μ-PPT electro-propulsion system Development and First fight Application

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Abstract: The sustainable development of micro-nano-satellite application puts forward new requirements for propulsion systems. The micro pulsed plasma thruster(μ-PPT) with low mass, small volume and low power consumption is particularly suitable for the task of orbit maintenance and attitude control of micro-nano-satellites including standard cubic satellites, which becomes the focus of micro-electro-propulsion system. The 5 W μ-PPT electro-propulsion technology research was carried out by Lanzhou Institute of Physics, for the mission of orbit maintenance and attitude control of micro-orbit satellites with power less than 30 w, weight less than 50kg, and orbit lifetime less than one year. The μ-PPT uses the ablation type of flat structure, the arc spark of coaxial semiconductor spark and the spring-fed PTFE, and the engineering prototype is completed. The main performance indicators for operating frequency of 1 Hz, the impulse-bit of 40 μN.s, the specific impulse more than 600 s, and the discharge voltage of 1600 V. In order to meet the requirements of orbital attitude control application for AX-III 12U standard cubic satellites, the key technologies such as high integration, high reliability, long life design and verification, energy storage and conduction, etc. have been broken through and the μ-PPT Micro-electro-propulsion system identification of products and the development of products and the development of flight products has been completed. The micro-electric propulsion system consists of a μ-PPT pulsed plasma thruster unit and a power supply unit. Wherein the thruster unit comprises three thruster heads, an energy storage module and a frame module; the power processing unit comprised a charging power module, an ignition module and a control module. The system has already completed the mechanics, heat and other environment testing assessment. The main technical indicators for the system weight of less than 2 kg, power is less than 5 W, the average thrust is greater than 40 μN, the specific impulse greater than 600 s, the total impulse is greater than 60 N.s. The μ-PPT micro-electro-propulsion system completed 2.2 million times of ground 1:1 long-life test, and the performance and long-life test results show that the μ-PPT micro-electric propulsion system will satisfy the mission requirement. The on-orbit flight attitude and orbit application strategy have been made and μ-PPT micro-electric propulsion system will be applied in the cubic satellite for the first time in china.

I.Introduction

NOWADAYS the development of the micro-nano satellite is very rapid. In order to improve its performance and functionality, it is necessary to use a propulsion system to carry out the launch track and position correction, mission required orbit and attitude control and adjustment for changes in attitude and orbit caused by atmospheric

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damping and other disturbances leading to affecting task completion or working life and formation flight or adjustment. Without the propulsion system, the service life of the spacecraft will be greatly reduced. However, micro-nano satellites have strict requirements for propulsion systems, including small size, low power consumption, light weight, and ease of integration. For example, for a 20-liter microsatellite, the system power is required to push the system more than a dozen watts, and the propulsion system weighs less than 2 kg. In addition, the micro-nano satellite part of the task to be accurately adjustable thrust. For the selection of the micro-propulsion system, the chemical propulsion system has a large weight and can not meet the requirement of weight limitation. The cold gas propulsion system can not meet the requirements for micro-nano satellite precision control (thrust required in the μN order) due to the low specific impulse (≤100 s) and the low thrust resolution (minimum thrust ≥10mN). In addition, the traditional electric propulsion system, such as ions propulsion system and Hall propulsion system, can not meet the micro-nano satellite for the weight and power requirements. Micro-electrical-thruster has the advantages of light weight, small volume, high integration and low power consumption making it suitable for micro-satellite propulsion system. Of which PPT has the highest technical maturity compared with other kinds of micro propulsion system such as VAT(Vacuum Arc Thruser), CT(Colloid Thruster) and become a research focus in micro-electric propulsion system[1-9].

The development and application of the PPT is probably divided into three stages: origin, miniaturization and standardization. Firstly, PPT originated in the sixties and seventies of last century. The former Soviet Union and the United States have developed a variety of PPT space flight test. However, because the expected large thrust failed to achieve, and long life and reliable problem is difficult to solve, the PPT engineering application process of PPT remained sluggish. Secondly, the miniaturization stage is from the nineties of the last century to around 2010. During this period, it was gradually realized that the application of low-power PPT in the field of micro-satellite has a unique advantage, miniaturization became the main research direction of the PPT. Among them, the most representative research work carried out in the United States and Japan. Finally, the PPT standardization originated in the Cubesats satellite project from 2010. Some typical research is carried out in the British Clyde Space (Clyde Space), Surrey University, the University of Stuttgart, Germany Institute of Space Systems and the Austrian Research Center[10-16].

This paper introduces the research work of the Lanzhou Institute of Physics about PPT based on the application of micro-nano satellite application under 50 kg. The configuration of this paper mainly includes technical scheme, key technology research and product development of first flight application system of the PPT.

II.Technical Scheme

A. Scheme Selection

Lanzhou Institute of Physics carried out the 10 W magnitude μ-PPT electro-propulsion technology research and product development for orbit maintenance and attitude control of the micro-satellite with the power consumption less than 50 W and weight less than 50 kg.

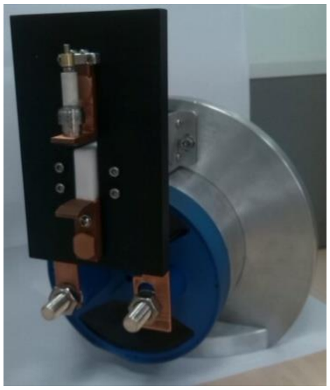

Figure 1. PPT prototype based on the selected scheme

From the review of the PPT technology development both domestic and international, there are two main types of structure as the parallel plate type and the coaxial type. Due to the parallel plate structure is of high maturity, the flying products are widely used. In addition, compared with the coaxial type, the performance of the parallel plate type is easy to be upgraded and also has many other advantages. Therefore, we selecte the ablation parallel structure as the main structure PPT electric propulsion system. On this basis, a coaxial semiconductor thin film type spark plug with a controllable frequency and stable discharge property is selected as a vacuum arc device. In addition, the demand for replenishment of the working fluid consumption, the use of torsion spring to promote the supply of the way to achieve stable supply of working fluid, this structure is simple and easy to integrate, easy to engineering. According to the basic scheme, the prototype was developed, as shown in Figure 1.

B. Performance of the prototype

Due to the small thrust and the short duration of the PPT, the thrust generated by the pulse has a certain discrete distribution characteristics. In engineering applications, the impulse characteristics of PPT are often evaluated as impulse-bit (I_{bit}), or the impulse generated by the thruster during a period of time, and serves as the main evaluation index of PPT propulsion performance. At the same time, the average pulse ablation mass (m_{loss}) was used to measure the propellant consumption during PPT work. Based on the above two basic parameters, combined with the PPT operating frequency, the specific impulse (I_{sp}), the average equivalent thrust (T), efficiency (η) and other performance parameters can be calculated.

I_{bit} can be directly calculated by the discharge current[17,18], but ignoring the effects of plumes in the calculation. In other words, if the plume is not strictly collimated, it could deflected and consumed in the discharge electrodes. In this condition, a large current can only contribution to a small thrust. Therefore, there are some disadvantages to calculation performance only use the discharge current. Based on this, a simple and effective method of PPT plasma process and performance characterization based on PPT discharge is proposed in the development of prototype. It can be expressed as follows: the electrical parameters are analyzed and the stable working range is obtained. Then, the optimal working condition is obtained by the plume diagnosis by the high speed camera. Finally, it is verified by the impulse-bit measurement and then a voltage-I_{bit} calibration curve can be generated. Using this method, the experimental research of the ignition, performance test and optimization of the prototype is carried out. The design index is achieved to 18 μN.s/J. The development baseline of the structure and performance parameters of the discharge chamber of the engineering prototype was determined. Typical discharge waveform of the PPT propulsion system and photograph of discharge plume is shown in figure 2.

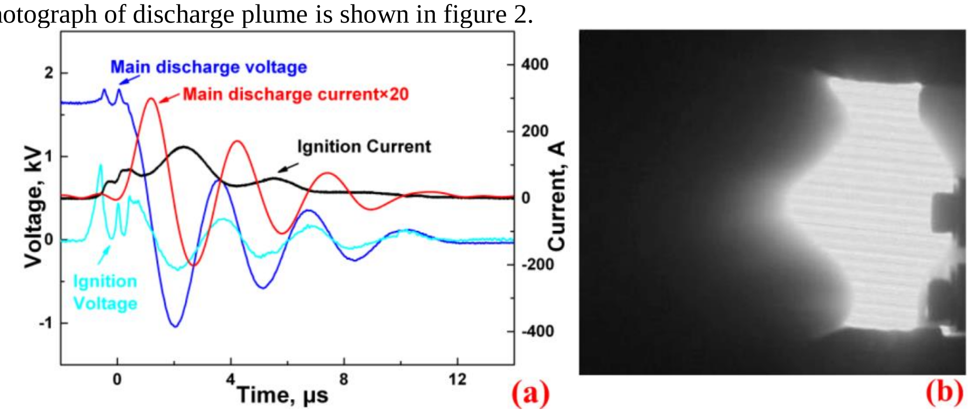

Figure 2. (a) Typical discharge U-I waveform, (b) Photograph of discharge plume

III.Key technology research

In the case of qualified performance, in order to achieve the engineering realization of the prototype, many key technologies need to break through. Specifically for the following aspects:

A. Micro-integrated technology

The breakthrough in integration technology of the micro-propulsion system is the integration in a limited space in the thruster. The thruster consists of frame module, energy storage module, the nozzle and the corresponding wires. The micro integration is conducted in these modules. Firstly, on the basis of structural features of the cubic satellite platform, the truss frame of modular structure is used. It has reliable structural mechanics. It can also be integrated in electrical fitting and modular design. Besides, ceramic capacitor with small volume and light mass is used as the energy storage module, so that high density of energy storage in the limited space is ensured and fully utilized. Ceramic capacitors are welded on both sides of the printed circuit boards to lower barycenter and enhance mechanical properties. Furthermore, the integration of the nozzle structural where discharge generated requires constant refueling of propellants as well as high-voltage insulation and carbon deposition limitation. In light of rigorous space limitation, other than adopting torsion spring and propellant refuel, in order to further enhance the stabilization in the end face, the propellant is also improved with the technology of parallel guide rail supply. In other words, the side face of propellant has a long bar. The ceramic insulation slip is installed with a long guide rail, in order to avoid unevenly stress and ensure the stable operation of propellant. As for high-voltage insulation, creepage-proof ceramic and polyimide frame are equipped and installed, so as to avoid high voltage discharge caused by carbon deposition. Trapezoid-shaped separation designs are used in side walls of polyimide. It effectively insulates the negative pole and positive pole, and avoid the connection induced by carbon deposition. Furthermore, through optimizing iteration experiment, it is feasible to select proper depth-width ratios and parameters, in order to ensure full ablation and utilization of performance. Lastly, frame module, the electric connection between energy storage module and nozzle of thruster is achieved through wire joining, so as to solve difficult technical issues concerning electric fitting in narrow space.

In summary, the small integrated design of the frame module, the energy storage module and the thruster nozzle achieves a narrow space thruster integration of a rail control nozzle and two attitude control nozzles at a capacity of 1U. At the same time with the attitude control and track maintenance function to meet the needs of satellite platforms.

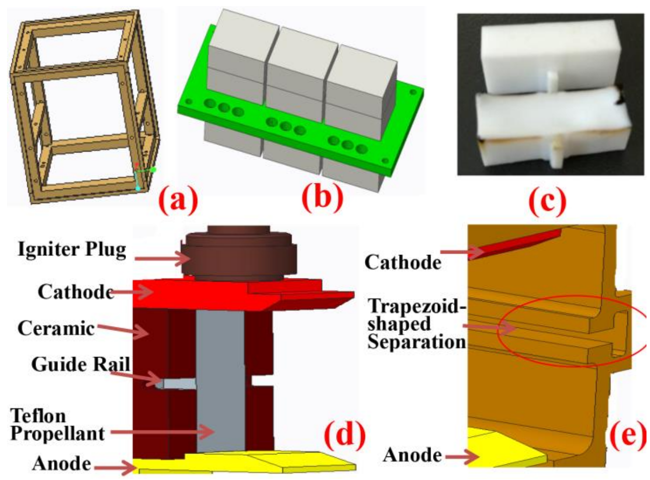

Figure 3. Schematic of integrating technology

B. Ignition system technology of highly reliability and long life

Ignition system mainly includes ignition circuit, spark plug and the corresponding wire. For the space mission, the basic principle of the ignition system is to generate a small pre-discharge to induce the main discharge using a small amount of energy, and maintain a high enough ignition success rate and reliability. Based on this, a coaxial semiconductor spark plug is used to strike arc in this work. This structure of the spark plug, due to the semiconductor metal oxide coating between the cathode and anode, with some advantages such as low resistance, not affected by air pressure, stable discharge, make the breakdown voltage is effectively reduce meanwhile the stability and consistency of the discharge is enhanced. The spark plug has a coaxial ring along the surface discharge structure, where the end is flat without tips and strong local electric field, almost do not disturb the main discharge electromagnetic acceleration. The ignition circuit adopts LC resonance booster technology. The working principle of this technology is as follows: First, charge storage capacitor to 900V, turn on the switching device, and then through the LC oscillation to produce a high voltage of twice the original value. After the spark plugs breakdown, the inductor is saturated due to large current and the capacitor discharges directly to the spark plug. The advantage of this topology is that the current on the switching device is just the discharge current, which avoids the high current caused by the transformer coil and reduces the current load of the switching device. In addition, even if the spark plug is not breakdown, or not connect to load, there will be no damage to other components. This topology can be adapt to the load impedance variation caused by film ablation, ensure ignition successful and form an efficient match with the coaxial semiconductor thin film spark plug. Above-mentioned specific system design effectively ensures the high reliability and long life requirements for the ignition system. The spark plug is shown in figure 4(a).

In addition, due to the harsh plasma environment where the spark plug work, the ablative product is deposited on the spark plug film, forming a carbon deposition, which can lead failure. Therefore, through the iterative optimization test and a reasonable set of spark discharge energy which could lead to the formation of short time high energy spark plug plasma discharge, can successfully induce the main discharge, meanwhile removal of carbon deposition and keep the semiconductor film not damage. In this way, the ignition system can has a long lifetime and good reliability. Typical discharge waveforms are shown in figure 4(b).

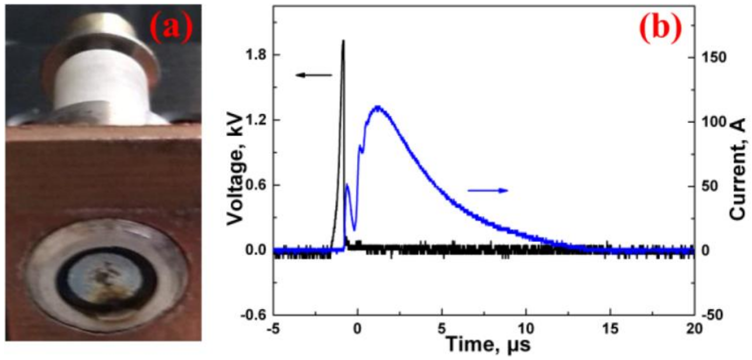

Figure 4. (a) spark plug structure，(b) Waveforms of igniter plug discharge voltage and current

C. Efficient energy storage and conduction technology

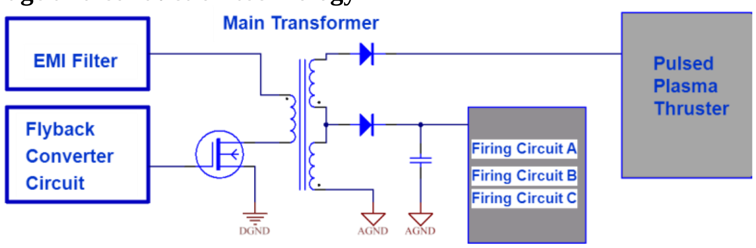

Figure 5. The topology of the charging circuit

Energy storage power supply system mainly includes charging circuit, energy storage capacitor and wire. In order to break through the key technologies, the ceramic capacitors are used as energy storage capacitors for strict limits of the size and quality on the satellite platform. In the case of satisfactory performance, a capacitor with a high rated voltage is selected and derating used. In order to realize the energy storage of the long life of the capacitor, the charging time of the charging module is required to meet the requirements of the working frequency. Further more, table power and small shocks are required to achieve high reliability and long life requirements. For this, the flyback converter is used as the charging circuit which is particularly suitable for 20 W low-power switching power supply, with some advantage such as simplicity, miniaturization, as shown is fig 5. Further more, the charging circuit adopts constant power charging scheme. PPU input power is approximately constant, which makes a low impact power to the bus, and can effectively reduce the electrical stress to storage capacitor and other component. In addition, Leeds soft wires are used as energy transfer conductors. The wire consists of dozens of thin wires with large cross-sectional area and high conductivity. So the current convergence effect is low, the energy transmission efficiency of the wire is high. This can effectively reduce the current stress to the capacitor and improve the life of energy storage capacitors. The waveforms of the charging voltage and bus current is shown in figure 6.

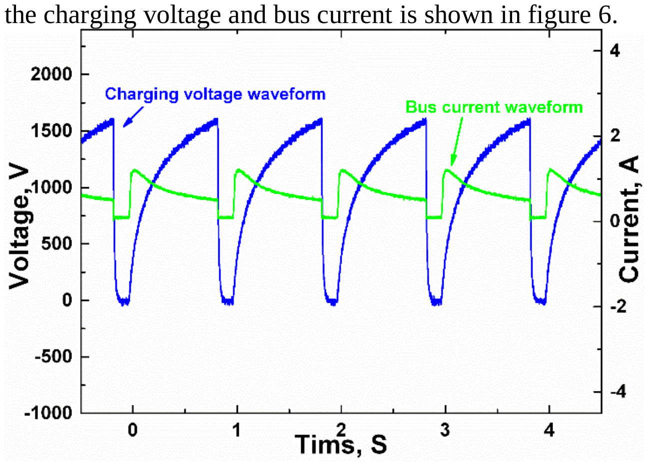

Figure 6. Waveforms of the charging voltage(blue) and bus current(green)

D. Performance verification technology for full life cycle

The key to the full life cycle test is the effective detection and verification of the performance in the process, especially the performance parameters consistency during the whole life cycle. Performance parameters mainly include two parts: first, the impulse-bit and specific impulse performance parameters; Second, the spark plugs and capacitors and other core components of the performance parameters.

In this paper, the performance parameters of the performance parameters such as impulse-bit and specific impulse are measured and characterized by state parameter detection and indirect calculation. By measuring the discharge current, discharge voltage, average ablation quality and other state parameters, and the impulse-bit, the specific impulse performance parameters can be calculated according to the formula in literature[17,18]. Although there is some error in this method, it is only necessary to obtain a relative value to meet the requirements in terms of consistency verification within the life cycle. In addition, compared to the micro-thruster platform during the life of the measurement, it is not necessary to exclude the complex system disturbance in the life cycle of and require a strict initial state consistency in each moment of opening and closing. Therefore, this method is a simple and effective method for performance consistency detection.

In addition, PPT thruster discharge is a kind of pulsed vacuum arc. From the basic physical principle, the discharge behavior has some uncertainty. For this feature, the consistency proposed in this paper is not means each performance value in the life cycle is the same, but the difference between a certain number of pulse discharge performance is not high that can meet the spacecraft mission control accuracy requirements. That is expressed as good consistency of performance.

Thus, to achieve the performance consistency of the detection thruster, the time step is set in the whole life cycle, the data for each step point is recorded, and the average ablation mass is weighed so that during this time, performance parameters I_{bit} and I_{sp} can be calculated to meet the limited time performance data statistical requirements and to achieve the purpose of performance data discreteness analysis. Further more, the performance curve of the performance parameters over time in the whole life cycle is represented so that the performance consistency can be obtained. Through this method, performance evaluation and verification are carried out. In addition, the assessment kit is required to perform a 1: 1 life test with the design flight status component to ensure the effectiveness of the test data.

In addition, the characteristics of the core components such as spark plugs and capacitors that affect the life of the thruster are measured in detail to analyze the characteristics of the thruster over time. Photographs of life test setup and cable connection is shown in figure 7.

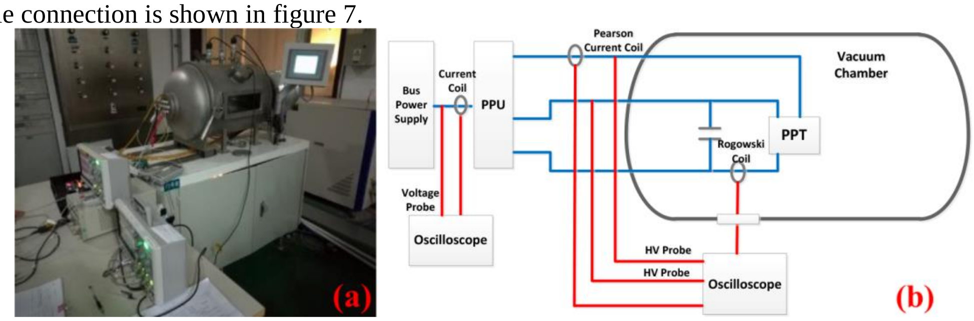

Figure 7. (a) Photographs of life test setup，(b) Schematic of cable connection

IV.Task analysis for first flight application

A. 12U cubic star PPT electric propulsion system composition and performance indicators

AX-III satellite is designed and manufactured by Northwestern Polytechnical University. The satellite design life is 3-6 months, the whole star volume is 12 U, where U is a volume unit equal to that of a rectangle with a side length of 100 mm. The power of the satellite is about 15-20 W and the orbital height is about 400 km. The satellite platform requires PPT to provide a certain impulse to offset the orbit environmental damping to extend satellite platform lifetime. In addition, the PPT electric propulsion system is required to do the attitude control task in the forward and reverse directions of the satellite X-axis (parallel to the flight direction). Through analysis, an average thrust of the satellite platform of no less than 37 μN is required to compensate the atmospheric damping and an impulse-bit of no less than 30 μN.s is required for attitude control task. Taking into account the lifetime, in the case of system power not higher than 5 W, the satellite platform orbit control need a total impulse of at lest 40 N.s and the attitude control need a total impulse of at lest 20 N.s. Some technical specification of the micro PPT propulsion system is shown in the table 1.

Tab.1. Summary of AX-III 12U standard cubic satellites propulsion requirements

	Type	Unit	Rated Value
1	Bus average input power	W	≤5
2	Impulse-bit（A single nozzle）	μN·s	≥40
3	Total impulse	N·s	Orbit Maintenance: 40N·s Attitude Control: 20N·s
4	Operating frequency	Hz	≥1
5	Specific impulse	s	≥450
6	Weight	kg	≤2
7	Interface temperature range	°C	-20~60
8	Lifetime	pulses	≥1.5×10⁶
9	External size (Thruster/PPU Unit)	mm	95.89mm×90.17mm×80mm
	Environmental adaptability	Meet the platform requirements	

PPT propulsion system in-orbit flight mission mainly includes the track maintenance and uniaxial attitude control of the AX-III satellite in orbit within the specified time. The whole satellite have two PPT electrical propulsion system. Each PPT electrical propulsion system contain a PPT thruster (each PPT thruster contain three nozzle, corresponding to three orthogonal directions) and a Power Processing Unit (PPU).

The main function of the PPT is generating the propulsion force through accelerating the ionized propellant to complete the track maintenance and attitude control mission of the satellite platform. In which, one thruster unit contain three thruster nozzle and a energy storage module. The operation of the three thruster nozzle is switched through the spark plug. In this way, the relay module can be eliminated and the weight can be reduced, the limited space can be effectively used.

The primary function of the PPU is to convert the power supply of the satellite into the power required for the PPT thruster to control the operating sequence of the thruster by receiving the satellite platform command and to output telemetry parameters that reflect the working state of the propulsion system. The I^2C bus was adopt for the communication between the PPU and satellite platform. To improve the compatibility of the product, the CAN bus is reserved. The bus provides a voltage of 6.4 V to 8.4 V and a power of no more than 5 W. PPU mainly includes three parts as the micro-controller, charging module and ignition module. The logical layout of the satellite and the propulsion system is shown in figure 8.

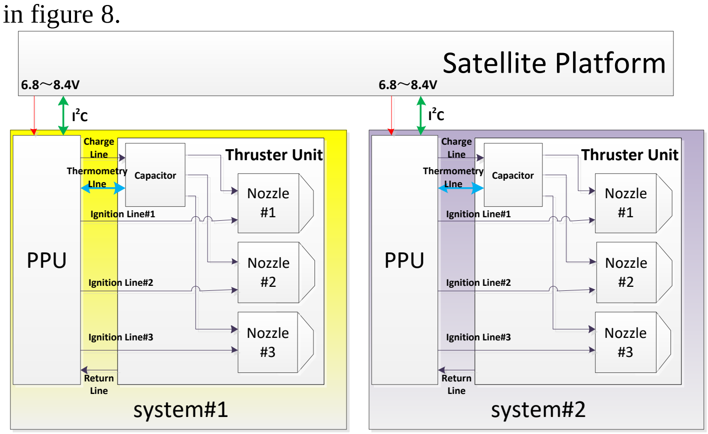

Figure 8. PPT electrical propulsion system configuration of the AX-III satellite

B. On board PPT electric propulsion layout and application strategy of the 12U cubic satellite

AX-III satellites are equipped with the same two sets of PPT electrical propulsion system, occupying 4U space of a total 12U of the cubic satellite. The thrust unit and PPU are installed adjacent to each other to reduce transmission losses. The two parts are connected using wire connection, with the terminal set in the thruster through which the PPU energy into the thruster. In order to prevent the interference from the PPU output to the bus input, the bus input and output should be in the symmetrically different side of the PPU surface. There are three sets of six nozzles, one of which is mounted on the tail to provide the thrust in the direction of the satellite orbit for orbit hoisting task; the other two sets of nozzles to provide the torque for satellite X-axis forward and reverse attitude control. The layout of the electrical propulsion system is shown in figure 9.

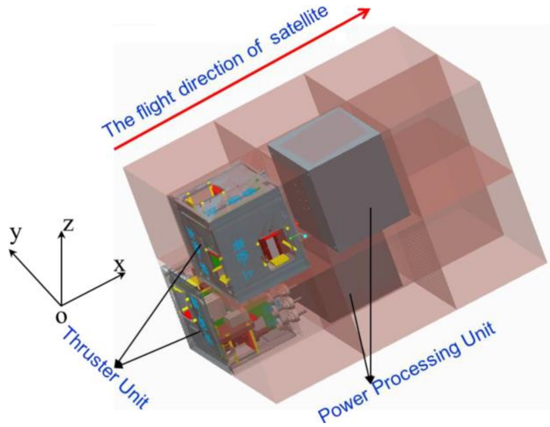

Figure 9. PPT electrical propulsion system configuration of the AX-III satellite.

System application strategy. PPT electric propulsion system tasks include track maintenance and attitude control, which is operated through command from the satellite platform. The application strategy mainly includes three main operation modes. First, the standby mode. The PPT electrical propulsion system is powered by the satellite platform, and the control system inside the power supply unit is in standby mode, waiting for the satellite platform to send track maintenance or attitude control commands so that the power supply module starts operating in sequence and supplies power to the thruster unit; Second, the track maintain mode. The satellite platform sends the track maintenance command when power applied. At this time, the PPT electric propulsion system starts to work in the order of charging and ignition, which makes the two nozzles of the system work which is used for orbit control. Third, (a) X axis forward attitude control instructions. The satellite platform sends the attitude command when power applied. At this time, the PPT electric propulsion system starts to work in the order of charging and ignition, so that the two sets of nozzles in the system for X-axis forward attitude control work; (b) X-axis reverse attitude control instructions. The satellite platform sends the attitude command when power applied. At this time, the PPT electric propulsion system starts to charge and ignition orderly, so that the two sets of nozzles in the system for X-axis reverse attitude control work.

V.Development of flight products for first flight application

According to the above discussion, based on the performance realization of the elementary prototype, the baseline of the size and performance parameters of engineering prototype are designed. On this basis, the integration in limited space, ignition modules and energy storage modules and other technical problems have been solved, and the flight products was developed. The flight product consists of two modules, a thruster module and a PPU module, as shown in figure 10. The performance of the flight prototype are also tested. The performance indicators of the final product is as follows: working discharge voltage is 1600 V, operating frequency is 1 Hz; impulse-bit is 40 μN.s; the specific impulse is 600 s and the system weighs is less than 2 kg.

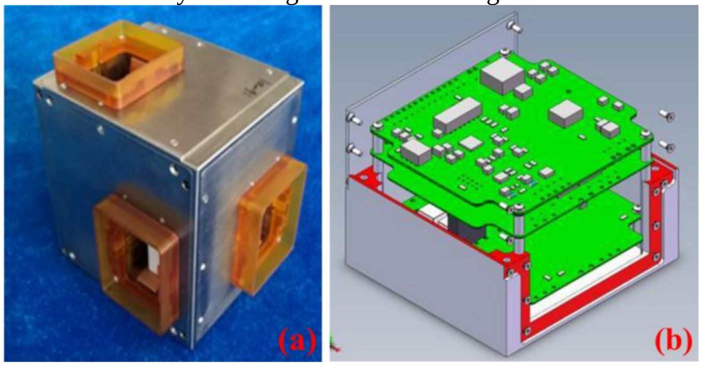

Figure 10. Flight prototype((a): PPT, (b): PPU)

VI.Environmental test assessment

A. Mechanical tests

Mechanical experiments include fundamental frequency scanning, acceleration mechanics experiment, sinusoidal vibration experiment, random vibration experiment and impact test. The experimental conditions are determined according to the general requirements. Before and after the experiment, the performance experiment was carried out to detect whether the product was affected by the mechanical experiment. The performance test items mainly include the discharge parameter measurement and the impulse-bit measurement. Each product test is synchronized between the product appearance inspection, the test of the spark plug impedance, capacitor capacity, insulation resistance and so other test.

Aiming at the installation structure and mechanical properties of the AX-III cubic satellite, the corresponding mechanical tooling is designed. The three-dimensional schematic diagram is shown in fig 11(a). During the experiment, each item was tested. After the test, the appearance is intact, no abnormal sound found inside, component parameters is not changed and the insulation resistance still properly. And the discharge performance parameters are compared before and after the mechanical test, as shown in figure 11(b). It is shown that the mechanical parameter the thruster keeps a good consistency before and after test, indicating that the thruster successfully pass the identification level and acceptance level mechanical test.

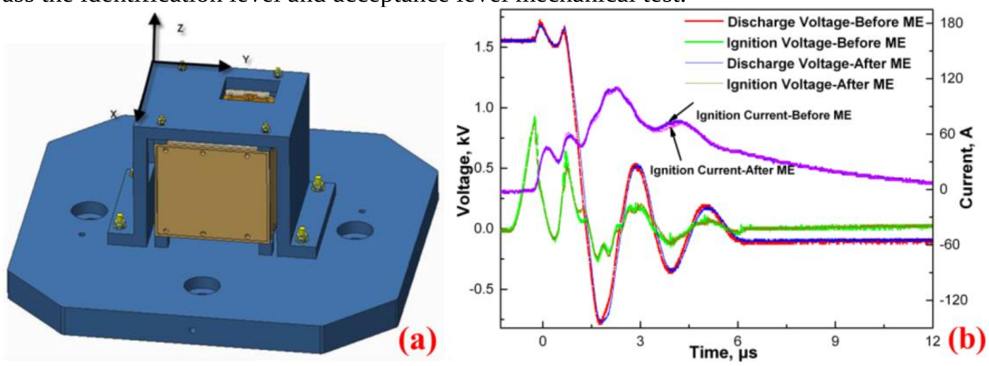

Figure 11. (a) Schematic of mechanics experiment setup，(b) Waveforms of discharge voltage and current before and after mechanics experiment(ME)

B. Thermal testing in vacuum

Thermal testing was designed to verify performance variation during the high and low temperature cycle. According to the mission requirements, the operating temperature range of the thermal testing is from -50 ℃ to +50 ℃, and the temperature change rate ≥ 1 ℃ / min. Thermal testing were carried out for 4 cycles. Due to the PPU is placed in the spacecraft cabin, according to the requirements, thermal vacuum testing temperature range is from -20 ℃ to +50 ℃. The high and low temperature of the test is controlled by the heat sink equipment of the vacuum chamber, and the temperature of the cabin and the product is controlled by thermal cage heating or heat sink liquid nitrogen circulation. The specific test process is that: First, ignition check before testing. The three nozzles are ignited in sequence with the ignition time is 5 min for each nozzle. Bus input voltage and current, ignition circuit voltage and current, charging voltage and other electrical parameters are monitored during the test. Then, high and low temperature cycle test begin after a insulation time of 30 minutes. Each nozzle is ignited for 20 minutes result in a total 1 hour testing time. Testing were carried out for 4 cycles. After four testing cycles, the final temperature rose to about 30 ℃ and holding 30min, then the test at room temperature ignition begin with each nozzle ignition time of 5 min. The performance test are carried out before and after the thermal test, which is of the same item as before mentioned mechanical test. It is shown that, the performance parameter in high and low temperature is the same as in room temperature indicating that that parts and the structure of the PPT have a good adaptability of high and low temperature environment and present a good performance stability.

C. Lifetime test

The key of the lifetime test is the effective measurement of the thruster performance parameters and its components during the test. Based on this, the first step is the electrical parameters record, waveform storage and discharge status monitoring. The waveforms are recorded every half hour, including the bus voltage, the bus current, the main discharge voltage, the main discharge current and ignition voltage and ignition current. The discharge monitoring to observe the plume discharge whether there is an exception is carried out simultaneously; after the ignition discharge totaled 100,000 times, the condition of the thruster was checked, including the ablation of the propellant surface and the carbon deposition caused by the ablation. The parameter of the core components of the spark plug and capacitor, such as impedance, inductance and capacitance are also tested simultaneously. Through the discharge current and ablation quality, the performance parameters of the impulse-bit and the specific impulse is calculated over time, as shown in figure 12. Discharge plume observation is carried out simultaneous, no abnormal phenomenon of the plume was observed during the lifetime test. The capacitor also shows a good consistence with the capacitance variation of less than 1%. The impedance of spark plug semiconductor film has been reduced over time, but the impedance decline rate becomes slower with the passage of time so that the ignition is normal all the time and can successfully induce the main discharge with no accidental discharge and failed discharge occurred. The number of ignition totaled 2.2 million times, fully meet the mission requirements of 1 million times.

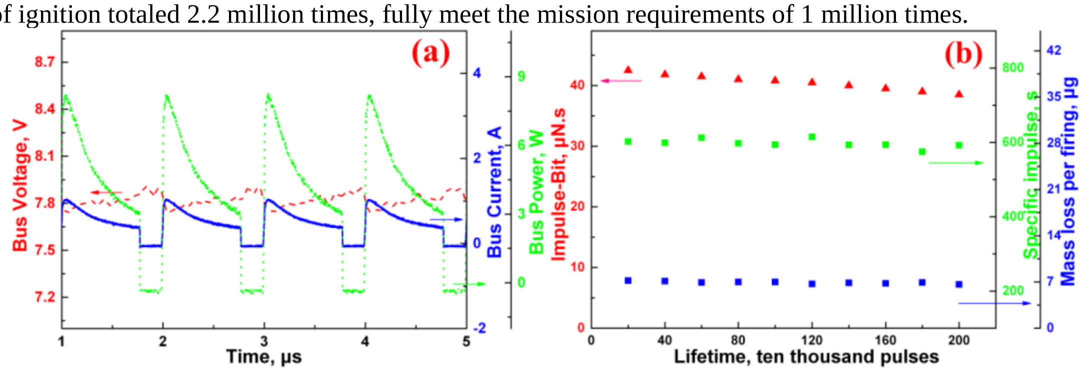

Figure 12. (a) Waveforms of the bus voltage and current，(b) Variation of Impulse-Bit, Specific impulse and Mass loss per firing with Lifetime

VII.Conclusion

A 10W-class μ-PPT electric propulsion system is developed by Lanzhou Institute of Physics using a flat plate structure ablation type, coaxial semiconductor spark plug arc and the propellant spring feed technology program. The development of the principle prototype and engineering prototype has completed. The main performance indicators are as follows: operating frequency is 1 Hz, impulse-bit is 40 μN.s, specific impulse is 600 s and discharge voltage is 1600 V. For the applications of the 12 U standard cubic satellite, several key technology such as highly micro integration, highly reliable long lifetime ignition, efficient energy storage and transmission and other technologies achieve a breakthrough. The development of the identification product is completed. The main performance indicators of the system products are as follows: the weight is less than 2 kg, the power consumption is 5 W, the average thrust is 40μN, the specific impulse is 600 s, the total impulse is 60 N.s. The requirements of the whole satellite task is fully achieved. The satellite will be in orbit operation in 2018, which is the China's first micro electric propulsion system of the cubic satellite for the first time in orbit applications.

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