한국추진공학회 2005년도 춘계학술대회 논문집 pp.399~402 2005 KSPE Spring Conference

초음속 동체후미부에서 발생하는 Plume-Induced Shock Wave에 관한 연구

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A Study of the Plume-Induced Shock Wave on Supersonic Afterbodies

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ABSTRACT

The present numerical study describes the flow physics on the interaction between the supersonic freestream and jet plume. The compressible flow past a simplified afterbody model with a sonic nozzle is investigated using mass-averaged Navier-Stokes equations, discretized by a fully implicit finite volume scheme, and the standard k-ω turbulence model. The results obtained through the present study are discussed specifically regarding the effect of the plume pressure ratio, freestream Mach number and base dimensions on the location of the plume-induced shock wave generated on the afterbody by the underexpansion of the jet plume.

초 록

본 연구에서는 초음속 자유유동과 제트 플룸 간의 간섭에 관련된 유동현상들을 수치해석적 방법을 이용하여 고찰하였다. 수치계산은 유한체적법으로 이산화한 압축성 mass-averaged Navier-Stokes 방정식에 표준 k-ω 난류모델을 적용하여 수행하였으며, 계산에 사용된 모델은 음속노즐을 가지는 동체후미부 만으로 단순화하였다. 수치계산결과는 플룸압력비, 기류마하수 및 베이스 직경이 플룸의 부족팽창에 의해서 동체후미부 상에 발생하는 plume-induced shock wave의 위치에 주는 영향에 대하여 주로 기술하였다.

Key Words: Afterbody(동체후미부), Plume (플룸), Shock Wave(충격파), Supersonic Flow(초음속 유동), Underexpansion(부족팽창)

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1. Introduction

Due to the demand of smaller, lighter and faster design, most of modern supersonic missiles have a very high thrust level within a limited cross sectional area. Such missile configurations generally produce a highly underexpanded jet plume[1] downstream of the exhaust nozzle exit, leading to considerable interaction between the plume and freestream near the tail of missile bodies. The boundary layer separation and pitching and yawing moments resulting from the interactions can have significant effects on missile stability and control[2].

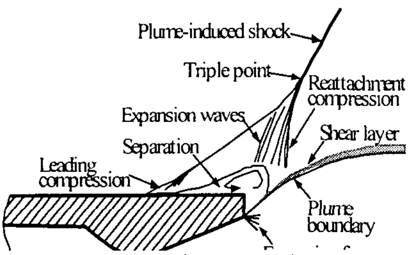

Fig. 1 Plume-freestream interaction

Plume interference schematically shown in Fig. 1 is a very complicated phenomenon, including the underexpansion of a plume, a λ-shaped plume-induced shock system, and a separation bubble and expansion waves inside the shock system. The shock wave can give self-oscillatory motions and additional drag so that does harm to the flight performance. Furthermore, the shock-induced separation occurs on the tail of the missile where fins are normally located, and therefore can have an adverse effect on control. The flow features changes basically depending on the dimensions of plume, affected by the plume pressure ratio, freestream Mach number and base geometry.

In the present study, the plume-induced shock wave generated on a simplified afterbody model with a sonic nozzle has been investigated using CFD as a tool. A fully implicit finite volume scheme was applied to compressible mass-averaged Navier-Stokes equations with the standard k-ω turbulence model. The results offer mainly an understanding of the effect of the plume pressure ratio, freestream Mach number and base dimensions on the location of the plume-induced shock wave, which is essential information for plume interference control[3,4].

2. Numerical Simulations

2.1 Model configuration

Figure 2 shows the schematic diagram of the axisymmetric afterbody model used in the present study. A propulsion system is simplified as a rectangular afterbody of 5D long (based on the model with 2.0D_e), where D is the diameter of afterbody, and a sonic nozzle with the exit diameter of D_e (= 31.75 mm). To find the effect of the afterbody diameter on the plume-freestream interaction, D is varied from 1.5D_e to 2.5D_e. In the figure, M, p and T are the Mach number, pressure and temperature respectively, and subscripts c and ∞ mean the combustion chamber and freestream respectively.

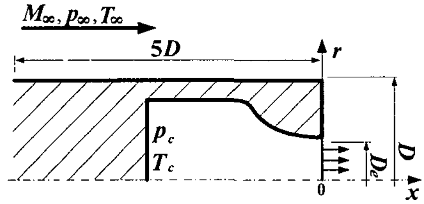

Fig. 2 Testing model

2.2 Computational domain and analysis

The computational domain was built up in consideration of strong plume underexpansion at high pressure ratios. In order to obtain grid independent solutions, the grid fineness was first examined for some of the flowfields under consideration. Shock waves and mixing layers are very thin and the convergence of solution strongly depends on the plume size and location of shock waves in the present flowfields. Grids were, therefore, clustered in regions with large gradients, such as shock waves, shear layers, and boundary layers to obtain more accurate predictions of the flowfield.

To provide various characteristics of plume-freestream interaction, the freestream Mach number M_∞ was changed up to 2.5 at a plume pressure ratio p_c/p_∞ from 50 to 600, which gave a moderately to highly underexpanded plume for the present afterbody model. Freestream pressure and temperature were assumed to be constant with the values of 1 atm and 300 K respectively.

In the computation, solutions were considered converged when the residuals for all equations were less than, typically, 1.0×10^{-4} with the mass imbalance check for flow inlet and outlet boundaries.

3. Results and Discussion

Figure 3 shows Mach number contours for D = $2.0D_e$ and M_∞ = 2.0. At p_c/p_∞ = 100, the highly underexpanded jet structure, called plume, forms a compression corner near the edge of the base, thus leading to a λ-shock system. As the plume pressure ratio increases, higher underexpansion gives a larger compression corner so that the plume-induced shock wave moves upstream and the separation bubble behind the shock wave becomes larger.

For D= $2.0D_e$, Fig. 4 shows shock locations with a change in plume pressure ratio. At M_∞ = 2.0 and 2.5, which are relatively higher Mach numbers, the shock location increases almost linearly as p_c/p_∞ becomes higher. At Mach 2.0, however, the rate of shock movement is considerably larger than other two cases and the rate becomes slightly smaller as p_c/p_∞ increases. The effect of the freestream Mach number on the shock location is, therefore, more significant at a higher pressure ratio.

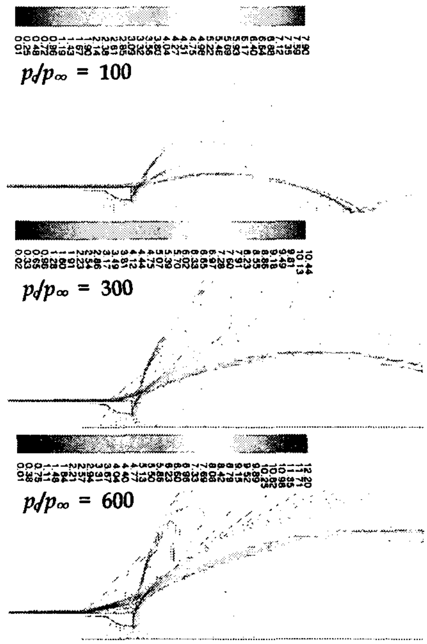

Fig. 3 Mach number contours for D = $2.0D_e$ and M_∞ = 2.0

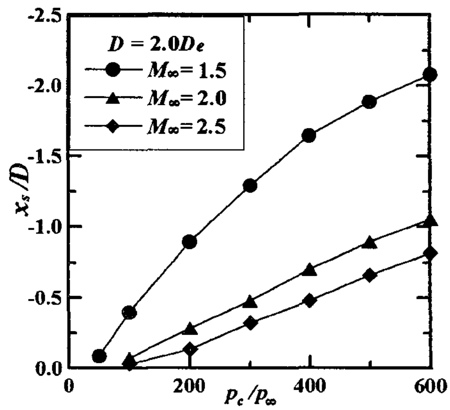

Fig. 4 Shock location vs. plume pressure ratio (D = $2.0D_e$)

For M_∞ = 2.0 and p_c/p_∞ = 300, Mach number contours given in Fig. 5 show the effect of the afterbody diameter on plume

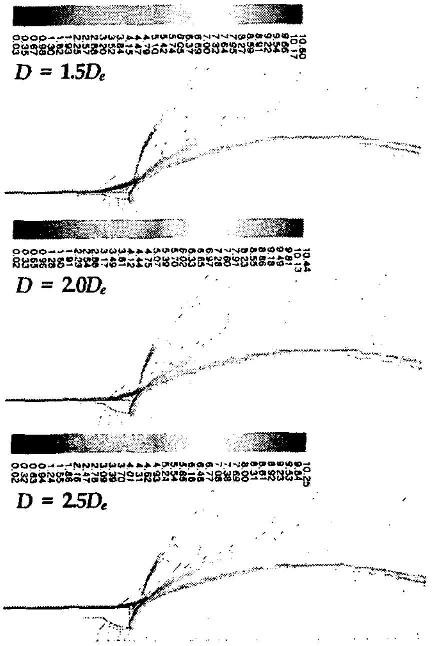

Fig. 5 Mach number contours for M_∞ = 2.0 and p_c/p_∞ = 300

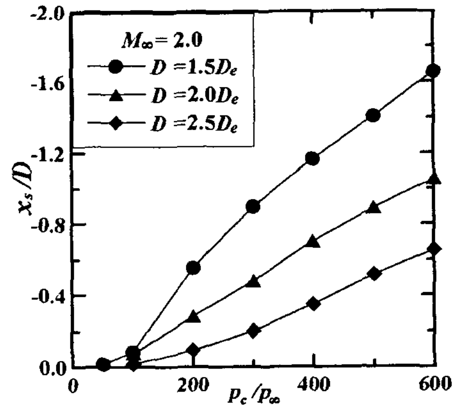

Fig. 6 Shock location vs. base diameter (M_∞ = 2.0)

intereference. It can be observed that a larger D value results in weaker shock-boundary layer interaction on the afterbody. As shown in Fig. 6, the shock location with a change in D is more significantly varied at a higher pressure ratio and it is considered negligible at relatively low pressure ratio (p_c/p_∞ < 100).

4. Concluding Remarks

The plume interference phenomenon was discussed especially on the location of a shock wave, which occurs on the afterbody tail by plume expansion. Supersonic freestream and plume interaction was simulated using mass-averaged Navier-Stokes equations and the standard k-ω turbulence model. A simplified afterbody model with a sonic nozzle has been tested for three base configurations at various freestream Mach numbers and plume pressure ratios. The results obtained through the present CFD analysis showed that the effects of the freestream Mach number and afterbody diameter on the shock location were more significant at a higher pressure ratio.

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