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Effects on Performance Measures of Second Throat Diffuser by Computational Analysis

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Abstract: A second throat diffuser using hydrogen and oxygen as the primary liquid is considered for the creation of the low vacuum in a high altitude testing facility for large are ratio rocket CE20 engine. Detailed pressure investigations have been carried out to evaluate the performance of the second throat for various operational conditions and geometric parameters. When the diffuser attains started condition, supersonic flow fills the entire inlet section and a series of oblique shock cells occurring in the diffuser duct seal the vacuum environment of the test chamber against back flow. The most sensitive parameter that influences the stagnation pressure needed for diffuser starting is the second-throat diameter Between the throat and exit diameters of the nozzle, there exists a second throat diameter value that corresponds to the lowest stagnation pressure for starting. When large radial/axial gaps exist between the nozzle exit and diffuser duct, significant reverse flow occurs for the unstarted cases, which spoils the vacuum .The predicted axial variations of static pressure along the diffuser are analyzed using ANSYS FLUENT and the modeled was carried out using CATIA V5.

Nomenclature

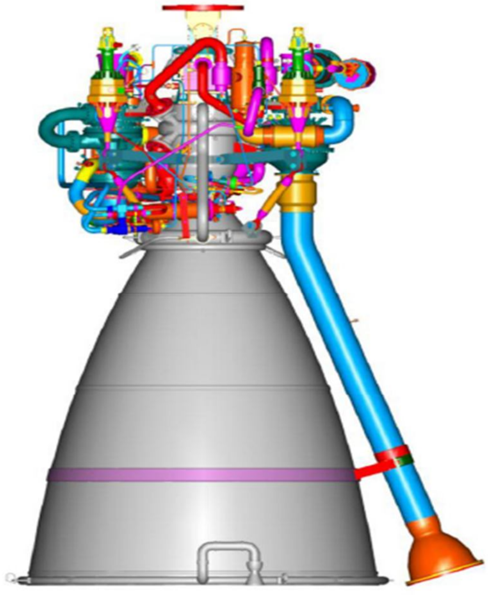
A_e = Thruster nozzle exit area
A_t = Thruster nozzle throat area
E = Specific internal energy
k = Thermal conductivity
M_i = Inlet Mach number
P_0 = Stagnation pressure
P = Static pressure
P_b = back pressure
P_v = Vacuum chamber pressure
q = Heat transfer per unit area
r = radial coordinate
T_0 = Stagnation temperature
V = Velocity
Z = Axial coordinate

I. INTRODUCTION

A. Cryogenic Engine

A cryogenic rocket engine is a rocket engine that uses a cryogenic fuel or oxidizer, that is, its fuel and oxidizer are gases liquefied and stored at very low temperatures. Rocket engines need high mass flow rate of both oxidizer and fuel to generate sufficient thrust. The liquid oxygen (LOX) oxidizer and liquid hydrogen (LH2) fuel combination is one of the most widely used.

The CE 20 is a cryogenic rocket engine developed by LPSC, ISRO. It is being developed to power the upper stage of the Geosynchronous satellite Launch Vehicle Mk III. It is the first Indian cryogenic engine to feature a gas-generator cycle. It is one of the most powerful cryogenic upper stage engines in the world.



CE 20 engine

B. Second Throat Ejector Diffuser

The secondary flow passage is essentially axisymmetric convergent divergent diffuser with a flow direction having both radial and axial components. The secondary air thus decelerated into the throat, followed by a normal shock process at a relatively low mach number in the divergent section. The primary purpose of an ejector diffuser is to reduce nozzle back pressure sufficiently to allow the nozzle to flow full at design chamber pressure, that is, without flow separation occurring in the divergent portion of the nozzle.

C. Boundary Conditions

The boundary conditions for the specified model are detailed as below:
No slip and adiabatic conditions at the ejector wall
Azimuthal symmetry on the axis
The prescribed mass flow rate liquid hydrogen of 0.459 kg/s with the stagnation temperature of 3700 k.
A segregated solver based on the SIMPLE technique is used to solve the partial differential equations,

CONDITON	VALUE
Solver type	Pressure based
Wall	No slip
Inlet mach	5.8
Model	Sparlart Allmaras

D. Operating Parameters

The geometric parameters of the cryogenic engine CE (20) to which the diffuser to be designed is,

Chamber pressure	60 bars
Mixture ratio	5.05
Area ratio	100
Specific impulse	444 s
Prop flow rate	0.459 kg/s
Vacuum thrust	200 kN
Ratio of specific heat	1.3
Stagnation temperature	3700 k

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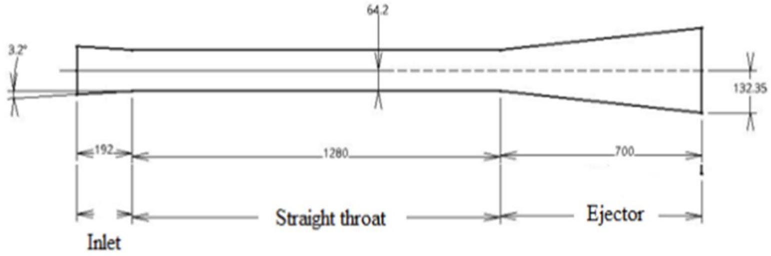
E. Geometric Parameters



F. Numerical Parameters

The geometry and the boundary conditions are used for the numerical simulation. The model was designed using the software package CATIA and the governing equations are solved using the CFD solver FLUENT.

The theoretical calculations are done for the operating parameters as listed

1) Exit Mach Number

$$\frac{A}{A^*} = \frac{1}{M}\left[\left(\frac{2}{\gamma+1}\right)\left(1+\frac{\gamma-1}{2}M^2\right)\right]^{\frac{\gamma+1}{2(\gamma-1)}}$$

$$100 = \frac{1}{M}\left[\left(\frac{2}{1.3+1}\right)\left(1+\frac{1.3-1}{2}M^2\right)\right]^{\frac{1.3+1}{2(1.3-1)}}$$

$M_e = 5.8$

2) Nozzle Exit Pressure

$$\frac{p_0}{p_e} = \left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{\gamma}{\gamma-1}}$$

$$\frac{60}{p_e} = \left(1+\frac{1.3-1}{2}5.8^2\right)^{\frac{1.3}{1.3-1}}$$

3) Nozzle Exit Temperature

$$\frac{T_0}{T} = \left(1+\frac{\gamma-1}{2}M^2\right)$$

$$\frac{3700}{T} = \left(1+\frac{\gamma-1}{2}5.8^2\right)$$

T = 611.975 k

$$M_y = \sqrt{\frac{\frac{2}{\gamma-1}+M_x^2}{\frac{2\gamma}{\gamma-1}+M_x^2-1}}$$

$= 0.3724$

4) Nozzle Throat Pressure

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$$\frac{p}{60} = \left(1 + \frac{\gamma-1}{2} M^2\right)^{\frac{\gamma}{\gamma-1}}$$

$$\frac{p}{60} = \left(1 + \frac{1.3-1}{2} 5.8^2\right)^{\frac{1.3}{1-1.3}}$$

p = 24.65 m bar

5) Temperature Downstream Of Shock

$$\frac{T_y}{T_x} = \left(\frac{1 + \frac{\gamma-1}{2} M_x^2}{1 + \frac{\gamma-1}{2} M_y^2}\right)$$

T_y = 3624.576 k

6) Pressure Recovery In Shock

$$P_1(1 + \gamma M_1^2) = P_2(1 + \gamma M_2^2)$$

$$p_y = 934.170$$

7) Nozzle Exit Velocity

$$V_e = M_e \sqrt{\gamma R T}$$

$$= 5.8\sqrt{\gamma \times 641.5 \times 611.975}$$

V_e =4143.475 m/s

The second throat diffuser can perform a high pressure recovery than straight throat diffuser,

So, The comparison was made for the different pressure recoveries.

$$M_g \times V_e \times \eta_d = \Delta P \times A \times \left(1 + \gamma M_y^2\right)$$

The pressure recovery calculations considered as,

i. 60% to 100%
ii. 65% to 100%
iii. 75% to 100%

a) Calculation For 65% To 100%

$$M_g \times V_e \times \eta_d = \Delta P \times A \times \left(1 + \gamma M_y^2\right)$$

For η = 0.65

d = 121.085mm

For η = 1

d = 150.18mm

b) Calculation For 60% To 100%

For η= 0.6

d = 116.335mm

For η = 1

d = 150.18mm

c) Calculation For 75% To 100%

For η= 0.75

d = 130.065mm

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For η= 1

d = 150.18mm

G. Design Methodology

From the numerical methodology,

We can conclude that the diameters of throat with respect to their percentage of pressure recovery as

For d = 121.085mm

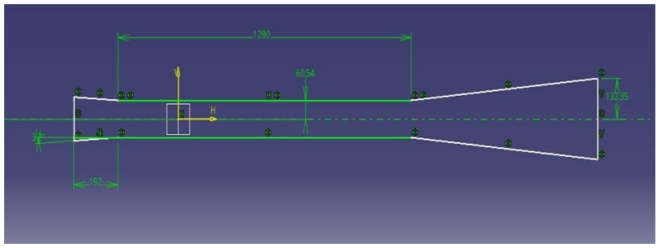

Fig 1

For d = 116.335 mm

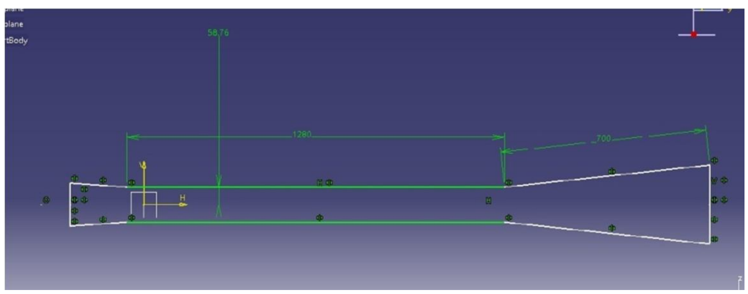

Fig 2

For d = 130.065 mm

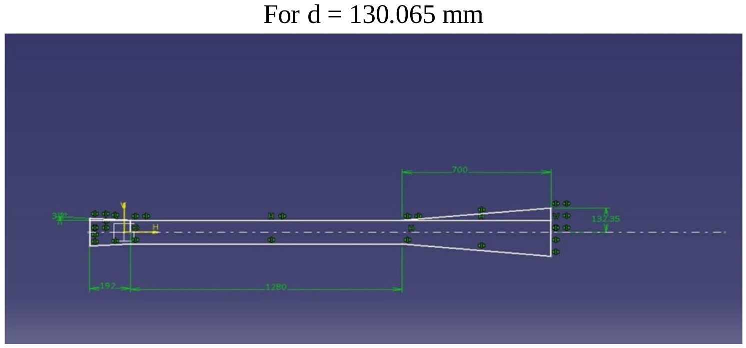

Fig 3

For d = 150.18 mm

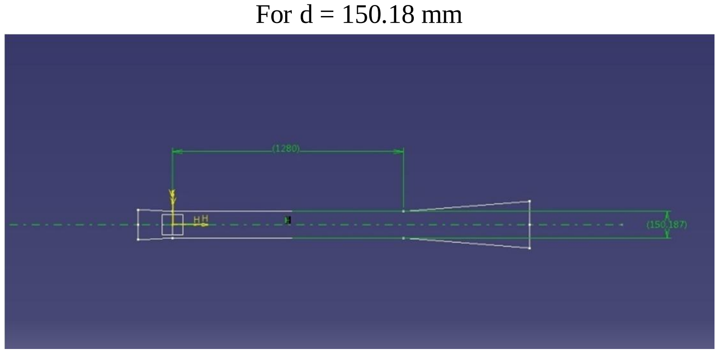

Fig 4

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For designing the diffuser the length, throat angles were standardized as
Length of the throat = 1280 mm
Inlet angle= 3.2 degrees
Length of the diffuser = 192 mm
Length of the nozzle= 700 mm
Diameter of the nozzle = 264.7 mm

II. SOLUTION METHODOLOGY

The most sensitive parameter that influences the stagnation pressure needed for diffuser starting is the second-throat diameter. Thus the analysis had been done for the pressure and other operational characteristics of a second throat diffuser for varying the diameter of the throat.

Between the throat and exit diameters of the nozzle, there exists a second-throat diameter value that corresponds to the lowest stagnation pressure for starting.

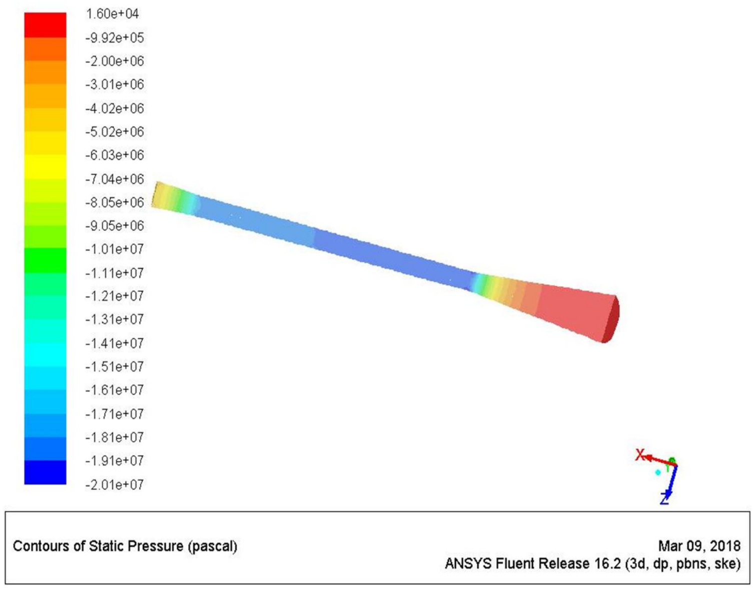

Fig 5 Static pressure contour for the throat dia = 116.335mm

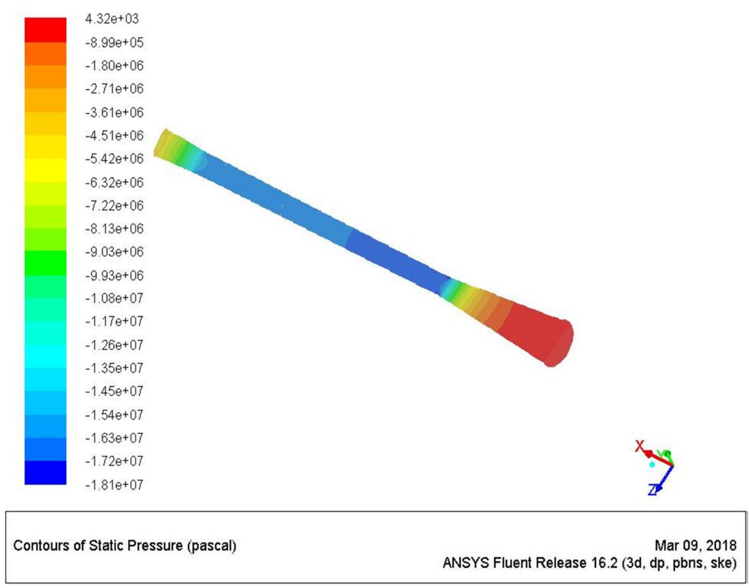

Fig 6 Static pressure contour for the throat dia= 130.065 mm

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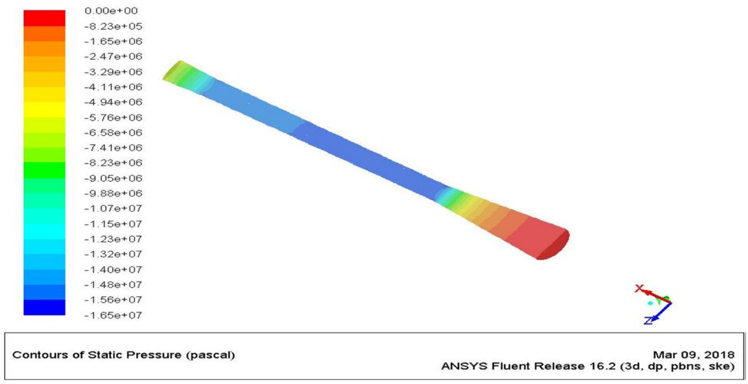

Fig 7 Static pressure contour for the throat dia = 150.15 mm

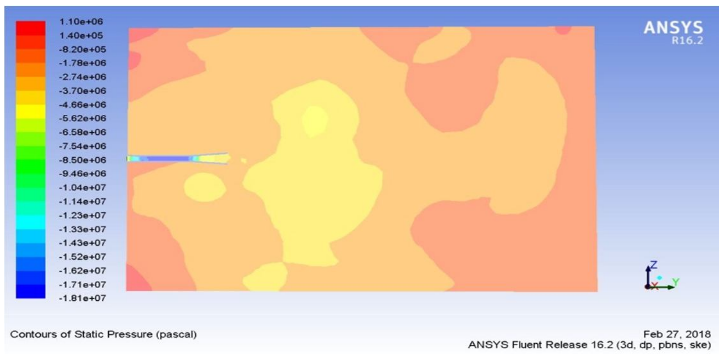

Fig 8 Static pressure contour for the throat dia = 121.085mm

As we know that the second diameter has to be large than the motor nozzle throat, since it has to accommodate additional mass flow due to entrainment.

If the diameter is very small the mass flow will be chocked and the desired conditions cannot be achieved. Among the four diameters , throat with diameter = 116.335 mm have less space inside the throat, thus have least static pressure which will not be sufficient conditions for the starting.

Throat with diameter = 130.065 mm have the more static pressure which seems to be having sufficient conditions for starting the STED

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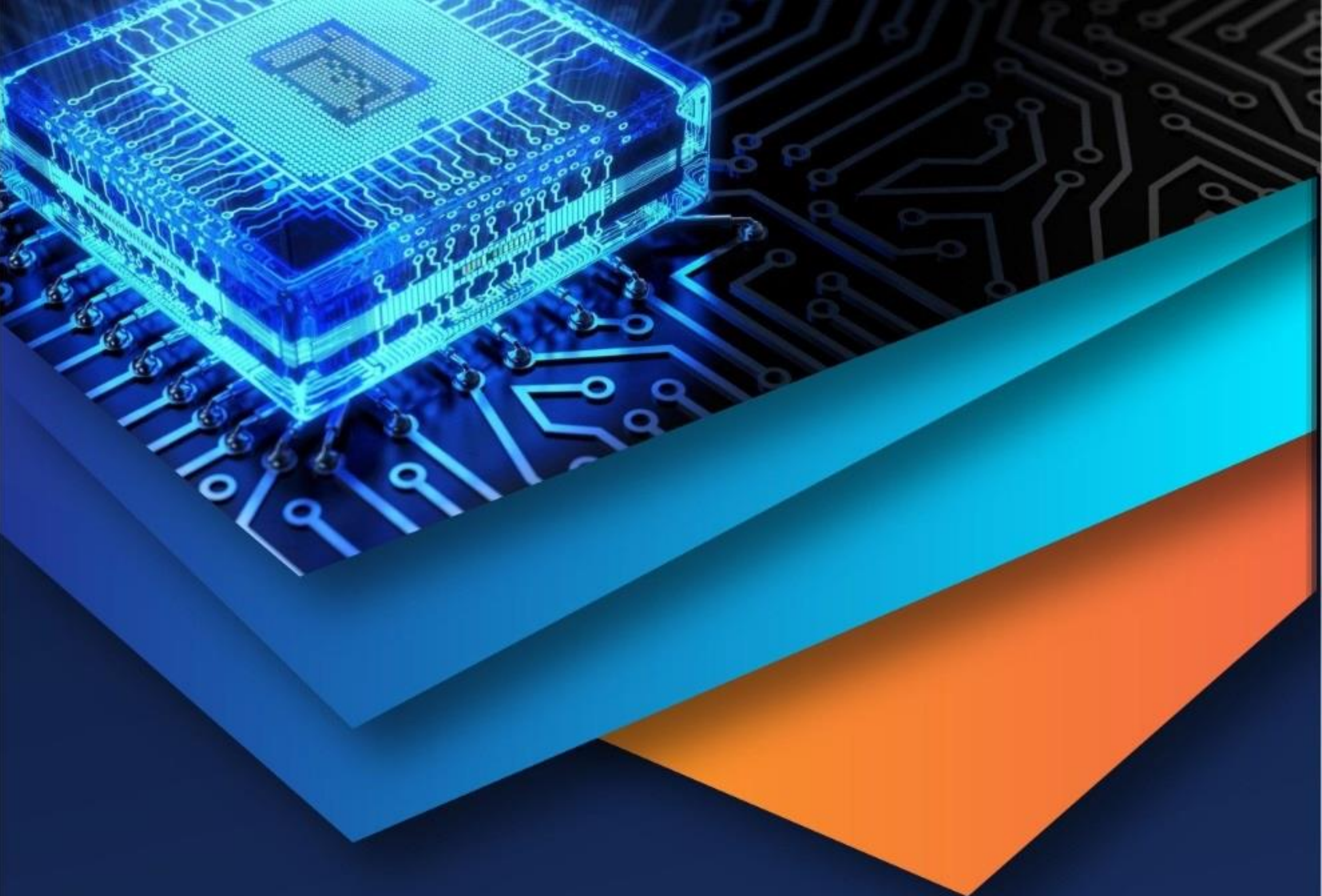

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