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Development of a propulsion system for a maneuvering module of a low-orbit nanosatellite

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Abstract

Nanosatellites (NS) are one of the fastest growing branches of spacecraft technologies. One of the attractive areas of application for such spacecraft is real-time low-orbit remote sensing of the Earth. This paper aims to develop a concept of a propulsion system (PS) and a maneuvering module (MM) for a CubeSat3U format low-orbit nanosatellite (NS), that would enable such spacecraft to maintain the required orbit altitude for a few weeks or months depending on insertion parameters. Requirements for the maneuvering module are stated. Helium, nitrogen and liquefied propane are considered as propellants. Modeling showed that helium provides the greatest specific impulse, but its density is too low to store a sufficient quantity. Under specified conditions, nitrogen may be able to provide required characteristics of the maneuvering module; specific impulse can be nearly doubled by heating the propellant to 700 °C with electric heaters.

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Keywords: nanosatellite; low orbit; maneuvering module; thruster; propellant; electric heating

1. Introduction

Today development of nano-class research and educational spacecraft is a common practice for leading airspace universities. A nanosatellite is usually defined as a satellite weighing from 1 to 10 kg, and developed primarily for purposes of research and education. With non-hermetic design and commercial components, creation of this class of satellite requires only a relatively small expense of time and finances.

Analysis of space launches for the last few years shows that the number of nanosatellite launches has been growing exponentially, with most of these being commercial and university spacecraft and satellite constellations. Projects under development include NS constellations for observation over other spacecraft, remote sensing of the Earth, communication and retranslation, monitoring of geophysical fields, and other missions that require precise station keeping and ability to maneuver for successful experimenting or otherwise achieving the mission goal.

The importance of development of a PS-equipped maneuvering module is further supported by some specifics of NS insertion. In most cases these spacecraft are inserted as way cargo, so the initial insertion orbits are determined by the requirements of the main payload, and do not always coincide with the desirable orbit for the nanosatellite.

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On the other hand, the NS should not turn into space debris after the end of their active life (normally no more than 1 or 2 years). For this reason nanosatellites ought to be inserted to low orbits, where they can be naturally de-orbited by air drag. All these factors make it desirable for the nanosatellites to be equipped with maneuvering modules that will enable their station-keeping across their one-two year operational life.

Development of MM for NS will improve their autonomy and extend their application range, which in turn will decrease the costs of many space exploration missions.

The goal of this paper is to develop a concept for a propulsion system for a maneuvering module of a low orbit aerodynamically stabilized CubeSat3U-class nanosatellite.

2. Selection of design scheme for a maneuvering module propulsion system

The design scheme for a maneuvering module for a low altitude aerodynamically stabilized CubeSat3U nanosatellite faces the following constraints: the thruster must be environmentally safe, accident-proof during testing and exploitation, and provide delta-velocity budget no smaller than 20 m/s, which is enough to keep required altitude or position in formation flying for up to half a year.

In view of these constraints, the most attractive design scheme for a maneuvering module will be a cold-gas thruster. The basic scheme of this propulsion system is represented in Fig. 1.

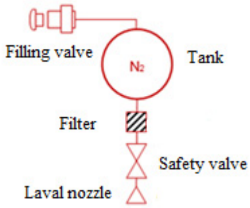

Fig. 1. Cold-gas thruster

The most important characteristic of the thruster is specific impulse of thrust, which characterizes exhaust velocity at nozzle exit w_e, m/s.

From the ideal gas law and the balance of energy in gas flow one can determine the exhaust velocity at de Lavale nozzle exit:

$$w_e = \sqrt{\frac{TR}{M} \cdot \frac{2k}{k-1} \cdot \left[1 - \left(\frac{p_e}{p}\right)^{(k-1)/k}\right]} \quad (1)$$

where w_e is the exhaust velocity at nozzle exit, m/s; T is absolute temperature of the gas at entry to the thruster; R= 8314,5 is the universal gas constant (J/kilomole/K); M is the molar mass of the gas (kg/kilomole); $k = c_p/c_v$ is adiabatic exponent; c_p is specific heat capacity at constant pressure, J/(kilomole K); c_v is specific heat capacity at constant volume, J/(kilomole K); p_e is absolute pressure of the gas at nozzle exit, Pa; p is absolute pressure of the gas at nozzle entry, Pa.

With p_e=0 it is possible to determine the theoretical maximum of exhaust velocity of the gas in vacuum, determined by the inner energy of the gas:

$$w_{max} = \sqrt{\frac{TR}{M} \cdot \frac{2k}{k-1}} \quad (2)$$

The biggest disadvantage of cold-gas thrusters is low specific impulse of thrust. This can be improved to some extent by heating the gas with electric heaters before it enters de Lavale nozzle (see Fig. 2).

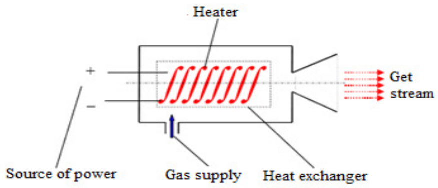

Fig. 2. Scheme of heater for electro-thermal propulsion system

3. Selection of propellant for the maneuvering module propulsion system

To select the propellant for the maneuvering module thruster one must compare the delta velocity budgets that different propellants provide. Propellant mass is determined both by the properties of the selected gas and by the parameters of the tank: volume and propellant storage pressure.

The propellant storage pressure at full load of the tank was selected to match the standard gas storage pressure adopted by Roscosmos enterprises: P_t = 35 MPa. This simplifies the filling of the tank, which is performed at the cosmodrome immediately before the launch.

The storage tank maximum dimensions for the CubeSat3U spacecraft were selected as 1.5U. To simplify the positioning of the valves, filler valve and filter, toroid shape of the tank was preferred. To ensure maximum safety, the tank was designed as a coil (see Fig. 3).


Fig. 3. Coil design of the tank

Considering the above constraints, it was possible to design the tank within required dimensions with displacement of $V_t = 0{,}36 \bullet 10^{-3}$ m^3. The suggested design of the tank has a number of additional advantages. In particular, it is easy to obtain legal certification for the unit, because the pipe that the tank is made of is certified by the maker, and as long as the technology of manufacturing meets specifications for the pipe, an object made out of it is also considered certified. Only a limited amount of testing that confirms the compliance with technological specifications during production is necessary. In addition, a coil design of the tank is safer in case of mechanical damage.

The delta velocity budget was calculated for three propellants: helium, nitrogen and liquefied propane C_3H_8 (refrigerating agent R290). The mass of gas propellant in the tank at full filling with temperature inside the tank T_t= 298K is calculated as for ideal gas:

$$m_{pr} = \frac{M \cdot P_t \cdot V_t}{R \cdot T_t} \ . \tag{3}$$

With the help of (3), initial propellant masses for various gases were calculated:

- for helium: m_{He}= 0,02 kg (M_{He}= 4 kg/kilomole);
- for nitrogen: m_N=0,142 kg (M_N=28 kg/kilomole);
- for propane m_{CH} = 0,177 kg (M_{CH} = 44,1 kg/kilomole; with the tank temperature 298K propane will be in liquid phase with density of 493 kg/m^3).

The exhaust velocities at de Lavale nozzle for cold gases (without heating) were found assuming the pre-nozzle temperature T= T_t= 298 K.

Then the maximum exhaust velocities in vacuum, with corresponding value of adiabatic exponent k, will be:

- for helium, w_{He}= 1757 m/s (k=1,67);
- for nitrogen, w_N =787 m/s (k=1,4);
- for propane, w_{CH} =1,13 (k=1,13).

The delta-V budgets were calculated according to Tsiolkovsky's formula:

$$\Delta V = w \cdot \ln(M_1 / M_2) \quad (4)$$

(supposing that the dry mass of the maneuvering module M_2 is 4 kg, while the mass of fully filled maneuvering module tank is $M_1 = M_2 + m_{pr.}$), and are:

- for helium: 8,8 m/s,
- for nitrogen: 27,4 m/s;
- for propane: 42,7 m/s.

These results are somewhat overestimated. The real-life exhaust velocities for helium and nitrogen will be approximately 10% lower, and for propane 15% lower.

The decrease in actual exhaust velocities can be explained by the following:

- the propellant mass for helium and nitrogen were calculated as for ideal gas, while at 35 MPa the molecules of the gas may occupy only a part of the geometric space of the tank;
- when the tank is filled with liquified propane, some part of it has to contain propane vapor to compensate for thermal expansion of the liquid in the tank;
- with such minor dimensions of the nozzle exit (the critical section diameter $d_{кр}$ is only .2mm) manufacturing tolerances may have a significant impact on exhaust velocity.

Nevertheless, calculations demonstrate that with helium as propellant, to achieve the specified delta-V budget of 20 m/s the propulsion system will require a tank of much bigger volume. Therefore, in view of the constraints imposed on the maneuvering module dimensions, the use of helium will not be practical.

If the goal is to achieve maximum delta-V budget, then the propane is preferable. Besides, the pressure in the tank in this case will not exceed 2 MPa, making it easier to achieve required strength of the tank, improve the conditions for valves, decrease leaks of propellant. However, propane is explosive, and thus unsafe for the launch vehicle when the nanosatellite is launched as way cargo.

As a result, nitrogen was selected as a propellant for the maneuvering module of the nanosatellite.

4. Calculations of thruster characteristics with electric heating of the propellant

Let us obtain more precise values of the maximum velocity at entry to and at exit from the nozzle, allowing for the propellant under expansion effect. For this, it is necessary first to obtain the relation between absolute pressures of the gases at the entry to and the exit from the nozzle p_e/ p. As follows from [1], this relation depends only on the value of propellant's expansion in the nozzle:

$$\frac{s_{ns}}{s_{cr}} = \sqrt{\frac{k-1}{2} \cdot \left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}} / \sqrt{\left(\frac{p_e}{p}\right)^{\frac{2}{k}} - \left(\frac{p_e}{p}\right)^{\frac{k+1}{k}}}} , \quad (5)$$

where S_{ns} is the area of the section of the nozzle exit; S_{cr} – is the area of critical section of the nozzle.

By substituting to (5) the values $\frac{s_{ns}}{s_{cr}} = 100$, k=1.4, the sought-for pressure relation is found: $\frac{p_e}{p} = 0.000256$.

Then, by calculating (1) with actual degree of expansion of the gas in the nozzle, we obtain actual exhaust velocity of nitrogen: $w_n = 749\ m/s$. The exhaust velocity value at nozzle edge is approximately 5% lower than the maximum exhaust velocity of nitrogen obtained earlier $w_n = 789\ m/s$ and the expected delta-V value will also decrease to 26.1 m/s.

Let us see if the condition of nearly complete emptying of the tank is met (by determining the mass of the nitrogen that can stay in the tank after the pressure in the tank will drop to such levels where the exhaust velocity in the critical section of de Lavale nozzle will be less than the local speed of sound and will depend on the pressure in the tank).

From [1] the critical relation of pressures is defined as:

$$z_{cr} = \left(\frac{2}{k+1}\right)^{\frac{k}{k-1}}, \quad (6)$$

For nitrogen, the de Lavale nozzle will operate according to calculated mode until the critical pressure relation (=0.5282) is achieved. From the condition that the pressure of the propellant at the entry to the nozzle equals the pressure in the tank P_t it is possible to find the minimum pressure of nitrogen in the tank $P_{t\,min}$ at which the nozzle will still be operating in the calculated mode. Assuming the pressure of the environment equals $p_{en} = 1\cdot10^{-6}$ MPa, then the minimum pressure in the tank will be $P_{t\min} = \frac{p_{en}}{z_{cr}} = 1.89\cdot10^{-6} MPa$, which corresponds to the residual mass of nitrogen in the tank approximately $5\cdot10^{-9}$ %.

$$m_{Nres} = \frac{M\cdot P_{t\min}\cdot V_t}{R\cdot T_t} = 0.00769\cdot10^{-9} kg.$$

Therefore, it can be assumed that nearly all mass of the nitrogen will be spent while the nozzle operates in calculated mode with supercritical pressure gap.

To increase the propellant exhaust velocity, which will in turn increase the delta-V budget, it is suggested to heat the propellant before the nozzle with the help of electric power generated by the nanosatellite's electric power supply system (EPSS).

An evident disadvantage of an electro-thermal propulsion system is that it requires significant increase in EPSS power. Additional power, required for heating the whole amount of nitrogen stored in the tank, can be estimated from the relation:

$$Q = C_{pN}\cdot m_N\cdot\Delta T \tag{7}$$

where: Q is the energy required to heat the whole supply of nitrogen, kJ; C_{pN} is specific thermal capacity of nitrogen at the given temperature, kJ/(kg*K); ΔT is the value of heating, K.

The estimate of additional electric power that EPSS has to generate for nitrogen heating must take into account the cyclogram of the thruster's operation for the whole operational life of the nanosatellite. For a hypothetical cyclogram of EPSS operation, where the thruster, during the mission, is to generate N_{imp} = 100 thrust impulses of the same length t_{imp} = 10, with the whole supply of the propellant spent to generate the impulses, the additional electric power P_m [Wt] required for propellant heating can be estimated from the relation:

$$P_m = \frac{Q}{N_{imp}}\cdot t_{imp} \tag{8}$$

Results of modeling for various temperatures to which the propellant is heated are shown in Table 1 and Figure 4 (the calculation results for Q are given not only in kJ, but also in the non-system unit watt-hour, which is more convenient to EPSS developers).

Table 1. Impact of propellant temperature on thruster parameters.

Nitrogen temperature before nozzle, K.	Nitrogen heating before nozzle, ΔT, K	Nitrogen exhaust velocity, m/s.	Delta-V budget, m/s.	Nitrogen specific thermal capacity, C_{pN}, kJ/(kg*K	Additional electric power required for heating the nitrogen in the tank, Q		ESSP power required for heating the total amount of nitrogen in the tank in 1000 s, W
					kJ	Wh	
298	0	749	26,1	1,040	0	0	0
398	100	866	30,2	1,042	14,8	4,11	14.8
498	200	968	33,8	1,056	30,0	8,33	30
598	300	1061	37,0	1,075	45,8	12,72	45.8
698	400	1146	40,0	1,092	62,0	17,22	62,0
798	500	1226	42,8	1,115	79,2	22,0	79,2

898	600	1300	45,3	1,140	97,1	26,97	97,1

Fig. 4 shows the graphs of the changes of delta V and required additional EPSS power according to the temperature to which nitrogen is heated.

According to the obtained results, let us state additional requirements for the nanosatellite EPSS. The power of on-board EPSS of CubeSat3U nanosatellites that is required for functioning of on-board systems is consumed over a long period of time and is, on the average, 3-5 Wt. The power required for propellant heating P_m, by contrast, is consumed in a short interval, as a so-called peak load. Let us estimate the average additional power, that on-board solar panels have to generate to charge on-board electric accumulators (batteries), from which the peak load is powered.

For this it is necessary to formalize the mission travel of the nanosatellite. Let us suppose that we must compensate for air drag of the nanosatellite for 30 days of flying at an altitude of 235 km. Then a calculation of two-impulse Zander-Hohmann's transfer shows the need to correct the orbit once in two days of the flight with total characteristic velocity expense approximately ΔV=30 m/s.

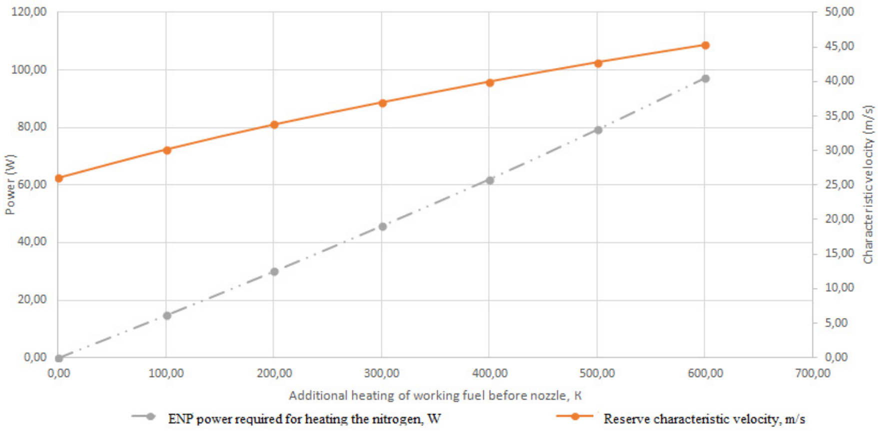

Fig. 4. Increase in delta-V budget and additional EPSS power required for propellant heating according to the range of heating

From Table 1 we can see that to achieve the ΔV=30 m/s the propellant must be heated by 100K, at the electric power expense of Q=8,33 watt-hour. Let us assume the efficiency ratio of solar-powered EPSS, according to expert evaluations, is η=0,6. With these assumptions,

$$P_{pc} = \frac{Q}{t_{ch} \cdot \eta} \tag{9}$$

where t_{ch} is the time when the batteries can be charged, in hours (in the case under consideration it is two days, or: t_{ch}=48h.)

Then P_{pc}=0,29 Wt.

With the altitude of 220 km, to compensate for air drag, all things being equal, the required delta-V is ΔV=45 m/s. Then, successful performance of this mission will require heating by 600 K and Q=26,97 watt-hour. Correspondingly, for this mission P_{pc}=0,936 Wt.

We can see that with sufficient time to charge the batteries the average electric power that the solar panels have to provide can be provided by a minor increase of solar panel size that does not necessitate substantial changes in the nanosatellite's design. The missions where the time when batteries can charge is small may be impossible for propulsion systems with propellant heating.

The peak power required from batteries is within the capacity of lithium ion batteries currently in use. It is also possible to use super capacitors that are beginning to find application in nanosatellite design [2].

5. Implementation of the maneuvering module propulsion system concept for a nanosatellite

The discussed propulsion system concept was used to design a maneuvering module for an aerodynamically stabilized nanosatellite that is being developed by the interuniversity chair of space exploration, Samara National Research University. The layout for prototype maneuvering module is represented in Fig. 5.

The space inside the tank is used to house a system of valves that feed pressurized nitrogen to the nozzle unit, which provides the required direction and value of velocity boost for the nanosatellite. The prototype sample will be tested to validate the feasibility of the design and technological decisions, after which the question of building a sample of a nanosatellite maneuvering module for an actual flight test will be discussed.

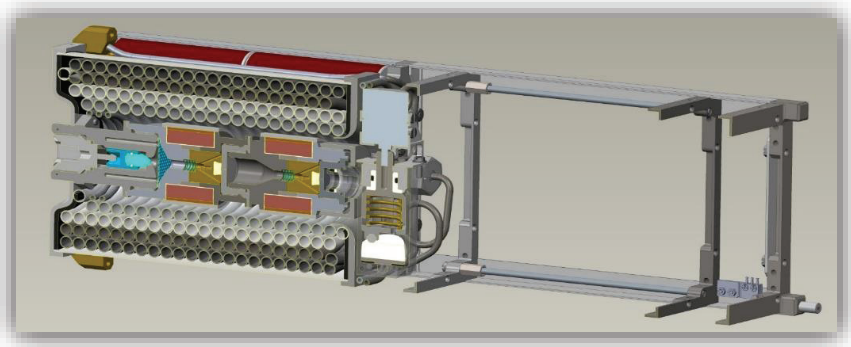

Fig. 5. Layout for a prototype nanosatellite maneuvering module

6. Conclusion

A concept of a propulsion system for a maneuvering module of a low-orbit aerodynamically stabilized CubeSat3U-class nanosatellite has been developed.

It is shown that electro-thermal propulsion system for a maneuvering module meets the requirements for low orbit station keeping of an aerodynamically stabilized nanosatellite.

Design documentation has been drawn up for a prototype that is being built for ground testing to validate the technological and design decisions.

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