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

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Abstract

In the context of using the thermal response method to measure remaining propellant mass in spacecraft tanks under microgravity conditions, this paper establishes a comprehensive thermal analysis model for coupled internal and external thermal environments. By treating the external thermal environment of the spacecraft tank as second-type floating thermal boundary conditions, decoupled thermal analysis calculations are achieved under gas-liquid two-phase distribution, providing an accurate temperature field calculation method for the thermal response method. Employing this method, numerical simulations are conducted for the on-orbit phase to obtain the temperature field distribution required for measuring propellant mass inside a spacecraft tank under microgravity conditions, capturing the overall coupling between internal and external thermal environments during thermal response heating operation. Based on transient temperature variations at specific monitoring points, the remaining propellant mass is retrieved through inversion. 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 on-orbit external thermal environment, which impacts the uniformity of tank wall temperature.

Full Text

Preamble

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 within the tank, providing an accurate temperature field calculation methodology for the thermal response method. Using this approach, the temperature field distribution under coupled internal and external thermal environments during thermal response heating operations in the on-orbit phase is obtained through numerical simulation for a specific spacecraft tank. The remaining propellant mass is then determined based on the transient temperature variations at designated monitoring points. The study reveals that during propellant mass measurement using the thermal response method, the tank temperature field is influenced not only by internal heating but also significantly affected by the in-orbit external thermal environment, which notably impacts the uniformity of tank wall temperatures.

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

0 Introduction

With the continuous advancement of China' s aerospace technology, various liquid propellants have been widely employed. Since the amount of propellant directly correlates with spacecraft lifespan, accurately estimating the remaining propellant mass within tanks during flight is critically important [?]. While mature technologies exist for liquid mass measurement on the ground, measuring liquid volume under microgravity conditions presents significant challenges due to the unique environmental constraints. Conventional methods relying on gravitational acceleration and buoyancy, such as weighing and hydrostatic pressure differential techniques, are inapplicable in microgravity or zero-gravity environments. Furthermore, because surface tension dominates in microgravity and tanks contain complex internal structures, the liquid position becomes unstable, rendering ground-based measurement technologies such as radiation and optical methods impractical [?].

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

Traditional thermal response method analysis only considers the influence of gas-liquid distribution under microgravity on tank thermal response. 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 tank surfaces. 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 accounts for both the influence of internal gas-liquid distribution on thermal response and the effects of external environmental thermal interactions under on-orbit conditions.

1 Theoretical Model

Under on-orbit conditions, spacecraft components exchange heat complexly with the space environment. The heat transfer model is illustrated in [Figure 1: see original paper]. The fundamental governing equation for two-phase flow is the momentum equation [?]:

$$\nabla \cdot (\alpha_q \rho_q \vec{v}_q) = \sum_{p=1}^{n} (\dot{m}_{pq} - \dot{m}_{qp}) + S_q$$

ANSYS FLUENT is employed to simulate the gas-liquid two-phase distribution of propellant under microgravity conditions using the Volume of Fluid (VOF) model from the finite volume method. By introducing the volume fraction variable α_q for phase q, the phase interface is tracked by solving continuity equations for the volume fraction of single or multiple phases. For phase q, the equation is as follows [?]:

$$\frac{\partial \alpha_q}{\partial t} + \vec{v} \cdot \nabla \alpha_q = \frac{S_q}{\rho_q}$$

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_q is the source term. The calculation of the primary phase volume fraction is based on the constraint:

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

The heat exchange between the spacecraft and external environment under on-orbit conditions is simulated using SINDA/FLUENT. 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 employed are the thermal network method and Monte Carlo method. The fundamental governing equation for the thermal network method is the energy conservation equation for nodes:

$$C_i \frac{dT_i}{dt} = \sum_j Q_{cond,ij} + \sum_j Q_{conv,ij} + \sum_j Q_{rad,ij} + Q_{int,i}$$

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 calculates radiative heat exchange in large complex systems using:

$$Q_{i\to j} = A_i \cdot RD_{i\to j} \cdot \sigma(T_i^4 - T_j^4)$$

where $Q_{i\to j}$ is the radiative heat flux from surface i to surface j [W]; A_i and A_j are areas [m^2]; and $RD_{i\to j}$ is the radiation 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.

2 Numerical Simulation of Propellant Gas-Liquid Distribution Under Microgravity

In numerical simulation, the first step involves establishing the internal geometry model and mesh file for the tank. [Figure 3: see original paper] shows the geometric model of the propellant tank system. Under microgravity conditions where surface tension dominates, propellant management devices are typically employed within tanks to ensure continuous propellant delivery. These devices significantly influence the gas-liquid distribution, so the tank model includes a simplified management device.

ANSYS CFD ICEM is used to generate 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 face cells, and 164,340 nodes. Using monomethylhydrazine (MMH) as the working fluid, simulations are performed under zero-gravity acceleration conditions for remaining propellant masses of 60 kg and 40 kg. [Figure 5: see original paper] and [Figure 6: see original paper] illustrate the liquid-phase volume fraction distributions within the tank for remaining propellant masses of 60 kg and 40 kg, respectively, where a value of 1 represents liquid-phase propellant and 0 represents gas-phase. The results show that 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.

3 Coupled Thermal Response Simulation

Since the thermal response process within the tank is a thermal operation conducted under on-orbit conditions, the influence of the spacecraft must be considered in real-time during tank thermal simulation. The spacecraft environment around the tank changes continuously due to external heat flux and internal power dissipation. Using a geostationary satellite as an example, SINDA/FLUENT simulates the internal thermal environment of the tank under on-orbit conditions. A geostationary orbit is adopted, and heat exchange between the tank and environment is calculated for the summer solstice. The resulting surface heat flux distribution under on-orbit conditions is extracted and output as a floating thermal boundary condition.

The tank surface heat flux distribution obtained under on-orbit conditions is 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 modeled by applying the thermal response heating flux at specific locations on the tank surface while monitoring temperature responses at designated detection points. The heating duration for thermal response method implementation typically ranges from 10 to 100 hours; this study employs a 25-hour heating period.

[Figure 7: see original paper] presents the temperature distributions after 25 hours of thermal response heating for remaining propellant masses of 40 kg and 60 kg. The results demonstrate that under the influence of external heat flux, the internal temperature distribution within the tank exhibits non-uniformity. Temperatures are generally lower for the 60 kg case compared to the 40 kg case. Temperature values at specific points on the tank wall are monitored to characterize the tank thermal response. [Figure 8: see original paper] shows the temperature response curves at a wall monitoring point during thermal response heating. The data indicate that temperatures for the 60 kg remaining mass case are consistently lower than those for the 40 kg case. Throughout the 25-hour heating period, the wall monitoring point exhibits significant temperature increases–approximately 70°C maximum for the 60 kg case and 170°C maximum for the 40 kg case.

The selection of wall monitoring point location is critical for the thermal response method. When remaining propellant mass is low, the liquid film thickness adhering to the wall due to capillary action is thinner, resulting in greater temperature rise under thermal response heating flux. It is particularly important to position heating elements within liquid-covered regions; placement in gas-phase regions would cause temperature to rise sharply, potentially damaging the tank wall. The results show that a mere 20 kg difference in propellant mass produces significant differences in temperature response at monitoring points, demonstrating that the thermal response method can achieve high measurement accuracy when remaining propellant mass is low.

5 Conclusions

This paper establishes an integrated analysis method for measuring remaining propellant mass in tanks under microgravity using the thermal response method, simultaneously considering both the internal gas-liquid two-phase distribution and on-orbit thermal boundary conditions. Tank modeling and simulation calculations are performed for internal gas-liquid two-phase distribution under microgravity and external environmental thermal effects under on-orbit conditions. The external heat flux is coupled with the internal two-phase distribution as a boundary condition to simulate the heating phase of the thermal response method, obtaining temperature responses at specific monitoring points for different remaining propellant masses.

The results demonstrate that on-orbit external environmental conditions influence tank surface temperature distribution, creating non-uniformity under external heat flux. Temperature responses at tank wall monitoring points are obtained for remaining propellant masses of 40 kg and 60 kg, with temperatures for the 40 kg case consistently higher than those for the 60 kg case—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 favorable measurement precision when remaining propellant mass is low.

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