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Features of deorbiting geostationary satellites

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The results of numerical simulation of the process of the disposal of a spacecraft from a quasi-geostationary orbit to a graveyard orbit with a height of about 280 km are presented. The calculations were carried out for a satellite with parameters very close to those of the Kazakh geostationary satellite Kazsat-2. Calculations have shown that, when using a stationary plasma thruster, the satellite reaches the disposal orbit in two days. And when a gas thruster is used, it will take eight days. At the same time, fuel consumption is approximately 60 kg more than in the case of SPT-70 and amounted to 61.826 kg. Taking into account this factor is very important when planning the movement of a satellite into a disposal orbit. Although raising the satellite's orbit will cause a westward drift, the change in longitude over the entire withdrawal period does not exceed 0.3 degrees.

Keywords: Geostationary satellite; KazSat-2; spacecraft; disposal; maneuvers; numerical simulations; orbit parameters

1 Introduction

In August 2009, the first Kazakh telecommunication satellite Kazsat-1 was transferred to the burial orbit. Almost all satellites operating in geostationary orbit will be disposed of at the end of their service life or as a result of an emergency situation. According to International Telecommunication Union (ITU) recommendations [1] as little debris as possible should be released into the geosynchronous orbit (GSO) region during the placement of a satellite in orbit; every reasonable effort should be made to shorten the lifetime of debris in elliptical transfer orbits with the apogees at or near GSO altitude; before complete exhaustion of its propellant, a geostationary satellite at the end of its life should be removed from the GSO region such that under the influence of perturbing forces on its trajectory, it would subsequently remain in an orbit with a perigee no less than 200 km above the geostationary altitude, and the transfer to the graveyard orbit removal should be carried out with particular caution

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in order to avoid radio frequency interference with active satellites. For satellites with orbital eccentricity e < 0.003, the minimum altitude of the disposal orbit perigee above GSO in kilometers is defined as:

$$\Delta H = 235 + 1000 \cdot \frac{C_R \cdot A}{m}, \tag{1}$$

where C_R is the solar radiation pressure coefficient (usually with a value between 1 and 2) at the beginning of the service life, A is the average cross-sectional area, m is the mass of the satellite.

In addition, according to [1] spacecraft operators are encouraged to monitor the use of propellant on board to ensure that adequate fuel is available to achieve the required maneuver at the end of life. It is recommended that a fuel margin be added to the budget in order to account for the effect of orbital determination inaccuracies and possible execution errors.

The results of work [2] show that for a spacecraft with an A/m of 0.01 m^2/kg, the minimum perigee of the disposal orbit should be 245 km above the GSO altitude and for a spacecraft with 0.10 m^2/kg and $C_R = 2$, it should be 435 km.

Thus, the disposal of spacecraft from a geostationary orbit is regulated by international legal acts. Although they are recommendatory in nature, in fact, they are mandatory for implementation.

The issues of space debris management and ensuring the safety of satellite flights are always one of the most pressing issues of modern astrodynamics. For a survey of recent works on this subject, see references [3–11]. These papers address issues such as requirements for the disposal systems of space objects, end-of disposal of geostationary satellites, new insights on the orbital debris collision hazards at geostationary orbits, stable disposal orbit for geostationary satellites, local debris congestion in the geosynchronous environment with population augmentation, passive end-of-life disposal from high-altitude orbits using solar radiation pressure, etc.

One of the most complete recent works devoted to the issue of removing a spacecraft from geostationary orbit is the paper by Gkolias and Colombo [12]. The authors explored the orbital dynamics of Earth satellites about the geosynchronous altitude, with the primary goal to assess current mitigation guidelines as well as to discuss the future exploitation of the region. A thorough dynamical mapping was conducted in a high definition grid of orbital elements, enabled by a fast and accurate semi-analytical propagator, which considered all the relevant perturbations. The results are presented in appropriately selected stability maps to highlight the underlying mechanisms and their interplay, which can lead to stable graveyard orbits or fast re-entry pathways. Moreover, they confirmed the existence of an effective cleansing mechanism for inclined geosynchronous satellites and discussed its implications in terms of current guidelines as well as alternative mission designs that could lead to a sustainable use of the geosynchronous orbital region.

The disposal simulation problem for a spacecraft in quasi-geostationary orbit is considered in Ashurov [13], where the necessary impulses, fuel consumption for maneuvers, the required time for the raising the orbit to 300 km, changes in the major semi-axis, eccentricity of the orbit and the satellite's drift in longitude are calculated. The process of raising the orbit is considered as successive starts of gas engines (GE). The author shows that, within the maximum duration of the engine operation time

of the GE, the duration of their individual starts does not significantly affect the final results of the disposal [13]. Then the parameters of the satellite were close to the parameters of the first Kazakh geostationary satellite Kazsat-1. And as mentioned above, the orbit was considered only as quasi-geostationary.

The purpose of this work is to simulate the process of deorbiting a geostationary satellite, similar to the Kazakh geostationary satellite Kazsat-2, which is currently successfully operating in orbit. Moreover, unlike [13], we will try to solve this problem in a more general case.

2 Assumptions and calculation method

To control the movement of satellites, large and small thrust engines can be used [14, 15]. In the case of high-thrust engines, it is assumed that the time required to create the necessary speed increase ΔV is negligible compared to the period of satellite revolution. This kind of control is called impulse control. In this case, the action of the traction force is reduced to a sudden change in the velocity of the spacecraft without changing the coordinates during the time the engine is running. Given that this time is usually much shorter than the time of the orbital transition, this assumption is justified.

In the case of low-thrust engines, the running time of the engine becomes of the order of the satellite revolution period. In this case the impact of the thrust on the orbit can be computed by integrating of the motion equation taking into account the acceleration due to the thrust of the engine [14, 16]. On this basis, in [17], the problem of controlling the planar parameters of the orbit of a geostationary spacecraft with the help of a low-thrust engine is solved. Since all maneuvers must be carried out from the optimality condition, in [18] the problem of optimizing the geostationary orbit with the use of an ionic low-thrust engine is considered.

In [13] the case of $e \neq 0, \quad i = 0$ (eccentricity and inclination of the orbit) and $T_s \neq T_E$ has been considered, where T_s is the sidereal period of the satellite revolution, T_E is the duration of the stellar day.

Now we consider the case of $e \neq 0, \quad i \neq 0$ and $T_s \neq T_E$. Thus, the elements of the orbit differ from the elements of the ideal orbit, that is they are more real. In this case, the satellite is displaced relative to the Earth's surface in both latitude and longitude [14]. However, since we consider the active motion of the satellite under the action of engine thrust, we do not take into account various perturbing factors that play a significant role in its passive motion.

Of the three kinds of impulses (tangential, normal and lateral, or binormal), a tangential impulse is used to change the major semi-axis of the orbit, the drift speed in longitude and the eccentricity vector. Taking into account that in the problem considered by us, as a result of one small impulse ($\Delta V \ll V$), the major semi-axis changes by a small amount ($\Delta a \ll a$), it can be shown that [14-16]:

$$\Delta a \approx \frac{2a^2}{\mu} V \Delta V, \tag{2}$$

where $\mu = fM$ is the gravitational parameter of the Earth.

The change of the major semi-axis is simultaneously accompanied by a change of the eccentricity e [14]:

$$\Delta e = \frac{2\Delta V}{V}(e + \cos\nu), \tag{3}$$

where ν is the true anomaly of the spacecraft.

The tangential impulse ΔV can be calculated as [13]:

$$\Delta V = \frac{F\cos\alpha\Delta t}{m}, \tag{4}$$

where F is the value of the thrust, m is the mass of the spacecraft, α is the angle between the directions of maneuvering and thrust of the engine and Δt is the thruster running time. Although the mass of the spacecraft is a function of time, but during the operation of the thruster it can be considered as a constant.

We assume that the orbital elements for initial moment of time (semi-major axis a_0, eccentricity e_0, inclination i_0, argument of perigee ω_0, true anomaly ν_0, mean anomaly M_0 and orbital period T_0) are known.

The process of raising of the orbit will be considered as a result of successive starts of the thrusters and corresponding changes in the indicated orbital elements. So, numerical simulation is performed in the following sequence.

1. Calculation of the mean anomaly: $M = M_0 + 2\pi t/T$.
2. Calculation of the true anomaly from the Kepler equation: $M = E - e\,\sin E$ and
$$\tan\frac{\nu}{2} = \sqrt{\frac{1+e}{1-e}}\,\tan\frac{E}{2}. \tag{5}$$
3. Calculation of the increment of the velocity ΔV by formula (4). In this case, for each individual maneuver, as spacecraft mass is taken its mean value $m = (m_{i-1} + m_i)/2$, where m_{i-1} and m_i are the masses of the spacecraft at the beginning and end of the maneuver. The spacecraft mass at the end of the maneuver is defined as $m_i = m_{i-1} - dm = m_{i-1} - \dot{m}\Delta t$, where $\dot{m}$ is the mass fuel consumption per second.
4. Calculation of the change in the major semi-axis Δa by the formula (2). At the same time, as the major semi-axis a and velocity V, values are taken that correspond to the beginning of the maneuver.
5. Calculation of the change in eccentricity Δe by formula (3). As in the previous case, as the velocity V, values are taken that correspond to the beginning of the maneuver. After the maneuver, the velocity, the major semi-axis and the eccentricity will become, respectively: $V = V_0 + \Delta V$, $a_i = a_{i-1} + \Delta a$, and $e_i = e_{i-1} + \Delta e$. Since the satellite maneuver is carried out only under the action of a tangential impulse, other orbital elements remain unchanged.
6. Calculation of satellite coordinates in the Earth-centered inertial (ECI) coordinate frames from orbital elements.
7. Calculation of satellite coordinates in the Earth-centered, Earth-fixed coordinate system.
8. Calculation of the geodetic longitude (λ), latitude (ϕ), and geodetic height (H).

3 Results of modeling and discussion

To simulate the disposal process, the following initial parameters were taken, which are close to the parameters of the Kazsat-2 satellite:

– initial longitude $\lambda_0 = 86°28'58''$ E and latitude $\phi_0 = 0°03'05''$ N;
– the major semi-axis of the orbit $a_0 = 42166.2$ km;
– eccentricity $e_0 = 0.0000440$;
– inclination $i_0 = 0.0541°$;
– argument of perigee $\omega_0 = 199.7178°$;
– true anomaly $\nu_0 = 231.7646°$;
– mean anomaly $M_0 = 231.7685°$;
– mass of the satellite $m_0 = 1271$ kg;
– angle between the directions of maneuvering and thrust $\alpha = 60°$;
– number of simultaneously operating in maneuvering thrusters $N = 4$.

The Kazsat-2 satellite has two types of thrusters: Stationary Plasma Thrusters SPT-70 and cold gas thrusters. Accordingly, for them, the thrust F is 0.039 N and 0.009 N.

The simulation results are presented in Table 1. We performed simulations up to the height of the disposal orbit, i.e. approximately 280 km.

Table 1 Simulation results.

Thruster	Deorbit time, days	Lifting height (h), km	Fuel consumption (Δm), kg	Limits of change in longitude ($\lambda_{\min}$–$\lambda_{\max}$), deg	Limits of change in latitude ($\pm\Delta\phi$), deg	Maximum change in eccentricity (Δe)
SPT-70	2	284	1.903	86.42–86.60	±0.054	0.001095
Gas	8	279.6	61.826	86.28–86.50	±0.054	0.000271

As can be seen from Table 1, the time required to put a satellite into a graveyard orbit depends on the type of thrusters used, which differ in thrust value. Approximately a fourfold advantage in the thrust of the SPT-70 thruster provides, respectively, a fourfold reduction in the time to perform this operation. However, the fuel consumption of the gas thruster is 32.5 times higher than that of the SPT-70 thruster. This is due to the high efficiency of the latter. Other parameters of the maneuver, such as changes in longitude, latitude, and eccentricity, differ very little. Because their changes depend more not on the characteristics of the thrusters, but on the initial elements of the orbit and on the features of the geostationary orbit.

Figures 1 and 2 show graphs of longitude, latitude, eccentricity, semi-major axis of the orbit, height above geostationary orbit, depending on time, as well as a plot of latitude versus longitude for the cases of the SPT-70 thruster and the gas thruster.

These figures show that a decrease in thrust will lead to a decrease in the amplitude of change in longitude. But this will lead to an increase in the drift of the satellite's position in the westerly direction. In addition, this will lead to a decrease in the amplitude of the eccentricity change by about a factor of four. From this we can conclude that if the maneuver must be carried out with a minimum eccentricity, then

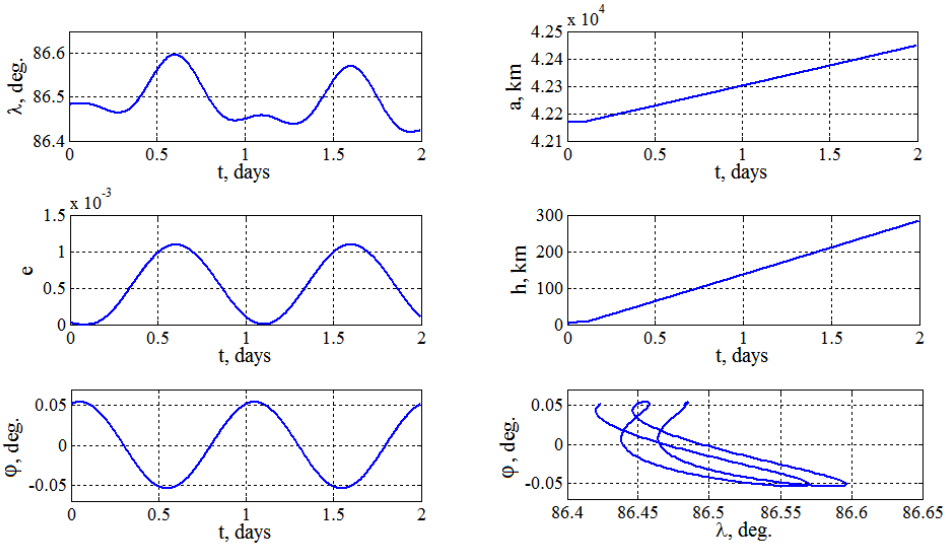


Figure 1 Changes in longitude eccentricity, latitude, semi-major axis of the orbit, height above geostationary orbit with time and latitude versus longitude for the case of the SPT-70 thruster.

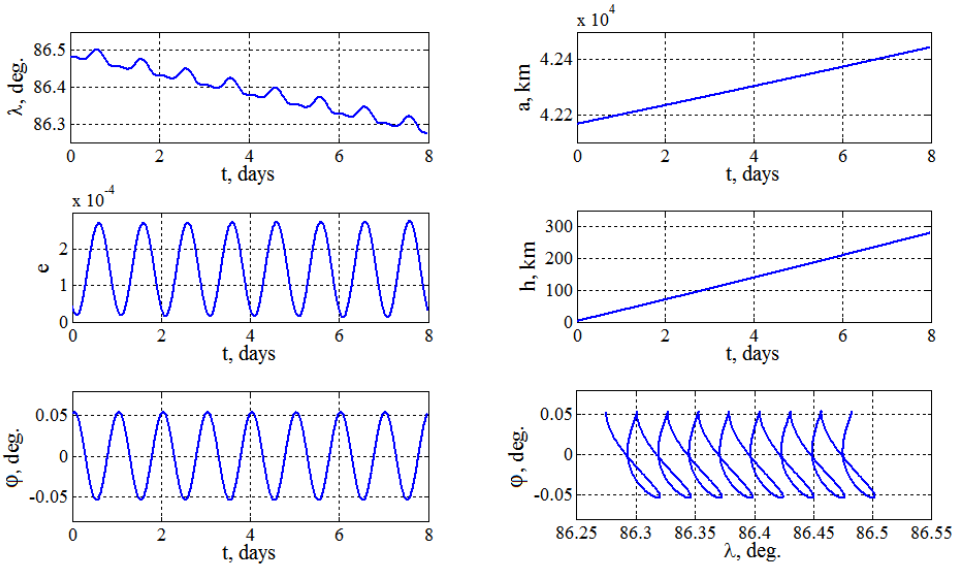


Figure 2 Changes in longitude eccentricity, latitude, semi-major axis of the orbit, height above geostationary orbit with time and latitude versus longitude for the case of gas thruster.

the use of gas thruster is preferable. But this will lead to an increase in the time it takes for the satellite to be removed to the disposal orbit.

The pattern of change in latitude is the same in both cases. Moreover, both

eccentricity and latitude have the same change period of one day. The semi-major axis and height above the geostationary orbit change linearly over time.

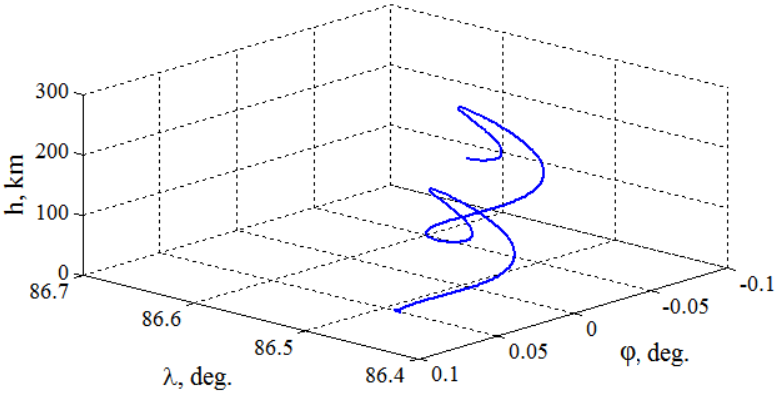


Figure 3 Trajectory of the satellite in the longitude, latitude and height space for the case of the SPT-70 thruster.

Figures 1 and 2 show that the plots of latitude versus longitude are specific, repeating patterns. An even more interesting picture can be seen in 3D Figures 3 and 4, where the height above the geostationary orbit is added vertically. In fact, this is the trajectory of the satellite in space during the maneuver.

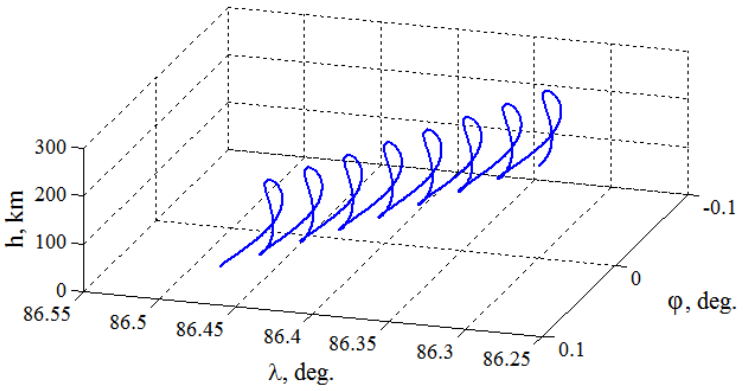


Figure 4 Trajectory of the satellite in the longitude, latitude and height space for the case of the gas thruster.

Thus, we have presented here a rather detailed picture of the disposal process of the geostationary satellite with parameters close to those of Kazsat-2. We think that taking into account the perturbing factors of the Sun and the Moon, as well as the nonsphericity of the Earth's gravitational potential, will not change the picture much. Evaluation of their influence on this type of maneuver is our long-term task.

4 Conclusion

We have carried out a numerical simulation of the process of moving a spacecraft from a quasi-geostationary orbit to a graveyard orbit with a height of about 280 km. The calculations were carried out for a geostationary satellite with technical parameters and orbital elements close to those of the Kazsat-2 satellite. Two variants of the maneuver are considered: using a stationary plasma thruster and a gas one. Calculations showed that it would take 2 days to reach the disposal orbit using SPT-70 and 8 days using a gas thruster. The fuel consumption in the first case was 1.903 kg, and in the case of a gas thruster it was 61.826 kg. This means that the choice of a gas thruster imposes a higher fuel requirement to ensure completion of the disposal process.

Raising the satellite's orbit will cause it to drift westward. Moreover, the change in longitude is periodically. And the presence of an inclination of the orbit will lead to a periodic change in latitude and eccentricity. Their change period is one day. Changes in the semi-major axis and height over time occur according to a linear law.

We have not yet considered the effect of the non-spherical potential of the Earth and the influence of the Sun and Moon. Because the acceleration from constantly running thrusters is much greater than from the acceleration of these perturbations. Taking them into account and making more accurate calculations is the goal of our future research.

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