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Engineering Notes

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Preliminary Design of Superorbital Earth Entry Flight Experiment Using the Volna Launcher

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I. Introduction

IT IS a challenge to predict heat-shield response for space mission involving significant radiant heating. More data are needed to increase the knowledge in the high-temperature flight environment [1–3]. The discrepancies between the calculated and the measurement results led one to inquire into the details of the interaction of radiation, convection, and ablation. Most of the radiation is absorbed in the boundary layer, which contains the ablation product. The radiatively heated boundary layer raises the convective heat transfer rate partly at the point at which absorption occurs [4,5]. It is very difficult to study these phenomena in a ground-test facility because of the limitations on enthalpy and dimensions. A new flight test with an ablating wall becomes highly desirable. We explore the possibility of a superorbital reentry flight experiment to test such interactions.

A series of flight tests [6,7] are proving the usefulness of the Russian submarine-launched Volna [8] as an efficient means of studying the reentry problems. However, it produces only suborbital entry velocities. We propose that the reentry velocity is increased by fitting the payload of Volna with a small solid rocket booster (SRB). An inviscid Newtonian analysis is made to evaluate the aerodynamic coefficients. The mass and performance are estimated using data on existing small SRB through flight trajectory calculations. The shape of the reentry vehicle (RV) and entry velocity is taken to be that of the Fire-II [4] so that comparison can be made. The interaction between radiative, convective heating, and ablation is calculated at the stagnation point for a carbon-phenolic (CP) ablator.

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II. Analysis Method

A. Mission Profile

The nominal gross liftoff mass of Volna is 35.4 t and the payload mass is 650 kg. The payload bay has a volume of 1.3 m^3. The vehicle is launched in the Barents Sea at the location of 69.3°N, 35.3°E with an inclination angle of 79 deg. During the third-stage ascent phase, the RV is initialized and oriented. After the separation, the RV reaches its maximum altitude. After this event, the RV is on a ballistic reentry trajectory. On the final part of the downward flight, the SRB is fired. At the end of this burn, RV separation occurs at the altitude of 120 km and the RV begins the reentry flight with 11 km/s velocity. During the reentry flight, the scientific measurements are made. The heating pulse finishes and the parachute opens at Mach 2 for stability reasons, with a rapid deceleration to a subsonic speed. The RV lands in a Kamchatka peninsula landing site of 57°N, 161°E.

B. Flight Parameters

The parameters of the basic launch system are deduced from the three known suborbital flights, which are the 1) maximum payload condition [8], 2) typical mission [8], and 3) inflatable reentry and descent technology (IRDT) mission [6]. A ballistic trajectory calculation is performed from the apogee to the landing position, applying two known shapes of RV: IRDT for the minimum payload and Volan [8] for all other payload conditions. The flight parameters deduced in this way are used in the calculation of the boosting period.

To determine the mass of the SRB-RV combination and the other parameters, the total impulse and the mass flow rate are needed as a function of firing time. For this purpose, the data [9,10] on the existing SRBs are collected. First, the propellant mass fraction and mass flow rate of existing SRBs are plotted against the mass of the SRBs. Mean values are fitted to their plots. From the mass fraction and mass flow rate values so estimated, the engine burn time is estimated as a function of SRB mass. Next, for each SRB-RV combination satisfying the requirement of entry velocity and flight corridor, a three-degree-of-freedom trajectory analysis is performed. The time for initiation of each burn is determined to satisfy the condition that the burning ends at the altitude of 120 km. The thrust structural mass supporting the SRB is estimated by the hypersonic aerospace sizing analysis method [11].

C. Calculation of Heating Rates

The convective heat flux in the absence of radiation, q_{c0}, is a sum of four components: catalytic recombination, surface reaction, sublimation, and conduction. To determine the radiation absorption in the boundary layer, the energy equation is written as

$$\left(\frac{\rho\mu}{\rho_e\mu_e}\frac{1}{P_r}g'\right)' + fg' + \frac{u_e^2}{H_e}\left[\left(1-\frac{1}{P_r}\right)\frac{\rho\mu}{\rho_e\mu_e}f'f''\right]' = \dot{s} \qquad (1)$$

where P_r, u, and subscript e are the Prandtl number, the tangential velocity, and the edge value; $\dot{s}$ is the normalized rate of energy addition by radiation absorption; and momentum function f satisfies the well-known Blasius-type third-order nonlinear ordinary differential equation. First, the mass-momentum equation for f, energy function g, and the species equation are solved assuming a frozen flow without $\dot{s}$. This procedure yields q_{c0}. The quantity $\dot{s}$ leads to an increase in the convective heat transfer rate because of the heating by radiation absorption. To obtain $\dot{s}$ accurately, one must carry out radiative transfer calculation through the entire shock layer. In the present work, such a lengthy work is impractical. Instead, a constant value that gives the correct results for Stardust [5], Fire-II [4], and Apollo 4 [4] are obtained. This constant value can be fitted against a variable containing freestream density ρ_∞, velocity v_∞, and nose radius r_{nose} in the form

$$\eta_r = 0.459\rho_\infty^{0.0512} v_\infty^{0.101} r_{\mathrm{nose}}^{-0.0937} \tag{2}$$

Using Eq. (2), we find that 55.5 to 88.1% of radiative heat flux at the boundary-layer edge is absorbed and 100 to 42.6% of absorbed radiative heat flux is converted to convective heat flux during the reentry flight proposed. The value of $\dot{s}$ is chosen to produce these results.

In [12], the radiative heat flux incident on the wall was presented over a wide range of flight conditions. In the present study, these data are represented by

$$q_{rw} = \frac{(1-\eta_r)}{(1-2\eta_r/3)} 10^{0.4865(v_\infty/1000)+1.4157+1.1149\log_{10}(\rho_\infty r_{\mathrm{nose}})} \tag{3}$$

Using Eqs. (2) and (3), the radiative heat flux at the boundary-layer edge can be deduced. The total convective heat flux with increased terms is expressed as

$$q_{cw} = q_{c0} + \frac{\eta_r(-2.134\eta_r + 2.307)}{(1-2\eta_r/3)} 10^{0.4865(v_\infty/1000)+1.4157+1.1149\log_{10}(\rho_\infty r_{\mathrm{nose}})} \tag{4}$$

To verify the accuracy of the present method, calculation is made for the Fire-II mission. The calculation results along the whole reentry trajectory are compared with the original calculation data [4]. For the peak heating period of 1642.48 s, the present model reproduces the original calculation closely: 90.4% of convective and 83.7% of radiative heat flux. The details are given in [13].

D. Calculation of Heat-Shield Recession

The heat shield of the present study is assumed to be a CP used for the Pioneer–Venus entry vehicle [14]. Fully coupled calculation with ablation and radiation is performed along the reentry trajectory. The catalytic coefficient for the oxygen recombination $\eta_{cr} = 0$ is assumed because the reactions are experimentally observed not to occur to any measurable extent. But we considered surface reactions: the carbon atoms combine with O and N to form CO and CN. At the conditions expected in this flight, sublimation of carbon will produce mostly C_2 and C_3. The probability of surface reactions η_{sr} and sublimation η_s are expressed in the form

$$\eta_{\mathrm{sr},s} = c_1\left(\frac{T_w}{1000}\right)^{c_2} \exp\left(\frac{c_3}{T_w}\right) \tag{5}$$

where T_w is the wall temperature.

In the surface reaction of O, the parameters c_1, c_2, and c_3 are known to be 0.63, 0, and −1160. For the reaction of N, $\eta_{\mathrm{sr}} = 0.3$. There is also a city-gas process of H_2O. The η_{sr} for this is negligibly small compared with the other two processes. The coefficient η_s must be determined by measurement. For C_2 and C_3, it is 0.003, 0, 0, and 0.01, 0, 0.

The behavior of CP during ablation is determined by the Korean Charring Materials Ablation (KCMA) program [1]. Unlike the original KCMA (the intrinsic density of char and resin), the initial and maximum void fractions of pure char are specified in the input. The material response calculation yields the recession as a function of time.

III. Results

A. Mass Budget and SRB Design Criteria

The calculated result in Fig. 1 shows RV masses as a function of SRB masses. The payload bay volume needed in packing is

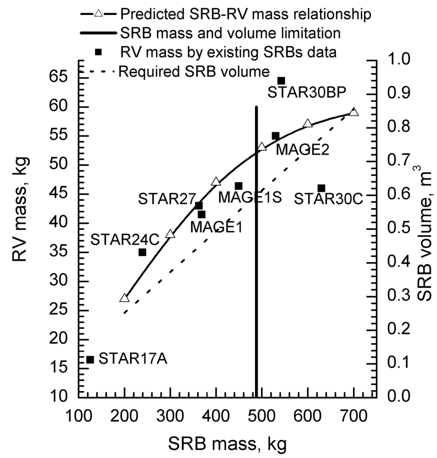

Fig. 1 Predicted mass of RV-SRB combination for reentry velocity of 11 km/s.

determined by the aspect ratio of SRB (i.e., diameter divided by its length and volume). The aspect ratio of selected SRBs ranges from 0.45 to 0.68 and their volume occupies between 0.15 and 0.69 m^3. However, because of the payload bay configuration, SRB volume must be selected in the range below 0.6 m^3 shown by the vertical line. This volume corresponds to SRB mass of 488 kg, propellant mass flow rate of 9.8 kg/s, and propellant mass fraction of 0.925. The effective specific impulse is selected as 289.2 s.

The preferred SRB is the Mage1S. Although it has 93% of the required propellant mass flow rate, it has the highest specific impulse of 291 s. In addition, its comparatively large aspect ratio (0.951, excluding the nozzle section) offers a high packing efficiency. Mage1S can accelerate a 46.4 kg mass up to the desired reentry velocity of 11 km/s. The thrust structural mass is taken to be 6 kg. The ballistic coefficient of the RV calculated is 73.5. It is especially close to that of the Stardust.

B. Boosted Flight Trajectory

Using the obtained aerodynamic characteristics and mass budget of RV-Mage1S combination, a three-degree-of freedom trajectory calculation was performed. The maximum altitude is 685.3 km at the location of 82.5°N, 116.4°E. The initial heading of 123 deg is predicted. At 479 s after its maximum altitude, the RV is boosted for 50 s before reaching the altitude of 120 km. A 11.05 km/s entry velocity with a flight-path angle of −17.7 deg is achieved.

C. Heat Transfer Rates

The results of the calculation show that the time point $t = 17$ s after 100 km is predicted as the peak heating point. It occurs 5 s earlier than the case without radiation coupling. The calculated convective peak heating rate of 1424 W/cm^2 is about 2.6 times the value in the absence of radiation. The increment of 1040 W/cm^2 is nearly 50% of the flux absorbed in the boundary layer: 2084 W/cm^2. The rest, 50% of absorbed radiative heat flux, is convectively discharged downstream. However, at 6 km/s, the increment of convective heat flux is reduced to below 20% of the value in the absence of radiation. The integrated total heat load of 25.54 kJ/cm^2 is about 120% of the Fire-II.

D. Heat-Shield Sizing

Stagnation point gas-surface interaction and thermal sizing was performed for the candidate CP heat shield. For the range of flight parameters considered, the total stagnation heat flux ranges from 12.6 to 1820 W/cm^2 and the dynamic pressure ranges from 1.31 to 165 kPa. The maximum recession is 10 mm. It is about 1.7 times the value in the absence of radiation. If we select the insulation requirement to maintain the bond-line temperature to be below 500 K, the needed heat-shield thickness is calculated to be 17 mm for the parachute opening points of Mach 2.

III. Conclusions

The payload package atop the Volna launcher can be fitted and accelerated by a small SRB. The SRB design criteria lead to a maximum mass of 488 kg, propellant mass fraction of 0.925, and mass flow rate of 9.8 kg/s. In this range, the most preferred SRB is Mage1S, which should be able to accelerate an RV, which has the Fire-II shape, mass of 46.4 kg to 11 km/s.

The increment of convective heat flux at the peak heating point due to radiation absorption was nearly 50% of the radiative heat flux absorbed in the boundary layer. According to the criterion set in the beginning, that the conversion of radiative heat flux to convective heat flux should be substantial, the proposed flight experiment is valid.

The calculation shows the convective and radiative peak heating rates to be $1424\ \mathrm{W/cm^2}$ and $394\ \mathrm{W/cm^2}$, respectively. The integrated total heat load is predicted to be $25.54\ \mathrm{kJ/cm^2}$. The maximum recession for the considered ablator was 10 mm. It is about 1.7 times the value in the absence of radiation. The needed heat-shield thickness is calculated to be 17 mm for the parachute opening at Mach 2.

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