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DEVISING A DEORBITATION STRATEGY FOR KAZAKHSTANI'S KazEOSat-1 SPACECRAFT

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The object of this study is the process of deorbiting the KazEOSat-1 spacecraft, which has completed its active service life in low Earth orbit. The main problem is the lack of an effective technique to deorbit KazEOSat-1, taking into account its technical characteristics, orbital parameters, and the need to minimize risks to the environment and other objects in orbit.

As part of the work, a software model was built that takes into account the initial orbital parameters of the device, which are essential for planning and performing deorbiting maneuvers. The model is designed to accurately calculate the descent trajectory, taking into account the laws of celestial mechanics and the influence of atmospheric conditions. The optimal deorbiting strategy was selected based on an analysis of various methods for calculating orbital maneuvers aimed at reducing fuel consumption and minimizing environmental risks. This included a comparative analysis of existing approaches and the selection of the most suitable ones under the given mission parameters.

The results of the simulation using precise modeling methods in the MATLAB software environment allowed us to determine the main deorbiting parameters, such as the altitude at which the maneuvers begin, the required velocity impulses, the total fuel consumption, and the expected time before entering the dense layers of the atmosphere. Based on the obtained data, practical recommendations were formulated for the KazEOSat-1 deorbit. The first stage, the active controlled deorbit, is carried out by operating the low-thrust engine and braking by the Earth's atmosphere, allowing the spacecraft to descend from 758 km to 444 km in 2.5 days. The second stage, the passive uncontrolled deorbit, continues the descent to 103 km in 969 days, using only atmospheric braking. The third stage, the uncontrolled drop, begins after reaching 103 km and ends with a drop to the Earth in 834 seconds

Keywords: Spacecraft, deorbiting, satellite disposal, LEO, space debris, orbit, CubeSat

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1. Introduction

Kazakhstan is actively developing its space technologies and aims to significantly expand its presence in the global space services market by 2030 [1]. The first high-resolution Kazakhstani remote sensing satellite (RSS) was launched on April 30, 2014, and the second medium-resolution satellite, KazEOSat-2, was launched on June 20, 2014. To date, the initial estimated service life of both satellites has come to an end, and work is currently underway to extend their operational activity. However, despite the successful extension of their operational activity, there will come a time when it will be necessary to dispose of them in accordance with international standards for reducing space debris and protecting space infrastructure [2]. Therefore, the need to devise a deorbit strategy is becoming increasingly important. This is due not only to the need to comply with international norms and standards for reducing space debris but also to the requirements for maintaining sustainable space activities. Spacecraft (SC) completing their missions must be deorbited in a manner that minimizes risks to operational satellites and avoids the creation of additional debris. This need is also exacerbated by the upcoming launch of a new remote sensing satellite, KazEOSat-MR, planned for the coming years, which will also require the development of a decommissioning strategy upon completion of its mission.

Currently, developed space nations use a variety of approaches to the disposal of spent spacecraft and other objects. For example:

– NASA (National Aeronautics and Space Administration) actively uses final orbit programming, especially for launching spacecraft into so-called disposal orbits or burning them up in the Earth's atmosphere. NASA is also testing robotic systems such as RRM (Robotic Refueling Mission), which can not only refuel satellites but also participate in their disposal in the future.

– ESA (European Space Agency) adheres to strict mission completion standards, including the transfer of spacecraft into disposal orbits or their controlled deorbit. ESA has developed projects such as Clear Space-1, the purpose of which is to capture and dispose of space debris using specially designed spacecraft.

– JAXA (The Japan Aerospace Exploration Agency) also uses final orbit programming methods for its space missions, ensuring that spent spacecraft do not pose a risk to other satellites and missions. The use of a solar sail in the Japanese IKAROS project is an example of the successful use of a solar sail, demonstrating the promise of the technology for a wide range of applications, including potential use in spacecraft control and deorbiting. However, its use in deorbiting requires further research.

– CNSA (China National Space Administration), like other major space agencies, uses methods in which satellites, after completing their active service phase, are placed into a so-called graveyard orbit, or directly burned up in the Earth's atmosphere. However, for large missions, such as Beidou satellites or Tiangong station modules, CNSA devises strategies for controlled deorbiting or transfer to safe orbits in order to minimize risks to other space objects.

However, for Kazakhstan, which is striving to expand its presence in the global space services market, devising its own method for disposing of space debris is critically important. This will not only allow it to comply with international standards but also contribute to the sustainable development of the national space program, minimize risks to space infrastructure, and maintain environmental safety in orbit.

The problem addressed in this paper is the lack of an effective deorbit strategy for the KazEOSat-1 spacecraft, which is nearing the end of its active life, so the goal is to devise a deorbit method taking into account the technical characteristics of the said spacecraft, taking into account the minimization of fuel costs and the reduction of risks of damage to surrounding objects in orbit, as well as reducing the negative impact on the environment. This includes the selection of suitable methods for calculating orbital maneuvers, developing a software model for simulating orbital parameters, and adapting them to real spaceflight conditions.

Therefore, research on devising a deorbit method for KazEOSat-1, KazEOSat-2 and the upcoming KazEOSat-MR satellites is a strategically important task for the national space program of Kazakhstan, the success of which determines the long-term and sustainable use of outer space.

2. Literature review and problem statement

The issue of deorbiting spacecraft that have completed their active service life remains one of the main tasks of the modern space industry. The gradual increase in the number of objects in Earth orbit leads to an increased risk of collisions and the formation of space debris, which threatens the safety of future space missions. Devising effective deorbiting methods that allow for the control over the deorbiting process, minimizing costs and risks, is becoming increasingly important.

Today, special attention is paid to aerodynamic methods of spacecraft deorbiting, which use atmospheric drag to slow them down and then deorbit them. One of the most well-known approaches is the use of membrane sails. Such structures increase the area of the spacecraft, which accelerates its deorbiting due to increased interaction with the atmosphere. However, despite the effectiveness of this method, there are still unresolved issues related to maintaining a stable position of the spacecraft. External cosmic factors, such as solar radiation and gravitational disturbances, significantly complicate control over spacecraft orientation, which requires additional research to improve the reliability of such systems [3]. Another option is square sails with controlled rotation, which provide more precise control over the deorbiting process. Such systems are especially effective for spacecraft with a large area relative to their mass, as they allow for a significant reduction in descent time and make the process predictable. However, the issue of optimizing control systems, which remain complex and expensive, remains unresolved. This is due to the objective complexity of designing dynamic orientation systems for square sails, as well as the insufficient concentration of research on simplifying such systems and reducing the requirements for equipment calibration accuracy [4].

A model for analyzing the stability of a deorbiting system using pyramidal sails is presented in [5]. This model takes into account such parameters as the dimensions of the support elements and the opening angle of the structure, which allows for the configuration to be optimized to ensure stable orientation. However, the question of the influence of the space environment on such systems remains open. This issue remains unresolved due to objective difficulties in modeling external influences such as solar radiation and atomic oxygen, as well as a limited number of orbital experiments. In addition, the focus of most studies was on the geometric optimization of the design, which led to insufficient consideration of the influence of the external environment. An innovative approach to deorbiting is the use of inflatable spherical devices. They are characterized by simple design, low energy consumption, and minimal mass, which makes them especially attractive for use in low orbits [6]. Such systems effectively increase atmospheric drag, facilitating rapid deorbit of the spacecraft. However, issues related to their reliability remain unresolved. This is due, on the one hand, to objective difficulties in designing damage-resistant materials, and on the other hand, to the limited number of orbital tests due to the high cost of such experiments.

Inflatable structures also increase the surface area of the spacecraft, which increases atmospheric drag and facilitates more efficient deorbiting. Studies such as [7] use Monte Carlo simulations to estimate the effect of altitude on the flow dynamics around an inflatable deorbiter, allowing for accurate modeling of the device's behavior under various conditions. However, issues related to the resistance of inflatable structures to damage from micrometeorites and space debris, as well as the reliability of their deployment in low-density atmospheric conditions, remain unresolved. This may be due to the difficulty of reproducing such conditions in experimental studies.

In [8], the effect of deorbiter geometry on thermal and dynamic characteristics is analyzed, which is critical for the design of reliable deorbiting systems. However, issues related to the selection of materials resistant to damage from micrometeorites and space debris, as well as to modeling the interaction of the deorbiter with high-speed flows in orbital conditions, remain unresolved. This is due to the difficulty of reproducing space environment conditions in ground-based experiments and the limitations of current computational modeling methods. Paper [9] studies the effect of an electrodynamic tether (EDT) on the reduction of the orbital altitude of satellites. It was found that the EDT effectively creates Lorentz resistance, which allows for a significant reduction in deorbit time without using fuel. However, issues related to the stability and controllability of the satellite during tether operation remain unresolved. This is due to the complexity of modeling ionospheric conditions, plasma density variability, and limitations of current computational models of the Earth's magnetic field. These limitations can be overcome by building more accurate models of tether interaction with the environment, which will allow for dynamic changes in the ionosphere and geomagnetic conditions in real time.

In [10], the vulnerability of the tether to collisions with space debris was assessed and it was demonstrated that they can be used to bring a 24-kilogram satellite down from an altitude of 600 km in less than 100 days. However, issues related to the optimization of the tether parameters, including its length, material, and shape, which affect the efficiency of interaction with the ionosphere and resistance to damage, remain unresolved. The reason for these limitations is the insufficient accuracy of the models describing the influence of micrometeorites and space debris, as well as limited data on the behavior of tether materials under real space conditions. To resolve these issues, additional research is required to design more reliable materials and refine the models of tether interaction with the environment.

Based on the above review, it can be argued that existing deorbitation methods are not always suitable for spacecraft with specific characteristics. Therefore, devising our own deorbitation method for the Kazakhstani remote sensing satellite KazEOSat-1 is a relevant task due to several factors. First of all, the proposed method will allow for a safe and controlled descent of the KazEOSat-1 spacecraft without causing damage to other space and ground objects. For this purpose, the characteristics and fuel reserves of a specific spacecraft are taken into account at each stage. Secondly, devising a national deorbitation method contributes to the development of local scientific and engineering competencies, strengthening the country's technological independence. In addition, it allows for more accurate compliance with international safety and space debris management standards [11, 12], optimized costs, and better managed risks associated with the completion of space missions. Thus, devising a proprietary deorbitation method is an important step towards improving the safety and efficiency of Kazakhstan's space program.

3. The aim and objectives of the study

The objective of this study is to devise and experimentally test a deorbit strategy for the KazEOSat-1 Earth remote sensing (ERS) spacecraft. The proposed strategy is aimed at minimizing fuel costs and the probability of collision with other space objects. In addition, the study involves the calculation of a safe drop zone for the residual elements of the spacecraft on the Earth's surface, while ensuring the longest possible preservation of control over the spacecraft at the final stage of its operation.

To achieve this goal, the following tasks were set:

– to analyze various methods for calculating orbital maneuvers to minimize fuel costs and reduce the risk of damage, as well as assess their applicability to KazEOSat-1;

– to develop an algorithm for a software model that takes into account the initial orbital parameters of the Kazakhstani ERS spacecraft KazEOSat-1, necessary for successful deorbiting, and calculate its descent trajectory taking into account the laws of celestial mechanics and atmospheric factors;

– using the model built, determine the main parameters of deorbiting the Kazakhstani remote sensing spacecraft KazEOSat-1, including altitude, velocity impulses, fuel consumption, and time, to ensure the effective execution of deorbiting maneuvers.

4. The study materials and methods

The object of our study is the deorbiting strategy for the Kazakhstani KazEOSat-1 remote sensing spacecraft. The study is aimed at devising approaches, methods, and tools to ensure the implementation of the deorbiting process in accordance with the strategy being developed. The main hypothesis of the study assumes that the use of theoretical and numerical methods for modeling orbital maneuvers in combination with calculations in the MATLAB environment makes it possible to effectively devise deorbiting strategies, thereby ensuring the accuracy of calculations and compliance with international safety requirements.

The following assumptions were accepted in the calculations.

The Keplerian orbital elements and TLE data fully characterize the motion of the spacecraft in a given orbit.

Atmospheric drag and other disturbing factors (for example, gravitational disturbances from the Moon and the Sun) can be taken into account in the calculations through standard models.

The formulas of classical mechanics and orbital dynamics used adequately describe the processes of the spacecraft's motion in orbit.

To simplify the calculations, the following simplifications were adopted:

– application of standard methods for TLE data processing without taking into account possible errors in the data transmitted from the spacecraft;

– consideration of the idealized spherical shape of the Earth without taking into account geophysical characteristics, such as gravitational field inhomogeneities;

– using MATLAB as the only tool for numerical modeling, excluding additional environments for analysis, which allowed us to concentrate on the development of individual scripts.

Deorbiting a spacecraft from orbit to Earth is a complex process that requires significant professional efforts. One of the main stages of strategy development is the construction of a software module that provides modeling of various deorbiting scenarios and makes it possible to choose the best option that takes into account potential difficulties and meets the requirements of international standards.

Currently, many software tools and environments are available for performing complex calculations. In this study, the MATLAB environment, specially designed for complex mathematical calculations, was chosen for modeling and calculating the parameters of spacecraft deorbiting. In this environment, scripts are created based on the formulas of classical mechanics and orbital dynamics, which makes it possible to obtain results in the form of numerical parameters of deorbiting and corresponding plots.

The first stage of calculations is to determine the initial altitude of the spacecraft in its working orbit at the moment of deorbiting. For this purpose, a software module is used in the form of a script that we developed in the MATLAB environment, as well as TLE elements (two-line orbital elements) obtained during the communication session of the satellite with the ground control segment. The script is designed to calculate the Keplerian orbital elements of the selected spacecraft and their subsequent use in the deorbiting process. The script's operation includes loading a .txt file with TLE data, reading and processing the lines of the source file to determine the orbital elements, as well as outputting the current location and graphical display of the spacecraft position at the time of calculation, using orbital dynamics formulas.

Fig. 1 shows the MATLAB program interface, which displays all the elements of the working orbit of the spacecraft at the time specified in the original TLE file. The interface demonstrates the calculated data based on orbital dynamics formulas, necessary for predicting the spacecraft's motion. The screen also displays the initial data on the Keplerian elements used for calculations during the deorbiting of the spacecraft.

After determining the orbital altitude, we can begin modeling the process of deorbiting the spacecraft to Earth. It is important to consider how exactly the deorbiting of the spacecraft occurs, which, as is known, is divided into two main stages: exoatmospheric and atmospheric. For a more complete understanding of these stages, their detailed explanation is given below.

Fig. 2 shows conditional boundary of the atmosphere, dividing the deorbiting of the spacecraft into several stages. Up to an altitude of 100 km, deorbiting is classified as exoatmospheric; after crossing this boundary, the spacecraft enters the atmospheric section of deorbiting. In addition to the conditional boundary, the figure shows the expected deorbiting trajectory and various parameters determined during the operation of the script for modeling the deorbiting of the spacecraft.

Thus, the orbital velocity of the spacecraft at point C, denoted as V_c, is the velocity of the spacecraft at the start of the maneuver to transition to the deorbit trajectory. After performing the braking maneuver, the spacecraft reaches velocity V_d, which is less than the initial orbital velocity V_c and corresponds to the start of motion along the deorbit trajectory. At point B, where the spacecraft enters the dense layers of the atmosphere, its velocity is denoted as V_e.

The radial component of the spacecraft velocity at the moment of transition to the deorbit trajectory, denoted as u_r, is the vertical component of the velocity relative to the center of the Earth at point C. The change in the trajectory angle during the transition from orbit to the deorbit trajectory, $\Delta\theta_d$, shows the change in the direction of motion of the spacecraft. The angle θ_c, which is the angle between the direction of the velocity V_c and the local horizon at point C, describes the inclination of the spacecraft trajectory relative to the surface of the Earth.

The orbit of the spacecraft is inclined relative to the equatorial plane of the Earth by an angle ψ, which characterizes the position of the orbit in space relative to the equator. The angle of entry into the atmosphere at point B, denoted as θ_e, describes the inclination of the trajectory of the spacecraft relative to the horizon at the moment of the beginning of the atmospheric vault. The angle β reflects the position of the landing point relative to the center of the Earth, and φ indicates the latitude of this point on the surface of the Earth.

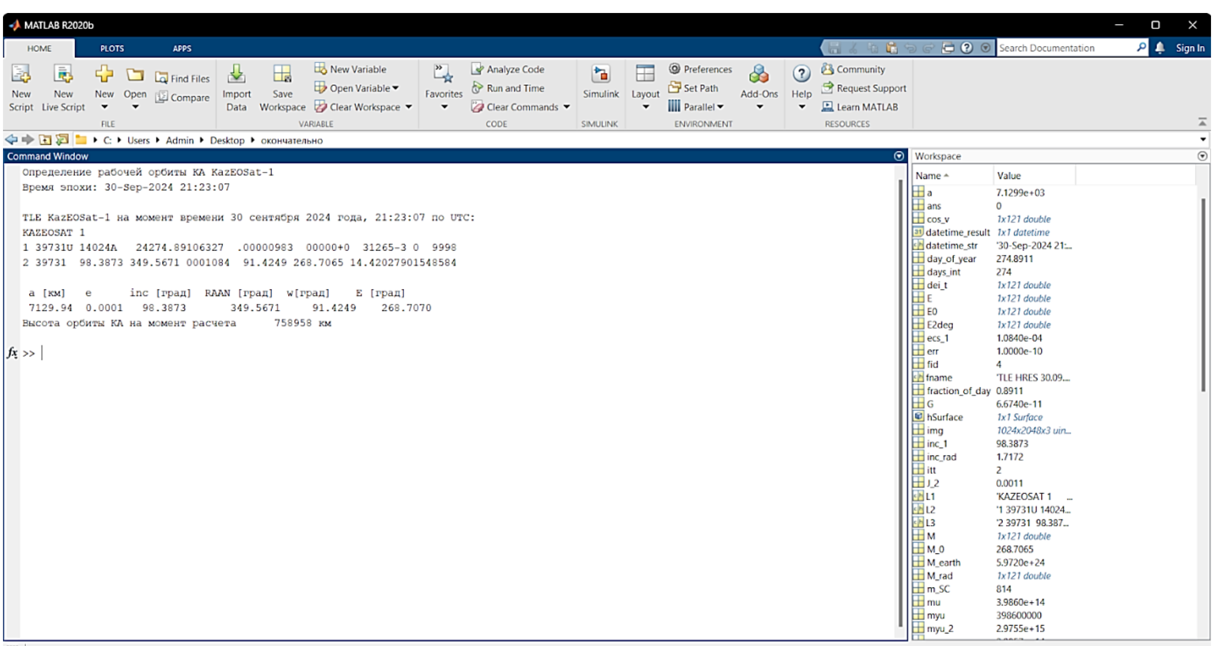


Fig. 1. MATLAB program workspace

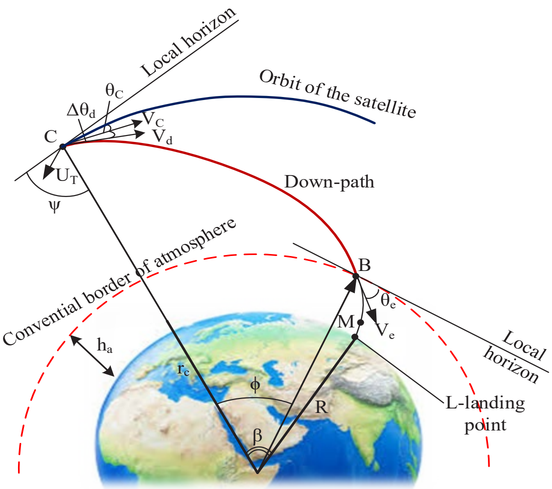


Fig. 2. Spacecraft deorbit trajectory and conditional atmospheric boundary

The radius of the Earth, denoted as R, is measured from the center of the Earth to the surface at the landing site of the spacecraft. The distance from the center of the Earth to the spacecraft at point C is denoted as r_c and is the radius of the spacecraft's orbit at the moment of the start of deorbiting. The height h_a at point C shows the height of the orbit above the Earth's surface at the place where the maneuver of transition to the vault trajectory begins. The end point of the deorbiting trajectory is the landing point L, where the spacecraft reaches the Earth's surface.

The orbit of the artificial Earth satellite (AES) is the orbit along which the spacecraft moves before the start of the deorbiting maneuver. Point C indicates the initial position of the spacecraft at the start of the deorbiting maneuver, and point B indicates the location of the spacecraft upon entry into the dense layers of the atmosphere.

After clearly defining the boundaries of the deorbiting stages and setting the initial altitude of the spacecraft, it is necessary to proceed to calculations for the extra-atmospheric portion of the deorbiting. This stage can be divided into several parts depending on the characteristics of the spacecraft. For example, deorbiting can be done gradually using onboard DMT or structured into active and passive phases. The latter option is determined by the amount of fuel that provides the active descent of the apparatus in the extra-atmospheric section.

After completing the calculations for the active descent of the spacecraft, two scenarios are possible:

1) the spacecraft descends to an altitude at which it intersects the dense layers of the atmosphere and continues its descent to the Earth's surface;

2) the spacecraft does not reach an orbit that is low enough, and then it is necessary to calculate its passive descent, which is carried out under the influence of atmospheric drag.

The scripts for the active and passive descent calculate the time to reach the altitude at which the spacecraft will enter the dense layers of the atmosphere for further descent to Earth. Additionally, calculations are performed of the orbital altitude, the period of revolution around the Earth, and the number of revolutions made per day.

The last script developed for the calculations is designed to simulate the deorbiting of the spacecraft in the dense layers of the atmosphere. When, after passive descent, the spacecraft orbit reaches the 100 km mark, the atmospheric deorbiting stage begins. This script calculates such parameters as the change in speed, the angle of entry, and the altitude of the orbit, completing the final stage of the spacecraft deorbiting. Below are the results of the calculations performed for the deorbiting of the KazEOSat-1 spacecraft. The specified theoretical and experimental methods of deorbitation calculations were selected due to their effectiveness in solving the tasks set. These methods provide integrated modeling of all stages of deorbitation of the Kazakhstani spacecraft KazEOSat-1 – from initial orbit calculations to final deorbitation in dense layers of the atmosphere, allowing us to obtain the necessary parameters for safe and effective completion of the mission.

5. Results of calculating the trajectory parameters using the KazEOSat-1 deorbitation strategy

5. 1. Analysis of various orbital maneuver calculation methods and evaluation of their applicability to KazEOSat-1

Deorbitation of the Kazakhstani KazEOSat-1 Earth remote sensing spacecraft from orbit to Earth is a complex task that requires taking into account many factors to ensure the safety and efficiency of the operation. A proper deorbitation strategy minimizes the risks of damage to the spacecraft, optimizes fuel consumption, and ensures precise achievement of the specified landing point. The process of devising a deorbitation strategy includes evaluating various orbital maneuver methods, modeling motion parameters at each stage, and optimizing the sequence of actions to achieve the best results. This paper discusses approaches to constructing an optimal deorbitation trajectory, including calculating the initial orbit parameters, active and passive descent stages, and maneuvers in dense layers of the atmosphere.

An important part of devising a deorbit strategy is choosing the most effective method that minimizes risks and optimizes the trajectory for safe descent of the spacecraft. Analysis of known methods for calculating orbital maneuvers was conducted. In addition to minimizing fuel costs and reducing the risk of damage, such criteria as applicability to the Kazakhstani spacecraft KazEOSat-1, ease of implementation, fuel costs, and high calculation accuracy were also taken into account.

The Patched Conic Approximation method divides the task into stages: movement around one body and transition to another. The method was rejected because it is not effective for missions in a sun-synchronous orbit [13].

The Lambert method solves the problem of movement between two points at a given time. It was rejected because it has extremely high computational complexity and requires taking into account disturbances [14]. This method is more applicable to interorbital and interplanetary maneuvers.

The method using low-thrust engines [15] is applicable for the extra-atmospheric part of deorbit. Optimization of the deorbiting scheme and development of the deorbiting strategy for the Kazakhstani KazEOSat-1 remote sensing spacecraft begins with calculations for various deorbiting options. The method involves increasing the orbital eccentricity and reducing the perigee to the dense layers of the atmosphere, which ensures the fastest deorbiting with minimal fuel costs. In this method, the impulse is applied at the apogee, where the speed is minimal, and the difference in speeds for deorbiting is insignificant. However, this method is not always applicable since a sharp change in altitude and the elongation of the orbit increase the risk of damage to the spacecraft. In addition, a significant difference in speeds can lead to an unpredictable change in the spacecraft orbit, up to an increase in altitude.

Among the methods considered, the Gohmann method stands out in that it allows maintaining a circular orbit and gradually reducing the altitude, which reduces the risk of damage to the spacecraft. However, this method requires significant fuel consumption. An alternative is continuous operation of the engines, which ensures a smooth descent, but requires even more resources due to the lack of passive sections, which leads to continuous fuel combustion and can lead to an undesirable change in orbit.

After comparing these methods, an optimal deorbit strategy was selected, combining elements of the Gohmann method for reducing the spacecraft's altitude in a transfer orbit while maintaining a circular orbit and the method of using impulses at perigee and apogee. This strategy minimizes the risks of damage to the spacecraft, while maintaining a relatively low consumption of fuel and resources.

The bi-elliptical method (Bi-Elliptic Transfer) is a more complex version of the Gohmann method using three impulses to transition through a highly elliptical orbit. It is not applicable in the strategy being devised due to the long time it takes to perform maneuvers. The choice of the deorbiting method was based on a comprehensive analysis of known approaches, taking into account the specificity of the mission of the Kazakhstani spacecraft KazEOSat-1. The main selection criteria were minimizing fuel costs, reducing the risk of damage to the device, ease of implementation, and high accuracy of calculations. After comparing the Gohmann, bi-elliptic maneuver, Lambert, and piecewise conical approximation methods, the strategy combining elements of the Gohmann maneuver and impulses at perigee and apogee was recognized as optimal, which ensures a balance between safety and economic efficiency.

5. 2. Development of an algorithm for calculating and modeling the initial orbital parameters of KazEOSat-1 for subsequent deorbiting

Our study has developed an algorithm for calculating and modeling the initial orbital parameters of a spacecraft for deorbiting. Special feature is the adaptation of approaches previously used only for orbit correction to the tasks of deorbiting a spacecraft, taking into account the technical characteristics of the spacecraft, orbit parameters, environment, and fuel reserves. The development of the algorithm is based on the adaptation of orbit correction methods for deorbiting tasks, taking into account the technical characteristics of the spacecraft, orbit parameters, and external factors. The algorithm includes the stages of calculating the initial orbital parameters, optimizing the trajectory using propulsion system impulses, and modeling passive deorbiting under the action of aerodynamic drag. It is based on numerical methods, integration of telemetry data, and visualization of results.

The spacecraft (SC) deorbiting algorithm includes several stages, starting with orbit construction. First, TLE data is obtained from the spacecraft telemetry or open sources such as CelesTrak, which are converted into ephemerides – coordinates x, y, z, and velocities v_x, v_y, v_z. Based on this data, a predicted orbit of the device is constructed, for example, for one orbit (~1.5 hours). For clarity, a 3D model of the orbit is built indicating the initial reference point, but this is not necessary for calculations.

After this, the active phase of deorbitation begins, during which the working orbit is determined and the points of application of the propulsion system impulses are calculated. The first impulse changes the orbit parameters, and the second, applied at the opposite point, corrects it to maintain a near-circular shape. This approach allows for effective control of orbit changes. Maneuvers are performed on each orbit until the fuel is completely consumed, which allows for a consistent decrease in the orbital altitude. At this stage, it is assumed that other objects in orbit are not taken into account since the calculation is focused exclusively on the spacecraft trajectory.

When the fuel runs out, the passive phase of deorbitation begins. At this stage, the spacecraft continues to descend under the action of the aerodynamic drag of the upper layers of the atmosphere. Gradually, the orbit loses altitude until the spacecraft enters the dense layers of the atmosphere.

The final stage is entry into the atmosphere, where braking, destruction, and combustion of the spacecraft occur due to friction with the atmosphere. At this stage, the time of the spacecraft's descent and its interaction with the dense layers of the atmosphere is calculated. The algorithm provides a full cycle of deorbiting, starting with calculations of orbital parameters and ending with safe completion of the mission.

Fig. 3 shows the algorithm for calculating and modeling the initial orbital parameters in the form of a diagram.

To start the calculations, it is necessary to set the initial data of the device, namely TLE, which are transmitted from the spacecraft during its approach to the ground segment. Such data contain information about the current position of the device in orbit at the time of transmission. TLE can be acquired from specialized electronic resources, at which specialists in tracking space objects and space debris update data daily in accordance with the established schedule of information transmission.

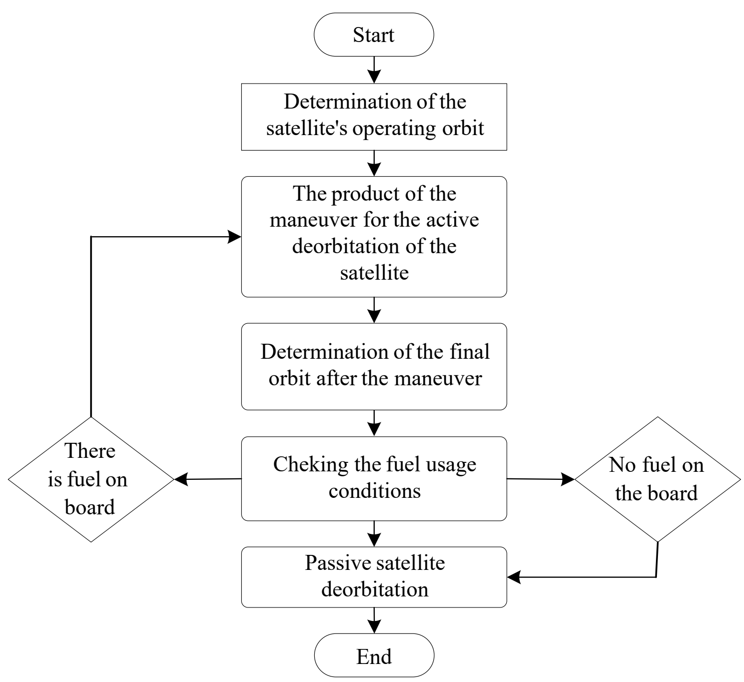


Fig. 3. Algorithm for calculating and modeling initial orbital parameters

For remote sensing spacecraft flying over ground stations approximately three times a day, the data is also updated with the same frequency. The algorithm for obtaining this data includes the following steps: to obtain information about a specific spacecraft, you need to visit the Celestrak website, enter its name in the search bar, and find the orbital parameters presented as two-line TLE elements. In this study, calculations are carried out for the Kazakhstani spacecraft KazEOSat-1, designed for remote sensing, the data for which are obtained in a similar way, entering the name of the device, and extracting the necessary orbital data. TLE KazEOSat-1 at the time of September 30, 2024, 21:23:07 UTC:

KAZEOSAT 1:
1 39731U 14024A 24274.89106327 .00000983 00000+0 31265-3 0 9998.
2 39731 98.3873 349.5671 0001084 91.4249 268.7065 14.42027901548584.

Thus, based on the TLE data, the location of the spacecraft at the start of the deorbiting calculations is considered. For this purpose, the MATLAB software was used in this work. The algorithm for entering data into the program includes the following stages:

1. Obtaining TLE data and saving them in the computer's workspace.
2. Opening the MATLAB program and running the script designed to determine the initial orbit of the spacecraft.
3. Using the saved file, the script is launched, providing the ability to select a TLE file previously saved on the computer.
4. Obtaining the necessary data on the working orbit of the spacecraft.

After executing the algorithm and working with the script in the MATLAB environment, the data necessary for further modeling of the deorbiting process were obtained.

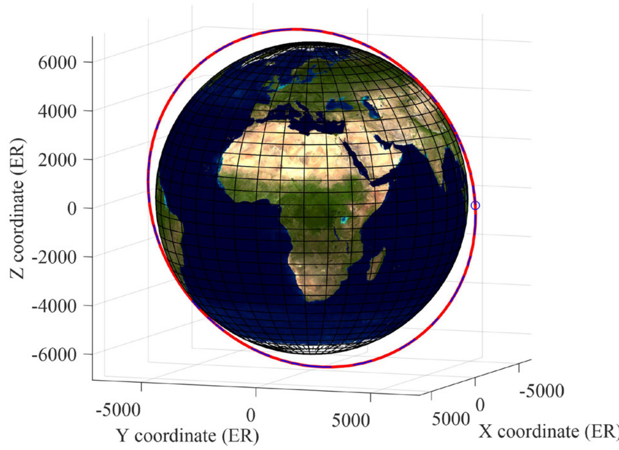
When entering the TLE data of the elements for the Kazakhstani KazEOSat-1 remote sensing spacecraft into the MATLAB script, the results were acquired as of September 30, 2024, 21:23:07 UTC. The provided results present the Keplerian orbital elements without taking into account external disturbances. Based on these elements, using the spacecraft orbital altitude, we calculate the spacecraft coordinates in space along the x, y, z axes.

The results are entered into the first script to calculate the spacecraft orbital altitude at the start of the modeling. This parameter serves as a starting point for further calculation of the active dome of the spacecraft. Formula (1) is used to determine the orbital altitude:

$$h=\sqrt{(x)^2+(y)^2+(z)^2}-R, \qquad (1)$$

where x, y, z are the coordinates of the spacecraft relative to the center of the Earth; R is the radius of the Earth.

According to the first script used to calculate the orbital altitude based on formula (1), as well as in accordance with the manual calculations, the initial altitude of the spacecraft in its working orbit was 758 km. This result serves as the initial parameter for further modeling of the spacecraft deorbiting. Fig. 4, *a* shows visualization of the spacecraft's working orbit for one full revolution, indicating its position at the time of the calculations. For a better visual representation of the orbital parameters, Fig. 4, *b* shows an image of the orbit without an image of the Earth.



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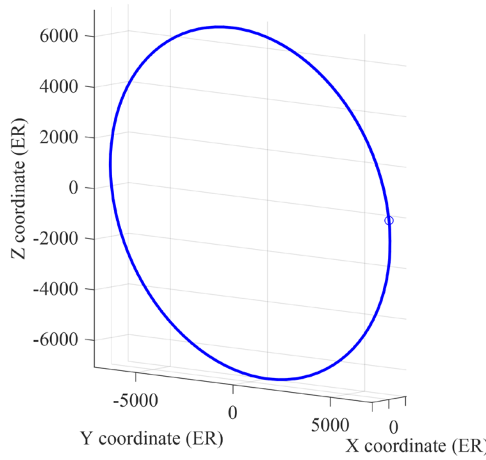

b

Fig. 4. The working orbit of the spacecraft and its location at the beginning of the calculation: *a* — with the image of the Earth; *b* — without the image of the Earth

In Fig. 4, *b*, the blue line in the image shows the trajectory of the spacecraft for one revolution around the Earth. The initial point of the calculation is the current position of the spacecraft on September 30, 2024, at 21:23:07 Coordinated Universal Time (UTC), which serves as the starting point for calculating the orbital parameters. The small blue circle in the image indicates the initial location of the spacecraft at this moment. When modeling the orbit, all the main parameters and scales were carefully taken into account, which ensured high reliability of the data. These calculations represent the orbit of the spacecraft for one revolution, starting from the specified time. However, it should be taken into account that minor factors, such as gravitational disturbances and atmospheric drag at low altitudes, can slightly change the orbit. Their influence in this case, however, is not significant.

Thus, the presented orbit of the spacecraft for one revolution around the Earth is the most accurate model built on the basis of all the initial data. This calculation serves as the basis for further modeling of the process of deorbiting the spacecraft to the Earth since the precise determination of the initial altitude and position of the spacecraft is an important step in devising an optimal deorbiting strategy.

To further simplify the calculations of the spacecraft deorbiting, we assume that its inclination remains unchanged ($\Delta i=0$), and the transition between orbits is performed coplanarly. A coplanar transition involves a transition from one orbit to another without changing the inclination of the spacecraft. Within the framework of this assumption, the spacecraft orbit can be represented in a two-dimensional plane along the x and y axes. This approach is applicable only in cases where both orbits have the same inclination, which allows for effective modeling of transition maneuvers.

The first developed script calculates the active descent of the KazEOSat-1 remote sensing spacecraft using the DMT installed on board the spacecraft. The data taken for the calculations are given in Table 1.

Table 1 give characteristics of the KazEOSat-1 spacecraft taking into account the fuel supply on board.

The spacecraft is designed for remote sensing of the Earth. It is located in a sun-synchronous low near-Earth orbit. The orbit characteristics are given in Table 2. Table 3 gives characteristics of DMT used in the KazEOSat-1 spacecraft:

Table 1

Characteristics of the KazEOSat-1 spacecraft

Characteristic	Description
Dry mass of the spacecraft	747 kg
Mass of the spacecraft with fuel, initial	830 kg
Mass of fuel, initial	83 kg
Mass of the spacecraft with fuel at the time of calculations	814 kg
Mass of the spacecraft fuel at the time of calculations	67 kg
Lifespan of active existence	7 years
Platform	Leostar-500-XO
Dimensions of the spacecraft	2.10×3.70 m
Power	1,200 W
Engine on board	1N, one-component, hydrazine.

Table 2

The orbit characteristics of KazEOSat-1 spacecraft

Spacecraft position	738×742 km
Orbital semi-major axis	750 km
Orbital inclination	98.5 degrees
Orbit classification	Low near-Earth
Orbit type	sun-synchronous

Table 3

1N engine characteristics

Characteristic	Value
Nominal thrust	1 N
Range of usable thrust	0.320–1.1 N
Nominal specific impulse	220 s
Range of usable specific impulse	200–223 s
Nominal mass fuel consumption	0.44 g/s
Range of usable mass fuel consumption	0.142–0.447 g/s
Minimum portion impulse	0.01–0.043 N
Mass of engine with nozzle section	290 g
Fuel	Hydrazine (N_4H_4)
Amount on board the spacecraft	4 units
Angle between thrust and direction of motion	30 degrees

Formulas (2) to (9) are used to calculate the active descent.

However, since the spacecraft has a DMT, a sharp transition from one orbit to another is impossible; only a total transition with a gradual decrease in the orbital altitude is necessary. In this case, the formulas are transformed as follows:

$$\Delta V_{1n}=V_{an}-V_{kp1n}, \tag{2}$$

where ΔV_{1n} is the first velocity impulse during braking, which is the difference between the velocities at the apogee of the transfer orbit and the initial circular orbit of the n-th order, m/s.

The second impulse for the transition from the transfer orbit to the final one is determined as follows:

$$\Delta V_{2n}=V_{kp2n}-V_{pn}, \tag{3}$$

where ΔV_{2n} is the second velocity impulse during braking, which is the difference between the velocities at the perigee of the transfer orbit and the final circular orbit of the n-th order, m/s.

The total velocity impulse is determined by summing all the applied impulses for deorbiting:

$$\sum_{0}^{n}\Delta V_n = \Delta V_{1n} + \Delta V_{2n}. \tag{4}$$

The mass required for transfer to the disposal orbit is determined by the parameters of the engine and the required impulse.

The mass of fuel for the first impulse is determined as follows:

$$m_f = m_{KA}\cdot\left[1-e^{\frac{\Delta V}{I_{si}}}\right], \tag{5}$$

where m_{SC} is the mass of the spacecraft, e is the Euler number equal to 2.7182818284, I_{si} is the specific impulse of the spacecraft in m/s, ΔV is the maneuver velocity impulse.

However, given that we take a fixed time of activation of the propulsion system, the mass of the burned fuel can be calculated as follows by modifying formula (5):

$$m_f = m\cdot\Delta t, \tag{6}$$

where m_f is the mass of fuel, kg, m is the mass of fuel consumed per 1 second of engine operation, Δt is the engine operating time.

In this case, the velocity impulse is expressed by the following formula:

$$\Delta V = \frac{F}{m_S}\cdot\Delta t,$$

where ΔV is the velocity change impulse, m/s; F is the thrust produced by the engine during operation, N; m_S is the spacecraft mass taking into account fuel, kg; Δt is the engine operating time, s.

However, it should be taken into account that the spacecraft has four engines located on the sides, with an angle α between the direction of thrust and the movement of the spacecraft.

Therefore, the spacecraft thrust will be as follows:

$$F=4\cdot F_N\cdot\cos\alpha, \tag{7}$$

where F_N is the nominal thrust of one engine, α is the angle between the direction of thrust and the spacecraft's motion.

We must also remember that we take into account the Earth's atmosphere as a factor influencing deorbiting, so we must also add the braking force of the spacecraft against the atmosphere:

$$F_A = C_{xa} S_M \frac{\rho V^2}{2}, \quad (8)$$

where C_{xa} is the spacecraft drag coefficient; S_M is the spacecraft area from the frontal part during the drop; ρ is the density of the atmosphere during the spacecraft drop; g is the acceleration of gravity equal to 9.81 m/s^2; V is the spacecraft speed.

So, to determine the braking impulse:

$$\Delta V = \frac{F + F_A}{m_S} \cdot \Delta t, \quad (9)$$

where ΔV is the velocity change impulse, m/s; F is the thrust produced by the engine during operation, F_A is the spacecraft braking force against the atmosphere, N; m_S is the spacecraft mass taking into account the fuel, kg; Δt is the engine operating time, s.

In this process, the spacecraft is deorbited due to the braking impulses created by DMT, as well as due to the effect of the Earth's atmosphere.

A visual example of the deorbiting process is shown in Fig. 5, where the initial orbit of the spacecraft is shown by the red line. This is the same orbit as in Fig. 4, *b*, but here it is shown as a two-dimensional diagram, which simplifies visualization for coplanar orbits. The active portion of deorbiting, implemented by turning on the engines and using fuel, is indicated by the black line, which maintains a constant length on the diagram since the pulse delivery time remains the same. The passive portion of deorbiting, occurring between two starts of the propulsion system, is shown by the green line; here braking is performed exclusively due to the resistance of the Earth's atmosphere. The Earth's radius is shown as a blue line for better visual perception of the spacecraft deorbitation.

The diagram of the KazEOSat-1 remote sensing spacecraft deorbitation during the impulsive braking created by DMT is shown in Fig. 5.

A significant decrease in the spacecraft orbital altitude is achieved by braking maneuvers performed by DMT. The main task in this process is to ensure a uniform descent and maintain a circular orbit. For this purpose, two braking impulses are used, each lasting 600 s, which are applied at opposite points of the orbit, with a separation angle of 180°. This distribution of impulses is explained by the need to recharge the engine before the next activation and is also the optimal DMT operating mode for the KazEOSat-1 remote sensing spacecraft.

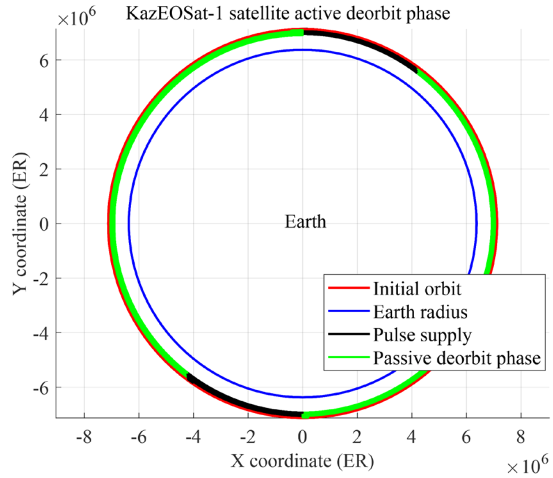

Fig. 5. Diagram of spacecraft deorbiting during impulsive braking of DMT

Both impulses are performed during one revolution around the Earth, and their combined effect leads to a decrease in the orbital altitude in one revolution. Our study contains detailed data on the deorbitation process, but for ease of analysis, the results were reduced to displaying changes in the orbital altitude with a step of 50 km. This allows us to highlight the main points of deorbitation and get a general idea of the dynamics of the process. All the data obtained are summarized in Table 4, which contains information on spacecraft deorbitation using DMT. As mentioned above, two impulses are performed during one orbital revolution. The first column of the table displays the current orbital altitude of the spacecraft before each maneuver. With each maneuver performed, the orbital altitude gradually decreases as the spacecraft approaches the dense layers of the atmosphere.

The second column of Table 4 gives the remaining fuel before each maneuver. Since fuel is required to operate the engine, its amount decreases with each impulse.

The third column displays the duration of DMT operation during each maneuver, which remains constant for all impulses and takes into account the total time of engine operation during maneuvers at the apogee and perigee of the orbit.

The fourth column of the table contains data on the spacecraft velocity impulse that occurs during each maneuver. Negative impulse values indicate a braking effect that reduces the speed of the spacecraft and, accordingly, its altitude. With each subsequent maneuver, the braking force increases, since, despite the descent, the speed of the spacecraft increases, requiring a more significant impulse for effective braking in the atmosphere.

Table 4

Calculation of orbital altitude with active spacecraft descent

Initial orbit altitude, m	Fuel weight at the beginning of the maneuver, kg	DU operating time, s	Velocity impulse, m/s	Passive descent time, s	Orbital altitude after the maneuver, m
758958	67	1,200	–5.3124	35,714	700,423
695376	54.32	1,200	–5.5149	29,424	651,494
650541	43.7728	1,200	–5.7206	29,118	602,423
597592	35.32	1,200	–5.930	28,812	553,207
545811	22.6506	1,200	–6.1436	28,506	503,840
495581	12.0975	1,200	–6.361	28,200	454,319
449103	1.5285	1,200	–6.4049	5,003	444,396

The passive descent time is shown in the fifth column. This period corresponds to the interval between maneuvers, during which the spacecraft does not use the engine, and its altitude gradually decreases under the influence of atmospheric drag and gravity. As the orbital altitude decreases, the speed of the craft increases, which reduces the duration of one revolution and, accordingly, the time of the passive descent.

Finally, the last column of the table displays the orbital altitude after the braking maneuvers. It follows from this data that, even when the spacecraft's fuel reserves are exhausted, the craft has not yet reached the altitude from which it could enter the dense layers of the atmosphere and burn up. Therefore, further altitude decrease will be modeled using the passive descent script, starting from the level of 444 km.

5. 3. Deorbitation parameters of the KazEOSat-1 remote sensing spacecraft obtained as a result of calculations

The initial data included the orbital parameters for the satellite as of September 30, 2024. Calculations of the orbital altitude, the magnitude of the velocity impulses, the fuel consumption, and the time required to effectively perform deorbitation maneuvers are considered.

The initial data for the calculations are the technical characteristics of the spacecraft, including its orbital parameters and the current fuel reserves on board. These parameters serve as the basis for modeling orbital maneuvers, determining the points of application of impulses and constructing a sequence of operations aimed at lowering the orbit to the level of the dense layers of the atmosphere.

Fig. 6 shows a plot of the passive descent of the KazEOSat-1 remote sensing spacecraft, where the data from Table 4 are displayed along the time (in days) and orbital altitude (in meters) axes. This plot clearly demonstrates the dynamics of the spacecraft's descent, displaying the process of passive descent until it enters the dense layers of the atmosphere. The plot shows that the passive descent is carried out due to atmospheric braking and lasts 969 days. The descent to a height of 250 km is relatively smooth since the density of the atmosphere at this level is still low. However, upon reaching a height of 250 km, the rate of decrease increases significantly, which indicates an increased density of the atmosphere at these levels.

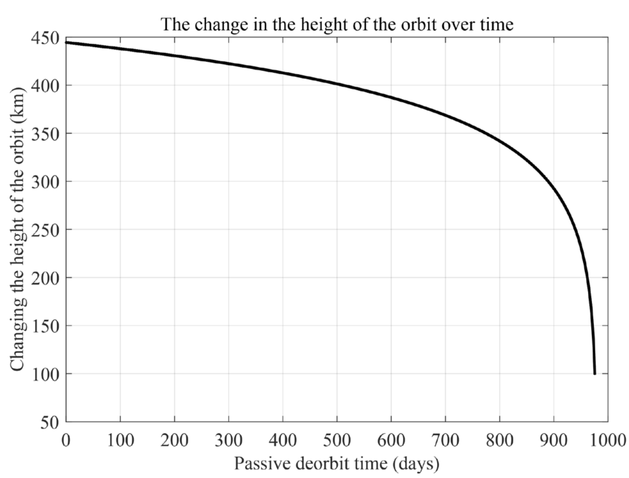

Fig. 6. Change in the orbital altitude of the KazEOSat-1 spacecraft with passive descent

For a more complete understanding of the deorbiting process, an additional plot (Fig. 7) is presented, showing the number of orbits performed by the KazEOSat-1 spacecraft during the entire period of the descent, in the form of a circular diagram. The plot, constructed along the *X* and *Y* axes, shows that the spacecraft's trajectory takes a spiral shape. This behavior is due to the fact that atmospheric braking acts similarly to the constant activation of the engine. However, due to the weak braking effect, the speed of the spacecraft changes slowly, which significantly increases the duration of deorbiting.

The black line on the diagram indicates the spiral trajectory of deorbiting, and the radius of the Earth is highlighted in blue for clarity. The dense arrangement of the orbits on the diagram indicates a significant number of orbital orbits that the spacecraft performed during its slow descent.

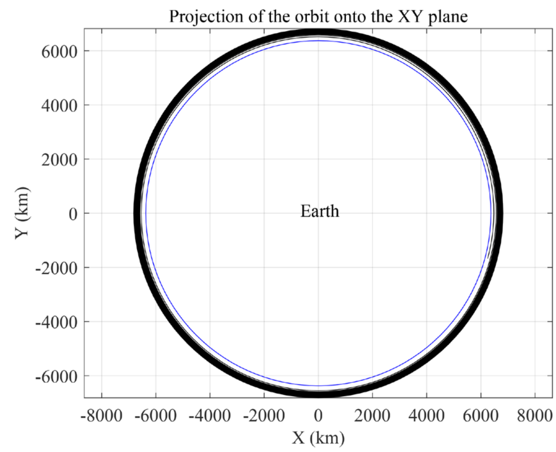

Fig. 7. Passive descent of the spacecraft KazEOSat-1

However, as the KazEOSat-1 ERS spacecraft approaches the Earth, its deorbiting accelerates and the change in orbital altitude becomes more noticeable, which is reflected in the increase in the density of the plot in Fig. 7 as it descends toward the surface. Table 5 gives results of the passive dataset of the spacecraft.

As in the previous case, a significant number of calculations were performed, and the amount of information was reduced to simplify the analysis, similar to Table 4. Table 5 gives data for every 50 km of descent. Table 5 has four columns representing the main parameters of the KazEOSat-1 ERS spacecraft deorbiting.

Table 5

Calculation of the orbital altitude for the passive descent of the KazEOSat-1 spacecraft

Descent time, day	Orbit altitude, m	Revolution period, s	Number of turns
0	444,396	5,596	15.43
548	393,988	5,535	15.61
787	343,976	5,474	15.78
891	293,967	5,533	15.96
937	243,984	5,352	16.14
957	193,915	5,292	16.32
965	143,798	5,231	16.51
969	103,306	5,183	16.67

The first column shows the deorbit time in days for each 50 km descent. As the spacecraft approaches the Earth, the deorbit time decreases due to the increasing density of the atmosphere, which has a braking effect on the spacecraft.

This process of orbital degradation under the influence of natural factors is often mentioned in scientific literature and in standards [2, 16, 17].

The second column shows the altitude of the spacecraft orbit at different stages of the deorbit with an interval of 50 km. However, in the last row, the interval is not observed since the spacecraft has reached an altitude of 100 km, from which it begins to enter the dense layers of the atmosphere.

The third column shows the time required for one revolution around the Earth. As the spacecraft approaches the Earth, this time decreases, since the orbital altitude and, consequently, the length of the revolution decrease.

The fourth column confirms this pattern, indicating the number of revolutions that the spacecraft makes in one day. The lower the orbit, the more revolutions the spacecraft makes per day.

According to the data in Table 5, in 969 days the KazEOSat-1 remote sensing spacecraft will descend sufficiently to begin active entry into the dense layers of the Earth's atmosphere, and the descent parameters will begin to change much faster. These changes will be taken into account in subsequent calculations.

After determining the active phase and passive phase of the descent in the extra-atmospheric section, the passive descent phase follows upon entry into the Earth's atmosphere. This phase begins upon entry into the Karman line at an altitude of about 100 km above the Earth's surface.

Upon entry into the atmosphere, the change in the spacecraft's velocity is calculated as follows:

$$\dot{V} = -C_{xa} S_M \frac{\rho V^2}{2m} - g\sin\theta, \tag{10}$$

where

$$\dot{\theta} = -C_{ya} S_M \frac{\rho V^2}{2Vm} + \frac{V}{R_E + h}\cos\theta - \frac{g\cos\theta}{V}. \tag{11}$$

Here C_{xa} and C_{ya} are the drag and lift coefficients of the spacecraft; S_M is the area of the spacecraft from the front during the drop, ρ is the density of the atmosphere during the drop of the spacecraft, m is the mass of the spacecraft, g is the acceleration due to gravity equal to 9.81 m/s^2; θ is the angle of entry of the spacecraft into the atmosphere; R_E is the average radius of the Earth equal to 6371 km; h is the changing altitude of the spacecraft orbit.

For a simpler solution to the problem, we shall assume that the spacecraft has a lift force C_{ya} equal to zero.

The change in the altitude of the spacecraft above the Earth's surface can be determined from the formula:

$$\dot{h} = V\sin\theta, \tag{12}$$

where $\dot{h}$ is the change in the altitude of the spacecraft orbit, V is the velocity of the spacecraft during the drop, θ is the angle of entry of the spacecraft during the drop.

The change in the flight range of the spacecraft during the drop is measured by the formula:

$$\dot{L} = \frac{VR\cos\theta}{R_E + h}, \tag{13}$$

where $\dot{L}$ is the change in the flight range of the spacecraft, V is the speed of the spacecraft, R_E is the average radius of the Earth, h is the altitude of the spacecraft.

Considering all:

$$F_A = \frac{c}{m}\frac{\rho V^2}{2}, \tag{14}$$

where ρ is the atmospheric density during the drop of the spacecraft, V is the spacecraft speed, and c is the spacecraft ballistic coefficient.

When the spacecraft enters the atmosphere, its speed begins to decrease, although up to this point, as the orbital altitude decreased, it increased. This is due to the sharp increase in atmospheric density. If at altitudes with low density, braking was relatively slow, then at denser layers, the atmosphere exerts significantly greater resistance on the apparatus, which slows it down both in altitude and speed, leading to a gradual drop of the spacecraft to the Earth's surface.

The angle of entry of the spacecraft into the atmosphere can change from the initial value to 90°. This is due to the fact that the spacecraft can enter the atmosphere at an angle to the horizon or almost vertically, when its speed becomes low enough that it stops counteracting the Earth's gravity and atmospheric resistance. In Fig. 8, it is clear that the spacecraft begins to descend at an angle, but after about 350 seconds it switches to an almost vertical drop. This indicates that at this stage the apparatus can no longer resist gravity and forces similar to those that affect falling objects on Earth begin to act on it.

It is believed that after the 350th second, only non-combustible parts of the spacecraft remain in the atmosphere, which can reach the Earth's surface and drop.

Fig. 8 shows a plot of dependence of the spacecraft's altitude on its flight range. This plot shows that up to an altitude of 30 km, the spacecraft's descent occurs relatively smoothly, after which there is a sharp transition to a virtually vertical descent. The apparatus falls along an arcuate trajectory, which allows its position to be determined at each moment in time.

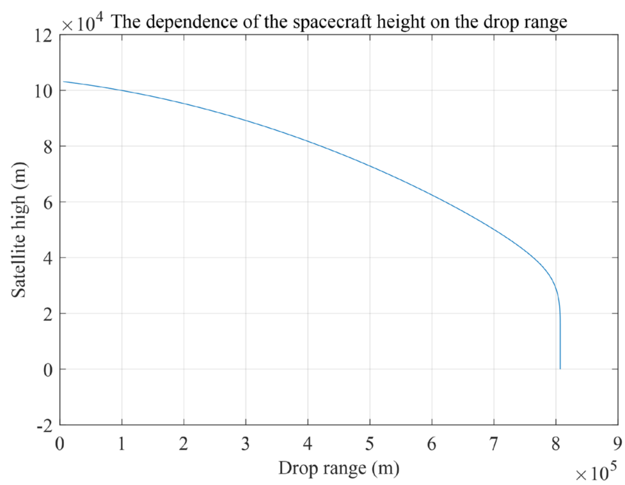

Fig. 8. Dependence of the altitude of the KazEOSat-1 spacecraft on the flight range during the drop

The plot clearly demonstrates the position of the KazEOSat-1 spacecraft along the X and Y axes, where the altitude is indicated along the Y axis, and the flight range is along the X axis. It is evident from Fig. 3 that the spacecraft no longer moves in a circular orbit, but rather moves along an arc-like trajectory, which gradually decreases toward the lower boundary of the plot. This indicates that the apparatus is falling, losing altitude relative to the Earth's surface.

Despite the obvious changes in the plot, Table 6 gives a more detailed analysis, which reflects the main parameters of the spacecraft's descent. Table 6 gives data on the descent time, changes in altitude and flight range, the angle of entry into the atmosphere, and the density of the atmosphere for a more visual representation of the process.

All parameters, except for the altitude and speed of the spacecraft, increase. This is due to the fact that during the drop, the angle of entry into the atmosphere increases, the density of the atmosphere increases as it approaches the Earth, and the flight range continues to increase, since the count is conducted from the point of entry into the dense layers of the atmosphere. The deorbit time is also increased since it is calculated from the moment the spacecraft begins to drop through the atmosphere. Table 6 contains six columns, each of which provides detailed information about the spacecraft's atmospheric deorbit process.

Table 6

Determining parameters of the spacecraft descent in the atmosphere

Descent time, s	Spacecraft altitude, m	Flight range, m	Descent velocity, m/s	Entrance angle, degree	Atmospheric density, kg/m^3
0	103,306	0	7,846	–1.54	0.0610^{-5}
100	71,967	509,169	7,464	–5.5	$2.38 \cdot 10^{-4}$
200	33,803	788,796	949	–18.1	$2.28 \cdot 10^{-2}$
300	15,991	806,496	134	–82.98	0.126
400	13,203	806,665	72	–89.99	0.267
500	9,026	806,665	54	–89.99	0.44
600	5,748	806,665	44	–89.99	0.65
700	3,031	806,665	37	–90	0.9
800	705	806,665	32	–90	1.18
834	0	806,665	30	–90	1.29

The first column of the table displays the time it took the KazEOSat-1 remote sensing spacecraft to drop from the entry point to the Earth's surface. As can be seen from the data, it takes only 834 seconds for the spacecraft to pass through the dense layers of the atmosphere.

The second column provides information on the change in the altitude of the KazEOSat-1 remote sensing spacecraft every 100 seconds of drop, starting from the altitude of entry to the moment of landing.

The third column indicates the distance traveled from the entry point to the atmosphere, which allows us to approximately calculate the location of the drop of the debris.

The fourth column shows the decrease in the speed of the KazEOSat-1 remote sensing spacecraft during the drop, which by the time of impact with the Earth is 30 m/s. Although this is a fairly high figure, it is significantly lower than the initial orbital speed of the spacecraft, equal to 7,846 m/s.

The fifth column displays the change in the angle of entry of the KazEOSat-1 remote sensing spacecraft into the atmosphere. While the entry angle was insignificant in the initial phase of the descent, in the atmosphere it increases until the spacecraft or its fragments reach the Earth's surface. It is important to note that the angle cannot exceed 90°. According to the data in Table 6, the angle changes vary from –1.5 to –90°, which indicates the transition of the apparatus to a vertical drop.

The last column is a parameter that affects the process of spacecraft deorbiting, although it does not change during the drop.

6. Discussion of results based on studying the deorbiting of the KazEOSat-1 spacecraft

Our study has analyzed the world experience of deorbiting spacecraft [4–10]. Large international space organizations such as NASA and ESA are actively developing standards and technologies for the planned set of devices [18, 19].

This study also defined the main approaches to the deorbiting of the Kazakhstani spacecraft, taking into account the parameters of the spacecraft and its orbit, as well as the remaining fuel supply. Fig. 2 demonstrates the general approach to deorbiting the spacecraft, indicates the boundary of the atmosphere, and specifies the parameters that must be taken into account for planning the deorbiting. Table 1 gives characteristics of the KazEOSat-1 spacecraft, taking into account the fuel supply on board. The orbital characteristics are in Table 2. Table 3 gives characteristics of the DMT used in the KazEOSat-1 spacecraft. Based on these data, deorbitation approaches for the KazEOSat-1 remote sensing spacecraft were optimized, taking into account its technical characteristics, orbit parameters, availability and parameters of the engine and fuel. Taking into account the above characteristics of the spacecraft, orbit and DMT, algorithms were developed for calculating and modeling the initial orbital parameters of the Kazakhstani KazEOSat-1 remote sensing spacecraft, necessary for successful deorbitation, consisting of three stages: active controlled descent, passive descent, and uncontrolled drop.

Formulas (2) to (9) were used to calculate the active descent. The necessary braking maneuvers performed by DMT to ensure a uniform descent and maintain a circular orbit were determined, and the time distribution between impulses for recharging the engine was given.

Formulas (10) to (14) were applied in the algorithms for modeling the descent after entering the atmosphere. The dependence of the KazEOSat-1 spacecraft altitude on the flight range during the drop is shown in Fig. 8.

As a result of our calculations, the main parameters for the deorbiting of the Kazakhstani remote sensing spacecraft KazEOSat-1 were obtained, which are given in the following tables: Table 4 – parameters for the active descent of the spacecraft, Table 5 – for the passive descent of the spacecraft, and Table 6 – parameters of the descent of the spacecraft in the atmosphere. Accordingly, Fig. 5 shows the plot of spacecraft deorbiting during active descent using DMT, and Fig. 6, 7 – plots of spacecraft deorbitation with passive descent; the dependence of the KazEOSat-1 spacecraft altitude on the flight range during atmospheric drop is shown in Fig. 8.

Comparison of the proposed method with alternative approaches, such as the use of membrane sails, revealed significant advantages in the stability of spacecraft attitude control. While traditional aerodynamic methods described in study [4] are subject to the risks of loss of control due to the dynamic effects of solar radiation and gravitational fields, our approach provides improved control. This is confirmed by the data from study [5], which emphasizes that minimizing external influences allows preventing uncontrolled spacecraft maneuvers, providing a more predictable and safer deorbitation process.

To determine the main parameters for deorbiting the Kazakhstani KazEOSat-1 remote sensing spacecraft, various methods were used, which made it possible to accurately calculate and model the initial orbital parameters of this spacecraft. Owing to the developed algorithms, it was possible to achieve high accuracy in calculations and determine the optimal conditions for each stage of deorbiting. The first stage, active controlled descent, was carried out using a propulsion system, allowing the spacecraft to descend from 758.95 km to 444.39 km in approximately 2.5 days (Table 4). The second stage, passive descent, took place using atmospheric braking from an altitude of 444.39 km to 103.31 km and took 969 days (Table 5). The final stage, uncontrolled drop, began after reaching an altitude of 103.31 km and lasted only 834 seconds (Table 6). Analysis reveals that the strategies devised for deorbiting the Kazakhstani KazEOSat-1 remote sensing spacecraft within the framework of this work not only effectively solve the problem of safe deorbiting of this spacecraft but also contribute to significant fuel savings. The advantages of this deorbiting strategy include the accuracy of maneuvers and the predictability of the final trajectory of the debris drop, which is critical to reducing the likelihood of collisions with other objects in orbit.

However, despite the effectiveness of the proposed methods, there are limitations, such as insufficient fuel for a fully controlled descent and the influence of external factors, which may require additional adjustments in the process of deorbiting the Kazakhstani KazEOSat-1 remote sensing spacecraft. Such factors require further study and can be taken into account in future studies to refine the models and algorithms for deorbiting control.

The strategy devised, adapted to the specific parameters of the spacecraft and its orbit, can be successfully applied to all future spacecraft of the country, including small devices such as cube sats and their orbital groupings. The developed algorithms allow taking into account the features of each device and optimizing the deorbiting process depending on its design and trajectory. Moreover, the software code developed provides the ability to assess the need to use an engine to deorbit a specific device, which makes it possible to determine whether it will leave orbit naturally under the influence of the atmosphere or whether active control will be required.

The devised approaches could be used:

– to reduce the risk of collisions in orbit;

– to ensure compliance with international requirements for the reduction of space debris;

– to plan spacecraft missions at the final stages of their operation;

– by the national operator of Kazakhstan in the field of space activities, JSC NC Kazakhstan Gharysh Sapary, and other national operators.

The conditions for applicability of the devised approaches and developed algorithms are the availability of a fuel reserve sufficient to perform the planned maneuvers and the availability of fresh and accurate data on the orbital parameters and the state of the spacecraft.

The potential expected effect of using the devised strategy is the deorbitation of the spacecraft, safe for the operation of other devices in orbit, and compliance with international standards in the space industry. Deorbitation of the spacecraft, in turn, helps reduce space debris. An indirect effect is an increase in the competencies of local specialists in the field of spacecraft control, the development and application of modern deorbitation methods, and thereby reducing dependence on foreign suppliers and technologies, as well as creating a basis for further research. Thus, the devised strategy for the deorbitation of the Kazakhstani remote sensing spacecraft KazEOSat-1 provides a basis for further improvement of the processes for managing the end of operation of spacecraft, contributing to ensuring space safety.

However, it is worth noting that the calculations did not take into account such factors as maneuvering to avoid space debris, residual fuel, atmospheric density during the drop of the spacecraft, depending on the orbital altitude, the braking force of the spacecraft against the atmosphere, minor deviations due to natural factors. It is recommended to take these factors into account during the actual test and to check with the sensors after each maneuver. It should also be taken into account that additional adjustments will increase the overall time for the disposal of the spacecraft. The study of these issues may become the subject of further research.

7. Conclusions

1. The inexpediency of using some methods, such as the Patch-Conic or Lambert methods, for deorbiting the KazEOSat-1 spacecraft has been substantiated. It was determined that for the KazEOSAT-1 remote sensing spacecraft, operating in a low Earth sun-synchronous orbit and having limited fuel reserves, the most suitable strategy is a combination of methods. It includes active maneuvers at the initial stage for rapid orbit reduction using the Hohmann transfer orbit method, a sharp decrease in perigee, gradual convergence using continuous engine operation, and passive braking in the upper atmosphere. This approach minimizes fuel consumption while maintaining control over the deorbiting process. When implementing it, it is necessary to take into account the technical limitations of the spacecraft and the features of orbital dynamics to reduce risks and ensure safety. It has been established that the use of the Hohmann transfer orbit when choosing a strategy for deorbiting the Kazakhstan remote sensing spacecraft ensures minimization of fuel consumption and a decrease in the risk of damage. A key step in devising a strategy is optimizing the trajectory and sequence of maneuvers.

2. To successfully deorbit the Kazakhstani KazEOSat-1 remote sensing spacecraft, an algorithm was developed that allows for the accurate calculation and modeling of the initial orbital parameters. Determining these parameters allowed us to build correct models for predicting the trajectory of the spacecraft and its interaction with the atmosphere. The key innovation was the use of approaches adapted for deorbiting spacecraft, which made it possible to take into account technical characteristics, orbital parameters, and fuel reserves. Based on TLE data and modeling in the MATLAB environment, the initial orbital parameters necessary for building an optimal deorbiting strategy were determined. Algorithms were developed that include impulsive maneuvers of the propulsion system, ensuring a uniform decrease in the altitude of the spacecraft and maintaining a circular orbit. Based on these algorithms, a software code was written that automates the calculations. Calculations showed that the total period of disposal of the KazEOSat-1 spacecraft from LEO will be 971 days – 2 days of active and 969 days of passive descent from the moment of the first activation of DMT. The calculations did not take into account such factors as maneuvering to avoid space debris, residual fuel, and minor deviations due to natural factors. It is recommended to take these factors into account during the actual descent and to check with the sensors after each maneuver. It should also be taken into account that additional adjustments will increase the overall time for the disposal of the spacecraft.

3. Based on our calculations, the key parameters for deorbiting the Kazakhstani ERS spacecraft KazEOSat-1 have been determined. The study showed that effective deorbiting maneuvers are possible under the following conditions:

– controlled descent of the Kazakhstani ERS spacecraft KazEOSat-1 due to the operation of DMT and atmospheric braking from 758.95 km to 444.39 km, which will take about 2.5 days;

– passive descent of the Kazakhstani ERS spacecraft KazEOSat-1 from 444.39 km to 103.31 km, where braking occurs due to the atmosphere and takes 969 days;

– uncontrolled drop and destruction of the Kazakhstani ERS spacecraft KazEOSat-1 at an altitude below 103.31 km, ending in 834 seconds.

Conflicts of interest

The authors declare that they have no conflicts of interest in relation to the current study, including financial, personal, authorship, or any other, that could affect the study, as well as the results reported in this paper.

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Data availability

The data will be provided upon reasonable request.

Use of artificial intelligence

The authors confirm that they did not use artificial intelligence technologies when creating the current work.

References

1. Strategiya "Kazahstan-2050": noviy politicheskij kurs sostoyavshegosya gosudarstva. Informacionno-pravovaya sistema normativnyh pravovyh aktov Respubliki Kazahstan. Available at: https://adilet.zan.kz/rus/docs/K1200002050
2. ISO 24113:2023. Space systems - Space debris mitigation requirements.
3. Zhang, R., Yang, K., Zhang, J., Bi, S. (2024). Overview and Key Technology of the Membrane Drag Sail for Low Earth Orbit Satellite Deorbit. Space: Science & Technology, 4. https://doi.org/10.34133/space.0115
4. Niccolai, L., Mengali, G. (2024). Performance Estimate of a Spin-Stabilized Drag Sail for Spacecraft Deorbiting. Applied Sciences, 14 (2), 612. https://doi.org/10.3390/app14020612
5. Zhang, J., Zhang, R., Yang, K. (2022). Attitude Stability Analysis and Configuration Design of Pyramid Drag Sail for Deorbit Missions. Journal of Aerospace Engineering, 35 (6). https://doi.org/10.1061/(asce)as.1943-5525.0001479
6. Xu, Y., Yang, Y., Huang, H., Jia, H., Fang, G. (2024). Numerical simulation of rigid-flexible coupled dynamics for an inflatable sphere deorbiting device. Advances in Space Research, 74 (1), 373–383. https://doi.org/10.1016/j.asr.2024.03.049
7. Rioseco Olave, D., Caqueo Jara, N., Cassineli Palharini, R., Santos Araujo Palharini, R., Gaglio, E., Savino, R. (2023). Inflatable aerodynamic decelerator for CubeSat reentry and recovery: Altitude effects on the flowfield structure. Aerospace Science and Technology, 138, 108358. https://doi.org/10.1016/j.ast.2023.108358
8. Caqueo, N., Palharini, R. C., Palharini, R. S. A., Gaglio, E., Savino, R. (2024). Inflatable aerodynamic decelerators for CubeSat reentry and recovery: Surface properties. Aerospace Science and Technology, 149, 109151. https://doi.org/10.1016/j.ast.2024.109151
9. Xiao, H., Huang, J., Liu, G., Lv, Y. (2020). Space Environment Modeling and Deorbiting Efficiency Analysis for Electro-dynamic Tether System. 2020 Chinese Control And Decision Conference (CCDC), 5063–5067. https://doi.org/10.1109/ccdc49329.2020.9164373
10. Sarego, G., Olivieri, L., Valmorbida, A., Brunello, A., Lorenzini, E. C., Tarabini Castellani, L. et al. (2021). Deployment requirements for deorbiting electrodynamic tether technology. CEAS Space Journal, 13 (4), 567–581. https://doi.org/10.1007/s12567-021-00349-5
11. IADC Space Debris Mitigation Guidelines. Available at: https://orbitaldebris.jsc.nasa.gov/library/iadc-space-debris-guidelines-revision-2.pdf
12. Vanessa, M. (Ed.) (2019). United Nations Handbook 2019–2020. Wellington: Ministry of Foreign Affairs and Trade of New Zealand, 58–60.
13. Curtis, H. (2020). Orbital Mechanics for Engineering Students. Butterworth-Heinemann. https://doi.org/10.1016/c2016-0-02107-1
14. Battin, R. H. (1999). An Introduction to the Mathematics and Methods of Astrodynamics, Revised Edition. AIAA. https://doi.org/10.2514/4.861543
15. Brophy, J. R. (2000). Low-Thrust Propulsion Technology. Journal of Propulsion and Power.
16. Janovsky, R., Kassebom, M., Lübberstedt, H., Romberg, O. et al. (2003). End-of-Life de -Orbiting Strategies for Satellites. 54th International Astronautical Congress of the International Astronautical Federation, the International Academy of Astronautics, and the International Institute of Space Law. https://doi.org/10.2514/6.iac-03-iaa.5.4.05
17. Everetts, W., Rock, K., Iovanov, M. (2020). Iridium deorbit strategy, execution, and results. Journal of Space Safety Engineering, 7 (3), 351–357. https://doi.org/10.1016/j.jsse.2020.07.013
18. NASA Orbital Debris Program Office. Available at: https://orbitaldebris.jsc.nasa.gov/
19. Space Debris. Available at: https://www.esa.int/Space_Safety/Space_Debris