Electrical Propulsion System in the Control of Earth Observing Spacecraft

IEPC-2015-34／ISTS-2015-b-34

Presented at Joint Conference of 30th International Symposium on Space Technology and Science
34th International Electric Propulsion Conference and 6th Nano-satellite Symposium
Kobe-Hyogo, Japan,
July 4 - 10, 2015
Vladimir P. Khodnenko[1]
'VNIIEM'Corporation' JSC Address: ROB496, Glavpochtamt, Moscow, 101000, Russia

Abstract: Electric rocket thrusters (ERT) are used in the systems for controlling the orbit of the Earth observing spacecraft (EOS) since only they can secure the required orbit's parameters with high accuracy (initial correction) and maintain their stability within the long-term lifetime (nominal correction).

In addition, for geosynchronous and high orbit Earth remote sensing satellites, the electric rocket thrusters can be used for generating the stable controlling torques for unloading mechanical actuators of the stabilization system

Nomenclature

H_{av} - average orbital altitude;
R_{Er} - Earth radius;
i - orbit inclination;
T_{Ω} - draconic orbital period;
Ω-$\alpha_{☉}$ - difference between orbital right ascensions and the mean Sun;
F – thrust;
I_{si} - specific impulse;
$\delta F/F$ - thrust instability;
I_{Σ} - total specific thrust;
M_{EPS} - EPS mass;
K_F - thrust instability factor;
K_y - knowledge accuracy of the flywheel inertia moment;
τ_c - initial correction duration;
Ψ - phase distance;
$\bar{E}$ - Laplace vector;
N_c - number of corrections a day.

[1] Chief Scientific Officer, e-mail:m6b@mail.ru

I. Introduction

The Earth, the Earth's atmosphere and the near-Earth space environment are objects for global geophysical monitoring, which is performed by the Earth observing spacecraft (EOS). The objectives of this monitoring are as follows:

- Earth natural resources exploration (ENRE);
- hydrometeorological support (HS);
- environmental monitoring (EM).

The electrical propulsion system's (EPS) placed onboard the Earth remote sensing satellites should perform the following operations:

- initial correction after spacecraft putting into the orbit;
- to maintain orbital parameters during nominal corrections;
- to direct the flight path to the monitoring area and to keep it within the given range;
- to vary phase disposition of spacecrafts that form the spacecrafts' constellation;
- to generate the controlling torques in the navigation systems of high orbit spacecraft.

II. Orbit Characteristics

The orbits of the Earth observing spacecraft should be characterized by minimum possible eccentricity and by stable geographical position of the perigee argument. Such orbits are called the stable orbits.

To obtain high quality remote sensing data, it is very important to know spacecraft (SC) orbit position with respect to the Sun.

Identity conditions of solar lighting can be secured if an angular position of SC orbit plane is constant with respect to the Sun, i.e. for so-called the sun-synchronized orbits (SSO) (Fig. 1).

The inclination of the sun-synchronized orbit can be determined quite precisely by the formula

I_{CCO} = arcos [-0,0986 $(1 + H_{cp}/R_3)^{3,5}$, where

H_{av} – average orbit altitude;

R_{er} – Earth radius.

As a rule the sun-synchronized orbits should be repeatable, i.e. they should pass over the same sub-satellite points on the Earth at the specified time (integer number of days).

These special characteristics of the EOS orbits (i.e. repeatability, Sun synchrony, stability) are generally related to the mean-orbit spacecraft (H = 600 – 1500 km).

The geosynchronous Earth orbit ($H \approx 36000$ km, $\imath = 0$, $T_\Omega = 24$ hours) is used for communication SC as well as for hydrometeorological SC which belong to EOS class.

The daily ($T_\Omega = 24$ hours) sub-polar ($\approx 90^o$) orbits are considered for promising SC designed for hydrometeorological support (SC HS). These orbits are characterized by important advantages combining globality (mean-orbit SC) and continuity (geo-synchronized SC) of atmosphere and the Earth surface monitoring. These orbits are characterized by daily repeatability.

To increase the frequency of the Earth global view by EOS equipped with information and measuring narrow-band facilities, the relative disposition of several spacecraft whose orbiting in the same plane should be maintained stable (lighting conditions should also be optimal). Therefore, it is necessary to maintain (with certain accuracy) equal rotation periods and orbit inclinations of these SC.

The modern carrier rockets (CR) put the SC into the orbit with considerable margins. Obviously, the initial correction is necessary to obtain the required orbit parameters.

Howeve,r the initial orbit parameters (after correction) are stable for a relatively shot period of time. It is known that SC orbit elements vary continuously due to different perturbing factors. Their interaction character and degree depend on size, shape and orbit plane position.

The effect of atmosphere retarding depends strongly on solar activity.

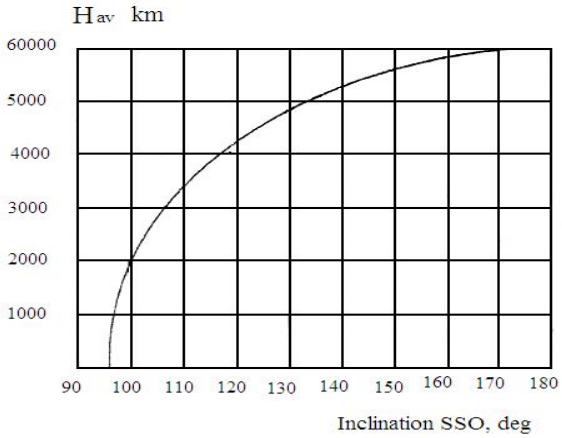

Image a– general view

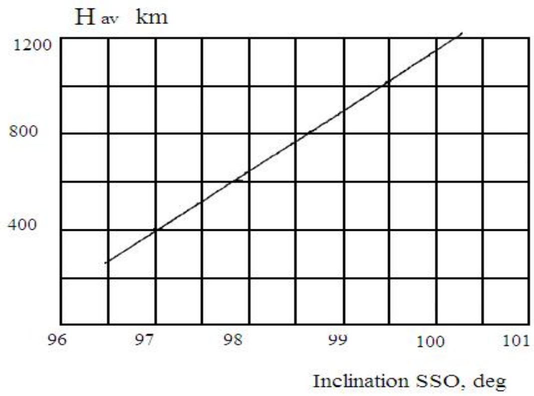

Image b– in EOS range

Fig. 1 The relationship between solar-synchronized orbit inclination and orbit altitude:

For example, when solar activity Ŧ ranges from 75 x 10^{-22} W/m^{2}Hz ("weak" Sun) up to 250 x 10^{-22} W/m^{2}Hz ("strong" Sun) the average fall of the rotation period per revolution δT_{Ω} ranges from 0.27 x 10^{-4} to 38.5 x 10^{-4} s/revolution, i.e. it is two orders greater.

The inclination of solar-synchronized orbit decreases due to the Sun gravitation by of 3,5 ang.min per year (for H ≈ 650 km), and in combination with initial errors it results in significant disagreement between the right orbit ascension value and the mean Sun value $\Omega\text{-}\alpha_{\Theta}$. Since the Earth's gravitational field is not spherical, variations in the Laplace vector and ascending node longitude are long-term periodical. Since the EOS lifetime is increased up to 7 - 10 years, it is possible to maintain the orbit elements within the acceptance limits only by executing periodical "support corrections".

III. Electrical Propulsion System's in the correction and navigation system's of earth observing spacecraft

For remote sensing the Earth remote sensing satellites should be maneuverable and for this purpose they are equipped by correcting power system's (CPS) whose total specific thrust makes it possible to maintain the given RES SC orbit parameters within their lifetime (5-7 years and more).

The following types of low-thrust engines are considered for real control systems (CS) of the RES SC as most tested under space conditions:

- termocatalytic hydrazine thrusters (THT);
- electric heated thrusters (EHT);
- stationary plasma thrusters (SPT);

Basic characteristics of the mentioned propulsions are presented in Table 1.

Table 1.

Characteristics	THT	EHT	SPT
Thrust, N	0,1 - 30	0,05 - 0,3	0,02 - 1
Thrust instability, %	Up to 10	Up to 10	Up to 5
Specific impulse, Ns/kg	1800 - 2300	2300 - 2800	$(1,1 -2)x10^4$
Thrust cost, W/mN	0,2 - 0,5	2,5 - 3,5	12 - 15
Thrust afteraction pulse, Ns	0,03*	(0,03 - 0,05)*	$(2-3)x10^{-3}$***

Note: Experimental data:* - for propulsions with the thrust of 0,1 N;
** - for propulsions with the thrust of 0,15 – 0,2 N;
*** - for propulsions with the thrust of 0,03 – 0,05 N.

Requirements to the space power plants jointly with structure-functional and design features of the EOS show that it is reasonable to use the stationary plasma thrusters (SPT) for EOS.

Stationary plasma thruster (SPT) is the most promising type of the electric rocket thrusters [2]. In addition to high specific impulse SPT has rather high efficiency ≥ 50 – 60% under relatively low discharge voltages U_p = 150 – 300V and small magnetic fields of the accelerating system Hr = $(1,2\text{-}1,6)\cdot 10^3$ A/m (150-200 Oe). Thrust control is possible for the SPT and it can operate properly both with low $(1\text{-}3)\cdot 10^{-2}$N and at relatively high thrust values ≥ (0.2-0.5) N, it is characterized by high reliability and high lifetime (≥10000 hours). Xenon used as a working medium is environmentally safe and makes it possible to use simple and quite reliable fuel feeding system.

For the SPT the time for running on to the steady motion of operation is not higher than 0.1 s and the aftereffect impulse is $(1\text{-}3)\cdot 10^{-3}$N·s under thrust level of (0.03-0.05) N. In addition the SPT is characterized by good reproducibility (about several percents).

It is reasonable to use the SPT in the high precision navigation systems of high-orbit Earth remote sensing satellites ($R_{orb} \geq 15\cdot 10^3$ km) for generating stable controlling torques of $(1,5\text{-}3)\cdot 10^{-2}$ Nm. They are a good addition to fly-wheel engines, if they are used for utilizing the kinetic torque since their thrust is low (≤ 0,05) N and stable (≤ 5%).

For choosing parameters of electro-rocket power plant and of the correction program it is reasonable to use a minimal duration criterion for initial and next corrections as an optimization parameter. I.e. it is necessary to minimize the period for putting EOS into exploitation and to minimize target information loss caused by corrections within the period of spacecraft active operation, if the requirements for mass limitations are met.

If the electrical propulsion system's is used both for correcting the orbit and for controlling SC motion with respect to the center of mass as it is for high orbit of Earth observing spacecraft, the problem of minimizing the total mass of the electrical propulsion system's becomes critical, if requirements for energy limitation and for navigation accuracy are met.

The electrical propulsion system's specifies the new requirements for the spacecraft in addition to mass-energy requirements. These new requirements are related to location accuracy especially for navigation task. The EPS should be compatible with the SC electro-radio-technical systems. The EPS plasma jets should be beyond of optic, solar arrays and equipment fields of view.

Therefore, type and characteristics of EPS are critical for design-layout and structural of the EOS.

Let's consider the ways of EPS orbit correction with the help of low thrust EPS for orbital parameters obtaining and maintaining. This is important for the proper spatio-temporal indicators of remote control target information.

Methods of EOS orbit correction with the help of corrector low-thrust EPS are described below. The EPS is used to obtain and maintain the specified orbital characteristics, which provide required space-time data of the remote sensing.

Repeatability properties are secured by correcting the draconic rotation period T_Ω. Orbit stability is maintained by separate and simultaneous correction of the rotation period T_Ω and the Laplace vector $\bar{E}$ (inside-plane correction). The Sun synchrony and phase position are secured by correcting the orbit plane inclination, as well as by correcting the rotation period while executing the phase correction.

It is possible to secure the specified rotation period without changing other parameters by igniting one of two thrusters placed onboard the spacecraft along axes ± X shall for the time period equal to an integer number of revolutions.

The Laplace vector is corrected by alternating activation of both thrusters at the same revolution and symmetrical orbit segments.

Segments are placed in the opposite hodograph's areas of alternating component of Laplace vector. If the rotation period and Laplace vector are corrected simultaneously, non equal orbit's segments are selected for thruster's operation and the highest is the segment that corresponds to the required variation in rotation period.

The phase is corrected by alternating activation of each thruster placed along axes ± X for the period equal to the integer number of revolutions. Between activations there is an interval needed for separating the spacecrafts at required angle due to the difference in rotation period.

Orbit inclination is corrected by thrusters placed along axes ± Y in the direction normal to orbit plane. The thrusters are ignited at the segments symmetrical with respect to ascending and descending nodes of the orbit.

The basic parameters of the EPS and of the orbit correction are as follows:

- nominal value (magnitude) of the thrust vector F;
- knowledge accuracy (permissible instability) of the thrust value $\frac{\delta F}{F}$;
- total specific thrust I_Σ required for initial and next corrections within SC lifetime;
- correction program, i.e. duration and periodicity of the space power plant activations within correction cycles.

Specific impulse $I_{sp,}$ thrust cost γ_{en} and "dry weight" of the EPS are determined by the type of the chosen electro-rocket thruster.

For specifying EPS parameters during design process the general and special limitations are set. General limitations area as follows:

- permissible total mass of the correcting EPS $M_{EPS} \leq M_{per}$ connected directly with the total specific thrust I_Σ and with the type of the EPT (specific pulse I_{sp});
- maximum electric power consumed by the EPS.

Limitations for EPS's sequence diagram are specified in the form of a relationship between permissible thruster's time of operation on the revolution τ_F and the rotation period T_Ω - $k = \frac{\tau_F}{T_\Omega}$ and also as a relationship between the number of revolutions where corrections are performed during a day n_k and the daily number of revolutions N_c.

For some cases, it is reasonable to use the EPS for several purposes (for example, in the navigation system to unload the fly-wheel engines). Therefore, the new limitations for EPS parameter should be considered.

An evaluation of navigation accuracy of the high orbit EOS shows that the maximum SC angular rate of rotation round the mass center at the unloading moment max φ(t) is proportional to the EPS thrust F and K_F+K_y where $K_F = \frac{\sigma F}{F}$ is thrust instability factor (spread) and K_y is the knowledge accuracy of the fly-wheel inertia moment. To secure the stabilization accuracy of geosynchronized EOS`s the thrust should be limited to $(5\text{-}7)^{-2}$N and $K_F \leq 10\%$. Such low thrust instability level is obtained only for the ERT and for low-thrust hydrazine engines, it is $\geq 20\%$.

Limitation of thrust instability $\frac{\delta F}{F}$ is needed for guarantied communication with the SC after a long correction period. Requirements and conditions effecting to correction parameters and ERS parameters are presented in Table 2.

Table 2.

Basic factors	Effected correction parameter	Additional conditions on which parameter depends
Requirements for minimizing the initial correction time $\tau_к$ ($\leq$ 20-30 days)	Nominal thrust	values of orbit elements which are corrected, energy limitations for the EPS sequence diagram, orbit radio checking-up (ORT) during correction
Requirements for the knowledge accuracy for the time of SC crossing the fixed direction under correction Δt ($\leq$ 30-60 s)	Thrust value and permissible thrust instability F;	values of orbit elements which are corrected, energy limitations for the EPS sequence diagram, orbit radio checking-up (ORT) during correction
Requirements for path keeping in the specified band ΔL ($\pm$10 km)	Periodicity of keeping correction Nк, value of corrected rotation period ΔTк, required specific thrust I	Reduction of the rotation period per revolution δTa; measurement accuracy of period δTи; relative error of atmosphere knowledge q and longitude measurement $\delta\Omega$, thrust instability
Requirements for minimizing the latitudinal stabilization of altitude spread	Correction periodicity, Laplace vector value Δ Eк, which is corrected, required specific thrust I	Permissible altitude variation δH, nominal value of orbit elements
Requirements for keeping the given phase position of the spacecraft	Periodicity of phase correction, values of inclination Δi and phase $\Delta\varphi$, which are corrected, required specific thrust I	Permissible variation of phase interval $\delta\varphi$ between SCs, value of inclination variation of the solar-synchronized orbit

The Table contains also the following limitations:

Nк – periodicity of keeping correction, i.e. the number of revolutions between two next corrections of path keeping within specified band ΔL;

δTa – reduction of rotation period per revolution depending on solar activity;

$\delta T_и$ – maximum computation error for rotation period according to the given trajectory measurements (accuracy of period measurement). This error depends on solar activity.
q – uncertainty knowledge factor for atmosphere in different directions of the SC path.

The problem of the regular EOS orbit correction which secures the repeatability, stability and also partially keeping the SC phase position should by solved by designers. Specific points of this problem are presented in the Table 3.

Table 3.

1. Factors requiring optimization	1. Application of low thrust ERTs 2. putting on errors T_Ω, e, w
2. Criterion of optimization and limitations	Minimal duration of the initial correction under the prescribed thrust value F, available total specific thrust I_Σ and also power of PPU which ensues the ERPP operation on correction revolutions $\tau к = к T_\Omega$ ($к \le 1$)
3. Controlling parameter	Location and duration of EPS segments of operation (program)
4. Boundary functionals	Mod $\Delta\overline{E} = \sqrt{(l-lcm)^2 + (h+hcm)^2} \to 0$ Arg $\Delta\overline{E} = Arctg\frac{h-hcm}{l-lcm} \to Arctg\frac{hcm}{lcm}$ $T_{\Omega 0} - T_\Omega \to 0$
5. Principles of optimal correction	1. Transition from e and Wк to Laplace vector $\overline{E} = \overline{E}cm + \Delta\overline{E}$ 2. Simultaneous correction execution and completion according to $\overline{E}$ and T_Ω

The table 4 depicts equations and relationships suitable for choosing and estimating correction parameters of the EPS for EOS. Information presented in [3] are used for deriving the presented relationships.

Table 4.

Correction parameters	Relationships
Schemes for separate correction of rotation period and Laplace vector	Correction of rotation on the whole revolution Laplace vector correction in two opposite sectors
The relationship between simultaneous correction scheme $T\Omega$ and E Dependence of the $T\Omega$ and E and the corrected parameter	$\frac{2\lvert\Delta T\kappa\rvert}{3T_{\Omega}\lvert\Delta E\kappa\rvert}\frac{\sin k\pi}{k\pi}\begin{cases}>1 & A\\ <1 & Б\\ =1 & B\end{cases}$
Rotation period variation per one revolution under space power plant operation	$\frac{dT_{\Omega}}{dN}=\frac{3T_0^2F}{\pi mv}\begin{cases}\Delta U & \textit{in the case A}\\ U_2-U_1+\Delta U & \textit{in the case B}\\ U_2-U_1 & \textit{in the case C}\end{cases}$
Variation of Laplace vector magnitude per one orbital period under space power plant operation	$\frac{dModE}{dN}=\frac{4T_0F}{\pi mv}\begin{cases}\sin\frac{\kappa\pi}{2}\cos\frac{\Delta U}{2} & \textit{в случае A}\\ \cos\frac{\kappa\pi}{2}\sin\frac{\Delta U}{2} & \textit{в случае Б}\\ \sin\frac{U_2-U_1}{2} & \textit{в случае В}\end{cases}$
Typical sizes for simultaneous correction segments $T\Omega$ and E	$\frac{\Delta U}{2}=\begin{cases}\frac{2\lvert\Delta T_k\rvert}{3T_{\Omega}\lvert\Delta E_k\rvert}\sin\frac{\kappa\pi}{2}\cos\frac{\Delta U}{2} & \textit{in the case A}\\ \arcsin\frac{3T_{\Omega}\lvert\Delta E_{\kappa}\rvert\kappa\pi}{\lvert\Delta T_{\kappa}\rvert\cos\frac{\kappa\pi}{2}} & \textit{in the case B}\end{cases}$ $U_2-U_1=\kappa\pi-\Delta U$

Required EPS impulse during simultaneous correction $T\Omega$ and E	$I^{p} = \dfrac{mv}{2}\Delta E_{\kappa}\begin{cases}\dfrac{\kappa\pi/2}{\sin\kappa\pi/2\,\cos\Delta U/2} & in\ the\ case\ A\\ \dfrac{\kappa\pi/2}{\cos\kappa\pi/2\,\sin\Delta U/2} & in\ the\ case\ B\\ \dfrac{\kappa\pi}{\sin\kappa\pi} & in\ the\ case\ C\end{cases}$
Required EPS impulse during separate correction $T\Omega$ and E	$I^{p} = mv\left(\dfrac{1}{3}\dfrac{\lvert\Delta T_{\kappa}\rvert}{T_{\Omega}} + \dfrac{1}{2}\lvert\Delta E_{\kappa}\rvert\dfrac{\frac{U_2-U_1}{2}}{\sin\frac{U_2-U_1}{2}}\right) = I_T + I_E$
EPS hrust value needed to start the initial correction in the specified point of time τκ,day	$F = \begin{cases}\dfrac{I^{*}\left(\tau_{Tu}+\tau_{\text{рбз}}\right)}{\tau_u T_{\Omega} n_{\kappa}\tau_{Tu}}\ under\ simmult.correction\ T_{\Omega}\ and\ E\\ \dfrac{I_E}{\kappa} = \dfrac{I_T N\left(\tau_u+\tau_{\text{рбз}}\right)}{n_{\kappa}\tau_{\kappa}\tau_{Tu}T_{\Omega}N}\ under\ contineous.correction\ T_{\Omega}\end{cases}$
Permissible thrust instability by considering the specified forecast error during correction (Δt)	$\dfrac{\partial F}{F} = \begin{cases}\dfrac{2\Delta t I^{*}}{FT_{\Omega}\Delta T_{\kappa}\left(\tau_{Tu}+\tau_{\text{рбз}}\right)^{2}N^{2}}\\ \dfrac{2\Delta t m v}{3T^{2}F\left[n_{\kappa}^{2}\left(\tau_{Tu}+\tau_{\text{рбз}}\right)^{2}+2\left(N-n_{\kappa}\right)\right]}\end{cases}$ −‖−‖−‖−
Time period (presented in revolutions) between two next corrections of keeping the path within ΔL	$N_{\kappa} = 2\left[\dfrac{\Delta L(1-q_1)}{w_3 R_0\lvert\partial T_a\rvert}(1-2\delta\Omega)\dfrac{1-2\,\delta F/F}{1-q_2}\right]^{1/2} - \dfrac{4\Delta T_u}{\delta T_a}\Big(1-2$
Inclination variation per one revolution under EPS operation in two orbit segments which are opposite with respect to ascending and descending orbit nodes	$\Delta i = 2\dfrac{FT_{\Omega}}{\pi m v}\sin\dfrac{\Delta U_{\Sigma}}{4}$ $\Delta U_{i\Sigma} = 2(U_2 - U_1)$

EPS impulse required for correcting the inclination	$I_i = \dfrac{mv\Delta i_K \dfrac{\Delta U_{i\Sigma}}{4}}{\sin\dfrac{\Delta U_{i\Sigma}}{4}}$; $\Delta i_K = \Delta i_{nK}$
Corrected period of revolution for varying the phase by Δφ under waiting of τφ [days]	$\Delta T_{K\varphi} = \dfrac{T_{\Omega H}\,\Delta\varphi + 2\pi N\tau_\varphi\left(1+\dfrac{\tau_\varphi N}{2}\right)\left\|\delta T_a\right\|}{2\pi N\tau_\varphi - \Delta\varphi}$
EPS impulse required for correcting the phase by Δφ under waiting time of τφ [days]	$I_\varphi = \dfrac{2}{3}mv\dfrac{\Delta T_\kappa\varphi}{T_{\Omega H}}$
Operation time of each EPS under phase correction by Δφ and waiting time t_φ	$\tau_{\partial y} = \dfrac{mv}{3T_\Omega\left\|F\right\|}\dfrac{T_{\Omega H}\,\Delta\varphi+2\pi\tau_\varphi N\left(1+\dfrac{\tau_\varphi N}{2}\right)\left\|\delta T_a\right\|}{2\pi\tau_\varphi N-\Delta\varphi}$
Scheme of phase correction with delay	……………………τ_φ………………………

The calculations made on the basis of the information presented in Table 4 make it possible to generate a graph (Fig. 2) that illustrates an approach for choosing the correction design parameters (nominal thrust value F, thrust instability $\frac{\delta F}{F}$, periodicity τ for keeping the path within ΔL.

Figure 2a depicts the required distribution function of thrust impulse I_Σ for initial correction. Optimal (simultaneous) and consequent corrections are marked by the solid lines and by the dotted lines, respectively.

Figure 2b depicts the relationship between the thrust needed for damping the putting-on error that corresponds to the required impulses presented in the abscissa axis for the period of correction not higher than 30 days and under correction number per day n_k=7, 8 и 15.

Figure 2c presents the relationships between permissible thrust spread and its value under forecast error not higher than 1 min for the forecast period of 9 days.

Figure. 3d presents the relationship between the periodicity of path keeping correction ΔL±10 km and thrust stability under different solar activities.
Scheme for selecting correction parameters for a concrete case is shown with arrows.

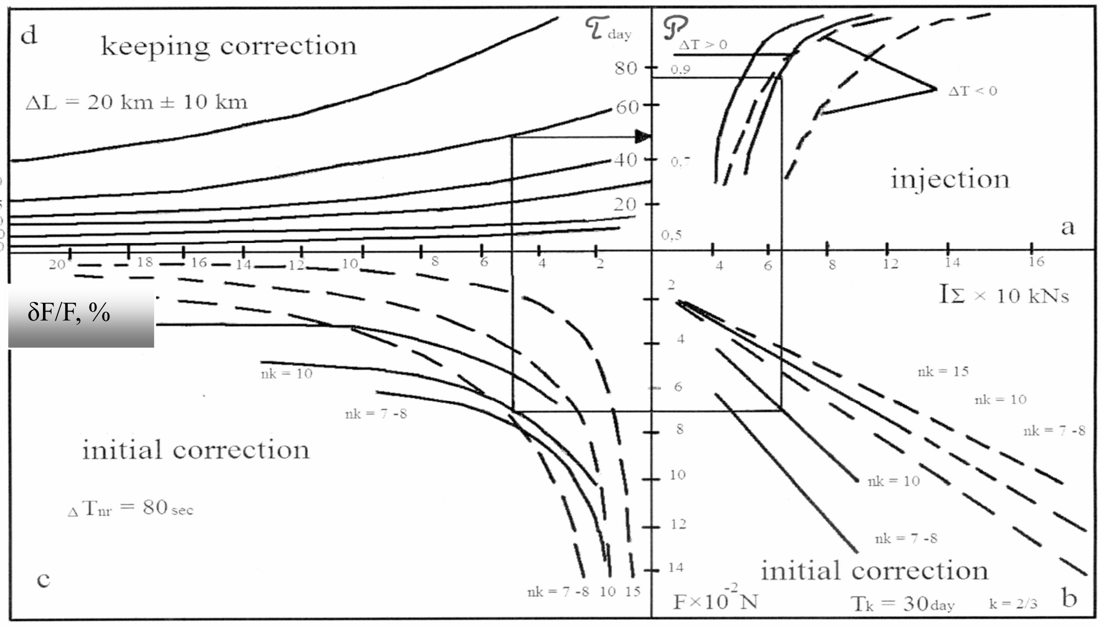

Fig. 2. Selection of the correction parameters

IV. Conclusions

1. The correlations of orbital parameters were generalized. They are able to meet the following requirements:
- ground track repeatability that secures the given periodicity of global (if only one SC operates in the constellation) and local Earth and atmosphere monitoring;
- ellipticity minimizing by the account of altitude and perigee argument consistency of that makes it possible to keep the geometric resolution of the target information;
- consistency of SC phase attitude in orbital constellation consisting of several SCs that secures stability in ballistic configuration of the SC constellation and minimization of the mission data loss.
- Sun synchrony, which provides good radiometric interpretation of the EPS mission data during ground data processing.
2. The usage of low-thrust EPS for initial correction and keeping of orbital parameters at the required achieved level are approved. The principles and methods of EOS correction with the help of EPS are presented.
3. Simplified equations for practical engineering design were derived on the basis of mass and energy limitations to EPS parameters and operation programs; of the basic ballistic flight equations and of the specific approach to the EPS orbit correction. This enables one to complete tasks of correction optimization at the project stage as well as SC orbital control. The examples of calculation of EPS orbital parameters were presented. They were formulated

using a special method developed and tested during the operation of the mean EOS of the "Resurs-01" type.

References

1. P.E. Eliasberg and et.al. Preliminary recommendations for evaluation accuracy of the Project "Resource-0" orbits.- M. Space Research Institute of the Academy of sciences of the USSR, 1979.
2. L.A. Artsimovich, K.N. Kozubskiy, A.I. Morozov, V.P. Khodnenko [and others]. Development and tests on "Meteor" AES of the fixed plasma engine (FPE). Space research. 1974. vol.XII.-Issue 3.-pp. 451-468.
3. P.E. Eliasberg. An introduction to the theory of artificial Earth satellite flight. Publishing house "Nauka". Moscow, 1965.