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ASCENT THERMAL ANALYSIS OF FAIRING OF SPACE LAUNCH VEHICLE

Sang-Ho Choi†, Seong-Lyong Kim, and Insun Kim
Dept. of Thermal/Aerodynamics, KARI, Daejeon 305-333, Korea
email: saintl@kari.re.kr

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ABSTRACT

The fairing of the launch vehicles has a role of protecting the spacecraft from outer thermal, acoustical, and mechanical loads during flight. Among them, the thermal load is analyzed in the present study. The ascent thermal analyses include aerodynamic heating rate on every point of the fairing, heat transfer through the fairing and spacecraft, and the final temperature during ascent flight phase. A design code based on theoretical/experimental database is applied to calculate the aerodynamic heating rate, and a thermal math program, SINDA/Fluint, is considered for conductive heat transfer of the fairing. The results show that the present design satisfies the allowing temperature of the structure. Another important thermal problem, pyro explosive fairing separation device, is calculated because the pyro system is very sensitive to the temperature. The results also satisfies the pyro thermal condition.

Keywords: fairing insulation, launch vehicle, aerodynamic heating, heat transfer

1. INTRODUCTION

The fairing of the launch vehicles has a role of protecting the spacecraft from outer thermal, acoustical mechanical loads. Among them, the thermal load is the most important for fairing design because the aerodynamic heating can destroy the fairing structure. Korea Aerospace Research Institute has developed some fairing structure appropriate for sounding rockets. In this study, a newly developed fairing adapted for space launch vehicle and its thermal load will be analyzed.

The general fairing configure is three parts, fairing nosecap of hemisphere type, fairing cone, fairing cylinder. In some cases, the fairing cone section is divided 2 or 3 part as the inclined angle. In the present study, the nosecap radius is 0.25 m, the inclied angle of the cone is 16°, and the radius of the cylinder is 1m. The fairing is composed of 4 layers, insulator (2 mm), CFRP sheet (1.6 mm) aluminium honeycomb (26 mm) and CFRP (1.6 mm) sheet.

2. TRAJECTORY AND AERODYNAMIC HEATING RATE

The aerodynamic heating rates strongly depends on the trajectory of the launch vehicle because it is a function of density and velocity. Launch vehicle's typical trajectory appears in Figure 1. At low altitude the vehicle velocity is relatively small, subsonic, while it is very fast in high altitude.

†corresponding author

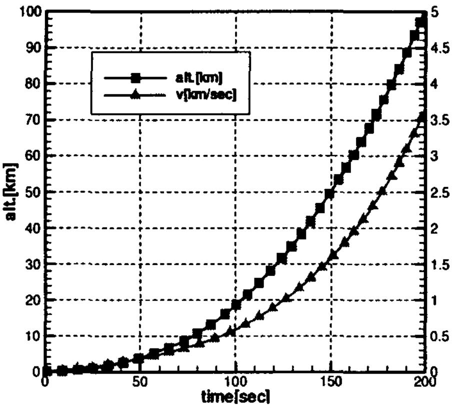

Figure 1. Trajectory profile.

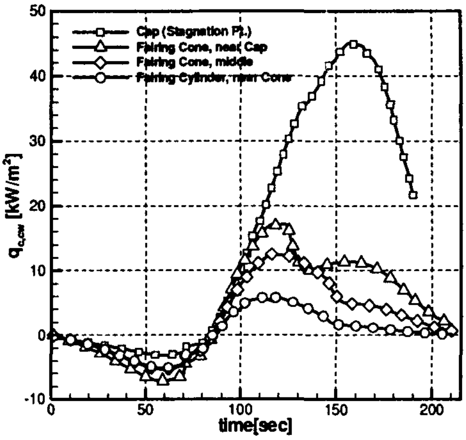

Figure 2. Aerodynamic heating rates.

The aerodynamic heating rates are shown in Figure 2 at the point of the fairing cap, cone and cylinder sections. At the nose cap, Fay & Riddel stagnation point aerodynamic heating rate equation with real gas effects is considered while turbulent boundary layer heat transfer equation is applied to the other point. Figure 2 shows that the aerodynamic heating rate at the nose cap is very large compared to the other point. As the distance from the nose cap increases, the aerodynamic heating rates decreases because the boundary layer thickness increases.

3. FAIRING TEMPERATURE

For the inner heat transfer calculation, the two method, intrinsic conduction solver of the MINIVER (Hender 1970) and the SINDA/Fluint solver (C&R Tech. 2004), are used. The two method show the almost same temperature at the fairing temperature. However, the heat analysis at the fairing separation device can be calculated with only the SINDA/Fluint because it is two dimensional heat transfer.

The resulting fairing temperature at outer and inner surfaces are shown in Figure 3 and 4. In the initial stage of the flight, the temperatures decrease, but they increases as the vehicle velocity increases. This cooling effects is due to the cooling of subsonic low temperature free stream flows because aerodynamic heating apparently appears in supersonic flows. Outer surface temperature increases up to 280°C, but the inner surface temperatures remain below 100, which satisfies the specified temperature 100°C.

The fairing separation device operates when the outer heating rate decreases below the specified value, 1135W/m^2. Because the device mechanism is pyro explosive, it is very sensitive to the temperature. The device is made of aluminium or SUS which are high thermally conductive. The fast heat conduction forces to use SINDA multi dimensional heat transfer routine rather than MINIVER intrinsic 1 D conduction routine.

The resulting temperature of the fairing device are shown in Figure 5. The maximum temperature is about 75°C regardless of the insulator, which is sufficiently below the explosion temperature limit. Generally if the insulator is applied to the material, the temperature variance and the maxi-

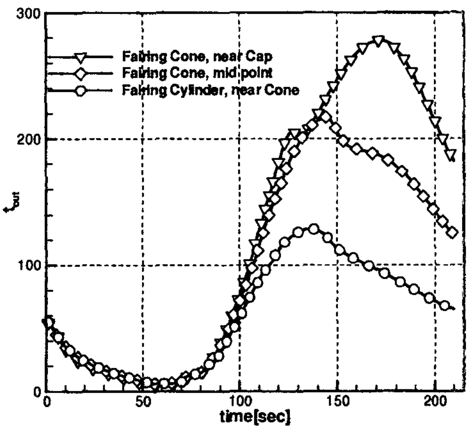

Figure 3. Outer surface temperature.

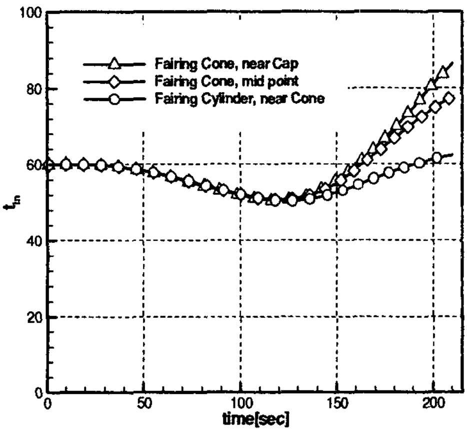

Figure 4. Inner surface temperature.

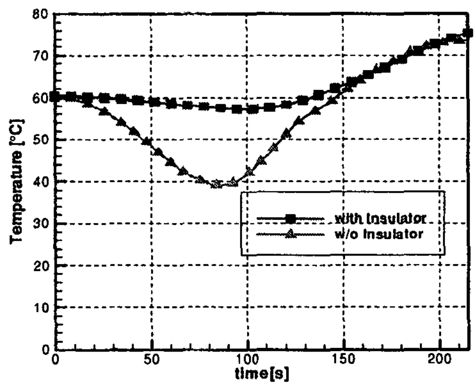

Figure 5. temperature of the separation device.

mum temperature decrease. The same effects appear in Figure 5. Without the insulator, the variation of the device temperature is about 36°C while it remains 18°C with the insulator. The maximum temperature is almost the same, which may be explained by relatively low aerodynamic heating rates.

4. CONCLUSIONS

The aerodynamic heating rate and the heat transfer analysis through the fairing and spacecraft has been calculated. The analyses show that the present design satisfies the launch vehicle thermal condition. Another important thermal problem in fairing design, the temperature of the pyro explosive fairing separation device, also satisfies the thermal condition.

REFERENCES

Hender, D. R. 1970, A Miniature Version of the JA70 Aerodynamic Heating Computer Program, H800 (MINIVER) (Athens: NASA COSMIC)

C&R Tech. 2004, SINDA/Fluint Version 4.6 User's Manual (Colorado: Cullimore and Ring Technologies Inc.)