RESEARCH ARTICLE | JULY 20 2023

Development of a thruster concept for deorbiting CubeSat 3U+ from altitude of 400 to 350 km FREE

D. K. Fedorova ✉; V. D. Telekh; C. V. Nosov; A. R. Gizzatullin; M. P. Shipilov

AIP Conference Proceedings 2549, 130006 (2023)
https://doi.org/10.1063/5.0109580

Development of a Thruster Concept for Deorbiting CubeSat 3U+ from Altitude of 400 to 350 km

D.K. Fedorova [1, a)], V.D. Telekh [1, b)], C.V. Nosov [1, c)], A.R. Gizzatullin [1, d)], M.P. Shipilov [2, e)]

[1)]*Bauman Moscow State Technical University, Moscow, 105005, Russia.*
[2]*Moscow Polytechnic University, Moscow, 107023, Russia.*

[a)]Corresponding author: fdaryakonst97@mail.ru
[b)]telekh@bmstu.ru
[c)]nosov@bmstu.ru
[d)]gizzatar@gmail.com
[e)]shipilov.mp@gmail.com

Abstract. The concept of a low-thrust electric propulsion system with a thrust of 9 μN and a specific impulse of about 1000 seconds for deorbiting a CubeSat 3U+ satellite from altitude of 400 to 350 km is proposed. A coaxial pulsed plasma thruster with a diameter 24 mm and 68 mm long with a power consumption of 5 W and frequency of 1 Hz, was considered.

INTRODUCTION

In the past decade, pico- and nanosatellites of the CubeSat format have been attracting more and more attention. These satellites' advantages are small dimensions, low mass, relatively inexpensive to manufacture and launch. According to [1, 2], a cost of a 1U CubeSat module is on average 50...85 thousand dollars.

These satellites are usually used for testing new technologies, software and hardware solutions, also for environmental monitoring or educational programs. For example, there is a table in [1], where some CubeSats and their missions can be found. It is clear, that most of these satellites were used for testing different systems. Some of satellites in that table were used for Earth remote sensing.

Sometimes nanosatellites are equipped with a propulsion system. It is used for orbit correction, increase of lifetime, disposal or group flight implementation.

In order to determine the propulsion system characteristics for a specific mission, a value of necessary velocity increment (ΔV) is required. It depends on many factors, such as parameters of initial and final orbits, the value of solar flux index and etc. There are different methods of finding these velocity increments. For example, flights between two non-coplanar circular orbits using $J2$ gravity effects are considered in [3]. It is shown how the value of ΔV depends on the inclination of the orbit and the number of passive revolutions while staying in the drift orbit.

ORBITAL DYNAMICS ESTIMATION

This work is focused on the deorbiting simulation of the CubeSat 3U+ satellite. The initial orbit is 400 km and the final orbit is 350 km, both circular. The initial orbit was chosen based on the fact, that such satellites can be deployed from the International Space Station (ISS). A simplified model was used for estimating the time of deorbiting. It was assumed, that the satellite descends along a spiral trajectory. The model is based on calculation of atmosphere density at instant orbit and iterative recalculation of descent rate.

The assumptions were made:

1. Earth radius is constant and its value is 6371 km.

XLV Academic Space Conference, Dedicated to the Memory of Academician S.P. Korolev and Other Outstanding National Scientists — Pioneers of Space Exploration
AIP Conf. Proc. 2549, 130006-1–130006-7; https://doi.org/10.1063/5.0109580
Published by AIP Publishing. 978-0-7354-4554-3/$30.00

2. Atmospheric drag and propulsion system's thrust are the only factors, which affect the satellite's descent.
3. Earth is the only celestial body with which the satellite gravitationally interacts.
4. The satellite moves in a circular orbit in the absence of atmospheric drag and thrust.

Main Calculation Formulas

Atmospheric drag can be written as:

$$F_D = C_d \frac{\rho_H V^2}{2} A,$$

where C_d is the drag coefficient which mainly depends on the spacecraft shape (for a satellite with flat mid-section on 200…400 km its value is about 2.4 [4]); ρ_H, kg/m^3 – atmosphere density at altitude H (was defined by GOST R 25645.166-2004); A, m^2 – mid-section area (the largest section normal to satellite's velocity's vector); V, m/s – velocity of the satellite.

The orbital motion is described by the following equation (2nd Newton's law):

$$\frac{M_S V^2}{R_E + H} = \frac{\mu M_S}{\left(R_E + H\right)^2},$$

where M_S is a mass of the satellite, kg; R_E, km – Earth radius; H, km – altitude of satellite; μ = 398600.4418 km^{3}s^{-2} – Earth gravity constant.

Expressed from the equation (2) velocity of the satellite results in:

$$V = \sqrt{\frac{\mu}{R_E + H}}$$

Gravitational force:

$$F_G = \frac{\mu M_S}{\left(R_E + H\right)^2}$$

Total energy of satellite (sum of kinetic and potential) with inserted from (3) velocity can be written as:

$$U_\Sigma = \frac{M_S V^2}{2} - \frac{\mu M_S}{R_E + H} = \frac{\mu M_S}{2\left(R_E + H\right)} - \frac{\mu M_S}{R_E + H} = -\frac{\mu M_S}{2\left(R_E + H\right)}$$

Atmospheric drag F_D and thrust F_T decrease this energy and lower the orbit by δh during time δt. Than changing of the total energy can be defined as:

$$-\frac{\mu M_S}{2\left(R_E + H + \delta h\right)} - \left(-\frac{\mu M_S}{2\left(R_E + H\right)}\right) = -F_D V \delta t - F_T V \delta t$$

Equation (6) can be transformed into:

$$-\frac{\delta h}{\delta t} = \frac{2\left(F_D + F_T\right) V \left(R_E + H\right)}{M_S V^2}$$

Considering that $\delta h \ll R_E + H$ and taking into account equations (2) and (3), it can be rewritten as:

$$-\frac{\delta h}{\delta t} = \frac{\left(R_E + H\right)^2}{\mu} \beta \rho_H \left(\frac{\mu}{R_E + H}\right)^{\frac{3}{2}} + 2F_T \sqrt{\frac{\mu}{R_E + H}} \frac{\left(R_E + H\right)^2}{\mu M_S},$$

where $\beta = \frac{C_d A}{M_S}, \frac{\text{m}^2}{\text{kg}}$ is the ballistic coefficient of the satellite.

Equation (8) presents the orbit descent rate. It can also be noticed that the rate of satellite's deorbiting strongly depends on the ballistic coefficient and the orbit altitude.

The Program Code Validation

A program code for determining satellite's lifetime was written. This code solves equation (8) with a small time step. In this way it integrates (8) and calculates deorbiting time from the initial orbit to the final one. Comparison with ever-flown CubeSat satellites was done for code validation. Real rate of deorbiting was compared with calculated one for real CubeSats.

ODERACS satellite (22994) was launched on 2nd of February, 1994 to the 350 km orbit. It reentered on the 3rd of March, 1995. Thus its lifetime was 395 days [5].

The reference area of the satellite was A = 0.018 m^2, the drag coefficient C_d = 1.96, mass M_S = 5 kg [6]. The ballistic coefficient β = 0.007 m^2/kg. Average solar flux index in 1994 was F_0 = 90 [7].

Calculated deorbiting time at F_0 = 100 is 331 days. Calculated deorbiting time at F_0 = 75 is 526 days. After linear interpolation of these results, it can be seen, that with index F_0 = 90 satellite's lifetime will be approximately 409 days.

Calculation error:

$$\delta = \frac{409 - 395}{409} \cdot 100\% = 3.6\%$$

The same comparison was done for SkySat-1 [8] (7 % error), RAIKO [9] (9 % error). In all cases calculation error was less than 10 %. Hence, the code functions satisfactorily and the method of deorbiting time estimation is acceptable.

After the validation, satellite's lifetime calculation can be done. According to the calculation results, CubeSat 3U+ deorbiting from 400 km to 350 km with solar flux index F_0 = 75 will take about 1600 days or less than 4.5 years. If it is required to speed up this process, the propulsion system could be installed onboard.

According to the calculations with the propulsion system of a thrust 9 μN, the satellite will deorbit twelve times faster (if propulsion system is active throughout the flight). Total deorbiting time will be 140 days instead of 1600 days.

CONCEPT OF PULSED PLASMA THRUSTER

It is proposed to use end-type coaxial pulsed plasma thruster (PPT) as the propulsion system. Schematic picture of such thruster is shown on Fig. 1. Similar works are being held in MAI (Russia) [10], Europe [11], Egypt [12], Japan [13] and others. Such thruster consists of a capacitor bank, electrodes, a propellant bar, usually Polytetrafluorethylene (PTFE), an insulator and an ignition electrode.

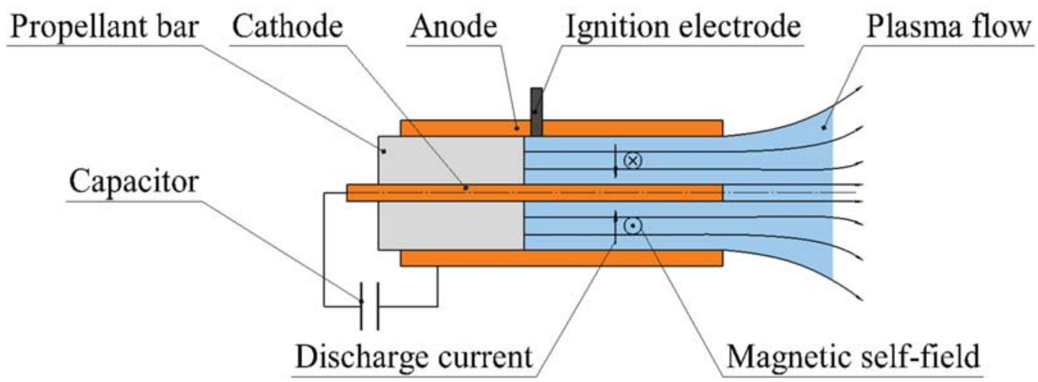

FIGURE 1. Scheme of coaxial PPT [14]

This thruster operates the following way. A voltage, which exceeds the one between cathode and anode, is applied to the ignition electrode and to the cathode. After that a charge occurs: an arc is struck between main electrodes. The radiation from this arc heats the surface of propellant bar. The process of ablation and ionization of light-erosion products starts. Lorentz force makes ions to accelerate. This force spring up due to appearance of a magnetic self-field of the current going through the electrodes and through the plasma filament. There are also uncharged particles in the flow, which can fly away from discharge channel, producing some thrust.

Propellant Mass Estimation

Considering the data, which was discussed above (9 μN thrust, operating time 140 days, propellant – PTFE), necessary propellant mass for such maneuver can be calculated. According to [15], the empirical equation of specific impulse can be written as:

$$I_{Sp} = 317\left(\frac{E_0}{A_P}\right)^{0.585},$$

where E_0, J – discharge energy; A_P, cm^2 – ablative area.

According to Tsiolkovsy equation, necessary velocity increment can be written as:

$$\left|\Delta V\right| = W_{eff} \ln\left(\frac{M_S + m}{M_S}\right),$$

where W_{eff}, km/s – effective exhaust velocity of plasma flow; M_S, g – satellite mass; m, g – propellant mass.

Effective exhaust velocity can be written as:

$$W_{eff} = I_{Sp} g,$$

where g = 9,81 m/s^2 – gravitational acceleration.

Length of the propellant bar can be written as:

$$l_P = \frac{m}{\rho_P A_P},$$

where ρ_T, g/cm^3 – propellant density.

By combining equations (9)-(12), mass and length of propellant bar as functions of ablation area can be written. As was mentioned above, propellant is PFTE. For satellite with mass 4 kg, with storage energy 5.3 J, the results are shown on Fig. 2 can be reached.

Figure 2 shows that it is necessary to choose optimum ablation area. It was chosen 2.8 sm^2. Thus, length of propellant bar is 40 mm, its mass is 25 g.

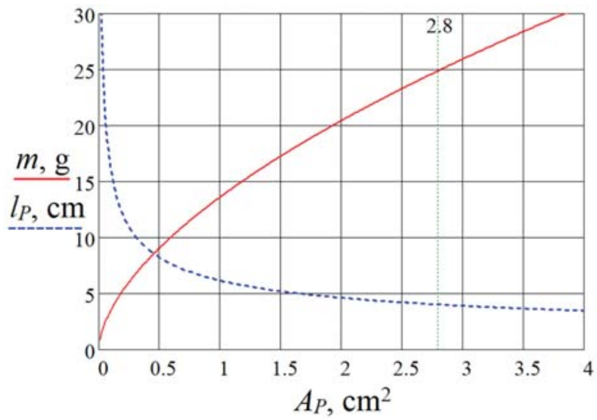

FIGURE 2. Length of propellant bar and its mass versus ablation area

Pulsed Plasma Thruster Parameters

A model, described in [16], was used for plasma flow numerical calculations in MATLAB. Input parameters for calculation can be found in Table 1.

TABLE 1. Input Parameters for Calculation

Parameter	Value
Energy	5.3 J
Mass of the Accelerated Plasma	0.5 μg
Frequency	1 Hz
Capacitance	4.6; 2.7; 1.3 μF
Charge Voltage	1500; 2000; 2800 V
Circuit Inductance	0.13 μH
Circuit Resistance	50 mΩ

The calculation was carried out for three various combinations of capacitance and discharge voltages (energy was still about 5 J). As a result, an option with capacitance 2.7 μF and voltage 2000 V was chosen as optimum. Results of the calculation are given in table 2.

TABLE 2. Performance Parameters of Coaxial PPT for 2.7 μF and 2000 V

Parameter	Value
Discharge Channel Length	15 mm
Exit Velocity	18 km/s
Discharge time	1.8 μs
Peak Current	7.4 kA

Based on the calculations results, PPT design was made. It is shown on the Fig. 3.

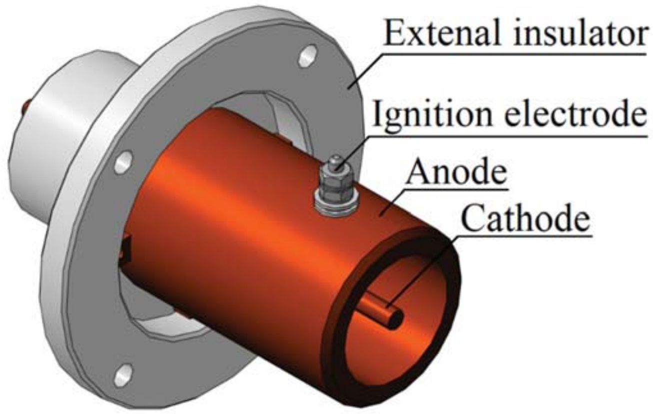

FIGURE 3. Developed concept of coaxial PPT

Thruster consists of copper electrodes, except the ignition one – it is made of molybdenum. There is also external insulator, made of polyamide-6, supposed to be attached to the satellite. Overall dimensions of the thruster are Ø 56 × 68 mm.

Figure 4 shows the position of PPT with layout of power processing unit and capacitor bank on the satellite layout. The whole propulsion system (with power processing unit – PPU) occupies no more than 100 mm long or 1U module in satellite.

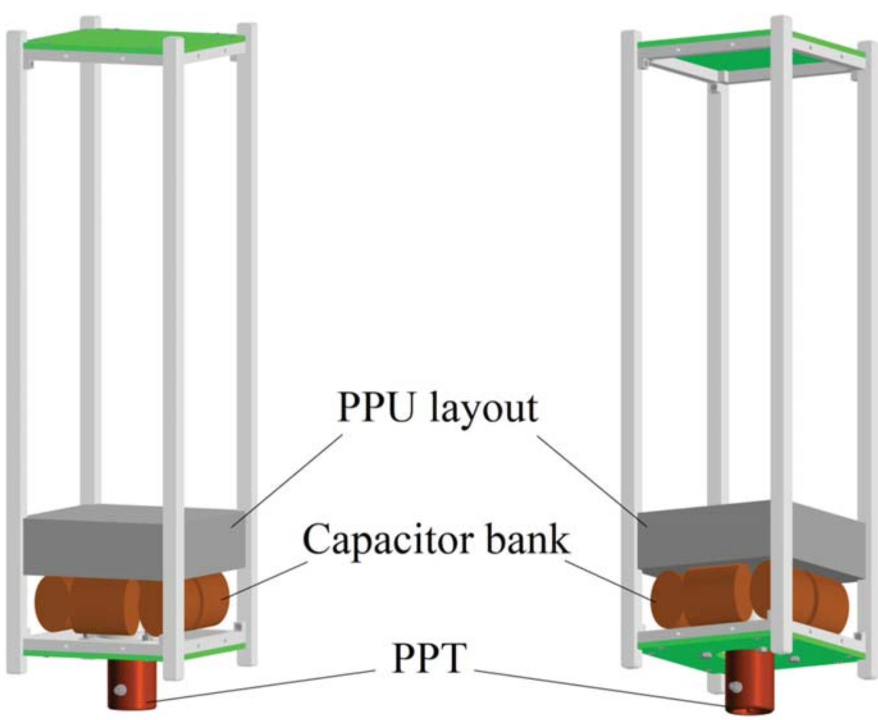

FIGURE 4. The model of thruster, placed on 3U+ CubeSat platform

CONCLUSION

The orbital dynamics estimation with the simplified model was developed. The results show that CubeSat 3U+ deorbits from 400 to 350 km in about four years and a half. At the same time, if the propulsion system with 9 μN thrust is installed onboard, the lifetime will decrease for almost twelve times and will last for 140 days.

The pulsed plasma thruster with 9 μN thrust and propellant mass of 25 grams was considered as the propulsion system. The design of such thruster has been proposed, estimation of thruster overall dimensions in satellite was done. The whole propulsion system with PPU and capacitor bank occupies no more than 1U CubeSat module.

ACKNOWLEDGMENTS

This research was funded by the Ministry of Science and Higher Education of the Russian Federation (government task 0705-2020-0046) and has been performed at large-scale research facilities «Beam-M» of Bauman Moscow State Technical University.

REFERENCES

1. D. A. Khramov, Kosmichna nauka I technologiya **20–31**, 15-3 (2009).
2. I.N. Gansvind, Issledovaniye Zemli iz kosmosa **82–88**, 5 (2019). doi:10.31857/S0205-96142019582-88.
3. A. A. Baranov, D. A. Grishko, V. I. Mayorova, Acta Astronautica **307-317**, 116 (2015). doi:10.1016/j.actaastro.2015.06.025.
4. M. M. Moe and C.D. Wallace, The Upper Mesosphere and Lower Thermosphere: A Review of Experiment and Theory **349-356**, 87 (1995). doi:10.1029/GM087p0349.
5. Space Track, see: https://www.space-track.org (date of application 15.04.2021).
6. B.R. Bowman and K. Moe, Advances in the Astronautical Sciences **117-130**, 123 (2006).
7. Canadian Space Weather Forecast Centre. The Penticton Observatory, see https://spaceweather.gc.ca/solarflux/sx-5-en.php (date of application 15.04.2021).
8. A. Vilkauskas. CubeSat mission LITSAT-1, see: PowerPoint Presentation (space-lt.eu) (date of application 08.05.2021).
9. NASA Space Science Data Coordinated Archive, see: NASA - NSSDCA - Spacecraft - Telemetry Details (date of application 8.05.2021).
10. N.N. Antropov et al., Procedia Engineering **53-60**, 185 (2017). doi:10.1016/j.proeng.2017.03.291.
11. C. Montag, H. Burghaus, G. Herdrich, T. Schönherr *Development of a new pulsed plasma thruster and a brief introduction of a planned test facility* in the *Proceedings of the 67th International Astronautical Congress* (Guadalajara, Mexico, 2016, October).
12. M. A. Othman et. al. *Firing test and evaluation of coaxial pulsed plasma thruster* in the *Proceedings of the 18th IOP Conference Series: Materials Science and Engineering* 610 (2019) 012070. doi:10.1088/1757-899X/610/1/012070.
13. K. Shintani et al.: *Research and development on coaxial pulsed plasma thruster with feed mechanism* in the *Proceedings of the 31st International Electric Propulsion Conference* (University of Michigan, Ann Arbor, Michigan, USA, 2009, September).
14. Hang Lia et. al., Acta Astronautica, **154-162**, 170 (2020). doi:10.1016/j.actaastro.2019.11.014.
15. W. J. Guman, J. Spacecr. Rockets, **51–54**, 13-1 (1976).
16. L.A. Artsimovich et al., JETP, **1**, 6-1 (1958).