Ion-Plasma Thrusters Thermal Balance Estimation

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***Abstract*—One of the main problem of the outer space exploration is the low specific impulse of spacecraft thrusters. The main limit of the development of high power and high specific impulse ion plasma thrusters is their destruction due to overheating. In this article was developed thruster losses and efficiency estimation method, helpful to determine the need in refrigeration system and main parameters of thruster thermal balance model.**

***Keywords*—Ion-plasma thrusters, thruster electric power looses, thruster heat looses, efficiency, thrust, specific impulse**

I. Introduction

In recent 20 years the outer space exploration has significantly decelerates. As the main obstacle of wide outer space exploration should be indicated a very long time of space expeditions, which main reason is non-solved problem of small specific impulse of spacecraft thrusters with high and medium thrust.

It seemed, that solution of specific impulse increasing exists, it's change of traditional chemical thrusters to electric thrusters (ion-plasma thrusters, hereinafter referred as IPT) with high specific impulse, because modern IPT have specific impulse up to 20 times more than chemical thrusters, but such IPT have very small thrust, mostly in millinewton range.

If spacecraft designers trying to increase the IPT thrust, that means to increase the thruster power, they encounter two main problems: need of spacecraft high potent power supply (nuclear, thermal-nuclear, solar etc.) and problem of IPT structure destruction.

In short, the topic of this article is to search a way to prevent the destruction of IPT structure.

II. Loses and IPT Destruction

IPT is a converter of electrical energy into kinetic energy of plasma stream.

The main energy characteristics of all rocket thrusters are: traction force or thrust T and specific impulse I_{sp} as well as the kinetic power of the working fluid, also called the reactive power P_{jet}:

$$P_{jet} = ½·G·V_{ex}^2, \tag{1}$$

where G is the working fluid mass flow and V_{ex} is the working fluid exhaust velocity.

Kinetic power can also be expressed through thrust and specific impulse:

$$P_{jet} = ½·T·I_{sp}·g. \tag{2}$$

For IPT the ratio of the effective power, i.e. its reactive power P_{jet}, to the electric power P consumed by IPT is IPT total efficiency η:

$$\eta = P_{jet}/P. \tag{3}$$

Losses of energy conversion into electric propulsion is IPT power loss P_L, determined as:

$$P_L = (1-\eta)·P, \text{ or } P_L = P-P_{jet}. \tag{4}$$

All losses of energy conversion in IPT can be represented as:

- heat loss: heat loss in current conductors, plasma heat transfers to IPT structure, losses for heating of chaotically moving neutral atoms of plasma flow, and
- divergence of ion beam (which can be minimized as will described lower) and
- electro-erosive destruction of IPT structures.

The mechanism of electro-erosive destruction due to the interaction with plasma flow is quite similar with electrode destruction mechanism in magnetron vacuum deposition process. It presents in almost any IPT, but can be minimized due to constructive measures. So, if special conditions are created in a magnetron sputtering system for plasma capture of cathode particles and their transfer to the target and the rate of electro-erosion destruction determines the efficiency of the deposition process. Then in IPT, on the contrary, in the design process, special measures are taken to prevent plasma contact and/or knockout, ionization and ion capture of neutral atoms of the surface of the conductive parts of the electrode system, the walls of the ion source and the accelerator channel. Magnetic traps are used to hold plasma from contact with the walls, and, where this is not possible, coating or lining the metal surface with highly resistant ceramic materials, therefore, the transfer rate of the electrodes material in IPT is much lower than in magnetron sputtering plants.

Evidence of a significant reduction in the effect of the electro-erosive destruction mechanism with the right approach to IPT design, is that for many IPT with power of hundreds of watts, operation time is thousands and tens of thousands of hours, and the number of starts can also be measured in tens of thousands. Thus, NASA confirms continuous operation of the “NEXT” IPT for a failure of more than *51,000* hours, and the “Deep Space 1” spacecraft overcame more than *262* million (km) with the “NSTAR” IPT and its speed ΔV was increased more than *10* (km/s) due to IPT operation was [1].

Thus, when appropriate structural measures are taken (an appropriate margin of electrode material thickness, measures to prevent or minimize plasma contact with the IPT elements, etc.), the destruction of the electrode system due to electrical erosion can’t cause sudden fatal failure and low IPT life.

As for the energy distribution during magnetron sputtering, as a result, all the energy consumed from the magnetron's power source is converted into the heat of the target, the cathode (cooled by water) and the hood of the vacuum unit (cooled by water or air).

Based on the foregoing, it was concluded that **the main reason of the IPT elements destruction are heat losses**.

III. IPT Thermal Balance

To show the heat transfer mechanisms in IPT, consider the design of IPT, patented in the USA in 2015 [2], shown in *Fig. 1*.

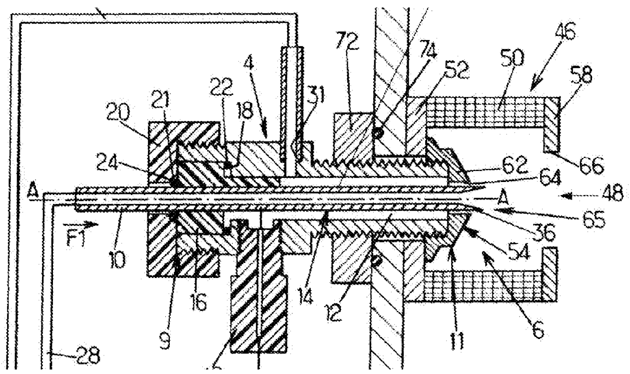

Fig.1. IPT design (patent US20150020502 A1).

In this IPT, the working fluid flows out from the nozzle 36, and a glow or arc discharge occurs in the discharge chamber 6 between the anode 66 and the cathode 62.

In contrast to magnetron sputtering, where all the energy from the power source is dissipated by the chamber, magnetron and target structures, in IPT part of the heat is removed by the flow of the working fluid (here flowing out through opening 48).

The remaining thermal energy released during electric discharge is absorbed by the structural elements of the electrode system and the discharge chamber (52, 62, 36, 50, 58). Overheating of these elements can lead to a loss of their strength and stiffness, the occurrence of spurious discharge and ultimately to their damage and destruction, that is, to IPT failure.

In order to determine the thermal power absorbed by the IPT structural elements, it is necessary to calculate the **heat balance of IPT, i.e. the condition of equality of the heat power input to the IPT structural elements and the heat output from them.**

Thermal power supplied to IPT structure can be determined by subtracting from the total amount of thermal power Q_t released during IPT operation (IPT emitted thermal power), the amount of thermal power allocated outside IPT with the working fluid flow Q_{wf}. Thus, the thermal power absorbed by IPT structure is determined. Since there are mechanical connections between the elements of its structure inside the thruster and the distance between them is relatively small, it means that the main mechanism of heat transfer in IPT is thermal conductivity.

Outer space vacuum is a highly efficient heat insulator. Thus, the processes of convection and heat conduction in outer space are absent and the only mechanism of heat transfer is the radiation.

In the temperature range up to *1000 (K)*, where the working areas of the main structural materials of IPT are located, thermal conductivity is a significantly more efficient heat transfer mechanism than radiation. Therefore, it can be assumed that the thermal power supplied to the IPT elements can be expressed integrally as absorbed thruster overall thermal power. Moreover, the thermal energy absorbed by IPT can be removed outside the thermodynamic system of IPT only by radiation: a heated IPT is a source of thermal radiation, and the power Q_r radiated by IPT is subtracted from the absorbed one.

The difference between the absorbed and radiated heat power determines the condition of IPT thermal (heat) balance:

$$Q_t - Q_{wf} = Q_r . \qquad (5)$$

The radiative heat transfer mechanism is described as a model of a radiating black body. In accordance with the Stefan-Boltzmann law, the total integral thermal power emitted by the black body in the entire spectral range is proportional to its area and the fourth degree of its temperature. Given the correction that a real IPT cannot absorb the entire thermal energy incident on it, because it is not a completely black body. It absorbs only part of it in proportional to its relative emissivity or degree of blackness, that is, the so-called gray body (not all the thermal energy incident on the IPT elements from the plasma stream is radiated), the thermal power Qr radiated by the IPT is:

$$Q_r = \delta \cdot \varepsilon \cdot S \cdot T_{ta}^{4}, \qquad (6)$$

where δ is the Stefan-Boltzmann constant $\approx 5.67 \cdot 10^{-8}$ $(W/(m^2K^4))$, ε is the relative emissivity or degree of blackness of IPT surface ≤*1*, S is the surface area of the IPT (m^2) and T_{ta} is the average temperature of the IPT surface (*K*),

$$T_{ta} = \sqrt[4]{\frac{Q_r}{\delta \cdot \varepsilon \cdot S}}\,. \qquad (7)$$

The average relative emissivity of the IPT surface is usually in the range *0.3 - 0.8* (e.g. copper wire in polyimide insulation $\varepsilon \approx 0.15$; aluminum vacuum ceramics $\varepsilon \approx 0.4 - 0.7$).

Thus, with an increase in the input electric power, a proportional increase in the thermal power released by IPT is occurred. In turn, this cause an increase in the thermal power emitted by the thruster; therefore, the temperature of IPT increase until the radiated heat power is in equilibrium with the absorbed thruster thermal power.

It should be noted that the temperature of micro thrusters with a thrust of several micronewtons, with a supplied electric power up to *500 (W)*, when it increases after turning IPT in operation, goes beyond the usual operating range of IPT structure materials, i.e., it does not create any negative effects. On the contrary, the discharge creates a slight heating of IPT, which is necessary to maintain it in the operating thermal regime corresponding to the regime of stable plasma combustion.

However, with an increase IPT power, its heat balance shifts toward an increase in radiated power and, consequently, in the temperature of the thruster structure, since the radiation surfaces of IPT cannot dissipate the supplied heat. IPT starts to overheat - the equilibrium temperature may be outside the limits of stability of IPT materials (insulation varnish of electromagnets, electrode material, it is possible to increase the temperature of magnets above the Curie point, etc.)

To prevent overheating of IPT, it is necessary to embed a thermal throttle into its control system, i.e. special controlled valve. This valve reduces the flow rate of the working fluid with increasing IPT temperature, as well as create an ion beam current control loop that reduces the beam current if the temperature of the structure is increased. That is, to artificially reduce thrust, specific impulse and power to prevent overheating and thruster failure. Such a need arises already for IPT with a consumed electric power of *660 (W)* (SPD-70) [3].

An alternative to artificial decreasing IPT performance is to create an IPT cooling system that allows excess heat output to be removed outside IPT and then dissipating it in outer space. Since overheating of IPT components can be prevented by efficient cooling.

IV. IPT Emitted Thermal Power

The thermal power, released during IPT operation, by its nature is the loss of the conversion of electrical energy into kinetic energy of an ion beam.

The most generalized approach to the formation of a loss model in IPT is proposed not from the point of view of describing the nature of the processes of these losses, but from the point of view of the resulting influence or the functional dependence of such losses on various discharge characteristics of IPT. This approach is expressed in the so-called utilization rates or the degree of utilization of certain resources used in the process of energy conversion.

A typical example of such approach proposed in [4] is illustrated in *Fig. 2*, and the overall IPT efficiency is equal to the product of resource utilization rates. The average values of the indicated resource utilization degrees for Hall-Effect Thrusters (hereinafter referred to as HET) are: for voltage utilization (accelerating voltage utilization factor) *>90%*, current utilization (discharge current utilization coefficient) *~70-80%*, divergence utilization (coefficient of divergence of ions velocity vectors) *>90%*, charge utilization (coefficient of utilization of ions charge) *>95%*, mass utilization (coefficient of utilization of the mass of the working fluid) *>90%*. Thus, the overall IPT efficiency (product of these coefficients) is *~50-70%* [4].

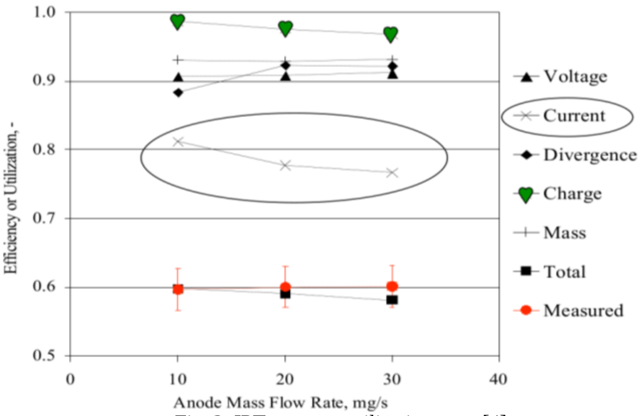

Fig. 2. IPT resource utilization rates [4].

Thus, the authors have assumed, that “primary efficiency loss is from current utilization i.e. electron backstreaming to the anode”, i.e. electron heating of the anode. Voltage utilization shows in fact electrons heating loses. Mass utilization shows the ratio between ionized and do not ionized atoms in IPT plasma. The neutral atoms are heating by plasms and they mostly transfer heat to the thruster structure. Divergence utilization shows the relarionship between number of ions, whose trajectory are parallel to the thruster axe and they took part in acceleration process, and number of ions that bombard the surface of IPT structure and heat it.

However, manufacturers' real IPT data also indicate lower values of total efficiencies (i.e. higher values of losses). For example, in some cases, the efficiency of well-developed serial “PPS-1350G” HET dropped to *39%* [5].

It should be noted that the **final result of all transformations of the energy loss in IPT is the creation of thermal energy**.

It is proposed to be guided by the assumption that the heat power released during the operation of IPT is the total power loss of the conversion of electric energy into kinetic energy of the ion beam P_L. The difference between P_L and Q_t is only the parasitic power of the part of the diverging ion flow, which do not fall on IPT structure, but whose trajectory is not strictly opposite to IPT thrust vector.

V. Thermal Power Reduced from IPT by Working Fluid Flow

The following approach is proposed to assess the value of thermal power diverted outside the thruster thermodynamic system by the flow of the working fluid.

The heat power diverted by the flow of the working fluid into the outer space, is equal to the rate of change of the thermal energy of the mass of the working fluid, and for the discrete mass of the working fluid emitted by IPT per unit time, it can be determined as:

$$Q_{wf} = C_{wf} \cdot G \cdot \Delta T_{wf}, \tag{8}$$

where C_{wf} is the specific heat of the working fluid in *(J/(kg·K)),* ΔT_{wf} – change in the average temperature of the working fluid when passing through IPT, *(K)*.

If the last 2 parameters - the average flow temperature and mass flow rate can be measured during experimental tests or specified when the IPT design is calculated, then the specific heat of the plasma is a certain parameter.

The heat capacity of the plasma is greater than the heat capacity of an ideal gas, since the energy supplied to the working fluid is spent in two ways: to change the kinetic energy of the ions of the working fluid and to change the average potential interaction energy between oppositely charged ions. Taking into account the electrical interaction between ions in a plasma is extremely complicated, which makes it possible to make only approximate calculations.

In the stationary mode of IPT operation the conversation is about specific heat, as heat capacity at constant pressure is C_p.

According [6] C_p of plasma is the combination of translational $C_p(T)$, rotational $C_p(R)$, vibrational $C_p(V)$, and electronics $C_p(E)$ energy modes. It is defined by:

$$C_p(T)= C_p(T)l+C_p(R)+C_p(V)+ C_p(E). \tag{9}$$

For this reason, the $C_p(T)$ function has several resonant peaks at various temperatures. The temperature dependence of C_p of the plasma of nitrogen, argon, air, helium, oxygen is shown in *Fig. 3*.

The heat capacity of inert gases (monatomic gases) under normal conditions (*20 (°C)* and *1 (bar)*) is *20.851 (kJ/(kmol·K))*, and the heat capacity of diatomic gases is close to *29.1 (kJ/(kmol·K)).*

Consider the worst and best cases, i.e. determination of the maximum and minimum limits of the value of the thermal power diverted outside the system by the flow of the working fluid.

Assume that the IPT operates in the temperature range of the first peak of the heat capacity amplitude (this assumption realistically reflects the operating temperature range of modern IPT).

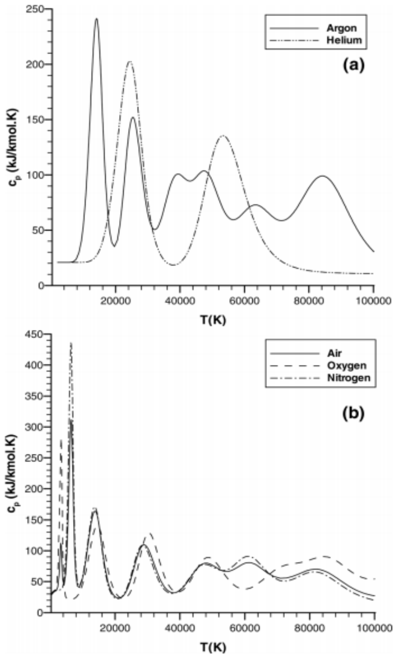

Fig. 3. The dependence of the heat capacity of the plasma [6].

The magnitude of the peak amplitude depends on the atomic mass and increases with its increase. The ratio of atomic masses of helium and argon is *9.9795*. The atomic mass ratio of argon and xenon is *3.2867*. Suppose that the ratio of atomic masses determines the ratio of the first peak amplitudes of heat capacity. Then the value of the first peak $C_p(T)$ for xenon is a maximum of the order of *270 (J/(mol·K))* or *2.0564 (J/(g·K)).*

According to manufacturers' data, “PPS 1350” HET with an average power consumption of *1350 (W)* consumes xenon in various operating modes from *2* to *7.5 (mg/s)* [5].

Different estimates of the average temperature of plasma ions, which determine the mass transfer of the HET plasma flow, are from *2500 (K)* [7] to *2900 (K)* or *0.22 (eV)* [8] and up to from *5454 (K) or 0.47 (eV)* for Helicon plasma (electrodeless) thruster [9] to *8703 (K) or 0.75 (eV)* [10].

Then the thermal power carried by the flow of the working fluid for the HET, calculated in accordance with (8), is from *9.5* to *42 (W)*, or from *0.7* to *3.1%* of the input power - these are estimates of the minimum and maximum possible values.

If it is IPT with an electrodeless radio frequency plasma source, then at similar costs, the thermal power carried out by the flow of the working fluid is from *21.4* to *131.1 (W)* or from *1.59* to *9.71%* of the power input, and it is also an estimate of the minimum and maximum possible values.

To calculate the thermal regime of the thruster and its cooling system, the IPT should be guided by a value close to the minimum value of the thermal power diverted by the flow of the working fluid — *0.7%* of the electric power supplied to the thruster. In this case, the error is equivalent to a decrease in the actual heat load on the cooling system compared to the calculated one, i.e. the cooling system has some power reserve (max. *9%*) and operate in a lighter mode. This reasoning also concerns the possible overstatement of the specific heat estimate of the working fluid.

From the above, it follows that: **there is a significant difference between chemical thrusters and IPT. It consists in differences of self-cooling mechanisms. In a chemical thruster, heat removal from structural elements is effectively carried out due to heat transfer by mass flow of the working fluid with the great mass flow rate (*kg/s)* and it is sufficiently effective. Then in IPT, due to the small mass flow rate (*mg/s)* in it, cooling due to heat removal by the working fluid is not effective and removes no more than *10%* of the generated heat.**

VI. Asessment of the Need to Create a Cooling System for IPT

The criterion for the need to create a cooling system for IPT is the risk of overheating. In this regard, at the initial stage of its development (when the basic characteristics of the electrode and magneto-optical systems are determined), it is necessary to assess the danger of possible overheating of structures.

The following method is proposed for a preliminary assessment of IPT overheating, that is, the need to create a cooling system and determine the maximum thermal load power:

- According to the thrust and specific impulse of the thruster in accordance with expression (2), the reactive power P_{jet} is determined;
- Based on the calculated or experimentally obtained value of electric power, in accordance with expression (3), the total IPT efficiency is determined and in accordance with expression (4) the total power loss P_L is determined;
- In accordance with expression (8), the value of the heat power Q_{wf}, diverted by the working fluid flow into the outer space, is determined or this value is taken equal to *0.7%* of the IPT electric power in accordance with the recommendation of Section V;
- The maximum value of the thermal power released during the thruster operation Q_t is taken equal to P_L;
- From the thermal balance condition (5), the power Q_r radiated by IPT is determined;
- Based on the obtained value of Q_r, and on the geometric and structural characteristics of the thruster, its average temperature is determined in accordance with (7).

If the obtained average temperature value is outside the working range of any elements of the electric propulsion system, then **there is a need to create a cooling system for IPT**.

In this case, based on the conditions for entering the calculated IPT temperature in the middle of the operating range, the amount of excess thermal power is determined, which should be removed outside the IPT thermodynamic system using a cooling system.

The thermal power value thus determined is the average estimate load of the IPT cooling system.

The maximum estimate of the thermal load of the cooling system is defined as the amount of excess thermal power using the lower IPT operating temperature. The obtained value can be taken as a possible estimate of the load power of the IPT cooling system, since all the assumptions in the evaluation of the variables included in the expression of this quantity were initially made in the direction of increasing the final value.

When designing a cooling system, it is recommended to be guided by the maximum estimate load power for the following reasons. During the IPT operation, the surface of the elements of the electrode system and the plasma path is destroyed due to the electroerosive effect of plasma. This effect is expressed in an increase in roughness and the appearance of friability of the surfaces of parts, which leads to a decrease the ε (relative emissivity) in (6).

At the same time, the degradation of the outer surfaces of IPT is much slower than the degradation of elements directly interacting with the plasma flow, i.e. an increase in the relative emissivity of the thruster as a radiator of thermal energy is much slower than an increase in its relative emissivity as a receiver of thermal energy.

Thus, during the operation of the IPT, over time, due to the degradation of its elements interacting with the plasma, the heat balance shifts toward an increase in the thermal power absorbed by the IPT and an increase in the thruster temperature – that means that the **thermal throttle and thermal control system will automatically decrease the thruster power**.

In this regard, it seems correct to accept the resulting estimate of maximum power as the calculated value of the load of the cooling system. There are two examples.

IPT with a thrust of *1 (N)* and specific impulse of *6000 (s)* in accordance with (2) develops the reactive power of *60 (kW)* with the total efficiency of *0.5*, thus the electric power consumed by this thruster is *120 (kW)* and the total power loss is *60 (kW)*.

Even assuming that the amount of heat power removed by the flow of the working fluid, Q_{wb} is not accepted in *0.7%* of the electric power in accordance with the recommendation of Section V, and it is evaluated in maximal rate *10%.* Thus only *90%* of the total power loss is the thermal power absorbed by IPT. In this case the absorbed power and, therefore, the power that must be radiated by the IPT surface into outer space is *54 (kW)*.

Assume that the maximum permissible long-term operating temperature of the thruster surface is *500(K)* (this is a realistic value taking into account the presence of sensors, coils of electromagnets, polymer insulating materials, etc.) and assume that the average value of the thruster relative emissivity is *0.5*.

In this case, in accordance with (6), *1* (m^2) of the IPT surface scatters *1772 (W)* into outer space, and for the complete "utilization" of the heat supplied the radiation surface of the thruster need to be at least *30.5* (m^2) area. In this case, the thruster should be in black space - outside the influence of the thermal radiation fluxes of the Sun and the Earth, as well as the spacecraft itself, creating an additional influx of thermal power to the radiating surface.

It should be noted that the surface area of the real IPT "DAS-200" with an electric power of *35 (kW)* is about *0.5* (m^2) [11]. This thruster does not have a cooling system and is cooled only by radiation. If the conditions of the above example are transferred to this IPT, then its steady-state temperature is about *1026(K)* or *753 (°C)*. This value is the **average** temperature of the thruster and not the temperature of its most heated elements (discharge and accelerator channels, electrode system). However, the temperature value obtained is already outside the working range of the insulating materials and above the Curie point of the elements of the magnetic system. Another problem is to ensure the strength and stiffness of loaded aluminum and copper electrodes of the electrode system and other electric propulsion elements. These facts explain that this IPT was not accepted for mass production and was not finalized, because it apparently, had a very small resource and a small number of repeated starts, despite the high traction characteristics.

Estimate the ultimate electric power of a real IPT, which has a thermal regime within the limits of the structurally permissible temperature *500 (K)*. Then, with a total thruster surface area of about *0.4* (m^2) and an average relative emissivity of *0.5* with a total efficiency of *0.5* and the assumption that only *90%* of the thermal power allocated to IPT is absorbed and re-emitted by it, thus the electric power is about *1575 (W)*. This is not the maximum power peak pulsed, but the average one. This value is in good agreement with the data of existing IPT designs, for example the "PPS 1350" Safran Snecma IPT [12].

Conclusion: as a rule, for an IPT with an electric power of more than *1.5 (kW)*, either artificial limitation of thruster power with increasing temperature or a cooling system device is necessary. Since the power limited option is not an acceptable solution, there is a need to create an IPT cooling system. The load power of the cooling system can be determined by the method proposed above. Theoretical values of specific impulse, thrust and electric power can be determined both experimentally and based on modeling of energy conversion processes into electric propulsion, which is discussed in the next section.

VII. IPT Thrust Processes Description

The IPT kinetic power generation process is described in detail in [13] - [16]. Here, a brief analysis is given from the point of view of calculating losses and the value of the thermal power released during the thruster operation Q_t evaluation is estimated.

Particles acceleration in IPT is the result of electrostatic forces. It affects only on charged particles; therefore, the neutral atoms and ions velocities in IPT output plasma differ by several orders. Consequently, neutral atoms play a low role or play no role in creating IPT thrust.

Taking into consideration that the IPT working fluid consist of particles and that IPT thrust T is determined by the accelerated ions outflow velocity V_{exi} and their mass flow rate G_i, equal to product of the ion mass m_i and its flow n_i (number of ions per second), it is possible to obtain the following expression:

$$T = -V_{exi} \cdot (m_i \cdot n_i). \tag{10}$$

The kinetic power of the particle beam is expressed as:

$$P_{jet} = \frac{T^2}{2 \cdot m_i \cdot n_i}. \tag{11}$$

That means that to increase the thrust without increasing of the mass flow rate, it is necessary to increase the reactive power of the thruster.

The energy of the electrostatic field E, which accelerates the working fluid ions, is equals to the product of the working fluid ion charge q_i and the potential difference V_b of the thruster accelerator electric field. After conversion to the kinetic energy of the ion, in accordance with the law of energy conservation, it is equal to the kinetic energy of the ion:

$$E = V_b \cdot q_i = ½ \cdot m_i \cdot v_{exi}^2. \tag{12}$$

From (12), the ion flow velocity is expressed as:

$$V_{exi} = \sqrt{\frac{2 \cdot V_b \cdot q_i}{m_i}}. \tag{13}$$

The ion beam current I_b, is equal to the product of the ion charge q_i and the number of ions n_i, which can be expressed through the mass flow of ions G_i:

$$I_b = q_i \cdot G_i / m_i \tag{14}$$

It means that the mass flow of ions can be expressed via ion beam current:

$$G_i = I_b \cdot m_i / q_i \tag{15}$$

From (10), which determines the IPT thrust, by substituting (13) and (14), taking into account that the ions in IPT have mostly a single unit degree of ionization, this means that the unit charge is equal to the elementary charge *e*, the following expression is obtained for IPT thrust (in Newtons):

$$T = \sqrt{\frac{2 \cdot m_i}{e}} \cdot I_b \cdot \sqrt{V_b}. \tag{16}$$

This form is convenient in that the first factor of expression (16) is a constant for each type of working fluid. The value of this constant for different working fluids are: for hydrogen = $1{,}44537 \cdot 10^{-4}$; for helium = $2{,}88042 \cdot 10^{-4}$; for neon = $6{,}467598 \cdot 10^{-4}$; for argon = $9{,}09981 \cdot 10^{-4}$; for krypton = $1{,}31796 \cdot 10^{-3}$; for xenon = $1{,}64968 \cdot 10^{-3}$; for air = $7{,}75058 \cdot 10^{-4}$. That is, the thrust of xenon IPT (in millinewtons) becomes:

$$T = 1{,}65 \cdot I_b \cdot \sqrt{V_b}. \tag{17}$$

Obviously, thrust increases with increasing atomic mass of the working fluid.

Expression (16) describes the **ideal model** of the acceleration process in IPT, since it does not take into account such factors:

- the real ion beam has a divergence, not all ions move along trajectories parallel to the thrust vector;
- in fact, the degree of ionization has a probability distribution;
- thermal effects must be taken into account. Some of them are positive for thrust and increase the kinetic energy of ions due to electronic heating. Some are negative ones, for example, ion beam scattering in collisions with neutral atoms in plasma flow of the thruster.

In general, expression (16) for real IPT can be represented as:

$$T = \gamma \cdot \sqrt{\frac{2 \cdot m_i}{e}} \cdot I_b \cdot \sqrt{V_b}, \quad (18)$$

where γ is a coefficient that take into account all the negative effects: ion divergence, ionization degree and thermal effects. For the ideal thruster $\gamma = 1$.

$$\gamma = \alpha \cdot \beta \cdot \delta, \quad (19)$$

where α is the ion beam divergence coefficient, β is the ionization degree coefficient and δ is the dissipation coefficient (taking into account scattering of the ion beam in collisions with neutral atoms etc.) *and* $lim(\alpha)= lim(\beta)=lim(\delta)=lim(\gamma)=1$.

Sometimes these coefficients are easy to calculate. For the ion beam with uniform conical divergence after exiting the IPT, constant current density, and a uniform electric field, the correction coefficient α is equal to the cosine of half the average beam divergence angle θ or the cosine of the angle θ between the generatrix and the axis of symmetry of the ion beam. Then:

$$T = \cos\theta \cdot \sqrt{\frac{2 \cdot m_i}{e}} \cdot I_b \cdot \sqrt{V_b}, \quad (20)$$

For instance, if the beam divergence is *60º* then θ = *30º* and $\cos(\theta)$ = *0,866,* which means that the IPT thrust loss due to the beam divergence is *13,4%.*

If the plasma source is not uniform, that is, the conditions described above with respect to the current density, field and beam geometry are violated, and then the correction coefficients must be integrated over these surfaces taking into account their curvature.

For cylindrical IPT:

$$\alpha = \frac{1}{Ib} \int_0^r 2 \cdot \pi \cdot J(r) \cdot \cos\theta(r)\, dr, \quad (21)$$

where $J(r)$ is the function of the current density depending on the radius. For $J(r) = const$, the expression (21) is transformed into (20).

The ion current distribution can be measured during the IPT test using Langmuir probes.

The correction for the ionization coefficient β takes into account the presence of multiply charged ions in the IPT plasma. If the IPT plasma contains simultaneously singly charged, doubly charged and triply charged ions, then the total beam current is assumed from the current of single I^+, double I^{++} and triple I^{+++} charged ions currents:

$$Ib = I^+ + I^{++} + I^{+++}, \quad (22)$$

and an ionization degree coefficient is:

$$\beta = 1 + \left(\frac{1}{\sqrt{2}} \cdot I^{++}\right)/I^+ + \left(\frac{1}{\sqrt{3}} I^{+++}\right)/I^+. \quad (23)$$

Practically, it is possible to estimate the number of single, double and triple charged ions quantity during thruster testing using a magnet sectoral charge analyzer, based on the effect of the Lorenz force and the dependence of the particle's Larmor radius on the particle's charge:

$$R_L = \frac{m_i \cdot v_{ipB}}{q_i \cdot B}, \quad (24)$$

where V_{ipB} is a part of the ion velocity perpendicular to the magnetic field of the analyzer magnets and B is their magnetic field. The number of charged ions I^+, I^{++} and I^{+++} in the working fluid is measured by *3* Faraday caps located in positions, appropriate to the ions radius.

The dissipative coefficient δ is determined mostly by two factors: - the ratio of neutral atoms to ionized atoms and the inhomogeneity of the magnetic field, electric field and plasma pressure. The development of reliable mathematical model for δ is a problem of great difficulty. A reliable model should describe the interaction of inhomogeneity in the plasma flow resulting from mechanical disturbances in the plasma source and the accelerating channel, perturbations of the magnetic and electric fields caused by inhomogeneity and differences in the real distribution of their intensity from laid down in the thruster. Such models are created for commercially available IPTs based on empirical data obtained during engine tests.

Known the scientific discovery of J. V. Kubarev, "The nature of the appearance of electrostatic instability of a plasma moving in inhomogeneous electric and magnetic fields" [17], which describe the theoretical principle of plasma acceleration optimization (equality of the relative gradients of pressure P and magnetic field H and the number of electrons n_e):

$$\frac{\nabla P}{P} = \frac{\nabla H}{H} = \frac{\nabla n_e}{n_e} \quad (25)$$

Based on the results of engine tests, δ can be described as a linear function of the number of ions on the number of rested neutral atoms:

$$\delta = k_1 \cdot n_i/n_n + k_2, \quad (26)$$

where k_1 is the empiric coefficient, taking into account the thermal interaction of neutral and ionized atoms and k_2 is the coefficient of the inhomogeneity effect.

In fact, this empirical method seems rather complicated, but there is another way.

VIII. Evaluation of the IPT Efficiency

Due to (16) an ideal IPT specific impulse:

$$I_{sp} = \frac{V_{exi} \cdot G_i}{g \cdot G}, \quad (27)$$

where g is gravitational constant.

The ratio of the ions mass flow per second to the mass flow per second of the working fluid, called the **index of the efficiency of the working fluid mass flow usage η_m**, in the case of a singly charged ions flow can be described by the following expression:

$$\eta_m = \frac{I_b \cdot m_i}{e \cdot G}. \quad (28)$$

In the case of a real IPT, taking into account all real thruster effects described above, with allowance for the correction coefficient γ expressed in (18), (19), (27), the expression for the mass utilization efficiency index (28), the expression for the ion velocity (13), the expression for the specific impulse is as follows:

$$I_{sp} = \frac{\gamma \cdot \eta_m}{g} \cdot \sqrt{\frac{2 \cdot e \cdot V_b}{m_i}}. \quad (29)$$

Substituting in (29) all the determined values, obtaining the value of the single conversion factor (in SI units):

$$I_{sp} = 1416{,}5288 \cdot \gamma \cdot \eta_m \cdot \sqrt{\frac{V_b}{M_a}}, \quad (30)$$

where M_a is the mass of the working fluid ion in atomic mass units.

Then the value (30) for the corresponding working fluid is equal to: for hydrogen $I_{sp}=1411{,}0065 \cdot \gamma \cdot \eta_m \cdot \sqrt{V_b}$; for helium $I_{sp}=708.0332 \cdot \gamma \cdot \eta_m \cdot \sqrt{V_b}$; for neon $I_{sp}=315.3311 \cdot \gamma \cdot \eta_m \cdot \sqrt{V_b}$; for argon $I_{sp}=224{,}11797 \cdot \gamma \cdot \eta_m \cdot \sqrt{V_b}$; for krypton $I_{sp}=154.7417 \cdot \gamma \cdot \eta_m \cdot \sqrt{V_b}$; for xenon $I_{sp}= 123.6258 \cdot \gamma \cdot \eta_m \cdot \sqrt{V_b}$; for air $I_{sp} = 263{,}1332 \cdot \gamma \cdot \eta_m \cdot \sqrt{V_b}$.

Thus, the rate of exhaust velocity (and, correspondingly, for specific impulse) increases with a decrease in the atomic mass of the working fluid..

An important indicator of IPT perfection is the **thrust to power ratio** ψ, which is the ratio of the thrust to the electric power P consumed by IPT:

$$\psi = T / P. \quad (31)$$

An indicator of the efficiency of the IPT electrical power usage is the **electric efficiency of the thruster η_e**, defined as the ratio of the ion beam power P_b to the electric power consumed by the thruster P:

$$\eta_e = \frac{V_b \cdot I_b}{V_b \cdot I_b + P_L}, \quad (32)$$

The overall efficiency η of IPT is defined as the ratio of the reactive power P_{jet} of IPT (11) to the electric power consumed by the thrust P_r:

$$\eta = \frac{T^2}{2 \cdot G \cdot P}. \quad (33)$$

It is easy to see, that all the needs for (33) parameters can be really determined and measured with sufficient accuracy during IPT tests (thrust, mass flow and electrical power consumption).

So, (33) is very important for the design and development of IPT, because IPT overall efficiency of IPT can be precisely determined experimentally.

But the measurement of these parameters makes it possible to evaluate such an important and difficult to determine parameter as a coefficient γ, and to evaluate the real difference between the total power loss P_L and the value of the thermal power Qt released during the thruster operation to refine the calculations.

IX. Conclusions

Based on the analysis, the following conclusions and recommendations were identified:

1. The main element of losses that cause the destruction of the structural elements of IPT is heat loss;

2. There is a difference between chemical thrusters in which the effective heat removal from structural elements is due to the large mass of the working fluid, and IPT, in which is a small mass flow rate and, therefore, previous concept of cooling mechanism has low inefficiency and removes no more than 10% of the heat generated;

3. An alternative to artificial reduction of thrust and specific impulse of IPT necessary to prevent thruster overheating, is refusing to artificially impair thruster performance, and creating IPT cooling system;

4. For IPT with an electric power of more than *1.5 (kW)*, as a rule, there is a need for artificial limitation of kinetic power, (T and I_{sp}) to limit the temperature increase or it is necessary to develop cooling systems.

In the article proposed a method for calculating the heat balance of IPT and the main input data the calculation of the IPT cooling system.

The proposed approach helps to create a mathematical model of the thruster heating and cooling processes and perspectives of further development of high power IPT for long space expeditions.

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