Numerical Simulation of Shock Wave-Boundary Layer Interaction with Supersonic Film Cooling

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Abstract. As an effective external cooling method, film cooling can significantly reduce the wall temperature of hot end components of aero engines and large liquid rocket engines. With the development of supersonic turbine blade, liquid rocket engine thrust chamber and ramjet thermal protection technology, the impact of shock wave-boundary layer interaction on the cooling effectiveness of film cooling is a problem that must be considered in the engine heat transfer design. In this paper, the two-dimensional supersonic film cooling is simulated based on the plane slot model, and the blowing ratio is similar to the actual engine. Impact of wave structures on film cooling was studied, and flow loss was evaluated by total pressure and entropy parameter to investigate the film cooling efficiency and uniformity distribution. Physical mechanism of shock wave-boundary layer interaction and its impact on film form was revealed. It shows that the shock wave leads to dramatic changes in flow parameters. Boundary layer separation is found due to the shock, and significantly affects the cooling efficiency. The blowing ratio is an important factor affecting the cooling efficiency. With blowing ratio increasing, local film cooling effectiveness and average cooling efficiency are improved and cooling uniformity decreases. However, the shock wave will cause the temperature of the insulation wall to rise, resulting in decreased cooling efficiency.

1. Introduction
As one of the main cooling methods of aero engines and liquid rocket engines, film cooling can provide effective thermal protection for the hot end components of the engine and greatly improve the reliability and service life of components. With the advancement of propulsion technology, supersonic combustion chambers and super/transonic turbines have become important aspects of engine design. Meanwhile, the film cooling of hot end components also presents new features, mainly manifested by the compressibility of air flow and the interaction between shock wave and boundary layer, which will greatly affect the mixed flow conditions of hot gas and cooling medium, thus affecting the film cooling effect.

The interaction between shock wave and boundary layer was first discovered by Ferri in the high speed tunnel [1]. Smith calculated and verified that the shock induced the boundary layer separation, and the area of the separation zone was directly related to the strength of the oblique shock wave, and he found that, the gas expansion accelerated after the shock wave, and the boundary layer gradually attached to the wall [2]. Ligrani investigated the interaction between shock wave and boundary layer by using the flat film cooling model (cylindrical film hole), and found that the cooling efficiency decreased with the increasing of Mach number [3]. Konopka studied the effect of shock wave on supersonic film cooling by large eddy simulation, and found that the effect of shock wave can...
significantly enhance the turbulence intensity of cooling air, and make the flow field appear obvious temperature fluctuation [4]. Through numerical simulation, Peng [5] studied the two-dimensional slot plate film cooling model under the action of shock wave, and the result shown that shock waves intensified the mixing of mainstream and cooling flow, resulting in the decrease of film cooling efficiency and the increase of adiabatic wall temperature. Wang investigated the change of the adiabatic temperature ratio of supersonic jets with the blowing ratio and the shape of the jet channels using a two-dimensional flat film cooling model. There have been few studies on the flow conditions and cooling characteristics forward the injection holes [6].

Whether the cooling film can maintain the wall attached state in a longer distance is the key to the effectiveness of film cooling. Generally, the strength of oblique shock has a significant effect on the size of the boundary layer separation region. When the shock wave enters the cooling film, it will generate a strong reverse pressure gradient after the wave, and propagate along the subsonic speed area to the upstream, which makes the cooling film to be lifted, causes the boundary layer to separate and generate vortex, and intensifies the mixing with the mainstream, resulting in the reduction of the film cooling efficiency. Therefore, it is particularly important to study the deterioration effect of the shock wave on the film cooling and how to suppress this effect. Fluent, a commercial computational fluid dynamics tool is employed in present paper to calculate the flow field. Flow pattern of supersonic film cooling is analysed in order to reveal the mechanism of flow separation induced by the shock wave and the shock wave-boundary layer interaction with film cooling. Furthermore, the influence of shock wave on the adiabatic wall temperature and the film cooling effectiveness is discussed.

2. Numerical simulation method

Two dimensional slot plate film cooling model has been established to simulate hot end components of aero-engine, and the computational domain is shown in figure 1. The cooling airflow is injected parallel to the mainstream, and the flow field height ratio of mainstream and cold flow is 9:1. A shock wave generator consisted of wedge angle with 10°is arranged on the upper-wall. The quadrilateral mesh is used to divide the structured mesh, and the total number of nodes is about 600,000. The turbulence model of SST $k-\varepsilon$ is applied to capture the mixing flow and flow separation phenomena in the near-wall area. Mach number of supersonic main flow at inlet is set to be 3, and Mach number of cooling air flow increased from 0.5 to 2.5, with 8 blowing ratios ($M=\rho_c V_c/\rho_\infty V_\infty$, “$c$” stands for cooling flow, and “$\infty$” stands for main flow) of 0.236, 0.283, 0.330, 0.354, 0.589, 0.707, 0.943 and 1.179. The boundary condition of pressure extrapolation is set at the outlet of the flow field to adapt to the characteristics of supersonic flow.

![Figure 1. Calculation model of supersonic flat film cooling](image)

3. Results and analysis

3.1. Flow field interference by shock wave

The influence of shock wave-boundary layer interaction on film cooling morphology and cooling effectiveness can be analysed according to the variation of aerodynamic parameters such as static
pressure, Mach number and velocity. As shown in figure 2&3, the first oblique shock wave is induced along the leading edge of the supersonic main stream in front of the wedge. At the corner of the top of the cold flow inlet, a cluster of expansion waves are generated due to the widening of the flow channel. At the intersection of the main flow and the cold flow, a second shock wave is generated due to severe compression. The first shock intersects the second shock under the action of the expansion wave. The shock wave extends to the cold flow boundary layer, and the pressure rise after the wave produces an inverse pressure gradient, which propagates upstream through the subsonic velocity region of the boundary layer. As a result, the thickness of the boundary layer increases and the streamline rises. Then the channel gets narrow and the reflection shock wave is formed. The reflected shock and the shock excited by the leeward surface intersect into a "λ" wave. The shock wave continuously reflects and intersects in the flow field; the intensity gradually weakens, and at the same time changes the flow field of the mixed flow.

When the reverse pressure gradient is sufficiently high and the velocity of cold flow is low, a local backflow phenomenon will occur near the shock wave reflection point (figure 4). However, when the cold flow is close to the Mach number of the main flow, although the shock wave will lift the streamline, no backflow will occur. Instead, the cold flow and the main flow maintain a constant flow path, and there will be no shock wave at the interface. At the same time, the higher the Mach number of the cooling flow is, the stronger the turbulence intensity is, and the greater the ability of turbulence to overcome the adverse pressure gradient is. Therefore, in a certain range, the interaction between shock wave and cold flow boundary layer will decrease with the increase of cold flow Mach number.

In order to evaluate the flow loss of supersonic film cooling, distributions of total pressure and entropy are researched (figure 5&6). The total pressure loss is obvious after the shock wave. The shock intensity increases with the increase of the Mach number of the main flow, which leads to the more serious loss of the total pressure after the shock; after the reflection, the shock intensity decreases, and the total pressure loss decreases obviously. At the interface of the two flows, there will be an obvious slipstream layer, and the momentum exchange of the layer will lead to the total pressure between the two. The entropy of the whole system (entropy generation) increases due to the significant
loss of the total pressure after the multiple shock waves. For the cooling flow, the entropy increase produced by shock wave consists of two parts. On the one hand, shock waves are reflected at the boundary layer of the cold flow, which increases the entropy of the airflow passing through the shock wave. However, the shock can only penetrate into a small area of the cold flow because of the reverse pressure gradient of the shock wave, which makes the sonic line move up. Therefore, this part of entropy increase is very small. On the other hand, it is because of the backflow inside the cold flow that leads to energy dissipation. This part of entropy increase accounts for the main part of total entropy increase. When the Mach number of the cold flow is low, a strong vortex will be generated below the shock reflection point, and when the Mach number of the cold flow is high, the ability to overcome the back pressure gradient is strong, so the vortex will not be generated and the entropy increases slightly.

3.2. Film cooling efficiency

The cooling effectiveness can be embodied by film cooling efficiency, which defines:

\[ \eta_i = \frac{T_g - T_{in}}{T_g - T_c} \]  

As shown in figure 7, with the increase of blowing ratio, the proportion of cooling air in the flow field increases, making the film cooling efficiency monotonously improved. The point of incidence of the shock wave occurs at x≈9cm, where a sudden drop of film cooling efficiency can be found. What’s more, as the blowing ratio increases, the impact of shock waves on the cooling efficiency will gradually decrease. Analysis on the mechanism level, the flow parameters will undergo a step change after the shock wave, resulting in the increase of gas pressure, density, and static temperature, and a decrease in the Mach number. The increase in temperature will cause the cooling efficiency of the film after the wave to decrease, but the shock wave will not affect the cooling efficiency of the film upstream of the incident point.

![Flow separation induced by shock wave](image1)

![Flow separation induced by shock wave](image2)

![Variation of film cooling efficiency on adiabatic wall](image3)
3.3. Average film cooling efficiency and non-uniformity of cooling efficiency

The average film cooling efficiency ($\eta_{av}$) is the area average of the local cooling efficiency, which can be used to evaluate the cooling effectiveness of a specific surface. The difference in the coverage of the film on the surface can be characterized by non-uniformity of cooling efficiency ($\sigma$). The expressions of the two variables are as following:

$$\eta_{av} = \frac{\iint \eta(x,y) dA}{A}$$

(2)

$$\sigma = \sqrt{\frac{1}{A} \iint [\eta(x,y) - \eta_{av}]^2 dA}$$

(3)

In figure 8, with the increase of blowing ratio, the average film cooling efficiency of adiabatic wall is gradually increased. With a smaller blowing ratio, the average film cooling efficiency increases rapidly. And when the blowing ratio increases to a certain extent, the average film cooling efficiency growth slows down and approaches 1. Figure 9 shows that, with the increase of blowing ratio, the non-uniformity of film cooling effectiveness on the adiabatic wall decreases gradually, that is to say, the coverage of cold air is uniform. The results show that with the increase of blowing ratio, the variation of cooling efficiency on the adiabatic wall is gradually stable, which makes the fluctuation of the variation degree decrease, so the cooling non-uniformity shows a decreasing trend.

4. Conclusion

In this paper, the interaction of shock wave, film cooling and boundary layer is investigated by the means of numerical simulation method. A shock wave generator is fixed up in the supersonic main
flow field to induce oblique shock. Flow patterns, wave structures and cooling effectiveness under different blowing ratios are analysed, the conclusions as following:

1. The shock generator generates multiple oblique shock waves in the main flow and mixing region of supersonic flow, and the expansion wave is formed by reflection in the expansion section. Shock waves reflect and intersect in the flow field, which makes the flow field much more complex.

2. In the channel interlaced by the primary shock, reflected shock, derived shock and expansion wave, the pressure and velocity of the fluid change drastically. The phenomenon of boundary layer separation occurs, which leads to flow loss and entropy increase, and significantly affects the film cooling effectiveness.

3. The blowing ratio is a key factor affecting the cooling effectiveness of compressible film cooling. With the increase of blowing ratio, both the local film cooling efficiency and average cooling efficiency are improved, and the cooling non-uniformity is reduced. The shock wave causes the adiabatic wall temperature to rise and the cooling efficiency to decrease obviously.

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