

FLIGHT TESTING OF VOLUME-IONIZATION ION THRUSTERS ON METEOR SATELLITE

Vladimir P. Khodnenko

NPP Russian scientific and Research Institute of Electromechanics
Moscow 101000, P.O. Box 496, Russia
Phone: (095) 924-94-98, fax (095) 207-49-62, E-mail: VNIEM@orc.ru

Flight testing of volume-ionization ion thrusters (VIIT) on Meteor satellite had been conducted in February 1972. Electric propulsion system (EPS) "Zefir" was jointly developed by Kurchatov AEI, VNIEM and EDB "Zaria". EPS "Zefir" consisted of two ion engines, propellant (mercury) and a power supply system comprising two high voltage rectifiers and a converter. Parameters of the EPS are as follows: power consumption (500-550)W, thrust (6-8) mW, lifetime 100h, ion effluent speed $\sim 5 \cdot 10^4$ m/s, ion current (55-75) mA, acceleration voltage $+2,7^{+0.2}_{-0.3}$ kV and mass 54 kg. EPS used 5 commands from Earth for control and 13 tele-metric channels. The first stage of flight testing qualified electric thrusters for space application, their effect on radio link with satellite as well as their compatibility with on-board instrumentation. The second stage of flight testing 102 min long allowed EPS thrust to be measured and it was found to be (5,0-7,1) mN and close to the expected value of 7 mN. Other parameters of ion thrusters the EPS as a whole compared favorably with the ground testing results. Endurance tests have not been completed due to partial malfunction of thruster (discharges in the acceleration system). In summary, flight-testing allowed us to get valuable information to be used as design guidelines for more advanced EPS with ion thrusters.

INTRODUCTION

Flight testing of ion engines on Meteor satellite took place simultaneously with Kaufman ion thrusters on SERT-2 spacecraft in 1970-1972 [1].

These thrusters were rather different. In Kaufman ion thrusters the acceleration of ions proceeds along the discharge axis while in Zefir type thrusters across the discharge line.

Both thrusters employed mercury as a propellant to be ionized in volume. The specific feature of these thrusters is discharge and thrust axes interrelation. Ions are accelerated in the ion-optical system along the discharge axis in the ionization chamber (Kaufman type). Ion acceleration takes place transverse to the discharge ("Zefir" type). In first case ion-optical system had grid configuration and in the second – slit type.

Flight missions in both cases were identical: qualification of ion engines for space conditions, their compatibility with spacecraft systems, determination of operational characteristics in space conditions and verification of the ground based conclusions.

Ion thruster

Ion thruster module (ITM) of operation is as follows (Fig. 1). Liquid mercury was forced by the spring device (4) into the porous gas generator being heated. The mercury vapor through the heated ceramic gas generator pores entered the gas generator chamber distribution cavity (1). From it vapor passed to the discharge chamber through holes. In this chamber a discharge between filament cathode (2) and anode (3) took place producing plasma. Plasma ions were setting uniformly on the chamber internal walls. Part of the ions that entered 4 output slits ($1,5 \times 70$ mm²) of the accelerating electrodes got additional energy in the electrical field.

The other part collected on the walls of the discharge chamber confirmed indirectly the ion generation efficiency and the propellant input rate. By measuring the ion current it is possible to stabilize the mercury vapor input by acting on the gas generator heater.

The focussing electrode (5) with negative potential relative to the chamber (1) and the frame (6) shields the chamber that

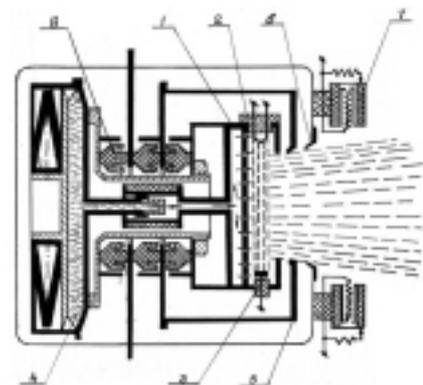


Figure 1: Ion thruster schematic

is under the high positive potential with respect to the frame because of the electron collection from the ion beam and cathode-neutralizer. Plasma neutralizer (7) generated the electron flux, which compensated the space charge of the accelerated ions. Neutralizer operated on the cathode thermoemission effect. The neutralizer propellant was chloride cesium contained in a vessel and evaporated during the independent stabilized heating.

High voltage insulator unit (8) distributed high voltage between chamber (1), focusing (5) and accelerating (6) electrodes.

“Zefir” EPS

“Zefir” EPS consisted of 5 units (Fig.2): two ion thruster modules ITM-1 and ITM-2, two blocks of high voltage rectifiers BHV-1 and BHV-2 and electroconverters and regulator unit (EAU)/

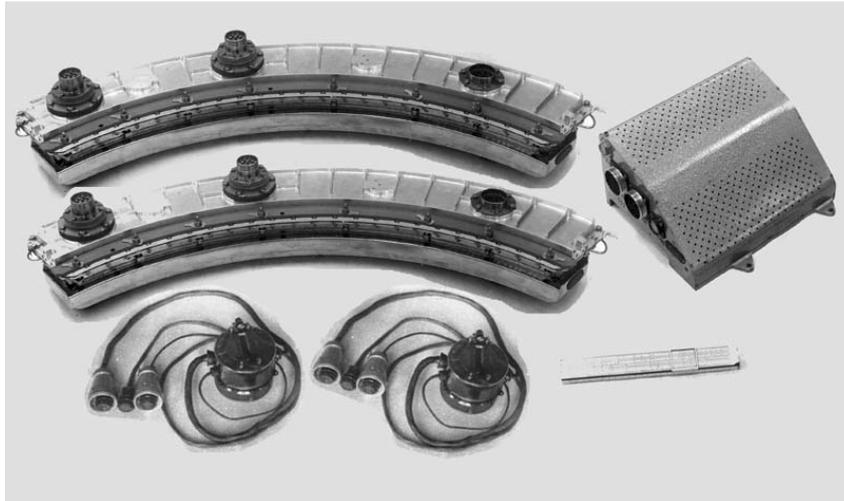


Figure 2: “Zefir” EPS

ITM-1 and ITM-2 thruster modules (Fig. 3 -1) were installed outside the Meteor spacecraft on the arms in XZ plane (Fig. 4) at a distance “R” of the thrust vector axis from the satellite mass center.

BHV-1 and BHV-2 high voltage rectifiers (Fig. 3 -2) were attached to the satellite bottom frame under the same own sealed enclosure with high voltage connectors. EAU electroconverters unit was located inside the satellite container. It comprises three power static converters, cathode filament voltage regulator, current regulator of the propellant evaporator heating, discharge current transducer, time relay, 14 telemetry converters, rectifiers and relay unit.

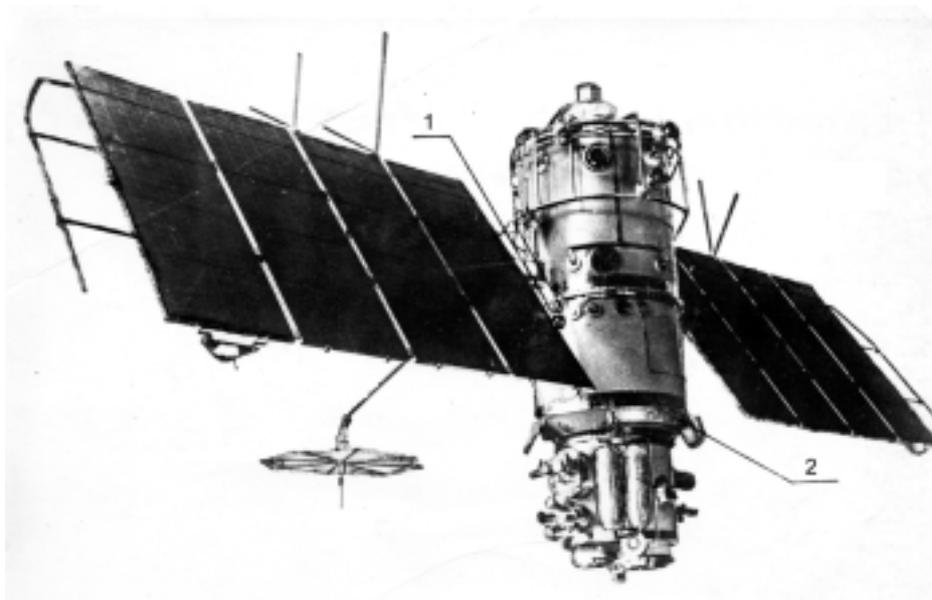


Figure 3: Meteor satellite with “Zefir” EPS

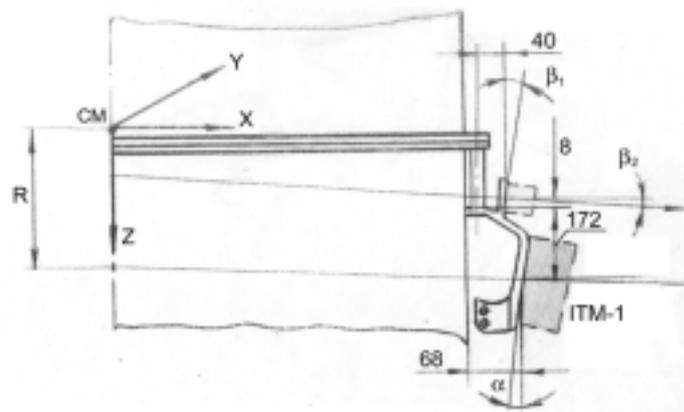


Figure 4: "Zefir" ITM-1 location on the Meteor satellite

The output parameters of the ITM power equipment are as follows:

- accelerating electrode	2700 $^{+200}_{-300}$ V;
- focusing electrode	1700 $^{+100}_{-200}$ V;
- discharge and starter heater	35 $^{+2}_{-3}$ V;
- bias voltage	15 $^{+2}_{-3}$ V;
- discharge chamber cathode heating	6,3 V; 1 kHz;
- neutralizer heating	4 V; 1 kHz;
- vaporizer heating	20 V; 1 kHz.

The main feature of the converters protection was their different responses to short-and long-term breakdowns in ITM. At the short-term breakdowns the protection reduced the voltage. During the long-term breakdowns, that were not eliminated by the voltage decreasing, the converters were switched off for 5...10 s and then switched on again for 1 s. If the breakdowns or short circuit has not been eliminated, EAU system would switch to the mode of cyclic sampling – 1 s of operation and 10 s of the off conduction. Such mode could be kept until the command to switch "Zefir" EPS "OFF". During the cathode circuitry shorting an automatic circuit breaker switched off EPS from the on-board mains system.

The time relay switched EPS off after 12 – 20 minutes of operation, if there were no commands from Earth. This relay could also be blocked by command from Earth.

"Zefir" EPS was controlled by the following commands:

1. ITM-1 "ON";
2. ITM-2 "ON";
3. Switching to stand by neutralizer;
4. Time relay blocked;
5. EPS "OFF".

Information on "Zefir" EPS operational data was as follows:

1. Accelerating electrode voltage U_a ;
2. Accelerating electrode current I_a ;
3. Focusing (intermediate) electrode current I_f ;
4. Discharge voltage U_p ;
5. Discharge current I_p ;
6. Bias current I_s ;
7. Neutralizer heating current I_n ;
8. Cathode heating current I_c ;
9. Gas generator heating current I_g ;
10. Neutralizer emission current I_e ;
11. Heating circuitry voltage U_h ;
12. Gas pressure in BHV-1
13. Gas pressure in BHV-2
14. Contact transducer of neutralizer

“Zefir” EPS ground tests

“Zefir” EPS prototype specimen has undergone all types of the mechanical thermal and vacuum as well as life tests in vacuum chambers. The EPS flight specimen was tested following the reduced run test program. Table 1 presents the “Zefir” EPS ground test results.

Table 1: The main “Zefir” EPS ground based parameters

Thrust, mN	6 – 8;
Power consumption, W	550
Full thrust impulse, kNs	$\geq 1,44$;
Number of firings	100;
Specific thrust, Ns/kg	5.10^4
Propellant consumption	
Hg through anode, mg/s	0,16;
Cs through neutralizer, mg/s	0,1;
Thrusters lifetime, hours	≥ 50 ;
Time to fire, s	270;
Propellants supply, kg	1,56;
EPS mass without propellant, kg	53,4.

The working fluid (mercury and chloride calcium) flow rate was estimated during the life tests on the basis of the ion current and the propellant efficiency coefficient under stationary conditions. Specific impulse was determined in terms of voltage U_a applied between discharge chamber and an accelerating electrode.

The ITM-1 and ITM-2 electrical parameters during nominal operating conditions were in good agreement with the values presented in Table 2.

Table 2

U_a , kV	I_a , mA	I_f , mA	U_p , V	I_p , A	I_n , A	I_c , A	P, W
2,7–2,6	65-70	3-4	34	2-2,5	6,5-6,8	6,5-7	507

“ZEFIR” EPS nests in space

Three stage “Zefir” EPS tests were conducted in space.

At first stage series of firings confirmed for the transmission of the both thrusters “ON” commands as well as commands on switching from the primary neutralizer to stand by one. Also the integrity of all electric circuits was checked, the absence of the short circuit and leakage currents in the accelerating and intermediate electrode circuits of and the mechanical damage was confirmed. The thruster power consumption from the on-board mains at heating-up condition was 316 W (current – 10,5 A at 30,1 V) and was the same as in the ground tests.

During the first 12 minutes of operation of ITM-1 the ion beam current I_a was recorded beginning from the 4th min while from 7 to 12 min I_a value was within 78...72 mA and corresponded to the maximum ground tests values. Thus, ITM-1 was started in 4 minutes after switching on. The accelerating voltage U_a reduction from 3700 V to 2600-2700 V after the load fully developed corresponded to the rectifier volt-ampere characteristic.

Since the 2nd minute neutralizer filament current had nominal value typical for the operating conditions: $I_n = 6,7$ A. The neutralizer emission current I_c could be measured from 9th min. to the communication session end. Its value (80...75 mA) was only slightly different from I_a . The focusing electrode current I_f was about 5% of I_a and found to be was consistent with values of the thruster ground test.

Table 3 gives the main thruster parameters for 9th min of its performance.

Table 3

U_a , kV	I_a , mA	I_f , mA	U_p , V	I_p , A	I_c , A	I_n , A
2,6	78	4,2	34	2,2	80	6,7

At the next session the large volume of the telemetry information allowed one to follow the details of the ITM-2 transition into the operating mode.

Close to the session end the bias current was close to the specified value of 0,65...0,7 A. Discharge current I_p changed respectively decreasing in time and approaching 3...3,2 A staying within the norm but exceeding the optimum value. Ion beam current I_a variations were attenuated approaching the nominal value of 65 mA. The current I_f varied over the wide range approaching 4 mA at the end of the communication session. The neutralizer current I_n reached the steady state value 7,1...7,2 A that was greater than the nominal value by 0,4...0,5 A fixed at the ground thruster adjustment. During the first 11 minutes of the thruster operation current I_e varied from 100 mA and more (overcompensation regime) and at maximum filament heating – to 80...95 mA.

The accelerating voltage during the thruster operation was by 15 % less than the nominal one. Due to longer than expected thruster operation it became possible to estimate the thrust based on the pitch wheel kinetic moment (K_c) changing $F = K_c/R\Delta t$ where Δt is thrust duration.

Values calculated on the basis of the telemetry data and specifying the EPS operation for several communication session are summarized in Table 4 and are follows: ion beam power P_a ; lost power P_i ; total power P_Σ ; power P calculated using electric conversion system efficiency; engine thrust determined from the equation $F_a = 0,208I_a\sqrt{U_a}$; thrust F defined by the pitch wheel kinetic momentum K_c and thrust quality γ (ratio of power to the thrust).

The bottom line of Table 4 gives the expected thruster parameters corresponding to those obtained at ground tests. It should be noted that ITM-1 parameters are close to specifications. Efficiency appeared to be rather low be cause of the reduced accelerating voltage.

Thus, the goals of the “Zefir” EPS 1st stage tests in space were fully accomplished. EPS seemed to be operative in space environment and its parameters were close to calculate and those obtained during the ground tests. No interference caused by the ion thruster operation with other satellite systems was observed.

2nd-stage test

ITM-2 was switched on for one revolution about the Earth (102 min). On 9th min ion beam current I_a has achieved 70 mA and up to 34th min. did not significantly change. From 9th min to 20th min focusing electrode current I_f has achieved 20 mA; and then up to 34th min gradually decreased (Fig. 5).

Table 4

Thruster	P_a	P_i	P_Σ	P	F_a	F	γ
	W	W	W	W	mN	mN	W/mN
ITM-1	202	185	387	516	8,3	-	62,1
ITM-2	149	202	351	469	6,4	7	73,4
ITM-2	156	211	367	490	6,8	7,1	72,1
ITM-2	154	224	378	504	6,5	5,0	77,6
Expected parameters ITM	189	191	380	507	7,6	-	66,7

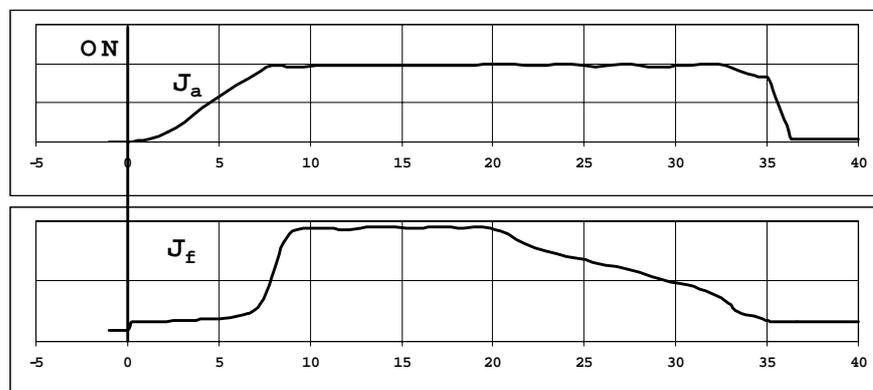


Figure 5: I_a and I_f current telemetry data at ITM-2 operation

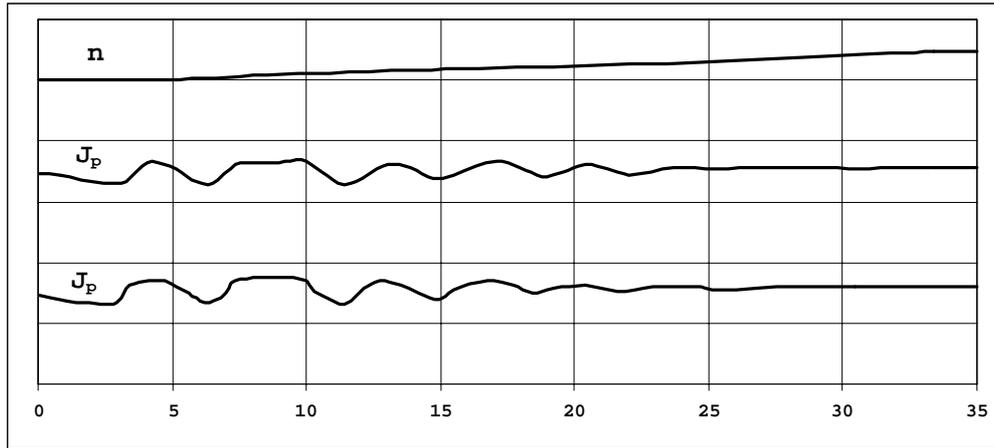


Figure 6: Telemetry of the wheel rotation rate n and current I_p at ITM-2 operation

By 24th min discharge current I_p was set at 3,4...3,5 A level (Fig. 6.) that exceeded the optimum value by 25 %. Discharge voltage was lower than the rated value according to the rectifier volt-ampere characteristic.

The neutralizer filament current I_n gradually increased at the beginning of the session from 6,4...6,7 A to 7,3...7,5 A after 25th min, in this time interval I_c current increased from 8,8 A to 9,9...10,1 A.

Soon after the thruster firing the ion beam overcompensation has occurred – the neutralizer emission current exceeded 100 mA at I_a about 70 mA. Overcompensation seems to be related to the large filament current of the neutralizer.

ITM-2 parameters with stand-by neutralizer were found to be close to the rated ones (Fig. 7.). Final stabilization of the operating regime did not occur until 35th min in the session.

During the present and further experiments the thrust was measured by means of the attitude control system. The external disturbances acting on the satellite along the pitch axis were estimated before the EPS switching on. These moments assessed by the pitch wheel revolution number changing appeared to be small, i.e. less than $1 \cdot 10^{-4}$ Nm.

The thrust calculated mean value of the thrust is given in Table 4.

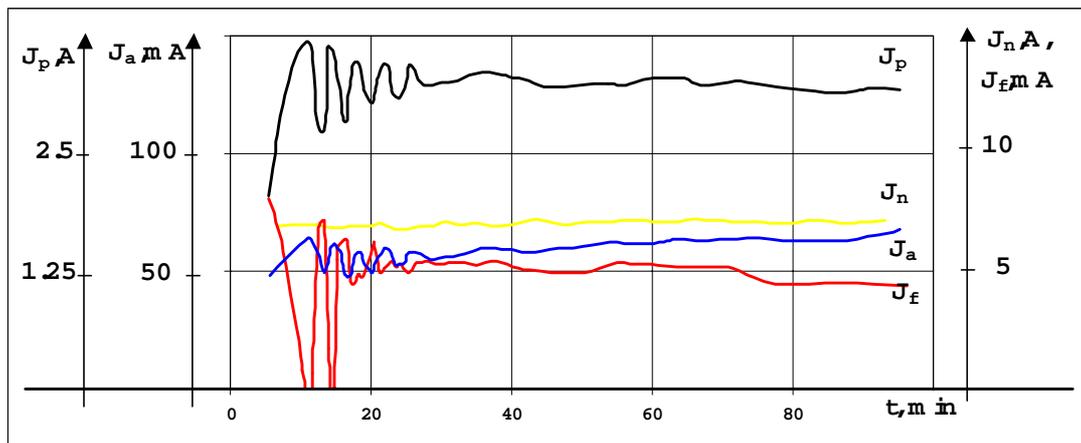


Figure 7: Time dependence of ITM-2 currents

During operation the number of the ITM-2 wheel (n) revolutions has increased from 28 rev/min to 220 rev/min (8th – 34th min) and thrust of 7,1 mN derived from these data was close to the designed values. (3rd line in Table 4).

The Table 4 presents values for the 30th min when the ITM-2 thruster operating conditions finally stabilized.

The 4th line in Table 4 gives the thruster parameters calculated on the basis of the 70th and 90th minutes data when the conditions were stable.

Thus, the goals of the 2nd-stage tests (for two month) have also been fulfilled.

The measured values of “Zefir” EPS thrust are close to the rated ones. Other propulsion module parameters are close to those obtained during the ground test.

3th-stage test

In six months time after two months testing there appeared favorable conditions for thruster switching on the illuminated orbit. In this period the Meteor spacecraft had the Earth’s shadow for more than 2500 times due to the orbit precession.

ITM-1 and ITM-2 were subjected to the profound thermal cycling over 250K-330K range. It had an effect on the valueless system of mercury and cesium supply. Mercury and cesium heated by the Sun were evaporated from the porous structures of the propellant containers and condensed on the ITM components. Vapor condensation on the accelerating electrode isolators and in the discharge chamber was proved to be particularly dangerous. As a result ITM-1 and ITM-2 switching had been accompanied by discharges between thruster electrodes.

The net effect of thruster operation (186 min) was orbital period increase by 0,06 s corresponding to the thrust impulse of 31 Ns. It follows that ITM-1 means thrust under unsteady conditions was 2,6 mN, i.e. three times less than at normal conditions.

At EPS emergency operation when breakdowns in ITM were observed during two turns, the receiving stations had recorded radio frequency emission noises of $(3 - 5) \cdot 10^{-6}$ V/m over 130...260 MHz frequency range that exceeded the background radiation intensity.

CONCLUSIONS

Electric thruster “Zefir” had been tested on the board Meteor spacecraft. First stage testing qualified ion engines for space conditions: their ability to be started and to become fully operational in a specified time interval as well as their ground based parameters.

Second stage testing included compatibility effects of ion engines on radio communication and other spacecraft systems. Also, using satellite orientation system ion engine thrust had been determined during 102 min of its operation. The measured value of the thrust 5,0 – 7,1 mN was found to be close to 7,0 mN resulting from both ground testing and computations based on operational parameters of the acceleration system. Other characteristics of ion thrusters compared favorably with ground tests.

Life tests had been concluded only partly due to breakdowns in the accelerator system.

In summary, flight testing of ion thrusters aboard a satellite allowed one to obtain valuable information to be used for design of advanced electric propulsion engines with ion thrusters.

REFERENCES

1. Kerslake W.R., Byers D.C., Rawlin V.K., Jones S.G., Berkopec F.D. Flight and ground performance of the SERT II thruster AIAA Paper, 1970, №1125, p.p. 1-12.