

Article

Variable Switching System for Heat Protection and Dissipation of Ultra-LEO Satellites Based on LHP Coupled with TEC

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Abstract: Ultra-low Earth orbit (LEO) satellites are widely used in the military, remote sensing, scientific research, and other fields. The ultra-LEO satellite faces the harsh aerothermal environment, and the complex variable attitude task requires the radiator of the satellite to not only meet the heat dissipation requirements of the load but also to resist aerothermal flux. In this study, the aerothermal flux of 160–110 km was calculated, and the loop heat pipe (LHP) coupled with a thermo-electric cooler (TEC) and multi-layer insulation (MLI) were applied to ultra-LEO satellites to determine the variable switching and fast response of heat dissipation and heat protection. An aerothermal flux simulation test platform was built. After the assessment of the ultra-LEO aerothermal flux test, even when the head temperature was as high as 350 °C and the side radiator temperature was as high as 160 °C, the temperature of the internal heat source could be controlled within 22.5 °C through the efficient work of the thermal variable switch system. The study confirms the accuracy and feasibility of the system, which provides an important reference for the subsequent actual on-orbit mission.

Keywords: ultra-LEO; satellite; aerothermal; LHP; TEC; heat protection; heat dissipation



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1. Introduction

Ultra-low Earth orbit (LEO) satellites with an orbital altitude of 100–300 km are considered to be a new solution for future space missions, as shown in Figure 1. Compared with traditional orbit altitude satellites, ultra-LEO satellites can significantly improve the Earth observation resolution, Earth communication capacity, and data transmission speed while reducing the launch costs of carrier rockets [1–3]. In recent years, ultra-LEO satellites have become a topic of interest in the field of space technology.

The Gravity field and Ocean Circulation Explorer (GOCE) satellite [4] launched in 2009 by the European Space Agency (ESA), designed for gravity field and ocean circulation exploration, operated at an orbit altitude of approximately 260 km. The Sparse Atmosphere Science Experiment Satellite [5] launched in 2016 by the Chinese Academy of Sciences, aimed to collect temperature and density data of the low orbit sparse atmosphere, with a minimum orbit altitude of 109 km, as shown in Figure 2. The Super Low Altitude Test Satellite (SLATS) [6] launched in 2017 by the Japan Aerospace Exploration Agency (JAXA) conducted high-resolution Earth observation while studying the characteristics of atomic oxygen in ultra-LEO, operating as low as 180 km.

As the flight speed of the satellite in the LEO is about 7.8 km/s when descending to altitudes of ≤ 160 km, the atmospheric density significantly increases compared with the higher orbit environment. The rarefied air particles at high speed collide with the satellite surface, resulting in aerothermal flux [7] which becomes more pronounced as the satellite orbit gradually lowers. At an orbit altitude of 120 km, the aerothermal flux in the windward region can reach up to 5100 W/m².

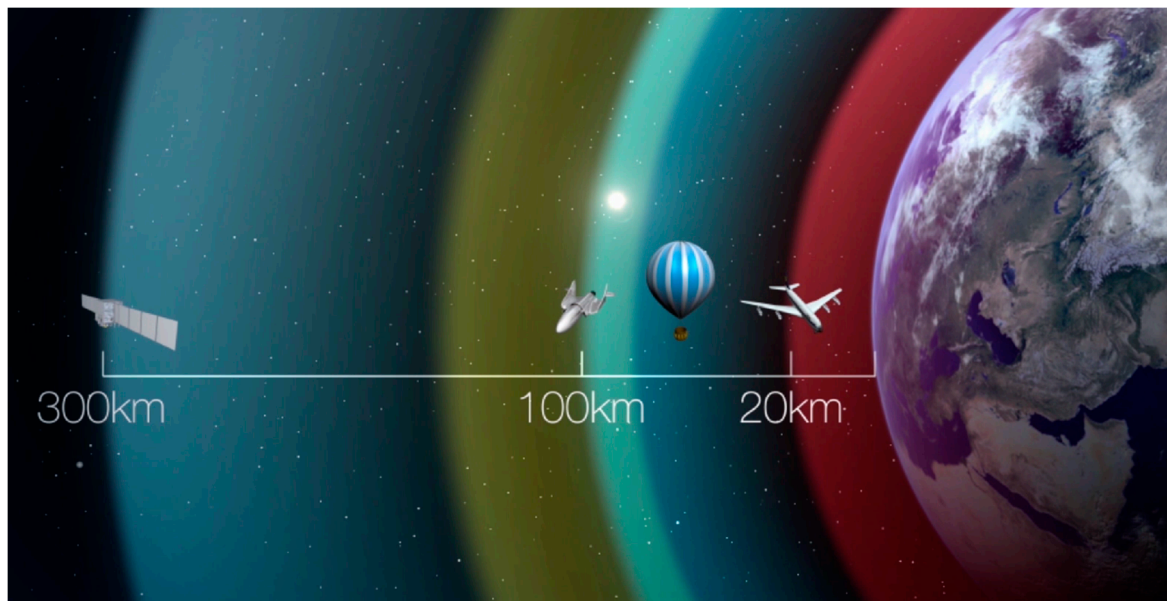


Figure 1. Ultra-low orbit flying area.



Figure 2. The Sparse Atmosphere Science Experiment Satellite.

Typically, the thermal protection systems of ultra-LEO spacecraft primarily use heat-resistant materials covering the external surface to minimize the impact of external aerothermal flux. The X-43A spacecraft tested by National Aeronautics and Space Administration (NASA, Washington, DC, USA) in 2001 employs C/C materials [8] on the nose and leading edges and alumina-enhanced thermal barrier rigid ceramic heat-resistant tiles on the fuselage and underside of the wings [9]. Other spacecraft experiencing lower aerothermal flux typically employ flexible thermal insulation blankets or multi-layer insulation (MLI) components, which are composed of metal foils and fiberglass cloth, with temperature resistance ranging from approximately 100 °C to 600 °C.

The application of these thermal protection materials inevitably obscures the valuable radiative cooling area of the satellite's surface. Therefore, the covering of thermal protection materials cannot achieve variable switching capabilities between thermal protection and heat dissipation.

Over recent years, with the continuous expansion of application demands, satellites have often needed to perform complex attitude variation tasks in ultra-low orbit, such as changing attitudes for communication, observation, and orbit alteration. Meanwhile, high-power payloads also have urgent heat dissipation requirements. Figure 3 shows the schematic diagram of the background mission in this paper.

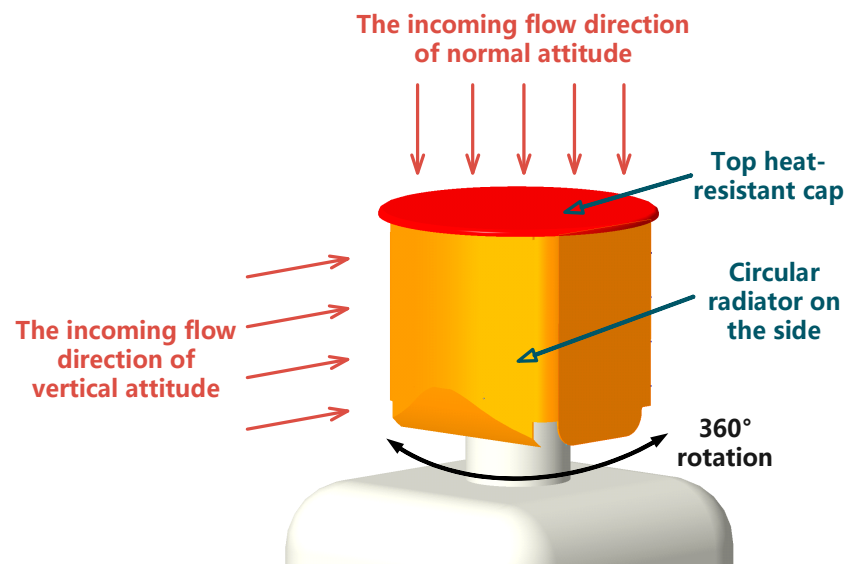


Figure 3. Schematic diagram of complex ultra-low orbit missions.

Adjusting the attitude in ultra-low orbit will alter the space thermal environment of the original radiator surface, leading to a decrease or even loss of heat dissipation capability. This necessitates that the radiator region on the satellite not only meets the heat dissipation requirements but also withstands the aerothermal flux impact after attitude changes, protecting the internal payload temperature from the influence of aerothermal flux. Therefore, research into a heat protection and dissipation variable switching system is urgently needed.

The heat dissipation and heat protection areas of traditional satellites are fixed areas divided independently, and the roles of the regions are generally not interchangeable. In traditional satellites, there are a few applications of variable thermal dissipation technology, mainly including adjustable thermal coatings based on the electrically-controlled change of surface emissivity [10,11], thermal switches using micro-electromechanical system (MEMS) louvers or electrostatically adjustable thermal resistance [12,13], and a deployable radiator coupled with an angle adjustment structure and flexible heat pipe [14,15], as shown in Figure 4. However, these technologies have limitations such as poor stability, high temperature intolerance, or the disruption of aerodynamic shapes, making them unsuitable for ultra-LEO missions.

Based on the previously discussed research, thermal protection materials and traditional variable heat dissipation technology do not meet the variable heat dissipation requirements of ultra-LEO satellites. This study aimed to apply the two-phase fluid loop controlled by a loop heat pipe (LHP) coupled with a thermo-electric cooler (TEC) to ultra-LEO satellites to achieve the variable switching ability of ultra-LEO heat protection and dissipation. Figure 5 shows a comparison diagram of the technical routes.

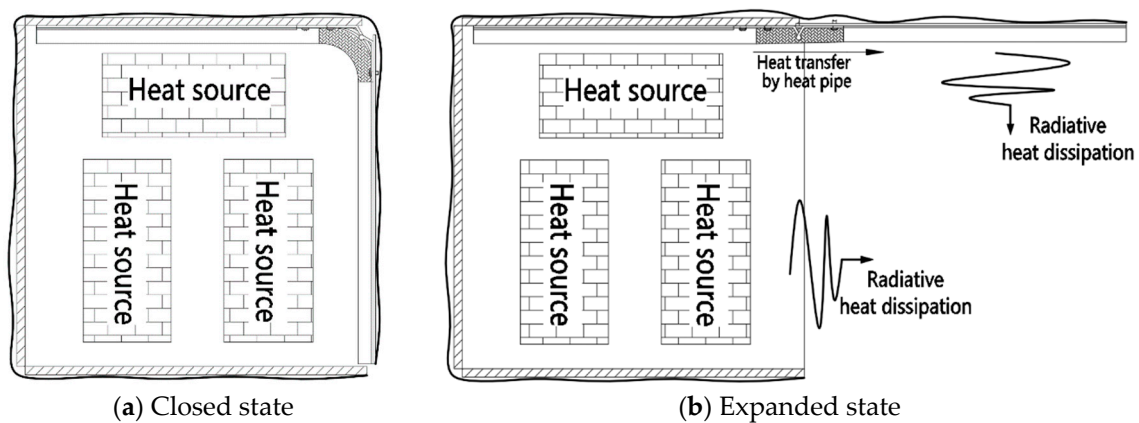


Figure 4. Diagram of an expandable radiator.

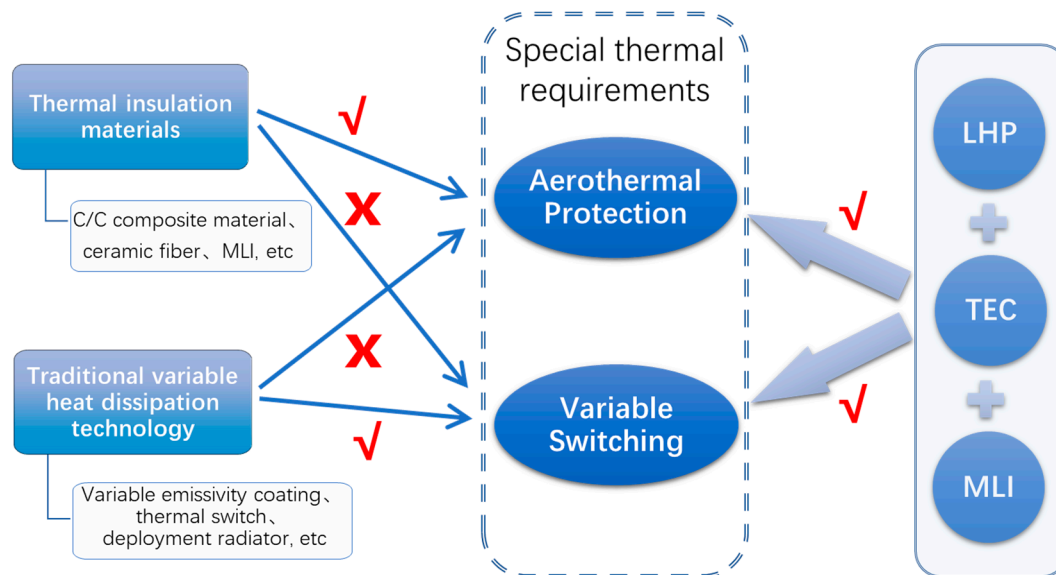


Figure 5. Diagram of technical roadmap comparison.

2. System Scheme

2.1. Design of Variable Heat Dissipation System

A system of LHP coupled with TEC and MLI was designed, which not only met the heat dissipation requirements of the internal payload but also resisted the external aerothermal flux during the task of changing attitude. Figure 6 shows the system diagram.

An LHP is a one-way heat transfer tool, but its startup speed and response speed pose difficult problems that have always hindered its application. The ultra-LEO mission is very compact and complex, so variable systems in ultra-LEO missions necessitate rapid response times.

Figure 7 shows the principal diagram of the LHP coupled with a TEC and MLI applied to ultra-LEO satellites. The circular radiator is made of aluminum alloy material and sprayed with a high-temperature resistant inorganic coating on the outer surface. When the radiator is not facing the airflow, it remains at a low temperature, enabling effective heat dissipation. It controls the operation of the LHP and transfers the internal payload heat to the cold radiator through the two-phase working medium. When the side radiator faces the airflow, the temperature increases under the influence of aerothermal flux and stops the LHP from running. The aerothermal flux on the radiator cannot be returned to the internal payload through the heat pipe, and the MLI material insulates the radiation heat transfer between the high temperature radiator and the internal payload.

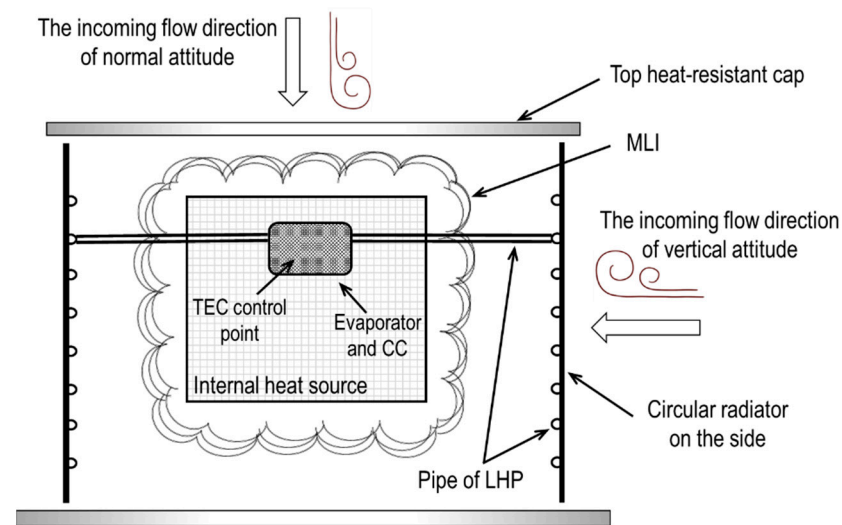


Figure 6. System design diagram.

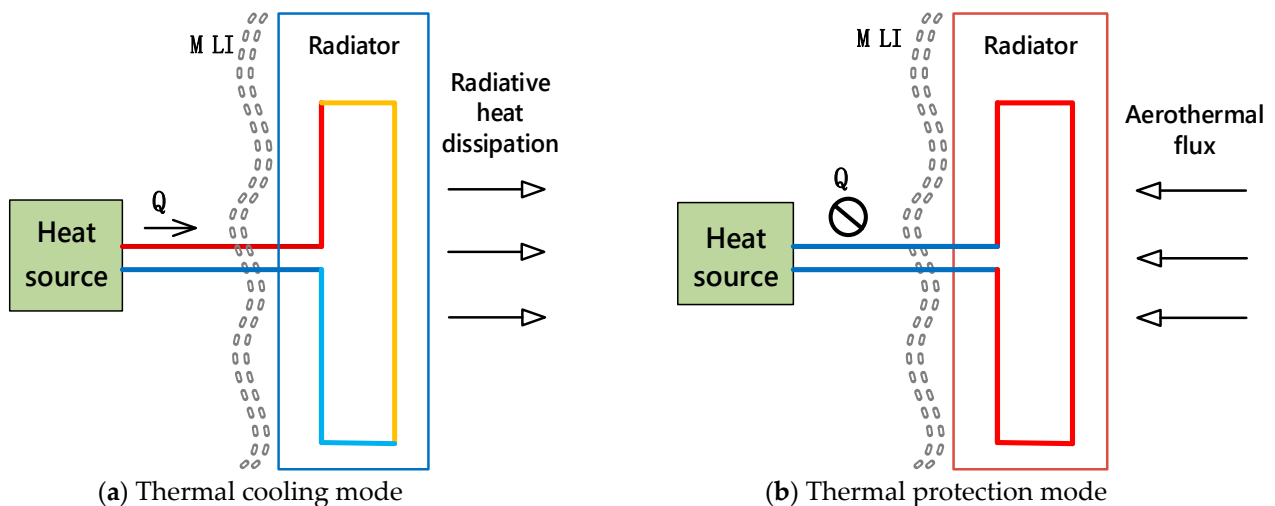


Figure 7. Thermal system working mode.

2.2. Design of TEC Control

An LHP is a closed loop heat pipe, which is usually composed of an evaporator, a compensation chamber (CC), a vapor pipeline, a condenser, a liquid pipeline, and a working medium [16,17]. The LHP condenser is a radiator arranged with pipes. The working medium absorbs heat and vaporizes in the evaporator and then enters the condenser to liquefy and release heat; the supercooled liquid is returned through the pipeline to the CC and is finally replenished to the evaporator.

In this paper, a TEC was used to control the LHP, and two sides of the TEC were in thermal contact with the evaporator and CC. The advantages of a TEC-controlled LHP are a higher control efficiency, a faster response, and better suitability for fine control [18,19].

Figure 8 shows the schematic diagram of a thermo-electric cooler (TEC)-controlled loop heat pipe (LHP). The temperature difference between the two sides of the TEC is controlled to form a positive pressure difference inside the LHP, which promotes the quick start of the LHP and drives the heat dissipation system. When the attitude changes and the radiator is heated by aerothermal flux, the power supply to the TEC is reversed and the reverse pressure difference is formed in the LHP, which hinders the operation of the heat pipe and forms a heat protection system.

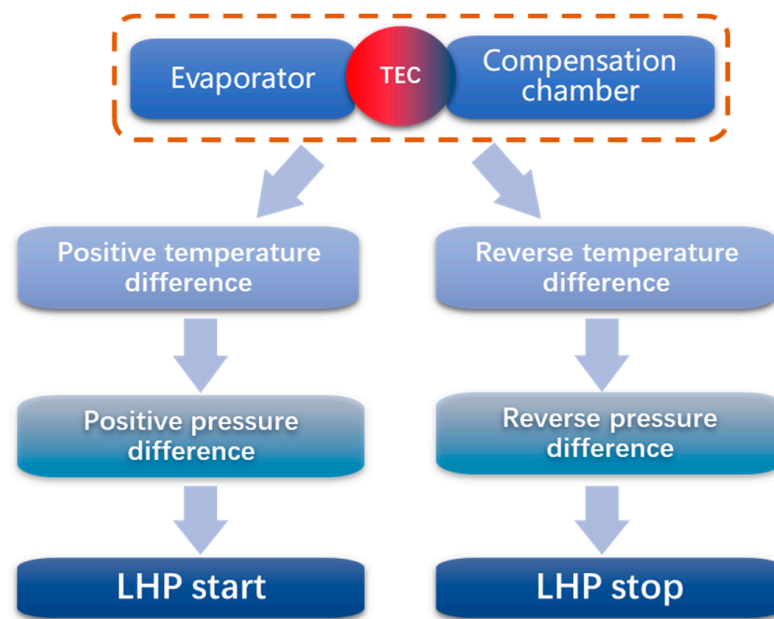


Figure 8. Schematic diagram of thermo-electric cooler (TEC)-controlled loop heat pipe (LHP).

The startup problem of the LHP is an important factor that limits its widespread use in the aerospace field. In practical engineering, problems such as failure to start, slow startup, and excessively high startup temperature of the LHP often occur. By using a TEC to start the LHP, cooling can be applied to the CC to form the same degree of superheat and push the heat pipe to start running. The principle is illustrated in Figure 9.

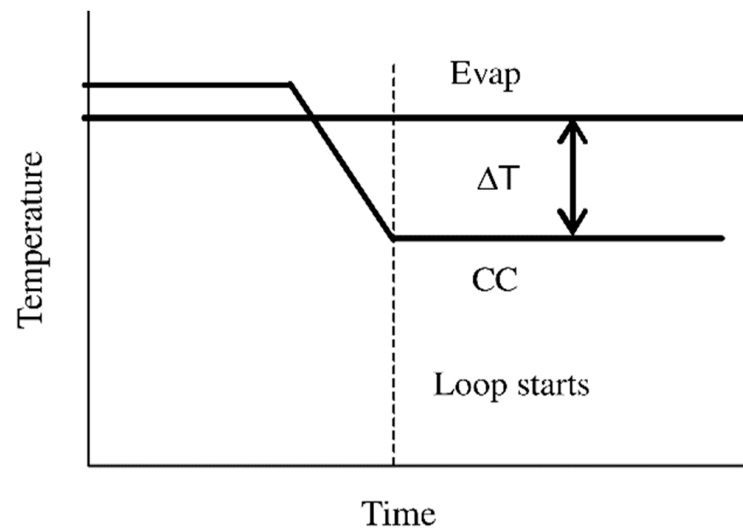


Figure 9. Schematic diagram of LHP startup by TEC.

3. Mission Flow and Aerothermal Flux Calculation

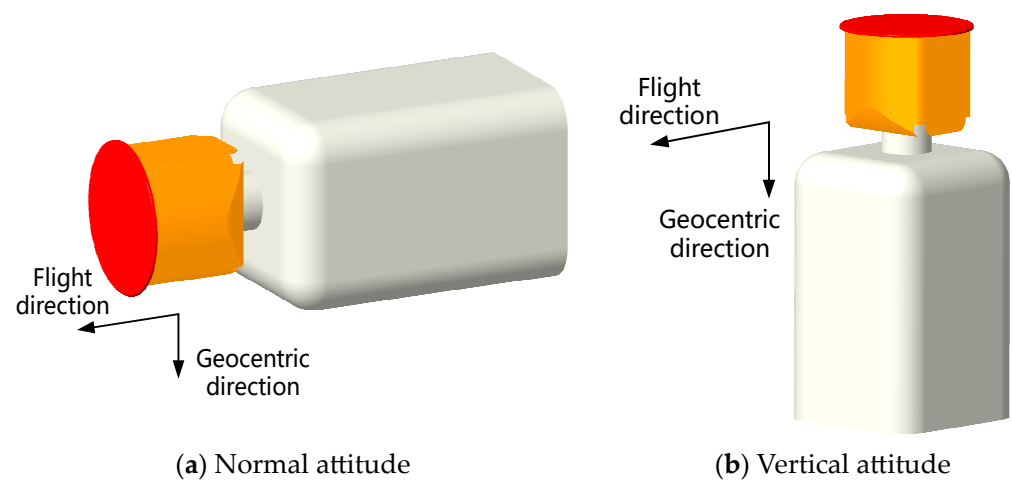
3.1. Introduction of Mission Flow

According to the mission plan, during the orbit descent of 160–120 km, to reduce the aerodynamic drag, the aircraft flies with a smaller windward attitude facing the head forward and carries out one vertical attitude task at 160, 150, 140, and 130 km and three vertical attitude tasks at 120 km altitude, as shown in Table 1.

Table 1. Introduction of the ultra-low earth orbit flight missions.

Orbit Altitude	Mission Statement	Vertical Flight Time	Internal Payload Heat Dissipation
160–130 km	One task per 10 km reduction	20 min	90 W
120 km	Three vertical attitude tasks	12 min	90 W

Figure 10 shows the schematic diagram of the two attitudes. Normal attitude: Top ahead flight reduces windward area while resisting aerothermal flux; Vertical attitude: According to the payload requirement, short-term side ahead flight within 20 min.

**Figure 10.** Flight attitude diagram.

This task requires an attitude change in ultra-LEO, which will change the space environment of the radiator surface, resulting in the decrease or even loss of its heat dissipation capacity. The side radiator should not only meet the internal payload heat dissipation requirement but also withstand the impact of aerothermal flux after an attitude change. Moreover, the variable switch system of heat protection and dissipation must be controlled autonomously and must respond quickly in orbit.

3.2. Calculation of Aerothermal Flux

For orbit altitudes (160–110 km) relevant to the background mission in this article, this altitude range belongs to the extremely rarefied region of free molecular flow [20]. In the free molecular flow regime, the average free path of gas molecules is much larger than the characteristic scale of the flow field. The flow characteristics are dominated by the interaction between gas molecules and the surface, while collisions between gas molecules can be neglected.

For the aerothermal flux problem of rarefied gas, an aerothermal engineering prediction method based on molecular collision theory is used. For a free molecular flow with an incident flow velocity of v_∞ , temperature T_∞ , and pressure P_∞ , on a surface element with wall temperature T_w and inclination angle θ , the aerothermal flux can be derived from the kinetic theory of molecules [21].

The number of molecules per unit volume is given by

$$N = \frac{P_\infty}{kT_\infty} \quad (1)$$

The molecular speed ratio is given by

$$S = \sqrt{\frac{\gamma}{2}} Ma_{\infty} \quad (2)$$

The number of molecules passing per unit area in unit time is given by

$$n = \psi \left[e^{-\eta^2} + \sqrt{\pi} \eta (1 + \operatorname{erf}(\eta)) \right] \quad (3)$$

where ψ , η and $\operatorname{erf}(\eta)$ are given by Equations (4)–(6).

$$\psi = \frac{N v_{\infty}}{2\sqrt{\pi} S} \quad (4)$$

$$\eta = S \sin \beta \quad (5)$$

$$\operatorname{erf}(\eta) = \frac{2}{\sqrt{\pi}} \int_0^{\eta} (e^{-x^2} dx) \quad (6)$$

The heat flux calculation formula of free molecular flow is as follows [22]:

$$q = -\alpha \left\{ \frac{\gamma + 1}{2(\gamma - 1)} n k T_w - \left[\left(S^2 + \frac{\gamma}{\gamma - 1} \right) n - \phi \right] k T_{\infty} \right\}. \quad (7)$$

where ϕ is given by Equation (8).

$$\phi = \frac{\psi}{2} e^{-\eta^2}. \quad (8)$$

Through calculation, the statistics of aerothermal flux at different orbital altitudes (160–110 km) and windward angles are shown in Table 2.

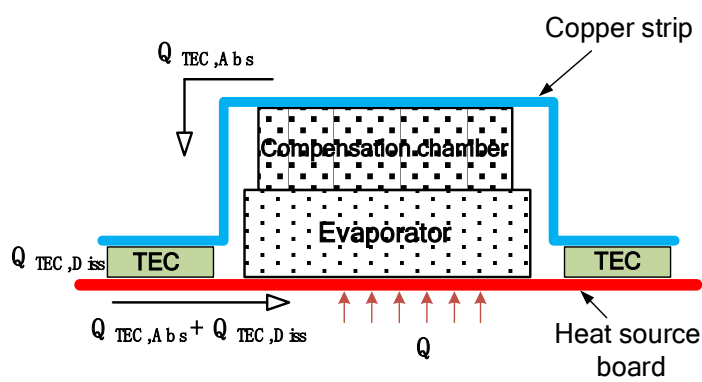
Table 2. Calculation results of aerothermal flux of ultra-LEO aircraft.

Orbit Altitude	Atmospheric Density (kg/m ³)	Aerothermal Flux in Various Regions (W/m ²)		
		$\theta = 90^\circ$	$\theta = 30^\circ$	$\theta = 0^\circ$
More than 200 km	Aerothermal Flux Need Not Be Considered			
160 km	1.23×10^{-9}	300	33	7.7
150 km	2.07×10^{-9}	500	55	13
140 km	3.83×10^{-9}	910	100	22
135 km	5.07×10^{-9}	1300	140	29
130 km	8.15×10^{-9}	1900	210	42
120 km	2.22×10^{-8}	5100	560	110
110 km	9.70×10^{-8}	21,000	2300	460

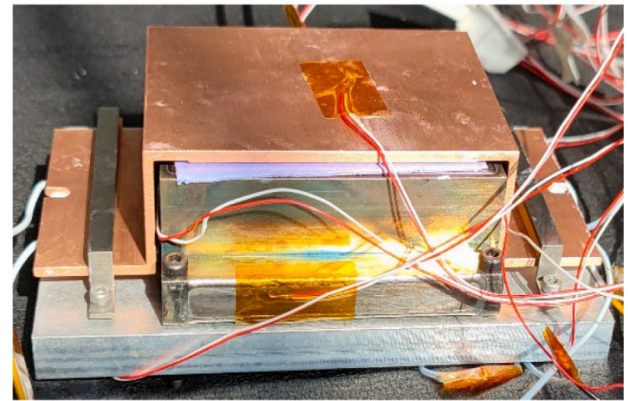
4. Experimental Setup

4.1. LHP Coupled with TEC

When used in conjunction with the flat LHP, two TECs were installed on either side of the evaporator and connected in series. One end of the TEC was installed on the heat source board and was in close contact with the evaporator, while the other end of the TEC was connected to the top cover of the CC via a thermal copper strip, and a flexible thermal pad was filled between the copper strip and the CC. In addition, special attention was paid to insulation and shock absorption when installing the copper strip to ensure the efficiency of TEC operation and prevent damage to the TEC during rocket launch. The schematic diagram and photo of the experimental status after assembly are shown in Figure 11.



(a) Schematic diagram



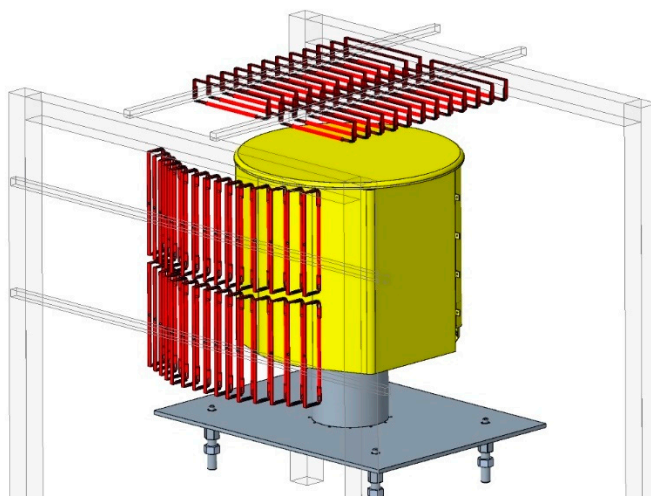
(b) Photo

Figure 11. LHP coupled with TEC.

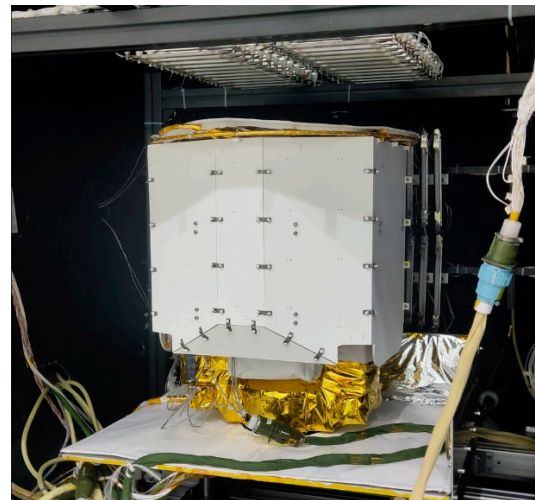
4.2. Aerothermal Flux Loading

Ultra-LEO aerothermal flux is caused by the high-speed collision of air particles with the aircraft surface, converting kinetic energy into thermal energy. Currently, there is no 1:1 scale experimental equipment for directly loading aerothermal flux on the ground.

This study designed an equivalent aerothermal flux simulation device, which consisted of 54 infrared lamps, divided into two arrays: the top and the side. Each lamp was powered by an independent DC power supply, and the maximum power of a single lamp was 500 W. The schematic diagram and photo of the aerothermal flux loading system are shown in Figure 12.



(a) Schematic diagram



(b) Photo

Figure 12. Infrared lamp array.

Due to the large aerothermal flux value to be simulated, the infrared lamp array arrangement was very dense. To ensure the accuracy of the flux in the plane, the heat flux loading system was specially calibrated before the formal test. A number of heat flux meters were arranged in the heating position, and the target temperature was set according to the different calculated aerothermal flux values. The power of each lamp was automatically adjusted by software program control, so that the temperature of each heat flux meter could reach the target temperature. This ensured the uniformity and accuracy of the experimental heat flux loading.

4.3. Test Methods

System vacuum thermal testing requires space environment simulation equipment, heat flux simulation equipment, a programmable regulated DC power supply, temperature data acquisition equipment, liquid nitrogen supply equipment, and vacuum extraction equipment. The test was conducted inside a KM II space environment simulator, which uses liquid nitrogen for refrigeration and has liquid nitrogen lines on the side walls and front and rear end faces to ensure that the temperature of different positions of the heat sink is lower than $-173\text{ }^{\circ}\text{C}$. The inner surface of the vacuum chamber was sprayed with a black coating with an infrared emissivity >0.88 . The internal pressure of the vacuum chamber is $<10^{-3}\text{ Pa}$ (convective heat transfer can be ignored). The testing system was placed at the center of the vacuum tank through support, as shown in Figure 13.

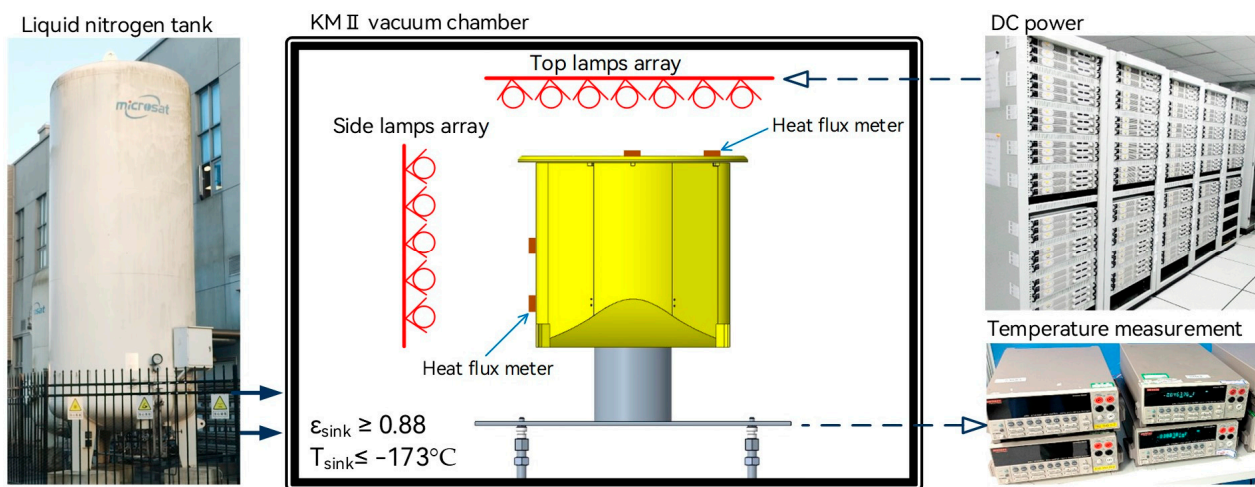


Figure 13. System of the experiment.

The temperature signal measurement system was mainly composed of a temperature sensor, a data acquisition board and a computer. T-type thermocouples with a temperature measurement accuracy of $\pm 0.5\text{ }^{\circ}\text{C}$ were used. The accuracy of the DC power supply used for the TEC and lamps can reach $\pm 0.01\text{ A}$.

5. Experiments and Results

5.1. Effect of TEC Power on System Temperature

The selection of the TEC power supply needs to be matched with the system. If the TEC power is small and the cooling capacity is not enough, it is difficult to quickly activate the LHP operation with a large flow rate. If the TEC power is too large, it will increase the heat dissipation of the TEC hot side and then increase the heat load of the entire cooling system, which has a negative effect on cooling. In this study, the effects of TEC power increasing from 0 to 25 W on the thermal resistance of the system, heat pipe temperature, and heat source temperature were tested.

Figure 14 shows the influence of TEC power on the system thermal resistance. The results indicate that the increase of TEC power can gradually reduce the thermal resistance of the system from $0.58\text{ }^{\circ}\text{C/W}$ to $0.34\text{ }^{\circ}\text{C/W}$. There is internal heat leakage during LHP operation, which will increase thermal resistance and reduce heat transfer efficiency, while the cooling capacity generated by TEC can compensate for the loss of heat leakage, thereby reducing the thermal resistance of heat transfer and improving heat transfer efficiency.

Figures 15 and 16 show the LHP operating temperature and heat source temperature under different TEC powers. The inclusion of a TEC greatly reduces the LHP operating temperature from approximately $-7\text{ }^{\circ}\text{C}$ to around $-22\text{ }^{\circ}\text{C}$. The cooling effect of the system was significantly enhanced, and the temperature of the heat source affected by this also reduced from the initial $11\text{ }^{\circ}\text{C}$ to around $3\text{ }^{\circ}\text{C}$.

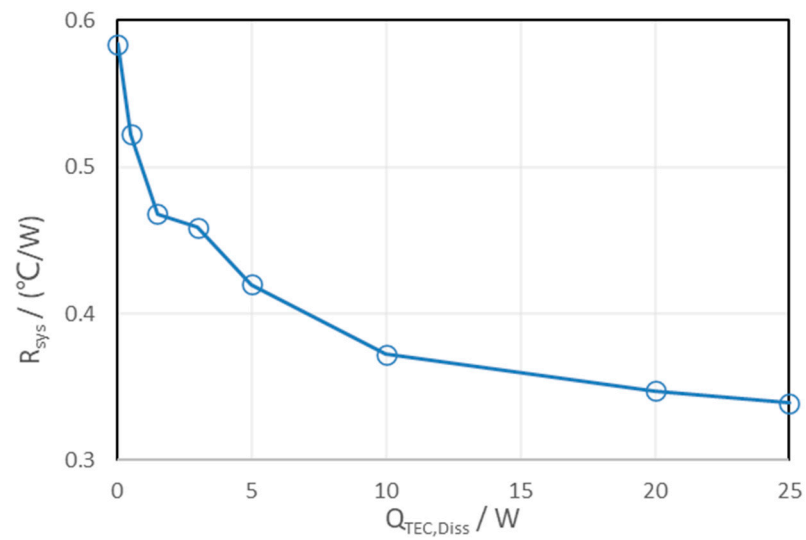


Figure 14. Influence of TEC power on system thermal resistance.

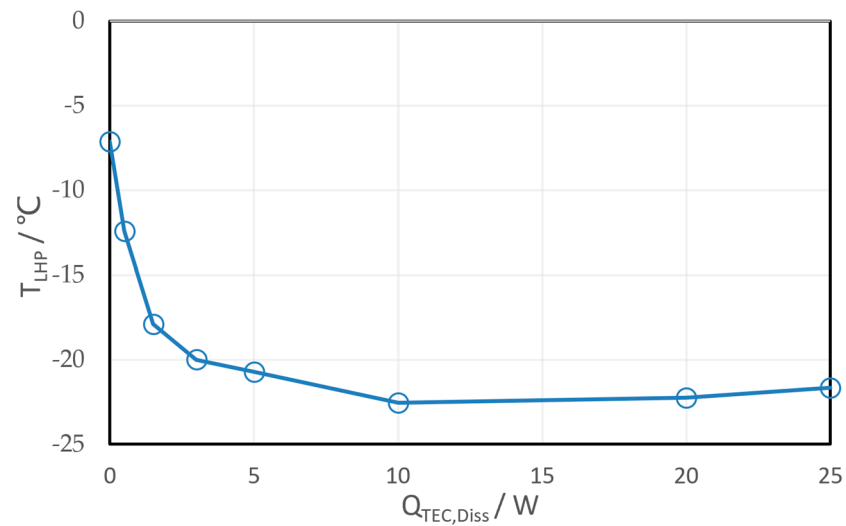


Figure 15. Influence of TEC power on LHP temperature.

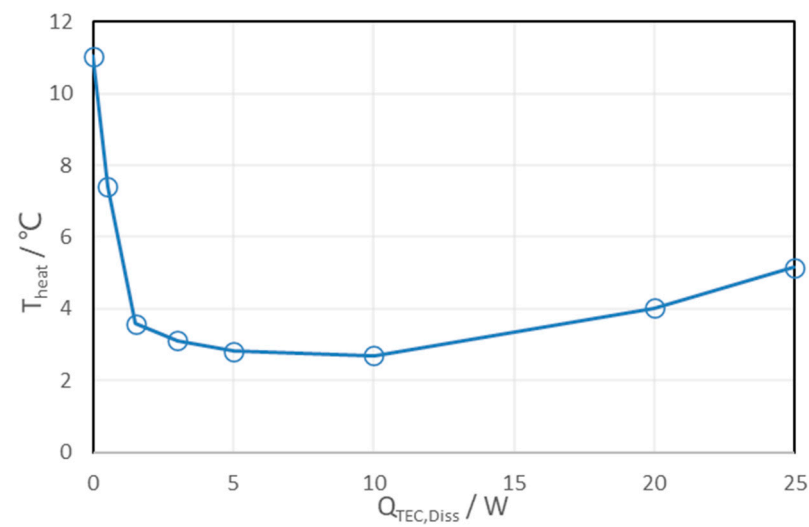


Figure 16. Influence of TEC power on heat source temperature.

When the TEC power is less than 3 W, it has a clear influence on the system—the cooling effect is greatly strengthened. When the TEC power is in the range of 3–10 W, the cooling effect of the system is further improved. When the TEC power is greater than 10 W, the temperature of the system does not decrease but gradually increases.

The test data show that when the TEC power was 10 W, the operating temperature of the system was the lowest and the cooling effect of the system was the best. When the TEC power was >10 W, although the thermal resistance of the system decreases slightly, the increased power has also become a heat burden to the system, and the temperature of the system increased. Therefore, 10 W was selected as the input power of the TEC in the subsequent ultra-LEO mission.

5.2. Heat Protection and Temperature Tolerance Test

In the ultra-LEO variable attitude mission, the side circular radiator will become the windward surface, and the temperature will rise sharply under the influence of aerothermal flux. There are some support rods between the external circular radiator and the interior of the system, as well as the pipe of the LHP (stainless steel pipe with 3 mm outer diameter and 0.5 mm wall thickness). Special heat protection and temperature tolerance tests were conducted to test the impact of the high-temperature external radiator on the internal device and to evaluate the LHP's tolerance to high-temperature environments.

The radiator dissipated heat energy into a cold and dark environment through radiation heat transfer, and the result of the energy balance was a return temperature. In the test, the temperature of the external radiator rapidly increased from 10 °C to 153 °C within 10 min. The test results in Figure 17 show that even though the temperature difference between the inside and outside was about 137 °C, the internal temperature of the system only had a small change of no more than 1.5 °C, which met the use requirements of the system.

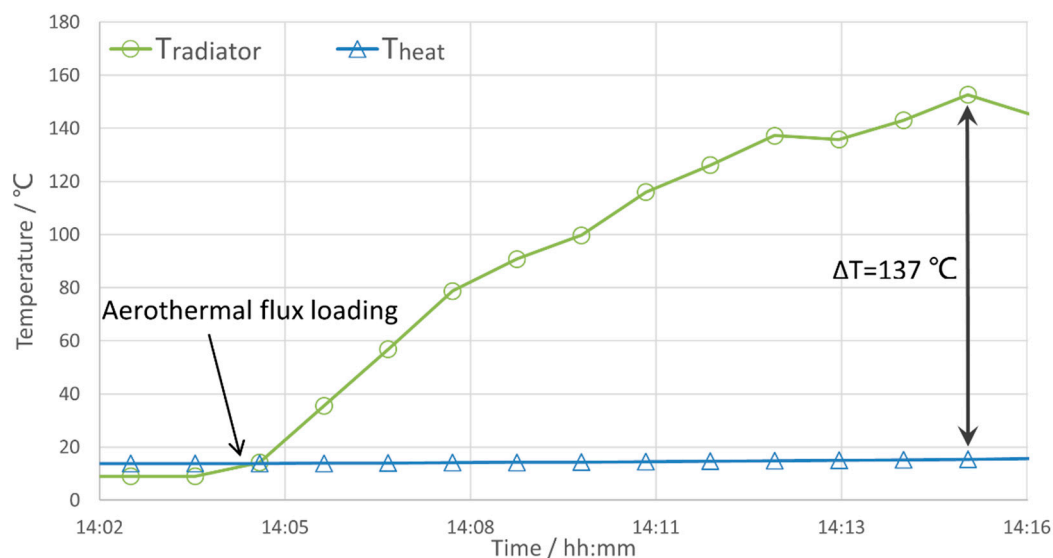


Figure 17. Temperature curve of system thermal protection effect.

After experiencing a high temperature test of about 150 °C, the LHP system did not show signs of high pressure damage or deformation. Because the LHP has a two-phase working medium inside, the pressure anywhere inside the LHP is consistent with the pressure inside the CC. The temperature of the CC inside the system is always lower than 20 °C, and the saturation pressure of 20 °C ammonia is 0.86 Mpa, so the internal pressure of the pipe on the radiator is always <0.86 MPa. The LHP system does not have the risk of high pressure due to aerothermal flux.

The test of heat protection and temperature resistance verified that the system designed in this paper can effectively prevent aerothermal flux.

5.3. LHP Startup Response and Cooling Test

In the ultra-LEO mission, the thermal management system was required to respond quickly and cool quickly during the mission interval to provide a good initial temperature for the next mission.

Based on the previous test results, the TEC power supply was set to 10 W to test the response and cooling effect of the system. The test data in Figure 18 show that when the TEC is turned on, the LHP can start quickly—within 1.5 min—and the return liquid pipe transfers the cooling source of the radiator to the internal system. Within only 10 min, the operating temperature of the LHP decreased from 11.1 °C to −2.5 °C, a decrease of 13.6 °C. Under the influence of LHP cooling, the temperature of the heat source inside the system decreased by around 6 °C.

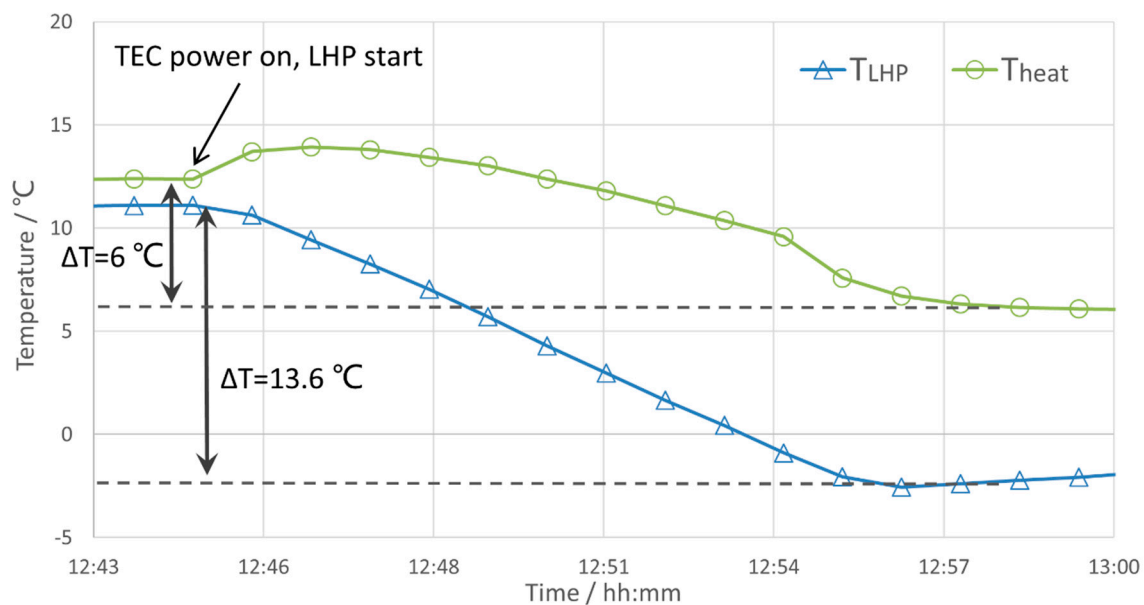


Figure 18. Cooling curve during system operation.

The test shows that the cooling system coupled with TEC and LHP responds quickly and can efficiently complete the internal cooling requirements of the system.

5.4. Whole Process Thermal Test of the Ultra-LEO Mission

The whole process test conditions at the system level were in accordance with the timing of the actual on-orbit mission. A total of seven vertical attitude missions were planned at an orbital altitude of 160–120 km, and the interval time of each vertical attitude mission was determined according to the calculation time of the natural decay of the orbit altitude. One vertical attitude mission was carried out at orbit altitudes of 160, 150, 140, and 130 km. When the orbit altitude was reduced to 120 km, the altitude was maintained by the propulsion system, and three vertical attitude missions were carried out.

Figure 19 shows the test results of the whole process of the ultra-LEO task, including the temperature curve of the top surface and the radiator. The test data show the influence of orbit altitude change and attitude change on temperature, reflecting the harsh high-temperature environment caused by the aerothermal flux.

The test results show that with the decrease of the orbit altitude, the aerothermal flux increases, and the top surface temperature rises clearly in a step shape with the decrease of orbit altitude, from the initial −40 °C to the highest 350 °C. The side radiator was only affected by aerothermal flux for a short time during the vertical attitude mission. As the aerothermal flux at 160–140 km was not large, the radiator temperature rise was not obvious. With the further reduction of orbit altitude, aerothermal flux increased significantly, and

the maximum temperature of the side radiator reached around 60 °C at 130 km and 160 °C at 120 km.

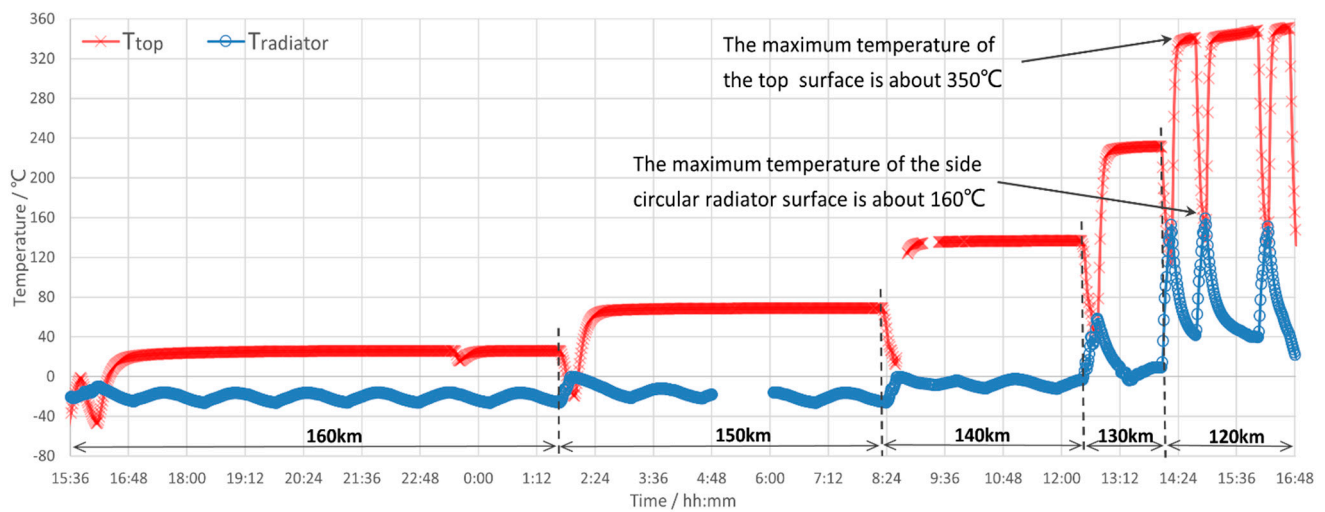


Figure 19. The outer wall temperature curve of the entire ultra-LEO mission.

Figure 20 shows the temperature curves of the LHP and the heat source inside the system at an orbit altitude of 130–120 km with the harshest aerothermal environment. With the help of the TEC, the LHP can be quickly started and cooled for each vertical attitude mission, greatly reducing the operating temperature of the LHP, and the temperature of the internal heat source also decreases slightly due to the cooling of LHP operation. During the entire ultra-LEO mission, the maximum temperature of the internal heat source was only 22.5 °C, which met the temperature requirements of the optical payload.

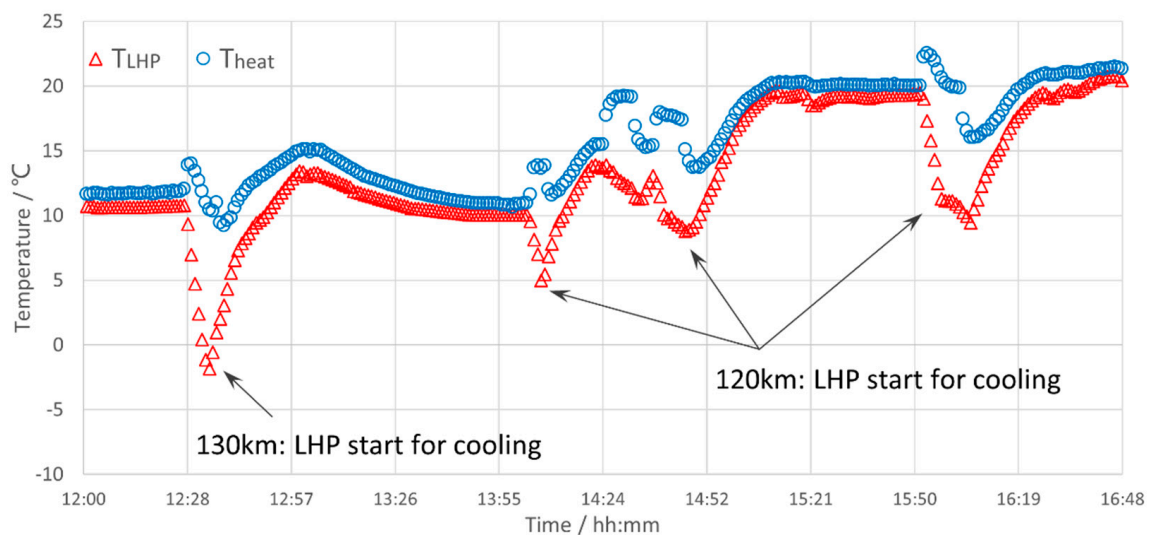


Figure 20. Internal temperature control effect at 130–120 km.

Throughout the whole process of the thermal test of the ultra-LEO mission, the thermal management technology, which combined aerothermal flux protection and heat load dissipation, proved to be correct and feasible, and the test data at system level were used for further optimization.

6. Conclusions

The ultra-LEO satellite needs to face the harsh aerothermal environment, and because of the special mission needs, the satellite also carries out complex attitude flight changes in

the aerothermal environment. In this study, an LHP coupled with a TEC was applied to the system, which not only prevented aerothermal flux and heat dissipation but also met the need of fast response. We calculated the aerothermal flux on the surface of the ultra-LEO satellite and conducted extensive experimental research on TEC control, system operation characteristics, and temperature control effect, resulting in the following conclusions.

1. When the orbital altitude is below 160 km, the surface of the satellite will produce non-negligible aerothermal flux. When the orbital altitude is 120 km, the aerothermal flux on the windward side of the satellite will reach 5100 W/m^2 .
2. TEC low power supply can greatly enhance the cooling effect of the system. At a TEC power supply power of 10 W, the system operating temperature dropped to its lowest—if the TEC power further increased, the system temperature began to gradually rise.
3. The heat protection test showed that when the temperature difference between the external radiator and the internal temperature was as high as 137°C , the internal temperature of the system showed only a small rise of 1.5°C , so the system has a good heat protection effect.
4. The LHP coupled with the TEC can start quickly within 1.5 min. The LHP operating temperature can be reduced by 13.6°C within only 10 min, and the temperature of the heat source inside the system can be reduced by about 6°C .
5. The system has passed the assessment of the whole aerothermal test process even if the top temperature rose to 350°C and the temperature of the side radiator rose to 160°C , the temperature of the internal heat source can be controlled within 22.5°C through the efficient work of the variable switch system for heat protection and dissipation.

Based on these findings, it is believed that the LHP coupled with the TEC thermal control system can complete the rapid switch of heat protection and dissipation functions in the harsh ultra-LEO aerothermal environment, which maintains the internal equipment at a very safe temperature. This study confirms the correctness and feasibility of the system and provides an important reference for the subsequent actual on-orbit mission.

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Nomenclature

k	Boltzman's Constant, J/K
M	Mach Number
N	Number of molecules per unit volume, mol/m^3
n	Number of molecules striking unit surface area per unit time, $\text{mol}/(\text{m}^2.\text{s})$
P	Pressure, Pa
q	Heating flux, W/m^2
Q	Heat load, W
R	Thermal resistance, K/W
S	Molecular speed ratio
T	Temperature, $^\circ\text{C}$

v	Velocity, m/s
CC	Compensation chamber
LHP	Loop heat pipe
ρ	Density
γ	Specific heat ratio
α	Coefficient accommodation
θ	Angle of incidence, °
Subscripts	
∞	Free-stream
e	Evaporator
w	Wall
heat	Heat source equipment
sink	Heat sink

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