Aerospace Science and Technology 74 (2018) 160–172

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

Design method of internal waverider inlet under non-uniform upstream for inlet/forebody integration

Guoping Huang, Fengyuan Zuo *, Wenyou Qiao

College of Energy and Power Engineering, Nanjing University of Aeronautics and Astronautics, Nanjing, Jiangsu, 210016, China

ARTICLE INFO

Article history:
Received 3 November 2017
Received in revised form 28 December 2017
Accepted 2 January 2018
Available online 31 January 2018

Keywords:
Internal waverider inlet
Three-dimensional inverse characteristic method
Non-uniform flow
Hypersonic

ABSTRACT

A novel Bump-integrated three-dimensional internal waverider inlet (IWI) design method is presented for high-speed inlet/forebody integration. The low-kinetic-energy (boundary layer) flow generated by a blunted leading-edge and forebody boundary layer represents an extreme challenge in the integration of aircraft forebody and inlet. In this method, such an inlet's flowpath is divided into the entrance shockwave segment, the isentropic compression segment and the isolator. First, a three-dimensional inverse method of characteristics (3D-IMOC) is developed to obtain a compression surface that can generate a requested entrance shock wave in non-uniform upstream flow. This configuration realizes the integration of IWI and aircraft fuselage by incorporating a Bump to remove most of the boundary layer flow. This is followed by a three-dimensional, isentropic compression flow-path with cross sectional areas conforming to the specified Mach number distribution. Finally, a new three-dimensional Bump-integrated IWI was tested in $M = 6$ wind tunnel, under a rather thick boundary layer upstream flow (37% height of inlet entrance). Both of the experimental data and numerical simulation results show that, the new method of IWI and Bump can overcome serious boundary layer flow problems and improve the inlet performance.

1. Introduction

High Mach number air-breathing vehicles (i.e., hypersonic vehicles for which the Mach number is higher than 5 and high-speed vehicles for which the Mach number is 3∼5) have become an intense research focus in the aerospace industry. The air-breathing propulsion system is crucial to achieving high-Mach-number flight [1]. Currently, the study of propulsion systems involves a multidisciplinary integrated system engineering research effort combining aerodynamics, thermodynamics, structural strength, flight trajectory simulation, and multidisciplinary optimization and involving a series of research directions [2,3]. Many studies have shown that integrating the design of the propulsion system and the fuselage is key to high-speed flight; most importantly, one crucial point is to design an integrated configuration of the aircraft forebody and inlet system. In the design process, the fuselage and inlet have their own requirements so that substantial contradictions and difficulties are likely encountered during integration [4].

The design of the aircraft forebody requires a high lift-to-drag ratio, pre-compression of the incoming stream to provide a high quality intake entry flow, and good mechanical strength and thermal protection [5]. Regarding thermal protection, the aircraft forebody must have an adequate leading edge radius. The heat load is generally inversely proportional to the square root of the radius of the stagnation point. In high-Mach number flight, the blunt leading edge produces a very small bow shock, and the shock wave in the front of the region is similar to normal shock compression, resulting in a thick layer of entropy [6]. The entropic layer wraps around the vehicle's surface from the leading edge and quickly mixes with the downstream boundary layer, which is also quite thick at high Mach numbers, forming a thick, boundary layer flow layer. Engine thermodynamic cycle studies have shown that for an M5-7 hydrocarbon-fueled vehicle, when the performance of the inlet decreases by 1%, the specific impulse of the propulsion system decreases by 3% to 5%. Furthermore, the thickness of such boundary layer on the fuselage is very large relative to the inlet system and may even be more than 50% of the height of the inlet; thus, it exerts a significant negative impact on the inlet performance. In these cases, the use of a conventional boundary layer diverter is clearly not realistic because it will cause great spillage drag, structural weight and other issues. However, if the boundary layer flow is swallowed by the inlet, it will seriously affect the inlet

* Corresponding author at: College of Energy and Power Engineering, Nanjing University of Aeronautics and Astronautics, 29 Yudao Street, Nanjing 210016, China.
E-mail address: zuofy@nuaa.edu.cn (F. Zuo).

https://doi.org/10.1016/j.ast.2018.01.012
1270-9638/

Nomenclature

IWI	internal-waverider inlet	ϕ	mass capture coefficient
3D-IMOC	three-dimensional inverse method of characteristics	V	velocity
3D	three-dimensional	α	the semi-apex angle with the major half axes of the ellipse cone
REST	rectangular-to-elliptic shape transition	β	the semi-apex angle with the minor half axes of the ellipse cone
R	radial distance	X, Y, Z	Cartesian coordinates
L	length		
H	height		
W	width		
γ	specific heat ratio		*Subscripts*
M	Mach number	0	free stream
π	pressure ratio	t	throat properties
σ	total pressure recovery	e	exit properties

performance, not only decreasing total pressure recovery, capture coefficient and surge margin but also potentially leading to starting problems that would result in the failure of whole propulsion system. Therefore, the design process of the fuselage should consider the issue of thermal protection to minimize the downstream effects of using a small blunt leading edge. In addition, the design integration of the aircraft forebody and the inlet must also be achieved to overcome the forebody boundary layer flow problem.

Based on the above statements, new methods of addressing the forebody boundary layer flow are required. Currently, based on research on the hypersonic inlet, several treatments have been developed for the boundary layer flow problem, including (a) boundary layer bleeding [7–11], (b) forced boundary layer transition (vortex generators) [12–16], and (c) magnetohydrodynamic control [17].

(a) For boundary layer bleeding, the temperature in the hypersonic boundary layer is relatively high. As a result, determining how to move such high-temperature airflow while considering the bleeding system settings is a serious problem; such high-temperature outflow also results in the loss of energy, which eventually affects the increment of the vehicle drag (similar to the diverter). These disadvantages dramatically limit the application of the bleeding method to hypersonic vehicles.

(b) The forced boundary layer transition method has been successfully applied in the X-43A hypersonic vehicle by setting a spanwise array of vortex generators [18]. Because the ingestion of a turbulent boundary layer suppresses the flow separation in the compression system and increases inlet operability (i.e., by reducing the susceptibility to flow separation within the engine), the overall engine performance is enhanced. Although the forced boundary layer transition can suppress the flow separation, the problem of swallowing the boundary layer flow has not been resolved.

(c) The magnetohydrodynamic control method, which is a recently developed research direction, has attracted the attention of many scholars in recent years. In this technique a magnetic field is applied to plasma to alter airflow direction. Unfortunately, the plasma excitation requires a large amount of energy, and the prospects for applying this method in the high-speed field are poor.

A novel type of Bump inlet was used in the U.S.-developed F-35 aircraft. This inlet compresses the airflow via a three-dimensional Bump surface. The Bump produces a transverse pressure gradient which diverts the forebody boundary layer to the sides of the intake entry [19]. This Bump surface not only weakens the interaction between inlet shock waves and boundary layer, but also obviates the diverter, bypass system and other related complex configurations of conventional inlets. As a result, it reduces the weight of the aircraft and decreases the flight drag, among other beneficial effects. In 2009, Lockheed Martin Corporation introduced a diverterless hypersonic inlet with a long and narrow shape, as shown in Fig. 1, and was granted a U.S. patent (U.S. Patent 7,568,347). This new Bump-compression surface has been computationally proven to be highly effective in boundary layer reduction above Mach 3, which is important for ramjet and scramjet applications [20]. Then, Benjamin J.T. collaborated with Boeing Company [21] to conduct experimental studies of a Mach 3 typical Bump-compression surface in a blowdown-type wind tunnel, shown in Fig. 2. The experiments showed that the Bump induced a curved shock system originating from the leading edge of the compression surface without causing flow separation and yielding favorable pressure gradients in both the streamwise and spanwise directions to push the boundary layer flow to the sides. However, this Bump-compression surface still has a relatively high height with large external drag, producing challenges associated with integration with the airframe-propulsion system.

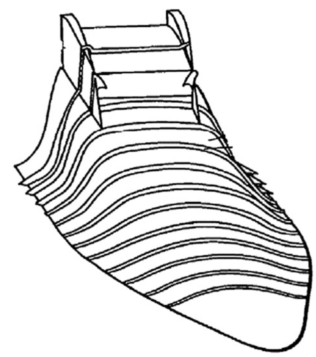

Fig. 1. The high-speed Bump-compression surface introduce by Lockheed Martin [20].

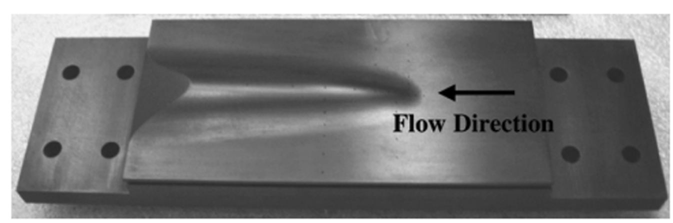

Fig. 2. Three-dimensional view and photograph of the Boeing Bump-compression surface [21].

On the another side, axial flow inlets, which utilize a series of compressive basic flows and have a specified entrance and exit shape [22], are receiving increasing attention as an optimum type of high-speed inlets in the context of academic research and practical aircraft development (e.g., the hypersonic SR-72 vehicle [23] developed by Lockheed Martin Corporation, shown in Fig. 3). Com-