Aerospace Science and Technology 81 (2018) 204–212

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

Inverse design and Mach 6 experimental investigation of a pressure controllable bump

Zonghan Yu, Guoping Huang *, Chen Xia

College of Energy and Power Engineering, Nanjing University of Aeronautics and Astronautics, Nanjing, Jiangsu 210016, People's Republic of China

ARTICLE INFO

Article history:
Received 7 April 2018
Received in revised form 31 July 2018
Accepted 3 August 2018
Available online 8 August 2018

Keywords:
Propulsion/airframe integration
Inverse design method
Permeable boundary method
Hypersonic boundary layer removing
Hypersonic wind tunnel test

ABSTRACT

The low kinetic flow starting from the airframe leading edge is a neglected aspect in designing hypersonic air-breathing flight vehicles. Compared with other boundary layer treating technologies, the bump concept can obtain a good balance of boundary layer removal, external drag control, shock system simplification, and integration design flexibility capabilities. On the basis of conventional conical-flow theory and the new 3D inverse design method, this study proposes a pressure controllable bump concept that can generate the bump configuration inversely by the prescribed pressure distribution. The bump/inlet integration pattern is analyzed, and the basic design methodology is presented. To validate whether the pressure distribution can be used in diverting the boundary layer, experimental study of the bump and numerical simulation are conducted. Results show that the bump has generated identical pressure distribution to the design. The bump can also divert approximately 50% of the boundary layer from the incoming low kinetic flow in Mach 6. Compared with the conventional cone-derived bump in Mach 6, the new bump is 25.8% shorter in height. The flow structure is adjusted nearly parallel to the *x*-direction, thereby promoting the flow quality of the inlet entrance. Hence, the new inverse design method of pressure distribution expands the applicable Mach range of the hypersonic airframe forebody.

1. Introduction

The high-speed air-breathing propulsion system can achieve global rapid reach from takeoff to high Mach cruise condition [1–3]. For the ramjet and scramjet, the importance of the inlet can be implied through an inlet–combustor ratio, which is the converted energy by air deceleration and compression (inlet part) divided by the generated energy of combustion (combustor part). The inlet–combustor ratio is 12% in Mach 1.8 when enthalpy is used to estimate the energy and the temperature of the combustion chamber is limited to 2000 K. As the Mach number increases to 3.4, the ratio increases to 2/3. When the Mach number is 4.5, the ratio increases to 2.3. Therefore, the inlet design is important to the air-breathing propulsion system with high Mach number [4–6].

Boundary layer removal is an inevitable aspect in supersonic/hypersonic inlet design. A strong compression (the magnitude is equivalent to that of a normal shock) of the inflow occurs in front of the airframe leading edge due to the bluntness effect [7–9], thereby leading to a rapid increase in the entropy. The entropy layer develops along the airframe and mixes with the boundary layer to form a low kinetic energy flow [10]. Studies have verified that the low kinetic energy is approximately 50% of the inlet entrance area in thickness, which affects the inlet compressing performance seriously [11,12].

Typical boundary layer removal techniques can be summarized into two categories according to the flow control method: active and passive flow control technologies. The boundary layer suction technology [13–16] uses a low stressor to divert the boundary layer into a low-pressure chamber. The boundary layer blowing technology [17–19] utilizes a high stressor to mix the boundary layer with the injected flow. The magneto-hydrodynamic flow control technology [20–22] uses external magnetic fields to ionized flows toward achieving flow behavior, such as inducing oblique shockwaves to change the flow structure. Different from the three active flow control technologies above, passive flow control technologies utilize specific aerodynamic configuration to change the original flow structure. The boundary layer compulsory transition technology (interchangeable trips) [23] suppresses the flow separation by turning the laminar boundary layer into turbulent condition. The vortex generator technology [24] adopts the vortex-inducing configuration to produce eddy flow and accelerate the mixing of mainstream and the boundary layer, thereby delaying the separation of the boundary layer. The ridge technology [25]

* Corresponding author.
E-mail address: hgp@nuaa.edu.cn (G. Huang).

https://doi.org/10.1016/j.ast.2018.08.006
1270-9638/

Nomenclature

β	shock wave angle
x	freestream direction
y	height direction
z	crosswise direction
K	intercepting height
π	static pressure ratio
p	static pressure
σ	total pressure recovery coefficient
k	specific heat ratio

Superscripts

$*$	stagnation properties

Subscripts

∞	free stream properties
local	true value of pressure

uses the pressure gap between the low-pressure area and the adjacent area to guide the boundary layer into the Ridge inner wall (the low-pressure area). Then, the low kinetic flow will be diverted away along the Ridge inner wall.

Apart from the technologies mentioned above, the bump concept is also adopted as a boundary layer removal technology [26–28]. This concept uses the pressure gradient at the streamwise and crosswise directions to divert the low kinetic flow. The concept is first designed by conical-flow theory, which obtains the leading edge of bump from the conical shock and uses streamline tracing method to generate the entire surface. In the design Mach number, the conical shock will completely attach on the leading edge, and the lift-to-drag ratio of bump considerably increases. Meanwhile, the bump is a waverider configuration. The bump configuration can be fully integrated with the airframe due to the smooth transition of the surface. Compared with other methods, the bump has potentials to hold a good balance in the aspects of boundary layer removal, external drag control, and shock system simplification. The Advanced Compact Inlet System project [29,30] obtained that the bump inlet is considerably better in aerodynamic performance than the conventional Pitot and Caret inlets. After a successful test on X-35, the bump concept was applied in the F-35 aircraft in 2006. Thereafter, the Lockheed Martin company applied for a patent about the bump inlet design with multistage shocks in 2007 [31]. Lo [32,33] combined the rounded bump with active blowing jet to investigate the flow pattern around the bump in Mach 1.3. Zhang [34,35] conducted experiments of a 2D bump in a supersonic inlet by using the changes in bump shape to control the shock system in Mach 3.38.

Nevertheless, the cone-derived bump faces several challenges in high Mach number. The bump surface is uniquely determined by the conical shock configuration and the streamlines after the cone shock. Thus, the overlarge bump height will induce large external drag in designing bump in supersonic or hypersonic speeds. On the other hand, the flow field around bump becomes non-uniform due to its concave shape. Therefore, the overlarge bump height also obtains a more non-uniform flow field. This configuration brings down the flow uniformity of the inlet inflow and aggravates the flow quality of inflow. Although many studies have been made to divert boundary layer by bump in Mach > 4, the design method of bump should be developed to efficiently divert boundary layer in hypersonic speeds.

According to the analysis above, a design method with improved flexibility to load pressure distribution on the bump surface is needed for high speeds. For realizing this target, the inverse design method is introduced to replace the conventional conical method. Different from other aerodynamic design methods, the inverse design method uses pressure/velocity distribution to generate the target surface. Hence, the inverse design method has high efficiency and is being developed. Braembussche [36] proposed a component design method for turbomachinery blades, and this method predicts the geometry modification values by using the results of the Navier–Stokes solver. Roidl and Ghaly [37] extended the inverse design method from the blade rows to the stage design and implemented iterative geometry modification based on CFD analysis. Wang and Li [38] used the adjoint method in the blade design. In this method, the geometry is changed to approach a target static pressure loading on the blade surface. Their continuous adjoint method made the design efficient by minimizing the gap between the modified geometry and the target geometry. Von Karman Institute for Fluid Dynamics [39,40] proposed the permeable boundary method, which modifies the geometry quickly until the transpiration values of the surface reach zero. Compared with other inverse design methods, the permeable boundary method is more efficient. The inverse design methods have been widely applied in turbomachinery blade design, and the efficiency is mainly determined by geometry modification mechanism, such as the performance of the methods in treating the feedback of CFD results. Meanwhile, optimizing the target is easier for 2D flows but more difficult for 3D geometries in which secondary flows have an important impact on losses.

This study uses the permeable boundary method to realize inverse design. Different from conventional conical-flow theory, the new method can generate bump surface inversely by the prescribed surface pressure distribution. To reach the goal, the permeable boundary method is extended to supersonic condition. Meanwhile, the way to prescribe pressure distribution is studied. To validate the design methodology, a wind tunnel experiment of Mach 6 is performed in the hypersonic wind tunnel (NHW) of Nanjing University of Aeronautics and Astronautics.

2. Description of the pressure controllable bump concept

2.1. Analysis of the integration pattern for hypersonic speeds

Two integration patterns of bump with a single-channel inward-turning inlet are depicted in Fig. 1. The two inlets have identical projected profile of the inlet entrance. The left one demonstrates the integration pattern of bump and a forward-sweep inlet. At the design condition, the incoming flow attaches on the inlet lip and the incident shock (colored in brown) impinges on the inlet bottom wall. The converging point is shown in the subplot of osculating plane arrangement. At the condition of low Mach number, the impinging position of the incident shock will move upstream, but no apparent flow spillage will occur until the incident shock moves to the inlet lip. Hence, this integration pattern cannot spill the mass flow away and induces inlet unstarting finally. The right pattern in the figure demonstrates the integration pattern of bump and a backward-sweep inlet. At the design condition, the incoming flow attaches on the inlet lip and the incident shock (colored in carmine) impinges on the most downstream of the lip. At the condition of low Mach number, the impinging position of the incident shock detaches from the inlet lip, and part of the inflow spills away after passing through the incident shockwave.

Apart from the low-Mach starting ability, the compression efficiency is also different between the two integration patterns.