Analysis of the possibility of using the system for small satellite de-orbiting based on an aerodynamic stabilizer, taking into account the physical features of the Earth’s upper atmosphere

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Abstract. The paper presents an analysis of performance of a satellite de-orbiting system that utilizes a deployable aerodynamic stabilizer for controlling the angular motion of a spacecraft in order to ensure the correct orientation of the breaking impulse. The problem of determining the shape of the stabilizer, considering its performance requirements and technological constraints has been addressed. The effectiveness of proposed design of the stabilizer for the prototype spacecraft based on the “AIST” small satellite was evaluated using two methods – calculation of aerodynamic forces generated by the rarefied gas flow using the experimental coefficients of the form and stochastic modelling using software. According to the calculations, the proposed aerodynamic system would be capable of providing sufficient stabilizing influence to satisfy the required stabilization time and permissible angle of attack deviation constraints. The range of orbits, where the aerodynamic stabilizer would remain efficient, have been determined with regards to atmospheric properties and design constraints, imposed by mass and size limitations of the system and the de-orbiting time.

1. Introduction
Nowadays, the problem of generation of space debris is nearing a critical level due to the avalanche-like growth of the number of small spacecraft being brought to low Earth orbit. In the near future, several multi-satellite groupings will be deployed in low Earth orbit. The problem of satellite’s post-mission de-orbiting have been extensively studied and reviewed in literature [1, 2]. In Ref. [1], the advantages and disadvantages of different methods of de-orbiting of small satellites were identified. A solid-fuel propulsion system proved to be the most effective; however, the correct orientation of the brake impulse requires the attitude control system of the satellite to remain functional till the very end of the mission, reducing the reliability of this de-orbiting method.

The experience of using passive systems of orientation and stabilization shows that such systems are capable to provide accuracy of spacecraft orientation in the range from 5 to 10 degrees, which makes such a system suitable for solving the problem of stabilizing the spacecraft before deceleration. The aerodynamic stabilization system has a number of advantages in comparison with other known orientation systems, since it does not require orientation sensors and special actuating elements. The paper presents a feasibility study for using inflatable aerodynamic stabilizer as a part of a satellite de-orbiting systems. The Projected effectiveness of the system is evaluated, as well as the permissible
range of orbits, where such a system could remain effective while still satisfying the design and technological constraints, imposed by the prototype satellite capabilities. The simulations were conducted on a prototype satellite, based on the “AIST” platform, a small satellite family, developed and launched as a joint project between JRSC “Progress” and Samara University [3].

2. Formulation of the problem of satellite de-orbiting

The equation (1) shows the conventional form of the satellite’s rotational motion. The de-orbiting system under consideration requires the satellite to be stabilized before the breaking impulse could be fired. The source of the stabilizing moment would be the aerodynamic stabilizer. It is worth noting, that deployment of the stabilizer causes additional perturbations imposed both by the deployment itself and the changes of satellite’s mass and distribution and characteristic area.

\[ \dot{\theta} + \frac{3\mu I_x - I_y}{I_z} \sin \theta \cos \theta = -\omega_{eb} + M_{xc} - M_{zp}, \]  

(1)

where \( I_x, I_y, I_z \) are the main central moments of inertia of the satellite, \( M_{xc}, M_{zp} \) – projections of the control and perturbing moments on the z axis, \( R \) – radius of the orbit, \( \theta \) – angle of deviation of the satellite from the main axis.

The most significant disturbing factors are the aerodynamic drag, the magnetic field, the solar pressure, the gravitational fields of the Earth and celestial bodies. At altitudes less than 600 km from the Earth, density of the atmosphere is relatively high, and therefore the aerodynamic forces acting on the spacecraft are not negligible and can be used to create aerodynamic moments. The characteristics of the atmosphere were taken from the ubiquitous JB2008 model, commonly used in modeling of orbit transfer maneuvers, as, for example, in [4].

If the center of pressure of the aerodynamic forces does not coincide with the center of mass of the space vehicle, an aerodynamic moment arises that can be used to orient and stabilize the space vehicle.

Aerodynamic control moments are calculated using approximations, since the exact scheme of interaction between the surface of the space vehicle and the stabilizer with a rarefied flow is unknown. In addition, there are significant variations in atmospheric density associated with solar activity and diurnal rotation of the atmosphere together with the Earth [5]. In the general form, the moment arising due to the aerodynamic drag is determined by the expression (2).

\[ M_a = \frac{\rho V^2}{2} \sum_{j=1}^{n} (C_a)_j S_j y_j, \]  

(2)

where \( (C_a)_j \) – drag coefficient of the j-th surface, \( \rho \) – atmospheric density at satellite altitude, \( V \) – satellites’ flight velocity, \( S_j \) – midsection area, \( y_j \) – arm of the drag force of the j-th surface (distance from center of mass to center of pressure). Accurate estimation of density at given parts of trajectory largely determines the overall accuracy of the simulation. An approximation of atmospheric density, depending on the orbital conditions, was adopted from [6].

\[ \rho = \rho_0 \left(1 + F_1 C + F_2 C^2\right) + \left(S^0 \varphi (1 + F_1) C + S^+ \varphi C\right) + \left(H, C + S^+ \varphi\right) \frac{\Delta \varphi}{H} + \frac{1}{2} \mu \left(\frac{\Delta \varphi}{H}\right)^2 + \delta K \exp \left(-\frac{\Delta \varphi}{H}\right). \]

(3)

This density model includes not only diurnal variation (factors \( F_1, F_2 \) and scale height variation with altitude (factor \( \mu \)) but also semiannual variation (include in \( \rho_0, H, \mu \), season-latitude variation in He (factor \( S^0, S^+ \)), as well, as the variation of diurnal, season-latitude, geomagnetic activity with height (\( H, S^+, K \)). Without conducting experiments, it is hard to accurately evaluate the accuracy of the method, but according to [6] the implemented approach allows obtaining orbital coordinates for a low-orbit high-drag satellite with accuracy in the range from 200 to 500 m, which is sufficient for the task.
3. Dependence of shape and size of the aerodynamic stabilizer on the characteristics of the de-orbiting system

When creating a system of aerodynamic stabilization, one of the main challenges is the correct choice of the shape of the stabilizer that has to solve the following problems:

1. Ensuring maximum aerodynamic control torque in a given range of angular deviations of spacecraft;
2. Ensuring an acceptable value of the aerodynamic resistance for solving the problem of braking the spacecraft;
3. Ensuring the maximum steepness of the torque characteristic at angles of attack, tending to zero.

The stabilizer must also meet a number of technological and design requirements, such as manufacturing accuracy, deviations from the design form, ease of deployment, manufacturability, reliability, minimum weight, etc. The technological decisions on the material of the stabilizer and the method of its deployment impose limitations on its possible shape, in particular, for the inflatable product, the shape of the convex body of rotation with the minimum number of seams appears to be the most appropriate. Several potential configurations of the stabilizer in the expanded state are shown in Figure 1.

For bodies of simple shapes, such as plates, cylinders, cones, spheres and hemispheres, there are experimentally confirmed dependencies of aerodynamic coefficients, forces and moments. Figures 2 and 3 show the dependences for some of the coefficients of aerodynamic forces and moments acting on some simple-shaped bodies: 1 - plate, 2 - cylinder, 3,4 - cone (respectively the angle of half-opening 15 and 75 degrees), 5 – hemispheres.

Based on the aforementioned technical and technological requirements for characteristics of the stabilizer and known dependences of the coefficients of aerodynamic forces, we have decided to make a stabilizer in a shape consisting of two spherical elements and a truncated cone. The form is shown in Figure 4. The chosen form provides sufficient aerodynamic characteristics, in particular the relative
position of the pressure center and the high value of the recovery torque at small angles of attack, which positively affects the stabilization time of the apparatus equipped with the proposed system of retraction from orbit.

Figure 4. Selected configuration of the stabilizer of the de-orbiting system

Figure 5. Characteristic dimensions of the stabilizer for the de-orbiting system

Figure 5 shows the stabilizer image with its characteristic dimensions. By changing the characteristic dimensions, we can control the magnitudes of the aerodynamic coefficients, thus changing the values of the aerodynamic forces and moments.

The choice of the cone blunting radius is determined by the nature of the variation of the drag coefficient.

It is proposed to use the form with the half-angle of the conical shape $\beta = 10^\circ$ and the radius $\rho' = 1.5 \text{ m}$ as a result of the study of the change in the behavior of the aerodynamic moment of the stabilizer with the change in the half-angle of the conical part and the corresponding change in the geometry of the anterior and posterior spherical segments.

The calculation was carried out in parallel by two methods - analytically, according to the methodology outlined in [7] and by stochastic modeling using the DS2V software package [8]. Due to the high computational complexity of the rarefied gas flow simulation (calculation of a single case takes several days), stochastic modeling was carried out for only a few scenarios to confirm the results of calculations obtained from simplified analytical dependencies based on the experimentally obtained data. Figure 6 shows the results of numerical simulations of flow around the AIST satellite with a deployed stabilizer. The parameters of the model are given in Table 1.

Figure 6. Flow speed distribution

The geometric dimensions of the stabilizer, shown in Table 1, are sufficient to create damping aerodynamic moments that are necessary for solving the problem. The process of approximate calculation of the coefficients of aerodynamic forces and moments by the theory of local interaction was carried out according the algorithm described in [7]. Calculated dependences of the dimensionless
coefficients of aerodynamic forces on the angle of attack are shown in Figures 7 and 8. It should be noted that the values of the own aerodynamic moments of the stabilizer are small in comparison with the moments that arise when the stabilizer deviates from the center of mass of the system “satellite-aerodynamic stabilizer”, which allows not to take into account their effect on the motion of the system as a whole.

| Parameter            | Range of admissible values | Units |
|----------------------|---------------------------|-------|
| Orbit altitude       | 650000                    | m     |
| Flow speed           | 7535                      | m/s   |
| Atmospheric density  | $1.52 \times 10^{-13}$    | kg/m$^3$ |
| $\varphi$            | 65                        | deg   |
| $\rho$               | 0.87                      | m     |
| $\varphi'$           | 60                        | deg   |
| $\rho'$              | 1.5                       | m     |
| $\xi$                | 2.6                       | m     |
| $\beta$              | 10                        | deg   |

As a result of stochastic modeling, the values of the aerodynamic force in $X$ direction $4.297 \times 10^{-5}$ N and the torque about $X$ axis $5.243 \times 10^{-7}$ Nm.

Comparison of the values obtained as a result of calculation by analytical dependencies, with parameters calculated by stochastic modeling methods, makes it possible to conclude that the obtained values are valid with an accuracy provided by approximate calculation methods.

The computed values of the aerodynamic moments were used in the equations of the orbital motion of the space vehicle and the rotation of the spacecraft relative to its center of mass.

![Figure 7](image1.png)  ![Figure 8](image2.png)

**Figure 7.** Dependence of the magnitude of the aerodynamic drag coefficient of the stabilizer on the angle of attack  
**Figure 8.** Dependence of the magnitude of the aerodynamic lift coefficient of the stabilizer on the angle of attack

4. Results and discussions

Figures 9 and 10 show the result of the simulation of rotary motion of a spacecraft under the influence of the aerodynamic stabilizing system in different orbits. According to the calculations, the proposed aerodynamic system would be capable of providing sufficient stabilizing influence to satisfy the required stabilization time and permissible angle of attack deviation constraints up to the altitude of 500 km. As was expected, the effectiveness of the system highly depends on the altitude of the orbit and the corresponding state of the atmosphere.
After completion of the satellite stabilization process, a solid-fuel propulsion system is engaged, providing the necessary braking pulse to quickly and efficiently change the satellite’s orbital parameters in order to reduce its ballistic lifespan. Simulation of the small satellite de-orbiting system’s performance on the example of “AIST” (RS-41at, RS-43as) spacecraft with an orbit height of about 500 km has shown the potential effectiveness of the proposed system. A reduction in the ballistic lifespan for “AIST” (RS43as) from 25 years to 94 days, “AIST” (RS41at) - from 50 years to 40 days.

5. Conclusion
The problem of determining the characteristics of an aerodynamic stabilizer for satellites, populating orbits in the range from 300 to 500 km was considered. The range of orbits, where the aerodynamic stabilizer would remain efficient, have been determined with regards to atmospheric properties and design constraints, imposed by mass and size limitations of the system and the de-orbiting time. Implementation of the proposed system into the design of small satellites, launched into low-Earth orbits would ensure reliable and relatively rapid descent from orbit even in case of a complete failure of satellite’s onboard systems.

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