{x}_0, \overline{x}_\kappa), \overline{p}', T_m], \; T_m = fixe. \tag{3}$$

In case V_{XK} does not depend on design parameters of EPPM, then the dynamic problem is formulating as the minimum characteristic speed and solving independently of the problem of parametric synthesis.

B. Calculation of expenses of characteristic speed

The general task of orbit elements control is to change the vector of orbit elements E_0 thus that it reached demanded value of E_K at the minimum expenses of characteristic speed V_X. When the thruster is operating continuously this task is equivalent to the one of speed. The scheme of separate control of orbit elements was proposed. The authors have received particular solutions corresponding to separate (locally optimal) control of orbit elements and top evaluation (guarantee value) of characteristic speed expenses for the maneuver:

$$V_X^* \approx V_{rnd} \cdot \nabla V_X(E) \cdot \Delta E \,, \tag{4}$$

where $\nabla V_X(E) = (\frac{1}{2A_0}, \frac{\pi}{4}, \frac{\pi e_0}{4}, \frac{\pi}{2}, \frac{\pi i_0}{2})^T$, $\Delta E = (\Delta A, \Delta e, \Delta\omega_a, \Delta i, \Delta\Omega_a)$, $V_{rnd} = 7.64 km/s$ - round characteristic speed.

In equation (4) $A_0, \Delta A$ - - the initial value and the deviation of the semimajor axis; ε - the gravitational parameter of the Earth; Δe - deviation of eccentricity of orbit; $\Delta\omega_a$ - deviation of the argument of perigee (excluding the precession); Δi - deviation of inclination; $\Delta\Omega_a$ - deviation of the longitude of the ascending node (excluding the precession); i_0 - the initial value of the inclination; e_0 - the initial value of the eccentricity of orbit. Figure 1 shows isolines of characteristic speed expenses (50 m/s and 100 m/s) for control of orbit elements: *A, e, ω*.

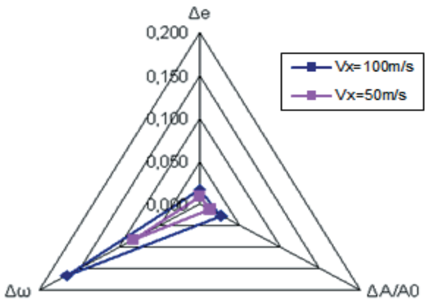

Figure 1. Isolines of characteristic speed expenses.

Equation (4) can be used to obtain upper estimates of the characteristic speed expenses for the maneuver, however, for more accurate decisions requires considering the effect of noncentrality the Earth's gravitational field. If in the expanding of geopotential we consider only the second zonal harmonic (J2), then its influence on the rate of perigee and the longitude of the ascending node translocation can be characterized by the following formulas[3]:

$$\Delta\omega_g = -\frac{\beta(5\cos^2 i - 1)}{2A^{3,5}\sqrt{\varepsilon}} t; \qquad \Delta\Omega_g = -\frac{\beta \cos i}{A^{3,5}\sqrt{\varepsilon}} t \,, \tag{5, 6}$$

where $\beta = 2.634 \cdot 10^{10}$ km^5/s^2.

Time *t* in equations (5, 6) can be expressed via the characteristic speed and jet acceleration: $t = V_X / a_0$. We can see that it includes the design parameter a_0, so V_X can be determined iteratively. It is shown in Ref. 1 the functional connection of the maneuver's characteristic speed V_X and jet acceleration a_0 for different models of motion. This connection can be approximated by an asymptotic curve of the form:

$$V_X = \widetilde{V}_X \cdot (1 + \frac{k}{a_0}) \,, \tag{7}$$

where k - is a constant coefficient, $\widetilde{V}_X$ - the value of the characteristic speed without considering the disturbing factors.

Table 1 shows the calculation of the characteristic speed expenses on the orbital elements control.

Table 1. Characteristic speed expenses on the orbital elements control.

ΔA/A₀ (*ΔA*, km)	V_{XA}, m/s	*Δe*	V_{Xe}, m/s	*Δω*, °	$V_{X\omega}$, m/s	*Δi*, °	V_{Xi}, m/s	*ΔΩ*, °	$V_{X\Omega}$, m/s
0.0001 (1km)	0.56	0.001	6.00	1	1.047	0.1	20.94	1	25.59
0.0003 (2 km)	1.12	0.002	12.00	2	2.095	0.2	41.89	2	51.18
0.0015 (10 km)	5.60	0.01	60.00	10	10.473	0.4	83.78	4	102.36
0.0022 (15 km)	8.40	0.015	90.01	20	20.945	0.5	104.73	5	127.95
0.0029 (20 km)	11.18	0.02	120.01	25	26.182	0.6	125.67	6	153.54
0.0044 (30 km)	16,76	0.03	180.01	40	41.891	0.8	167.56	8	204.72
0.0058 (40 km)	22.34	0.04	240.02	50	52.364	0.9	188.51	9	230.31
0.0073 (50 km)	27.90	0.05	300.02	60	62.836	1.0	209.45	10	255.90

C. Control of relative movement of two satellites

The authors consider the problem of control of the relative motion of two satellites, one of which is passive (I), the other one is active (II - satellite-inspector) equipped with EPPM of small thrust. To simulate the relative motion of two satellites we use linear perturbation theory, assuming small deviations in the parameters. After the procedure of linearization relatively to unperturbed Keplerian orbit the model of the relative motion can be written as[4]:

$$\begin{cases} \frac{d\Delta r}{dt} = \Delta V_r; \\ \frac{d\Delta L}{dt} = \Delta V_u - \lambda \Delta r; \\ \frac{d\Delta V_r}{dt} = 2\lambda \Delta V_u - \lambda^2 \Delta r + \frac{2\varepsilon}{r_1^3}\Delta r; \\ \frac{d\Delta V_u}{dt} = -\lambda \Delta V_r - \frac{V_{r1}}{r_1}\Delta V_u + \frac{V_{r1}}{r_1}\lambda \Delta r + a_T; \\ \frac{d\Delta z}{dt} = \Delta V_z; \\ \frac{d\Delta V_z}{dt} = -\lambda^2 \Delta z + a_W; \\ \frac{\partial \vartheta}{dt} = \frac{\sqrt{\varepsilon p}}{r_1^2}, \end{cases} \tag{8}$$

where Δr - deviation of the distance from the Earth's center to the projection of the satellite on the plane of the unperturbed circular orbit; Δu - argument of latitude; $\Delta L = \Delta u \cdot r$ - projection of the distance between satellites (I and II) on the arc of unperturbed orbit; Δz - deviation of the distance from the plane of the undisturbed orbit to the satellite; a_T, a_W - projections of the jet control acceleration on the axes of the orbital coordinate system; ΔV_r - deviation of the radial speed, ΔV_u - deviation of the transverse speed, ΔV_z - deviation of the normal speed (the projection of the speed on the normal to the plane of the undisturbed orbit), ε - is the gravitational parameter, t - is the current time, $\lambda = \sqrt{\varepsilon(1-e^2)/p^3}$ - average angular speed of the motion of the satellite-inspector along the reference orbit of passive satellite I; $p = A(1-e^2)$ – focal parameter.

We divide the relative movement into two components: a planar movement (control of elements $\Delta r, \Delta u, \Delta V_r, \Delta V_u$) and lateral movement (control of elements $\Delta z, \Delta V_z$).

Firstly we analyze a planar movement. We bring forward the first four equations of the system (8) to a simpler mind, ignoring the difference between the gravitational acceleration (here r_1, λ_1 refer to the orbit of the satellite I):

$$\begin{cases} \frac{d\Delta r}{dt} = \Delta V_r; \\ \frac{d\Delta u}{dt} = r_1^{-1}(\Delta V_u - \lambda_1 \Delta r); \\ \frac{d\Delta V_r}{dt} = 2\lambda \Delta V_u + 3\lambda_1^2 \Delta r; \\ \frac{d\Delta V_u}{dt} = a_T - 2\lambda_1 \Delta V_r. \end{cases} \tag{9}$$

The optimization problem is formulated in the form: we need to take the system from an arbitrary position to the coordinate origin as fast as possible, i.e. we need to find the form of the control law that delivers minimum to the functional

$$I = \int_{t_0}^{t_k} 1 dt \to \min . \tag{10}$$

Applying the algorithm of the Pontryagin maximum principle[5] to solve the optimization problem, we receive approximately-optimal control law:

$$a_T = -a_T sign\left[-\frac{1}{3}(\Delta u - 2\lambda_1^{-1} r_1^{-1} \Delta V_r) + \lambda_1^2 \frac{(\Delta r + \lambda_1^{-1}\Delta V_u)\left|\Delta r + \lambda_1^{-1}\Delta V_u\right|}{2\overline{a_T} r_1} \right]. \tag{11}$$

Figure 2 shows the results of modeling of two satellites' relative motion, one of which is satellite-inspector, equipped with EPPM of small thrust. The initial conditions of the simulation: Δr = 1 km, Δu = 3°, the modulus of transverse component of the acceleration: a_0 = 6.67·10^{-5} m/s^2, radius vector r_1 = 6778 km.

The control law in the form (11) is the exact solution to the problem if the deviations of the speed $\Delta V_r, \Delta V_u$ remains negligibly small during the whole time of the maneuver. In this case, the components of the status vector remain the only deviation along the radius Δr and along the orbit Δu. The structure of the optimal control contains only one switch. For this case we obtain the formula for expenses of the characteristic speed:

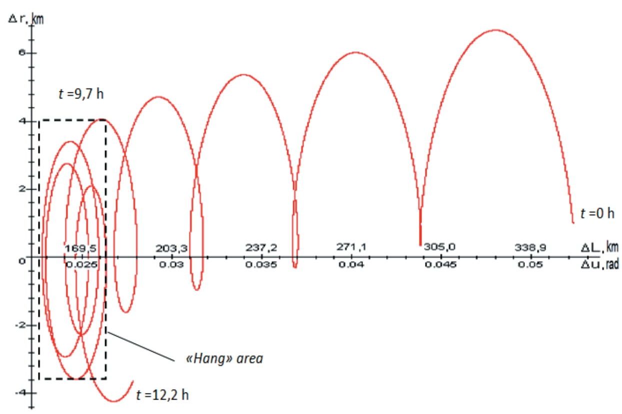

Figure 2. The results of modeling of relative movement of satellite-inspector and satellite-target.

$$V_{X\min} = \min_{\delta(0)} \left(\lambda_1^{-1} \left(-\frac{\Delta r_1 \lambda_1^2}{\delta(0)} + 2\sqrt{\left(\Delta r_1 \lambda_1^2\right)^2 + \frac{\Delta u \lambda_1^2 r_1 a}{3\delta(0)}} \right) \right), \quad (12)$$

where $\delta = \pm 1$ - switch function.

The expenses of characteristic speed according to (12) required for maneuvers of this type, are significantly lower the expenses of V_X, required to put the satellite-inspector in the area of the satellite-target (these expenses for initial conditions for which modeling was performed (Fig. 2) are estimated at 2 to 5 m/s).

Secondly we analyze a lateral component of the satellite-inspector movement, for which we write the fifth and sixth equations of system (8):

$$\begin{cases} \dfrac{d\Delta z}{dt} = \Delta V_z; \\ \dfrac{d\Delta V_z}{dt} = -\lambda^2 \Delta z + a_W. \end{cases} \quad (13)$$

To find the optimal control ensuring the fastest taking the system from an arbitrary position to the coordinate origin we apply the algorithm of the Pontryagin maximum principle again. In the result the control law of lateral component of satellite-inspector movement is:

$$a_W = -a_W \, sign\left[\Delta z + \frac{\Delta V_z \left|\Delta V_z\right|}{2 a_W} \right]. \quad (14)$$

III. Determination of basic design parameters of the EPPM and satellite-inspector

The choice of EPPM parameters defining a design shape of the satellite-inspector, is carrying out stage-by-stage, using at each stage a more complete and accurate model of the motion of the inspector. Basic values of EPPM principal design parameters $a_0^{(0)opt}, c_{opt}^{(0)}$ are determined on the basis of the upper estimate characteristic speed expenses. After determining these values $a_0^{(0)opt}, c_{opt}^{(0)}$ they are compared with the range of parameters of the real propulsion systems, formed by combining the variety of electric propulsion design parameters of and a variety of energy performances of launchers (launch payload). Further, the synthesis process is carried out iteratively. On the first iteration of the synthesis the disturbing factors having a significant impact on the satellite-inspector movement are taking into account. After finding these values $a_0^{(1)opt}$ and $c_{opt}^{(1)}$ they are again compared with the range design parameters of the real engines, which leads to a narrowing of the initial variety (number of alternatives is reduced). On the second iteration of synthesis for orbit altitudes H < 800 km the perturbing effects of the upper atmosphere are taken into account. For altitudes not above 400 km, it is possible to use the average maximum value on revolution of perturbing acceleration.

As a result, the EPPM parameters possess new values $a_0^{(2)opt}, c_{opt}^{(2)}$, which are again superimposed on the range of real electric propulsion alternatives that leads to further narrowing of the variety.

Thus, the process of determining the basic design parameters of EPPM is iterative, which is shown schematically in Fig. 3.

The synthesis algorithm of satellite-inspector basic design parameters is formed on base of the above-described iterative process of determining the basic design parameters of EPPM.

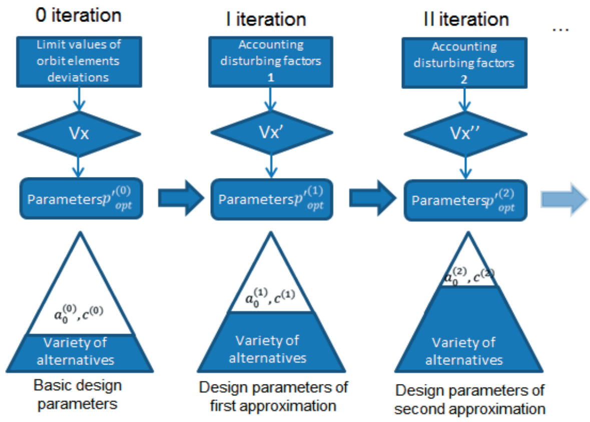

Figure 3. Scheme of the process of determining EPPM basic design parameters

A. The synthesis algorithm of the satellite-inspector's principal design parameters

1. The range of the extreme deviations of orbit elements that defines the area of satellite-inspector control relative to initial operational orbit is set out. The characteristic speed expenses necessary for maneuvers of each type are defined (table 1).

2. Maneuver with $V_{X\max}$ is selecting from a range (table 1) using (15). Maximum permissible time of the most "power-consuming" maneuver implementation $T_{m\ fix}$ is setting:

$$V_{X\max} = \max \sum_{E=1}^{5} V_{XE} = 432.94 m/s; \quad T_{m\ fix} = 2000\ \ h. \tag{15}$$

3. Basic values of jet acceleration and exhaust velocity of a working fluid are defining:

$$a_0^{(0)opt} = \frac{V_{X\max}}{T_{m\,fix}} = 6.01 \cdot 10^{-5} m/s^2; \; c_{opt}^{(0)} \cong \sqrt{\frac{2\eta V_X (1+\gamma_{SFS})}{\alpha_{PU} a_0}} \cong 12.41 km/s. \qquad (16, 17)$$

4. The set of design parameters of real propulsion systems is forming (database created by the method of morphological analysis). Principle of formation of alternatives variety is shown in Table 2. In the Table 2: M_{SATi} – mass of satellite-inspector (kg), $i = \overline{1,m}$; P_j – thrust of engine (N), $j = \overline{1,k}$; $a_{0\,ij}$ - values of control accelerations (m/s^2).

To form the range of EPPM real design parameters we used the following preliminary data (Fig. 3).

Table 2. Principle of formation of alternatives variety.

P_j, N / M_{SATi}, kg	P_1	…	P_k
M_{SAT1}	$a_{0\ 11}$	…	$a_{0\ k1}$
…	…	…	…
M_{SATm}	$a_{0\ 1m}$	…	$a_{0\ km}$

Electric rocket engines of small thrust	"SPD-25"	"SPD-35"	"SPD-50"	"SPD-60"	"SPD-70"	"SPD-100"	"AIPD-95"
Thrust, mN	7	10	20	30	40	85	3.5

Figure 3. Performances of modern Russian thrusters.

The expected design mass of the satellite-inspector is defined by power opportunities of the carrier rocket (mass of payload launched to circular orbit with the height of 400 km and inclination of 97º) (Fig. 4).

In the result it was formed a variety of alternatives which includes 56 variants (Table 3).

This variety can be expanded due to introduction of additional parameters: n - number of engines as a part of EPPM (working simultaneously in one direction of thrust) and N_{SAT} - number of the satellite-inspectors which are placed in a carrier payload zone.

Launcher	"Rokot"	"Strela"	"Start-1"	"Volna"	"Ciklon"	"Shtil"	"Cosmos"	"Soyuz-2.1v"
Payload, kg	1100	900	350	70	1400	260	800	1500

Figure 4. Mass of payloads of Russian carrier rockets.

Table 3. Initial variety of alternatives.

Mass of rocket's payload (mass of satellite-inspector), kg \ Thrust of engines, N		"SPD-25"	"SPD-35"	"SPD-50"	"SPD-60"	"SPD-70"	"SPD-100"	"AIPD-95"
		$7,00*10^{-3}$	***$1,00*10^{-2}$***	***$2,00*10^{-2}$***	***$3,00*10^{-2}$***	***$4,00*10^{-2}$***	***$8,50*10^{-2}$***	***$3,50*10^{-3}$***
"Rokot"	1100	$6,36*10^{-6}$	$9,09*10^{-6}$	$1,82*10^{-5}$	$2,73*10^{-5}$	$3,64*10^{-5}$	$7,91*10^{-5}$	$4,09*10^{-6}$
"Strela"	900	$7,78*10^{-6}$	$1,11*10^{-5}$	$2,22*10^{-5}$	$3,33*10^{-5}$	$4,44*10^{-5}$	$9,67*10^{-5}$	$5,00*10^{-6}$
"Start-1"	350	$2,00*10^{-5}$	$2,86*10^{-5}$	$5,71*10^{-5}$	$8,57*10^{-5}$	$1,14*10^{-4}$	$2,49*10^{-4}$	$1,29*10^{-5}$
"Volna"	70	$5,83*10^{-5}$	$8,33*10^{-5}$	$1,67*10^{-4}$	$2,50*10^{-4}$	$3,33*10^{-4}$	$1,14*10^{-3}$	$6,43*10^{-5}$
"Ciklon"	1400	$5,00*10^{-6}$	$7,14*10^{-6}$	$1,43*10^{-5}$	$2,14*10^{-5}$	$2,86*10^{-5}$	$6,01*10^{-5}$	$3,21*10^{-6}$
"Shtil"	260	$2,69*10^{-5}$	$3,85*10^{-5}$	$7,69*10^{-5}$	$1,15*10^{-4}$	$1,54*10^{-4}$	$3,35*10^{-4}$	$1,73*10^{-5}$
"Cosmos"	800	$8,75*10^{-6}$	$1,25*10^{-5}$	$2,50*10^{-5}$	$3,75*10^{-5}$	$5,00*10^{-5}$	$1,09*10^{-4}$	$5,63*10^{-6}$
"Soyuz-2.1v"	1500	$4,67*10^{-6}$	$6,67*10^{-6}$	$1,33*10^{-5}$	$2,00*10^{-5}$	$2,67*10^{-5}$	$5,80*10^{-5}$	$3,00*10^{-6}$

As a result of synthesis of a general variety of alternative the morphological table contains 504 alternatives: n = 1 to 3; N_{SAT} = 1 to 3 (Table 4).

Table 4. Final variety of alternatives contains 504 variants.

Mass of rocket's payload (mass of satellite-inspector), kg		Number of engines, n																				
		n = 1							n = 2							n = 3						
Number of satellite-inspectors, N_{SAT}	Thrust of engines, N	$7,00*10^{-3}$	$1,00*10^{-2}$	$2,00*10^{-2}$	$3,00*10^{-2}$	$4,00*10^{-2}$	$8,70*10^{-2}$	$3,50*10^{-3}$	$1,40*10^{-2}$	$2,00*10^{-2}$	$4,00*10^{-2}$	$6,00*10^{-2}$	$8,00*10^{-2}$	$1,74*10^{-1}$	$7,00*10^{-3}$	$2,10*10^{-2}$	$3,00*10^{-2}$	$6,00*10^{-2}$	$9,00*10^{-2}$	$1,20*10^{-1}$	$2,61*10^{-1}$	$1,05*10^{-2}$
N_{SAT} = 1	1100	$6,36*10^{-6}$	$9,09*10^{-6}$	$1,82*10^{-5}$	$2,73*10^{-5}$	$3,64*10^{-5}$	$7,91*10^{-5}$	$4,09*10^{-6}$	$1,27*10^{-5}$	$1,82*10^{-5}$	$3,64*10^{-5}$	$5,45*10^{-5}$	$7,27*10^{-5}$	$1,58*10^{-4}$	$8,18*10^{-6}$	$1,91*10^{-5}$	$2,73*10^{-5}$	$5,45*10^{-5}$	$8,18*10^{-5}$	$1,09*10^{-4}$	$2,37*10^{-4}$	$1,23*10^{-5}$
	900	$7,78*10^{-6}$	$1,11*10^{-5}$	$2,22*10^{-5}$	$3,33*10^{-5}$	$4,44*10^{-5}$	$9,67*10^{-5}$	$5,00*10^{-6}$	$1,56*10^{-5}$	$2,22*10^{-5}$	$4,44*10^{-5}$	$6,67*10^{-5}$	$8,89*10^{-5}$	$1,93*10^{-4}$	$1,00*10^{-5}$	$2,33*10^{-5}$	$3,33*10^{-5}$	$6,67*10^{-5}$	$1,00*10^{-4}$	$1,33*10^{-4}$	$2,90*10^{-4}$	$1,50*10^{-5}$
	350	$2,00*10^{-5}$	$2,86*10^{-5}$	$5,71*10^{-5}$	$8,57*10^{-5}$	$1,14*10^{-4}$	$2,49*10^{-4}$	$1,29*10^{-5}$	$4,00*10^{-5}$	$5,71*10^{-5}$	$1,14*10^{-4}$	$1,71*10^{-4}$	$2,29*10^{-4}$	$4,97*10^{-4}$	$2,57*10^{-5}$	$6,00*10^{-5}$	$8,57*10^{-5}$	$1,71*10^{-4}$	$2,57*10^{-4}$	$3,43*10^{-4}$	$7,46*10^{-4}$	$3,86*10^{-5}$
	70	$1,00*10^{-4}$	$1,43*10^{-4}$	$2,86*10^{-4}$	$4,29*10^{-4}$	$5,71*10^{-4}$	$1,24*10^{-3}$	$6,43*10^{-5}$	$2,00*10^{-4}$	$2,86*10^{-4}$	$5,71*10^{-4}$	$8,57*10^{-4}$	$1,14*10^{-3}$	$2,49*10^{-3}$	$1,29*10^{-4}$	$3,00*10^{-4}$	$4,29*10^{-4}$	$8,57*10^{-4}$	$1,29*10^{-3}$	$1,71*10^{-3}$	$3,73*10^{-3}$	$1,93*10^{-4}$
	1400	$5,00*10^{-6}$	$7,14*10^{-6}$	$1,43*10^{-5}$	$2,14*10^{-5}$	$2,86*10^{-5}$	$6,21*10^{-5}$	$3,21*10^{-6}$	$1,00*10^{-5}$	$1,43*10^{-5}$	$2,86*10^{-5}$	$4,29*10^{-5}$	$5,71*10^{-5}$	$1,24*10^{-4}$	$6,43*10^{-6}$	$1,50*10^{-5}$	$2,14*10^{-5}$	$4,29*10^{-5}$	$6,43*10^{-5}$	$8,57*10^{-5}$	$1,86*10^{-4}$	$9,64*10^{-6}$
	260	$2,69*10^{-5}$	$3,85*10^{-5}$	$7,69*10^{-5}$	$1,15*10^{-4}$	$1,54*10^{-4}$	$3,35*10^{-4}$	$1,73*10^{-5}$	$5,38*10^{-5}$	$7,69*10^{-5}$	$1,54*10^{-4}$	$2,31*10^{-4}$	$3,08*10^{-4}$	$6,69*10^{-4}$	$3,46*10^{-5}$	$8,08*10^{-5}$	$1,15*10^{-4}$	$2,31*10^{-4}$	$3,46*10^{-4}$	$4,62*10^{-4}$	$1,00*10^{-3}$	$5,19*10^{-5}$
	800	$8,75*10^{-6}$	$1,25*10^{-5}$	$2,50*10^{-5}$	$3,75*10^{-5}$	$5,00*10^{-5}$	$1,09*10^{-4}$	$5,63*10^{-6}$	$1,75*10^{-5}$	$2,50*10^{-5}$	$5,00*10^{-5}$	$7,50*10^{-5}$	$1,00*10^{-4}$	$2,18*10^{-4}$	$1,13*10^{-5}$	$2,63*10^{-5}$	$3,75*10^{-5}$	$7,50*10^{-5}$	$1,13*10^{-4}$	$1,50*10^{-4}$	$3,26*10^{-4}$	$1,69*10^{-5}$
	1500	$4,67*10^{-6}$	$6,67*10^{-6}$	$1,33*10^{-5}$	$2,00*10^{-5}$	$2,67*10^{-5}$	$5,80*10^{-5}$	$3,00*10^{-6}$	$9,33*10^{-6}$	$1,33*10^{-5}$	$2,67*10^{-5}$	$4,00*10^{-5}$	$5,33*10^{-5}$	$1,16*10^{-4}$	$6,00*10^{-6}$	$1,40*10^{-5}$	$2,00*10^{-5}$	$4,00*10^{-5}$	$6,00*10^{-5}$	$8,00*10^{-5}$	$1,74*10^{-4}$	$9,00*10^{-6}$
N_{SAT} = 2	550	$1,27*10^{-5}$	$1,82*10^{-5}$	$3,64*10^{-5}$	$5,45*10^{-5}$	$7,27*10^{-5}$	$1,58*10^{-4}$	$8,18*10^{-6}$	$2,55*10^{-5}$	$3,64*10^{-5}$	$7,27*10^{-5}$	$1,09*10^{-4}$	$1,45*10^{-4}$	$3,16*10^{-4}$	$1,64*10^{-5}$	$3,82*10^{-5}$	$5,45*10^{-5}$	$1,09*10^{-4}$	$1,64*10^{-4}$	$2,18*10^{-4}$	$4,75*10^{-4}$	$2,45*10^{-5}$
	450	$1,56*10^{-5}$	$2,22*10^{-5}$	$4,44*10^{-5}$	$6,67*10^{-5}$	$8,89*10^{-5}$	$1,93*10^{-4}$	$1,00*10^{-5}$	$3,11*10^{-5}$	$4,44*10^{-5}$	$8,89*10^{-5}$	$1,33*10^{-4}$	$1,78*10^{-4}$	$3,87*10^{-4}$	$2,00*10^{-5}$	$4,67*10^{-5}$	$6,67*10^{-5}$	$1,33*10^{-4}$	$2,00*10^{-4}$	$2,67*10^{-4}$	$5,80*10^{-4}$	$3,00*10^{-5}$
	175	$4,00*10^{-5}$	$5,71*10^{-5}$	$1,14*10^{-4}$	$1,71*10^{-4}$	$2,29*10^{-4}$	$4,97*10^{-4}$	$2,57*10^{-5}$	$8,00*10^{-5}$	$1,14*10^{-4}$	$2,29*10^{-4}$	$3,43*10^{-4}$	$4,57*10^{-4}$	$9,94*10^{-4}$	$5,14*10^{-5}$	$1,20*10^{-4}$	$1,71*10^{-4}$	$3,43*10^{-4}$	$5,14*10^{-4}$	$6,86*10^{-4}$	$1,49*10^{-3}$	$7,71*10^{-5}$
	35	$2,00*10^{-4}$	$2,86*10^{-4}$	$5,71*10^{-4}$	$8,57*10^{-4}$	$1,14*10^{-3}$	$2,49*10^{-3}$	$1,29*10^{-4}$	$4,00*10^{-4}$	$5,71*10^{-4}$	$1,14*10^{-3}$	$1,71*10^{-3}$	$2,29*10^{-3}$	$4,97*10^{-3}$	$2,57*10^{-4}$	$6,00*10^{-4}$	$8,57*10^{-4}$	$1,71*10^{-3}$	$2,57*10^{-3}$	$3,43*10^{-3}$	$7,46*10^{-3}$	$3,86*10^{-4}$
	700	$1,00*10^{-5}$	$1,43*10^{-5}$	$2,86*10^{-5}$	$4,29*10^{-5}$	$5,71*10^{-5}$	$1,24*10^{-4}$	$6,43*10^{-6}$	$2,00*10^{-5}$	$2,86*10^{-5}$	$5,71*10^{-5}$	$8,57*10^{-5}$	$1,14*10^{-4}$	$2,49*10^{-4}$	$1,29*10^{-5}$	$3,00*10^{-5}$	$4,29*10^{-5}$	$8,57*10^{-5}$	$1,29*10^{-4}$	$1,71*10^{-4}$	$3,73*10^{-4}$	$1,93*10^{-5}$
	130	$5,38*10^{-5}$	$7,69*10^{-5}$	$1,54*10^{-4}$	$2,31*10^{-4}$	$3,08*10^{-4}$	$6,69*10^{-4}$	$3,46*10^{-5}$	$1,08*10^{-4}$	$1,54*10^{-4}$	$3,08*10^{-4}$	$4,62*10^{-4}$	$6,15*10^{-4}$	$1,34*10^{-3}$	$6,92*10^{-5}$	$1,62*10^{-4}$	$2,31*10^{-4}$	$4,62*10^{-4}$	$6,92*10^{-4}$	$9,23*10^{-4}$	$2,01*10^{-3}$	$1,04*10^{-4}$
	400	$1,75*10^{-5}$	$2,50*10^{-5}$	$5,00*10^{-5}$	$7,50*10^{-5}$	$1,00*10^{-4}$	$2,18*10^{-4}$	$1,13*10^{-5}$	$3,50*10^{-5}$	$5,00*10^{-5}$	$1,00*10^{-4}$	$1,50*10^{-4}$	$2,00*10^{-4}$	$4,35*10^{-4}$	$2,25*10^{-5}$	$5,25*10^{-5}$	$7,50*10^{-5}$	$1,50*10^{-4}$	$2,25*10^{-4}$	$3,00*10^{-4}$	$6,53*10^{-4}$	$3,38*10^{-5}$
	750	$9,33*10^{-6}$	$1,33*10^{-5}$	$2,67*10^{-5}$	$4,00*10^{-5}$	$5,33*10^{-5}$	$1,16*10^{-4}$	$6,00*10^{-6}$	$1,87*10^{-5}$	$2,67*10^{-5}$	$5,33*10^{-5}$	$8,00*10^{-5}$	$1,07*10^{-4}$	$2,32*10^{-4}$	$1,20*10^{-5}$	$2,80*10^{-5}$	$4,00*10^{-5}$	$8,00*10^{-5}$	$1,20*10^{-4}$	$1,60*10^{-4}$	$3,48*10^{-4}$	$1,80*10^{-5}$
N_{SAT} = 3	367	$1,91*10^{-5}$	$2,73*10^{-5}$	$5,45*10^{-5}$	$8,18*10^{-5}$	$1,09*10^{-4}$	$2,37*10^{-4}$	$1,23*10^{-5}$	$3,82*10^{-5}$	$5,45*10^{-5}$	$1,09*10^{-4}$	$1,64*10^{-4}$	$2,18*10^{-4}$	$4,75*10^{-4}$	$2,45*10^{-5}$	$5,73*10^{-5}$	$8,18*10^{-5}$	$1,64*10^{-4}$	$2,45*10^{-4}$	$3,27*10^{-4}$	$7,12*10^{-4}$	$3,68*10^{-5}$
	300	$2,33*10^{-5}$	$3,33*10^{-5}$	$6,67*10^{-5}$	$1,00*10^{-4}$	$1,33*10^{-4}$	$2,90*10^{-4}$	$1,50*10^{-5}$	$4,67*10^{-5}$	$6,67*10^{-5}$	$1,33*10^{-4}$	$2,00*10^{-4}$	$2,67*10^{-4}$	$5,80*10^{-4}$	$3,00*10^{-5}$	$7,00*10^{-5}$	$1,00*10^{-4}$	$2,00*10^{-4}$	$3,00*10^{-4}$	$4,00*10^{-4}$	$8,70*10^{-4}$	$4,50*10^{-5}$
	117	$6,00*10^{-5}$	$8,57*10^{-5}$	$1,71*10^{-4}$	$2,57*10^{-4}$	$3,43*10^{-4}$	$7,46*10^{-4}$	$3,86*10^{-5}$	$1,20*10^{-4}$	$1,71*10^{-4}$	$3,43*10^{-4}$	$5,14*10^{-4}$	$6,86*10^{-4}$	$1,49*10^{-3}$	$7,71*10^{-5}$	$1,80*10^{-4}$	$2,57*10^{-4}$	$5,14*10^{-4}$	$7,71*10^{-4}$	$1,03*10^{-3}$	$2,24*10^{-3}$	$1,16*10^{-4}$
	23	$3,00*10^{-4}$	$4,29*10^{-4}$	$8,57*10^{-4}$	$1,29*10^{-3}$	$1,71*10^{-3}$	$3,73*10^{-3}$	$1,93*10^{-4}$	$6,00*10^{-4}$	$8,57*10^{-4}$	$1,71*10^{-3}$	$2,57*10^{-3}$	$3,43*10^{-3}$	$7,46*10^{-3}$	$3,86*10^{-4}$	$9,00*10^{-4}$	$1,29*10^{-3}$	$2,57*10^{-3}$	$3,86*10^{-3}$	$5,14*10^{-3}$	$1,12*10^{-2}$	$5,79*10^{-4}$
	467	$1,50*10^{-5}$	$2,14*10^{-5}$	$4,29*10^{-5}$	$6,43*10^{-5}$	$8,57*10^{-5}$	$1,86*10^{-4}$	$9,64*10^{-6}$	$3,00*10^{-5}$	$4,29*10^{-5}$	$8,57*10^{-5}$	$1,29*10^{-4}$	$1,71*10^{-4}$	$3,73*10^{-4}$	$1,93*10^{-5}$	$4,50*10^{-5}$	$6,43*10^{-5}$	$1,29*10^{-4}$	$1,93*10^{-4}$	$2,57*10^{-4}$	$5,59*10^{-4}$	$2,89*10^{-5}$
	87	$8,08*10^{-5}$	$1,15*10^{-4}$	$2,31*10^{-4}$	$3,46*10^{-4}$	$4,62*10^{-4}$	$1,00*10^{-3}$	$5,19*10^{-5}$	$1,62*10^{-4}$	$2,31*10^{-4}$	$4,62*10^{-4}$	$6,92*10^{-4}$	$9,23*10^{-4}$	$2,01*10^{-3}$	$1,04*10^{-4}$	$2,42*10^{-4}$	$3,46*10^{-4}$	$6,92*10^{-4}$	$1,04*10^{-3}$	$1,38*10^{-3}$	$3,01*10^{-3}$	$1,56*10^{-4}$
	267	$2,63*10^{-5}$	$3,75*10^{-5}$	$7,50*10^{-5}$	$1,13*10^{-4}$	$1,50*10^{-4}$	$3,26*10^{-4}$	$1,69*10^{-5}$	$5,25*10^{-5}$	$7,50*10^{-5}$	$1,50*10^{-4}$	$2,25*10^{-4}$	$3,00*10^{-4}$	$6,53*10^{-4}$	$3,38*10^{-5}$	$7,88*10^{-5}$	$1,13*10^{-4}$	$2,25*10^{-4}$	$3,38*10^{-4}$	$4,50*10^{-4}$	$9,79*10^{-4}$	$5,06*10^{-5}$
	500	$1,40*10^{-5}$	$2,00*10^{-5}$	$4,00*10^{-5}$	$6,00*10^{-5}$	$8,00*10^{-5}$	$1,74*10^{-4}$	$9,00*10^{-6}$	$2,80*10^{-5}$	$4,00*10^{-5}$	$8,00*10^{-5}$	$1,20*10^{-4}$	$1,60*10^{-4}$	$3,48*10^{-4}$	$1,80*10^{-5}$	$4,20*10^{-5}$	$6,00*10^{-5}$	$1,20*10^{-4}$	$1,80*10^{-4}$	$2,40*10^{-4}$	$5,22*10^{-4}$	$2,70*10^{-5}$

5. Basic values of the EPPM optimum design parameters compare to parameters of real propulsion systems under the terms:

$$a_0 > a_0^{(0)opt},\ c > c_{opt}^{(0)},\ R > T_{m\,fix}\,. \tag{18}$$

Values of parameters of propulsion systems, unsatisfactory to the conditions (18) are rejecting, and thus the variety of alternatives is narrowed.

6. The EPPM design parameters of the first approximation are defining (accounting disturbing factor - zonal harmonic gravity model J2).

7. The received values compare to the variety of real engines parameters again. The variety is narrowed further.

8. Depending on average height of a working orbit the maximum level of the disturbing acceleration (an average for a revolution) from influence of the top atmosphere[6] is estimating (the disturbing factor 2, Fig. 3). Values of the EPPM design parameters are specified.

9. Procedure of comparison of optimum design parameters of the second approximation with a set of design parameters of real propulsion systems repeats. The variety of alternatives is narrowed again.

10. Process of more precise definition of the EPPM optimum design parameters can be continued taking into account other disturbing factors. End of process becomes it the application of more precise models of satellite-inspector movement doesn't lead to noticeable reduction of a variety of alternatives. Application of this algorithm based on method of the morphological analysis allowed to reduce consistently quantity of options from 504 to 210. Variants which do not satisfy conditions (18) are shaded in the table 4 with a dark color.

11. For the values of real design parameters of propulsion systems (subset of 210 variants) found on the last iteration, the relative payload μ (1) is defining. Incompatible options, for which $\mu < 0$, are excluding from the variety.

12. The EPPM universal design parameters for range of maneuvers Z are defining.

B. Design parameters universal for the range of dynamic operations

The vector of parameters $\overline{\overline{p'}}$ is universal for the range of dynamic maneuvers Z, if the satellite with parameters $\overline{\overline{p'}}$ can execute any dynamic operation from Z; and the maximum degree of nonoptimality $\rho(z,\overline{p'})$ reaches on Z the minimum value at $\overline{p'} = \overline{\overline{p'}}$. Nonoptimality degree is a loss measure in criterion of efficiency when replacing vector of optimum design parameters on some other[7]:

$$\rho(z,\overline{p'}) = \frac{\max\limits_{p'\in P} \mu(\overline{p'}, V_X(z))}{\mu(\overline{\overline{p'}}, V_X(z))}. \tag{19}$$

Measure of nonoptimality of the design decision is $F = \min\limits_{p'\in P} \max\limits_{z\in Z} \rho(z,\overline{p'})$. Vector $\overline{\overline{p'}} = \min\limits_{p'\in P\to} \max\limits_{z\in Z} \rho(z,\overline{p'})$ is a vector of design parameters universal for a range of dynamic operations Z[8].

IV. Conceptual design shaping of satellite-inspector

The EPPM universal parameters received by using the synthesis algorithm are $\overline{\overline{p'}}(\overline{\overline{a_0}},\overline{c}) = (8.00\cdot 10^{-5} m/s^2; 12.8 km/s)$. These universal parameters identify: type (“SPD-100”) and number of thrusters in EPPM (8 – by two for each direction of thrust); mass of satellite-inspector (~750 kg) and their quantity in payload zone (two); type of carrier rocket (“Soyuz 2.1v”) and also an expanded set of the EPPM design parameters: power of EPPM (1000W); total thrust of EPPM (60 mN); solar panel square necessary for work EPPM (3 m^2); mass of a working fluid (32.1 kg of xenon that corresponds to $V_{X\max} = 432.94$ m/s); mass of working fluid (kg), resource of the block of thrusters (2500 h); motor time for maneuver realization (1902.2 h). These parameters allow to determine in a first approach conceptual design shape of the satellite-inspector (Fig. 5).

Figure 5. Conceptual design shape of the satellite-inspector.

V. Conclusion

The results of work shown in this paper allow to make a conclusion about possibility of creating a satellite-inspector, equipped with electric propulsion power module. The synthesis algorithm of the satellite-inspector's principal design parameters, containing the iterative scheme of determination of electric propulsion power module parameters, was developed and realized. This algorithm allows not only to find the optimum basic design parameters of a module from a variety of different variants, but also to generate conceptual design shape of the satellite-inspector.

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