SUPERSONIC COMBUSTION IN RAM ACCELERATOR AND SCRAMJET ENGINE COMBUSTOR

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OVERVIEW

A comprehensive DES quality numerical analysis has been carried out for reacting flows in a superdetonative mode ram accelerator and in constant-area and divergent scramjet combustor configurations with and without a cavity. Ram accelerator flow was modeled for the experiments of German scientists in the French-German Research Institute-Saint Louis, RAMC30. Stoichiometric Hydrogen and Oxygen mixture diluted by Carbon dioxide was used as propellant gas. This numerical study showed some possible supersonic combustion flow of shock-induced combustion partially sustained by near normal detonation waves.

For the case of scramjet engine, model combustor flight tested by the program HyShot was modeled. Transverse injection of hydrogen is considered over a broad range of injection pressure. The corresponding equivalence ratio of the overall fuel/air mixture ranges from 0.167 to 0.50. The work features detailed resolution of the flow and flame dynamics in the combustor, which was not typically available in most of the previous studies. In particular, the oscillatory flow characteristics are captured at a scale sufficient to identify the underlying physical mechanisms. Much of the flow unsteadiness is related not only to the cavity, but also to the intrinsic unsteadiness in the flowfield. The interactions between the unsteady flow and flame evolution may cause a large excursion of flow oscillation. The roles of the cavity, injection pressure, and heat release in determining the flow dynamics are examined systematically.

1. INTRODUCTION

In the morning of 17 December 1903, Orville Wright made the first flight at about 10:35 am, a bumpy and erratic 12 seconds in the air. And then, Wilbur flew the plane 175 feet-just a few feet shorter than the wingspan of a Boeing 747. During the final flight of the day, piloted by Wilbur, the Wright Flyer remained airborne for 59 seconds and flew 852 feet.[1]

One of heroic flight lifted from Hounslow, England landed in Darwin, Australia on 10 December 1919, by the pilots Sir Ross and Sir Keith Macpherson Smith born in Adelaide, was completed in 27 days and 20 hours for the long journey of 11,294 miles.[2]

About 100 years late, in 30 July 2002, the world first scramjet combustor, HyShot flight test payload, demonstrated a successful supersonic combustion during hypersonic flight of Mach 7.7 for 5 seconds flying about 12 km.[3] Soon, in 27 March 2004, NASA made the first successful hypersonic flight of Mach 6.8 by X-43A hypersonic airplane installed a scramjet engine for 10 seconds of powered flight, and again in 16 November 2004, made the second successful hypersonic flight of Mach 9.8.[4] Flight time of these hypersonic scramjet flight tests lasted about one-tenth of the Wright Flyer, while the flight distance extended to the hundred times longer range. Recently, there were another scramjet engine flight test campaigns, so called HyShot 3, 4, and HyCAUSE. We are dreaming the flight to any places on the earth within 2 hours in the near future by a hypersonic transport powered by the scramjet engines.

As mentioned, there have been several activities on the scramjet engines and the supersonic combustion from the Asia-Pacific countries, such as USA, Australia, China, and Japan. [5]

Even though, there are so many important issues related with better understandings on the supersonic combustion phenomena in the scramjet engine, we limit our topics only on the effect of grid refinement, scheme unsteadiness, and phenomenological differences on the solution quality of numerical simulation onto supersonic combustion. For these purposes, we selected the case of numerical simulation for the HyShot scramjet combustor with a poor grid of coarse spacing[6], second case for a virtual scramjet engine composed by a straight/divergent flow path with/without cavity area of a good grid system of fine spacing[7], and finally third case of a ram accelerator for the superdetonative mode flight speed, which was filled by Hydrogen-air/diluted by Nitrogen/ Carbon dioxide combustible premixed gas.[8] It is believed that first case and second case would be controlled by turbulent mixing limited, while third case would be controlled by reaction limited.

2. COMPUTATIONAL FORMULATIONS AND ALGORITHMS

2.1. Conservation Equations

The combustor configuration is assumed to be two dimensional for computational efficiency. The conservation equations for a multi-component system are employed to analyze the chemically reacting flow in a scramjet combustor. The coupled form of the species conservation equations, fluid dynamics equations, and turbulent transport equations can be summarized in a conservative vector form as follows.

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

where the conservative variable vector, Q, convective flux vectors, F and G, diffusion flux vectors, Fv and Gv, and reaction source term W are defined. Details of the governing equations and thermal properties are described in the literature[8].

$$\mathbf{Q}=\begin{bmatrix}\rho_i\\ \rho u\\ \rho v\\ \rho e\\ \rho k\\ \rho w\end{bmatrix}\ \mathbf{F}=\begin{bmatrix}\rho_i u\\ \rho u^2+p\\ \rho uv\\ \rho Hu\\ \rho ku\\ \rho\omega u\end{bmatrix}\ \mathbf{G}=\begin{bmatrix}\rho_i v\\ \rho uv\\ \rho v^2+p\\ \rho Hv\\ \rho kv\\ \rho\omega v\end{bmatrix}$$

$$\mathbf{F}_v=\begin{bmatrix}-\rho_i u_i^d\\ \tau_{xx}\\ \tau_{xy}\\ \beta_x\\ \mu_k\,\partial k/\partial x\\ \mu_\omega\,\partial\omega/\partial x\end{bmatrix}\ \mathbf{G}_v=\begin{bmatrix}-\rho_i u_i^d\\ \tau_{xy}\\ \tau_{yy}\\ \beta_y\\ \mu_k\,\partial k/\partial y\\ \mu_\omega\,\partial\omega/\partial y\end{bmatrix}\ \mathbf{W}=\begin{bmatrix}w_i\\ 0\\ 0\\ 0\\ s_k\\ s_\omega\end{bmatrix}$$

2.2. Combustion Mechanism and Turbulence Closure

The present analysis employs the GRI-Mech 3.0 chemical kinetics mechanism for hydrogen-air combustion [9]. The mechanism consists of eight reactive species (H, H_2, O, O_2, H_2O, OH, H_2O_2 and HO_2) and twenty-five reaction steps. Nitrogen is assumed as an inert gas because the oxidation process does not have significant effect on the fluid dynamics in a combustor. Turbulence closure is achieved by means of Mentor's SST (Shear Stress Transport) model which is based on the k-ω two-equation formulation [10]. 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[11]. Baridna et al. reported that the SST model shows good prediction for mixing layer and jet flow problems, and that it is also less sensitive to initial values [12]. In the case of ram accelerator calculation simple Baldwin-Romax model was employed.

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 much useful advances were achieved, the improvement was significant in comparison with the results from laminar chemistry and existing experimental data, as evidenced in the results by Möbus et al [13]. By examining the existing results, such as Norris and Edwards [14], it is thought that the solution accuracy seems to be more dependent on grid resolution than the modeling of turbulence-chemistry interaction. 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.

2.3. Computational Algorithms

The governing equations were discretized numerically by 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 along with a differentiable limiter function. The spatial discretization strategy satisfies TVD conditions and shows high-resolution shock capturing capability. The discretized 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. Since detailed descriptions of the governing equations and numerical formulation are documented in the previous literature [15], it will not be recapitulated here. The numerical methods have been validated against a number of steady and unsteady simulations of shock-induced combustion phenomena that showed good agreement with existing experimental data [16-17].

3. HYSHOT SCRAMJET COMBUSTOR

3.1. Combustor Configuration

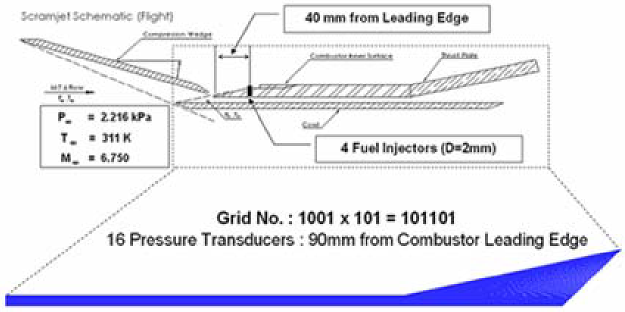

FIG 1. HyShot Geometry

HyShot scramjet combustion flow path is shown in Fig.1. The channel type combustor of 9.8mm height, 75mm width, and 300mm length is composed of transverse fuel injection and constant area straight duct[18]. The incoming airflow to the combustor inlet is set to Mach number 2.74 at 1256K and 83.48 kPa. Grid spreading over this combustor section is 1001 x 101.

3.2. Results

One result of numerical simulation is shown in Fig. 2. However, when it is compared with experimental case, as shown in Fig. 2, numerical result is thought as the sequence of less mixing than actual experimental case. We can notice that pressure distribution predicted by numerical simulation along the combustor for the case of

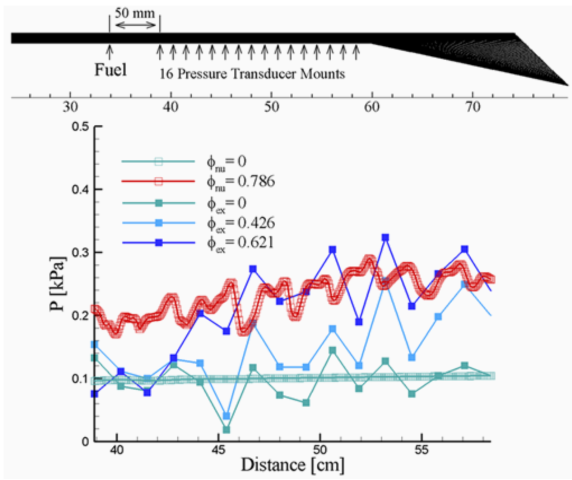

FIG 2. Pressure Distribution

equivalence ratio Ø=0.786 is about same level as those of experimental case Ø=0.621.

4. CONFIGURATION OF SCRAMJET COMBUSTOR

4.1. Combustor Configuration

The supersonic combustor considered in this study is shown in Fig. 3. The channel type combustor of 10 cm height and 131 cm length is composed of transverse fuel injection and a cavity. This combustor configuration is quite similar to the HyShot test model, except for the cavity, in which a swallowing slot is employed to remove the boundary layer from the inlet and the combustor starts with a sharp nose [18]. A cavity of 20 cm length and 5cm depth, having an aspect ratio, L/D of 4.0, is employed at 20 cm downstream of the injector.

4.2. Operating Conditions

The incoming air flow to the combustor is set to Mach number 3 at 600 K and 1.0 MPa. This combustor inlet condition roughly corresponds to a flight Mach number 5-6 at an altitude of 20 km, although the exact condition depends on the inlet configuration. Gaseous hydrogen is injected vertically through a slot of 0.1 cm width to the combustor through a choked nozzle. The fuel temperature is set to 151 K. The injector exit pressures are 0.5, 1.0 and 1.5 Mpa, and the overall equivalence ratios are 0.167, 0.33 and 0.5.

4.3. Combustor Conditions

A total of 936×160 grids are used for the main combustor flow passage, and 159×161 grids for the cavity. The grids are clustered around the injector and the solid surfaces and the injector. 54 grid points are included in the injector slot and the minimum grid size near the wall is 70 μm. All the solid surfaces are assumed to be no-slip and adiabatic, except for the upper boundary. For convenience and reduction of the number of grid points required to resolve the boundary layer, the upper boundary is assumed to be a slip wall, which is equivalent to the flow symmetric condition in the present configuration. Extrapolation is used for the exit boundary. Time step is set to 6 ns according to the minimum grid size and the CFL number of 2.0. Four sub-iterations are used at each time step..

4.4. Results

Numerical simulations were carried out for twelve cases, including non-reacting and reacting flows, with/without cavity for three different injection pressures of 0.5 1.0 and 1.5 MPa. The following sections will discuss the results for several cases. All the cases were run for 6 ms starting from the initial condition, which is longer than the typical test time of the ground based experiments.

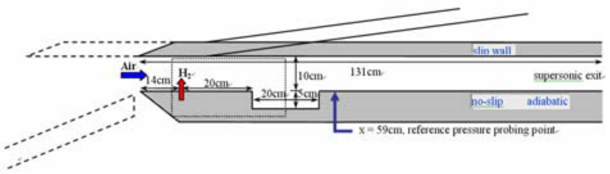

FIG 3. Scramjet combustor configuration

4.4.1. Reacting Flows Without Cavity

Here, to elucidate the height of variable geometry combustor, case of straight flow path and 4°divergent channel flow path is compared for the case of with/without cavity and injector exit pressure of 0.5, 0.75, 1.0 and 1.5 MPa. Subsequently, the instantaneous temperature fields for reacting flows without a cavity were shown. For the injection pressure ratio of 5.0, combustion occurs in the frontal separation region, but is not fully established along the shear layer. This separation region contains a pool of radicals and acts as a preheating zone. The flame is not anchored there, but in the region containing shock-wave/shear-layer interaction where the instability is generated. Downstream of this location, heat release from chemical reactions takes place, accompanied with large vortices convecting downstream. The overall phenomena seem quite similar to a typical turbulent diffusion flame generating large vorticities.

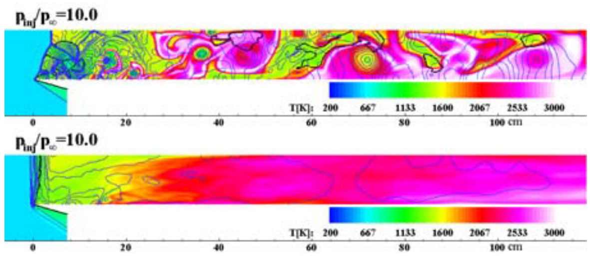

FIG 4. Constant area combustor without cavity

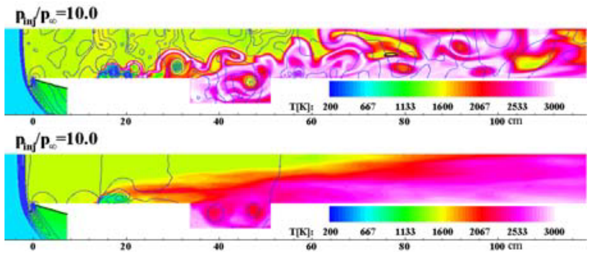

FIG 5. Constant area combustor with cavity

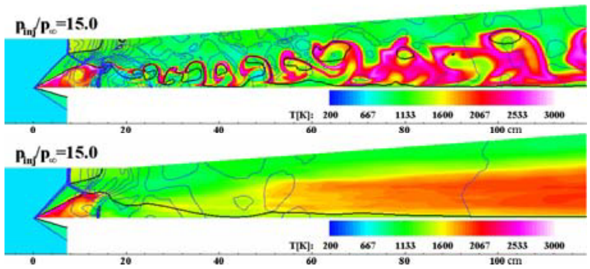

FIG 6. Divergent combustor without cavity

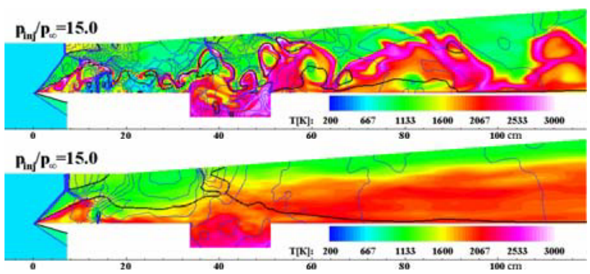

FIG 7. Divergent combustor with cavity

It is thought from this result that chemical reactions do not intensify the disturbance to an extent sufficient for triggering the instability of the injector flow.

For the injection pressure ratios 10.0 and 15.0, the temperature fields show different characteristics. Due to the large heat release, the pressure behind the injector builds up and leads to a Mach reflection across the combustor. A large subsonic region is formed downstream of the injector, and the injector flow no longer shows a structure composed of a leading oblique shock wave, a frontal separation region, etc. Instead, the fuel is injected though a narrow zone of subsonic jet, but can penetrate much deeper. The frontal separated flow region, which has a role as a radical pool and a pre-heater, still exists for the injection pressure ratio of 10.0, but disappears for the injection pressure ratio of 15.0 due to the fully subsonic environment near the injector. The frontal separated flow region and the oblique shock wave extends to the leading edge of the combustor and is stabilized there by the fixed inlet boundary condition.

Fig 4 and 6 show the temporal evolution the pressure for the reacting flows without a cavity. For the injection pressure ratio of 5.0, the final pressure is slightly higher than that of the non-reacting case and the pressure builds up very slowly. In contrast to this case, the results for the pressure ratios of 10.0 and 15.0 indicate that the injector instability is triggered around 1 ms and the pressure build-up is established around 3 ms for the pressure ratio of 10.0 and 1.5 ms for the ratio of 15.0. The pressure reaches approximately 0.5MPa for the pressure ratio of 10.0 and 0.6 MPa for the pressure ratio of 15.0. The flow fields for both cases are nearly stabilized around this condition.

4.4.2. Reacting Flows With Cavity

Lastly, in Fig 5 and 7, the instantaneous temperature fields for the reacting flows with a cavity. For the injection pressure ratio of 5.0, combustion is fully established over the cavity, and the cavity acts as a flame holder or a radical pool. The pressure builds up to around 0.3MPa, which is much greater than the case without a cavity. For the injection pressure ratios of 10.0 and 15.0, the pressure builds up rapidly and the combustor seems to be choked. The Mach reflection continuously develops and is finally disgorged out of the inlet.

5. CONFIGURATION OF RAM ACCELERATOR

Ram Accelerator is a novel concept using a principle of ramjet propulsion as a propelling mechanism of a projectile similar to the LEO satellite in a barrel, so a combustible mixture gas is compressed by a series of shocks and them generates thrust force, consequently in a similar way as Oblique Detonation Wave Engine(ODWE) or scramjet engine.

One of an example is listed here for a projectile of 28mm diameter, and 220mm length flying inside the barrel of 40mm diameter Flow condition is such as initial combustible fill pressure of 25atm H2/O2 diluent gas mixture at 300K, projectile highest Mach number 6-7. Mainly 400 x 60 grid is used, while grid refinement test has been done for the case of 800 x 120 and 1600 x 240 grids. Quasi-Steady wall pressure distribution for the case of $2H_2+O_2+7N_2$ mixture is tested, and shows almost same results. In Fig. 8, flow development around flying projectile is shown along the height distance until 100cm elapsed. We can notice shock waves, boundary layer, separation bubble, shock-induced combustion detonation waves, and global flow field considerably realistically. It may be said that less fine grid system still works not losing any important flow physics hence numerical simulation for premixture supersonic combustion is less sensitive to the grid spacing. It is thought premixture combustion is basically reaction controlled phenomena but not mixing controlled phenomena as previously shown for the case of scramjet combustion.

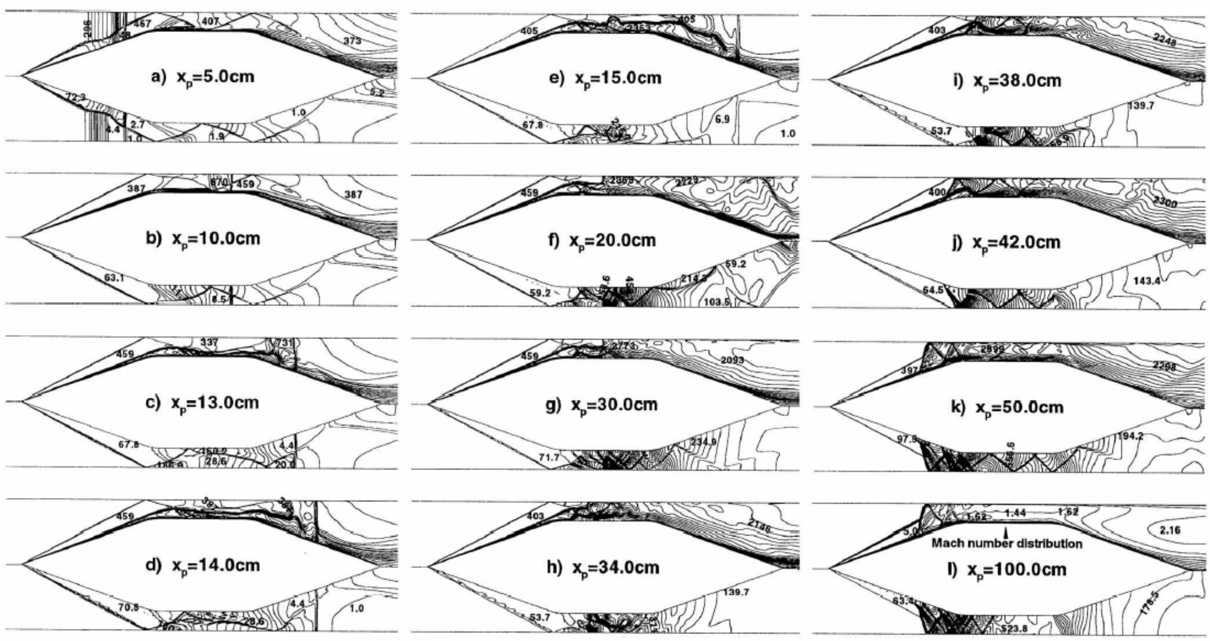

FIG 8. Temperature Distributions(upper half of each figures) and Pressure Distributions (lower half of each figures) in Marching Sequence for 2H2+O2+5N2 Mixture. Mach numbers are plotted for the case after xp=100.0cm

6. SUMMARY

The reacting flow dynamics in a scramjet combustor was carefully studied by means of a comprehensive numerical analysis. The present results show a wide range of phenomena resulting from the interactions among the injector flows, shock waves, shear layers, and oscillating cavity flows. As a conclusion of the present study, new findings can be summarized as follows.

1) Strong unsteady flow characteristics were identified for a scramjet combustor. The work appears to be the first of its kind in the numerical study of combustion oscillations in a supersonic combustor.

2) Transverse injected jet may remain stable without disturbance, but can be triggered to become unstable with disturbances from a shear layer or a cavity. Disturbed transverse injected jet has deeper penetration and improved fuel/air mixing than the stabilized one. A more careful study is necessary to characterize the stability of transverse injection jets.

3) The roles of the cavity as a source of disturbance for the transverse jet, fuel/air mixing enhancement, and flame holder were clarified.

4) Unstable flow characteristics for the reacting cases are similar to that of non-reacting flows except for the cases where pressure builds up rapidly.

5) Variable geometry can mitigate thermal choking, hence enhance combustor operation range.

6) Boundary layer development and flow separation are very importantly effecting to the onset of thermal choking.

7) Adequate numerical grid refinement is seriously demanded.

8) Premixed combustion simulation has less sensitive to the grid spacing.

9) The present study can be extended to a more realistic combustor configuration, but further investigations are necessary to achieve better understanding of detailed fluid and flame a scramjet combustor.

7. ACKNOWLEDGMENTS

This research was partly supported by ADD through FVRC-SNU and by National Research Laboratory program (M10500000072-05J000007210). The authors wish to acknowledge this financial support.

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