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Effect of Mach number and equivalence ratio on the pressure rising variation during combustion mode transition in a dual-mode combustor

Chenlin Zhang, Juntao Chang *,1, Jingxue Ma, Wen Bao, Daren Yu, Jingfeng Tang

Harbin Institute of Technology, 150001 Heilongjiang, People's Republic of China

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A B S T R A C T

Experiments of equivalence ratio linear increasing are utilized to employ to investigate the characteristics of combustion mode transition. For the experiments with kerosene of different increasing slope, it is found that the fuel equivalence ratio has a significant impact on pressure rising slope variation during combustion mode transition. In the experiments, the big equivalence ratio slope would not lead to slope variation of pressure rising. Similarly, this phenomenon also could not occur under some high Mach number of incoming flow experiments. Different equivalence ratio slope would lead to different border shape of the boundary layer. And the border shape of the boundary layer and normal shock wave at thermal throat restrains the main flow is an important factor to alter the pressure. Gas dynamics analysis is employed to explain the effect of the boundary layer on the pressure rising slope variation. Further, the analysis also reasonably explains that incoming flow Mach number impacts on pressure rising slope variation.

1. Introduction

Dual-mode supersonic ramjet (scramjet) is deemed to be the most promising option for hypersonic flight [1–3]. By operating in scramjet mode of high flight Mach number and in ramjet mode of low flight Mach number, the dual-mode combustor could fulfill a series of flight mission in a wide range of flight Mach number [4–8]. Under low flight Mach number, the ramjet-mode combustor could keep a good performance because the main flow is compressed into subsonic airflow by shock train in isolator and subsonic is effective to reduce the Rayleigh loss in combustor. Under high Mach number flight, strong shock compression loss would lead to static temperature increasing in isolator, so supersonic combustor is required to work in a scramjet mode. The combustion mode would be transformed when the flight Mach number or equivalence ratio is altered. The combustion modes in dual-mode combustor are different from the combustion modes which include deflagration and detonation [9,10]. It is produced during dual-mode combustor working process. How to regulate the combustion modes of combustor in flight process has a significant effect on obtaining a better performance. Additionally, some special phenomena (thrust sudden change, hysteresis loop) during combustion mode transition would arouse some special problems and increases the control difficulty during its decelerated or accelerated flight process [11,12]. These mechanisms of the extraordinary phenomena need to be further investigated.

Just as mentioned above, many special phenomena during combustion mode transition process have been observed and investigated in previous research. T. Kouchi [13] found there is a sudden thrust increase during the process of combustion mode transition when the overall fuel equivalence ratio was added to 0.4. By analyzing OH radical distribution and flame index distribution, the driving force of the mode transition was combustion-generated high pressure. J.C. Turner [14] undertook experiments to investigate the process of equivalence ratio increasing, and found that an abrupt thrust increase at an equivalence ratio less than 1. B.G. Xiao [15] also found an experimental pressure sudden change accompanying with the equivalence ratio increasing in a combustor with cavity, and further identified and classified two combustion modes by pressure. When the pressure ratio was less than 1.5, the combustor operated in a scram-mode, otherwise a ram-mode. In the process of combustion mode transition, Matthew [16–18] obtained different pressure behavior of combustion mode transition and found that there was a sudden change in the wall pressure. The flame position was moved as the downstream boundary condition was changed abruptly and the flow became un-chocked. The sud-

* Corresponding author.
E-mail address: changjuntao@hit.edu.cn (J. Chang).
1 Academy of Fundamental and Interdisciplinary Sciences.

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Nomenclature

α	deflection angle	k	specific heat ratio
β	expansion angle	t	time
θ	shock wave angle	x	distance in the flow direction
Φ	equivalence ratio	Ma	Mach number
A	cross section area of core flow	*Subscripts*	
F	force	1	isolator entrance
p	static pressure	2	isolator exit
p_b	pressure ratio		

den change of pressure had been claimed from many different supersonic combustor configurations from different papers. This is a common phenomenon in combustion mode transition. With equivalence ratio and incoming flow Mach number variation, the abrupt boost of pressure on sidewalls occurs would lead to the thrust and control issues. Additionally, other special phenomenon was also found in combustion mode transition. W. Bao [11] found a nonlinear hysteresis behaviors of mode transition. For the same equivalence ratio in the loop path, wall pressure presented a disparate distribution. It means that pressure distribution is influenced by the equivalence ratio path. T. Cui [19] promoted a mechanism model to explain the hysteresis phenomenon of mode transition. In supersonic combustor, the combustion mode has a direct effect on the combustion zone. D.J. Micka [20] found that there are two distinct combustion stabilization locations in combustion modes by altering the total temperature of the incoming flow. Under the ramjet mode, one is the jet-wake stabilization location and another is the cavity stabilization location. The scramjet mode operation only exists in the cavity stabilized location. Similarly, H.B. Wang [21,22] indicated there are three combustion modes in the configuration with cavity acts as a flame holder by increasing cavity length or equivalence ratio, or decreasing distance between the fuel injection and the cavity. In the process of combustion mode transition, the combustion zone could be transferred the locations and arouse airflow parameters alteration. In a word, combustion mode transition could bring many complex flow and combustion issues.

Supersonic and subsonic zone owns a different property for combustion and flow. The complicated interaction of combustion and flow produces the special complex phenomenon. Many special nonlinear phenomenon (sudden change, hysteresis loop) during combustion mode transition increasing difficulties of engine control. Some of them have been reported in many papers. To support dual mode scramjet engine development, there should be a deep need for greater understanding for the special phenomenon in the combustion mode transition process. The pressure rising slope variation phenomenon accompanying with combustion mode transition is found by detecting of pressure in the experiment. It is distinct from the pressure sudden change and its mechanism has been expounded in the reference [23].

In present paper, we found that the pressure rising slope variation has a close relationship with the equivalence ratio and the incoming flow Mach number by a series of experiments. The pressure rising slope variation would not occur in some experiments of high incoming flow Mach number and big equivalence ratio slope increasing. Numerical simulation indicates that the border of the boundary layer and normal shock wave at thermal throat restrains the mass flow rate and arouse pressure change in the previous paper [23]. Based on the compression and expansion process for the main flow, a flow analysis is proposed to clarify the pressure variation during normal shock wave occurrence. Additionally, the flow analysis is good to explain the effect of equivalence ratio slope variation and incoming flow Mach number on the pressure.

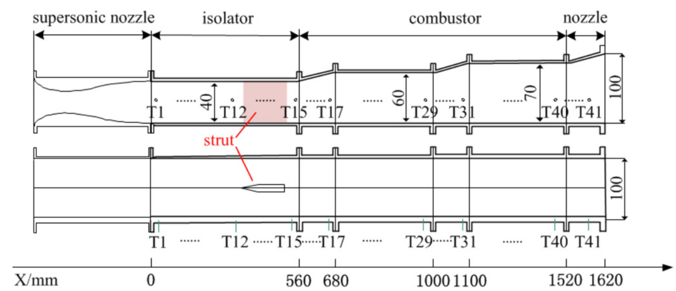

Fig. 1. Schematic diagram of isolator–combustor configuration with strut.

2. Experimental setup

The experiment was employed using the direct-connected scramjet experiment system at the Harbin Institute of Technology, China. Gas source system contains high-pressure air, oxygen and nitrogen source. It could ensure effective run time is kept over 100 s. To obtain the high enthalpy incoming flow, the gas from the air source could be heated to different total temperature through the alcohol-air combustion in a heater, which was located upstream of the supersonic nozzle. Additional oxygen is injected into the heater to maintain a 0.21 O_2 mole fractions in the heated products because of the oxygen is consumed in combustion. A two-dimensional Laval nozzle with an exit area of 40 mm × 100 mm was used to accelerate the high enthalpy air. By using nozzles with different area of throat, the different incoming flow Mach number could be obtained.

A schematic of the scramjet model was used in the current experiment is shown in Fig. 1. The model has a constant width of 100 mm. To connect with the nozzle exit, the area of isolator has a 40 mm × 100 mm cross section. The isolator–combustor configuration contains six segments. And three segments have different divergent angles and others are constant area. Detail sizes of every segment are shown in Fig. 1. A strut using to ignite and maintains flame stabilization is mounted at about 460 mm in the direction of flow, that there are 42 holes with a diameter of 0.6 mm spreading over both sides and trailing edge of the strut. Kerosene fuel at room temperature is injected from the strut in the direction normal to the air flow direction and it can also serve as active cooling agent and effectively relieves the aerodynamic heating on the strut.

To measure the static pressure in the combustor, a series of pressure-tap ports are distributed on the center line of the combustor side wall. Each port is connected with a low-frequency pressure transducer by hollow metal ducts, which could protect transducers from the high temperature of combustion. The measurement error of the transducers is 0.2% in full scale, and the range is 0 to 1.0 MPa. Hollow metal ducts are 2.5 mm diameters and 200 mm length, which will decay the cutoff frequency of pressure measurements. To decrease this decay as much as possible, the hollow metal ducts were filled with kerosene. According to a conservative estimation, the cutoff frequency of the system