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Procedia Engineering 67 (2013) 218 – 229

Procedia Engineering

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7th Asian-Pacific Conference on Aerospace Technology and Science, 7th APCATS 2013

# Mixing Effectiveness Study in Scramjet Combustion

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**Abstract**

Scramjet propulsion provides promising opportunities for high-speed transportation and cost reduction for satellite and spacecraft launch operations. This paper provides some selected numerical examples among the abundance of available test cases for the purpose of demonstrating to what extends the current computational models can do in assisting the design and test programs of scramjet combustors. Numerical examples that cover a wide range of flight speeds are investigated with benchmark data validations. Ideas of mixing enhancement and flame holding mechanisms in scramjet combustion is analyzed and planned for the future experimental investigations.



*Keywords*: Scarmjet, Computational Fluid Dynamics, Finite-rate Chemistry, Fuel Injection, Mixing Enhancement.

**Nomenclature**

| | |
|---|---|
| $C_1, C_2, C_3, C_\mu$ | turbulence modeling constants, 1.15, 1.9, 0.25, and 0.09 |
| $C_p$ | heat capacity |
| $D$ | diffusivity |
| $H$ | total enthalpy |
| $K$ | thermal conductivity or stiffness |
| $k$ | turbulent kinetic energy |
| $Q$ | heat flux |
| $T$ | temperature |
| $t$ | time, sec |
| $u$ | mean velocities |
| $V^2$ | $\sum u^2$ |
| $x$ | Cartesian coordinates or non-dimensional distance |

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Selection and peer-review under responsibility of the National Chiao Tung University
doi:10.1016/j.proeng.2013.12.021

| | |
|---|---|
| *Greek symbols* | |
| $\alpha$ | species mass fraction |
| $\varepsilon$ | turbulent kinetic energy dissipation rate |
| $\mu$ | viscosity |
| $\mu_t$ | turbulent eddy viscosity ($=\rho C_\mu k^2/\varepsilon$) |
| $\Pi$ | turbulent kinetic energy production |
| $\rho$ | density |
| $\sigma$ | turbulence modeling constants, 0.9, 0.9, 0.89, and 1.15 for Eqs. (2), (4-6) |
| $\tau$ | shear stress |
| $\omega$ | chemical species production rate or natural frequency |
| *Subscripts* | |
| $r$ | radiation |
| $t$ | turbulent flow |

## 1. Introduction

It has been a long standing issue in boosting the payload mass fraction performance enhancement of human's access to space technical capabilities. During the Apollo era in the 1960s, the Saturn V was capable of putting 120 tones of payload mass to low Earth orbit. That is about 4.2 percent of the total takeoff mass of the rocket. Ironically, none of today's launch vehicles can match that performance. In searching for possible breakthrough in enhancing the payload mass fraction performance of launch vehicles, supersonic combustion ramjet (Scramjet) design study has been a long-term research effort in the aerospace community since the 1960's. Major programs were launched between 1984 and 1995, including NASA's National Space Plane (NASP, X-30) program, which were unfortunately terminated in 1994 due to budgetary problem after the Challenger incident. However, a rich amount of research experience and test data was obtained during that period from design modeling, analyses and ground test programs [1-5].

At the turn of the new millennium, new efforts in flight demonstrations have also revealed valuable research data and pushed the flight envelop closer to the ultimate goal of Mach 18~25 flight into space. HyShot flight test of the University of Queensland, Australia, using Terrier/Orion sounding rocket test bed, demonstrated for the first time a successful scramjet combustion experiment in flight on July 30, 2002. Shock tunnel test data and numerical studies are also investigated before and after flight experiments. Hyper-X (X-43) of NASA demonstrated a successful positive thrust scramjet flight to Mach 7, using B-52 air-launch Pegasus booster test bed, in late 2002. Hyper-X then increased its maximum flight speed to Mach 9.8 in the second test on Nov. 16, 2004. Later, X-51 also demonstrated its Mach 5 to 6 flights with hydrocarbon fuel. These new developments plus data collected in the past 45 years has laid a promising foundation for the aerospace research community that we are one step closer to the goal of reaching space and hypersonic air travel using scramjet combustion technology. In these research activities, computational fluid dynamics (CFD) methods have played an important role in resolving combustor design problems.

In this paper, numerical study of scramjet combustion based on radical farming concept is performed for several combustor designs. A Navier-Stokes flow solver with finite-rate chemistry model is employed for supersonic combustion flowfield analysis. As part of the key design issues, fuel mixing and flame holding mechanisms are critically important in supersonic combustion. In the present study, design concepts of mixing enhancement with flame holding characteristics are proposed.

## 2. Numerical Approach

The present CFD methodology is based on a multi-dimensional, finite-volume, viscous, chemically reacting, unstructured grid and pressure-based formulation. Time-varying transport equations of continuity, species continuity, momentum, total enthalpy, turbulent kinetic energy, and turbulent kinetic energy dissipation are solved using a time-marching scheme. These transport equations are written as:

$$\frac{\partial \rho}{\partial t}+\frac{\partial}{\partial x_j}\left(\rho u_j\right)=0 \tag{1}$$

$$\frac{\partial \rho \alpha_i}{\partial t}+\frac{\partial}{\partial x_j}\left(\rho u_j \alpha_j\right)=\frac{\partial}{\partial x_j}\left[\left(\rho D+\frac{\mu_t}{\sigma_\alpha}\right)\frac{\partial \alpha_i}{\partial x_j}\right]+\omega_i \tag{2}$$

$$\frac{\partial \rho u_i}{\partial t}+\frac{\partial}{\partial x_j}\left(\rho u_j u_i\right)=-\frac{\partial p}{\partial x_i}+\frac{\partial \tau_{ij}}{\partial x_j} \tag{3}$$

$$\begin{aligned}\frac{\partial \rho H}{\partial t}+\frac{\partial}{\partial x_j}\left(\rho u_j H\right)=\frac{\partial p}{\partial t}+Q_r+\frac{\partial}{\partial x_j}\left(\left(\frac{K}{C_p}+\frac{\mu_t}{\sigma_H}\right)\nabla H\right)+\frac{\partial}{\partial x_j}\left(\left((\mu+\mu_t)-\left(\frac{K}{C_p}+\frac{\mu_t}{\sigma_H}\right)\right)\nabla\left(V^2/2\right)\right)\\+\frac{\partial}{\partial x_j}\left(\left(\frac{K}{C_p}+\frac{\mu_t}{\sigma_H}\right)\left(u_k\frac{\partial u_j}{\partial x_k}-\frac{2}{3}u_j\frac{\partial u_k}{\partial x_k}\right)\right)-\rho f\bullet V\end{aligned} \tag{4}$$

$$\frac{\partial \rho k}{\partial t}+\frac{\partial}{\partial x_j}\left(\rho u_j k\right)=\frac{\partial}{\partial x_j}\left[\left(\mu+\frac{\mu_t}{\sigma_k}\right)\frac{\partial k}{\partial x_j}\right]+\rho(\Pi-\varepsilon) \tag{5}$$

$$\frac{\partial \rho \varepsilon}{\partial t}+\frac{\partial}{\partial x_j}\left(\rho u_j \varepsilon\right)=\frac{\partial}{\partial x_j}\left[\left(\mu+\frac{\mu_t}{\sigma_\varepsilon}\right)\frac{\partial \varepsilon}{\partial x_j}\right]+\rho\frac{\varepsilon}{k}\left(C_1\Pi-C_2\varepsilon+C_3\Pi^2/\varepsilon\right) \tag{6}$$

A predictor plus corrector solution algorithm is employed to provide coupling of the above equations. A second-order central-difference scheme is employed to discretize the diffusion fluxes and source terms. For the convection terms, a second-order upwind scheme with adaptive total variation diminishing flux limiter is employed for smooth solutions across shock discontinuities. To enhance the temporal accuracy, a second-order backward difference scheme is employed to discretize the temporal terms. Details of the numerical algorithm can be found in Ref's [6-12].

Eqn. (5) and (6) above show the extended k-ε turbulence model [13], which is suitable for supersonic flow applications, is used to provide the turbulence effects. A modified wall function approach is employed to provide wall boundary layer solutions that are less sensitive to the near-wall grid spacing. Consequently, the model has combined the advantages of both the integrated-to-the-wall approach and the conventional law-of-the-wall approach by incorporating a complete velocity profile and a universal temperature profile [11]. A finite-rate chemistry chemical reaction mechanism [8,14] is used to describe the combustion process occurs inside the scramjet combustion chamber.

In the present numerical investigation, hydrogen/air combustion conditions are of interest. A reduced 9-step hydrogen/oxygen reaction chemical kinetics mechanism [15] is employed for simulations. This chemistry model is summarized in Table 1.

Table 1. Hydrogen/oxygen chemical kinetics in Arrhenius form, $k = A\, T^N\, e^{-E/RT}$.

| Equation | A | N | E/R |
|---|---|---|---|
| $O_2 + H_2 = 2\,OH$ | 1.700E+13 | 0 | 24070 |
| $H_2 + OH = H_2O + H$ | 2.190E+13 | 0 | 2590 |
| $2\,OH = H_2O + O$ | 6.023E+12 | 0 | 550 |
| $H_2 + O = H + OH$ | 1.800E+10 | 1 | 4480 |
| $O_2 + H = O + OH$ | 1.220E+17 | -0.91 | 8369 |
| $O + H + (M) = OH + (M)$ | 1.000E+16 | 0 | 0 |
| $2\,O + (M) = O_2 + (M)$ | 2.550E+18 | -1 | 59390 |
| $2\,H + (M) = H_2 + (M)$ | 5.000E+15 | 0 | 0 |
| $H + OH + (M) = H_2O + (M)$ | 8.400E+21 | -2 | 0 |

## 3. Benchmark Validations

### 3.1. GASL Supersonic Combustion Test Case

A gas hydrogen/air reacting flow was simulated with the present finite-rate chemistry model. In the experiment, two supersonic cold hydrogen slot jets are injected into a parallel supersonic hot air stream in a dump combustor geometry, the schematic of the experimental setup is illustrated in Fig. 1. The experimental investigation of this supersonic reacting shear layers was conducted by GASL and Calspan in the Calspan 96-inch reflected shock tunnel [16]. In the present study, the finite-rate chemistry model plus the extended two-equation turbulence model of the present numerical model is employed to predict the combustion phenomenon of the high-speed shear layers and the associated heat fluxes along the combustor walls. Reaction mechanisms given in Table 1 are used for the present computation. A mesh system of 77,400 elements is generated to model the geometry of this test case. A converged solution was obtained in 10,000 time steps, with a 1.0E-06 sec time step size. The predicted pressure contours is shown in Fig. 2, which reveals the shock structure that promotes radical production mechanism in the ignition process. Fig. 3 shows the predicted Mach number contours. Figs. 4 and 5 show the predicted H2O and OH mass fraction contours indicating that the reaction starts in the mid-channel between the shear layers. The predicted and measured heat flux distributions along the top and bottom walls are shown in Fig. 6. It is evident that the present model captures the combustion starting location and obtains an accurate wall heat flux predictions. The ignition delay phenomenon can be clearly observed from the predicted numerical results.

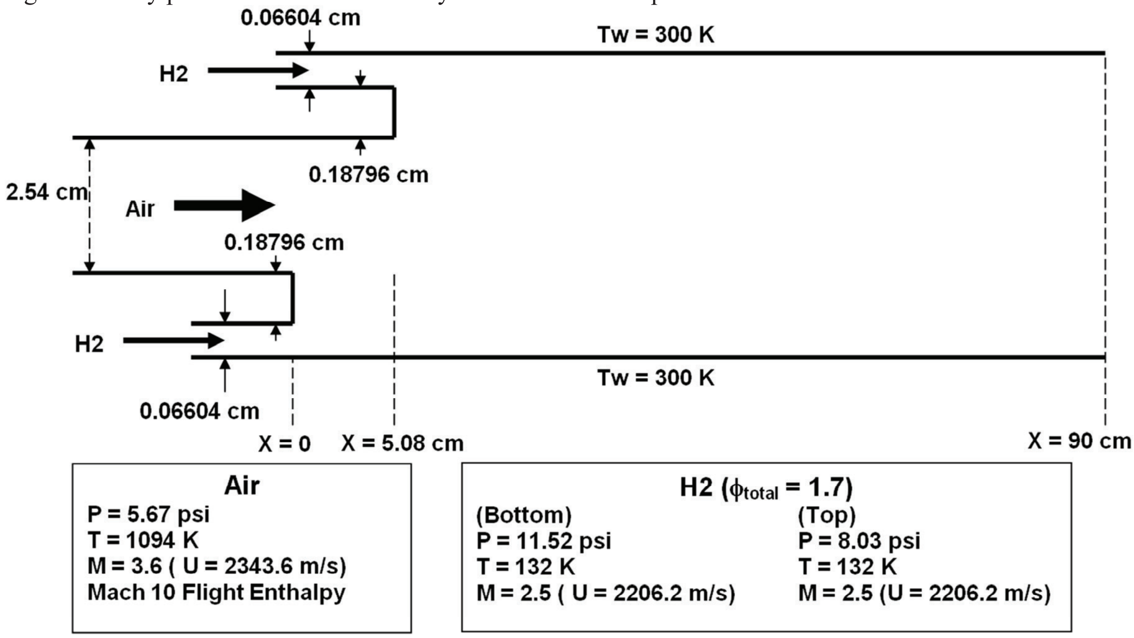



Fig. 1. GASL combustion test conditions

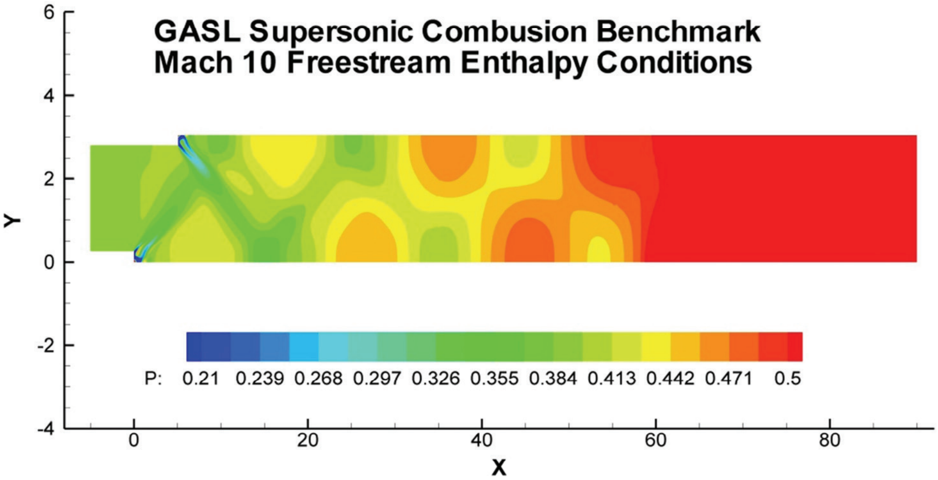



Fig. 2. Predicted pressure contours

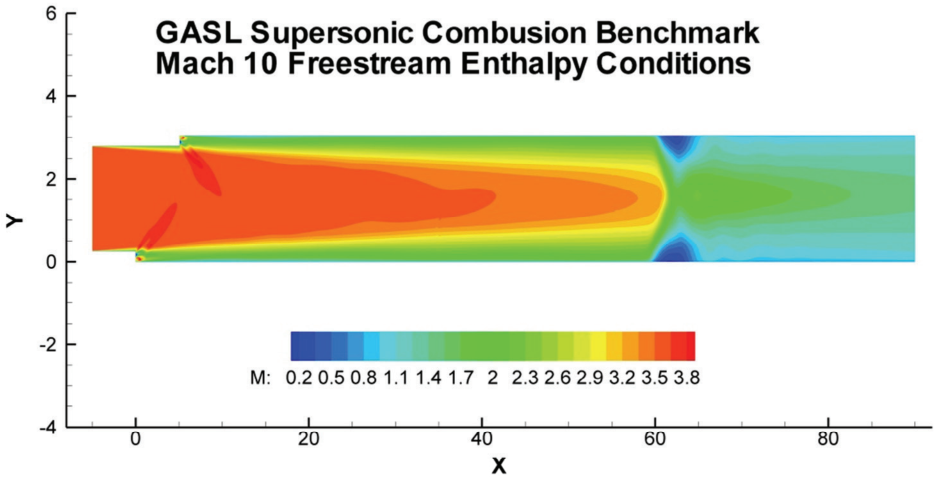



Fig. 3. Predicted Mach number contours

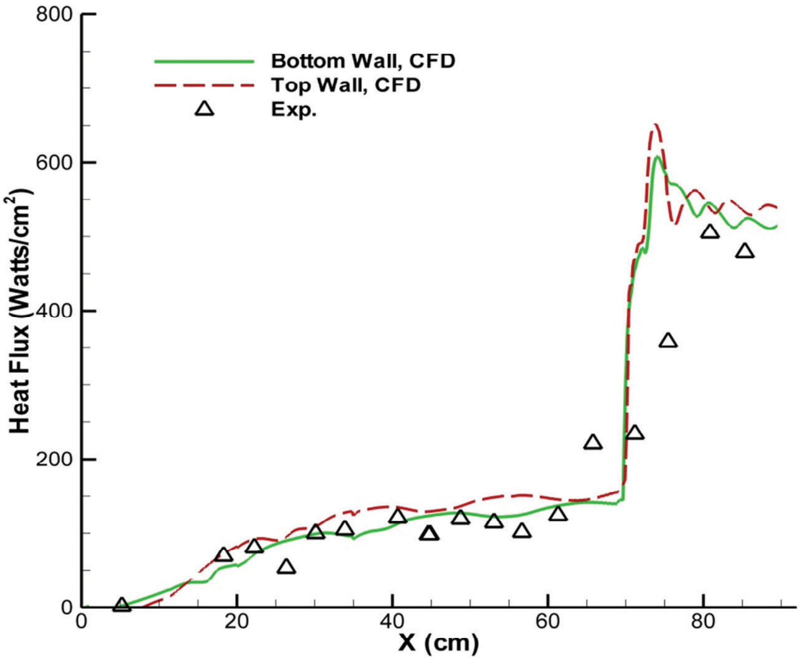



Fig. 4. Comparisons of wall heat flux distributions

### *3.2. Flow Injection into Supersonic Stream*

Injection flow has been used as an effective method for fuel mixing in supersonic stream that suffers less drag and loss penalties. In combining cavity designs [17] to increase the mixture residence time in the combustor, improved overall combustion efficiency has been observed in experiments. Turbulence effects play an important role in simulating the complex flowfield around the injection jets. To improve numerical predictions, a very large eddy Simulation (VLES) model is implemented in the present method based on an extended turbulence model [13]. An experiment of a 2D helium injection flow into a 2.9 Mach number freestream was performed by Weidner and Drummond in 1981 [18], which showed an almost 4 cm long recirculation bubble, and thus the pressure rise along the wall, upstream of the jet. The helium jet was injected with stagnation pressure and stagnation temperature of 2.551 MPa and 289.37 K, respectively. The freestream conditions are 0.0663 MPa of pressure and 108 K for the temperature. Fig. 5 shows that the predicted upstream recirculation zone length is 3.3 cm with the VLES turbulence model. Without the VLES treatment, the predicted recirculation length is only about 1.8 cm. Fig. 6 shows the predicted flowfield of a 2D N2 injection flow of Aso et al. [19] with similar improvement of the VLES model.

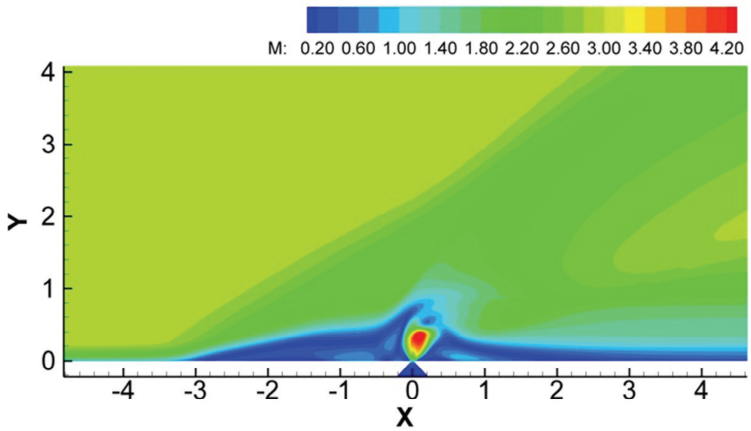



Fig. 5 Predicted Mach number of a He injection

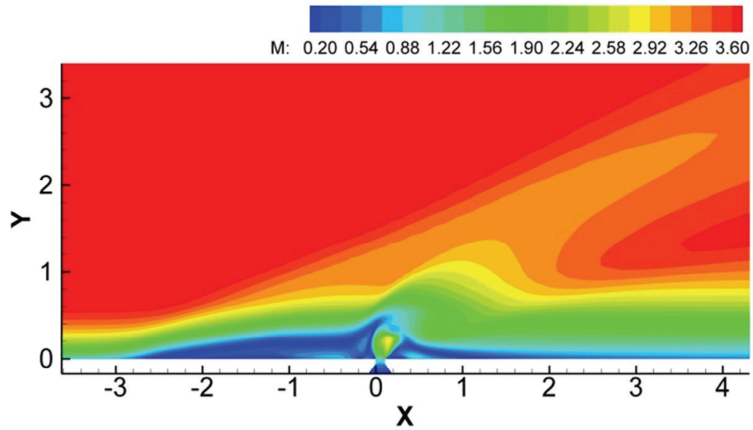


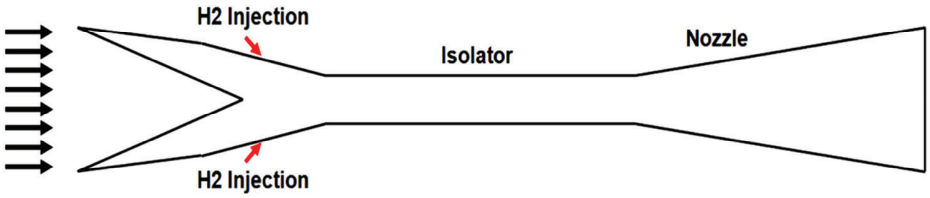

Fig. 6 Predicted Mach number of a $N_2$ injection

### *3.3. Scramjet Combustor Test Case*

Accompanied with fuel injection methods of scramjets, the shock system created in hypersonic flows, forms locally high pressure cells that assist in the creation of radical species from the fuel/oxidizer mixture in the stream [20-21]. The test case selected for the present benchmark validation was designated Run #7678 in [21], which has a Mach 6.42 free-stream air flow (2612 m/sec velocity), fuel-lean 0.51 equivalence ratio, 8958 Pa static pressure and 412 K static temperature. As illustrated in Fig. 7, the combustor height is 0.024 m and the model length is 0.625 m. The hydrogen gas injection was done using 8 injection holes (4 each equally spaced for the bottom and top walls) with 0.002 m diameter at choked conditions. The injector angle was 45 degrees measured relative to the wall surfaces. A 3D CFD model is generated to represent half the combustor model with symmetry simplification. The total number of elements is 3.888 million. A converged solution was obtained in 5,000 time steps with 1.0E-6 sec time step size.

The predicted pressure contours (ATM) and temperature contours (K) on the boundaries and the symmetry plane are shown in Fig. 8 (a) and (b), respectively. The shock system formed shows high pressure regions that help radical species creation in the supersonic stream and leads to the major combustion in the isolator. Fig. 9

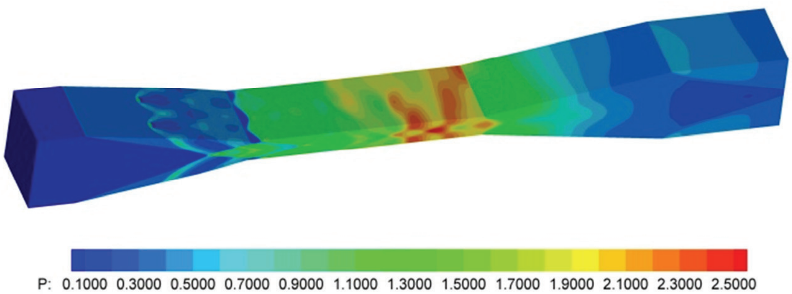
shows the predicted H2O mass fraction distributions. Fig. 10 shows data comparisons of wall pressure distributions, which gives general agreements between the present model prediction and the measured data. For sensitivity study purpose, a case with fuel mixture equivalence ratio of 0.215 is also calculated. As expected, the overall pressure level in the combustor is reduced with decreasing mixture equivalence ratios.



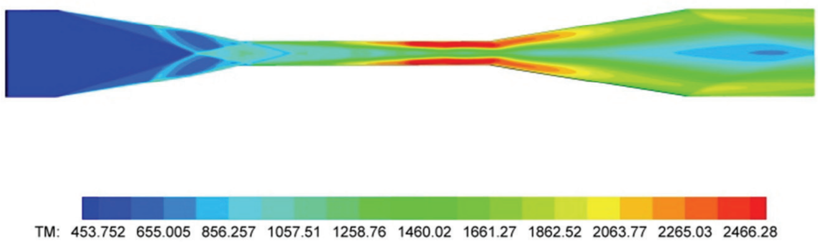

**Test Conditions**

Combustion Chamber Height = 2.4 m
M = 6.42 (U = 2612 m/sec)
H = 4.1 MJ/kg
P = 0.088 ATM
T = 412 K

Fig. 7. Geometry and test conditions of the University of Queensland supersonic combustor




(a)




(b)
Fig. 8. Predicted (a) pressure contours (ATM) and (b) temperature contours (K)

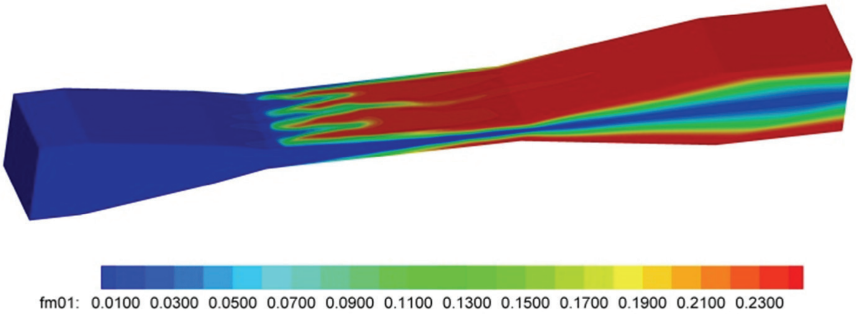



Fig. 9. Predicted H2O mass fraction

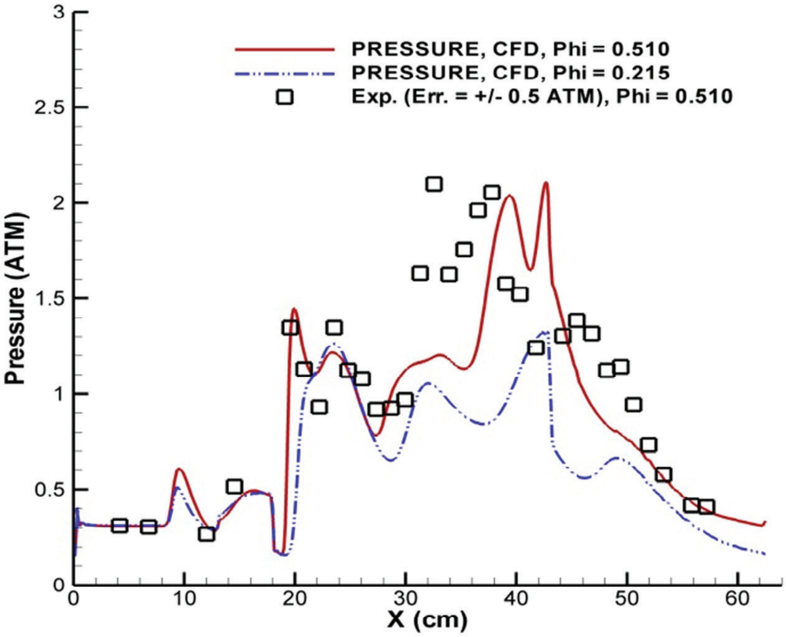



Fig. 10. Surface pressure data comparisons

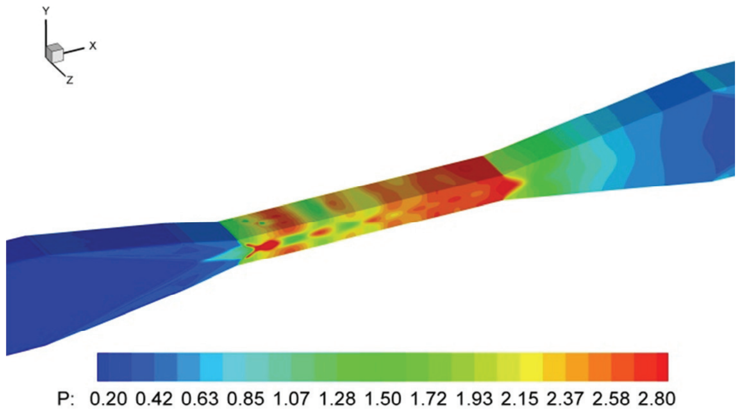
## 4. Mixing Enhancement Study

Many fuel injection and flame holding mechanisms have been proposed and studied that include wall injection, strut injection, cavity injection, etc. One important design issue in scramjet is the fully integrated propulsion system with the airframe, aiming at boosting the thrust performance by reducing losses. In this extreme flight regime, even friction losses become very significant. So, the main objective of the present study is to investigate viable methods of enhancing the fuel mixing and combustion efficiency without increasing drag penalties. To achieve this goal, drag-producing injection methods are excluded in this study. Then, the focus here is to design wall injection patterns that provide mixing enhancement and flame holding effects. And, the injection patterns are located in the straight section of the isolator such that no additional friction, pressure and momentum drags are introduced. The scramjet combustor geometry of the University of Queensland experiment discussed in section

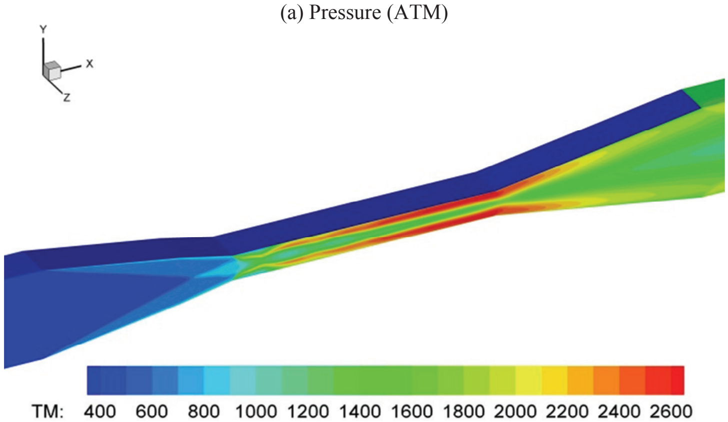
3.3 is adopted here. Just downstream of the isolator, a 5-injector pattern design is proposed to form a V-gutter shape against the incoming supersonic air. Sonic injection conditions are imposed for the hydrogen jets. The jet pressure is then adjusted for different mixture equivalent ratios for thrust performance analysis.

In the numerical model, symmetry condition is also applied for this case to save the computational efforts. Fig. 11 shows flowfield solutions of this case with equivalence ratio of 0.742. The present fuel injection pattern shows good mixing and flame holding effects by observing the flowfield solutions. The predicted thrust performance data are compared in Table 2. It is clear that the present injection method produces positive thrust with Isp ranging from 341.7 sec to 503.8 sec. It is worth noting that the friction losses are very significant for both cases.

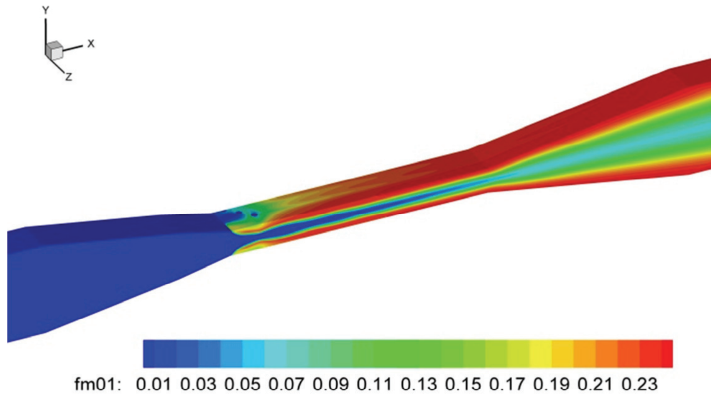
To validate the present design concept and explore its overall performance, some low cost flight experiments of the present design concept will be planned using sounding rockets after further numerical optimization study to minimize the friction losses through the flow passage.



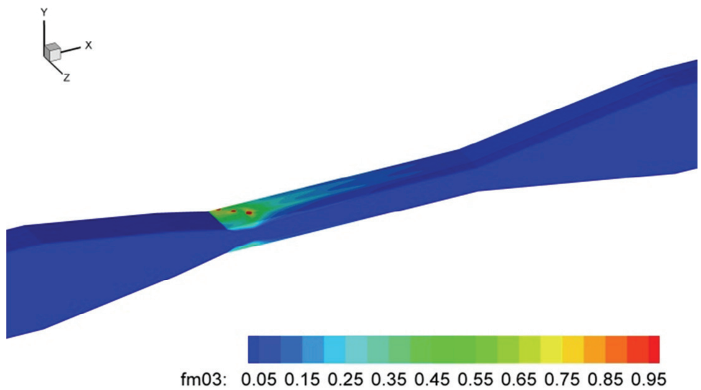

(a) Pressure (ATM)




(b) Temperature (K)




(c) H2O Mass Fraction




(d) H2 Mass Fraction

Fig. 11 Predicted flowfield of the proposed hydrogen injection pattern near the entrance of the isolator.

Table 2. Comparisons of thrust performance for different equivalence ratios.

| Equivalence Ratio | Thrust (N) | $H_2$ Flow (kg/s) | Isp (sec) | Friction Drag (N) |
|---|---|---|---|---|
| 0.594 | 36.04 | 0.01076 | 341.7 | 45.94 |
| 0.742 | 66.42 | 0.01345 | 503.8 | 48.05 |

## 5. Conclusions

A comprehensive computational model has been developed in this study for the analysis of scramjet combustion. The present numerical model is based on a Navier-Stokes flow solver that includes turbulence, finite-rate chemistry and radiative heat transfer models with all-speed capability in parallel computing. Benchmark validation cases of parallel and vertical injections in supersonic streams are presented with good results in data comparisons with experimental measurements. A flame-holding hydrogen injection has been introduced to providing enhanced mixing of air and the fuel. This arrangement gives good combustion efficiency and thrust performance of the scramjet engine.

Thrust performance degradation due to friction losses is still an important issue in scramjet design and applications. The predicted performance data indicate that the friction drag can be as high as the net thrust for the scramjet system under investigation. Fuel injection scheme and combustor internal flow passage optimization for combustion efficiency enhancement while keeping the friction losses low is a useful guideline for numerical modelling work and planning for sounding rocket flight experiments in the future research.

## Acknowledgements

This work is supported by the Suborbital Science Experiments Program of the National Space Organization of the National Applied Research Laboratory. The computational resources and services are provided by the National Center for High-performance Computing.

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