https://doi.org/10.15407/knit2023.03.034
UDC: 629.764

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ESTIMATION OF THE POSSIBILITY OF USING ELECTRIC PROPULSION SYSTEMS FOR LARGE-SIZED ORBITAL DEBRIS POST-MISSION DISPOSAL

The steady increase in the amount of large-sized orbital debris represents a substantial threat to satellite missions. Currently, many methods of cleaning near-Earth space with the use of various means based on various physical principles are considered. Out of them all, the active method using a rocket propulsion system is the most commonly implemented. Considering the high specific impulse, small size, and mass of electric propulsion systems, they are a particularly attractive choice as means of post-mission disposal. Despite their advantages, such systems have certain peculiarities that need to be considered in the process of designing and implementing modern post-mission disposal means. These peculiarities include the maximum time of a single firing of the electric propulsion system, the maximum time of the battery charging, and the time of operation of the control system.

The purpose of this work is the determination the capabilities of the modern Hall thrusters ST-25 and ST-40 developed by Space Electric Thruster Systems in solving the problem of post-mission disposal of large-sized orbital debris from low-Earth orbits taking into account the limitations on the power supply system. To achieve this goal, methods of analysis, synthesis, comparison, and computer simulation were used. In the course of the carried-out research, the following problems were solved. A scheme for post-mission disposal of large-sized orbital debris from low-Earth orbit was developed with consideration of the use of an electric propulsion system. The dependence was determined of the minimum delta-v increment required for post-mission disposal of an object within 25 years on the initial altitude of the orbit and the ballistic coefficient of the orbital debris. The upper boundary of the combinations of masses of orbital debris, the altitude of the initial orbit, and the ballistic coefficient were determined, for which post-mission disposal from near-Earth orbits is possible with the use of electric propulsion systems.

The obtained results can be used in solving problems of the development of modern means of active post-mission disposal of orbital debris with the use of Hall thrusters developed by Space Electric Thruster Systems.

Keywords: *orbital debris, active post-mission disposal, electric propulsion system.*

Цитування: Golubek A. V., Dron M. M., Petrenko O. M. Estimation of the possibility of using electric propulsion systems for large-sized orbital debris post-mission disposal. *Space Science and Technology*. 2023. **29**, № 3 (142). P. 34—46. https://doi.org/10.15407/knit2023.03.034

INTRODUCTION

The ecological state of near-Earth space is characterized by a huge number of space objects. One of the ways to slow down the growth of the number of such orbital debris objects (ODO) in low-Earth orbits (LEO) is the post-mission disposal of the upper stages of launch vehicles and spacecrafts into the dense layers of the atmosphere, where they will decay.

Currently, various methods of disposal of space objects out of their orbits are considered [20, 23, 32, 34]. These can be active post-mission disposal with the use of jet propulsion systems [4, 8, 12—16, 22, 25, 29, 31] or autophage systems [37, 38], electrodynamic tether systems [9, 19], ion beam shepherd systems [1, 18], magnetodynamic systems [17, 35], sailing structures [3, 5, 24, 26, 30], and laser systems [33, 36]. The use of combined methods is considered in [2, 6, 10, 11, 21].

One of the most promising methods is the use of a jet propulsion system. The main advantage of this approach is to ensure the disposal of a space object within a given time frame and at a given re-entry point in near-Earth space. However, this approach has a significant drawback related to the need for fuel components on board a space object. This leads to a significant increase in the mass-dimensional characteristics of the propulsion system.

One of the most promising types of propulsion systems is the electric propulsion system (EPS), which has a number of significant advantages compared to other types:

• the specific impulse (1500...1600 sec.) significantly exceeding that of propulsion systems of other types;

• operation with the use of electrical energy, which can be obtained directly in space from solar panels;

• possibility of long service life (thousands of hours);

• possibility of multiple on/off switching (thousands of cycles).

Along with these advantages, EPSs have a number of features that must be taken into account when using them:

• the need for sufficient electric power on board (after the degradation of solar panels at the end of the period of active existence of the space vehicle);

• small thrust value (from several mN to tens of mN). If necessary, to increase the thrust of the EPS, one can use a bunch of 2...3 engines (if there is sufficient electric power on board).

The complexity of the design of the propulsion system, which, in addition to the electric rocket engine, includes a system for storing and supplying the propellant and the energy conversion and control system necessary for the functioning of the EPS.

LITERATURE REVIEW

As it was noted earlier, of all multitude of methods of post-mission disposal of ODOs [20, 23, 32, 34], up to date, active deorbiting with the use of a rocket propulsion system is predominantly used.

Authors of [31] were among those who first noticed the possibility to use EPS in the problem of post-mission disposal of ODOs. In this work, the problem was formulated, the ways of its solution were determined, and a comparative analysis of energy expenses on implementation of active deorbiting with the use of chemical propulsion systems and the EPS was made.

For a fast evaluation of the preliminary requirements for the accomplishment of typical debris deorbiting with the use of the EPS, a new algorithm is proposed [29]. It is based on Edelbaum's method of approximation, which takes into account eclipses, atmospheric drag, and the oblateness of the Earth.

A concept of active post-mission disposal of cubesat-sized ODOs was developed in [13]. The law for controlling the thrust vector during continuous operation of the EPS, ensuring the minimum time of disposal, was defined more precisely. A simulation for an example of the Kompsat-1 satellite and Pegasus launch vehicle was carried out. A possibility of disposal within two years was shown.

The work [15] is devoted to the development of a method of active post-mission disposal of a coplanar constellation of ODOs with the use of the EPS. A scheme was developed for disposal that ensures the minimal time of disposal of the constellation in the presence of limitations on the risk of collisions within the constellation. A quasi-optimal minimum time control law for controlling the thrust vector was developed. Simulation for an example of the OneWeb constellation was carried out, the results of which confirmed the workability of the proposed solutions.

In [22], an autonomous strategy of active post-mission disposal of ODOs with the use of the EPS was developed. A scheme of disposal consisting of four stages was proposed. They include migration to a target orbit, approach, and others. Control laws for the thrust vector at each stage were developed. A simulation of post-mission disposal was carried out on an example of several debris objects, which confirmed the effectiveness of the proposed strategy.

A new concept of active post-mission disposal with the combined use of the thrust force of an EPS and the aerodynamic drag was proposed in [25]. The scheme of disposal developed in this work consists of two stages. The first stage is active, involving the firing of the EPS. The second one is passive, relying on the atmospheric drag. A study on the choice of the power of a Hall and ion EPSs required for disposal of the upper stage of the H-2A launch vehicle from the altitude of 600 km within one year was carried out. The dependence of the mass of the disposal object on the power of the EPS was determined. The influence of the time limit for the orbital existence on the flight trajectory was studied.

A method of post-mission disposal with the reentry of a small satellite equipped with a hybrid propulsion system was developed in [14]. A scheme of disposal consisting of two distinct segments was considered: movement in the upper layers of the atmosphere and movement in the dense layers of the atmosphere with landing in a specified area of the Earth. A study of the processes of motion in the atmosphere for different scenarios was carried out.

The work [16] is devoted to research on the active method of post-mission disposal from LEO with the use of the EPS. For active disposal with the use of Lyapunov methods, a stable time-sub-optimal control law for the thrust vector of the EPS was developed. Dependences of the increment of velocity required for post-mission disposal and the time of disposal on the altitude of the initial orbit and the altitude of the perigee of the beginning of the passive segment of the trajectory were determined. It was shown that in the absence of limitations on the delta-v, disposal in the entire range of LEO is done in less than five years. In the opposite case, the altitude of the initial orbit, its inclination, and the ballistic coefficient have to be taken into account. It was revealed that for the majority of inclinations, the most reasonable is the post-mission disposal by reducing the altitude of the orbit perigee, and, for the orbits close to 90°, change of the inclination can be of particular interest.

As for Ukraine, this matter was considered for the first time in [8]. The authors proposed a conceptual model of a servicing spacecraft for cleaning near-Earth space with the use of the EPS. An analysis of its main features was carried out, and the area of its applicability was determined as altitudes from 800 to 1200 km.

Further, the problems of active post-mission disposal of ODOs from LEO with the use of the EPS were developed in [11]. The authors proposed a method of combined post-mission disposal of ODOs with the use of the EPS and aerodynamic sail device (active post-mission disposal is a particular case of this method). A peculiarity of this method is the minimization of consumption of the propellant in the presence of limits on the time spent in the orbit.

In the dissertation work [10], extended research was carried out for the processes of active and combined post-mission disposal of the ODOs with the use of ESP. Besides, the author developed a method for determining the mass of ODO that can be disposed of with the use of an EPS with given parameters. For typical ODOs, the maximum delta-v, the energy expenditures, and the range of masses of the objects being disposed of were determined.

Besides, a study of the effectiveness of ODO post-mission disposal systems is carried out in [7]. The authors proposed the use of a criterion of the total integral relative effectiveness for the choice of the optimal disposal of ODOs from LEO. It was shown that the combined method of post-mission disposal with the use of a servicing apparatus of an autophage launch vehicle helps to reduce compensation expenses. The possibilities of creating combined disposal systems with multiple-firing engines for the reduction of operational costs were considered.

FORMULATION OF THE RESEARCH PROBLEM

Based on the literature review, we can conclude the following. For solving the problem of determining the capabilities of modern Hall thrusters ST-25

Figure 1. Hall thruster ST-25

Figure 2. Hall thruster ST-40

Table 1. **Hall thruster's characteristics**

Hall thruster	ST-25-1	ST-25-2	ST-40-1	ST-40-2
Input power, W	150	250	180	300
Thrust, mN	5.9	10.3	9.5	17
Specific impulse, s	650	1160	900	1490
Lifetime, hr	3000	3000	5000	5000

(Fig. 1) [28] and ST-40 (Fig. 2) [27] developed by Space Electric Thruster Systems (Ukraine) for active post-mission disposal of ODOs from LEO within limitations of the energy supply system we will use results presented in [10], specifically, the analytical dependences of the optimal increment of the characteristic velocity per one firing and the number of firings on the altitude of the initial orbit, ballistic coefficient, time of recharging of the battery and time of active operation of the control system. Besides, we will use the proposed method for determining the range of masses of ODOs that can be disposed of with the EPSs with given parameters.

In the course of research, it is necessary:

• to develop a scheme for the post-mission disposal of large-sized ODOs from LEO using EPS;

• to determine the dependence of the minimal delta-v increment per one firing and the number of firings on the altitude of the initial orbit, the ballistic coefficient, and the charging time of the battery in the presence of limitations on the active operation of the control system (CS);

• to determine the dependence of the upper margin of the mass of an ODO being disposed on the altitude of the initial orbit, the ballistic coefficient, and the charging time of the battery in the presence of limitations on the active operation of the CS;

• to determine the dependence of the minimal total time of operation of Hall thrusters ST-25 and ST-40 on the altitude of the initial orbit, the ballistic coefficient, and the charging time of the battery in the presence of limitations on the active operation of the CS;

The following initial data were assumed:

• the altitude of the initial orbit of the ODO in the range from 500 to 1500 km;

• the ballistic coefficient is 0.001 m^2/kg and 0.01 m^2/kg, which corresponds to the range of large-sized ODOs;

• the time of active functioning of the CS as equal to 1 year;

• the charging time of the accumulator battery of 1, 3, and 5 hours;

• the post-mission disposal time is 25 years;

• technical data of Hall thrusters are shown in Table 1.

THE MAIN MATERIAL OF THE RESEARCH

Consider the requirements for the process of post-mission disposal ODOs from into the dense layers of the Earth's atmosphere. On the one hand, the process of disposal should minimize the costs of the propellant. On the other hand, according to generally accepted norms, the disposal should not last longer than 25 years. At the same time, the active existence of most modern flight CS does not exceed 5 to 15 years.

Consequently, the process of post-mission disposal of these conditions should consist of two characteristic stages: the active and the passive [6, 10—12, 25]. The first is the formation of an elliptical orbit using the EPS during the active operation of the CS. The second is the passive trajectory of the orbit under the influence of the force of aerodynamic drag of the Earth's atmosphere of the time interval remaining out of the required 25 years from the moment of the beginning of the disposal.

Consider the active stage of the disposal. Let us introduce the assumption that the EPS is "perfectly controllable". Despite its advantages, EPS has one drawback — a small thrust force. It basically does not exceed one Newton. Consequently, ensuring the delta-v increments of hundreds of meters per second needed to remove an ODO from LEO will require long-term firings of the EPS. At the same time, it is necessary to take into account the dependence of the EPS operating time on the capabilities of the power supply system of the means of disposal, which are limited. That is, the process of forming the required values of speed increments using EPS for most ODOs will be a sequence of alternating EPS firing and downtime due to the need to charge the battery [10, 11]. In addition, the efficiency of the thrust force in the problem of forming an elliptical orbit depends significantly on the position of the ODO in orbit. Thus, around the apogee, it is more than 90 %, and around the perigee, less than 10 %.

Considering the above, we will present a scheme for post-mission disposal of an ODO from LEO using EPS in the form of the following scheme (Fig. 3).

Designations in Fig. 3: *a* — the Earth, *b* — the initial orbit of the ODO, *c* — the ODO with EPS turned off, *d* — the ODO with EPS operating, *e* — the trajectory of the active stage of disposal, *f* — the trajectory of the passive stage of disposal, *1* — reorientation of the ODO before the first ignition of the EPS, *2* — switching of EPS on, *3* — simple EPS, battery charging, *4* — passive flight after the end of the active work of the CS.

Let us consider the method of estimating the capabilities of the EPS to ensure the post-mission disposal of the ODO from LEO [10].

1. Determination of the minimum delta-v required for post-mission disposal, the corresponding number of firings of the EPS, and the delta-v per one firing. According to [10], these values will be determined as follows.

The number of EPS ignitions is determined by the relation

$$n_{EM} = a_{0N}(h_{\Pi} - a_{1N}) + a_{2N} \,, \tag{1}$$

where a_{0N}, a_{1N} , and a_{2N} are the coefficients.

a_{0N} is determined by relations

$$a_{0N} = b_{00N}\tau_{CS} + b_{01N} \,, \tag{2}$$

$$b_{00N} = c_{01N}\exp(c_{02N}\tau_{CB}^{c_{03N}}) \,, \tag{3}$$

$$b_{01N} = d_{01N}\exp(d_{02N}\ln\tau_{CB}^{d_{03N}}) \,, \tag{4}$$

$$c_{0iN} = e_{0iN}\sigma_{LO}^2 + e_{1iN}\sigma_{LO} + e_{2iN} \,, \tag{5}$$

$$d_{0iN} = f_{0iN}\sigma_{LO}^2 + f_{1iN}\sigma_{LO} + f_{2iN} \,, \tag{6}$$

$$\sigma_{LO} = \log\sigma_S \,,$$

where τ_{CS} is the service time of the CS (years), τ_{CB} is the battery charging time (hours), e_{0iN} , e_{1iN} , and e_{2iN} are given in Table 2, f_{0iN} , f_{1iN} , and f_{2iN} are given in Table 3, σ_S is the ballistic coefficient of ODO.

a_{1N} is given by

$$a_{1N} = 35\sigma_{LO}^2 + 307\sigma_{LO} + 1098 \,. \tag{7}$$

a_{2N} is determined by relations

$$a_{2N} = b_{20N}\tau_{CS} + 1 \,, \tag{8}$$

$$b_{20N} = c_{21N}\exp(c_{22N}\tau_{CB}^{c_{23N}}) \,, \tag{9}$$

$$c_{2iN} = h_{0iN}\sigma_{LO}^2 + h_{1iN}\sigma_{LO} + h_{2iN} \,, \tag{10}$$

where h_{0iN} , h_{1iN} , and h_{2iN} are given in Table 4.

The delta-v in one firing of the EPS is

$$\Delta V_{EM} = \frac{\Delta V_{EM\Sigma}}{n_{EM}}, \qquad (11)$$

where $\Delta V_{EM\Sigma}$ is the EPS's minimum delta-v during post-mission disposal time, which is determined by the relation

$$\Delta V_{EM\Sigma} = \nu_{1\Sigma}\{1 - \exp[\nu_{2\Sigma}(h_{\Pi} - \nu_{3\Sigma})^{\nu_{4\Sigma}}]\}, \qquad (12)$$

where $\nu_{1\Sigma}$, $\nu_{2\Sigma}$, $\nu_{3\Sigma}$, $\nu_{4\Sigma}$ are the coefficients.

$\nu_{1\Sigma}$ is determined by relations

$$\nu_{1\Sigma} = \alpha_{01\Sigma}\tau_{CS}^3 + \alpha_{11\Sigma}\tau_{CS}^2 + \alpha_{21\Sigma}\tau_{CS} + \alpha_{31\Sigma}, \qquad (13)$$

$$\alpha_{01\Sigma} = -0.01469\sigma_{LO} - 0.17884, \qquad (14)$$

$$\alpha_{11\Sigma} = 0.424856\sigma_{LO} + 3.00379, \qquad (15)$$

$$\alpha_{21\Sigma} = 1.289592\sigma_{LO}^3 - 0.83654\sigma_{LO}^2 - 1.55223\sigma_{LO} - 66.6, \qquad (16)$$

$$\alpha_{31\Sigma} = 31.3\sigma_{LO}^3 + 44.3\sigma_{LO}^2 + 198.9\sigma_{LO} - 924.3. \qquad (17)$$

$\nu_{2\Sigma}$ is given by

$$\nu_{2\Sigma} = (\alpha_{02\Sigma}\tau_{CS}^3 + \alpha_{12\Sigma}\tau_{CS}^2 + \alpha_{22\Sigma}\tau_{CS} + \alpha_{32\Sigma})\cdot 10^{-9}, \qquad (18)$$

$$\alpha_{02\Sigma} = 0.373019\sigma_{LO}^3 + 1.021142\sigma_{LO}^2 + 2.602269\sigma_{LO} + 32.5, \qquad (19)$$

$$\alpha_{12\Sigma} = -12.0\sigma_{LO}^3 - 33.0\sigma_{LO}^2 - 84.1\sigma_{LO} - 1050, \qquad (20)$$

$$\alpha_{22\Sigma} = 154.3\sigma_{LO}^3 + 422.2\sigma_{LO}^2 + 1081\sigma_{LO} + 13509, \qquad (21)$$

$$\alpha_{32\Sigma} = -7804\sigma_{LO}^3 - 21360\sigma_{LO}^2 - 54711\sigma_{LO} - 683719. \qquad (22)$$

$\nu_{3\Sigma}$ is determined by the relation

$$\nu_{3\Sigma} = 6.333472\sigma_{LO}^3 + 63.0\sigma_{LO}^2 + 334.7\sigma_{LO} + 1098. \qquad (23)$$

$\nu_{4\Sigma}$ is given by

$$\nu_{4\Sigma} = -(0.00015\sigma_{LO} + 0.00516)\tau_{CB} + 0.027609\sigma_{LO} + 0.919621. \qquad (24)$$

2. Determination of the minimum total mass of the propellant necessary to ensure the disposal

$$\Delta m_{EM} = m_s\left[1 - \exp\left(-\frac{\Delta V_{EM} n_{EM}}{9.80665 I_E}\right)\right], \qquad (25)$$

where m_S is the ODO's mass; I_E is the EPS's specific impulse.

3. Determination of the minimum total operating time of the EPS.

$$t_{EM} = \frac{\Delta m_{EM}}{\dot{m}_E}, \qquad (26)$$

where $\dot{m}_E$ is the EPS's flow mass rate.

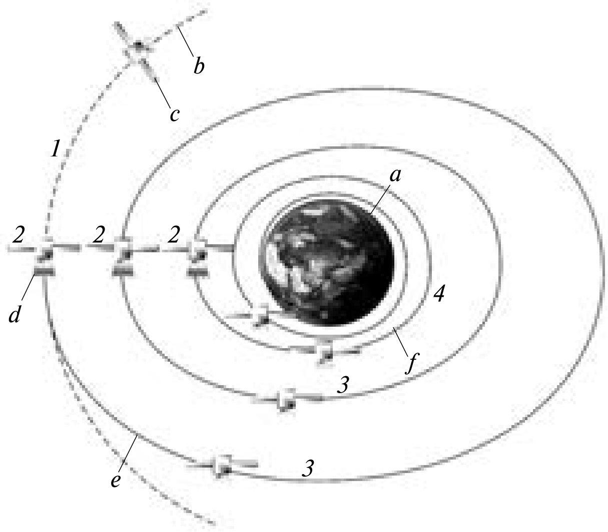

Figure 3. Post-mission disposal scheme

Table 2. **Equation (5) coefficients**

i	1	2	3
e_{0iN}	-1.178452×10^{-2}	1.291436×10^{-4}	7.536625×10^{-4}
e_{1iN}	-2.201379×10^{-1}	-8.937730×10^{-2}	-1.083015×10^{-2}
e_{2iN}	-3.569176×10^{0}	-1.632239×10^{0}	3.325187×10^{-1}

Table 3. **Equation (6) coefficients**

i	1	2	3
f_{0iN}	1.278853×10^{-4}	1.868910×10^{-3}	1.300126×10^{-2}
f_{1iN}	1.081764×10^{-3}	1.209050×10^{-2}	8.016459×10^{-2}
f_{2iN}	3.497105×10^{-2}	-9.594542×10^{-1}	1.151486×10^{0}

Table 4. **Equation (10) coefficients**

i	1	2	3
h_{0iN}	1.665162×10^{1}	-2.229510×10^{-3}	-1.841633×10^{-3}
h_{1iN}	4.559628×10^{0}	-1.106098×10^{-2}	-1.191240×10^{-2}
h_{2iN}	5.472710×10^{3}	-6.460616×10^{-1}	1.145981×10^{0}

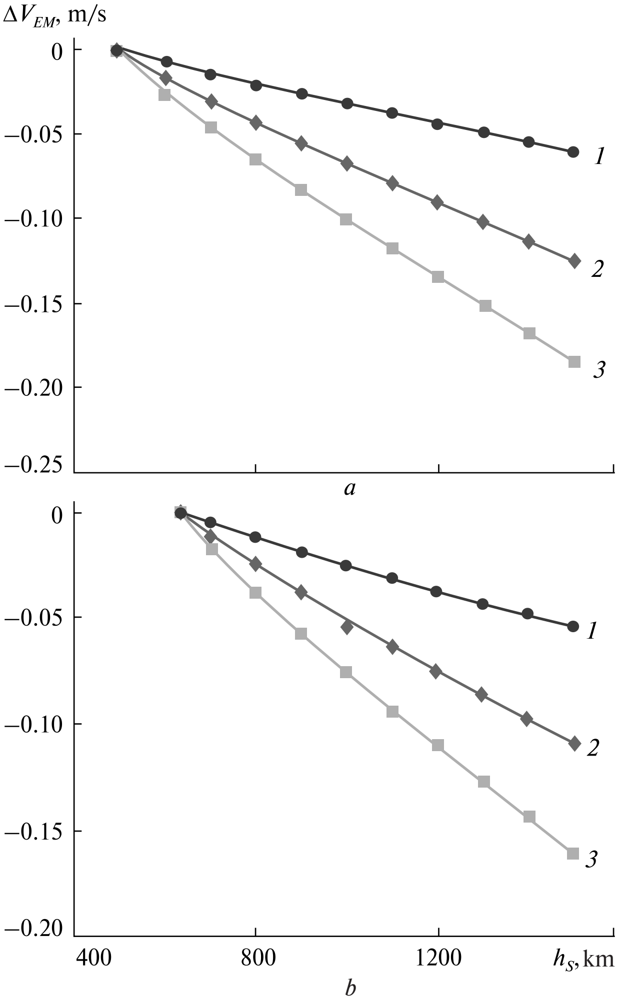

Figure 4. Dependences of the minimum delta-v increment for one ignition on the altitude, the ballistic coefficient, and the battery charging time: *a* — for σ_S = 0.001 m^2/kg, *b* — for σ_S = 0.01 m^2/kg. (Curve *1, 2, 3* — for τ_{CB} = 1, 3, 5 hr, respectively)

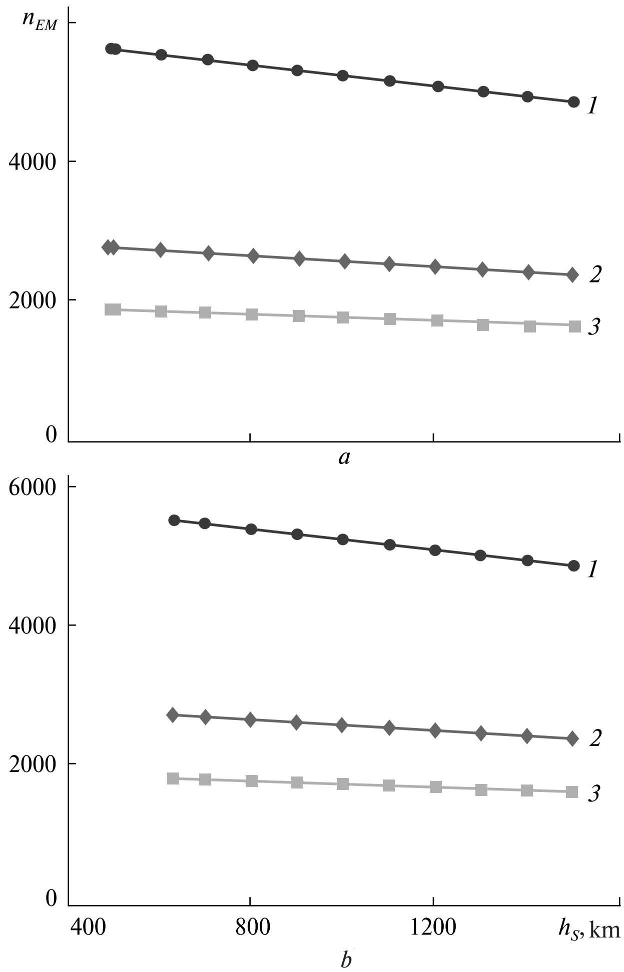

Figure 5. Dependences of the number of EPS ignitions on the altitude, the ballistic coefficient, and the battery charging time: *a* — for σ_S = 0.001 m^2/kg, *b* — for σ_S = 0.01 m^2/kg. (Curve *1, 2, 3* — for τ_{CB} = 1, 3, 5 hr, respectively)

4. If the resulting mass of the propellant and the minimum operating time of the EPS meet the specified requirements for the operation of the EPS, we can move on to the next operations.

5. Calculation of the minimum energy spending to ensure the process of disposal

$$E_{EM} = N_E t_{EM}\,, \qquad (27)$$

where N_E is the EPS's input power.

6. Calculation of the time of one firing of the EPS

$$\tau_{EM}(m_s) = \frac{m_s}{\dot{m}_E}\left[1 - \exp\left(-\frac{\Delta V_{EM}}{9.80665 I_E}\right)\right]. \qquad (28)$$

7. Determination of the upper boundary of the range of masses of ODOs, which can be disposed of using EPS. To do this, we will define the level of efficiency of using EPS as 10 % of its maximum value. This corresponds to about 1/4 of the part of the orbital period of the ODO. The upper boundary we will find by solving the functional

$$m_{SM} = \underset{t_{EM}=0,25T_S}{\arg}\left[\tau_{EM}\left(m_S\right)\right], \qquad (29)$$

where T_S is the period of ODO's orbital motion.

As an outcome of mathematical modeling of post-mission disposal of large-sized ODOs from LEO, the

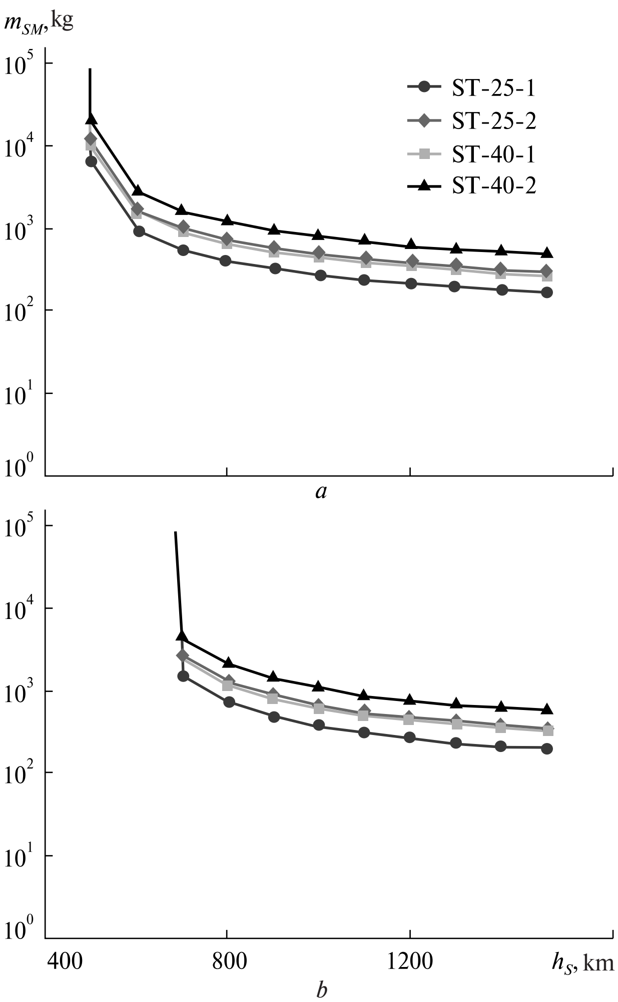

Figure 6. Dependence of the upper limit of the mass of the ODO on the initial orbit altitude, the ballistic coefficient for the battery charging time 1 hr: *a* — for σ_S = 0.001 m^2/kg, *b* — for σ_S = 0.01 m^2/kg

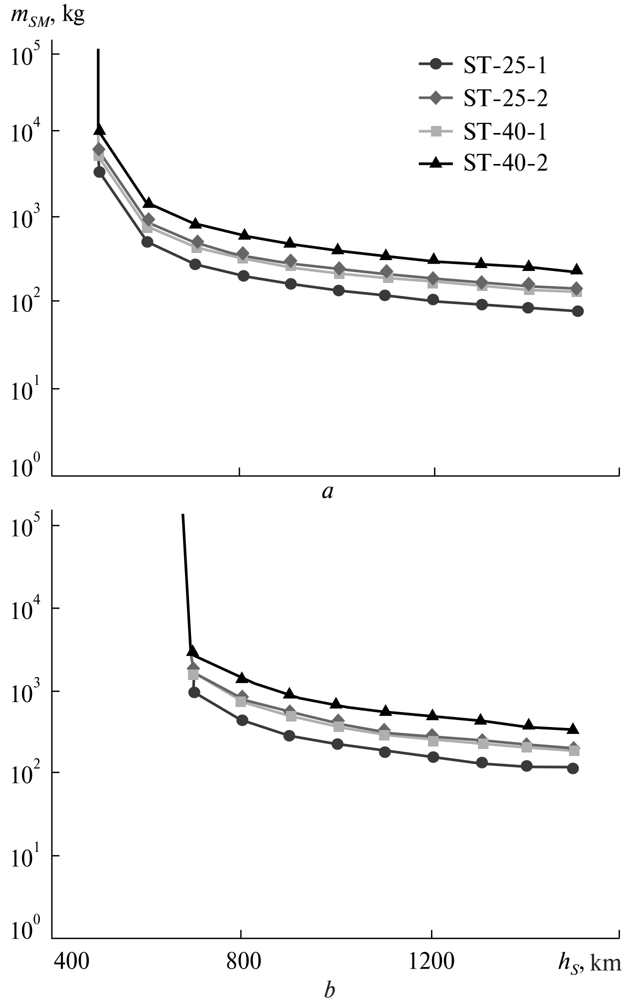

Figure 7. Dependence of the upper limit of the mass of the ODO on the initial orbit altitude, the ballistic coefficient for the battery charging time 3 hr: *a* — for σ_S = 0.001 m^2/kg, *b* — for σ_S = 0.01 m^2/kg

following dependences on the altitude of the initial orbit, the ballistic coefficient, and charging time of the battery were obtained:

- minimum delta-v for one ignition of the EPS (Fig. 4);
- number of the EPS ignitions (Fig. 5);
- maximum mass of the ODO (Fig. 6—8);
- total operating time of the EPS (Fig. 9—11).

DISCUSSION OF RESULTS

Consider the results obtained. As can be seen in Fig. 4, the minimum delta-v increases exponentially with the increase in the altitude of the initial orbit, is inversely proportional to the ballistic coefficient, and is directly proportional to the battery recharging time. An increase in the altitude of the initial orbit requires a substantial delta-v for ensuring post-mission disposal. An increase in the ballistic coefficient results in a stronger influence of the atmospheric drag and a decrease of the required delta-v. An increase in the battery charging time leads to the reduction of the number of firings of the EPS over the active functioning of the CS. This leads to the growth of the required delta-v.

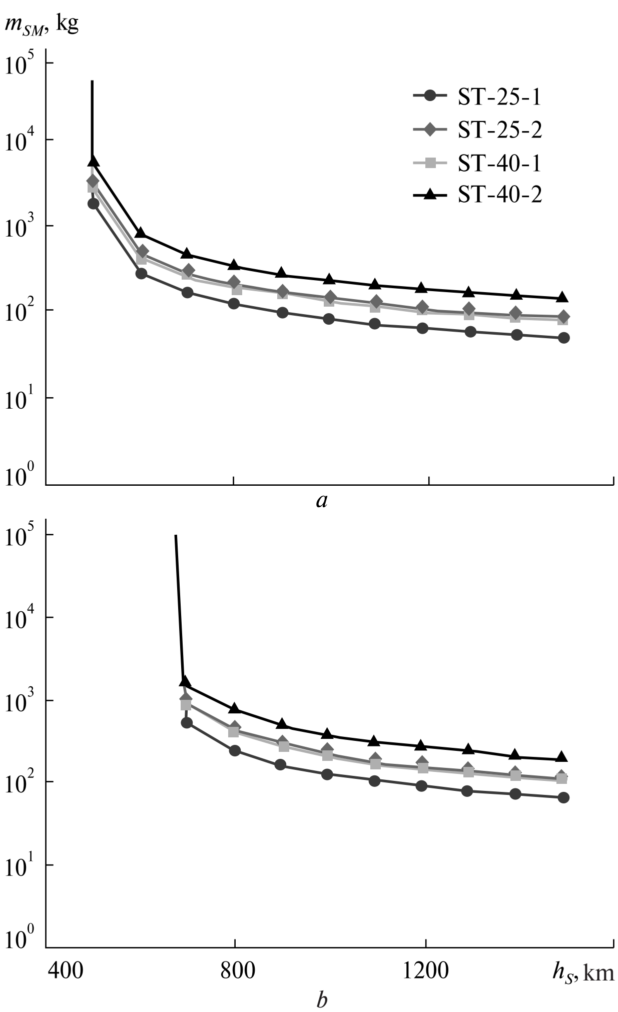

Figure 8. Dependence of the upper limit of the mass of the ODO on the initial orbit altitude, the ballistic coefficient for the battery charging time 5 hr: *a* — for σ_S = 0.001 m^2/kg, *b* — for σ_S = 0.01 m^2/kg

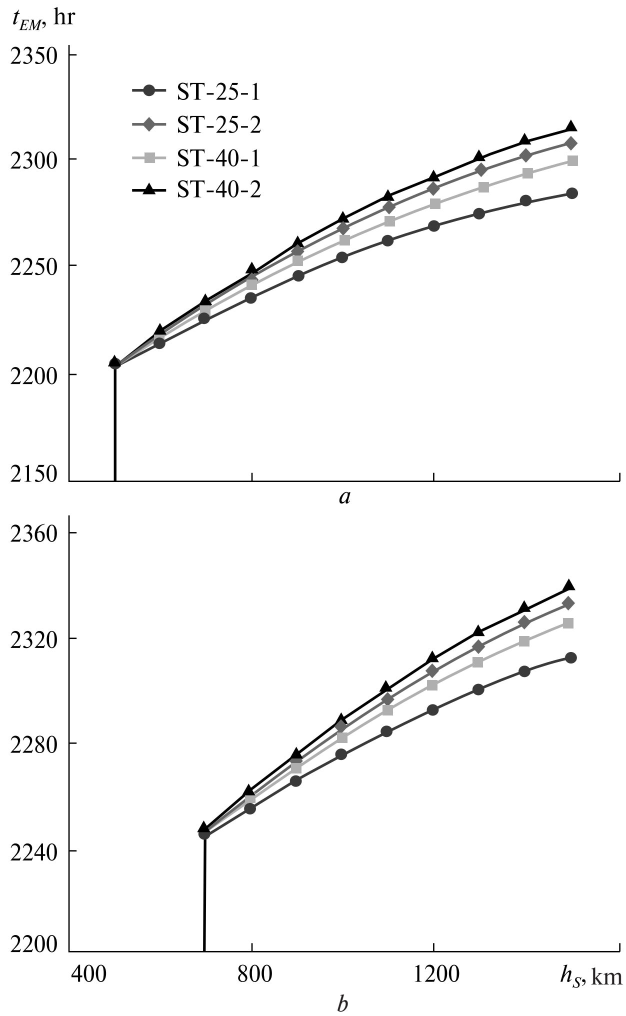

Figure 9. Dependence of the minimum total operating time of the EPS on the initial orbit altitude, the ballistic coefficient for the battery charging time 1 hr: *a* — for σ_S = 0.001 m^2/kg, *b* — for σ_S = 0.01 m^2/kg

According to Fig. 5, we can see the following. The number of firings reduces with the growth of the altitude. A similar situation must be observed in the case of impulse operation of the EPS. With the growth of the altitude, the orbital period of an ODO increases. Consequently, the number of passages of the apogee of the orbit, around which the EPS is fired, during the period of active operation of the CS reduces. The influence of the ballistic coefficient on the number of firings of the EPS is insignificant, but with its increase, the number of firings increases insignificantly. This is stipulated by a stronger influence of the atmospheric drag and, therefore, by a smaller value of the orbital period.

Consider the dependence of the mass of the ODO (Fig. 6—8). It has a complicated nature and, for the considered range of variation of the parameters, lies within limits from 60 kg to 4 tons. An inversely proportional dependence on the altitude of the initial orbit is observed. With the increase in the ballistic coefficient of the ODO by order of magnitude, its disposal mass increases by approximately 2.5 times.

 ISSN 1561-8889. Космічна наука і технологія. 2023. Т. 29. № 3

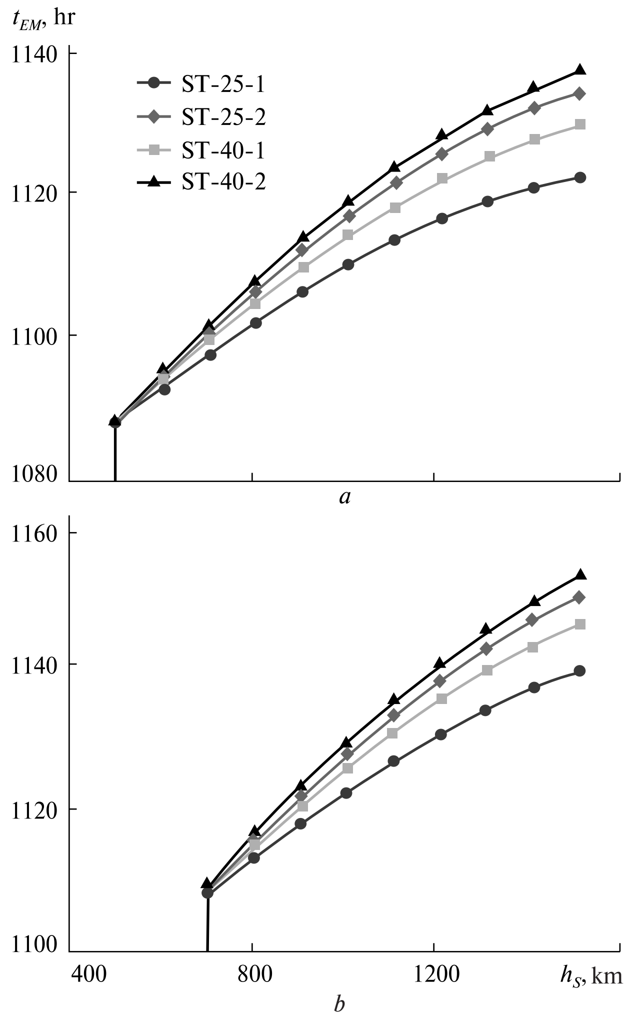

Figure 10. Dependence of the minimum total operating time of the EPS on the initial orbit altitude, the ballistic coefficient for the battery charging time 3 hr: *a* — for $\sigma_S = 0.001$ m^2/kg, *b* — for $\sigma_S = 0.01$ m^2/kg

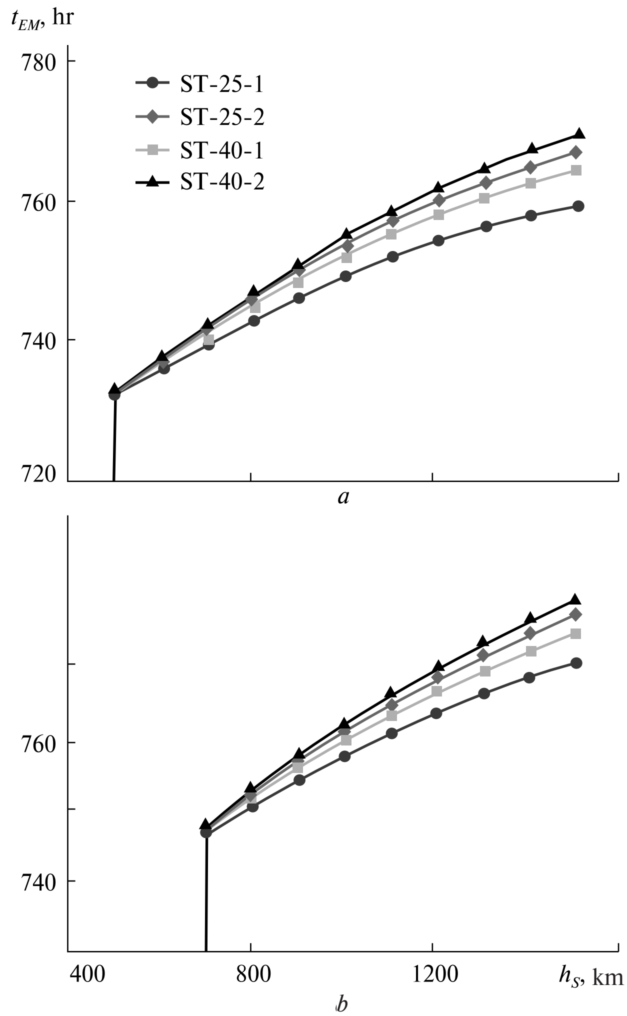

Figure 11. Dependence of the minimum total operating time of the EPS on the initial orbit altitude, the ballistic coefficient for the battery charging time 5 hr: *a* — for $\sigma_S = 0.001$ m^2/kg, *b* — for $\sigma_S = 0.01$ m^2/kg

An increase in the battery charging time from 1 hour to 5 times causes a decrease in the disposal mass by approximately 3 times. For ST-25, the disposal mass ranges from 60 kg to 2.5 tons. And for ST-40, from 100 kg to 4 tons.

Concerning the total time of operation of the EPS over the time of post-mission disposal, the results are given in Fig. 9—11. As can be seen from the obtained dependences and Table 1, the total minimum time of operation does not exceed the given values. The altitude of the initial orbit influences this value insignificantly, with inverse proportionality within 5 %. Its increase by an order of magnitude increases this time by approximately 5 %. The increase in the battery charging time from 1 hour by a factor of 5 causes a lowering of the minimal total time of firing of the EPS by three times.

Note that the dependences ST-25-2 and ST-40-2 correspond to the upper limit of thruster performance and are, therefore, optimal in terms of the mass to be disposed of. In turn, ST-25-1 and ST-40-1 correspond to the lower limit of engine performance and represent the upper limit for any operating modes of the thrusters.

CONCLUSIONS

In the research, the following results were obtained:

• a scheme for post-mission disposal of large-sized ODOs from low-Earth orbits using EPS has been developed;

• dependences of the minimum increment of delta-v per one firing and the number of firings on the altitude of the initial orbit, the ballistic coefficient, and the battery charging time in the presence of a limitation on the time of active operation of the CS were determined;

• dependence of the upper margin of the mass of the ODO to be disposed of on the altitude of the initial orbit, the ballistic coefficient, and the battery charging time in the presence of limitations on the time of active operation of the CS was determined;

• dependence of the minimum total time of operation of Hall thrusters ST-25 and ST-40 on the altitude of the initial orbit, the ballistic coefficient, and battery charging time in the presence of limitations on the time of active operation of the CS was determined;

• it was shown that, depending on the operational conditions for the Hall thruster ST-25, the maximum disposal mass ranges from 60 kg to 2.5 tons, and for the Hall thruster ST-40, from 100 kg to 4 tons;

• it was shown that the total time of operation of the EPS does not exceed the limiting values.

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Стаття надійшла до редакції 24.02.2023 *Received 24.02.2023*
Після доопрацювання 24.02.2023 *Revised 24.02.2023*
Прийнято до друку 01.03.2023 *Accepted 01.03.2023*

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ОЦІНКА МОЖЛИВОСТЕЙ ВИКОРИСТАННЯ ЕЛЕКТРОРАКЕТНИХ РУШІЙНИХ УСТАНОВОК ДЛЯ ВІДВЕДЕННЯ ВЕЛИКОГАБАРИТНОГО КОСМІЧНОГО СМІТТЯ

Постійне зростання кількості космічного сміття є суттєвою загрозою польотам супутників. В даний момент розглядається багато методів очистки навколоземного простору з використанням різноманітних засобів, робота яких будується на різних фізичних принципах. Але, незважаючи на це, найбільше застосування з них отримав активний відвід з використанням реактивної рушійної установки. З огляду на високу величину питомого імпульсу та низькі габаритно-масові характеристики вибір електроракетних рушійних установок становить особливий інтерес. Незважаючи на свої переваги, такі установки мають ряд особливостей, які необхідно враховувати у процесі проектування та використання сучасних засобів відведення. До них належать максимальний час роботи установки за одне включення, максимальний час заряду акумуляторної батареї та час функціонування системи керування. Метою даної роботи є визначення можливостей сучасних електроракетних рушійних установок ST-25 і ST-40 розробки Space Electric Thruster Systems в задачі відведення великогабаритних об'єктів космічного сміття з низьких навколоземних орбіт з урахуванням обмежень системи живлення. Для досягнення поставленої мети використано методи аналізу, синтезу, порівняння і комп'ютерного моделювання. В результаті досліджень вирішено такі задачі. Розроблено схему відведення великогабаритних об'єктів космічного сміття з низьких навколоземних орбіт, яка враховує особливості використання електроракетної рушійної установки. Визначено залежність мінімальної зміни швидкості, яка необхідна для забезпечення відведення протягом 25 років, від висоти початкової орбіти і балістичного коефіцієнта космічного об'єкта космічного сміття. Визначено верхню межу області мас об'єктів космічного сміття від висоти початкової орбіти і балістичного коефіцієнта, які можна відвести з низьких навколоземних орбіт електроракетними рушійними установками. Отримані результати можна використовувати в задачах проєктування сучасних засобів активного відведення космічного сміття з використанням електроракетних рушійних установок розробки Space Electric Thruster Systems.

Ключові слова: космічне сміття, активне відведення, електроракетна рушійна установка.