Numerical Investigation of Non-premixed Oblique Detonation Operations in Scramjet Combustor

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Abstract

The present study aims to develop strategies for hydrogen injection to operate scramjet combustor configuration in detonation mode at higher Mach number flights conditions. The reactive multi-species unsteady Navier-Stokes equations along with turbulence modelling are solved with detailed chemical kinetics for a two-dimensional computational domain of cavity based scramjet combustor. In order to establish detonation mode combustion a finite length wedge at angle is attached to the downstream of cavity in a scramjet combustor configuration. Initial simulations are performed at Mach 7 incoming air flow with freestream pressure of 40 kPa and temperature of 300 K for 2 ms time duration. The hydrogen fuel is injected at 30 mm upstream of cavity with angle of injection 15° using straight pipe of 2 mm width to provide mass flow rate equivalent to $\phi = 0.34$ with respect to incoming air mass flow rate. It is found that the presence of cavity between fuel injector and wedge stabilizes the detonation mode combustion and suppress the intermittent transition between deflagration and detonation modes in comparison to without cavity case. Further the flow conditions at the starting of combustor based on hypersonic intake, operating at an altitude of 25 km with flight Mach number 9 are simulated for cavity based combustor with wegde. The outcome suggest that high temperature of incoming flow can have adverse effect to develop detonation mode combustion, but with cavity and distance between wedge and cavity stable detonation front can be established.

Keyword: *Numerical Simulations, Shock induced Ignition, Detonation, Non-premixed Combustion, Scramjet*

1. Introduction

Detonation mode combustion differs than deflagration mode combustion as the former has flame coupled with the shock wave (called detonation wave) and it travels at supersonic speed with reaction products behind shockwave. The pressure and temperature can have higher jump across the detonation wave in comparison to the subsonic deflagration flame. There are mainly three propulsion concepts [1] based on detonation combustion: pulse-detonation engines (PDE), rotary detonation engine (RDE) and oblique detonation wave engine (ODWE). Among these oblique detonation wave engine (or sometimes called shock induced combustion ramjet, Shcramjet) has simplest design and possibility to integrate with scramjet engine configuration. The advantages of detonation-based propulsion systems are: higher rate of energy release, higher thermodynamic efficiency as well as shorter combustion chamber in comparison to conventional deflagration mode combustion systems.

At higher flight Mach numbers, the scramjet engines suffers inefficient heat release due to higher total enthalpy of incoming flow is comparable to the heat of combustion, resulting in insufficient fuel heat release and the inability to generate effective net thrust [2]. The scramjet configuration operates in deflagration mode combustion. Hence, the operation of scramjet engine configuration in detonation mode operation can providing higher heat release, leading to extend its operational flight range. A typical air-breathing hypersonic inlet of scramjet engines consist of multiple compression design, enabling multiple oblique shocks generated by external compression (ramp) and internal compression by cowl lip. Figure 1(a) shows three compression inlet (two almost equal strength external compression) and one internal compression to provide higher pressure and temperature at the combustor inlet. Among various strategies of fuel injection to provide better mixing

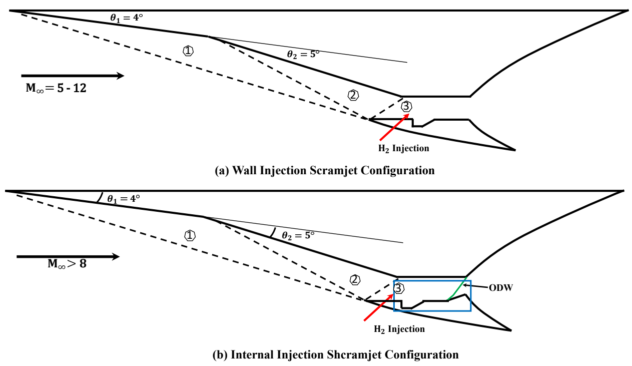

Fig. 1: Non-premixed (a) Scramjet Configuration, (b) Oblique Detonation Wave Engine Configuration

and flame stabilization, cavity based scramjet combustors are studied extensively [3]. In cavity based scramjet, the fuel is injected upstream of cavity as well ignition mechanism is provided to place stable flame. The oblique detonation mode combustion in high-speed flows has been studies extensively for premixed fuel-air mixture[4, 5], with strategies to inject fuel at the upstream locations, at the starting of external compression ramp so the homogenous well mixed fuel-air mixture can be obtained before entering the combustor [6]. A finite length ramp can be added in the combustor to provide shock induced ignition and formation of detonation wave. However, there has been recent interest in non-premixed mode oblique detonation wave engines to avoid pre-ignition of premixed combustible mixture before combustor entrance as well as provide shorter combustor length and flexible operations.

In recent numerical study [7] unsteady oblique detonation wave for sonic fuel injection before the wedge tip has been obtained by fuel injection at angles less than or closer to wedge angle, while higher angles lead to deflagaration mode combustion at incoming air with Mach number of 7 and freestream pressure of 40 kPa and freestream temperature of 300K. Further the intermittent transitions between deflagration and detonation mode combustions are observed even for lower fuel injection angles. The current study proposes placing a cavity to stabilize the detonation mode combustion between fuel injection location and the wegde (as shown in Fig. 1(b)) as well as performing simulations for real flight conditions, corresponding to Mach 9 flight at altitude of 25 km to understand the effects of real flight operations.

2. Numerical Method

In the current investigation, the numerical simulations are performed by solving unsteady Reynolds-Averaged Navier-Stokes equations with reactive multi-species equations along with turbulence model in two-dimensional computational domain. The modelling of detonation phenomenon for non-premixed fuel injection in high speed flow requires resolving multi-scales in space and time to efficiently model fuel-air mixing as well as gas dynamics and chemistry effects. The commercial CFD-package CONVERGE-CFD has been used through out the study, due to its efficient adaptive grid refinement, incorporating turbulence modelling as well efficient chemistry solver SAGE which utilize adaptive zoning [8]. Gas properties are calculated using an ideal gas equation. The second order spatial discretization with step flux limiter and fully implicit first-order accurate time integration method were used with PISO solver scheme. Variable time-stepping method with minimum time-step of 0.1 ns was employed, which was automatically calculated based on maximum CFL assigned. The numerical method and automated grid refinement strategies are adapted based on previous study of Vashishtha et al. [7]. The solution was initialized in first 0.4 ms with air only in the domain and hydrogen was injected at 0.4 ms with total simulation time of 2 ms. The chemistry is modelled using Westbrook Mechanism [9] for H2-Air combustion (with 12 species and 46 reactions), assuming Nitrogen as inert gas.

2.1. Computational Domain

The two-dimensional computational domain consists of sonic fuel injector at an angle 15° upstream of a cavity of dimension similar (not exactly) to HiFire 2 scramjet combustor [10] as shown in Fig. 2(a). The bottom flat wall of cavity has width of 65 mm and the depth below the injector plane of 18 mm and angle of cavity ramp is 30°. The cavity is starting at 30 mm downstream of fuel injection location. The wedge

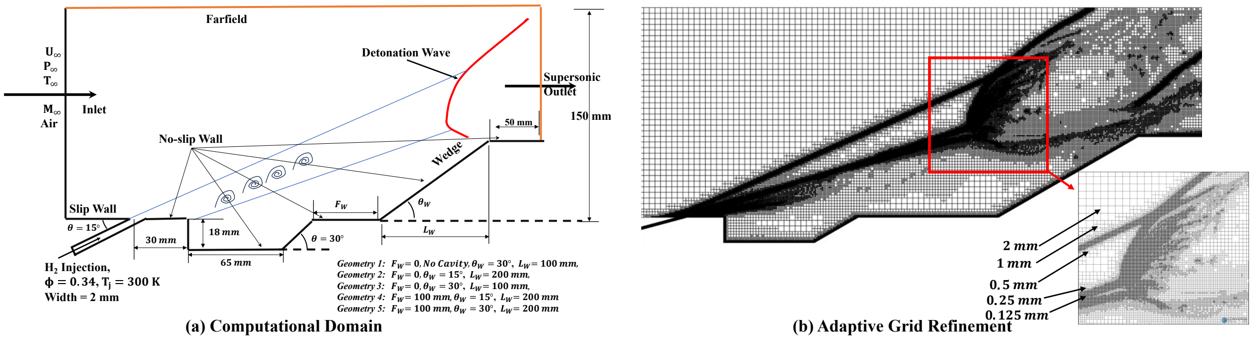

Fig. 2: (a) Computational Domain, (b) Adaptive Grid Refinement for Geometry 4 at time 0.2 ms

of length L_w at angle θ_w to establish oblique detonation wave is located at the downstream of cavity with distance F_w from end of cavity ramp. The five geometric variations are investigated in this study as 1) reference case (Geometry 1): without cavity and wedge located 30 mm downstream of injector, 2) Geometry 2: wedge starting at cavity end ($F_w = 0$) with length $L_w = 200$ mm and angle $\theta_w = 15°$, 3) Geometry 3: wedge starting at cavity end ($F_w = 0$) with length $L_w = 100$ mm and angle $\theta_w = 30°$, 4) Geometry 4: wedge starting at 100 mm downstream of cavity end ($F_w = 100$ mm) with length $L_w = 200$ mm and angle $\theta_w = 15°$, and 5) Geometry 5: wedge starting at 100 mm downstream of cavity end ($F_w = 100$ mm) with length $L_w = 100$ mm and angle $\theta_w = 30°$. During each numerical simulations, the base grid of 4 mm is used to initialize the simulation before fuel injection up to 0.35 ms. At time 0.35 ms the base grid is refined to 2 mm. Further 4 levels of adaptive grid refinements are performed based on velocity, temperature, mass fractions of H_2, O and OH. Figure 2(b) shows the various levels of grid refinements for Geometry 4 at the end of simulation.

2.2. Boundary and Initial Conditions

Tab. 1: Simulation Conditions ($\phi = 0.34$, Inlet height = 150 mm)

Property	M_∞ **= 7.0 incoming air**		**Mach 9 Flight at altitude 25 km**	
	Inlet Air	**Inlet Fuel**	**Inlet Air**	**Inlet Fuel**
Mach Number	$M_\infty = 7.0$	$M_j = 1.0$	$M_\infty = 5.26$	$M_j = 1.0$
Pressure (kPa)	$P_\infty = 40$	$P_j = 793.0$	$P_\infty = 52.84$	$P_j = 565.6$
Temperature (K)	$T_\infty = 300$	$T_j = 300$	$T_\infty = 582.05$	$T_j = 300$
Velocity (m/s)	$V_\infty = 2435.4$	$V_j = 1316.4$	$V_\infty = 2551.3$	$V_j = 1316.4$
Total Flow Enthalpy (MJ/kg)	$H_{0,air} = 3.3$	-	$H_{0,air} = 3.8$	-
Mass Flow Rate (kg/s)	$\dot{m}_{air} = 169$	$\dot{m}_{H_2} = 1.69$	$\dot{m}_{air} = 120$	$\dot{m}_{H_2} = 1.20$
H_2 Mass Fraction	0	1	0	1
O_2 Mass Fraction	0.233	0	0.233	0
N_2 Mass Fraction	0.767	0	0.767	0
Momentum Flux Ratio (J)	-	0.4	-	0.39

All five geometries, mentioned in previous section are simulated for two conditions: 1) $M_\infty = 7.0$ air flow condition with freestream pressure 40 kPa and freestream temperature 300 K, accordingly hydrogen injected from fuel injector with equivalence ratio $\phi = 0.34$, 2) $M_{flight} = 9.0$ according to altitude of 25 km with hypersonic inlet consists of two external compression $\theta_{compression1} = 4°$ and $\theta_{compression2} = 5°$ and internal compression $\theta_{compression3} = 9°$ as shown in Fig. 1(b). The relevant simulation conditions are highlighted in Table 1. The wall before the fuel injection is considered as slip wall with zero gradient thermal boundary, while other bottom walls including cavity and wedge are modelled as no-slip with adiabatic thermal boundary. The outlet and freestream conditions are modelled as zero gradient. The fuel inlet boundary is modelled with total pressure and total temperature, which is activated at 0.4 ms after the flow-field with incoming air is established in the domain.

3. Results and Discussions

The operations of non-premixed shock induced combustion ramjet (Shcramjet) or oblique detonation engine is governed by efficient mixing of fuel-air as well as ignition provided by compressed region by the wedge. The initial study [7] highlights that direct fuel injection in front of wedge, placed in high-speed flow at certain angle leads to formation of detonation wave with intermittent transitions to deflagration mode combustion by interaction of hot combustion products into jet shear layer and producing ignition near the upper jet boundary. In this study, a cavity is placed to control the intermittent deflagration transition and analyzed by time dependent heat release rate as well as temperature contours. The flow condition of Mach 7 incoming air flow with freestream temperature 300 K and pressure 40 kPa are analyzed as well as real flight conditions according to Mach 9 flight at altitude of 25 km, which leads to higher incoming air temperature, are investigated for five different geometries with fuel injection according to $\phi = 0.34$.

3.1. Mach 7 incoming air conditions

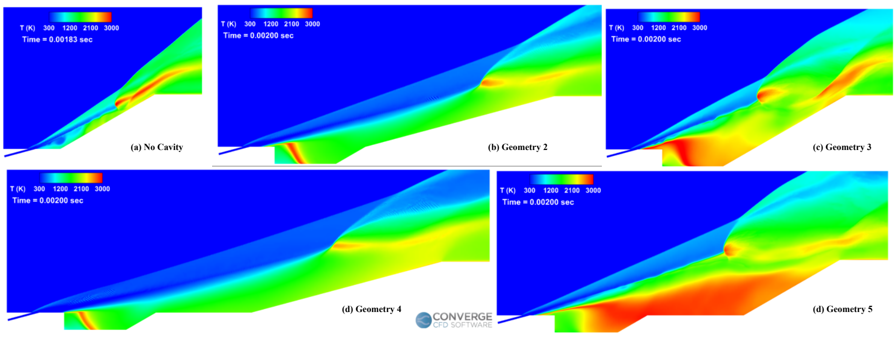

Fig. 3: Temperature Contours for Mach 7 incoming flow with various geometries

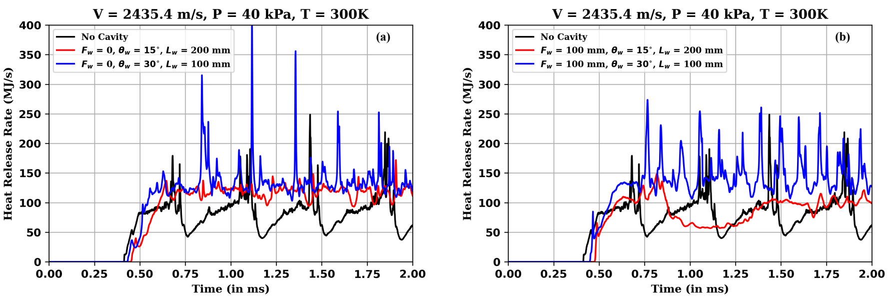

Fig. 4: Time-dependent Heat Release Rate for Mach 7 incoming flow

The results from hydrogen injection at equivalence ratio 0.34 for Mach 7 incoming air flow (air velocity of 2435.4 m/s) are presented in Fig. 3 as temperature contours with detonation wave formation and Fig. 4 as time history of heat release rate. In without cavity case, the intermittent transition between detonation and deflagration waves are observed, as seen in sudden drop of heat release at various instances in Fig. 4. The lifted detonation wave forms near the end of wedge in Fig. 3(a). For geometry 2, with no gap between cavity end and wedge at angle 15°, a stable detonation wave forms at downstream location in smooth manner as seen in Fig. 3(b). For geometry 3, with no gap between cavity end and wedge at angle 30°, a strong detonation wave is formed (Fig. 3(c)) in the middle of wedge with many incidents of ignitions without transitioning into deflagration mode (Fig. 4). By providing the flat region between cavity end and wedge starting, the formation of detonation wave becomes smoother for wedge angle 15° (Fig. 3(d)), which is observed as lower heat release rate in Fig. 4 right. The flat region between cavity and wedge for wedge angle 30° leads to smoother detonation wave formation (Geometry 5: Fig. 3(e)) with instances of multiple detonation in heat release curve (Fig. 4 right).

3.2. Mach 9 flight Conditions for scramjet inlet

Mach 9 flight condition at altitude of 25 km with three compression hypersonic inlet (as shown in Fig. 1(b) corresponds to incoming air velocity of 2551.3 m/s with freestream temperature of 582 K and freestream pressure of 52 kPa. The increased freestream air temperature (in comparison to Mach 7 air flow) leads to early ignition of fuel air mixture (at same $\phi = 0.34$), overall affecting detonation wave formation adversely.

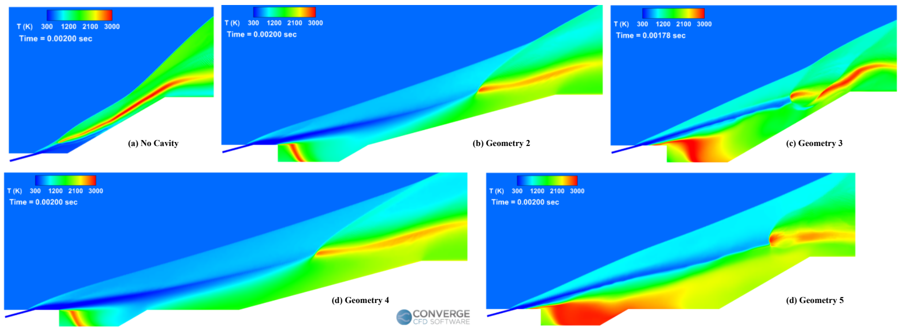

Fig. 5: Temperature Contours for for Mach 9 flight conditions at altitude 25 km

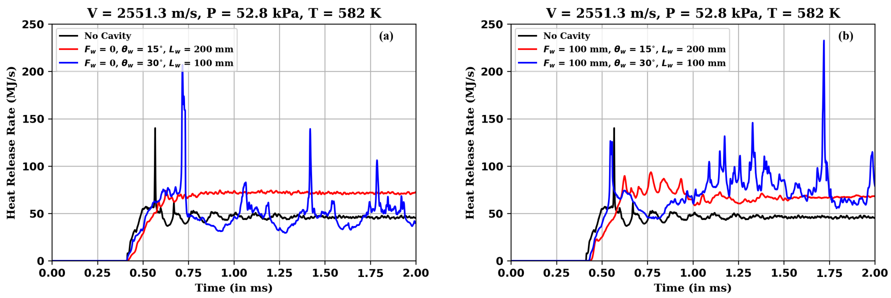

Fig. 6: Time-dependent Heat Release Rate for Mach 9 flight conditions at altitude 25 km

For Mach 9 flight condition, with no cavity case, direct fuel injection in front of wegde at angle 30° leads to no detonation wave formation, only deflagration wave along the wegde (in Fig. 5(a)) and no sudden heat release rate in Fig. 6. For geometry 2 with wedge starting near the end of cavity at angle 15° an inclined smooth stable detonation front (Fig. 5(b)) is formed with higher constant heat release rate (Fig. 6 left). For geometry 3 with wedge starting near the end of cavity at angle 30°, an unstable detonation wave forms (in Fig. 5(c)) with intermittent transitions to deflagration modes as seen in heat release curve in Fig. 6 (left). The geometry 4 with gap between cavity end and wedge tip, with wedge angle 15°, a smoother detonation wave forms with lower heat release rate (in Fig. 6 (right)) than compare to geometry 2. The geometry 5, with gap between cavity end and wedge tip and wedge angle 30° a stronger detonation wave forms (in Fig. 5(e)) with higher heat release rate (in Fig. 6 (right)) with instances of multiple ignitions in comparison to geometry 3.

3.3. Thermal and Combustion Efficiencies

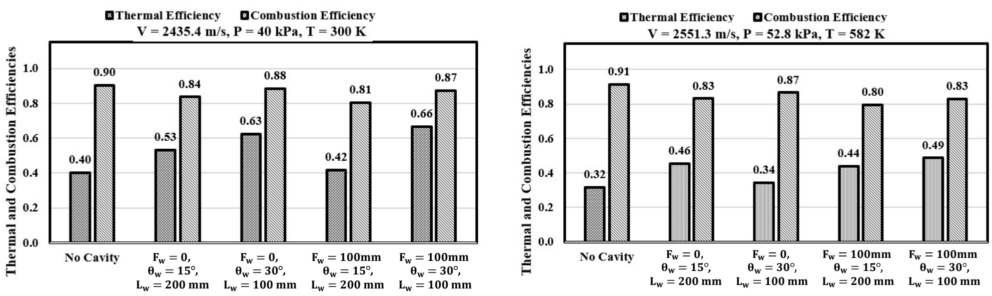

Fig. 7: Thermal and Combustion Efficiencies: a) Mach 7 incoming air flow (left), b) Mach 9 flight at altitude 25 km (right)

To understand the overall effect of non-premixed traverse injection for cavity based scramjet combustor operation for detonation mode, the overall thermal and combustion efficiencies are computed based on previous study[7] and compared in Fig. 7. The higher thermal efficiencies reflects the detonation mode operations, while lower thermal efficiencies reflects intermittent or more duration of deflagration mode combustion. The higher combustion efficiencies reflects the higher consumption of injected hydrogen, without considering the consumption due to deflagration or detonation. For Mach 7 incoming air flow with lower freestream air temperature, the higher thermal efficiencies are observed with higher wegde angles of 30° for $F_w = 0$ and $F_w = 100$ mm. The gap between cavity end and wedge provides better mixing and slightly better thermal efficiencies. The real flight conditions of Mach 9 at altitude 25 km leads to higher freestream temperature, which has adverse effects on thermal efficiencies. The higher thermal efficiencies are observed for stable detonation formation for geometries 4 and 5, with gap between cavity end and wedge tip as well as for geometry 2 without gap with wedge angle 15°.

4. Conclusions

This study concludes that the presence of cavity between the traverse fuel injection and inclined wedge can lead to stable smooth formation of detonation wave formation at hypersonic Mach 7 incoming air flow with freestram temperature of 300 K and pressure of 40 kPa. The operation of cavity based scramjet combustor with addition of wedge in detonation mode can possible provide pathway to extend flight range with higher thermal efficiency. The flow-field at combustor inlet corresponding to real flight condition of Mach 9 at altitude of 25 km, has higher freestream temperature. The operation of cavity based scramjet combustor with wedge in detonation mode is adversely affected by higher incoming air temperature because of early ignition in jet layers, however it can be mitigated by providing relaxation zone by lowering wedge angle or provinding gap between cavity and wedge.

5. Acknowledgments

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