Plume Temperature Measurements of a Spin-Stabilized Rocket

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ABSTRACT

Rocket plume temperature distribution plays an important role on the heat transfer analysis of launch carriers. Thus, a non-intrusive IR scanner is used in this paper to analyze the influence of spin motion on plume centerline temperature measurements of spinning solid-propellant rockets. Results show that the average error of temperature measurement is below 3.5% even with a 10% error of estimated emittance value. At the early burning stage under no-spin condition, the rocket plume contains a high percentage of flammable gas and is cooled by the vortex of ambient air, which is induced by the transient plume jet of the rocket. Hence, the plume at the nozzle exit has a low temperature during the early burning stage. After a short period, high-temperature particles apparently are ejected and emit high-intensity IR that causes the increase of plume temperature at the nozzle exit. Regardless of plume temperature level at the nozzle exit and the existence of rocket spin, the plume temperature for secondary combustion of flammable gas is consistent, and the average temperature error is less than 6 percent. This study also offers an economical and practical method for the study of plume temperature distribution of a spin-stabilized rocket.

Introduction

The radiation property of rocket plumes is an important parameter for the analysis of radiation heat transfer in the rocket motor burn phase, because the radiation is the main heat transfer mode in space. Hence, in the development process of spinning solid-propellant rockets, rocket plume temperature measurement and radiation property studies are important. They are also helpful in increasing the ability of infrared (IR) detection and launch system development. Research on heat insulation of launch carriers, such as the space shuttle and satellites, are very important because the thermal radiation emitted from the rocket plume causes electronic equipment in the carrier or satellite become risky during rocket motor burn phase. Furthermore, the radiation property study of rocket plumes also plays an important role in the R&D of insulators and in the heat transfer analysis of the nozzle surface. Nevertheless, for the analysis of radiation characteristics, the rocket plume temperature must be measured. These motivations have inspired R&D on intrusive and non-intrusive technology of flame and/or combustion gas temperature measurement in recent years.

Key Words: rocket plume, infrared, spinning solid-propellant rockets, thermal radiation, plume emissivity

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There are many non-intrusive methods for measuring temperature. An Infrared scheme has some advantages and disadvantages in comparison to other methods. First, it does not disturb the flow field, and it is easy to make accurate calibrations using a blackbody source whose temperature is known. Second, the optical scanning system can be either mechanical or electrical, as long as it is able to measure the temperature distribution at one point, along one line or on a surface. If high-resolution detectors and high-speed scanning mechanisms are used, the unsteady, two-dimensional temperature of solid surfaces and liquid surfaces can be detected. Third, in accordance with the optical setup, the temperature of a small or large object can be measured, and the distance between object and scanner can be changed. The measured temperature range can be arbitrarily chosen as long as the electronic equipment is appropriately arranged. Fourth, the operation is convenient for industrial applications, and the cost of an IR system is cheaper than other types of non-intrusive methods, such as LIF and CARS, although the latter two methods can yield temperature at a point without assumptions of flame gas geometry. The drawback of the IR system is that the medium between the object and scanner absorbs and scatters the IR intensity, which may in turn affect the results of IR measurements. Therefore, the appropriate wavelength for the detector must be carefully selected. The window material is very special and the cost of manufacture is expensive. The emissivity of the object must be calculated.

The results of previous studies show that if the wavelength is in the range of CO$_2$ species emission, measurement of flame temperature by IR brightness thermometry is very accurate. The traditional infrared pyrometer must use an external blackbody source for calibration in temperature measurement and an external laser for optical path alignment. Owing to the above inconveniences, this research adopts a new infrared temperature measurement system for rocket plume temperature measurement. The calculation theory of the IR system follows the Kirchoff and Planck Laws. Both reflections of surrounding temperature on the object and the absorption and emission of the atmosphere are considered.

Many chemical species are created in the burning gas of the solid propellant, including H$_2$, CO, and HCL, where H$_2$ and CO species react in oxygen to form H$_2$O and CO$_2$ species, and the HCL species reacts in oxygen to form H$_2$O and Cl$_2$ species, while the burning gas is ejected from the nozzle. Thus, the analysis of solid propellant burning gas shows that carbon dioxide is the main species that has a particularly strong effect on the thermal radiation in the rocket plume. To improve properties of the solid propellant, metallic and/or nonmetallic particles are added. Therefore, solid-propellant rocket plume compositions become complex when the motor burns, and there are strong effects on the plume emissivity prediction.

The field of thermal radiation in rocket plumes is located on the short band of IR. At this near infrared region, the observation data and physical characteristic knowledge are sufficient for the thermal radiation study of rocket plumes. Reasons for this are: (1) The temperature distribution of solid propellant rocket plume is from 2000 to 2500 K, and the Planck blackbody spectral distribution peaks at this temperature are located on the near infrared region; (2) Infrared detection measurement techniques are well-developed for works in this IR region; (3) The emission band of the rocket plume lies in the near infrared region due to the asymmetric stretch carbon vibration of carbon dioxide molecules that are the main species of solid propellant rocket plumes. Hence, the analysis of rocket plume temperature distribution at spin conditions in this work is conducted using the IR scanner with the filter band in the near infrared region.

**EQUIPMENT SETUP AND PROCEDURE**

A solid-propellant rocket motor test stand is employed in the present study, as shown in Fig. 1. The test stand is manufactured by stainless steel to avoid corrosion caused by the rocket plume. Four supporting stainless steel tubes are used as the main platform support bracket. The main platform contains a gimbals arrangement, two servomotors, a gearbox, a spin disk, and a linkage. The digital servomotor rests upon the gimbals arrangement, and the high-speed slip ring is under the gimbals. In order to keep the test stand level, a circular level vial is set on top of servomotor. The shaft of the servomotor connects a spin shaft with a coupling. The spin shaft is made of
high-carbon heat-treated stainless steel to avoid distortion. The surface of the spin shaft incorporates a layer of chromium coating to increase hardness of the shaft.

The test stand employs a 0.5 HP DC digital servomotor as the driver of the rocket spin motion and uses an SNC series control board to drive the servomotor. The spin rate can be adjusted by use of an IBM compatible computer with SNC-912 software. The maximum spin rate of the servomotor is 3000 rpm. This test stand allows for testing at different spin rates due to the servomotor. In this study, three spin rates, 600, 960, and 1200 rpm, are adopted to satisfy the requirement for dynamic similarity. That is, the tangential velocity of the rocket model must be the same as that on the real motor. The outside diameter of the real motor is near 120 cm (the outside diameter of star-48 is 48 inches with a 50 rpm of spin rate), and the motor can run from 30 to 120 rpm. Since the subscale rocket is smaller than the real motor, the traveling time of particles in the combustion chamber of the subscale rocket is shorter. Thus, the spin rate of the subscale rocket must be increased.

A subscale solid-propellant rocket model is employed in the simulated experiment and hot fire test. The chamber pressure and thrust distribution curves versus burn time of the subscale rocket motor test are shown in Fig.2. The material of the rocket nozzle is made by highly dense graphite. The throat diameter is 7 mm with the half angles of the entrance are 60 degrees against the center line of the nozzle. The expansion half angle is 15 degrees. Manufactured by a CNC (computer numerical control) lathe, these nozzles have highly precise dimensions. Each nozzle is only used once because the throat diameter varies with particle accumulation during each burning test. For safety concern the nozzle is designed to automatically break at the location of step when the chamber pressure is higher than the designed value.

The main equipment of the infrared temperature measurement system, used to measure the centerline temperature of the solid propellant rocket plume, is produced by AGEMA. This system contains a 7\(^\circ\) (View angle) lens with a wavelength range of 2 ~ 5.6 \(\mu\)m and a main controller with an external trigger device that can be used to select measurement range and system resolution. The scanner is a measurement device that can accept infrared radiation from the object. Two infrared black body sources built inside the scanner body are used as reference light sources to increase the IR system accuracy. There are two scanning types of scanner: a line type with 2500Hz scanning frequency, and a surface type with 25Hz scanning frequency. In normal conditions, the maximum measurement temperature is 773K, but the maximum measurement temperature can be increased to 2100K by use of a special filter. A personal computer with special software is used to analyze the thermal image during experimental measurements.

**RESULTS AND DISCUSSION**

**Emissivity Prediction**

Determining the radiation emissivity is an important step\(^{9,36}\) in the analysis of radiation of a solid propellant rocket plume. The calculation becomes even more complex, if the rocket plume contains metal particles, because such factors as particle size distribution\(^{1,5,10}\), refraction index (a function of the temperature and wavelength), scattering and absorption of the cross section area, radiation property of particles cloud\(^{16,17,22,32}\), the temperature and pressure distribution in the plume\(^{13,33,39}\) must be taken into account. It is very difficult to achieve these data. Hence, to enhance the measurement accuracy and to analyze the effect of emissivity predicted error on plume temperature measurement, two methods are used to obtain plume emissivity for the temperature determination of the rocket plume in this study.

**Gas Emittance**

As previous discussed, the solid-propellant rocket plume contains both burning gas and metal particles. To predict the rocket plume emissivity, the emittance of burning gas must be obtained
first. Since, a special filter is built into the IR scanner, only 4.3 µm wavelengths can pass the filter, so the emissivity of a rocket plume which contains metal particles at this band must be determined through other means. According to the study of Tien and Lee on luminous flame emissivity, the following equation can be used to calculate flame emissivity if the soot particles are assumed as a gray body in the luminous flame\textsuperscript{36}.

\[ \varepsilon = \varepsilon_g + \varepsilon_s - \varepsilon_g \times \varepsilon_s \quad (1) \]

where \( \varepsilon \) is flame emissivity, subscript \( g \) indicates gas, and \( s \) indicates soot particles. A comparison of the equation results with the experimental data by Markstein’s work showed that a good agreement could be achieved\textsuperscript{41}. Thus, the emissivity of a rocket plume that contains metal particles can be obtained by through this equation, if the emissivity of plume gas and metal particles can be obtained.

Previous researches on rocket plume composition indicate that the main gas species in plumes is \( \text{CO}_2 \), and a 4.3 µm wavelength is emitted from \( \text{CO}_2 \). This is due to the asymmetric stretch vibration of carbon dioxide molecules, if local thermodynamic equilibrium (LTE) and plume opaqueness are assumed\textsuperscript{14,34}, with IR scanners, the emissivity of carbon dioxide molecules can be taken as rocket plume emissivity. In the study of radiation emission, many models have been used to analyze band emissivity of carbon dioxide molecules, including the Elsasser model, Statistical model, Wide-Band model and others\textsuperscript{2,8,21,25}. The partial pressure and temperature of carbon dioxide gas must be obtained for these models. It is very difficult to obtain accurate data in solid propellant rocket plumes because the necessary reaction mechanism analysis is troublesome and laborious in solid propellants. In order to avoid a large amount of complex analysis and to acquire the emissivity of carbon dioxide molecules in rocket plume, professor Kuo et al used a FT-IR spectrometer to measure combustion gas emissivity of AP/HTPB solid propellant at a temperature of 1500 K and pressure of 15 psia. The emissivity value of 4.3 µm wavelength is about 0.107 in the experiment. Hence, in this study, the gas species emissivity of solid propellant rocket plumes adopts the value of 0.107, as obtained by professor Kuo.

**Emissivity of Metal Particles**

For the metal particle emissivity study, usually, an average property of particle clouds is used to replace the emissivity of metal particles because of the difficulty of obtaining metal particle size data. The calculation of particle cloud emissivity is complex. Thus, Plass\textsuperscript{11}, Bartky and Bauer\textsuperscript{5} assume on isotropic scattering and optically thick cloud to simplify the calculation as below:

\[ \varepsilon_p = \eta \left( \frac{\sigma_a}{\sigma_s} \right)^{1/2} \quad (2) \]

where subscript \( p \) indicates the particle cloud, \( \sigma_a \) is the absorption of the cross section, \( \sigma_s \) is the scattering of the cross section, \( \eta \) is a constant with a value of about 2.3. The data needed to calculate \( \sigma_a \) and \( \sigma_s \) are offered by the table of Bauer and Plass.

**Rocket Plume Emissivity**

According to the study of Bauer, the 4 µm band emissivity of metal particle clouds at 1973 K is 0.0303. After using extrapolation for the 4.3 µm band, emissivity of metal particle cloud could be obtained as 0.033. Finally, the equation offered by Tien is used and the emissivity of solid
propellant rocket plumes with metal particles could be achieved as 0.138. Another method to predict plume emissivity is adopted by Morizum et al. For which the 4.3 µm band emissivity of a solid propellant rocket plume is found to be 0.125 by integrating the radiant intensity spectral distribution diagram of a 4.3 µm band. Comparing values produced by Tien and Morium shows that the error of the rocket plume emissivity prediction is about 10 percent. However, the 10 percent prediction error of emissivity is acceptable in practical applications. In this paper, these two values of emissivity are used as the emissivity of rocket plume, and the effect of emissivity prediction error on the temperature measurement of a solid propellant rocket plume is investigated.

**Spin Effect**

A star-shaped grain is designed for the solid rocket. The performance of subscale solid propellant rockets is shown in Fig. 2, which indicate that chamber pressure and thrust in the ignition transient and aft-burning period are higher than the mean value of the test. This is the main characteristic of the star-shaped solid propellant rockets. The effect of estimated emittance value for plume center-line temperature measurement is shown in Fig. 3, where two types of estimated emittance are used to attain the temperature distribution of plumes under the no-spin condition at the start of motor burning. In this figure, the measured positions of IR thermal image are indicated. The curve distribution shows that the average error of temperature measurement is below 3.5% in spite of the 10% error of estimated value of emittance. Thus, the IR system is used to analyze the rocket plume temperature distribution with the single value of 0.138 as emittance for the simplified analysis in this study.

Figs. 4 to 7 represents the influence of spin rate on rocket plume temperature distribution. The data shown in Fig. 4 is rocket plume temperature distribution at various burning times under no-spin condition. At early stage, the rocket plume contains high-percentage of flammable gas cooled by the entrance vortex of cooling air that is induced by the transient plume jet of the rocket. Hence, the plume at the nozzle exit has a low temperature at early stage. When the rocket plume ejects into the atmosphere and mixes with the oxygen of the air, a secondary combustion of the exhausted flammable gas occurs. The plume temperature increases continuously due to the secondary combustion until the reaction is completed. At later burning times, the temperature of the nozzle exit clearly increase until it reached a constant temperature range of about 1000°C. This phenomenon is caused by melted metal particles that are produced by the combustion of propellant at high temperature. The high temperature particles apparently emit high-intensity IR that causes the plume temperature at the nozzle exit increaseing. Since the metal particles cause high temperatures at the nozzle, the temperature distribution curve presents the retroflex phenomena. This is due to the cooling effect of the mixing of high temperature particles and the cold air vortex which is induced by the plume jet. In order to keep thermal equilibrium, the cooled particles absorb the heat energy of the exhaust plume, and part of the heat energy is swept into the cold air vortex. For this reason, the plume temperature decreases abruptly while combustion gas is ejected from the nozzle and increases when the secondary combustion of the plume occurs.

The temperature distributions of the rocket plume shown in Figs. 5 to 7 are the results for various spin rates of 600, 960 and 1200 rpm, respectively. These figures demonstrate the influence of spin rate on plume temperature. In contrast to the plume temperature of the no-spin condition, the plume temperature at the nozzle exit of the spin case decreases, and are almost the same values as the initial burning stage. Thus, the retroflex phenomenon never occurs in the plume temperature distribution curve under the spin condition. The plume temperature of the nozzle exit is low because of the cooling effect of the cold air vortex and the lack of high
temperature particles, which are already accumulated inside the combustion chamber nozzle due to the spin behavior of the rocket.  

The IR thermal image of rocket plumes under spin and no spin conditions during various burn times are shown in Figs. 8 to 10 which illustrate the centerline temperature distribution of the rocket plume. As the thermal images reveal, the rocket plume has a larger expansion angle and higher centerline temperature under the no-spin condition in comparison with the measured data in spin condition. Fig. 8 indicates that the plume at the start burn has a low temperature at the nozzle exit because of the lack of high temperature particles, regardless of spin motion. In Figs. 9 and 10 after burning for a short period, the plume of a no-spin rocket contains large amounts of high-temperature particles, and the plume temperature is larger than that in the spin case. This is because particles accumulation in the nozzle portion for the spin effect, hence, the rocket plume contains small amounts of metal particles. Then, the cold air vortex enhances the cooling effect and the plume temperature of spinning rockets is reduced to a value below average. This is also a reason for the small ejection expansion angle of a spinning rocket plume at the nozzle exit. However, the plume temperature for secondary combustion of flammable gas is consistent, and the average temperature errors are less than 6 percent, except for the plume tail end and nozzle exit. The maximum temperature appears at 30 cm below the nozzle exit. The temperature errors in the plume tail end are caused by ignition delay time that affects the amount of flammable gas in the combustion chamber and the large amount of cold air induced by turbulence in the plume tail end.

CONCLUSIONS

The study of spin behavior of the rocket plume temperature indicates that the average error of temperature measurement is below 3.5%, despite the 10% error existed in the estimated emittance value. The continuous increase in rocket plume temperature is caused by the secondary combustion when the plume is ejected from the nozzle. The rocket plume temperature at the nozzle exit clearly increases are the burning time goes on under the no-spin condition, however it will decrease due to cold ambient air vortex and a lack of high-temperature metal particles at the spinning condition. The plume temperature for the secondary combustion of flammable gas is consistent. The average error of center-line plume temperature is less than 6 percent, and the maximum temperature appears at the position of 30 cm below of the nozzle exit.

REFERENCE

Figure 1. A diagram of the experimental test stand setup

Figure 2. The chamber pressure and axial thrust versus time distribution of the motor test
Figure 3. The effects of emissivity on plume temperature

Figure 4. Temperature distribution of rocket plume at no spin condition
Figure 5. Temperature distribution of rocket plume at 600 rpm

Figure 6. Temperature distribution of rocket plume at 960 rpm
Figure 7. Temperature distribution of rocket plume at 1200 rpm

Figure 8. Temperature distribution of rocket plume at start burn in different spin rate
Figure 9. Temperature distributions of rocket plume at burn 0.8 sec in different spin rate

Figure 10. Temperature distributions of rocket plume at burn 1.6 sec in different spin rate