21st Australasian Fluid Mechanics Conference
Adelaide, Australia
10-13 December 2018

Simulation of a Rotating Detonation Ramjet Model in Mach 4 Flow

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Abstract

Rotating detonation waves within a ramjet combustor will cause periodic pulses of high pressure which could degrade or unstart the ramjet inlet. To explore the upstream influence of rotating detonation waves, simulations of a hydrogen-fuelled rotating detonation ramjet engine model in a Mach 4 flow have been performed using the CFD code, 'Eilmer4'. The full three dimensional flow will be split into two, essentially two-dimensional flows: (1) the axisymmetric inlet compression process; and (2) a rotating detonation wave process in the annular combustor. The study proceeds via three main steps. (1) A ramjet inlet is designed for Mach 4 flight and the inlet is simulated using CFD; (2) rotating detonation simulations for an 'unwrapped' annular combustor are performed by imposing the out-flow conditions obtained in Step 1 as the in-flow boundary conditions in this step; and (3) the periodic pressure pulses from the rotating detonation simulations in Step 2 are imposed at the out-flow boundary of the inlet defined in Step 1. The inlet simulations performed in Step 3 show that this inlet is resilient to the downstream pressure disturbances generated by the rotating detonation wave at the chosen operating conditions. However, isolation of the inlet considered herein is achieved with impractically-high total pressure losses. Work is currently being performed to improve the design of the inlet and isolator and re-assess the influence of the rotating detonation wave.

Introduction

Detonation is a form of shock-induced premixed combustion in which the shock wave propagation is sustained by the energy released in the combustion process. Analyses show that detonation-mode engines can have a theoretical thermodynamic cycle efficiency more than 20% higher than conventional, deflagration-mode combustion engines [1]. Early studies on detonation ramjets focused on oblique detonation wave (ODW) arrangements where the detonation is induced by a stabilized shock anchored within the ramjet flow path. For such configurations, the velocity of the reactant mixture relative to the combustion chamber has to be greater than the local Chapman-Jouguet (C-J) velocity. The stabilisation of oblique detonation waves through a range of transient conditions, has been demonstrated experimentally [2].

For cases where the velocity of the reactant mixture does not exceed the local C-J velocity, two different detonation engine configurations – pulse detonation engines (PDEs) and rotating detonation engines (RDEs) – have attracted attention in recent years due to the extremely rapid, isochoric combustion occurring within these devices which results in an improved theoretical thermal efficiency relative to other engines [3].

PDEs achieve thrust through an unsteady cycle involving the repetitive filling, ignition, and discharging of a combustor tube. Integration of the unsteady PDE cycle with a steady inlet and nozzle for an air-breathing engine is challenging. The RDE cycle differs substantially from the PDE cycle in that it consists of a detonation wave continuously rotating around an annular combustor [4]. The RDE cycle avoids the transient combustor filling and discharge issues of the PDE cycle since there is a continuous injection of fresh reactants into the combustor in the case of RDEs. Although there will still be cyclic pulsations associated with the rotation of the detonation waves, once the combustion is initiated, the detonation front will be sustained, avoiding the cyclic initiation and deflagration-to-detonation transition process of PDEs. The annular shape of the RDE combustor also offers ease of integration with axisymmetric engine designs. Simplified engine structures might also be possible through the development of RDEs, resulting in further enhancement of the thrust-to-weight ratio and making it a promising engine technology for future aerospace propulsion applications.

Most of the numerical and experimental RDE research has focused on rotating detonation in rocket-like combustors in which fuel injection is controlled by arrays of sonic micro-nozzles within a pressurized plenum chamber [5]. Three key geometric parameters for such RDE combustors have been identified by [6], and as depicted in Figure 1 within the context of a ramjet combustor: Δ which is the minimum distance between the walls of the annular channel, d which is the minimum inner diameter of the annular channel, and L is the minimum length of the chamber. Parametric studies based on the mathematical modelling of rotating detonation waves coupled with an inlet and isolator have been reported [7, 8]. Although the potential advantages of applying rotating detonation in ramjet engines has been acknowledged in previous publications [9, 10], an assessment of the upstream influence of the rotating detonation waves on the supersonic inlet – the topic of the current study – is yet to be reported.

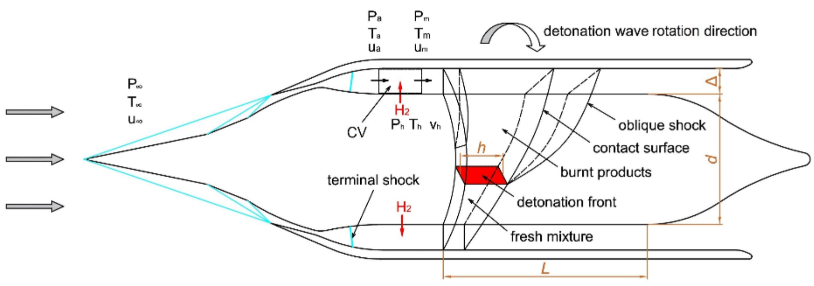


Figure 1. Schematic illustration of a rotating detonation ramjet model.

Rotating Detonation Ramjet Model

Mach 4 flight at an altitude of 15 km has been selected as the target operating condition for the rotating detonation wave ramjet engine. The free stream flow parameters corresponding to this operating condition are listed in Table 1. These conditions have been chosen in part because it should be possible to physically simulate this Mach 4 flight condition in the hypersonic wind tunnel at the University of Southern Queensland in the near future, thus enabling practical assessment of modelling and simulation efforts.

A ramjet model consisting of a supersonic inlet, a rotating detonation combustor and an aerospike nozzle is proposed in this paper as shown in Figure 1. Hydrogen is injected downstream of inlet to form a stoichiometric mixture which will be detonated in the annular combustor. The computational simulations reported herein were performed using Eilmer4 [11] on USQ's High Performance Computing facility.

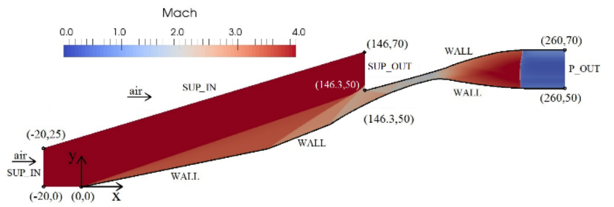
M	P_∞ (kPa)	T_∞ (K)	u_∞ (m/s)
4	11.9	216	1178

Table 1. Mach 4 free stream conditions at an altitude of 15 km.

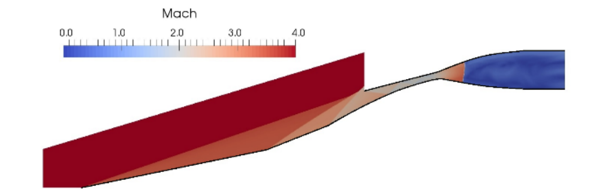
Step 1: Inlet Design

An axisymmetric, three-shock inlet with an overall contraction ratio of 7.76 was designed and simulated using the transient, compressible flow solver, Eilmer4. The key dimensions of the inlet and the computational domain with the specified boundary types are illustrated in Figure 2(a). For these simulations, Eilmer4 was operated in an inviscid mode with multi-block structured grids having a total of 84000 cells.

The Mach number contours within the inlet are presented in Figure 2 for different outlet boundary pressures of 180 kPa and 340 kPa when the simulated flows have stabilised. The numerical results show that the axisymmetric, three-shock inlet was able to tolerate a varied back pressure from 180 kPa up to 340 kPa with the terminal shock being positioned further upstream, closer to the throat of the inlet at the higher outlet pressure. For steady outlet pressures higher than 340 kPa, the inlet unstarts.



(a) Outlet pressure of 180 kPa



(b) Outlet pressure of 340 kPa

Figure 2. Computed Mach number contours for the inlet operating at a simulated time of 5 ms after initialisation of the computation for two different outlet boundary pressures: (a) 180 kPa; and (b) 340 kPa. The key dimensions (units: mm) of the computational domain are indicated in part (a).

Step 2: Rotating Detonation Combustion

To define flow conditions for the combustion calculation, transverse sonic injection of hydrogen with a total pressure and temperature of 500 kPa and 300 K, respectively, was assumed to occur at a location upstream of the entrance of the annular combustor, as shown in Figure 1. The mass flow rate of air (1.776 kg/s) was calculated based on the inlet capture area and the properties of Mach 4 free stream air in-flow. The mass flow rate of hydrogen (0.052 kg/s) was calculated from the mass flow rate of the air so as to achieve a stoichiometric mixture feeding the combustor. Constant area mixing was assumed within the control volume (CV, illustrated in Figure 1), and friction, heat transfer and other losses are neglected. The flow leaving the CV was assumed to be fully mixed and the flow conditions of the mixture were calculated using

$$\dot{m}_{air} + \dot{m}_{H_2} = \dot{m}_{mix} \tag{1}$$

$$P_{air} + \rho_{air} u_{air}^2 = P_{mix} + \rho_{mix} u_{mix}^2 \tag{2}$$

$$\dot{m}_{air}\left(h_{air} + \frac{u_{air}^2}{2}\right) + \dot{m}_{H_2}\left(h_{H_2} + \frac{u_{H_2}^2}{2}\right) = (\dot{m}_{air} + \dot{m}_{H_2})\left(h_{mix} + \frac{u_{mix}^2}{2}\right) \tag{3}$$

in combination with the ideal gas equation of state:

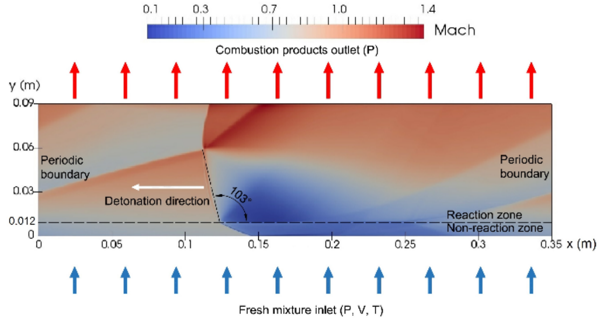
$$P_{mix} = \rho_{mix} R_{mix} T_{mix} \tag{4}$$

In the energy equation (3), the enthalpies are calculated using $h = c_p T$, where the c_p for mixture is obtained based on the mass fraction of air and hydrogen. The conditions for the air entering the CV were obtained from the computational results averaged across the flow exiting the inlet for an outlet boundary pressure of 200 kPa. The air flow, hydrogen flow, and the calculated conditions of the mixed flow, which provide the steady in-flow conditions for the rotating detonation simulations are listed in Table 2.

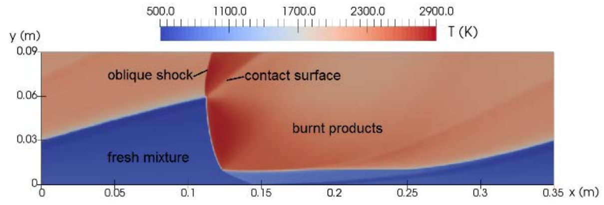
	Air	Hydrogen	Mixture
Pressure (kPa)	200	264	233
Temperature (K)	880	250	780
Velocity (m/s)	291	1201	358

Table 2. Flow conditions for the air-hydrogen mixing process.

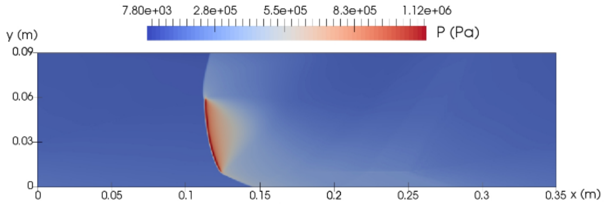
The modelling of the rotating detonation was performed based on a 2-dimensional 'unwrapped' annular combustor by treating the width of the annulus Δ (the parameter indicated Figure 1) as having unit thickness and ignoring curvature and radial effects. A computation domain consisting of a non-reaction zone and a reaction zone was discretized as a 2-dimensional uniform grid (1750 × 450 cells) with periodic boundary conditions, as shown in Figure 3(a), which also illustrates the computed contours of Mach number at an instant in time during a cycle. The pre-mixed hydrogen-air was fed continuously into the computational domain at steady conditions (as specified in Table 2). The non-reaction zone functioned as a buffer region for the steady in-flow conditions against the transient pressure rises induced by the moving detonation wave and having this buffer region improved the computational convergence. The combustion products exhausted from the combustor and a fixed outlet boundary pressure of 25 kPa was specified. The transient simulation of viscous compressible flow was carried out with a detailed H_2/O_2 oxidation chemistry mechanism including 8 species (H_2, O_2, H_2O, HO_2, OH, O, H, N_2) and 18 elementary reactions proposed by [12].



(a) Mach number contours.



(b) Temperature contours.



(c) Pressure contours.

Figure 3. Computational results within the 'unwrapped' rotating detonation combustor at an instant in time.

A single detonation wave was initiated within the domain by introducing a small zone with a temperature of 2000 K, a pressure of 1.0 MPa, and a velocity of -2000 m/s; this zone was a small rectangular region (2 mm × 10 mm) located in the South-East corner of the reaction zone, as viewed in Figure 3(a). Once established, the detonation wave travelled from right to left across the computational domain as illustrated, simulating to a circumferential flight of the wave within the annular combustor. The computed Mach number, temperature and pressure throughout the 'unwrapped' annulus at a particular instant in time are shown in Figure 3. The typical RDE structures observed in other studies have also been simulated in the present CFD work, as shown in Figure 3(b). The results illustrate an inclined detonation front with an angle of 77° relative to the wave propagation direction. The inclination of the detonation front arises because of the non-zero in-flow speed of the fresh mixture. The numerical simulations show that the moving detonation generates an oblique compression wave propagating upstream (downwards, into the non-reaction zone as shown in Figure 3(c)) processing the incoming fresh mixture, and generating a pulse of high pressure. The distribution of the main species in a direction perpendicular to the detonation front is illustrated in Figure 4, which is consistent with the description of the ZND detonation model [13].

Step 3: Periodic Pressure Effect on the Started Inlet

The pressure distribution along the boundary at the entrance of the combustor was extracted from the RDE simulation results in Step 2. The spatial distribution of pressure along this boundary at two different times is shown in Figure 5. The speed of the detonation wave across the computational domain is calculated to be 1703 m/s and, when the inclination of the detonation wave is considered, the magnitude of the velocity vector perpendicular to the wave front is found to be 1748 m/s, which is still less than C-J detonation wave speed of 1915 m/s estimated using well-established thermodynamic tools [14].

The spatial distribution of pressure around the perimeter of the combustor at two particular times, as determined in Step 2, is illustrated in Figure 5. To proceed with Step 3, the periodic variation of pressure at any particular location (which has a frequency of 4.865 kHz) as determined in Step 2 is now imposed as the outlet pressure boundary condition for the axisymmetric inlet and subsonic diffuser simulations. Clearly this is quite a simplification because at any instant in time, the pressure around the perimeter of the combustor inlet cannot be axisymmetric because of the presence of the rotating detonation wave. Nevertheless, axisymmetric simulations are performed as an initial attempt to simulate the effects of pressure pulsations in the combustion chamber. The simulations in this step were initialised using the results from the steady-flow started inlet obtained in Step 1.

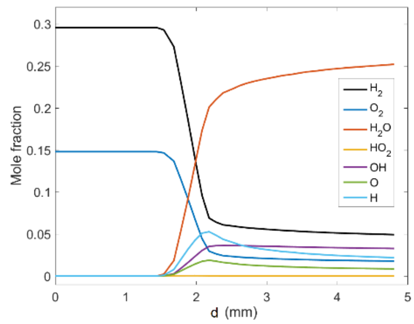


Figure 4. Mole fraction for species across the detonation wave at a location corresponding to x ≈ 0.115 m, y ≈ 0.03 m in Figure 3.

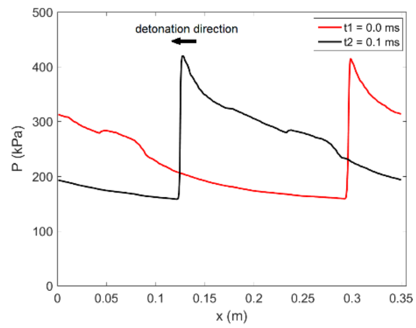
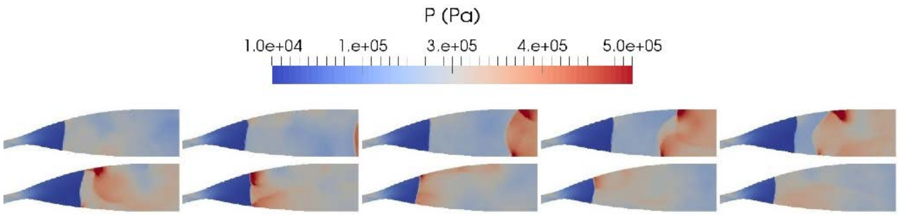
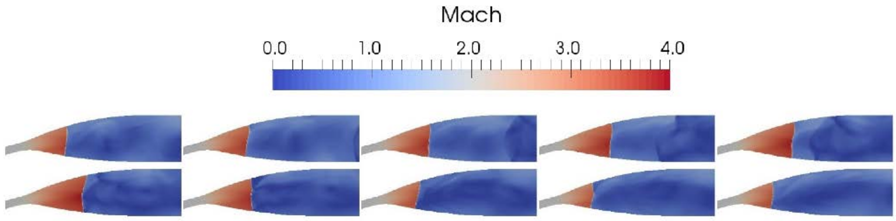


Figure 5. Simulated pressures at two different times (separated by the interval of 0.1 ms) at the entrance of the combustor.

Figure 6 presents the transient results as sequential colour maps of pressure and Mach number within the subsonic diffuser throughout one cycle of the rotating detonation wave. The upstream movement of the pressure wave arising from detonation, followed by a more gradual decay of pressure within the diffuser, can be observed in Figure 6(a). The terminal shock is able to adjust its position within the diffuser, and the pressures after the terminal shock consequently change with time. Although the peak pressure generated by the rotating detonation wave was around 420 kPa, which exceeded the maximum steady value of outlet pressure at which the ramjet inlet remained started of 340 kPa, the periodic pulses of high pressure induced by the detonation wave did not un-start the inlet.



(a) Pressure contours



(b) Mach number contours

Figure 6. Image sequences showing computational results for the pressures (part a) and Mach number distributions (part b) within the subsonic diffuser throughout one cycle of the rotating detonation wave. Sequential images are separated in time by 0.02 ms.

Conclusions

To explore the upstream influence of periodic pulses of high pressure induced by detonation waves, two-dimensional simulations of a hydrogen-fuelled rotating detonation ramjet engine model in a Mach 4 flow have been performed. The simulations show that the axisymmetric, three-shock inlet was able to tolerate steady outlet pressures up to 340 kPa with the existence of a terminal shock and desired subsonic out-flow at the selected Mach 4 conditions. A simulated rotating detonation wave with a circumferential speed of 1703 m/s generated periodic pressure rises that propagated upstream. When the started inlet was subjected to these pressure pulses imposed at the outlet boundary of the diffuser, it was observed that the inlet did not unstart, despite the transient pressure peak of 420 kPa exceeding the maximum steady flow outlet pressure for the started diffuser.

The ramjet inlet considered in the present work was isolated from the combustion chamber via a subsonic diffuser with a relatively strong shock wave stabilised in its diverging section. The pressure losses associated with the single-shock subsonic diffuser are too high for the arrangement to be considered practical. Future simulations will target a supersonic inlet and isolator configuration having a higher total pressure recovery, and will also include viscous effects and a more complete treatment of injection and mixing. Future simulations of the inlet should also consider the non-axisymmetric nature of the periodic pressure fluctuations in the combustion chamber when assessing the potential of inlet unstart.

Acknowledgments

Simulations using Eilmer4 were performed using the University of Southern Queensland (USQ) Fawkes HPC which is co-sponsored by QCIF (Queensland Cyber Infrastructure Foundation) - www.usq.edu.au/eresearch/hpcwere.

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