Postprint of Thermal Model for Spacecraft Propellant Mass Measurement Based on Thermal Response Method

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Abstract

This paper, in the context of measuring remaining propellant mass in spacecraft tanks under microgravity conditions using the thermal response method, establishes an integrated thermal analysis model that accounts for coupling effects between the internal and external thermal environments of spacecraft tanks. By treating the external thermal environment of the spacecraft tank as a second-type floating thermal boundary condition, decoupled thermal analysis calculations under gas-liquid two-phase distribution are achieved, providing an accurate temperature field calculation method for the thermal response method. Employing this method, numerical simulations for the on-orbit phase are conducted to obtain the temperature field distribution required for measuring propellant mass inside a spacecraft tank under microgravity conditions using the thermal response method. The temperature field distribution under integrated coupling of internal and external thermal environments of the tank during thermal response heating operations is acquired, and the remaining propellant mass is retrieved based on transient temperature variations at specific monitoring points. The study reveals that when measuring propellant mass using the thermal response method, the tank temperature field is influenced not only by internal heating but also significantly by the on-orbit external thermal environment, which notably affects the uniformity of tank wall temperatures.

Full Text

A Thermal Model for Spacecraft Propellant Mass Measurement Based on the Thermal Response Method

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Abstract: This paper addresses the measurement of remaining propellant mass in spacecraft tanks under microgravity conditions using the thermal response method. A comprehensive thermal analysis model is established that couples the internal and external thermal environments of spacecraft propellant tanks. By treating the external thermal environment as a second-type floating thermal boundary condition, the thermal analysis is decoupled from the gas-liquid two-phase distribution calculations, providing an accurate temperature field computation methodology for the thermal response method. Using this approach, the temperature field distribution under coupled internal and external thermal environments during thermal response heating is obtained through numerical simulation for the on-orbit phase. The remaining propellant mass is then determined through inverse calculation based on the transient temperature variations at specific monitoring points. The study reveals that when using the thermal response method for propellant mass measurement, the tank temperature field is influenced not only by internal heating but also significantly affected by the external thermal environment in orbit, which impacts the temperature uniformity of the tank wall.

Keywords: Thermal response method; microgravity; propellant mass; temperature field
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0 Introduction

With the continuous development of China' s space technology, various liquid propellants have been widely applied. Since the amount of propellant directly affects spacecraft lifetime, accurately estimating the remaining propellant mass in tanks during flight is crucial. While mature technologies exist for liquid mass measurement on the ground, measuring liquid volume under microgravity conditions presents significant challenges due to environmental constraints. In microgravity or zero-gravity conditions, conventional methods relying on gravitational acceleration and buoyancy—such as weighing and static pressure differential methods—cannot be applied. Furthermore, because surface tension dominates in microgravity and tanks contain complex internal structures, the liquid position becomes unstable, making ground-based measurement techniques such as radiation and optical methods difficult to implement.

Since the 1960s, more than a dozen measurement techniques for on-orbit propellant gauging have been proposed, among which the thermal response method offers very high accuracy when propellant remaining quantity is low, enabling precise determination of remaining propellant at the end of spacecraft life. The thermal response method determines propellant mass by analyzing the transient thermal response under microgravity conditions. The measurement mechanism involves applying a specific heat flux to the tank during operation. Different remaining propellant masses result in different gas-liquid distributions within the tank, leading to distinct thermal responses when the same heat flux is applied. Based on these differences, inverse calculations are performed to obtain the remaining propellant mass.

Traditional thermal response analysis only considers the influence of gas-liquid distribution on tank thermal response under microgravity conditions. However, under actual on-orbit conditions, spacecraft and tanks continuously exchange heat with the external environment, and these complex thermal interactions inevitably affect the thermal response of the tank surface. This paper establishes a comprehensive thermal analysis model that couples the internal and external thermal environments of satellite tanks for thermal response method applications. By treating the external thermal environment as a second-type floating thermal boundary condition, the model considers both the influence of internal gas-liquid distribution on thermal response and the effects of external environmental thermal interactions under on-orbit conditions.

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1 Theoretical Model

To obtain accurate thermal response of the tank, three main calculations are required: first, determining the gas-liquid two-phase distribution under microgravity for different remaining propellant masses; second, establishing a complex heat exchange model between the spacecraft interior and external environment under on-orbit conditions; and finally, coupling the external heat exchange as boundary heat flux with the microgravity gas-liquid distribution to simulate the thermal response during thermal response method heating. The coupled computation process is illustrated in [Figure 2: see original paper].

Under on-orbit conditions, spacecraft internal and external components undergo complex heat exchange with the space environment, as shown in the heat transfer model schematic [Figure 1: see original paper]. The fundamental governing equation for two-phase flow is the momentum equation. For simulating the gas-liquid two-phase distribution of propellant under microgravity conditions, ANSYS FLUENT is employed using the Volume of Fluid (VOF) model from the finite volume method. By introducing the volume fraction variable _q for phase q, the phase interface is tracked by solving continuity equations for volume fractions of single or multiple phases. For phase q, the equation is:

$$\frac{\partial \alpha_q}{\partial t} + \mathbf{v} \cdot \nabla \alpha_q = \frac{S_{\alpha_q}}{\rho_q} + \frac{1}{\rho_q} \sum_{p=1}^{n} (\dot{m}_{pq} - \dot{m}_{qp})$$

where $\dot{m}_{pq}$ represents mass transfer from phase p to phase q, $\dot{m}_{qp}$ represents mass transfer from phase q to phase p, and S_{α_q} is the source term. The calculation of the primary phase volume fraction is based on the constraint:

$$\sum_{q=1}^{n} \alpha_q = 1$$

For calculating heat exchange between the spacecraft and external environment under on-orbit conditions, SINDA/FLUENT is used. By specifying on-orbit operating conditions and orbital parameters, a thermal model of the spacecraft under on-orbit conditions is established to calculate heat distribution. The primary computational methods are the thermal network method and Monte Carlo method. The basic governing equation for the thermal network method is the nodal energy balance equation:

$$C_i \frac{dT_i}{dt} = Q_{cond} + Q_{conv} + Q_{rad} + Q_{int}$$

where Q_{conv} represents convective heat transfer between surfaces, Q_{rad} represents radiative heat transfer between surfaces, and Q_{int} represents internal heat sources from equipment. The Monte Carlo method for calculating radiative heat transfer in large complex systems uses the equation:

$$\Phi_{i,j} = A_i F_{i,j} \sigma (T_i^4 - T_j^4)$$

where $\Phi_{i,j}$ is the radiative heat flux from surface A_i to surface A_j [W], A_i and A_j are areas [m^2], and $F_{i,j}$ is the radiative transfer factor from A_i to A_j.

The external thermal influence on the tank is output in the form of boundary heat flux. Subsequent coupled calculations continue to use ANSYS FLUENT software to obtain the temperature response at specific points within the tank.

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2 Numerical Simulation of Propellant Gas-Liquid Distribution Under Microgravity

For numerical simulation, the first step is to establish the internal geometry model and mesh file of the tank. [Figure 3: see original paper] shows the geometric model of the propellant system. Under microgravity conditions, surface tension dominates liquid behavior. To ensure continuous propellant delivery, propellant management devices (PMDs) are typically employed inside tanks, which significantly influence gas-liquid distribution. Therefore, the tank model includes a simplified management device.

ANSYS CFD ICEM is used to create an unstructured mesh file, as shown in [Figure 4: see original paper]. To ensure calculation accuracy, a high mesh density is employed with 818,394 volume cells, 1,689,015 surface cells, and 164,340 nodes.

Using monomethylhydrazine (MMH), a common liquid propellant, the two-phase distribution under zero-gravity acceleration is simulated for remaining propellant masses of 60 kg and 40 kg. [Figure 5: see original paper] and [Figure 6: see original paper] show the liquid-phase volume fraction distributions inside the tank for remaining propellant masses of 60 kg and 40 kg, respectively, where a value of 1 indicates liquid-phase propellant and 0 indicates gas-phase. Under the influence of the propellant management device, liquid propellant concentrates primarily in the lower portion of the tank with a concave liquid surface. The propellant climbs upward along the liquid guide channels, submerging the lower hemispherical surface.

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3 Coupled Thermal Response Simulation

Since the thermal response process inside the tank is a thermal action implemented under on-orbit conditions, the influence of the spacecraft must be considered in real-time during tank thermal simulation. The spacecraft is affected by external heat flux and internal power dissipation, causing the environment around the tank to change continuously. Taking a geosynchronous satellite as an example, SINDA/FLUENT is used to simulate the internal thermal environment of the tank under on-orbit conditions. Using a geosynchronous orbit and calculating heat exchange on the summer solstice, the surface heat flux distribution of the tank under on-orbit conditions is obtained and output as a floating heat flux boundary.

The surface heat flux distribution obtained under on-orbit conditions is extracted and coupled with the previously calculated internal propellant distribution. Using the surface heat flux distribution as the external boundary condition, the thermal influence of the on-orbit environment on the tank is calculated. Simultaneously, the heating process of the thermal response method is simulated by applying thermal response heating flux at specific locations on the tank surface while monitoring temperature responses at designated detection points. Typical thermal response method heating durations range from 10–100 hours; this study employs a 25-hour heating period.

[Figure 7: see original paper] shows the temperature distribution in the tank after 25 hours of thermal response method heating for remaining propellant masses of 40 kg and 60 kg. The results demonstrate that under external heat flux influence, the internal temperature distribution becomes non-uniform. At 60 kg remaining propellant mass, temperatures are generally lower than at 40 kg. By monitoring temperature values at specific points on the tank wall, the thermal response can be characterized. [Figure 8: see original paper] shows the temperature response curve at a wall monitoring point during thermal response method heating.

The temperature at the monitoring point for 60 kg remaining propellant mass is consistently lower than for 40 kg. Throughout the 25-hour heating period, the wall monitoring point temperature increases significantly. For 60 kg remaining propellant, the maximum temperature rise is approximately 70°C, while for 40 kg remaining propellant, the maximum temperature rise reaches 170°C. The selection of wall monitoring point location is critical for the thermal response method. When remaining propellant mass is low, the liquid film thickness attached to the wall due to capillary action is thinner, resulting in greater temperature rise under thermal response heating flux. It is particularly important to note that heating elements must be placed in liquid-covered regions; placement in gas-phase regions could cause temperature to rise sharply and potentially damage the tank wall.

Despite only a 20 kg difference in propellant mass, the temperature response at monitoring points shows significant variation. The thermal response method establishes a database of temperature responses at monitoring points for different remaining propellant masses to estimate actual remaining mass. Larger temperature response differences indicate higher method accuracy, demonstrating that the thermal response method can achieve high precision when remaining propellant mass is low.

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5 Conclusions

This paper develops an integrated analysis method for measuring remaining propellant mass in tanks under microgravity using the thermal response method, simultaneously considering both internal gas-liquid two-phase distribution and on-orbit thermal boundary conditions. The tank is modeled to simulate internal gas-liquid two-phase distribution under microgravity conditions. The thermal influence of the external environment under on-orbit conditions is calculated and coupled with the internal two-phase distribution in the form of boundary heat flux to simulate the heating phase of the thermal response method, obtaining temperature responses at specific monitoring points for different remaining propellant masses.

The results show that the on-orbit external environment significantly affects the surface temperature distribution of the tank. Under external heat flux, the tank surface temperature distribution becomes non-uniform. Temperature responses at tank wall monitoring points are obtained for remaining propellant masses of

40 kg and 60 kg. The temperature for 40 kg remaining propellant is consistently higher than for 60 kg, with temperature differences reaching nearly 100°C. Despite only a 20 kg difference in propellant mass, the temperature response gap at monitoring points is substantial, indicating that the thermal response method will achieve good measurement accuracy when remaining propellant mass is low.

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Note: Figure translations are in progress. See original paper for figures.