

DESIGN AND NUMERICAL ANALYSIS OF ASYMMETRIC NOZZLE OF SHCRAMJET ENGINE

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ABSTRACT

The aim of this paper is to present a design approach for the nozzle of Shcramjet vehicle and its optimization in terms of flow and geometrical parameters. The flow properties of the asymmetric nozzle of Shcramjet vehicle are examined through analytical and numerical analysis. Method of Characteristics (MOC) was solved to obtain the two dimensional geometries of the nozzle taking the slant angle of initial MOC line as the varying parameter. The geometries were designed as per specific combustor outflow conditions at the speed of 12.5 Mach and the altitude of 32.5 km and were numerically simulated in ANSYS Fluent using density based solver. The pressure expansion ratio of the nozzle was expressed as a function of slant angle of initial MOC line with its maximum value obtained for nozzle with largest sizing. The deviation between numerical and analytical result induced upon by assuming flow to be calorically perfect is also studied and presented.

General Terms

Hypersonic, aerospace, flow expansion, SSTO.

Keywords

Characteristic lines, expansion waves, Method of Characteristics.

1. INTRODUCTION

Shock Induced Combustion Ramjet (Shcramjet) [1,2,3,4] engine is a hypersonic air breathing propulsion system in which combustion of fuel-air mixture flow takes place across induction shock waves. The ignition in Shcramjet engine is initiated by oblique shock waves with sudden rise in pressure and temperature which is in contrast with the Scramjet engine with slow diffusive mode of combustion. The igniting oblique shock waves (OSWs) can be stabilized over a wedge, strut or a ramp in a combustion chamber, thus called 'standing OSWs'. In particular, instead of multiple shock waves, if a single strong shock wave is used to ignite the mixture flow at once, the OSW transitions into a single standing oblique detonation wave (ODW).

The Shcramjet engine fully retains the conceptual simplicity of scramjet engine and incorporates the same components, namely, inlet, combustor and nozzle, but has two important benefits- significantly shorter combustor length and requirement of lesser inlet diffusion [2]. The entire Shcramjet 'waverider' vehicle forebody is used as the inlet for compression and the main engine (an elongated channel) on the ventral side of the vehicle and the entire afterbody is used as the nozzle for generation of thrust. However, the compression at inlet required is not limited by combustor entrance speed but instead mainly by entrance temperature such that no premature ignition of fuel occurs during mixing. In comparison to Scramjet, Shcramjet is argued to have better performance above Mach 12 [3, 4]. This makes it the candidate for next stage propulsion system for hypersonic flight as well as SSTO after Scramjet, or as an alternative to chemical rockets for atmospheric flight up to Mach 22 [2].

The application of shock induced combustion and detonation waves for propulsion applications in mass launchers and hypersonic air breathing engines was first studied by Dunlap et al. in Reference [5]. The early application for oblique detonation wave (ODW) was mainly studied for ram accelerator [6] where a dual-wedge-shaped body is accelerated to hypersonic speeds through a channel filled with fuel-air mixture. Experimental studies for wedge-stabilized ODWs were carried out in Ref. [7], and later experimental studies on shock-induced combustion and ODW were reported in Ref. [8, 9, and 10]. In a study by Chan J. et al. in Ref [11], a three dimensional numerical simulation of the aero propulsive performance characteristics of a Scramjet and a Shcramjet at a flight Mach number of 11 was studied and compared. Another comparison study was performed by Dubeabout R. et al. in Ref. [3], where the fuel-specific impulse of scramjet given by Heiser H. et al. in Ref. [12] was compared to that of a Shcramjet.

Method of Characteristics (MOC) is a method to reduce PDEs to a family of ODEs along which the solutions can be integrated. In 1949, Guentert C. in Ref. [13] at NASA Lewis implemented the analytical approach in practice for designing

an axisymmetric method of characteristic nozzles for wind tunnel. Korte J. et al. in Ref. [15] developed a design program which incorporates a modern approach to the design of supersonic/hypersonic wind-tunnel nozzles. Ali H. et al. in Ref. [16] calculated minimum length of the supersonic nozzle for the optimum Mach number at the nozzle exit, with uniform flow at both converging and diverging section of the nozzle using method of characteristics.

Zebbiche T. et al. in Ref. [17] used Method of Characteristics to develop the profile of supersonic axisymmetric Minimum Length nozzle to have a uniform and parallel flow at the exit section. Furthermore, this paper studied the changes in geometry by varying the stagnation temperature and exit Mach numbers. Mbuyamba J. et al. in Ref. [18] calculated and designed the internal profile of a supersonic nozzle for Cold Gas Dynamic Spray (CGDS) process by the application of Method of Characteristics. Moreover, this paper developed a new MATLAB code capable to calculate and design the internal profile of the high performance (high gas speed with no or reduced shock waves) MOC nozzles.

The present paper reports the two dimensional numerical simulation of asymmetric nozzle of Shcramjet. MATLAB code has been developed to solve the method of characteristics at a specific combustor outflow condition of Shcramjet taking slant angle of initial MOC line as varying parameter in order to determine the optimum nozzle based on its sizing and its performance.

2. ANALYTICAL METHOD

The analytical model assumes the flow to be inviscid, irrotational, isentropic and calorically perfect with specific heat ratio (γ) assumed as 1.22. The equation governing steady, adiabatic, two dimensional, irrotational, supersonic flow is given as:

$$(V_x^2 - a^2) \frac{\partial V_x}{\partial x} + V_x V_z \left(\frac{\partial V_x}{\partial z} + \frac{\partial V_z}{\partial x} \right) + (V_z^2 - a^2) \frac{\partial V_z}{\partial z} = 0$$

$$\frac{\partial V_z}{\partial x} - \frac{\partial V_x}{\partial z} = 0$$

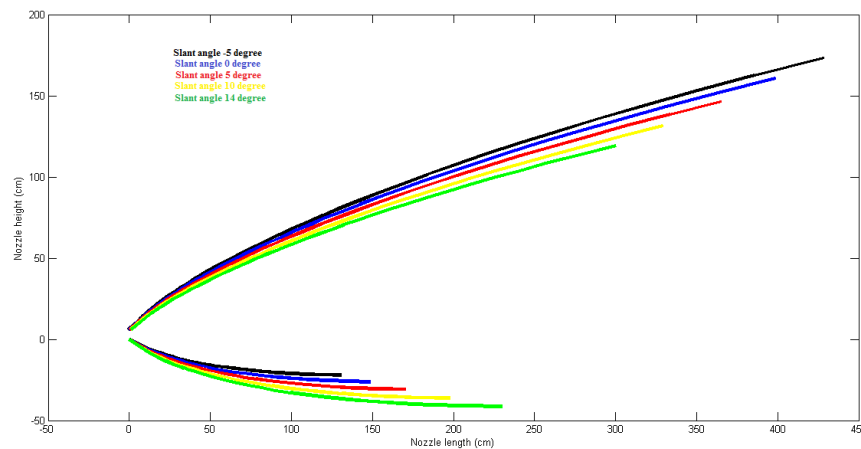


Fig 1: Profile for body and cowl sections of the nozzle obtained from MATLAB

V_x and V_z are velocity components in x and z direction respectively and 'a' is the speed of sound. The compatibility equation is solved along the characteristic lines representing expansion waves to calculate the flow properties and the nozzle contour. Compatibility equation is given as:

$$\theta \pm \gamma(M) = \text{constant}$$

The left running characteristic line is represented by the positive sign whereas, the right running characteristic line is represented by the negative sign. The Prandtl Meyer function is the dimensionless measure of speed and is the function of Mach number, expressed as:

$$\gamma(M) = \int \frac{\sqrt{M^2 - 1}}{1 + \frac{\gamma-1}{2} M^2} \frac{dM}{M}$$

The nozzle operating conditions as listed, were taken as per the combustor outflow conditions of Shcramjet vehicle. The slant angle of initial MOC line was varied as per -5° , 0° , 5° , 10° and 14° to obtain five geometries. The flow from the combustor is assumed to be at pressure 807018 Pa, Mach 2.4679 and at flow angle of 14° following deflection from the combustor ramp.

Table 1: Operating conditions

Parameters	Value
P_{in}	807018 Pa
P_{atm}	1500 Pa
M_{in}	2.4679
Θ_{in}	14°

3. ANALYTICAL RESULT

Analytically, MOC was solved to expand the combustor outflow from pressure of 807018Pa to the atmospheric pressure of 1500Pa for different geometries which were obtained by varying the slant angle. The variation in the nozzle sizing with change in slant angle is as shown in figure 1 and 2. With the increase in slant angle, the height of the nozzle and cowl length increased whereas the body length decreased with minimum nozzle sizing obtained at the slant angle of 14°. Nozzle with minimum size are desirable as larger nozzle contributes to greater vehicle weight and increased drag. On increasing the flow angle from the combustor, the length of the body increased whereas that of the cowl decreased. The increase in slant angle was seen to counteract this effect.

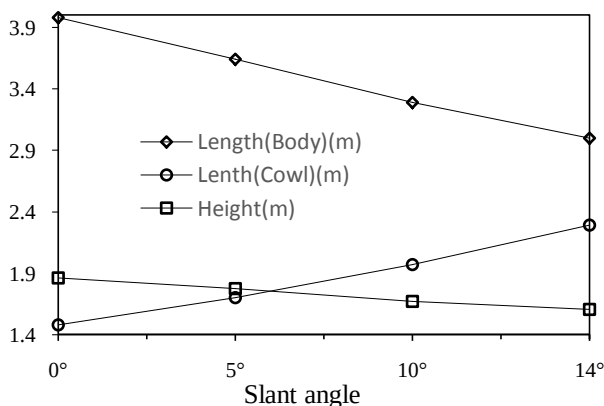


Figure 2: Nozzle sizing

4. NUMERICAL METHOD

The Shcramjet inlet flowfield is computed using 2-D Euler equations. The mass, momentum and energy balance equations for inviscid, compressible and unsteady flow, with species scalar equation, is given as:

$$\frac{\partial U}{\partial t} + \frac{\partial F}{\partial x} + \frac{\partial G}{\partial x} = S$$

Where,

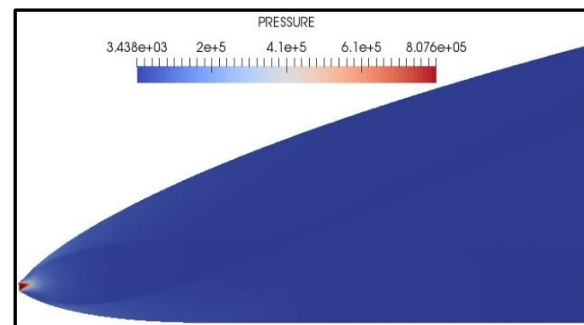
$$U = \begin{pmatrix} \rho \\ \rho u \\ \rho v \\ \rho \left(e + \frac{v^2}{2} \right) \\ \rho Y_i \end{pmatrix}, \quad F = \begin{pmatrix} \rho u \\ \rho u^2 + p \\ \rho uv \\ \rho u \left(e + \frac{v^2}{2} \right) + pu \\ \rho u Y_i \end{pmatrix}$$

$$G = \begin{pmatrix} \rho v \\ \rho uv \\ \rho v^2 + p \\ \rho v \left(e + \frac{v^2}{2} \right) + pv \\ \rho v Y_i \end{pmatrix}, \quad S = \begin{pmatrix} 0 \\ 0 \\ 0 \\ 0 \\ R_i \end{pmatrix}$$

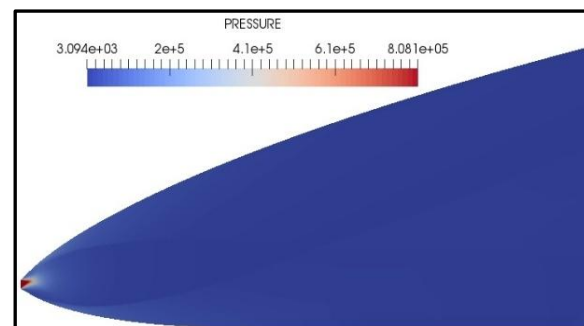
U, **F** and **G** are solution vector and flux vectors respectively. ρ is the density with u and v the velocity components in x and y direction. The simulation was performed in ANSYS Fluent using a density based solver. Advection Upstream Splitting Method, AUSM [Liou et al. 1993] flux vector splitting scheme was used for flux splitting with third order MUSCL

extrapolation for primitive variables [vanLeer B. 1979]. Inlet was set as pressure far field with outlet an extrapolated boundary and body and cowl set as adiabatic wall. The flow was assumed to be chemically frozen with concentration of species H_2 , O_2 , H_2O and N_2 being 0.002912, 0.02352, 0.209568 and 0.764 respectively as per 88% combustion of stoichiometric mixture in combustion chamber.

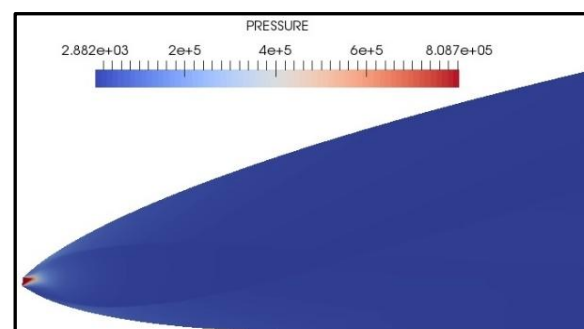
5. NUMERICAL RESULTS



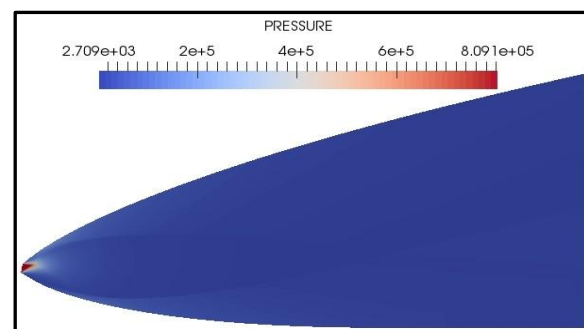
a) Slant angle = -5°



b) Slant angle = 0°



c) Slant angle = 5°



d) Slant angle = 10°

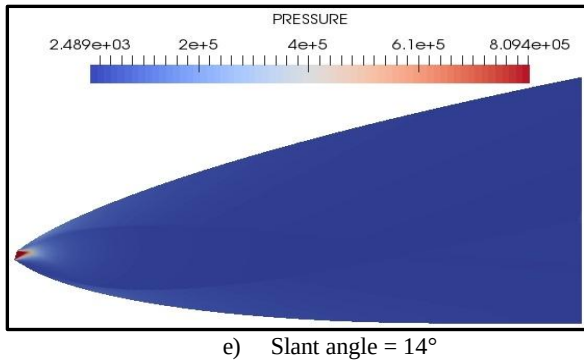


Figure 3: Static pressure (in Pa) Contour of numerically simulated geometries of five different cases

Analytically, the nozzles were designed to reduce the pressure 807018 Pa from combustion chamber to the atmospheric pressure of 1500 Pa. But the pressure obtained at the outlet of the nozzle from numerical analysis as shown in figure 4 for all cases were above 1500 Pa and thus the nozzles were under expanded. Analytical designs were based on the assumption that the flow is calorically perfect in contrast to numerical analysis, in which specific heat ratio varies with temperature, inducing the variation in result between analytical and numerical analysis.

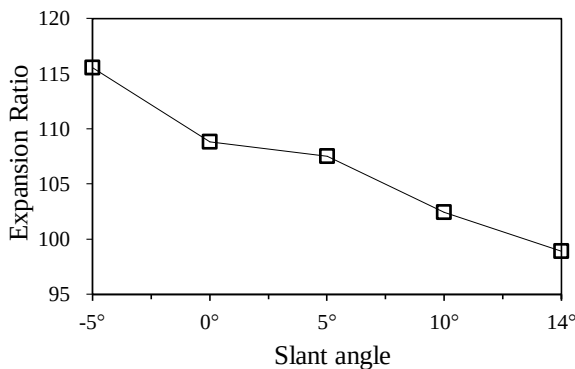


Figure 4: Variation in expansion ratio with slant angle

The area weighted average of the static and total pressure at the inlet and outlet of the nozzle were evaluated from the numerical analysis to calculate the expansion ratio and total pressure recovery respectively which are plotted in figure 2 and 3. The expansion ratio decreased with the increase in slant angle with nozzle having slant angle -5° the maximum value and 14° the minimum value. There was only a slight variation in the total pressure recovery with the change in slant angle. Further estimates of this pressure recovery can only be obtained with a viscous flow simulation, with significant pressure losses occurring in the boundary layer and boundary layer thickness becomes a significant parameter for performance measurement.

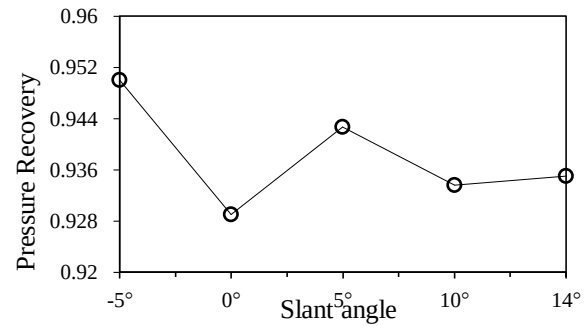


Figure 5: Variation in pressure recovery with slant angle

6. CONCLUSION

The nozzle for Shcramjet vehicle was designed to expand the combustor outflow from the pressure of 807018 Pa to the atmospheric pressure of 1500 Pa at an altitude of 32.5 km by varying the slant angle of initial MOC line to obtain five different geometries. The study shows the tradeoff between the performance of the nozzle in terms of expansion ratio and the nozzle sizing with the largest nozzle at slant angle -5° having the better performance than the smallest nozzle at slant angle 14°. The tradeoff also exists between the performance and sizing of the nozzle and its compatibility with the height of the inlet of the vehicle for straight upper body of the vehicle. In absence of the boundary layer, total pressure recovery cannot be taken as a primary selection parameter as most of the losses occur in the boundary layer.

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