Development and testing of low-power EP based on the ablative pulsed plasma thruster for an ERS SSC

A V Bogatyi, G A Dyakonov*, D A Kashirin, G A Popov and S A Semenikhin
Research Institute of Applied Mechanics and Electrodynamics, Moscow Aviation Institute, 5 Leningradskoe shosse, 125080, Moscow, Russian Federation

*E-mail: riame3@mai.ru

Abstract. The paper presents the appearance of a low-power electric propulsion system for a small remote sensing satellites based on an ablative pulsed plasma thruster. Preliminary analysis for the purpose and fields of application of ablative pulsed plasma thrusters (APPT) was carried out. The design appearance of a thruster for small remote sensing satellites was identified, and the main directions of APPT further development were defined. Methods for calculating the APPT propulsion characteristics were developed. The breadboard model of APPT for small remote sensing satellites was developed and made (APPT-350). Research tests of the breadboard model of APPT for Earth remote sensing small spacecrafts were carried out.

1. Introduction
Nowadays small spacecraft (SSC) with the mass from 10 kg to 1000 kg can solve such actual tasks as the Earth remote sensing (ERS), navigation, mapping, and communication quite efficiently, especially if they are combined into orbital systems, including two or more SSC equipped with optoelectronic, radar and other equipment providing high resolution due to adding apertures of individual SSC’ equipment [1].

The operating conditions of low-orbit SSC require maintenance and regular correction for their orbits, which makes it necessary to use small-size electric propulsion systems (EPS) capable of operating efficiently in conditions of limited consumption of electric power [2].

2. The task of maintaining a low Earth orbit (LEO) for ERS SSC
One of the typical tasks of a low-power EPS is to maintain the SSC low circular Earth orbit [3].

The aerodynamic drag force \( F_a \) acting on the spacecraft orbiting at velocity \( V \) is:

\[
F_a = \frac{1}{2} C_d \rho V^2 S_m
\]

here \( \rho \) is the atmospheric density, which in the first approximation, when its fluctuations due to solar radiation are not taken into account, depends on the orbit height \( h \) above the Earth level only and is regulated by the government standard GOST 4401-81 for the International Standard Atmosphere (ISA); \( C_d \) is the aerodynamic drag coefficient (for a free-molecular gas flow that occurs at densities corresponding to the upper layers of the atmosphere \( h > 200 \text{ km} \), \( C_d \approx 2.3 \)), and \( S_m \) is the mid-section area of the spacecraft.

In the simplest case of a circular orbit with height \( h \), the spacecraft velocity is defined by the following formula:

\[
V^2 = \frac{G M}{(R_E + h)}
\]

here \( G \) is the gravitational constant, \( M \) is the Earth mass, and \( R_E \) is the average radius of the Earth.
The characteristic velocity $V_x$ necessary to maintain a conditional circular orbit with height $h$ during time $T$ is: $V_x = F_a \cdot T/m_{sc}$, where $m_{sc}$ is the spacecraft mass.

Figure 1 shows the calculated dependences of the aerodynamic drag force $F_a$ averaged in accordance with ISA, and of the characteristic velocity $V_x$ necessary to maintain the circular orbit of a conventional small satellite with mass $m_{sc} = 100$ kg and with a mid-section area of $1 \text{ m}^2$ on the orbit height $h$ during one year ($T=3.16 \times 10^7 \text{ s}$). The calculation assumed an aerodynamic drag coefficient $C_d = 2.3$.

The characteristic velocity is related to the parameters of the propulsion system by the Tsiolkovsky formula:

$$V_x = J_{sp} \cdot \ln[m_{sc}/(m_{sc} - m_p)] \quad (3)$$

Here $J_{sp}$ is the specific impulse (mass-average efflux velocity) of the propulsion system; $m_{sc}$ is the total SSC mass (with the propellant storage); $m_p$ is the propellant storage.

For the electric propulsion systems, as a rule, $m_p << m_{sc}$, so without compromising accuracy, the latter formula can be replaced by the simpler ratio: $V_x = J_{sp} \cdot m_p/m_{sc}$ or $V_x = J_{sp} \cdot m_{sc}/m_{sc}$, where $J_{sp} = J_{sp} \cdot m_p = m_{sc} \cdot V_x$ is the total thrust pulse of the propulsion system.

Figure 2 shows the calculated dependences of the required total thrust pulse $J_{sp}$ on the time of low circular orbit maintenance for a conventional SSC of 100 kg in mass having a mid-section area of $1 \text{ m}^2$ for various heights $h$ of a low circular near-Earth orbit.

![Figure 1](image1.png) ![Figure 2](image2.png)

**Figure 1.** Calculated dependences of the aerodynamic drag force $F_a$ and of the characteristic velocity $V_x$ necessary to maintain circular orbit of conventional SSC ($m_{sc} = 100$ kg, $S_m = 1 \text{ m}^2$) on the orbit height $h$, for one year.

**Figure 2.** Calculated dependences of the required EPS total thrust pulse on the given time of the low circular orbit maintenance for a conventional SSC ($m_{sc} = 100$ kg, $S_m = 1 \text{ m}^2$) for different orbit heights $h$.

It is known that the International Standard Atmosphere is not recommended for ballistic analysis of the orbits of artificial Earth satellites, since it does not take into account significant fluctuations in the density of the upper atmosphere depending on the time of day, season, and solar activity. However, such a simplified approach allows us to obtain estimates for the minimum thrust and total thrust pulse of the EPS required to maintain the orbit with a given height and is quite acceptable for assessing the feasibility of using a propulsion system of one or another type.

It is obvious from data presented in figure 2 that the total thrust pulse necessary to maintain the SSC orbit with the height in range of 400-700 km for a period from 1 to 10 years is in the range from 1 to 30 kN·s. If it is required to remove SC from its orbit after the end of lifetime, the required total thrust pulse is approximately doubled. In this case, the averaged aerodynamic drag force of a spacecraft with an average midsection of $1 \text{ m}^2$ at the height of 400 km and higher does not exceed 0.4 mN, which allows the use of various electric propulsion thrusters with a thrust of at least 1 mN for maintaining such orbits. The ratio of the averaged aerodynamic drag force to the available thrust is
approximately equal to the ratio of EPS-on time at each revolution to the orbit period – the relative engine time.

3. **Ablative pulsed plasma thrusters**

One of the promising thrusters for solving the problems of controlling the motion of remote sensing spacecraft is the ablative pulsed plasma thruster (APPT), which uses plasma of the products of ablation of solid substance (Teflon-4) as the propellant. The place of pulse plasma thrusters in a number of other EPS is shown in figure 3.

The principle of operation of APPT with rail geometry and side feed of the solid plasma-forming substance is explained by figure 4. The thruster comprises: an energy storage unit consisting of one or several high-current low-inductance capacitors; the plane electrodes made of copper usually; one (in the case of end feed) or two (for the side feed) Teflon bars with rectangular cross-section used as propellant; and a high-voltage discharge igniter. The energy storage unit, electrically connected to the thruster electrodes, is charged by the power supply unit. A pulsed discharge lasting a few microseconds occurs between the electrodes following the discharge of the high-voltage igniter. Due to the ablation of Teflon bars in the process of discharge along their surfaces and subsequent ionization of the evaporated substance (propellant), plasma is formed, which is accelerated by gas-dynamic and electromagnetic forces. The highest efficiency of acceleration process is reached with the domination of electromagnetic accelerating force formed in the discharge (accelerating) channel as follows:

In the accelerating channel, the discharge current generates magnetic field, the basic component \( B_z \) of which is directed along the transverse axis \( z \). When interacting with discharge current with the density \( j_y \), this component generates a bulk electromagnetic force \( F_x = j_y \times B_z \) directed along the flow axis \( x \). The resulting plasma is accelerated by the electromagnetic force to a velocity of 20...30 km/s. The plasma flow is sufficiently well focused (the flow divergence angle is \( \sim 30° \)). After the discharge ends, the remaining plasma flows out at thermal velocities, which also makes a noticeable contribution to the thrust produced by the thruster.

The APPT-based EPS has the following primary advantages:
- independence of specific characteristics of EPS on power consumption (linear thrust dependence on power);
- simplicity of design and electrical circuits and absence of expensive materials resulting in low EPS cost in total;
- low mass and simplicity of the power processing unit (PPU);
- simplicity and reliability of the solid propellant storage and feed system (SPSFS);
- cheap and non-deficient propellant (Teflon-4);
- continuous operation readiness and precise value of the thrust pulse, which is explained by the low value of impulse bit.

4. A family of APPT-based EPS developed by RIAME MAI

Currently, a number of electric propulsion systems based on ablative pulse plasma thrusters with the discharge energy from 8 to 155 J are developed at RIAME MAI (Research Institute of Applied Mechanics and Electrodynamics of the Moscow Aviation Institute), which are shown in figure 5 [4-6]. All propulsion systems of the APPT series are mainly intended for correcting and maintaining the orbit of low-orbit SSC. The most advanced of them are: APPT-45-2, APPT-155 and APPT-95, which passed the full range of ground experimental testing (in the case of APPT-95, with the exception of lifetime tests). The APPT-45-2-based EPS, designed for the scientific small spacecraft “MKA-FKI PN2”, was launched into LEO in 2014 (ref. to figure 6). The APPT-155-based EPS was designed for the small remote-sensing satellite “Soyuz-Sat-O” and differed from the APPT-45-2 by the increased total pulse, since its functions included not maintaining the orbit only, but the satellite deorbiting at the end of its active life also. The most powerful of the presented EPS family is the APPT-95-based, which is being developed jointly with the Research Institute of Electromechanics (NIIEM) for the scientific SSC “Ionosphere-M” [7].

![Figure 5. A family of APPT-based EPS developed at RIAME MAI (APPT-95 is being developed jointly with NIIEM) [4].](image1)

![Figure 6. Small scientific spacecraft MKA-FKI PN2 with the APPT-45-2-based EPS [8].](image2)
Primary performance of APPT developed by RIAME MAI are presented in table 1.

### Table 1. Primary performance of APPT developed by RIAME MAI.

| EP Model | APPT-120 | APPT-45-2 | APPT-250 | APPT-155 | APPT-95 |
|----------|----------|-----------|----------|----------|--------|
| Discharge energy, J | 20 | 55 | 62 | 88 | 155 |
| Power consumption, W | 60 | 75…150 | 60…120 | 70…140 | 170 |
| Propellant storage lifetime, h | 220 | 3860 | 3600 | 5950 | 3970 |
| Specific impulse, m/s | 7160 | 11000 | 12000 | 13200 | 16000 |
| Average thrust, mN | 0.9 | 1.44…2.9 | 1.2…2.4 | 1.4…2.8 | 3.5 |
| Total pulse, kN·s | 0.7 | 20 | 15.6 | 30 | 50 |
| Propellant storage, kg | 0.1 | 1.8 | 1.3 | 2.4 | 3.15 |
| Total EPS mass, kg | 3.0 | 10.5 | 8.2 | 14.0 | 20.0 |
| Thrust cost, W/mN | 67 | 52 | 50 | 52 | 49 |
| Effective specific impulse, m/s | 233 | 1900 | 1900 | 2140 | 2600 |

5. **Main indices characterizing an APPT-based EPS**

Main external characteristics of the thruster, which are to be determined at the test facility, are the following:

- average thrust
  \[ F = p \cdot f \]  

- power consumption
  \[ N = W \cdot f \]  

- and mass consumption per pulse \( m \) that is defined by the weight method. Here \( p \) \([N\cdot s]\) is the thrust impulse bit, \( W \) \([J]\) is the discharge energy, and \( f \) \([Hz]\) is the pulse repetition rate.

The technical perfection of a thruster is characterized by its specific parameters:

a) thrust efficiency (it should not be confused with thermal and electrodynamic efficiencies, which are usually much higher than \( \eta_t \)):

\[ \eta_t = \frac{p^2}{2 \cdot m \cdot W} \]  

b) specific impulse that is equal to mean-mass velocity of the plasma flow

\[ J_{sp} = \frac{p}{m} \]  

c) thrust cost

\[ C_t = \frac{N}{F} \]  

Not all specific characteristics of EPs, though reflecting their technical perfection, are equally important for the development of propulsion systems for real spacecraft. The accomplishment of its target function by the electric rocket propulsion system is determined, first of all, by the characteristic velocity \( V_x \), which it provides to the spacecraft with the mass \( m_{sc} \). The characteristic velocity, in its turn, defines the total thrust pulse of the propulsion system \( P_2 \) (a SC is considered a body of constant mass, as the EPS propellant storage \( M_p \) is normally substantially lower than \( M_{sc} \)):

\[ P_2 = M_{sc} \cdot V_x \]  

Thus, the total thrust pulse is the most important characteristics of EPS; it is defined by the APPT specific impulse and propellant storage:

\[ P_2 = M_p \cdot J_{sp} \]  

The specified total thrust pulse of EPS can be achieved equally by increasing the specific thrust pulse and by increasing the propellant storage.
The second EPS characteristics that is the most important for spacecraft developers is its total mass $M_{EPS}$. The EPS mass depends on the following:
- mass and size of propellant bars, i.e. on the given total pulse;
- specific mass of capacitors;
- mass and redundancy level of the PPU electronic modules.

Thus, the most important specific characteristics that shows the perfection of an EPS is the effective specific impulse $J_{sp}$ [N·s/kg], equal to the ratio of the EPS total thrust pulse to its total mass, including the propellant storage:

$$J_{spe} = \frac{P_z}{M_{EPS}}$$

Absolute values of thrust and power consumption, and thus of the thrust cost, are of substantially lower value for an APPT-based EPS. In particular, the lack of power on board the SSC can be compensated by reducing the pulse rate and, accordingly, by increasing the thruster operation time. In this case, the discharge energy will remain the same and, accordingly, all specific characteristics of the thruster will remain unchanged. The lack of thrust is also compensated by the increase in the thruster operation time.

6. Semi-empirical method for calculating APPT characteristics

Formulas (4)...(11) are the basic computational relationships that define the APPT external appearance at the design stage. According to the results of multiple experimental works, which are reviewed in [4], the thrust impulse bit $p$ depends linearly on the discharge energy $W$, while the average thrust $F$ is, correspondingly, directly proportional to the power consumption $N$. The behavior of this empirical relationship is shown in figure 7.

![Figure 7. The APPT impulse bit as a function of discharge energy.](image)

The greatest uncertainty in the calculation, in particular, in the most important relation (10), is introduced by the dependence of the specific impulse on the discharge energy. At present, it is not possible to obtain such a relationship by calculation. The available analytic and numerical models of plasma acceleration in pulsed plasma accelerators, ref. to [9], for example, include the varying empirical coefficients allowing the calculated current and voltage oscillograms to be adjusted to the experimentally measured.

However, the extensive experience in the development and testing of pulsed plasma thrusters available at RIAME MAI allows us to identify simple empirical relationships, common to the entire family of flight APPTs and their prototypes, presented in table 1. In particular, the experimentally obtained dependence of the specific impulse on the discharge energy is shown in figure 8.
It follows from figure 8 that dependence of the APPT specific impulse on discharge energy is interpolated quite well by the following empirical formula:

$$J_{sp}[m/s] = 3000 \cdot \sqrt[3]{W[J]}$$  \hspace{1cm} (12)

However, it should be noted that in the case of considerable variations in the operating processes in the discharge (accelerating) channel of APPT, the empirical formula (12) will likely lose its validity, and thus a large amount of additional experimental research will be required.

7. Possibilities of reducing the total mass of APPT-based EPS

The total mass structure of the APPT-based EPS is graphically shown in figure 9 by the example of APPT-250. In [10], the following ways for reducing the APPT-based EPS mass are analyzed:

- reducing the mass of the energy storage unit (ESU) by optimizing its energy content and increasing the storage density of high-power capacitors;
- reducing the mass of propellant and of its storage and feed system by increasing the specific impulse;
- reducing the mass of electronic units (PPU and discharge initiation unit – DIU) by using new circuit designs, new components and transition from the analog power and control circuits to the digital ones;
- reducing the mass of structure by using new structural materials.

As follows from the diagram in figure 9, the ESU mass and the mass of propellant with its storage and feed system are the main components of the total mass of the APPT-based EPS. Together they make up 70-75% of its total mass. In addition to the accumulating capacitors, the ESU comprises the discharge buses, insulators, primary structural elements and a compound used for casting high-voltage circuits.

![Figure 8](image8.png)

**Figure 8.** The APPT specific impulse as a function of discharge energy [10].

![Figure 9](image9.png)

**Figure 9.** The total mass structure for the APPT (ESU – energy storage unit, PSFS – propellant storage and feed system, PPU – power processing unit, DIU – discharge initiation unit) [10].

It is possible to conclude that the main directions for improving the mass-dimensional characteristics of APPT-based EPS are:
Reducing the mass of the energy storage unit by optimizing its energy content and using capacitors with increased specific storage density;

- Reducing the mass of propellant by increasing specific impulse using new plasma acceleration schemes.

Figure 10 shows the results of the calculated estimates for the total mass of the APPT-based EPS as a function of discharge energy with the given values for the total thrust pulse $P_\Sigma = 10 \text{kN} \cdot \text{s}; 30 \text{kN} \cdot \text{s}; 50 \text{kN} \cdot \text{s}$ and for the specific storage density of capacitors $\omega_C = 28 \text{J/kg}$, that corresponds to the modern pulsed energy storage capacitors.

It is obvious that the discharge energy of available models of the APPT-based EPS (points corresponding to APPT-155 and APPT-95 are shown in figure 10) is substantially higher than the optimum in terms of obtaining the minimum mass of propulsion system. However, according to the experience of RIAME MAI in the APPT development and testing, a decrease in the discharge energy is associated with an increase in the thrust cost.

According to the results of analysis for the structure of total mass of the APPT-based EPS and for the possibilities of its reduction, it can be concluded that the discharge energy in the range from 40 to 80 Joules is close to the optimum in terms of the minimum total mass of the EPS with the storage density of the available capacitors ranging from 28 J/kg to 35 J/kg and while maintaining acceptable specific performance of the thruster. With further improvement in the mass-dimensional characteristics of the capacitors, the optimum of the discharge energy will shift toward higher values with a corresponding increase in the specific impulse and reduction in the thrust cost.

8. APPT-350 experimental model

An experimental model of the ablative pulsed plasma thruster APPT-350 was developed by RIAME MAI in 2020 for maintaining and correcting the orbit of a small SC intended for the remote Earth sensing (figures 11, 12). The new thruster differs from the earlier model APPT-250 of the same level in thrust and power consumption in the following:

- modular design that allows separate assembly, adjustment and testing for the thruster units and the energy storage unit;
- energy content of the energy storage capacitor bank close to the optimum for a given thrust level;
- characteristics of the discharge circuit providing better local-temporal matching of the plasma-forming substance energy and mass input into the discharge channels.

The above features allowed reduction for the total mass of the thruster with the propellant store from 7.2 kg down to 6.7 kg while keeping specific impulse of 12000 m/s and total thrust pulse of 15.6 kN·s.

Figure 11. The ablative pulsed plasma thruster APPT-350.

Figure 12. The APPT-350 breadboard model.
The APPT-350 breadboard model testing at the test facility IU-1 verified the following thruster performance:

- discharge energy 50 J;
- average thrust 1.2…2.4 mN with the power consumption of 50…100 W;
- specific impulse 12000 m/s;
- total thrust pulse 15.6 kN·s with the propellant store of 1.3 kg;
- total EP mass, propellant store including 6.7 kg.

9. Conclusion
Within the frames of works on development of low-power EP based on ablative pulsed plasma thrusters for small remote sensing satellites, the following works were carried out:

- preliminary analysis for the purpose and fields of application of APPT was carried out, the design appearance of a thruster for ERS SSC was identified, and the main directions of APPT further development were defined;
- methods for calculating the APPT propulsion characteristics were developed;
- the breadboard model of APPT for ERS SSC was developed and made (APPT-350);
- research tests of the breadboard model of APPT for ERS SSC were carried out.

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