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Effects of trips on the oscillatory flow of an axisymmetric hypersonic inlet with downstream throttle

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Abstract Experimental investigations are conducted on an axisymmetric hypersonic inlet to evaluate the effects of trips on oscillatory flows. The model exit is throttled with a fixed block to generate oscillatory flows at a freestream Mach number of 6 in a conventional wind tunnel and a shock tunnel. Schlieren imaging and pressure measurements are adopted to record unsteady flow features. Results indicate that trips with a 1 mm thickness prominently suppress external separations, shorten oscillatory cycles, and modify pressure magnitudes. Trips can reduce the upstream movement ranges of separated shocks from nose regions to locations axially 142 mm downstream. The oscillatory cycles are shortened from 3.75 ms to 3.25 ms and from 4 ms to 3.13 ms in two facilities. Tripped cases generally exhibit higher pressure magnitudes than those of untripped cases, of which the increment is up to 21 times the freestream static pressure for the farthest downstream transducer in the shock tunnel. The effects of trips are related to the streamwise vortexes in wake flows, in which interactions between external separations modify the separated flow patterns and enhance the sustainment of the forebody boundary layers to backpressure. Flow processes causing increments of oscillatory frequencies and pressure magnitudes are analyzed, while the flow mechanisms dominating the processes still need to be clarified in the future.

1. Introduction

Hypersonic inlets must operate in a started mode for efficient operations as the air capture and compression portion of hypersonic airbreathing vehicles.[1] However, a hypersonic inlet can be unstarted through various factors, such as improper design, low flight Mach number, and high backpressure, among others. Among these factors, a high backpressure is induced by fuel injection and combustion in the combustors,

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which is encountered at the acceleration or cruise stages of flights.

Unstart problems caused by combustor backpressures have been largely investigated in wind tunnels[2–5] or through numerical simulations.[6,7] High combustor backpressures are mimicked primarily with mechanical throttling of flow,[2–4,8–10] mass injection,[11,12] or heat release.[5,13] According to prior studies, separated flows emerge and propagate upstream under the influences of backpressures. These separated flows can remain at a certain location with a quasi-steady motion of shock trains under a relatively "low" backpressure. Once the backpressure is sufficiently high, separations would propagate upstream and eventually unstart the inlet flow. Then, oscillatory flow can appear as a violently unsteady mode of unstart flow.[2–4,8] Wagner et al.[2] observed a high-amplitude oscillatory flow in an inlet/isolator model as the inlet unstarted. Tan et al.[3,8] investigated oscillatory flows and the unstarting process of a rectangular inlet. Substantial separations and pressure fluctuations were observed and specified as "big" or "small" buzzes according to oscillatory amplitudes. Li et al.[4] studied the unsteady behaviors of a two-dimensional inlet in a shock tunnel; the results indicated that the unstart flows were accompanied with a shock wave oscillation.

Oscillatory flows are highly detrimental to inlet performances and flight operations. Flow separations and spillages reduce the captured oxidizer, leading to combustor flameout and thrust loss. Pressure fluctuations and shock oscillations generate unsteady aerodynamic loads that may harm structural safety. Therefore, preventing oscillatory flows is necessary. Based on the findings of previous works, some studies have been conducted to prevent or delay oscillatory/unstart flows,[14–16] using mainly active control methods. Valdivia et al.[14] investigated the effects of energizing the sidewall boundary layer of an isolator on the unstart process. Vortex generators, vortex generator jets, and their combinations were tested, and the combined method improved the inlet's sustainment to backpressure and reduced pressure fluctuations. Im et al.[16] used a plasma actuator to manipulate unstart flow. The plasma actuation could arrest boundary layer separations and delay the unstart process.

Additional instruments of manipulation would increase flight loads and complicate vehicle operations, although the above control devices can provide effective mitigations on unstart flows. Passive flow control devices, such as vortex generators, may be suitable and are worth studying in terms of robustness. Furthermore, discrete roughness arrays (also called trips) mounted on the forebodies of flight vehicles[17,18] can be a more suitable choice than vortex generators, because these trips are part of the vehicles. Trips are designed to enhance boundary layer transitions through streamwise vortexes generated in wake flows. Previous studies showed that trips could suppress the separations of started flows[19] or mitigate unstarts caused by large bluntness,[20] which are similar to the roles that vortex generators exert on separated flows.[21] However, to the best of the authors' knowledge, few studies have focused on the effects of trips on oscillatory flows.

Consequently, the effects of trips on the oscillatory flows of a hypersonic inlet are investigated experimentally in this paper. An axisymmetric inlet model is selected to avoid corner vortexes encountered in configurations with sidewalls. Schlieren imaging and transient pressure measurements are adopted to record unsteady flow features during experiments. The effects of trips on oscillatory flows are analyzed, and instructions for practical applications are provided.

2. Experimental facilities and methods

2.1. Facilities and experimental conditions

Experimental studies are conducted in a conventional wind tunnel running in a blowdown mode at the Nanjing University of Aeronautics and Astronautics[22] and a shock tunnel at the University of Science and Technology of China.[4] Detailed descriptions of the two facilities can be found in the previous literatures. The shock tunnel is a facility of the authors' institution and can be operated conveniently. Corresponding studies are also conducted in the conventional wind tunnel, because studies on oscillatory flows of a hypersonic inlet have rarely been conducted in impulse facilities.

In Table 1, the freestream Mach numbers of the conventional wind tunnel and the shock tunnel are 6.0 (nominal) and 5.9, respectively. A corresponding total temperature and a total pressure of both wind tunnels are regulated to supply the freestream with unit Reynolds numbers of $5.41 \times 10^6\ \mathrm{m}^{-1}$ and $5.03 \times 10^6\ \mathrm{m}^{-1}$, respectively. The experimental time of the conventional wind tunnel is much longer than that of the shock tunnel. Based on the time (t) history of the total pressure(p_t) in Fig. 1, the conventional wind tunnel can supply 4 s of stable flows in a blowdown mode, while the experimental time of the shock tunnel is approximately 20 ms for equilibrium interface operations. Each case is fired twice in the shock tunnel to ensure repeatability of experiments in the impulse facility.

2.2. Test model and measurements

The test model is an axisymmetric hypersonic inlet designed with a shock-on-lip Mach number of 6.5.[23] The model shown in Fig. 2(a) has an inlet capture diameter of 128 mm, and the forebody nose is blunted with a radius of 0.8 mm. The captured flows are compressed with an initial conical surface inclined at 10° and the following curved surfaces with a total turning angle of 9.7° upstream from the inlet entrance. The flow-turning angle at the entrance is 9°, while the internal compression section smoothly transits to a 50 mm horizontal isolator with a height of 4.72 mm. The total and internal contract ratios of the inlet are 6.41 and 1.58, respectively. A 90 mm long

Table 1 Test conditions of the conventional wind tunnel and the shock tunnel.

Parameter	Conventional wind tunnel	Shock tunnel
Freestream Mach number	6.0 (nominal)	5.9
Total temperature (K)	570	920
Total pressure (MPa)	0.8	1.5
Unit Reynolds number (10^6/m)	5.41	5.03
Experimental time (s)	4	0.02
Diameter of nozzle exit (mm)	500	300