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# Control of flow separation on a contour bump by jets in a Mach 1.9 free-stream: An experimental study

Kin Hing Lo*, Hossein Zare-Behtash, Konstantinos Kontis

University of Glasgow, School of Engineering, University Avenue, G12 8QQ, UK

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ABSTRACT

Flow separation control over a three-dimensional contour bump using jet in a Mach 1.9 supersonic free-stream has been experimentally investigated using a transonic/supersonic wind tunnel. Jet total pressure in the range of 0–4 bar was blowing at the valley of the contour bump. Schlieren photography, surface oil flow visualisation and particle image velocimetry measurements were employed for flow visualisation and diagnostics. Experimental results show that blowing jet at the valley of the contour bump can hinder the formation and distort the spanwise vortices. The blowing jet can also reduce the extent of flow separation appears downstream of the bump crest. It was observed that this approach of flow control is more effective when high jet total pressure is employed. It is believed that a pressure gradient is generated as a result of the interaction between the flow downstream of the bump crest and the jet induced shock leads to the downwards flow motion around the bump valley.



## 1. Introduction

Research on two- and three-dimensional contour bumps is an active research topic in the aerospace sector because of their applicability in both transonic and supersonic vehicles. Studies on wave drag reduction using contour bumps in transonic aircraft wings have been well documented in the literature [1–14]. It was found that around 10 to 20% of wave drag reduction could be achieved by using this flow control strategy. In addition, NASA proposed the concept of the Diverterless Supersonic Inlet (DSI) which implemented three-dimensional contour bumps as part of the supersonic inlet in 1950s [15]. Later studies concluded that compared to the other conventional supersonic inlet configurations, DSI could achieve higher total pressure recovery and lower flow distortion in supersonic speeds [16–20]. The results from these studies eventually materialised and DSI was first implemented into the engines of the Lockheed-Martin F-35 Lightning supersonic fighter aircraft [21].

Although using contour bumps could provide desire performance in drag reduction and high total pressure recovery in transonic and supersonic aircraft, it is known that adverse effects can be induced by flow separation and spanwise vortices formation appear downstream of the bump crest of the bumps [3,14,17]. As a result, it is important to investigate the flow separation characteristics of contour bumps in order to have better understanding in the physics of bump flow. Surprisingly, only a few studies have been conducted in this area for both subsonic and supersonic speeds. Although the subject matter of the present study is contour bump flow separation control in supersonic speed, some background information about the flow physics of contour bumps in subsonic free-stream is also included to provide a more complete literature survey.

In subsonic flow, Byun [22] and Byun et al. [23] investigated experimentally the formation of spanwise vortices downstream of the bump crest for a range of three-dimensional rounded contour bumps. The authors concluded that the number and size of the spanwise vortices that formed depended on the width and apex height of the bump. In addition, large-scale three-dimensional flow structures were observed which indicated the presence of a wake region downstream of the bump crest. Recently, Yakeno et al. [24] numerically investigated the streamwise flow pattern over a two-dimensional rounded contour bump in laminar flow. The results obtained suggest that flow separation appears immediately downstream of the bump crest which leads to the formation of three-dimensional flow structures in the wake region. The authors found that their size and shape depend on the Reynolds number of the flow and a similar conclusion was also drawn by Iaccarino et al. [25] in turbulent flow.

Lo [26], Lo and Kontis [27] and Lo et al. [28,29] investigated experimentally the flow pattern around a three-dimensional rounded contour bump in both Mach no. 1.3 and 1.9 supersonic free-stream. Experimental data showed that flow separation did appear immediately downstream of the bump crest which led to the formation of a large wake region. In addition, the authors in these studies showed that counter-rotating spanwise vortices were formed in the bump valley which is agreed with the finding obtained by Konig et al.

* Corresponding author.
E-mail addresses: kinhing.lo@glasgow.ac.uk (K.H. Lo), hossein.zare-behtash@glasgow.ac.uk (H. Zare-Behtash), kostas.kontis@glasgow.ac.uk (K. Kontis).

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[3]. In addition, the results shown in these studies suggested that the size of the wake region and spanwise vortex pairs that formed downstream of the bump crest decreased when the free-stream Mach number was increased from $M_\infty = 1.3–1.9$.

Svensson [30] conducted a numerical study to investigate the streamwise and spanwise flow patterns over different rounded three-dimensional contour bumps in both subsonic, transonic and supersonic free-stream to investigate the applicability of contour bumps with various geometries in DSI. The author concluded that for a contour bump with a given bump width and apex height, the size of the wake region and the spanwise vortices that formed in the bump valley increases with increasing free-stream Mach number ($M_\infty$) when $M_\infty < 1$. In contrast, when $M_\infty > 1$, the size of the wake region and the spanwise vortices decreases with increasing free-stream Mach number. This is agreed with the results shown by Lo [26] and Lo et al. [28,29].

It is clear that the occurrence of flow separation and the formation of the spanwise vortices increase the pressure drag generated by the bumps. In addition, when contour bumps are used in DSI, these effects lower the total pressure recovery that can be achieved and also affect the uniformity of flow entering the engines [17,30]. Therefore, effective measures must be established in order to achieve flow separation control in contour bumps. Jet blowing is one effective way to achieve flow separation control and it has been extensively investigated in the subsonic flow regime. In contrast, using continuous jet blowing in flow separation control in supersonic flow is less common. The studies conducted by Zubkov et al. [31], Glagolev et al. [32,33], Glagolev and Panov [34] and very recently Beketaeva et al. [51] were some of the earliest experimental studies to investigate the interaction between the injected sonic/supersonic jet and the supersonic free-stream. In these studies, the gaseous jet was injected from a flat plate through orifices of different sizes. Experimental data in these studies showed that complicated three-dimensional compression waves are formed upstream of the injected jet. In addition, a pair of counter-rotating horseshoe-shaped vortices appears immediately downstream of the injected jet and the tips of these horseshoe vortices propagated downstream from the jet.

One of the first experimental studies that investigated the flow physics of supersonic flow past a flying object with sonic and supersonic jet injection employed was conducted by Zubkov et al. [35]. Experimental data in [35] showed that the interaction between the injected jet and the supersonic free-stream leads to the formation of a shock wave immediately upstream of the jet. In addition, counter-rotating vortices were observed in the leeward side of the model. Koike et al. [36] found that by blowing jet continuously at the leeward face of a micro-ramp in a Mach 2.6 free-stream could reduce the extent of flow separation that occurred. The authors explained that the blowing jets reduced the size of the spanwise vortices that formed and also led to the formation of a streamwise vortex pair downstream of the micro-ramp. These streamwise vortices facilitate flow mixing between the low energy boundary layer and the high energy free-stream. As a result, the boundary layer was re-energised and thus delayed flow separation. Lo et al. [26,28] experimentally investigated flow separation control in a three-dimensional contour bump using active blowing jet in a Mach 1.3 free-stream. Similar to the conclusion obtained by Zubkov et al. [35] and Koike et al. [36], the authors found that the blowing jet hindered the formation of the spanwise vortices and also reduced the size of the wake region that appeared in the bump valley.

The present study aims to extend the previous studies conducted by Lo et al. [26,28] to a higher free-stream Mach number to simulate the working conditions of a diverterless supersonic inlet. This experimental study aims to firstly look at the flow physics of a three-dimensional rounded contour bump with and without active sonic jet blowing involved in a Mach 1.9 free-stream. Secondly, the effect of the jet total pressure in affecting the streamwise and spanwise flow patterns over the contour bump is also included in this study.

## 2. Experimental setup

### 2.1. Tri-sonic wind tunnel

All of the experiments in this campaign were conducted in an intermittent in-draught type tri-sonic wind tunnel. An intermittent in-draught type wind tunnel means that the airflow inside it is maintained by means of a pressure difference between the atmosphere (upstream) and vacuum (downstream). A schematic of this wind tunnel facility is shown in Fig. 1. The same trisonic wind tunnel was also employed in the experimental studies conducted by Lo [26], Lo and Kontis [27], Lo et al. [28,29], Zare-Behtash et al. [46,47], and Ukai et al. [48–50]. The wind tunnel has a rectangular test section with dimensions of 485.5 mm (length) × 150 mm (width) × 216 mm (height). The three-dimensional rounded contour bump model was floor mounted at the middle of the wind tunnel test section. Optical access is achieved through the two quartz side and the top windows which is also made of quartz. A quick opening butterfly valve is situated between the test section and the vacuum tank. When the butterfly valve is opened, a pressure difference is generated between the upstream of the wind tunnel and the vacuum tank. As a result, a stable airflow is developed inside the wind tunnel. The required Mach 1.9 supersonic free-stream was generated by expanding the airflow inside the wind tunnel through a pair of convergent–divergent nozzles situated upstream of the test section.

The free-stream Mach number ($M_\infty$) at the wind tunnel test section was calculated from the total pressure ratio between the upstream and at the test section of the wind tunnel [37]. Pitot probes were inserted into the wind tunnel to obtain information about total pressures at the two different locations. The end of each pitot probe was connected to a Kulite XT-190M pressure transducer via flexible tubes. The voltage signals from the pressure transducer were captured by a National Instruments (NI)

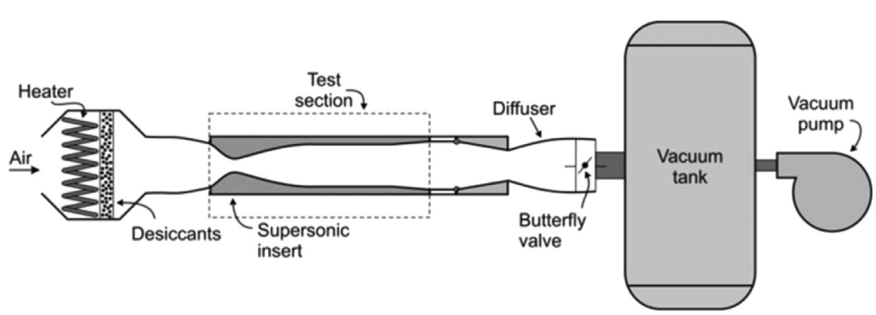


**Fig. 1.** Schematic of the tri-sonic wind tunnel.