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DESIGNING THE CONFIGURATION AND SELECTING THE DESIGN PARAMETERS OF DRAG SYSTEMS FOR DEORBITING SPACECRAFT CREATED BY *PIVDENNE* DESIGN OFFICE

Introduction. *To stabilize the space debris environment, defunct spacecraft and mission-related debris shall be deorbited.*

Problem Statement. *The analysis of drag sail systems for deorbiting spacecraft has shown that they are effective for spacecraft deorbiting from orbits having an altitude of up to 800 km, but have some disadvantages: vulnerability of the shell material to space debris fragments that may damage it and electrostatic breakdown.*

Purpose. *The purpose of this research is to design the configuration and to select the design parameters of drag systems for deorbiting spacecraft created by Pivdenne Design Office.*

Materials and Methods. *Methods of space flight mechanics, mathematical modeling of design problems have been used in this research.*

Results. *The calculations have shown that the time of deorbiting Sich-2-1 spacecraft from the design orbit is about 6.5 years for a mass of the drag deorbit system of 9 kg that is 5% of the mass of Sich-2-1 spacecraft. It has been determined that in the case of increasing the deorbit time from the design orbit after the end of operational life to 25 years, the mass of the drag system may be reduced to 4.5 kg. With a mass of the drag deorbit system of 9 kg, the effective use of this DAD system is limited to an altitude from 730 to 750 km, in the case of close to circular orbits of different dislocations, and to an altitude of at most 700 km in perigee and 842 km in apogee in the case of low-elliptical orbits.*

Conclusions. *Based on the requirements of Pivdenne Design Office for the mass and dimensions of the drag augmentation device, the configuration and design view of the drag augmentation device (DAD) have been developed. This design is notable for its compactness that is due to the use of spring mechanisms and low-cost micro-motors, which deploy drag elements. In this design, the device occupies a little space on Sich-2-1 spacecraft.*

Keywords: *space debris, drag deorbit system, deorbit time, design parameters of the deorbit system.*

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The problem of clogging outer space is one of the obstacles to the proper operation of spacecraft (SC), orbital groups (OG), and orbital stations (OS). The dynamics of growth of cataloged space debris (SD) in Earth orbits has been given in [1]. Also, according to [1] on April 1, 2019, in Earth orbits, about 14,432 space debris with characteristic dimensions of more than 10 cm were cataloged.

The main sources of space debris include defunct spacecraft, rocket bodies, mission-related debris, and fragmentation debris. According to studies [2, 3], the most clogged areas of the near-Earth space are low Earth orbit (LEO) areas with an altitude of up to 2000 km, and high geosynchronous elliptical orbits of communication satellites within the range of inclination 50°...60°.

In order to prevent an increase in the number of SD, the Inter-Agency Space Debris Coordination Committee (IADC) has made recommendations on reducing the lifetime of SD on LEO down to, at most, 25 years [4]. However, given the current upward trends in the number of SD in LEO [1], the orbital lifetime of SD may be shortened more.

Today, there are four techniques that can move debris from heavily trafficked orbits:

1) deorbiting of SD with subsequent burn in the dense layers of the atmosphere [5—7];

2) controlled re-entry of large SD, i.e. the ability to force the entry over a pre-determined area, region, within which the debris is to fall [8];

3) moving communications satellites at the end of their operational lives to graveyard (dump) orbits [9];

4) in-orbit recycling of SD (a new concept) [10, 11].

These problems are solved by means of active and passive deorbit systems. The active deorbit systems are implemented through using scavenger satellites [12, 13] that deorbit space debris or through equipping spacecraft with additional propulsion systems [14, 15] that generate the braking pulse and transfer the spacecraft to low orbits, where their lifetime satisfies the conditions [4]. However, in the case of scavengers with robotic manipulators and cable systems, there are difficulties in capturing rotating space debris, as well as in controlling the scavenger — SD relative motion with a stable position of tethered space debris [16]. In turn, for the formation of the braking pulse with the help of active propulsion systems, a necessary condition is the efficiency of the orientation and stabilization systems and power supply of the spacecraft, which is quite difficult to provide at the end of operational life. Thus, there are many technical difficulties that unable the use of the active deorbit systems for all space debris.

Unlike the active ones, the passive deorbit systems require neither fuel consumption for their operation (almost do not require on-board energy consumption, except for electromagnetic systems) nor control of the relative motion of the removed spacecraft. Today, the known passive systems for deorbiting space debris from LEO are drag augmentation devices (DADs) [17—19], bare electrodynamic tethers (BET) [20, 21], electromagnetic tethers (EMT) [22], and space debris deorbit systems with the use of permanent magnets (PMS) [23]. All these systems have a good scientific and theoretical justification and confirmation by space or ground laboratory experiments. Thus, each of the passive systems has its technical advantages and disadvantages, and some of these systems have difficulties in practical implementation.

The BET systems are the "fastest" deorbit systems as they have the shortest deorbit time. However, there are significant difficulties in their practical implementation. For the wires being rather long (from 1 to 10 km), it is difficult to stabilize their relative position, preventing them from being entangled, and keeping a constant tension after deployment. As for the disadvantages of EMT, these systems require uninterrupted power supply of electromagnets with electric current, and hence the efficiency of the spacecraft power system, which is a quite troublesome task at the end of the spacecraft operational life.

Therefore, the most promising passive deorbit systems are DADs and PMS. However, for the application of PMS, it is necessary to design capsule shields for permanent magnets (PM), which protect the spacecraft equipment from magnetic fields and from the interaction of PM with dynamic ionospheric plasma flow until the end of the operational life of the spacecraft. It should be noted that these capsule shields occupy a lot of useful space on the spacecraft.

Thus, given the lack of free space on the *Sich-2-1* spacecraft designed by *Pivdenne* Design Office, the most rational technical solution is to make a compact drag-sail system to be deployed by a special mechanism for deorbiting the spacecraft after the end of its operation life.

Designing a configuration of DAD for deorbiting Sich-2-1 spacecraft. In accordance with the requirements of *Pivdenne* Design Office for designing DAD for *Sich-2-1* spacecraft, the device shall be located at the base plate of the spacecraft. The arrangement of DAD is shown in Fig. 1.

Given the requirements of *Pivdene* Design Office, a compact drag augmentation system with a special deployment device has been designed. The stowed view is shown in Fig. 2.

The body (Fig. 2) of the box-shaped container is proposed to be made of sheet aluminum (alloy D16) with a protective coating (PF-115 enamel, gray), with the lid of the container made of sheet aluminum (alloy D16) and protected by coating (PF-115 enamel, gray). The container serves as a system for storing drag elements (folded sails and deployment mechanisms) until the end of the operational life of the spacecraft. After the end of the operational life of the spacecraft, the air drag of the spacecraft in orbit increases as a result of increase in its cross-sectional area during the deployment of additional areas of sails. The design of the sail device and the deployment system folded is presented in Fig. 3.

The sail is made of polyimide film PM-A, in the form of two cylinders connected to each other by a strip. After opening the container lid *5* by means of pyrolock *3* and spring mechanisms 6, the drag sail elements 7 are deployed with the use of special electric drives 8 (Fig. 4).

The DAD for deorbiting the *Sich-2-1* spacecraft operates as follows. The device in the closed position of container *2* is installed on spacecraft *1* (Fig. 2). The electrical connector of the device is connected to the power system of the spacecraft.

Upon the command from the spacecraft, voltage is supplied to pyrolock *3* of the device. The pyrocartridge is triggered and the pyrolock rod *3* releases the tongue *4* of the lid *5* of the container *2*. The lid *5* under the action of spring mechanism *6* of the container rotates on the axes and opens the container *2* (Fig. 3). Inside the container, at an angle to each other, there are installed two mechanisms of sails *7* in the compressed state (Fig. 3). The springs of the

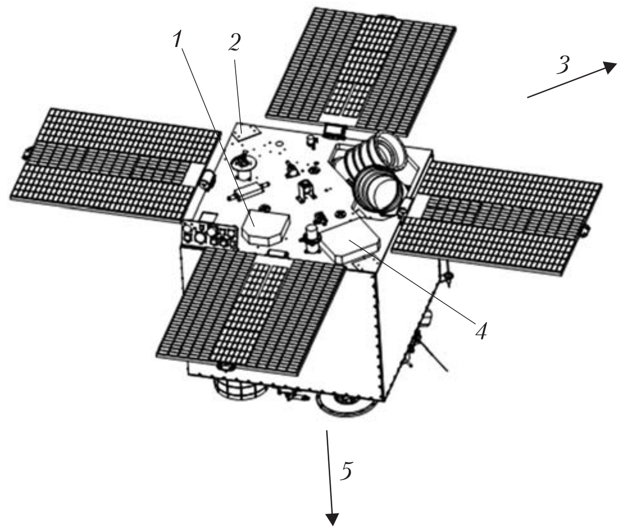


Fig. 1. The area of DAD location on the base plate of *Sich-2-1* spacecraft: *1* —area of DAD location; *2* — base plate; *3* — flight direction; *4* — area of DAD location; *5* — direction towards the Earth

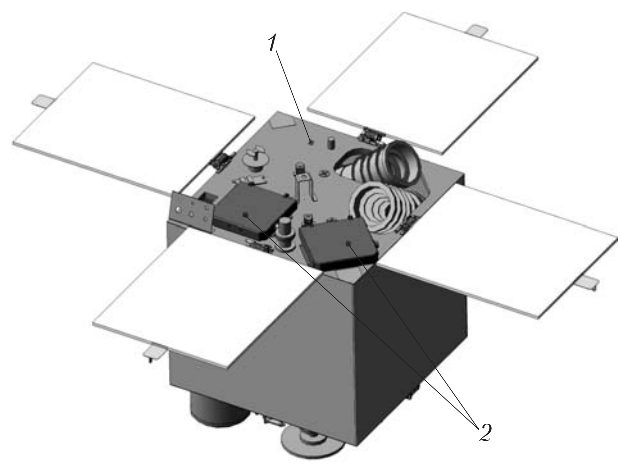


Fig. 2. Stowed view of *Sich-2-1* spacecraft DAD:
1 — *Sich -2-1* spacecraft; *2* — containers for onboard storage of DAD

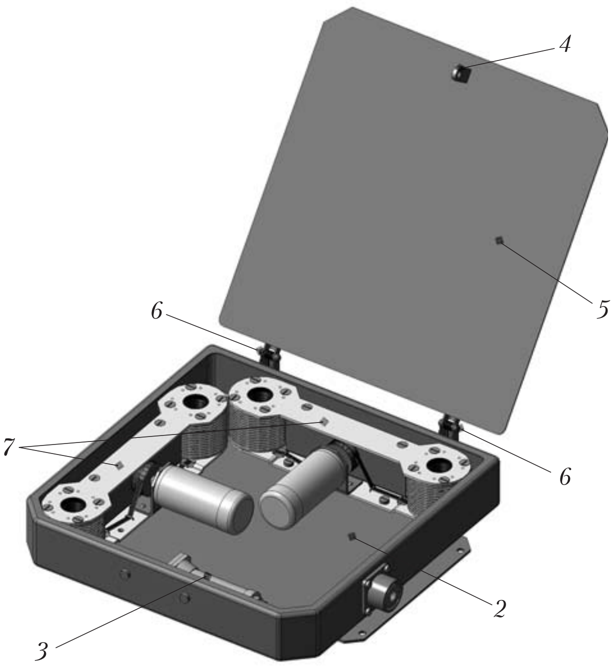


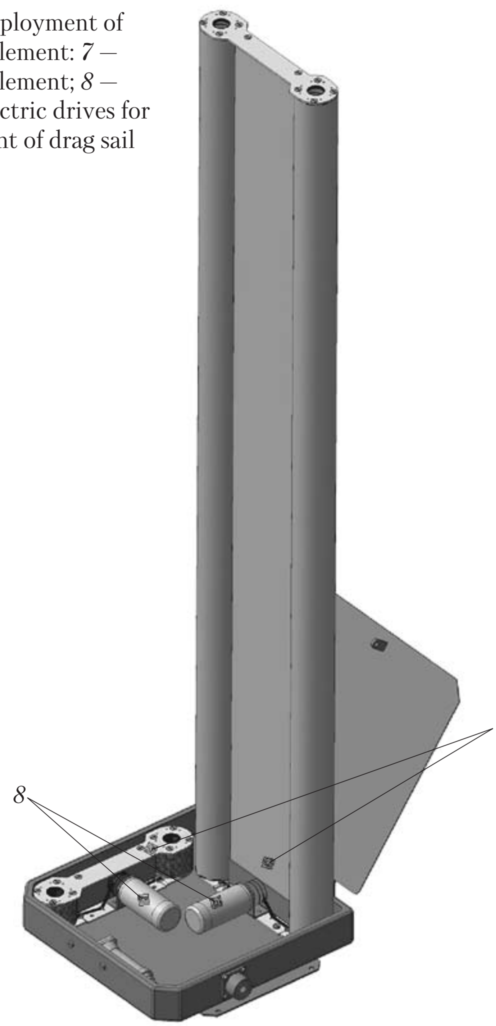
Fig. 3. Configuration of the sail device and the deployment system in folded form: *2* — container for onboard storage of DAD; *3* — pyrolock; *4* — tongue with a hole for fixing the lid on the body with the pyro lock rod in the closed position of the container; *5* — container lid; *6* — spring mechanisms for opening the lid of the container; *7* — drag sail elements

sail rods push out the upper plates of the sail frames with the sail blades attached to them and overcome the forces of tension of the drive wires.

Upon the spacecraft command, the voltage is supplied on engine *8* of one of the sails mechanisms *7*, or on both engines at once, to rotate the engine so to unwind the wires from the drive coil (Fig. 4). The tension of the wires weakens. The springs of the sail rods push out the upper plate of the sail frame with the sail blade fixed on it, on one (Fig. 4) or on both (Fig. 5) mechanisms. The magnitude of the sail opening is regulated by the spacecraft and can be changed, if necessary. As soon as the sails *7* are deployed, spacecraft *8* starts deorbiting (Fig. 5).

The initial data and mathematical models for determining the limits of DAD application for Sich-2-1 spacecraft. The following initial data are used to estimate the limit of applicability of DAD for deorbiting *Sich-2-1* spacecraft:

Fig. 4. Deployment of drag sail element: *7* — drag sail element; *8* — special electric drives for deployment of drag sail element



The parameters of the osculating elements of the expected orbit of *Sich-2-1* spacecraft:

- major half-axis a = 7046 km;
- inclination i = 98.08 degrees;
- argument of perigee ω = 69 degrees.

The mass and dimensions of *Sich-2-1* spacecraft:

- mass m_{KA} = 180 kg;
- midship section area $S_M = 0{,}58$ m^2.

The requirements for DAD are as follows:

- the orbital life should be 25 years;
- the mass of DAD should not exceed 5% of the spacecraft mass.

For calculating the time of deorbiting *Sich-2-1* spacecraft from the specified expected orbit and similar orbits for spacecraft of *Sich-2-1* class, it has

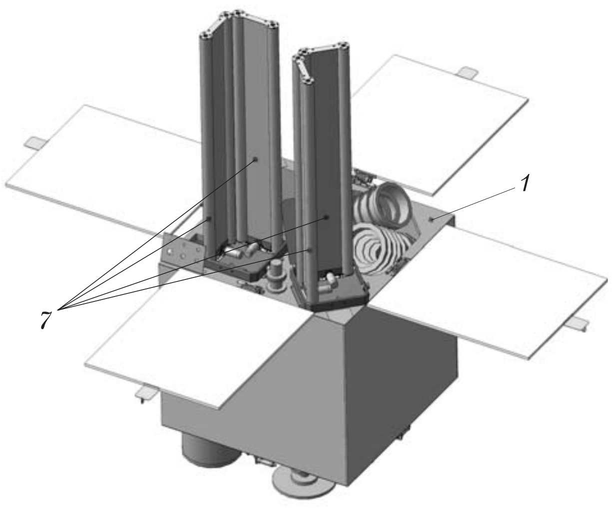


Fig. 5. Drag sail elements in the deployed form

been proposed to use two systems of differential equations in osculating elements [24, 25]. The first system of differential equations is a classical approach to the application of Lagrangian planetary equations in the Gaussian form [24]:

$$\left.\begin{aligned}
&\frac{\mathrm{d}a}{\mathrm{d}\vartheta}=\frac{2pr_{KA}^{2}}{\mu\left(1-e^{2}\right)^{2}}\left(S\cdot e\sin\vartheta+T\cdot\frac{p}{r_{KA}}\right)\\
&\frac{\mathrm{d}e}{\mathrm{d}\vartheta}=\frac{r_{KA}^{2}}{\mu}\left\{S\cdot\sin\vartheta+T\cdot\cos\vartheta\left(1+\frac{r_{KA}}{p}\right)+T\cdot e\frac{r_{KA}}{p}\right\}\\
&\frac{\mathrm{d}i}{\mathrm{d}\vartheta}=\frac{r_{KA}^{3}}{\mu p}\cos\left(\vartheta+\omega\right)\cdot W\\
&\frac{\mathrm{d}\Omega}{\mathrm{d}\vartheta}=\frac{r_{KA}^{3}\sin\left(\vartheta+\omega\right)}{\mu p\sin i}W\\
&\frac{\mathrm{d}\omega}{\mathrm{d}\vartheta}=\frac{r_{KA}^{2}}{\mu e}\left\{-\cos\vartheta S+\left(1+\frac{r_{KA}}{p}\right)\sin\vartheta T\right\}-\\
&\qquad-\cos i\frac{r_{KA}^{3}\sin\left(\vartheta+\omega\right)}{\mu p\sin i}W\\
&\frac{\mathrm{d}t}{\mathrm{d}\vartheta}=\frac{r_{KA}^{2}}{\sqrt{\mu p}}\left\{1+\frac{r_{KA}^{2}}{\mu e}\left[\cos\vartheta S-\left(1+\frac{r_{KA}}{p}\right)\sin\vartheta T\right]\right\}
\end{aligned}\right\}, \tag{1}$$

where a is the major half-axis of the orbit; e is the eccentricity; Ω is the right ascension of the ascending node; ω is the argument of the perigee; i is the orbital inclination; $r_{ÊÀ}$ is the radius vector of the spacecraft,

$$r_{\mathrm{KA}}=\frac{a(1-e^{2})}{1+e\cos\vartheta};$$

p is the focal parameter of the orbit, $p=a\left(1-e^{2}\right)$; μ is the gravitational constant; ϑ is the true anomaly; t is time of the spacecraft motion in the orbit; *S, T,* and *W* are the projections of radial, transverse and normal perturbing accelerations on the axis of the orbital coordinate system.

However, it should be noted that system of differential equations (1) has a significant disadvantage i.e. possible degeneracy of the solution of the system at low values of eccentricity or inclination. Also, this system is not suitable for the analysis of spacecraft motion in the orbits close to the polar ones. The limits on the eccentricity and the inclination $e\geq 0{,}005$, $10^{\circ}\leq i\leq 81^{\circ}$ are not suitable for the analysis of the orbit to launch *Sich-2-1*spacecraft. Given the listed shortcomings of model (1), the system of differential equations [25] is written as follows:

$$\left.\begin{aligned}
&\frac{di}{dt}=z\cos u\cdot W^{*}\\
&\frac{d\Omega}{dt}=z\frac{\sin u}{\sin i}\cdot W^{*}\\
&\frac{du}{dt}=\sqrt{\frac{\mu}{R_0^3}}\left(\frac{s^{\frac{1}{2}}}{z^{2}}-1\right)-z\frac{\sin u}{\sin i}\cdot\cos i\cdot W^{*}+\sqrt{\frac{\mu}{R_0^3}}\\
&\frac{d\gamma}{dt}=2\cdot z\cdot s\cdot T^{*}\\
&\frac{db_1}{dt}=\sqrt{\frac{\mu}{R_0^3}}b_2\\
&\frac{db_2}{dt}=\sqrt{\frac{\mu}{R_0^3}}\frac{\gamma-b_1}{z^{3}}+S^{*}
\end{aligned}\right\}, \tag{2}$$

where u is the latitude argument; R_0 is the radius of the undisturbed circular orbit;

$$b_1=\frac{R}{R_0}-1$$

is deviation of the current radius of the perturbed orbit R from the radius of the undisturbed or-

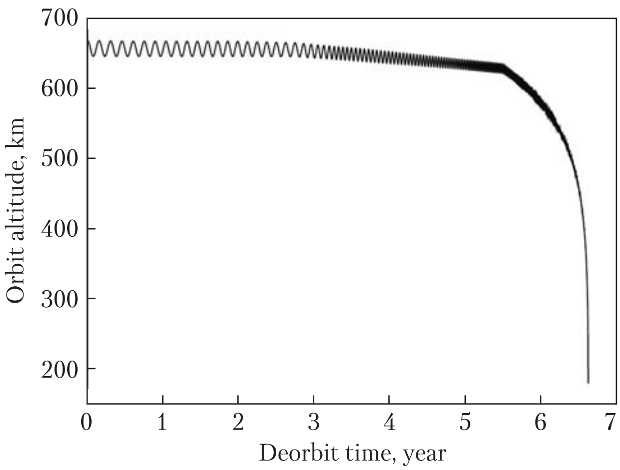


Fig. 6. Time of *Sich-2-1* space deorbiting with the use of designed DAD

bit; b_2 is the radial velocity in the perturbed orbit, which is related to the velocity in the perturbed circular orbit;

$$\gamma = \frac{p}{R_0} - 1$$

is the infinitesimal coefficient;

$$z = 1 + b_1 ; \; s = 1 + \gamma ;$$

$$S^* = \sqrt{\frac{R_0}{\mu}} \cdot S ; \; T^* = \sqrt{\frac{R_0}{\mu \cdot s}} \cdot T ; \; W^* = \sqrt{\frac{R_0}{\mu \cdot s}} \cdot W .$$

According to [27], the time derivative of the argument of the latitude of the undisturbed circular orbit is written as

$$\dot{u}_0 = \sqrt{\frac{\mu}{R_0^3}} .$$

With this in mind, for computer calculations of *Sich-2-1* spacecraft deorbiting time, in order to optimize the machine time, in model (2), the transition for differentiation by independent variable u_0 is made.

Thus, with the use of model (1) [24], it has been proposed to analyze the time of *Sich-2-1* deorbiting from low-elliptical orbits with the specified limits on the inclination and the eccentricity. Further, with the help of model (2) [25, 26], the time of *Sich-2-1* deorbiting from polar, close to circular, solar-synchronous orbits will be calculated.

Determining the limits of DAD application to Sich-2-1. The following parameters of DAD are selected to determine the limits of DAD application for *Sich-2-1* spacecraft:

- the total surface area of the sail elements $S_{пови.в.е.}$ is equal to 13 m^2;
- the area of the midship section of the sail elements, with the maximum area being perpendicular to the aerodynamic flow of the approaching atmosphere, $S_{мід.в.е.}$ is equal to 4.5 m^2;
- the DAD mass m_{ACB} = 9 kg (5% of the mass of *Sich 2-1* spacecraft).

Thus, the time of spacecraft deorbiting with the use of the developed DAD that has the specified parameters, based on the atmosphere model [27], from the expected orbit whose main parameters are presented above in this paragraph is about 6.5 years (Fig. 6).

Hence, based on the obtained results, it may be concluded that the specified DAD is an effective means for deorbiting *Sich-2-1* spacecraft from the specified orbit to which the spacecraft is planned to be launched. The time of deorbiting from the specified orbit is much less than the maximum allowable life of defunct spacecraft in near-Earth orbits [4]. This indicates the possibility to reduce the DAD mass and dimensions. For this orbit, it is possible to reduce the total surface area down to 4.5 m^2 (the midship section area $_{і\,³ä.â.â}$ is 1.8 m^2), and the mass down to 4.5 kg that is 2.5% of the spacecraft mass.

For analyzing the limits of effective use of DAD for deorbiting *Sich-2-1* class spacecraft from other low Earth orbits we take the following mass and dimensions of DAD, with the DAD mass accounting for 5% of the spacecraft mass. The results of the calculations of the time for *Sich-2-1* deorbiting from low close to circular and low-elliptic near-Earth orbits of different dislocations are shown in Table.

Based on the analysis of the obtained results (Table), we may conclude that the limit of effective use of this DAD is an altitude of 730—750 km in orbits close to circular. In low-elliptical orbits, this DAD may be used at an altitude of, at most, 700 km in the perigee and 842 km in the apogee. Thus, we may conclude that this DAD meets the require-

Calculations of *Sich-2-1* Spacecraft Deorbiting Time from Low Earth Orbits of Different Dislocations

Close to circular LEO with eccentricity e = 0.001			
Orbit altitude, km	Inclination, deg.	Deorbit time	
600	90	3.56 years	
600	30	3.71 years	
700	60	16.4 years	
700	100	15.9 years	
750	80	28.6 years	
Low elliptical LEO with eccentricity e = 0.01			
Perigee altitude	Apogee altitude	Inclination, deg.	Deorbit time
600	740.82	80	5.9 years
700	842.85	20	24.8 years

ments for limiting the orbital life of *Sich-2-1* spacecraft class in LEO to 25 years.

If there are requirements for limiting the volume of DAD, it is possible to use a smaller drag element while maintaining the time of deorbiting the defunct spacecraft. This may be achieved by orienting and stabilizing the drag element that has a rigid connection with the spacecraft with the maximum area directed towards the aerodynamic flow of approaching atmosphere [28]. The studies have shown that such stabilization [29] results in increasing the midship section area for flat drag elements by 50% as compared with the calculations of the average midship section area in the case of undirected deorbiting. The orientation and stabilization can be made with the help of special control elements with permanent magnets [28] or with the use of motion control methods and residual life of electromagnets (magnetorks). These methods allow saving much onboard energy for stabilization as compared with the classical approaches, which is important for long-term missions. Also, it should be noted that the stabilization should be done up to an altitude of 550 km, the time of deorbiting to which accounts for 80—90% of the total mission time. Stabilization below 550 km is not efficient, because aerodynamic disturbances are significant and require a lot of onboard energy for compensation, and the saving of time for deorbiting the spacecraft from a given altitude to a dense atmosphere is 1—2 months, which is insignificant in long-term missions.

Conclusions and recommendations. Based on the analysis of the means of spacecraft deorbiting from LEOs and the requirements of *Pivdenne* Design Office to develop a deorbiting system for *Sich-2-1* spacecraft, it has been found that the use of deployable DAD is the most appropriate. As a result, DAD for *Sich-2-1* spacecraft (Figs. 3—5) has been designed. The mass and dimensions of this DAD have been calculated, given the limitation that the DAD mass should not exceed 5% of the spacecraft mass.

To calculate the deorbit time and to determine the limits of the effective use of DAD, two systems of differential equations in osculating elements have been used. This has allowed modeling the orbital motion of spacecraft in low-elliptical and close to circular orbits. Two mathematical models of orbital motion have been used in order to extensively study the spacecraft deorbit from orbits of different dislocations with the help of DAD in order to more accurately determine the limits of effective application. Studying with the use of only one of the models of orbital motion does not allow the analysis of spacecraft motion in the orbits of different dislocations, because of the limitations of each model.

The calculations have shown that the time of deorbiting *Sich-2-1* spacecraft from the expected orbit is about 6.5 years. The mass of DAD is 9 kg that is 5% of the spacecraft mass. In the case of increasing the time of deorbiting the *Sich-2-1* spacecraft from the design orbit after the end of operational life to 25 years, the DAD mass may be reduced to 4.5 kg. For a DAD mass of 9 kg, the limit of effective use of this DAD is an altitude of 730 to 750 km, in close to circular orbits of different dislocations, and at most, 700 km in perigee and 842 km in apogee, in low-elliptical orbits.

The results of the research have been implemented at *Pivdenne* Design Office, for choosing the parameters of drag system for deorbiting *Sich-2-1* spacecraft.

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РОЗРОБКА КОНСТРУКТИВНОЇ СХЕМИ ТА ВИБІР ПАРАМЕТРІВ АЕРОДИНАМІЧНОЇ СИСТЕМИ ВІДВЕДЕННЯ КОСМІЧНИХ АПАРАТІВ РОЗРОБКИ ДП КБ «ПІВДЕННЕ» З ОРБІТИ

Вступ. Для стабілізації середовища космічного сміття відпрацьовані космічні апарати та верхні ступені ракет-носіїв необхідно відводити з орбіти.

Проблематика. Проведений аналіз надувних аеродинамічних систем відведення космічних апаратів з орбіти показав, що вони є ефективним засобом відведення космічних апаратів з орбіти на висотах до 800 км, однак мають певні недоліки: ймовірність пошкодження фрагментами космічного сміття через чутливість матеріалу оболонки, а також ймовірність електростатичного пробою.

Мета. Розробка конструктивної схеми та вибір параметрів аеродинамічної системи відведення космічних апаратів, розроблених ДП «КБ «Південне», з орбіти.

Матеріали й методи. Методи механіки космічного польоту, математичне моделювання задач проєктування.

Результати. Розрахунки показали, що час відведення космічного апарату «Січ-2-1» із планованої орбіти складає близько 6,5 років при масі аеродинамічної системи відведення 9 кг, що складає 5% від маси зазначеного космічного апарата. Визначено, що у разі збільшення часу відведення космічного апарата «Січ-2-1» з планованої орбіти після завершення експлуатації до 25 років, масу аеродинамічної системи можна зменшити до 4,5 кг. При масі аеродинамічної системи відведення в 9 кг, межею ефективного застосування зазначеної аеродинамічної системи відведення є висоти від 730 до 750 км на близьких до кругових орбітах різної дислокації і висоти не більше 700 км в перигеї та 842 км в апогеї на малоеліптичних орбітах.

Висновки. Виходячи із вимог ДП КБ «Південне» до масових і габаритних параметрів засобу відведення, було розроблено конструктивну схему і проєктний вигляд аеродинамічної системи відведення, що розгортається. Особливістю конструкції є компактність, що забезпечується застосуванням пружинних механізмів і маловитратних мікроелектродвигунів, що розгортають аеродинамічні елементи. Така конструкція займає незначний об'єм на космічному апараті «Січ-2-1».

Ключові слова: космічне сміття, аеродинамічна система відведення, час відведення, проектні параметри системи відведення.