УДК 629.78

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ANALYSIS OF TECHNOLOGIES FOR SPACECRAFT REMOVAL FROM LOW EARTH ORBITS USING ONBOARD-PRODUCED ELECTROMAGNETIC AND MAGNETIC FIELDS

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Мета статті – аналіз ефективності застосування двигунних установок з постійними магнітами у якості альтернативного способу відведення об'єктів космічного сміття з низьких навколоземних орбіт.

У роботі розглянуто поточні завдання, пов'язані з розробкою методів і створенням засобів відведення космічних апаратів, термін активного існування яких закінчився, з робочих орбіт за допомогою власних електромагнітних і магнітних полів, що генеруються за допомогою різних магнітних пристроїв. Проведено огляд сучасних систем відведення космічних апаратів з низьких навколоземних орбіт, що базуються на генерації власного електромагнітного поля, що при взаємодії з динамічним потоком іоносферної плазми і магнітним полем Землі створює силу гальмування і таким чином здійснює відведення космічного апарату з орбіти. Виявлено недоліки та переваги електромагнітних систем відведення космічних апаратів. Запропоновано альтернативний метод і систему відведення об'єктів космічного сміття за допомогою пристроїв з постійними магнітами. Представлено конструктивну схему пристрою з постійними магнітами і запропоновано алгоритм принципу її дії. Проведено аналіз екранів для магнітних і електромагнітних полів і обрано найбільш відповідний екрануючий матеріал. Обраний екрануючий матеріал являє собою багатошаровий екран, що складається з алюмінієвих, мідних і магнітних шарів. Розроблено математичну модель орбітального руху космічного апарату з пристроєм на постійних магнітах. За допомогою пакету прикладних програм SciLab розраховано час відведення для різних космічних апаратів і висот орбіт. Виходячи з отриманих результатів розрахунку зроблено висновок, що ефективність впливу сили гальмування залежить від співвідношення між інерційними характеристиками космічних апаратів та об'ємами постійних магнітів, що встановлюються на дані апарати. Було визначено, що для великих космічних апаратів, масою понад 2 т, використання двигунних пристроїв з постійними магнітами в якості систем відведення є неефективним. Це обумовлюється непропорційністю між зростанням сили гальмування в залежності від об'єму постійного магніту та зростанням інерційних характеристик космічного апарата при збільшенні його маси. Спираючись на дані результати визначено межі ефективного застосування двигунних пристроїв з постійними магнітами.

Цель статьи – анализ эффективности применения двигательных установок с постоянными магнитами в качестве альтернативного способа увода объектов космического мусора с низких околоземных орбит.

В работе рассмотрены текущие задачи, связанные с разработкой методов и созданием средств увода космических аппаратов, срок активного существования которых закончился, с рабочих орбит с помощью собственных электромагнитных и магнитных полей, генерируемых с помощью различных магнитных устройств. Проведен обзор современных систем увода космических аппаратов, базирующихся на генерации собственного электромагнитного поля, которое при взаимодействии с динамическим потоком ионосферной плазмы и магнитным полем Земли создает тормозящую силу и таким образом уводит космический аппарат с орбиты. Определены преимущества и недостатки электромагнитных двигательных систем космических аппаратов. Предложены альтернативный метод и система увода с орбиты объектов космического мусора с помощью двигательных устройств на постоянных магнитах. Представлена конструктивная схема устройства с постоянными магнитами, и предложен алгоритм принципа ее действия. Проведен анализ экранов для магнитных и электромагнитных полей, и выбран наиболее подходящий экранирующий материал. Выбранный экранирующий материал представляет собой многослойный экран, состоящий из алюминиевых, медных и магнитных слоев. Разработана математическая модель орбитального движения космического аппарата с устройством на постоянных магнитах. С помощью пакета прикладных программ SciLab рассчитано время увода для разных космических аппаратов и высот орбит. Исходя из полученных результатов расчета сделан вывод, что эффективность воздействия тормозящей силы зависит от соотношения между инерционными характеристиками космических аппаратов и объемов постоянных магнитов, которые устанавливаются на данные аппараты. Определено, что для больших космических аппаратов, массой свыше 2 т, использование двигательных устройств с постоянными магнитами в качестве систем увода является неэффективным. Это обуславливается непропорциональностью между ростом тормозящей силы в зависимости от объема постоянного магнита и ростом инерциальных характеристик космического аппарата при увеличении его массы. Опираясь на данные результаты определены границы эффективного применения двигательных устройств с постоянными магнитами.

The aim of this paper is to analyze the effectiveness of propulsion devices with permanent magnets as an alternative space debris deorbit system for low earth orbits.

The paper considers current problems in the development of methods and means for deorbiting used spacecraft with the help of electromagnetic and magnetic fields produced by different onboard magnetic devices and overviews state-of-the art spacecraft deorbit systems that use an onboard-produced electromagnetic field whose interaction with the incident flow of the ionospheric plasma and the Earth's magnetic field produces an additional drag force, thus deorbiting the spacecraft. The advantages and disadvantages of electromagnetic spacecraft propulsion systems are identified. An alternative method and system are proposed for deorbiting space debris objects using permanent-magnet propulsion devices. A construction diagram of a permanent-magnet device is presented, and an algorithm of its operation is proposed. Magnetic and electromagnetic field shields were analyzed, and the most appropriate shielding material was chosen: a multilayer shield that consists of aluminum, copper, and magnetic layers. A mathematical model of the orbital motion of a spacecraft with the permanent-magnet device was developed. Using SciLab, the deorbit time was calculated for different spacecraft and different altitudes. From the calculated results it was concluded that the effectiveness of the magnet-produced drag force depends on the relation between the spacecraft's inertial characteristics and the permanent magnet volume. It was found that permanent-magnet propulsion devices as deorbit systems are ineffective for large spacecraft heavier than 2 t. This is due to the fact that the increase in the magnet-produced drag force with the permanent magnet volume is not in proportion to the increase in the spacecraft's inertial characteristics with the spacecraft mass. Using these results, the range of effective use of permanent-magnet propulsion device was determined.

Key words: *permanent magnets, spacecraft, de-orbit system, magnetic field, electromagnetic field, electrical field, dynamic flux of ionospheric plasma.*

Introduction. According to the tendency of increasing numbers of space debris objects, the issue of Near-Earth Space protection is becoming more and more actual. So, on April, 2018, about 14156 space debris objects were cataloged by the National Aeronautics and Space Administration (NASA) in USA [1]. As a result, there is a task of the development of most effective de-orbit systems and technologies. For today, there are two methods of the space debris objects removal from Low Earth's Orbits (LEO): active debris removal and passive debris removal. The active debris removal (ADR) involves the use of the Space Servicing Vehicles (SSV) or propulsion systems which are integrated into the spacecraft for removing it from LEO after the expiration of its lifetime [2]. The ADR includes SSV systems such as LEOSweep, robotic captured manipulators, etc. and integrated propulsion systems such as micro reactive propulsion thrusters (chemical thrusters, electrical reactive thrusters ets.) [3]. The advantages of ADR systems are the opportunity to control the removal process and the possibility to determine the accurate place of falling of large space debris objects. However, significant fuel consumption, control problems, and additional costs for launching of SSV make ADR not so attractive for today. That's why, another concept with using passive debris removal (PDR) is more popular in practice due to its simplicity. The PDR systems include aerodynamic de-orbit systems, electromagnetic passive systems, solar sails, magnetic and electrical sails [4]. The aerodynamic de-orbit systems have such good efficiency, described in [5], but due to their large size, there is a probability of impingement with space debris fragments and damages of the shell. The same can be determined for the solar sails systems. That's why, a new approach using contactless interaction with the Near-Earth Environment and using magnetic and electromagnetic fields has been proposed by many Space Agencies. This approach is based on the using self-generated magnetic or electromagnetic field of spacecraft with the use of additional devices. This magnetic or electromagnetic field interacts with Earth's magnetic field and dynamic flux of ionospheric plasma and braking force is generated due to this interaction. The most popular proposed such systems are: electromagnetic deorbit devices and electromagnetic sails.

Electromagnetic deorbit devices. Electromagnetic deorbit devices are based on generating magnetic field of spacecraft with the use of special electrical devices, such as conductive coils, electromagnetic tethers, difficult lattice conductive constructions, etc. According to Bio Savar's Law, electrical current which is flowed to these devices generates electromagnetic field around conductive wires and on the whole spacecraft [6]. This electromagnetic field which is generated by special electrical devices interacts with dynamic flux of ionospheric plasma and Earth's magnetic field. As a result of this interaction and according to the theory of physics of plasma, braking force is generated [7]. One of the well-known examples of such systems is electromagnetic tether. The best example of electromagnetic tethers, which has full scientific description, is Terminator Tether (TT) concept (fig.1) [8 – 10].

Figure 1 – Terminator Tether deorbit system

TT concept is based on the Electromagnetic Induction Law (EIL) and magnetic field – ionospheric plasma interaction. After deploying of the long conductive TT and stabilizing it with extra controller, the electrical current is induced in this tether. Then TT begins interaction with ionospheric plasma flux and Earth's magnetic field. The result of its interaction is electrical current which is induced in TT. According to the EIL, self-electromagnetic field is generated around this tether. The drag (braking) force is the effect of the interaction between flux of ionospheric plasma and self-electromagnetic field.

The estimations of the deorbit time with the use of TT have perfect results, which are presented in table 1 [8].

Table 1

Constellation	Altitude (km)	Inclination (degrees)	Deorbit Time no TT (Derelict)	Initial Orbit Decay Rate (km/day)	Deorbit Time with TT
Orbocomm 1	775	45	100 years	44	11 days
Orbocomm 2	775	70	100 years	11.6	41 days
LEO One USA	950	50	600 years	32	18 days
GlobalStar	1390	52	9000 years	22.3	37 days
Skybridge	1475	55	11000 years	18.5	46 days
FaiSat	1000	66	800 years	13.5	45 days
Iridium	780	86.4	100 years	2.1	7.5 months
M-Star	1350	47	7000 years	27	28 days
Celestri	1400	48	9000 years	26	32 days
Teledesic	1350	~85	7000 years	1.7	17 months

While analyzing these results, it can be concluded that exploiting TT as a deorbit system is very effective, because it can be used for great diapason of altitudes and has very short deorbit time compared with analogical systems. However, exploitation of TT has some difficulties, connected with stabilization of the relative position of the tether. The declared length of TT about 5 – 10 km makes stabilization of stretching very difficult realized in practice. Therefore, TT regardless of their effective key indicators has not found high popularity.

Another system is **the electromagnetic sail** which was presented by Japanese scientists Ikkoh Funaki and Hiroshi Yamakawa from Japanese Aerospace Research Agency [7]. The best description of this concept is given in PhD dissertation of the Japanese junior scientist Yasumasa Ashida [11]. This concept is based on interaction between flux of solar wind and generated dipole electromagnetic field. As a result of this interaction, thrust force is generated (fig. 2). The electromagnetic field around spacecraft is generated using large super conductive coils. The radius of these coils is proposed about 20 km. The magnetic sails are offered to use in interplanetary space for long missions of researching solar system. However, such difficult design of coils can be not realized in practice today because of their large size. Another side of difficulties connected with practical realization is the shortage of onboard power. According to the calculations of Japanese scientists, the levels of current and voltage which are needed for circuit power supply are very high.

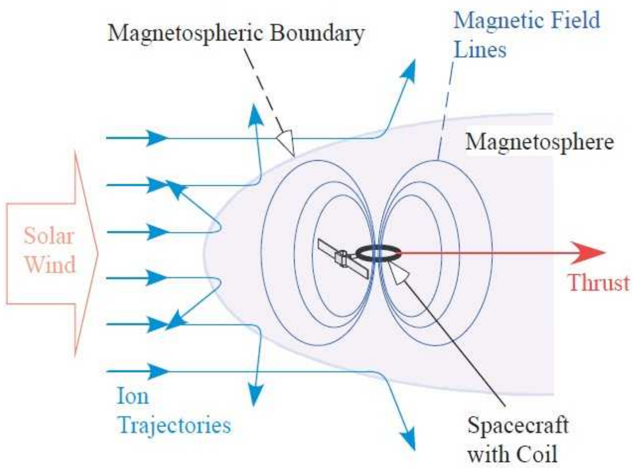

Figure 2 – The principle of work of magnetic sails

So, the required current for coil is 2 kA and the required voltage is about 20 kV. It means that super power source of onboard electrical energy is needed for these requirements. That's why this concept has only theoretical side of rationale. It can be concluded, that implementation of electromagnetic solar sails as a deorbit systems are not expedient. Considering all difficulties connected with the use of electromagnetic solar sails, the new approach based on using **permanent magnets (PM)** was developed [12 – 17]. Professor V. Shuvalov has proved the existence of forces arising from the interaction of a permanent magnet with a stream of ionospheric plasma, which can be used to control the orbital motion of a spacecraft, including removing space debris from orbits. He carried out an experimental verification of this effect and developed the corresponding mathematical model [12 – 15]. On this basis, studies [16, 17] were carried out to obtain an estimate of the duration of the deorbiting of spacecraft from orbits of various dislocations. Applying PM as the thrust devices requires additional protective screens, which will protect electrical equipment from magnetic interference.

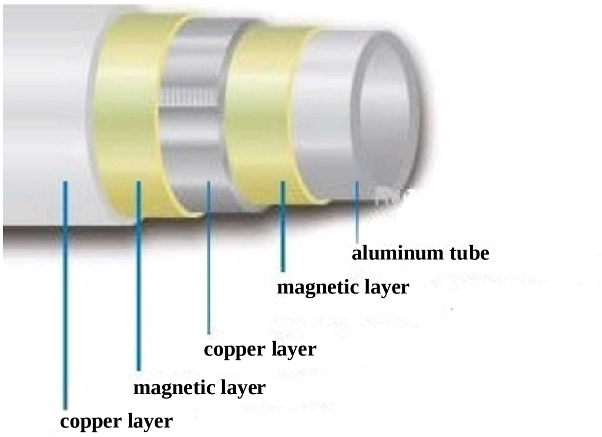

Figure 3 – Protective multi-layer screen

There are many ways of screening magnetic field described in research works of different scientific centers, the studies of which are connected with exploration of materials [18 – 22]. One of the most popular approaches of screening permanent magnetic field is using multilayered screens which consist of different materials [18]. This approach has a good description in the research of scientists from National research Nuclear University “MEPhI” [22]. The multi-layer screen consists of five layers from different screening materials such as cuprum, magnetic layer and aluminum tube (fig. 3). This screen is proposed to use in devices with PM. The principle scheme of devices with PM is shown in fig. 4.

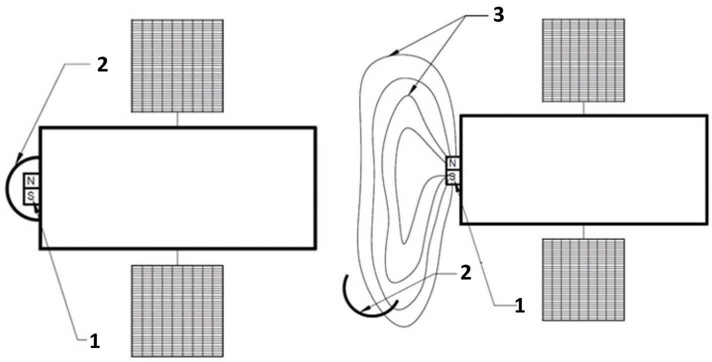

Figure 4 – Principle scheme of devices with permanent magnets, where 1 – permanent magnet, 2 – special protective screen, 3 – generated magnetic field.

The principle of work of these devices is described by the following algorithm which consists of three steps:

1. Activation of the device after the expiration of spacecraft lifetime;
2. Detachment of the special protective screen;

3. Generation of the self-magnetosphere and braking force, starting deorbit process.

Calculations of effectiveness of the devices with PM were carried out with the use of equations of the spacecraft orbital motion in osculating elements:

$$\left.\begin{aligned}
&\frac{da}{d\vartheta}=\frac{2pr_{KA}^2}{\mu\left(1-e^2\right)^2}\left(S\cdot e\sin\vartheta+T\cdot\frac{p}{r_{KA}}\right),\\
&\frac{de}{d\vartheta}=\frac{r_{KA}^2}{\mu}\left\{S\cdot\sin\vartheta+T\cdot\cos\vartheta\left(1+\frac{r_{KA}}{p}\right)+T\cdot e\frac{r_{KA}}{p}\right\},\\
&\frac{di}{d\vartheta}=\frac{r_{KA}^3}{\mu p}\cos\left(\vartheta+\omega\right)\cdot W,\\
&\frac{d\Omega}{d\vartheta}=\frac{r_{KA}^3\sin\left(\vartheta+\omega\right)}{\mu p\sin i}W,\\
&\frac{d\omega}{d\vartheta}=\frac{r_{KA}^2}{\mu e}\left\{-\cos\vartheta S+\left(1+\frac{r_{KA}}{p}\right)\sin\vartheta T\right\}-\\
&\qquad-\cos i\,\frac{r_{KA}^3\sin\left(\vartheta+\omega\right)}{\mu p\sin i}W,\\
&\frac{dt}{d\vartheta}=\frac{r_{KA}^2}{\sqrt{\mu p}}\left\{1+\frac{r_{KA}^2}{\mu e}\left[\cos\vartheta S-\left(1+\frac{r_{KA}}{p}\right)\sin\vartheta T\right]\right\},
\end{aligned}\right\}, \qquad (1)$$

where μ – gravitational constant; a – semi-major axis of the orbit; e – eccentricity; i – inclination; Ω – argument of perigee; ω – right-ascension of the ascending node; ϑ – true anomaly; t – time of the orbital motion of spacecraft; r_{KA} – radius-vector of the spacecraft; p – focal parameter of the orbit; S, W, T – projections of the perturbing acceleration on the axis of the orbital coordinate system.

The projections of the perturbing acceleration consist of aerodynamic perturbing acceleration, gravitational perturbing acceleration and perturbing acceleration of braking force which is generated by interaction of self-magnetosphere of spacecraft and dynamic flux of ionospheric plasma. This can be written in the next form:

$$\begin{aligned}
S &= S_G + S_A\\
T &= T_G + T_A + T_{BF}\ ,\\
W &= W_G + W_A
\end{aligned} \qquad (2)$$

where S_G, W_G, T_G – projections of the gravitational perturbing acceleration; S_A, W_A, T_A – projections of the aerodynamic perturbing acceleration; T_{BF} – projection of the perturbing acceleration of braking force.

Projection of the perturbing acceleration of braking force T_{BF} is projected only to T-projection of the orbital coordinate system in the direction of braking force. Perturbing acceleration T_{BF} is calculated from the mathematical model of interaction between ionospheric plasma and generated magnetic field [12 – 15].

In turn, the altitude of the orbit H is connected with the semimajor axis a by the following expression:

$$a = \frac{H + R_E}{1 - e}, \tag{3}$$

where $R_E = 6371$ km – the radius of Earth.

Using mathematical model [12 – 17] and equations (1) – (3), there have been obtained the following estimations of the deorbit time for some near circular orbits and spacecrafts (Table 2).

Table 2

Mass of space-craft/mass of device with PM (kg)	Altitude (km)	Inclination (degrees)	Eccentricity	Average cross-section area of space-craft m^2	Deorbit Time with devices with PM
661.5/19.845	600	45	0.005	20	2.485 years
1753.3/52.599	650	30	0.005	6.48	5.5 years
508/15.24	700	60	0.005	12.9	6.1 years
743/22.29	750	52	0.005	10	7.2 years
1923/57.69	800	55	0.005	15	11.3 years
300/9	850	66	0.005	4	10.25 years
1000/30	900	86.4	0.005	6	31.4 years
400/12	900	86.4	0.005	4	23.14 years

Another calculation has been carried out with fixed mass, inclination of the orbits and average cross-section area of spacecraft (Table 3). All calculations have been carried out according to the requirements of mass ratio (mass of the deorbit system = 3 % of mass of the spacecraft).

Table 3

Mass of space-craft/mass of device with PM (kg)	Altitude (km)	Inclination (degrees)	Eccentricity	Average cross-section area of space-craft m^2	Deorbit Time with devices with PM
400/12	600	45	0.005	4	1.15 days
400/12	650	45	0.005	4	2.458 years
400/12	700	45	0.005	4	4.517 years
400/12	750	45	0.005	4	7.173 years
400/12	800	45	0.005	4	9.523 years
400/12	850	45	0.005	4	13.8 years
400/12	900	45	0.005	4	18.081 years

Analyzing these estimates, the next conclusions can be made:

1. At first, deorbit time depends on ratio between inertial characteristics of spacecraft and mass of device with PM which is loaded to this spacecraft.

Though, else if the requirements of mass ratio correspond, it can be seen that for different masses and sizes of spacecraft we observe different values of deorbit time. It can be explained that generated braking force which depends on the size of PM increases not as fast as inertial characteristics of spacecraft depending on mass. It can be seen from the seventh row in Table 2.

2. It should be emphasized that deorbit time depends on the orbit parameters of spacecraft. Thus, with the same altitudes of orbits and the same masses and sizes of spacecraft, but with different inclinations different values of deorbit time have been obtained (the 7-th row of Table 2 and the 7-th row of Table 3).
3. To obtain rational parameters of deorbit system based on device with PM many factors should be considered that affect the deorbit time of spacecraft. The main factors which should be considered are: mass and size of the spacecraft, mass and size of the device with PM, orbit parameters, value of generated braking force, ratio between generated braking force and inertial characteristics of spacecraft. After analyzing these factors, the conclusion can be made about expedience or inexpedience of implementation of the device with PM. Thus, analyzing the estimates from Table 2 and Table 3 it can be concluded that with the increasing of the mass of spacecraft the effectiveness of using devices with PM decreases.

Conclusions. As a result of the analysis of the deorbit technologies of spacecraft from LEO using its self-electromagnetic and magnetic fields, their advantages and disadvantages have been determined. The new approach which is based on using devices with PM has been explored. The obtained estimates of the results of the study show that efficiency of implementation of the devices with PM depends on many factors. To achieve maximum efficiency, these parameters should be analyzed first of all when designing a deorbit system for each spacecraft considering its future use. So, within the requirements of mass ratio, with the increasing of inertial parameters of spacecraft the efficiency of the use of the devices with PM decreases. That's why the maximum efficiency of the use of the devices with PM is observed for small and medium spacecrafts.

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Received on September 14, 2018,
in final form on December 10, 2018