Numerical Analysis of Dynamic Combustion in HyShot Model Scramjet

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1 Introduction

Development of hypersonic air-breathing propulsion engines is the key issue for the success of future high-speed air transportation. Although there are many technical challenges for the hypersonic engines, combustor is one of the core technologies. The flow entering a combustor at hypersonic flight speed should be maintained supersonic to avoid the excessive heating and dissociation of air. The residence time of the air in a hypersonic engine is on the order of 1ms for typical flight conditions. The fuel must be injected, mixed with air, and burned completely within such a short time span. A number of studies have been carried out worldwide and various concepts have been suggested for scramjet combustor configurations to overcome the limitations given by the short flow residence time. Among the various injection schemes, transverse fuel injection into a channel type of combustor appears to be the simplest and has been used in several engine programs, such as the HyShot scramjet engine, an international program leaded by the University of Queensland[1]. The present study attempts to achieve improved understanding of unsteady flow and flame dynamics in a real scramjet combustor configuration employing a transverse fuel injection.

2 Numerical Approach

2.1 Governing Equations and Numerical Methods

The flowfield is assumed to be two-dimensional for computational efficiency, and can be described with the conservation equations for a multi-component chemically reactive system. The coupled form of the species conservation, fluid dynamics, and turbulent transport equations can be summarized in a conservative vector form as follows.

$$\frac{\partial \mathbf{Q}}{\partial t}+\frac{\partial \mathbf{F}}{\partial x}+\frac{\partial \mathbf{G}}{\partial y}=\frac{\partial \mathbf{F}_\mathrm{v}}{\partial x}+\frac{\partial \mathbf{G}_\mathrm{v}}{\partial y}+\mathbf{W}$$

where **Q** is the conservative variable vector, **F** and **G** are convective flux vectors, $\mathbf{F}_\mathrm{v}$ and $\mathbf{G}_\mathrm{v}$ are viscous flux vectors, and **W** is reaction source term. The governing equations were treated numerically using a finite volume approach. The convective fluxes were formulated using Roe's FDS method derived for multi-species reactive flows along with the MUSCL approach utilizing a differentiable limiter function. The spatial discretization strategy satisfies the TVD conditions and features a high-resolution shock capturing capability. The discretized

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equations were temporally integrated using a second-order accurate fully implicit method. A Newton sub-iteration method was also used to preserve the time accuracy and solution stability.

2.2 Chemistry Model and Turbulence Closure

The present analysis employs the GRI-Mech 3.0 chemical kinetics mechanism for hydrogen-air combustion. The mechanism consists of 9 species (H, H_2, O, O_2, H_2O, OH, H_2O_2, HO_2 and N_2) and 25 reaction steps. Nitrogen is assumed as an inert gas because its oxidation process only has a minor effect on the flame evolution in a combustor. Turbulence closure is achieved by means of Mentor's SST (Shear Stress Transport) model derived from the k-ω two-equation formulation. This model is the blending of the standard k-ε model that is suitable for a shear layer problem and the Wilcox k-ω model that is suitable for wall turbulence effect. Baridna et al. reported that the SST model offers good prediction for mixing layers and jet flows, and is less sensitive to initial values in numerical simulations[2]. Another important issue is the closure problems for the interaction of turbulence and chemistry in supersonic conditions. Recently, there were many attempts to address this issue using LES methods, PDF approaches, and other combustion models extended from subsonic combustion conditions. Although many useful advances were achieved, the improvement was insignificant in comparison with the results obtained from laminar chemistry and experimental data, as discussed by Möbus et al[3]. A careful review of existing results, such as Norris and Edwards suggests that the solution accuracy seems to be more dependent on grid resolution than the modeling of turbulence-chemistry interaction[4]. In view of the lack of reliable models for turbulence-chemistry interactions, especially for supersonic flows, the effect of turbulence on chemical reaction rate is ignored in the present work.

3 Combustor Configuration and Operating Conditions

T4 free piston shock tunnel of UQ was used for the ground test under the conditions M=6.5, p=0.9-5.8kPa and T=285-291K. From the given conditions the total enthalpy is 3.0MJ/kg. The model scramjet consists of an intake, a combustor and a thrust plate and each size of components is of 305mm x 100mm, 300mm x 75mm and 200mm x 75mm respectively. The intake is a 17° inclined wedge which compresses the incoming hypersonic flow. The flow is further compressed by the combustor cowl, after which hydrogen is injected. Combustion occurs in the combustor and hot gases from the combustion process are expanded through the thrust plate hence producing thrust. The combustor has a constant rectangular area and 16 pressure transducers which are mounted orderly 90mm downstream from the combustor inner surface leading edge. Each distance between pressure transducers is 13mm. Four injectors with a 2mm diameter are located 40mm downstream from the combustor inner surface leading edge with hydrogen injected transversally into the incoming supersonic flow. For the two-dimensional numerical analysis, the four fuel injectors were assumed to be one long slot of 75mm x 0.168mm with the same area. In the flight test the sounding rocket reaches a maximum altitude of 315km and the scramjet is maneuvered into the experimental attitude before re-entry. Between altitudes of 35km and 23km, gaseous hydrogen is injected into the scramjet and pressure measurements are recorded. During the flight, the scramjet/rocket vehicle exposes changing conditions such as altitude (h), angle of attack (AOA), equivalence ratio (φ), spin, etc. Among these variations, the ground experiment using T4 free piston shock tunnel considered altitudes (h=35, 28, and 23km), angle of attack (AOA=0°, 4°, and -4°) and equivalence ratios (φ=0-0.75). The simulation focuses on the design point which corresponds to h=28km, AOA=0°, φ=0.426 respectively. In addition, two more cases (h=35 and 23km) are simulated to explain the effects according to altitude. The experimental data were taken at 1.2ms after the flowfield was established in the combustor and the same procedure was followed in simulations. The detail conditions are summarized in Table 1 for freestream, combustor inlet, and injector exit.

Table 1 Simulation Conditions (AOA=0°, ϕ = 0.426)

	Freestream (35 / 28 / 23km)	Combustor Inlet (35 / 28 / 23km)	Fuel Injector (35 / 28 / 23km)
P [kPa]	0.95 / 2.22 / 5.47	32.74 / 82.11 / 188.05	113.72 / 307.34 / 648.60
T [K]	306 / 311 / 307	1161 / 1229 / 1164	250 / 250 / 250
M	6.53 / 6.75 / 6.53	2.75 / 2.79 / 2.75	1.0 / 1.0 / 1.0

4 Results and Discussion

4.1 Surface Pressure and Flow Instability

In the typical flowfield generated by transverse injection, the underexpanded jet is injected normal to the freestream with sonic conditions over the injector exit. As the jet leaves the injector exit, it expands rapidly and penetrates the boundary layer, developing a system of shock and recirculation regions. In the upstream of the jet, a jet-induced bow shock is generated due to blockage of freestream and the induced shock causes the boundary layer separation which includes two counter-rotating vortices.

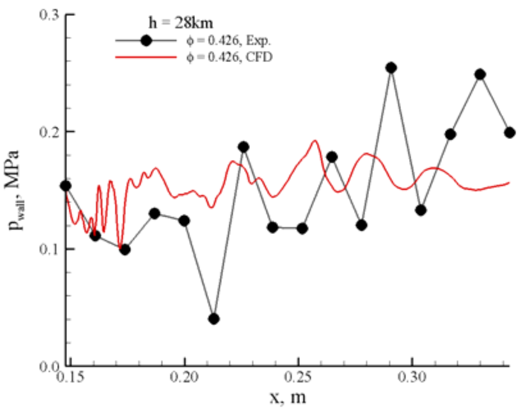

Fig. 1 Surface pressure between experiment and simulation (h=28km, AOA=0°, ϕ=0.426)

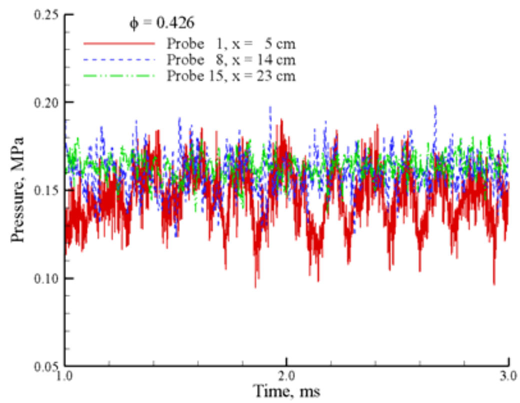

Fig. 2 Pressure-time history at x=5,14,23cm from fuel injector (h=28km, AOA=0°, ϕ=0.426)

Surface pressure distributions between numerical and experimental results for design condition were compared with each other in Fig. 1. Both experimental and numerical data were taken at 1.2ms after the flowfield was established in the combustor. In the experiment, the combustor wall pressure rise due to combustion is observed, particularly towards the rear of the combustion chamber. In the simulation, the pressure rise is also observed but the slope is somewhat gentle compared with experiment. This attributes to the absence of three-dimensional flow structures such as counter-roating vortex and horseshoe vortes. The pressure distributions, however, reasonably agree with each other. The temporal variations of a combustor wall pressure are shown in Fig. 2. The pressure was recorded at 5, 14, and 23cm from the fuel injector and these measurement points correspond to #1, #8, and #15 pressure transducers respectively. The mixed states between high-frequency related with intrinsic supersonic flow instability and low-frequency related with thermo-fluidic instability appear in the graph and this phenomena show the unsteadiness of turbulent combustion. The pressure-time history of #1 transducer reveals clear periodic characteristics as well. The principal low-frequency is about 6kHz.

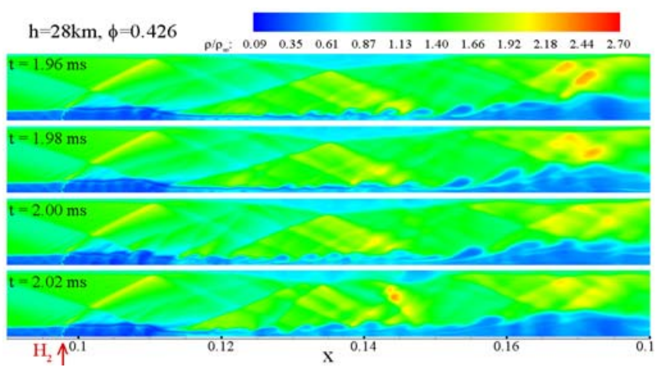

Fig. 3 Temporal variation of density flowfield (h=28km, AOA=0°, ϕ=0.426)

For a detail analysis of periodic characteristics, density contours are revealed in Fig. 3. The figures which include flow structures around injector are displayed at intervals of 0.02ms. The Richtmyer-Meshkov instability, when an interface between fluids of differing density is impulsively accelerated by the passage of a shock wave, is propagated into upstream through the subsonic boundary layer. This instability disturbs the flow structures around injector and then flows to downstream. As a result of disturbance, the mixing and combustion are enhanced. These processes are repeated and recorded in the pressure-time history as periodic forms.

4.2 Altitude Effects

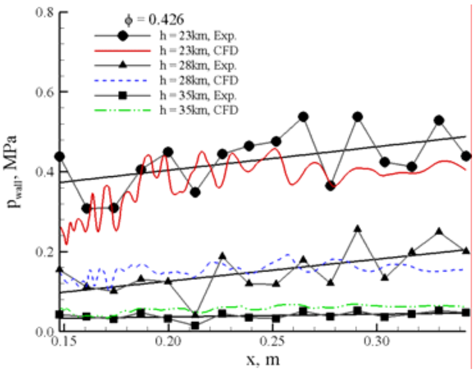

Fig. 4 Surface pressure between experiment and simulation (AOA=0º, ϕ=0.426)

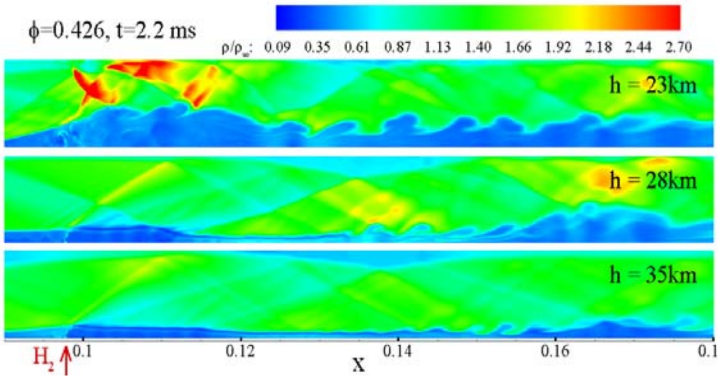

Fig. 5 Density contour comaprison along altitude (AOA=0º, ϕ=0.426)

Altitude effects are examined in Fig. 4 by comparing surface pressure distributions between numerical and experimental results. During altitude variation, the AOA and equivalence are fixed as design condition. The pressure rising characteristics as the measurement point moves towards the rear combustor are common regardless of altitude. The pressure distributions from two-dimensional simulations reasonably agree with all experimental cases. However, the pressure rising slope increases with an altitude decrease in both experimental and simulation results. Figure 5 shows the density contours and distinct differences are observed. According to an altitude decrease, the subsonic boundary layer thickness as well as the strength of Richtmyer-Meshkov instability increases. Here the subsonic boundary layer is a passage which the instability propagates through. As a result of above effects, there exist an ordinary instability in 23km altitude, a periodic instability in 28km altitude, and a flow converging to stable state as an initial instability disappears in 35km altitude. The combustion efficiencies along the altitude increase are 83.5, 61.8, and 45.0% respectively and therefore the pressure rising slope increases with an altitude decrease.

5 Summary and Conclusion

The turbulent combustion flow in HyShot scramjet combustor was carefully studied by means of a comprehensive numerical analysis. The simulations focused on the design condition (h=28km, AOA=0°, φ=0.426) and were extended to off-design conditions to investigate altitude effects. Comparisons between experimental and numerical results present that the two-dimensional simulations reasonably predict pressure distributions in combustor. The temporal variations of a combustor wall pressure reveal periodic characteristics and the principal frequency is about 6kHz at design conditon. These periodic characteristics can be explained by the Richtmyer-Meshkov instability propagation. Altitude effects are also investigated as off-design conditions. The residence time of instability is controlled by the strength of instability and subsonic boundary layer thickness. This affects the combustion efficiency and pressure rising slope according to altitudes.

References

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