Sustained combustion with Ramp-Cavity enabled Scramjet Combustor

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Abstract

Combustion community is interested in Sustained supersonic combustion at low earth orbits of air breathing regime. Achieving thorough mixing of fuel-air, sustained combustion coupled with lower stagnation pressure losses and positive thrust are the key areas of concern. Combustion is effective with proper mixing pattern and method of fuel injection inside the supersonic combustor. There have been many studies on the effectiveness of cavities in supersonic flow. While ramps are used for mixing, cavities act as flame holders. Present paper deals with a numerical study on the performance of supersonic combustor exploring the effect of combustor entry Mach number on a model scramjet combustor. Three Mach numbers have been chosen. Ethylene is used as fuel in the studies. In the numerical experiments performed, ramp-cavity supersonic combustor is designed to study the combined effect of ramps and cavities with gaseous ethylene as fuel for achieving effective mixing. Fluent version v 15 has been used to carry out the studies. Based on the numerical experiments, it is observed that ramps in supersonic flow generate axial vortices that help in macro-mixing of fuel with air. Recirculation zones present in the cavities increase the residence time of the combustible mixture. Comparison is made on the effect of combustor entry Mach number on the flow field in the combustor. Among the three Mach numbers selected, it is observed that the higher the Mach number, the better is the performance in terms of flow field parameters

Key words: Supersonic combustion, scramjet, ramps, cavities, mixing, CFD

1. Introduction:

Supersonic combustion continues to be interesting research area for the combustion community. Air breathing propulsion has been the choice for the low earth orbit as it eliminates the need to carry oxidizer and in turn increases the pay load. Maximum cycle temperature imposes a limit on turbine functioning and its efficiency. The ramjet cycle works efficiently in subsonic speeds. Deceleration of high speed air for subsonic combustion leads to dissociation of particles, increased temperature of airstream into which further
heat cannot be added for useful thrust. Also, as speed increases, the terminal shock is generated and leads to significant pressure losses resulting in energy loss. It becomes more efficient to maintain the flow at supersonic speed throughout the engine and to add heat through combustion at supersonic speed [1]. However, supersonic combustion needs to be achieved in a short length of the combustor with residence time of supersonic air stream is of the order of 1 ms.

Both Hydrogen and Hydrocarbon fuels have been used for supersonic combustion applications. Hydrogen is used for space requirements and hydrocarbon fuels are suitable for defence applications due to reasons such as storability. Hydrogen because of its high specific impulse is used as reference fuel. Being a gas, Hydrogen is easily miscible with supersonic airstream.

Combustion community is working to devise methods to achieve ignition and sustained supersonic combustion. Achieving sustained supersonic combustion needs attention of the issues such as thorough mixing of fuel with supersonic air stream, flame holding and lower stagnation pressure losses. Mixing of the fuel and air involves both near field and far field mixing. Fuel injectors should be designed optimally and placed suitably in the combustor to reduce flow losses and to minimize the blockages to flow for good mixing.

Providing stream wise vortices in the combustor enhances mixing [2, 3]. Backward facing step has been used to generate recirculation zone with hot gases in it and that serves as a continuous ignition source [4]. However, backward facing step results in higher stagnation pressure losses. Ramps are used to generate axial vortices which help in macro-mixing of fuel with air. Wilson et al [5] conducted experimental studies on the role of ramps in improving the penetration and mixing of fuel (Mach1.9) with supersonic air stream at Mach 2.9. They identified mechanisms like baro-clinic torque, Vorticity, Magnus force for increasing the mixing. They observed that the penetration increased by 22%, plume expansion, a measure of mixing efficiency by 39% and concentration decay rate by 27%. There was a total pressure loss of 17% with the use of ramps in the low angled injection. Manoharan et al [6] carried out experimental work on supersonic cold flow mixing with Ramp Mixers. Baro-clinic torque is generated at the air-liquid interface and enhances micro-mixing due to interaction of shocks generated by ramps with the fuel stream. Cavities are first designed and used by CIAM (Central Institute of Aviation Motors), Moscow in a joint Russian/French dual-mode scramjet flight test [7]. Cavities are also used as fuel injectors and flame holders. Cavities provide re-circulation zones and act as flame holding devices. Supersonic combustion with cavity-strut injection of supercritical kerosene in a model scramjet engine was experimentally investigated by Sun Ming-bonet al [8] in Mach 2.92 facility with the stagnation temperatures of approximately 1430 K. The experimental results showed that the injection by thin struts could not acquire a steady flame when mixing field cannot match well with the cavity separation region. Hangbo Wang et al [9] investigated the characteristics of cavity assisted hydrogen jet combustion in a supersonic flow at a Mach number of 2.52, simulating flight conditions at Mach 6.

Experimental studies involve cost intensive facilities for arriving at optimal design with the consideration parameters affecting supersonic combustion. Simulation studies permit to study variation in parameters to arrive at optimization of systems and processes. Mishra and Sridhar [10] carried out a numerical study on the effect of fuel injection angle on the performance of a 2-D supersonic cavity combustor. Direct injection of hydrogen fuel into the cavity with various injection angles for non-reacting and reacting flow conditions was studied using Fluent software. They found that 1350 injection angle leads to maximum pressure losses in non-reacting conditions and 1200 in reacting conditions. 1200 injection angle shows highest combustion efficiency due to mixing in reacting conditions. Tahsini and Mousavi [11] investigated the effect of impinging oblique shock on combustion efficiency when hydrogen was injected into the supersonic cross flow. The 2D simulation showed that the shock impinging upstream of the injection slot, tilting the fuel jet to the upstream, increasing the oblique shock strength, and using hydrogen peroxide in fuel stream increase the combustion efficiency.

Combustion enhancement with Hydrogen injection through a transverse slot nozzle in a cavity combustor was studied by Kyung Moo Kim et al. [12]. Fundamental studies on cavity based flame holders in a non-reacting supersonic flow were studied by Gruber et al. [13]. Based on L/d ratio (length to depth ratio of cavity) and shear layer reattachment, cavities are of two types; open and close cavities [14, 16]. Due to low pressure losses experienced by open cavities (L/d>10), it is essential to prefer open cavities in supersonic flow conditions. Fuel air mixing is also aided by large scale shear oscillations associated with cavities. Hillier et al studied numerically the open cavity flows for L/d >10 of annular, rectangular-section cavities in a Mach 2.2 flow [14]. Flow characteristics and flow field structures for three types of rectangular cavities for different
Mach numbers were studied by Dang Guo Yang et al. [15]. Ben-Yakeret al. [16] used the cavities for flame stabilization in a solid fuel supersonic combustor and demonstrated self-ignition as well as sustained combustion of polymethyl-methacrylate (PMMA) under supersonic flow conditions.

Present numerical investigations provide insight into the flow field conditions in reacting conditions when ethylene fuel is injected into the ramp-cavity combustor to evaluate full-scale model scramjet combustor performance. For these conditions, combustor entry Mach number of 2, 2.5 and 3 are considered in the present study. Making use of advantages of both Ramps and Cavities, a combustor is designed to study its geometry and flow field conditions in order to analyze the combustor performance/flow-field at combustor entry Mach numbers with Ethylene as fuel.

2. Methodology:

2.1 Combustor design geometry:

The schematic view of a combustor configuration used for the simulations is shown in Fig. 1. To integrate with the flight vehicle, rectangular combustor section is used. Hence, for this study, a full-scale combustor model of cross section 86 mm X 275 mm with a length of 1850 mm is considered as illustrated in Fig. 1. The combustor has one constant area section and other diverging sections with top wall divergence as given in the figure. Cantilevered ramps are configured on both the top and bottom walls. The ramps are located alternately on top and bottom walls as shown in Fig. 2. First three sets of ramps consist of two ramps each on top and bottom walls of the combustor. Final set consists of one ramp each on top and bottom wall of the combustor. Open type cavities are located on the top wall of the combustor to achieve flame holding in supersonic combustion. Diverging portions of the combustor are proposed to avoid thermal choking.

Fig. 1 Supersonic Ramp-Cavity Combustor

Fig. 2 Ramp Arrangement in the Combustor
In the present work, gaseous ethylene is considered as fuel. Fuel injection is carried out through injectors on both sides of each ramp.

### 2.2 Fuel injection schemes:

The combustor configuration is designed with cantilevered ramps arranged in the top and bottom walls of the combustor alternately. Fuel injection through ramp is shown in Fig.3.

![Fuel injection scheme in the Ramp-Cavity combustor](image)

### 2.3 Simulation studies:

In the present work, gaseous ethylene is considered as fuel. For the present study, fuel injection has been considered in such a way that heat addition should not cause any upstream interaction leading to combustor on-start condition.

Simulations are carried out with combustor entry Mach number of 2, 2.5 and 3. ANSYS Fluent v15.0 software is used. The software has been validated for flows involving supersonic combustion. In the present work, the density based solver is used to transport the multi-species system. The transport equations are solved using the explicit discretization scheme. The realizable k-ε model with standard wall function is used to transport the multi-species system which is better compared to k-ω model. Single step chemistry model is considered. The laminar finite-rate model is used for solving the species volumetric reaction. This model computes the chemical source terms using Arrhenius expressions, and ignores the effects of turbulent fluctuations. The parallel processing is achieved through Message Passing Interface (MPI) technique.

### 2.4 Governing equations for computational studies:

The following governing equations are used for simulation in Fluent code

**The Mass conservation equation:**

The equation for conversation of mass, or continuity equation, can be written as follows:

$$\frac{\partial \rho}{\partial t} + \nabla \cdot (\rho \mathbf{v}) = S_m$$

The Source $S_m$ is the mass added to the continuous phase from the dispersed second phase
Momentum Conservation Equations:

Conservation of momentum in an inertial reference frame is given as:

\[ \frac{\partial}{\partial t} (\rho \vec{v}) + \nabla \cdot (\rho \vec{v} \vec{v}) = -\nabla \rho + \nabla \cdot (\vec{T}) \]

Where \( \rho \) is the static pressure, \( \vec{T} \) is the stress tensor

The Stress tensor is given by \( \vec{T} = \mu \left[ (\nabla \vec{v} + \nabla \vec{v}^\top) - \frac{2}{3} \nabla \cdot \vec{v} \right] \)

Where \( \mu \) is the molecular viscosity, \( I \) is the unit tensor and the second term on the right hand side is the effect of volume dilation.

Energy Equation:

\[ \frac{\partial}{\partial t} (\rho E) + \nabla \cdot (\vec{v} (\rho E + P)) = -\nabla \left( \sum_j h_j I_j \right) + S_h \]

Equations of State:

The transport equations described above must be augmented with constitutive equations of state for density and for enthalpy in order to form a closed system. In the most general case, these state equations have the form

\[ \rho = \rho(p, T) \]

\[ dh = \frac{\partial h}{\partial T} l_p \, dT + \frac{\partial h}{\partial p} l_r \, dp \]

\[ = c_p \, dT + \frac{\partial h}{\partial p} l_r \, dp \]

\[ = c_p = c_p(p, T) \]

Various special cases for particular material types are described below.

Incompressible Equation of State

This is the simplest case: density is constant and \( c_p \) can be (at most) a function of temperature

\[ \rho = \rho_{\text{spec}} \]

\[ dh = c_p \, dT + \frac{dp}{\rho} \]

\[ c_p = c_p(T) \]

Ideal Gas Equation of State

For an ideal gas, density is calculated from the ideal gas law and \( c_p \) can be (at most) a function of temperature

\[ \rho = \frac{w_{\text{abs}}}{k_0 T} \]
\[ dh = c_p \, dT \]
\[ c_p = c_p(T) \]

Where \( w \) is the molecular weight, \( P_{abs} \) is the absolute pressure and \( R_0 \) is the universal gas constant.

**Transport Equations for the Realizable k-\( \epsilon \) Model**

\[
\frac{\partial}{\partial t}(\rho k) \frac{\partial}{\partial x_j} + \left( \rho k u_j \right) = \frac{\partial}{\partial x_j} \left[ \left( \mu + \frac{\mu_t}{\sigma_k} \right) \frac{\partial k}{\partial x_j} \right] + G_k + G_b - \rho \epsilon - Y_M + S_k
\]

and

\[
\frac{\partial}{\partial t}(\rho \epsilon) \frac{\partial}{\partial x_j} + \left( \rho \epsilon u_j \right) = \frac{\partial}{\partial x_j} \left[ \left( \mu + \frac{\mu_t}{\sigma_\epsilon} \right) \frac{\partial \epsilon}{\partial x_j} \right] + \rho C_1 S \epsilon + \rho C_2 \frac{\epsilon^2}{k + \sqrt{\epsilon}} + C_{1s} \frac{\epsilon}{k} C_{3s} G_b + S_\epsilon
\]

Where

\[ C_1 = \max \left[ 0.43, \frac{\eta}{\eta + \sqrt{2S}} \right] \]
\[ \eta = S \frac{k}{\epsilon}, S = \sqrt{2SS} \]
\[ \mu_t = \rho C_{\mu} \frac{k^2}{\epsilon} \]
\[ C_{\mu} = \frac{1}{A_0 + A_s \frac{kr}{\epsilon}} \]
\[ U^* = \sqrt{S_{ij} S_{ij} + Q_{ij} \tilde{Q}_{ij}} \]
\[ A_0 = 4.04, A_s = \sqrt{6} \cos \phi \]
\[ \phi = \frac{1}{3} \cos^{-1} \left( \sqrt{6} W \right), W = \frac{S_{ij} S_{jk} S_{ki}}{S^3} \sqrt{S_{ij} S_{ij}}, S_{ij} = \frac{1}{2} \left( \frac{\partial u_j}{\partial x_i} + \frac{\partial u_i}{\partial x_j} \right) \]

\[
\frac{\partial}{\partial t} (\rho Y_i) + \nabla \cdot (\rho \bar{v} Y_i) = \nabla \bar{J}_i + R_i + S_i
\]

**Mass Diffusion in Turbulent Flows**

In turbulent flows, ANSYS FLUENT computes the mass diffusion in the following form

\[ \bar{J}_i = -\left( \rho D_{i,m} + \frac{\mu_t}{S_{Ci}} \right) \nabla Y_i - D_{T,i} \frac{\nabla T}{T} \]

Where \( S_{Ci} \) is the turbulent Schmidt number, by default \( S_{Ci} \) is 0.7.

**The Laminar Finite – Rate Model**

The laminar finite – rate model computes the chemical source terms using Arrhenius expressions,
\[ R_i = M_{wi} \sum_{r=1}^{nR} \tilde{R}_{ir} \]

Where \( M_{wi} \) is the molecular weight of species \( i \) and \( \tilde{R}_{ir} \) is the Arrhenius molar rate of species \( i \).

The complete computational domain is meshed with hexahedral elements with prism layers near the wall. Sufficient care has been taken to capture the flow phenomena.

### 2.5 BOUNDARY CONDITIONS:

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Air</th>
<th>Ethylene</th>
</tr>
</thead>
<tbody>
<tr>
<td>( P_0 ), bar</td>
<td>5</td>
<td>12</td>
</tr>
<tr>
<td>( T_0 ), K</td>
<td>1000</td>
<td>253</td>
</tr>
<tr>
<td>( M )</td>
<td>3</td>
<td>1</td>
</tr>
</tbody>
</table>

The stagnation pressure at the inlet to the combustor is 5 bar corresponding to the high altitude conditions. The inlet of the combustor model is defined as Pressure Far Field with two different Mach Numbers 2 and 3. In each simulation, constant Mach number of 2 or 2.5 or 3 is considered at the inlet and hence Far field pressure condition is employed. The Outlet of the model is defined as Pressure Outlet. The inlet of the ramp holes is defined as the Mass Flow Inlet with total mass flow rate of 133 g/s for equivalence ratio 0.6. Only half of the combustor is considered for simulation purpose, the wall along the width is considered as a symmetry plane. The combustor walls are considered adiabatic wall condition.

- **Turbulence model:** \( k-\varepsilon \) Realizable model with enhanced wall functions
- **Species:** \( C_2H_4, O_2, CO_2, \) and \( H_2O \) for the case of Ethylene studies
- **Reaction model:** Laminar volumetric model
- **Fuel mass flow rate:** 133 gm/s
- **Fuel Temperature:** 253 K
- **Fuel Pressure:** 1200000 Pa

### 2.6 Simulation methodology:

In the present work density based solver with an explicit scheme has been used. All the convective and diffusive terms are discretized with a second order accurate scheme. Navier- stokes, 3-D simulations were carried out for the given configuration with solver and models in two steps.

Step -1: Fuel is injected into the combustor through ramps for mixing with supersonic airstream

Step-2: Reaction (combustion) is initiated after completing the cold flow with fuel addition.
3. Grid Independence Study

The flow path of the full-scale combustor is shown in Fig. (4). The computations with different entry Mach numbers have been carried out. Fluent software is used to carry out the computational studies on a ramp-cavity based scramjet combustor for non-reacting and reacting flow conditions. 2 million, 4 million and 8 million grids have been generated and grid independence study has been carried out.

![Flow path of the combustor](image)

Ethylene fuel combustion with supersonic airstream at combustor entry Mach number 2 and 3 has been analyzed for the studies. Mach number, static pressure, static temperature and mass fractions of species have been plotted along the combustor length and compared for 2 million grid, 4 million grids and 8 million. Mach number distribution shows that the flow features have been captured by the grids and plotted along the combustor. The grids follow the same contour and 2, 4 million grids closely match each other. Similarly, the static pressure plots for 2 million and 4 million grids show that 2 and 4 million grids follow the same trend. Static pressure plot with 4 million grids simulates slightly lower pressure at about 0.8 m and 0.9 m along the length of the combustor. Static pressures for both cases closely match in the diverging portion of the combustor. Static temperature plots for both 2 million and 4 million grids show that both closely match along the combustor. Static temperature plot with 4 million grids predict higher temperature than 2 million grids. The mass fraction contours of O2, CO2 and H2O also show that the variation with the use of both the grids is minimum and comparable. Static pressure plot for 2, 4 and 8 million grids has been generated and presented in the figure (5). Static pressure for 4 million grids is closely matching with the 8 million grids. Though 8 million grids are finer, in order to capture flow field characteristics and optimize the machine time a trade-off is considered. Hence, the simulations were carried out with 4 million grids throughout the computational studies.

![Static Pressure variation along the combustor](image)
4. Results and Discussion

Computational studies of Ramp-Cavity combustor have been carried out with ethylene as fuel. Combustor entry Mach numbers of 2, 2.5 and 3 have been studied. The flow field analysis has been done for all the cases. The Mach number, static pressure and static temperature contours have been evaluated.

Fig.(6) Flow field along combustor for entry Mach number M3. Mach number variation along combustor:- (a) Non reacting flow with ethylene at combustor entry M 3.0 (b) Reacting flow with ethylene at combustor entry Mach 3.0. Static Pressure (Pa) along the combustor:- (c) Non reacting flow with ethylene at combustor entry Mach 3.0 (d) Reacting flow with ethylene at combustor entry Mach 3.0. Static Temperature (K) contours along the combustor:- (e) Non reacting flow with ethylene at combustor entry Mach 3.0 (f) Reacting flow with ethylene at combustor entry Mach 3.

The flow properties for the combustor entry Mach number 3 have been presented. To appreciate the flow characteristics along the width of the combustor, Mach number, Static pressure and Static temperature were shown across the combustor section at regular intervals. The combustor entry Mach numbers are considered constant for the simulation. Combustor configuration is simulated with 3-D analysis. Comparison plots of the flow field for three different entry Mach numbers are shown in the fig.10. The plots are drawn along the centre-line of the symmetry plane. This dimension, y is exactly at the centre of the symmetry plane. The variation of Mach number along the length of the combustor is depicted in Fig.6 (a) for cold flow with fuel injection (only mixing, with no combustion). It can be seen that there is a drop in Mach number along the length. At the vicinity of ramps, there appeared to be a significant drop in Mach number due to thorough mixing. Also, three dimensional oblique shocks are observed in the ramp zone. The shocks have resulted in enhanced mixing of the fuel with the supersonic airstream. Also, ramps develop contra rotating vortices in the supersonic air stream and induce baro-clinic torque which results in proper mixing of fuel with incoming supersonic air. However, at the zone adjacent to end of ramps and cavities, the flow got decelerated due to entrainment of airstream in the cavities. It can be clearly observed that within the cavities, the flow is subsonic due to the presence of recirculation zones. The shear layer is decreasing along the combustor while the Mach number is increasing.
In case of reacting flow (combustion), as in Fig. 6 (b), the flow is still supersonic in the close regions of ramps but got decelerated heavily at the entry regions of cavities when compared to non-reacting flow. This can be substantiated that the combustion of fuel has resulted in drop in Mach number but with significant increase in static pressure. The combustion of fuel has driven the flow with more energy that would lead to higher thrust. It can further be seen that the fluctuations in Mach number as well as static pressure are more significant in the zone of cavities than in the regions of ramps. This is demonstrating the fact of the mandatory provisions such as cavities in achieving flame stabilization.

![Mach number contours](image)

(a) Combustor entry Mach Number 2 
(b) Combustor entry Mach Number 2.5 
(c) Combustor entry Mach Number 3

The Mach number contours at different planes along the combustor for reacting flow are shown in Fig. 7. It is seen that the Mach number is mostly supersonic in the combustor except in the cavities during combustion. There is deceleration of the flow but observed to be supersonic in the core of the combustor.

The static pressure along the combustor is shown in Fig. 6. (c) for air flow with fuel injection. It is observed that the pressure increases in the ramp zone. Ramps are arranged alternately in the combustor section and there is an angle of divergence on the top wall of the combustor. Due to these variations in the geometry of the combustor, there is a corresponding variation in static pressure in the combustor with ramps. At the ramps, due to better mixing of fuel with air, there is an increase in pressure and the pressure reduces after ramps. Fuel mixing is also enhanced due to multiple three dimensional shocks and results in the pressure rise. After the ramps, the pressure decreases because of expansion and increases due to recirculation zone in the cavity. There is a pressure rise in the cavity zone due to recirculation and further the pressure reduces in the diverging combustor. The pressure variation along the combustor follows supersonic flow pattern.
In the reacting flow (combustion), shown in Fig. 6(d), the pressure remains low near to the ramp zone and the variation is dominant in the ramps zone due to multiple shocks. The pressure rise is high locally near the fuel injecting ramps indicating thorough mixing and combustion. There is continued pressure rise in the cavity zone which is caused due to recirculation in the cavities and higher pressure is clearly seen in the combustor. The pressure reduces to the ambient towards the end of the diverging combustor. It can be observed that the pressure rise is marginal with fuel mass addition to the supersonic airstream. The static pressure rise is high and more dominant in the cavities zone with heat addition compared to ramps zone. This may be due to mixing occurring in the ramps and heat addition that takes place downstream indicating high pressure rise. The static pressure rise is about 1 bar between cold flow, with fuel addition and combustion. Also, there is a static pressure rise of about 0.5 bar between ramp zone and downstream ramps, near cavities. As with the Mach number, static pressures across the planes are shown along the combustor in Fig.8. It is seen that the static pressure increase is high in the combustor flow field with entry Mach number 3. However the pressure decelerates towards the exit of the combustor. The combustor flow field with entry Mach number 3 shows uniform pressure increase towards the end of the ramps and near cavities.

![Static Pressure variation at combustor entry Mach Number 2 during combustion](image)

Fig.8 Static Pressure variation at combustor entry Mach Number 2 during combustion. (a) Static pressure for Combustor entry Mach Number 2. (b) Static pressure for combustor entry Mach Number 2.5. (c) Static pressure for combustor entry Mach Number 3.

The variation of static temperature along the combustor is depicted in Fig.6 (e) for cold flow with mass addition (non-reacting flow). The static temperature increases slightly due to mixing of the air stream with ethylene. The three dimensional shock structure and rise in temperature are clearly observed in the ramp zone. The rise in temperature may be due to mixing in the ramp zone. At the point of fuel injection from the ramps and up to cavities in the combustor, the increase in temperature is noted. However, there is a marked difference in temperature rise at the cavities. The shear layer along the top wall of the combustor is observed to carry high temperature, about 1200 K, air fuel mixture and the temperature in the diverging combustor is also locally high.
The static temperature variation for the three Mach numbers, 2, 2.5 and 3 is shown in Fig. 9. It is observed that the static temperature rise is more uniform along the planes near ramps and cavities in the combustor with air entering at Mach number 3 compared to the other cases.

![Fig. 9 Static Temperature variation at combustor entry Mach Number 2 during combustion (a) Static temperature for combustor entry Mach Number 2 (b) Static temperature for combustor entry Mach Number 2.5 (c) Static temperature for combustor entry Mach 3](image)

Fig. 9 Static Temperature variation at combustor entry Mach Number 2 during combustion (a) Static temperature for combustor entry Mach Number 2 (b) Static temperature for combustor entry Mach Number 2.5 (c) Static temperature for combustor entry Mach 3

Fig. 10 depicts the plots of Mach number, static pressure and static temperature for reacting flow (combustion) with combustor entry Mach numbers of 2.0, 2.5 and 3.0. The analysis of the flow field is shown in Fig. 10. Fig. 10 (a) shows the Mach number distribution along the combustor. There are variations in Mach number due to mixing in the ramps zone, decrease in Mach number at cavities and Mach number rises in the diverging combustor. However, the Mach number suddenly drops to about 1.3 from 2.0 at a distance of about 120 mm from the entry of the combustor due to strong oblique shocks in the case of combustor with entry Mach number 2.0. With Mach 2.5 and Mach 3.0 combustor entry Mach number cases, multiple oblique shock structure and mixing due to ramps cause variations in the Mach number plots. It is observed that locally there will be subsonic pockets with Mach number below 1.0 in the case of combustor entry Mach 2.0. In other cases, the Mach number remains supersonic throughout. Decrease in the Mach number can be found at the cavities also and then the Mach number increases in the diverging combustor. From these plots, it can be observed that with the combustor entry Mach number of 3, the Mach number in the combustor follows the trend and remains supersonic throughout the combustor.
Fig. 10 Flow field Plots for Combustor entry Mach numbers. (a) Mach number variation along the combustor (b) Static Pressure variation along the combustor. (c) Static Temperature variation along the combustor

Fig. 10 (b) depicts the static pressure variation along the combustor for entry Mach numbers 2, 2.5 and 3. It can be seen that in all the cases, the static pressure at the entry to the combustor is the nozzle exit pressure. In the case of combustor entry Mach number of 2, there is a sudden rise in the static pressure at about 120 mm from the entry of the combustor corresponding to the Mach number decrease. This may be due to the strong oblique shocks in the combustor. The pressure rises and falls in the ramp zone due to mixing because of vortices. The pressure reduces at the cavities and again increases because of reflecting shocks emanating from the cavities. The pressure reduces further in the diverging combustor. In the case of entry Mach number 2.5, similar pattern is observed with that of Mach 2. However, the pressure has risen to above 2.5 bar and then varies in the ramp zone as with the Mach number contour. The pressure decreases in the cavities due to sudden expansion and increases again due to pressure recovery. In the plot showing static pressure for combustor entry Mach number 3, the pressure rise in the ramp zone due to mixing of fuel and air is following the trend. The static pressure increased to 1.7 bar in the combustor. After the ramps, the pressure decreased due to diverging portion of the combustor. The pressure fall in the cavities is due to expansion which rises again because of reflecting shocks from the cavity base. The pressure decreases in the diverging portion of the combustor in line with the supersonic combustion. The difference in the static pressure between entry and exit is not high.

The static temperature variation along the combustor for the different combustor entry Mach numbers is depicted in Fig.10(c). The static temperature plot is also similar to the trend in Mach number and static pressure plots along the combustor. In the case of combustor entry Mach number 2, the static temperature rise started at 120 mm, increased further in ramps and after cavities. In the case of combustor entry Mach number of M2.5, the static temperature fluctuated in the ramps zone due to mixing, alternate compression and expansion. The static temperature rise can be seen in the cavity zone which reduces in the
diverging combustor. In the case of combustor entry Mach number 3.0, the static temperature variation in the ramps indicate better mixing as the temperature rise can be observed after 550 mm from the beginning of the combustor which increased continuously except in the cavities because of expansion in the area. The static temperature further increases indicating the recirculation zone in the cavity due to which the flame stabilization and supersonic combustion is sustained in the combustor. The static temperature reduces further in the diverging combustor.

The mass fractions of species are depicted in Fig.11 in which the combustion of supersonic air stream at different entry Mach numbers mixes with ethylene fuel of 0.6 equivalence ratio. In the case of oxygen, mass fraction decreases gradually in the ramps zone and after 1 m length there is a decrease in the content. The oxygen mass fraction plot in the case of Mach 2.5 also shows similar pattern. In the case of combustor entry Mach number of 3, the oxygen content decreases after 400 mm from the start of the combustor where mixing and combustion happens. This continues upto 1.2 m of the combustor and then the oxygen mass fraction increases. It may be due to the completion of most of the combustion of ethylene with air in the combustor upto the cavities.

![Flow field Plots for Combustor entry Mach numbers 2, 2.5 and 3](image)

Fig. 11 Flow field Plots for Combustor entry Mach numbers 2, 2.5 and 3 (a) Mass fraction of oxygen along the combustor. (b) Mass fraction of H2O along the combustor. (c) Mass fraction of CO2 along the combustor

The mass fraction contour of CO2 shows that for combustor entry Mach number 2, the mass fraction increases from about 200 mm and continuously increases upto 0.8 m of combustor length, slightly reduces and then increases upto 1.4 m of the combustor length. This indicates combustion to continue in the diverging portion of the combustor also. In the case of combustor entry Mach numbers 2.5 and 3 the increase in CO2 mass fraction starts after 400 mm from the start of the combustor and continues upto 1.1 m of combustor.
length before decreasing. This may be due to complete mixing and combustion between 400 mm to 1100 mm length of the combustor. The mass fraction of H2O for combustor entry Mach number 2 shows a similar trend as that of CO2 mass fraction in the combustor. The increase in the case of combustor entry Mach number 3.0 is at about 550 mm from the start and continued up to 1000 mm length of the combustor. This indicates the combustion could be completed in a relatively short length of the combustor. It can be observed that when the combustion entry Mach number is 3, the Mach number in the combustor is supersonic throughout, the pressure rise is as per the expected trend and the temperature rise is substantial indicating supersonic combustion, compared to the combustor entry Mach numbers 2 and 2.5. It may be concluded that for the combustor configuration with ramps and cavities as located in this combustor, combustor entry Mach number 3 is more suitable for sustained supersonic combustion.

5. Conclusions

Numerical simulations have been carried out for the geometry of the combustor with gaseous ethylene as fuel. Studies are aimed at understanding the net effect of combustor entry Mach number on the ramp-cavity combustor. Based on the study, the following conclusions are drawn.

i. With ramp-cavity arrangement, effective mixing and combustion are achieved.
ii. The blockage due to ramps is well within the limit and did not affect the flow.
iii. Among three Mach numbers studied, a case of Mach number 3 (highest) has yielded satisfactory results.

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