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Project Report on

"ANALYSIS OF ROCKET NOZZLE WITH 1 AND 4 INLETS"

Submitted in partial fulfillment of the requirements for the award of the degree

BACHELOR OF ENGINEERING
IN
MECHANICAL ENGINEERING

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CERTIFICATE

This is to certify that the project work entitled **"ANALYSIS OF ROCKET NOZZLE WITH 1 AND 4 INLETS"** is a bonafide work carried out by **RAHUL KUAMR (1MV18ME065), RISHI SHARMA (1MV18ME068), SHANKAR SHARMA (1MV18ME076)** And **VAISHNAVI BHARDWAJ (1MV18ME091)** of **Sir M. Visvesvaraya Institute of Technology**, Bangalore, in partial fulfillment for the award of degree of Bachelor of Engineering in **Mechanical Engineering** of the **Visvesvaraya Technological University**, **Belagavi** during the academic year 2021-2022. It is certified that all corrections and suggestions indicated for Internal Assessment have been incorporated in the report. The project report has been approved as it satisfies the academic requirements in respect of Project Work prescribed for the Bachelor of Engineering degree.

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ABSTRACT

A De Laval nozzle is put under study based on the number of inlets (one and four) from the combustion chamber based on rapid increases in temperature in the combustion chamber and the convergent part of the nozzle with decreases in temperature at the exit. The working fluid consider as air for the analysis. This project aims to evaluate the exit velocity, exit pressure, and exit temperature. The analysis is carried out using Computational Fluid Dynamics (CFD) software and ANSYS Fluent. This study also aims towards comparing the performance of a single inlet and four inlet nozzles at different Mach numbers (subsonic, sonic, and supersonic).

Table of Contents

LIST OF FIGURES

LIST OF TABLES

Chapter 1

INTRODUCTION

High-performance engines in rockets incorporate some form of a convergent-divergent nozzle. The analysis is carried using software like CATIA for designing the nozzle and ANSYS FLUENT for analysing the flows in the nozzle.

A comparative analysis is carried out for a single inlet and four inlet rocket nozzles at subsonic, sonic, and supersonic speeds by varying the Mach Numbers.

The flow through a convergent-divergent nozzle is one of the benchmark problems used for modelling the compressible flow through computational fluid dynamics. In this project analysis of a convergent divergent rocket nozzle is done by varying the Mach Number and obtaining results for parameters like pressure, temperature, and velocity at the exit of the nozzle.

A rocket engine uses a nozzle to accelerate hot exhaust to produce thrust as described by Newton's third law of motion. The amount of thrust produced by the engine depends on the mass flow rate through the engine, the exit velocity of the flow, and the pressure at the exit of the engine. The value of these three flow variables is all determined by the rocket nozzle design

A nozzle is a relatively simple device, just a specially shaped tube through which hot gases flow. Rockets typically use a fixed convergent section followed by a fixed divergent section for the design of the nozzle. This nozzle configuration is called a convergent-divergent**,** or CD, nozzle. In a CD rocket nozzle, the hot exhaust leaves the combustion chamber and converges down to the minimum area, or throat, of the nozzle. The throat size is chosen to choke the flow and set the mass flow rate through the system. The flow in the throat is sonic which means the Mach number is equal to one in the throat. Downstream of the throat, the geometry diverges, and the flow is isentropically expanded to a supersonic Mach number that depends on the area ratio of the exit to the throat. The expansion of a supersonic flow causes the static pressure and temperature to decrease from the throat to the exit, so the amount of the expansion also determines the exit pressure and temperature. The exit temperature determines the exit speed of sound, which determines the exit velocity. The exit velocity, pressure, and mass flow through the nozzle determines the amount of thrust produced by the nozzle.

A convergent-divergent or CD also called as a De Laval nozzle.

1.1 Nozzle

De Laval nozzle was invented by Gustaf de Laval, a Swedish inventor. It is a converging-diverging type of nozzle, generally employed to provide supersonic jet velocity at the exit of the nozzle. In this paper, analysis of De Laval nozzle is carried out theoretically by formulating required nozzle equations and the results have been validated by computer simulation using the CFD software ANSYS FLUENT.

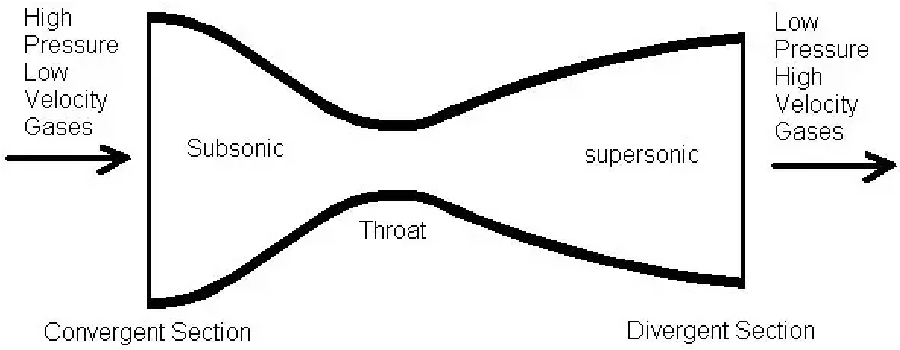


Fig 1. De Laval Nozzle

It is widely used in some types of steam turbine and is an essential part of the modern rocket engine and supersonic jet engines.

1.2. How it works:

Its operation relies on the different properties of gases flowing at subsonic and supersonic speeds. The speed of a subsonic flow of gas will increase if the pipe carrying it narrows because the mass flow rate is constant (grams or pounds per second). The gas flow through a de Laval nozzle is isentropic (gas entropy is nearly constant) and adiabatic (heat loss or gain is nearly zero). At subsonic flow the gas is compressible; sound, a small pressure wave, will propagate through it. Near the nozzle "throat", where the cross-sectional area is a minimum, the gas velocity locally becomes transonic (Mach number = 1.0), a condition called choked flow. As the nozzle cross sectional area increases the gas continues to expand and the gas flow increases to supersonic velocities where a sound wave will not propagate backwards through the gas as viewed in the rest frame of the nozzle (Mach number > 1.0).

Mach number:

The Mach number is the ratio of flow velocity after a certain limit of the sounds speed. In simple words, it is the ratio of the speed of a body to the speed of sound in the surrounding medium.

The formula of Mach Number is:

$$M = u/c$$

Where,

- The Mach number is M
- Based on the limits the local flow velocity is *u in m/s.*
- The speed of sound in that medium is *c in m/s.*

The various regimes of Mach values are:

Regime	**Subsonic**	**Transonic**	**Sonic**	**Supersonic**	**Hypersonic**	**Hypervelocity**
Mach	<0.8	0.8–1.3	1.0	1.3–5.0	5.0–10.0	>10.0

Subsonic:

Commercial aircraft with aerodynamic features such as the rounded nose and leading edges. The Mach is below 0.8.

Transonic:

Aircraft that are built with swept wings. The Mach value is between 0.8-2.1.

Supersonic:

The aircraft created to go supersonic have a definite design, it has the complete movement of the canards, thin aero foil sections, and sharp edges. The Mach levels are between 1.2 and 5.0

Hypersonic:

These planes have several distinctive features such as nickel-titanium skin that is cooled and small wings. The Mach values are between 5.0 and 10.0. The U.S. plane X-15 created the world record of flying at Mach 6.72.

High-Hypersonic:

The Mach levels are between 10.0 -25.0. When flying at such huge speeds thermal controls becomes an integral portion of the design. The hotness of the surface must be considered beforehand.

Chapter 2
LITERATURE SURVEY

This chapter includes the related studies and topics for the realization of this project. Referring to the previous studies on Rocket Nozzle Analysis which are published, patented and papers were of enormous help.

[1] performed a CFD Analysis on four inlet rocket nozzle with a Mach Number of 2.1. The paper shows light on the CFD analysis of the pressure and temperature of the nozzle with four inlets.

[2] Conducted a CFD Analysis of a Rocket Nozzle with Two Inlets at Mach 2.1 and their findings are, pressure and temperature for a rocket nozzle with two inlets at Mach 2.1 are analysed with the help of fluent software. When the fuel and air enter the combustion chamber according to the x and y plot, it is burning due to high velocity and temperature, and then temperature increases rapidly in the combustion chamber and convergent part of the nozzle, and after that temperature decreases in the exit part of the nozzle.

[3[Conducted numerical analysis to determine an optimum divergent angle for the nozzle which would give the maximum outlet velocity and meet the thrust requirements. The inlet dimensions and the boundary conditions are kept constant and the divergent angles are varied to understand how the variation in divergent angle affects the flow pattern through the nozzle. Conducted numerical analysis to determine an optimum divergent angle for the nozzle which would give the maximum outlet velocity and meet the thrust requirements. The inlet dimensions and the boundary conditions are kept constant and the divergent angles are varied to understand how the variation in divergent angle affects the flow pattern through the nozzle.

[4] This paper contains analysis over a convergent divergent rocket nozzle which is performed by varying the number of divisions in mesh. The various contours of nozzle like Cell equi-angle skew, Cell Reynolds number, Pressure, Velocity, Mach Number, and above are calculated at each type of mesh using CFD analysis software ANSYS Fluent.

Madhu BP, Syed Sameer, Kalyana Kumar M & Mahendra Mani G (2017) conducted a study on supersonic flow through the rocket nozzle which was simulated using numerical method. The parameters like Mach number, static pressure and shocks were observed for conical and contour nozzles using axi-symmetric model in ANSYS FLUENT software. The occurrences of shocks for the conical nozzles were observed along with the other parameters for various divergent angles. The parameters under observation were compared with that of contour nozzle for respective divergent angles by maintaining the inlet, outlet and throat diameter and lengths of convergent and divergent portions.

2.1 Research Gap

Literature survey suggests that:

- The expansion analysis has been done only for individual rocket nozzle at a particular Mach Number. So, there is no sufficient work done on the subsonic, sonic, and supersonic speeds at the rocket nozzle.
- None of the studies have compared the subsonic, sonic, and supersonic speeds of single and four inlet rocket nozzles.

2.2 Objectives

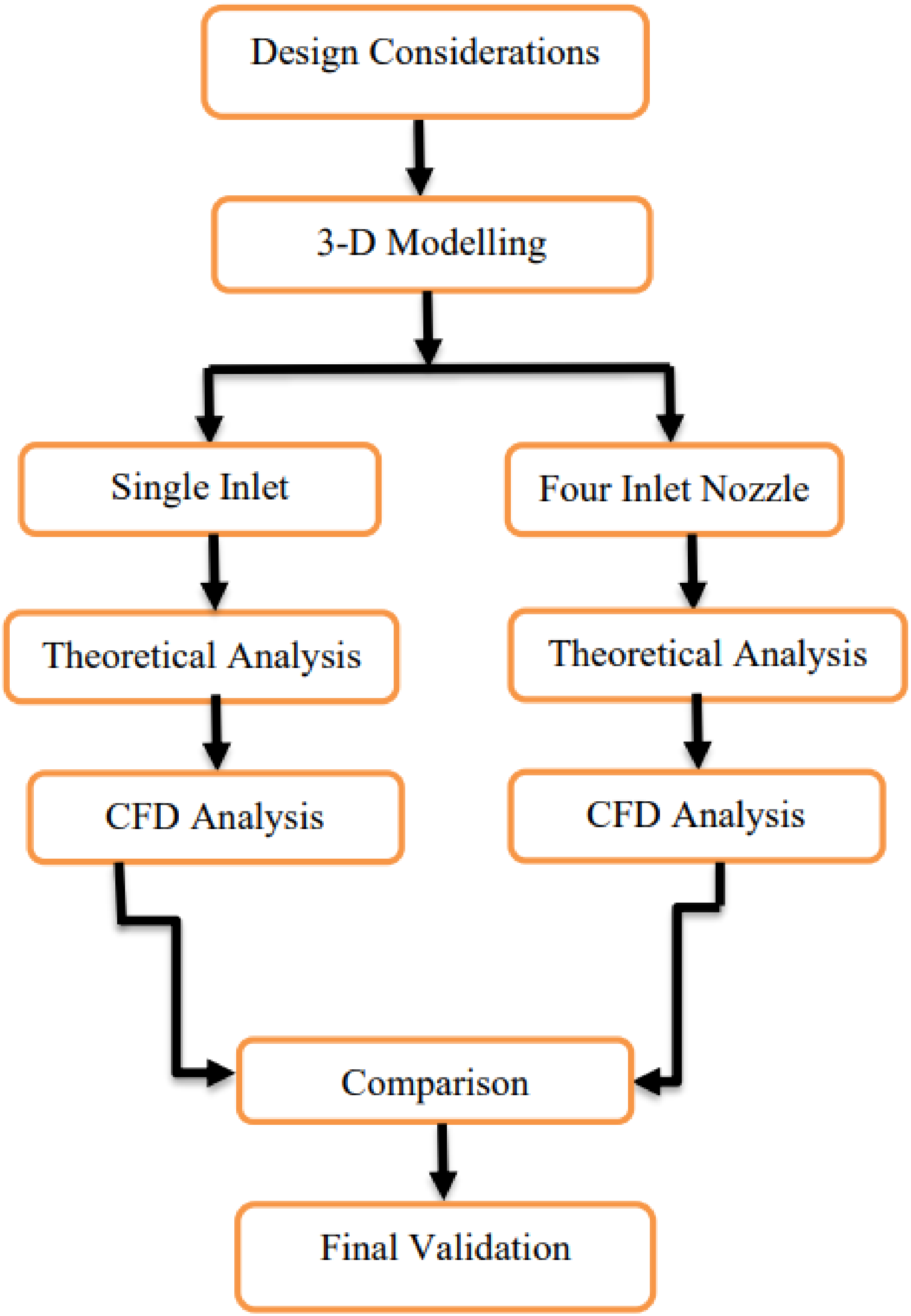
- The objective of the present work is to simulate supersonic, sonic, and subsonic flow through a convergent-divergent rocket nozzle of single and four inlets, to precisely understand the flow dynamics and variation of flow properties in the combustion chamber with the nozzle. This simulation is carried out using ANSYS software.
- To perform CFD Analysis of a De Laval Nozzle with one and four inlets using ANSYS FLUENT.
- To perform a comparative study of the performance of De Laval Nozzle for a different number of inlets based on exit temperature, exit pressure and velocity, by varying the Mach Number (subsonic, sonic and supersonic).
- The Mach no. range will be decided in the range of 0.5, 1.0 and 2.5. This based on the literature survey to achieve best result in theoretical calculations and get appropriate result in the software as well

Chapter 3
METHODOLOGY

- Related information about numerical analysis of expansion through convergent-divergent nozzles has been done.
- Geometry is designed using CATIA.
- Meshing and suitable boundary conditions are applied using ANSYS.
- Analysis of the models is done by using ANSYS FLUENT.
- Quantitative Analysis and Grid sensitivity study is performed.
- Results for different Inlets at various Mach numbers are obtained and compared. The creation of two-dimensional geometry of the nozzle is done on CATIA and meshing is done on ANSYS.
- The meshing method will be Automatic Method and the mesh type will be selected as All Quad. The analysis will be done for different inlets and Mach numbers.
- The variation of the number of divisions will be done on the inlet, exit, and on the walls of the nozzle.
- A numerical method adopted to approximate the governing equations, along with the relevant boundary conditions, by a system of linear algebraic equations is known as a discretization method.
- Thus, a problem involving calculus is transformed into an algebraic problem which can then be solved on a computer by using a solution methodology. A discretization technique and a solution methodology constitute the numerical methodology used to solve a heat transfer and fluid flow problem.

3.1 Workflow of Project:

The project methodology is depicted in the flow char



Chapter 4

3-D Modelling of DE LAVAL Rocket

4.1 Design

The design of the De Laval rocket nozzle is done based on the literature survey. This design is used for both, single and four inlets, nozzles.

The rocket nozzle is first designed using CATIA. The geometry of the nozzle is based on ideal conditions.

BOUNDARY CONDITIONS: -

Inlet Diameter	25mm
Exit Diameter	35mm
Throat	10mm
Length of the nozzle	75mm

Table 4.1 Dimensions of the Rocket Nozzle

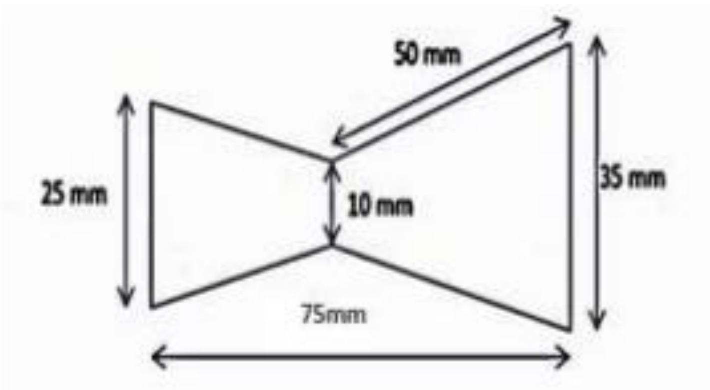


Fig 4.1 Side view of rocket nozzle

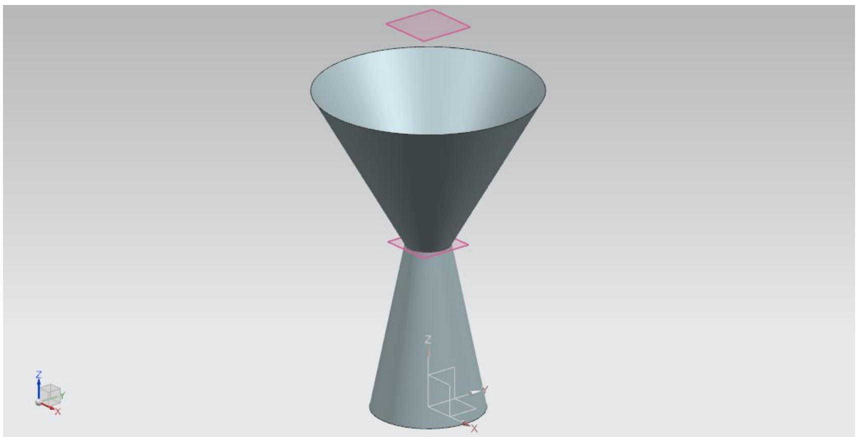


Fig 4.2 A 3D view of the rocket nozzle

4.2 Theoretical Calculations for Exit Temperature, Exit Pressure, and Exit Velocity

The working fluid is taken as air.

Ideal combustion chamber pressure is taken between 1-20 MPa.

Ideal combustion chamber temperature is taken as 3200 ^{O}C

We consider,

Combustion Chamber Pressure (P_t) = 20 MPa

Combustion Chamber Temperature (T_t) = 3500K

Specific heat ratio (γ) = 1.2

Gas Constant (R) = 0.287 KJ/kg-K

4.2.1 Single Inlet Nozzle

To calculate the exit temperature, pressure and velocity,

Exit Temperature Ratio $\frac{T_e}{T_t} = \left(1 + \left(\frac{\gamma-1}{2}\right) M^2\right)^{-1}$ ………………………………(1.1)

Exit Pressure Ratio $\frac{P_e}{P_t} = (1 + \frac{\gamma-1}{2} M^2)^{-\frac{\gamma}{\gamma-1}}$ ……………………………………(1.2)

Exit Velocity V_e=M$\sqrt{\gamma R T_e}$ ……………………………………………………..(1.3)

Mass Flow Rate $\dot{m} = \frac{AP_t}{\sqrt{T_t}} \sqrt{\frac{\gamma}{R}} M \left(1 + \frac{\gamma-1}{2} M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$ …………………………....(1.4)

Nozzle Exit Thrust F = $\dot{m}$ V_e + (P_e – P_o)A_e..……………………………………(1.5)

Where,

Mach Number - M, Exit Temperature – T_e, Exit Pressure – P_e, Area of cross section at the exit – A_e, Free Stream Pressure – P_o , Mass Flow Rate - $\dot{m}$, Area of Throat – A, Diameter of Throat = dt, Diameter at the exit of nozzle = d_e .

P_o=1.01 MPa

Area of throat A=$\frac{\pi}{4}$ d_e^2

A=1.72 m^2

The area at the exit of nozzle A=$\frac{\pi}{4}$ d_e^2

A_e=21.07 m^2

<u>At Mach 2.5:</u>

Using equation (1.2) to find exit pressure,

$$\frac{P_e}{P_t} = (1 + \frac{\gamma-1}{2}M^2)^{-\frac{\gamma}{\gamma-1}}$$

$$P_e = 20\left(\left(1 + \left(\frac{1.2-1}{2}\right)(2.5)^2\right)\right)^{-\frac{1.2}{0.2}}$$

P_e=1.0862 MPa

Using equation (1.1) to find exit temperature,

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

$$T_e = 3200\left(1 + \left(\frac{1.2-1}{2}\right)(2.5)^2\right)^{-1}$$

T_e=1969.23 ^{O}C

Using equation (1.3) to find the exit velocity,

$$V_e = M\sqrt{\gamma R T_e}$$

$$V_e = 2.5\sqrt{1.2 * 0.287 * 1969.23}$$

V_e= 61.539 m/s

Using equation (1.4) to find the mass flow rate,

$$\dot{m} = \frac{AP_t}{\sqrt{T_t}}\sqrt{\frac{\gamma}{R}}M\left(1 + \frac{\gamma-1}{2}M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$$

$$\dot{m} = \frac{1.72*20}{\sqrt{3200}}\sqrt{\frac{1.2}{.287}}2.5\left(1 + \frac{1.2-1}{2}2.5^2\right)^{-\frac{1.2+1}{2(1.2-1)}}$$

$\dot{m}$=0.35 kg/s

Using equation (1.5) to find the Nozzle Exit thrust,

$F = \dot{m}\, V_e + (P_e - P_o)A_e$.

$F = 0.35 * 61.539 + (2.2338 - 1.013) * 21.07$.

F= 25.72 MN.

At Mach 1:

Using equation (1.2) to find exit pressure,

$\frac{P_e}{P_t} = (1 + \frac{\gamma - 1}{2} M^2)^{-\frac{\gamma}{\gamma - 1}}$

$P_e = 20\left(\left(1 + \left(\frac{1.2-1}{2}\right)(1)^2\right)\right)^{-\frac{1.2}{0.2}}$

$P_e = 11.28$MPa

Using equation (1.1) to find exit temperature,

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

$T_e = 3200\left(1 + \left(\frac{1.2-1}{2}\right)(1)^2\right)^{-1}$

$T_e = 2908.8^{O}C$

Using equation (1.3) to find the exit velocity,

$V_e = M\sqrt{\gamma R T_e}$

$V_e = 1\sqrt{1.2 * 0.287 * 3181.8}$

$V_e = 33.103$ m/s

Using equation (1.4) to find the mass flow rate,

$\dot{m} = \frac{AP_t}{\sqrt{T_t}} \sqrt{\frac{\gamma}{R}} M \left(1 + \frac{\gamma - 1}{2} M^2\right)^{-\frac{\gamma + 1}{2(\gamma - 1)}}$

$\dot{m} = \frac{1.72 * 20}{\sqrt{3200}} \sqrt{\frac{1.2}{.287}} 1 \left(1 + \frac{1.2-1}{2} 1^2\right)^{-\frac{1.2+1}{2(1.2-1)}}$

$\dot{m} = 0.736$ kg/s

Using equation (1.5) to find the Nozzle Exit thrust,

$F = \dot{m} V_e + (P_e - P_o)A_e$.

$F = 0.736 * 33.103 + (11.28 - 1.013) * 21.07$.

F= 216.32 MN.

At Mach 0.5:

Using equation (1.2) to find exit pressure,

$$\frac{P_e}{P_t} = (1 + \frac{\gamma-1}{2} M^2)^{-\frac{\gamma}{\gamma-1}}$$

$$P_e = 20 \left(\left(1 + \left(\frac{1.2-1}{2} \right) (0.5)^2 \right) \right)^{-\frac{1.2}{0.2}}$$

P_e= 17.246 MPa

Using equation (1.1) to find exit temperature,

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

$$T_e = 3200 \left(1 + \left(\frac{1.2-1}{2} \right) (0.5)^2 \right)^{-1}$$

T_e= 3123.2 ^{O}C

Using equation (1.3) to find the exit velocity,

$$V_e = M\sqrt{\gamma R T_e}$$

$$V_e = 0.5\sqrt{1.2 * 0.287 * 3181.8}$$

V_e=17.1 m/s

Using equation (1.4) to find the mass flow rate,

$$\dot{m} = \frac{AP_t}{\sqrt{T_t}} \sqrt{\frac{\gamma}{R}} M \left(1 + \frac{\gamma-1}{2} M^2 \right)^{-\frac{\gamma+1}{2(\gamma-1)}}$$

$$\dot{m} = \frac{1.72*20}{\sqrt{3200}} \sqrt{\frac{1.2}{.287}} 0.5 \left(1 + \frac{1.2-1}{2} 0.5^2 \right)^{-\frac{1.2+1}{2(1.2-1)}}$$

$\dot{m}$=0.542 kg/s

Using equation (1.5) to find the Nozzle Exit thrust,

$F = \dot{m}V_e + (P_e - P_o)A_e$.

F = 0.542 ∗ 17.1 + (17.246 – 1.013) ∗ 21.07.

F= 342.02 MN.

4.2.2 Four Inlet Nozzle

Taking Ideal conditions,

Pressure at the exit of one combustion chamber = 20 MPa

Pressure at exit of 4 combustion chambers = 4*20 = 80MPa

Ideal combustion chamber temperature is taken as 3200 OC

We consider,

Combustion Chambers Pressure (Pt) = 80 MPa

Combustion Chamber Temperature (Tt) = 3500K

Specific heat ratio (γ) = 1.2

Gas Constant (R) = 0.287 KJ/kg-K

To calculate the exit temperature, pressure and velocity,

Exit Temperature Ratio $\frac{T_e}{T_t} = \left(1 + \left(\frac{\gamma-1}{2}\right) M^2\right)^{-1}$ ……………………………….(1.1)

Exit Pressure Ratio $\frac{P_e}{P_t} = (1 + \frac{\gamma-1}{2} M^2)^{-\frac{\gamma}{\gamma-1}}$ ……………………………………(1.2)

Exit Velocity $V_e = M\sqrt{\gamma R T_e}$ ……………………………………………………..(1.3)

Mass Flow Rate $\dot{m} = \frac{AP_t}{\sqrt{T_t}} \sqrt{\frac{\gamma}{R}} M \left(1 + \frac{\gamma-1}{2} M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$ …………………………..…..(1.4)

Nozzle Exit Thrust $F = \dot{m} V_e + (P_e - P_o)A_e$ …………………………….……(1.5)

Where,

Mach Number - M, Exit Temperature – T_e, Exit Pressure – P_e, Area of cross section at the exit – A_e, Free Stream Pressure – P_o , Mass Flow Rate - $\dot{m}$, Area of Throat – A, Diameter of Throat = dt, Diameter at the exit of nozzle = d_e .

P_o=1.01 MPa

Area of throat $A=\frac{\pi}{4} d_e^2$

$A=1.72 \text{ m}^2$

The area at the exit of nozzle $A=\frac{\pi}{4} d_e^2$

$A=\frac{\pi}{4} 5.18^2$

$A_e=21.07 \text{ m}^2$

At Mach 2.5:

Using equation (1.2) to find exit pressure,

$\frac{P_e}{P_t} = (1 + \frac{\gamma-1}{2} M^2)^{-\frac{\gamma}{\gamma-1}}$

P_e=80 $\left(\left(1 + \left(\frac{1.2-1}{2}\right)(2.5)^2\right)\right)^{-\frac{1.2}{0.2}}$

P_e=4.3448 MPa

Using equation (1.1) to find exit temperature,

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

T_e=3200$\left(1 + \left(\frac{1.2-1}{2}\right)(2.5)^2\right)^{-1}$

T_e=1969.23 ^{O}C

Using equation (1.3) to find the exit velocity,

V_e=M$\sqrt{\gamma R T_e}$

V_e=2.5$\sqrt{1.2 * 0.287 * 1969.23}$

V_e= 65.1058 m/s

Using equation (1.4) to find the mass flow rate,

$\dot{m} = \frac{AP_t}{\sqrt{T_t}} \sqrt{\frac{\gamma}{R}} M \left(1 + \frac{\gamma-1}{2} M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$

$\dot{m} = \frac{1.72*80}{\sqrt{3200}} \sqrt{\frac{1.2}{.287}}\, 2.5 \left(1 + \frac{1.2-1}{2} 2.5^2\right)^{-\frac{1.2+1}{2(1.2-1)}}$

$\dot{m}$=1.4 kg/s

Using equation (1.5) to find the Nozzle Exit thrust,

F = $\dot{m}$ V_e + (P_e – P_o)A_e.

F = 1.4 ∗ 61.539 + (8.935 − 1.013) ∗ 21.07.

F= 166.917 MN.

At Mach 1:

Using equation (1.2) to find exit pressure,

$\frac{P_e}{P_t} = (1 + \frac{\gamma-1}{2} M^2)^{-\frac{\gamma}{\gamma-1}}$

$P_e=80\left(\left(1+\left(\frac{1.2-1}{2}\right)(1)^2\right)\right)^{-\frac{1.2}{0.2}}$

P_e= 45.158 MPa

Using equation (1.1) to find exit temperature,

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

$T_e=3200\left(1+\left(\frac{1.2-1}{2}\right)(1)^2\right)^{-1}$

T_e= 2908.8^{O}C

Using equation (1.3) to find the exit velocity,

$V_e=M\sqrt{\gamma R T_e}$

$V_e=1\sqrt{1.2 * 0.287 * 3181.8}$

V_e=33.103 m/s

Using equation (1.4) to find the mass flow rate,

$\dot{m} =\frac{AP_t}{\sqrt{T_t}}\sqrt{\frac{\gamma}{R}}M\left(1+\frac{\gamma-1}{2}M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$

$\dot{m} =\frac{1.72*80}{\sqrt{3200}}\sqrt{\frac{1.2}{.287}}1\left(1+\frac{1.2-1}{2}1^2\right)^{-\frac{1.2+1}{2(1.2-1)}}$

$\dot{m}$=6.184 kg/s

Using equation (1.5) to find the Nozzle Exit thrust,

$F = \dot{m} V_e + (P_e – P_o) A_e$.

F = 6.184 ∗ 33.103 + (45.158 – 1.013) ∗ 21.07.

F= 930.135 MN.

At Mach 0.5:

Using equation (1.2) to find exit pressure,

$\frac{P_e}{P_t} = (1 + \frac{\gamma - 1}{2} M^2)^{-\frac{\gamma}{\gamma-1}}$

$P_e = 80\left(\left(1 + \left(\frac{1.2-1}{2}\right)(0.5)^2\right)\right)^{-\frac{1.2}{0.2}}$

P_e= 68.984 MPa

Using equation (1.1) to find exit temperature,

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

$T_e = 3200\left(1 + \left(\frac{1.2-1}{2}\right)(0.5)^2\right)^{-1}$

T_e= 3123.2 ^{O}C

Using equation (1.3) to find the exit velocity,

$V_e = M\sqrt{\gamma R T_e}$

$V_e = 0.5\sqrt{1.2 * 0.287 * 3181.8}$

V_e=17.1 m/s

Using equation (1.4) to find the mass flow rate,

$\dot{m} = \frac{AP_t}{\sqrt{T_t}}\sqrt{\frac{\gamma}{R}}M\left(1 + \frac{\gamma-1}{2}M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$

$\dot{m} = \frac{1.72*80}{\sqrt{3200}}\sqrt{\frac{1.2}{.287}}\,0.5\left(1 + \frac{1.2-1}{2}0.5^2\right)^{-\frac{1.2+1}{2(1.2-1)}}$

$\dot{m}$=9.12 kg/s

Using equation (1.5) to find the Nozzle Exit thrust,

$F = \dot{m}V_e + (P_e - P_O)A_e$.

F = 9.12 ∗ 17.1 + (68.984 – 1.013) ∗ 21.07.

F= 1432.149 MN.

Chapter 5
ANALYSIS OF THE ROCKET NOZZLES

The analysis is carried out mainly using ANSYS Software. It allows transferring of the model from CAD to ANSYS.

5.1 Single Inlet Nozzle Analysis

This section deals with the procedure of analysis of the single inlet rocket nozzle giving output for temperature and pressure.

5.1.1 Modelling of Single Inlet Nozzle

The geometry of the nozzle is created using NX CAD and then transferred to ANSYS.

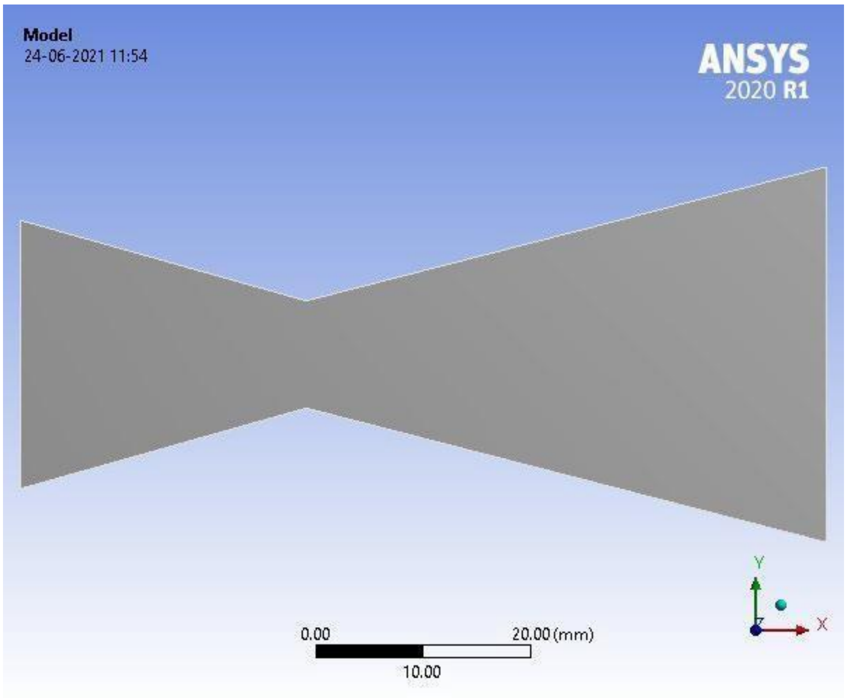


Fig 5.1 Model of single inlet rocket nozzle

2-Dimensional modelling of the nozzle is done using ANSYS. It is then converted to a surface model and split into different sections. The dimensions of the Nozzle are stated below.

Table 5.1 Dimensions of the Rocket Nozzle

Inlet Diameter	25mm
Exit Diameter	35mm
Throat	10mm
Length of the nozzle	75mm

5.1.2 Meshing of the Single Inlet Nozzle

The meshing is done using the ANSYS Software. The Automatic Mesh function is used forMeshing for the Single Inlet Nozzle.

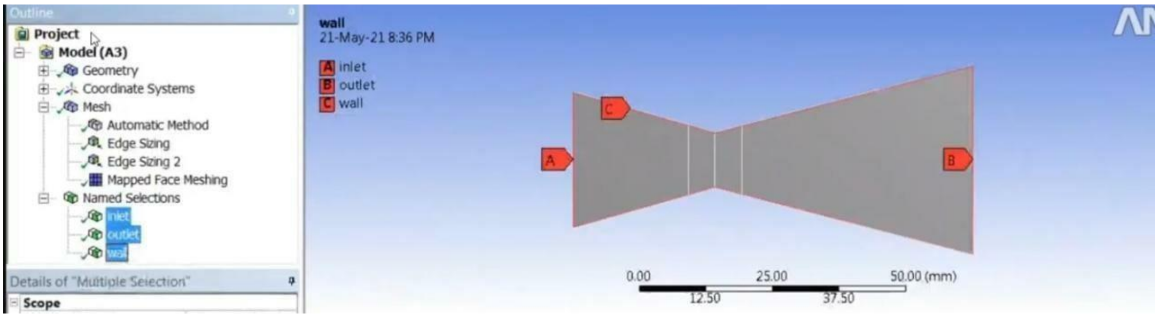


Fig 5.2 Meshing of various regions of the nozzle

Domain	Nodes	Elements	Wedges	Hexahedra
Single Inlet Nozzle	131	105	3	102

Table 5.2 Mesh details of the single inlet nozzle

The mesh is split over three sections of the nozzle, inlet, wall and, outlet.

The mapped face meshing is used to have fine divisions over the throat of the nozzle.

5.1.3 Analysis of the Single Inlet Nozzle

The meshed nozzle is then used for analysis to find the exit temperatures and pressures under the effect of different Mach Numbers. The parameters setup is discussed in the table below.

Table 5.3 Problem Setup for Single Inlet Nozzle

General	Type: Density-based Velocity: Absolute Time: Steady 2D space: Planar
Models	Energy: On Viscous: Inviscid
Materials	Fluid: Air Density: Ideal Gas Viscosity: Sutherland
Boundary Conditions	Inlet: Pressure Inlet Outlet: Pressure Outlet Gauge Pressure: 0 Type: Symmetry
Initialization	Standard Initialization Compute from Inlet

5.1.4 Solutions for Single Inlet Nozzle Analysis

Considering the single inlet rocket nozzle functions under the ideal conditions and the flow is Laminar, the following analysis has been done.

5.1.4.1 At Mach 0.6

5.1.4.1.1 Pressure

Pressure The Combustion chamber pressure is taken to be between 1-20 MPa, with a gas constant of 0.287 KJ/Kg-K. The output of the analysis is shown in the figure below.

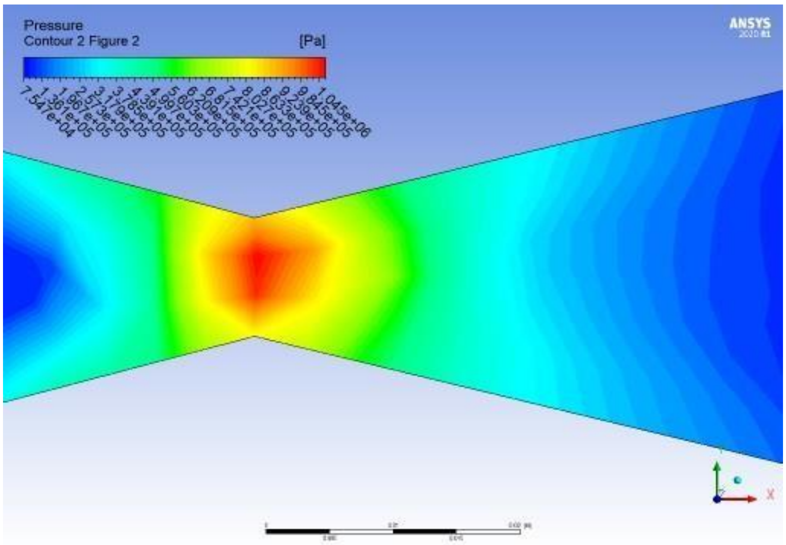


Fig 5.3 Pressure variation at Mach 0.6 for single inlet nozzle

The minimum pressure of 7.547 e^04 Pa and the maximum pressure of 1.045 e^06 Pa isexperienced in the single inlet rocket nozzle at Mach 0.6.

5.1.1.1.1 Temperature

The Ideal temperature for a combustion chamber is taken as 3200K and the specific heatratio as 1.2. The fluid considered is Air. The output of the analysis is shown below.

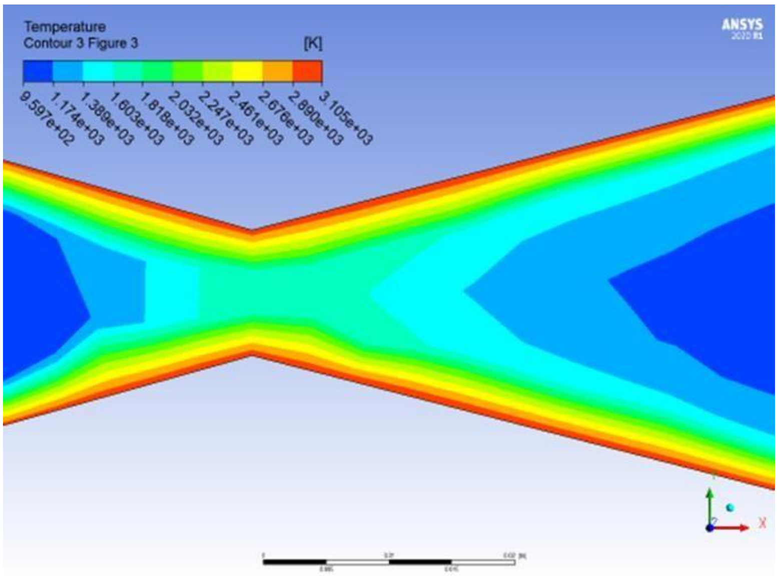


Fig 5.4 Temperature variation at Mach 0.6 for single inlet nozzle

The minimum temperature of 9.597 e^02 K and the maximum temperature of 3.105e^03 Kis experienced in the single inlet rocket nozzle at Mach 0.6.

5.1.1.1.2 Velocity

The variation of velocity at Mach 0.6 for a single inlet nozzle is shown.

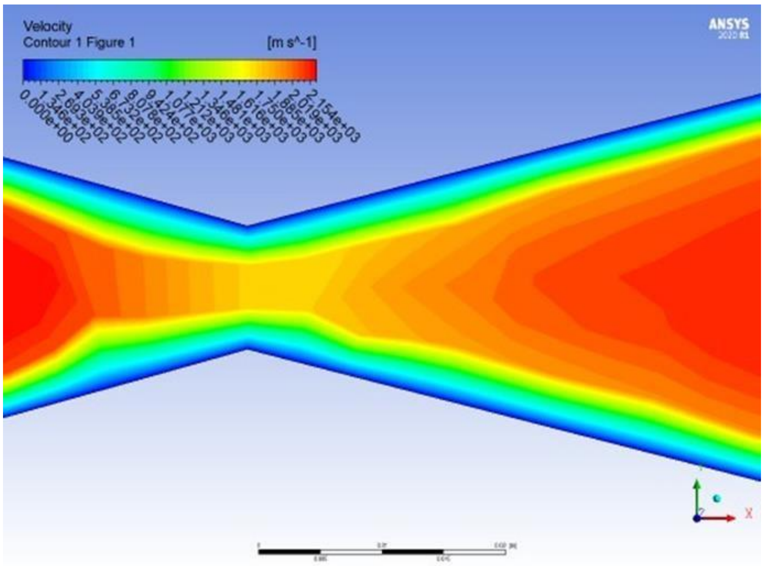


Fig 5.5 Velocity variation at Mach 0.6 for single inlet nozzle

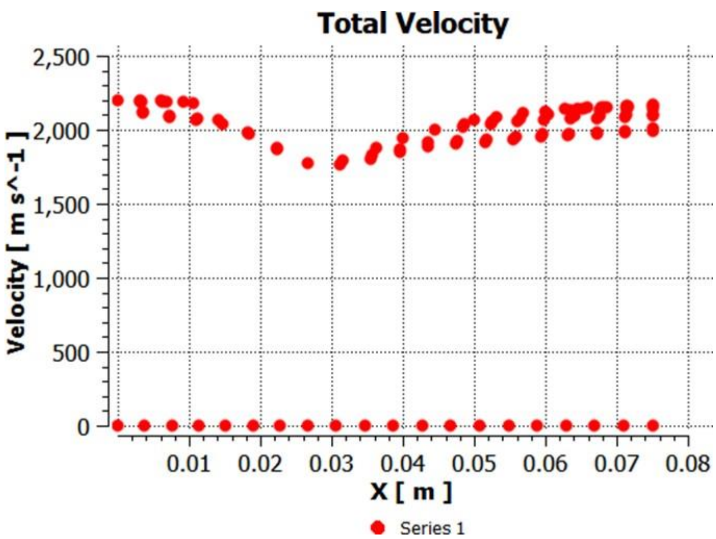


Fig 5.6 Total velocity variation along X-direction at Mach 0.6 for single inlet nozzle

The minimum velocity 0 m/s and the maximum velocity of 2.154 e^03 m/s is experiencedin the single inlet rocket nozzle at Mach 0.6.

5.1.1.2 At Mach 1

5.1.1.2.1 Pressure

The Combustion chamber pressure is taken to be between 1-20 MPa, with a gas constantof 0.287 KJ/Kg-K. The output of the analysis is shown in the figure below.

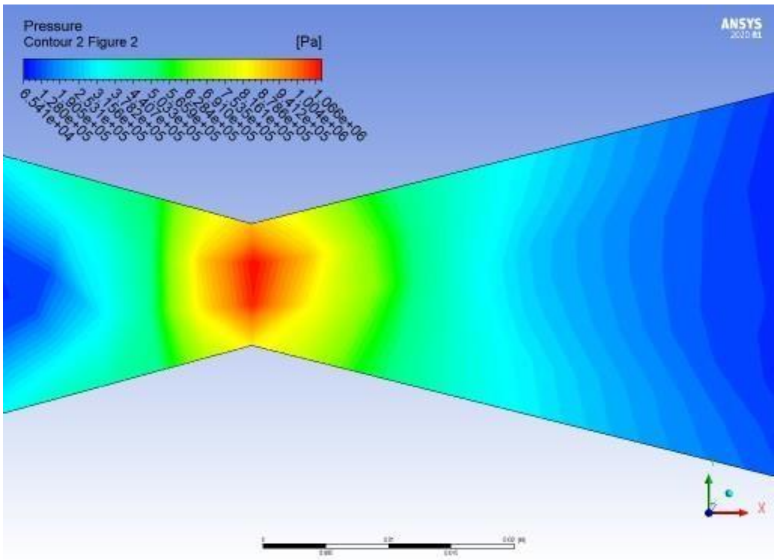


Fig 5.7 Pressure variation at Mach 1 for single inlet nozzle

The minimum pressure of 6.541 e^04 Pa and the maximum pressure of 1.066 e^06 Pa isexperienced in the single inlet rocket nozzle at Mach 1.

5.1.1.2.2 Temperature

The Ideal temperature for a combustion chamber is taken as 3200K and the specific heatratio as 1.2. The fluid considered is Air. The output of the analysis is shown below.

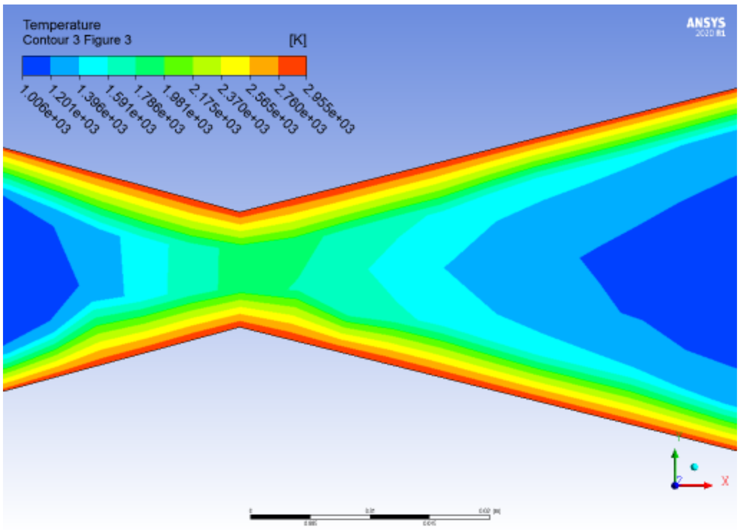


Fig 5.8 Temperature variation at Mach 1 for single inlet nozzle

The minimum temperature of 1.006 e^03 K and the maximum temperature 2.955 e^03 Kis experienced in the single inlet rocket nozzle at Mach 1.

5.1.1.2.3 Velocity

The variation of velocity at Mach 1 for a Single Inlet Nozzle is shown.

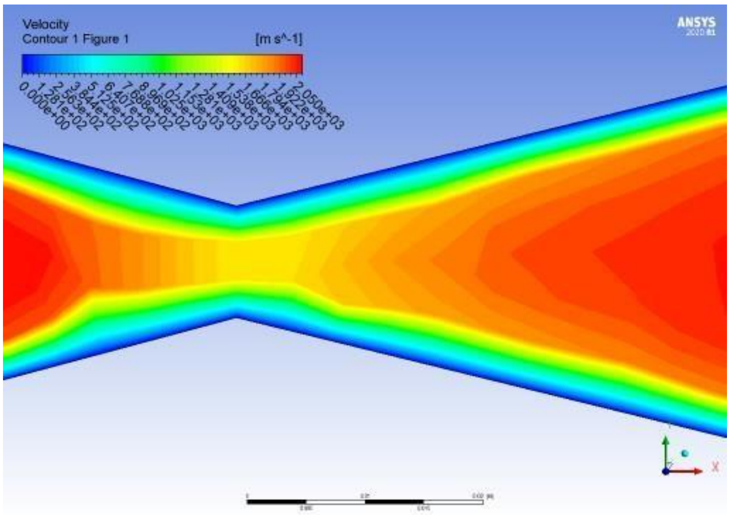


Fig 5.9 Velocity variation at Mach 1 for single inlet nozzle

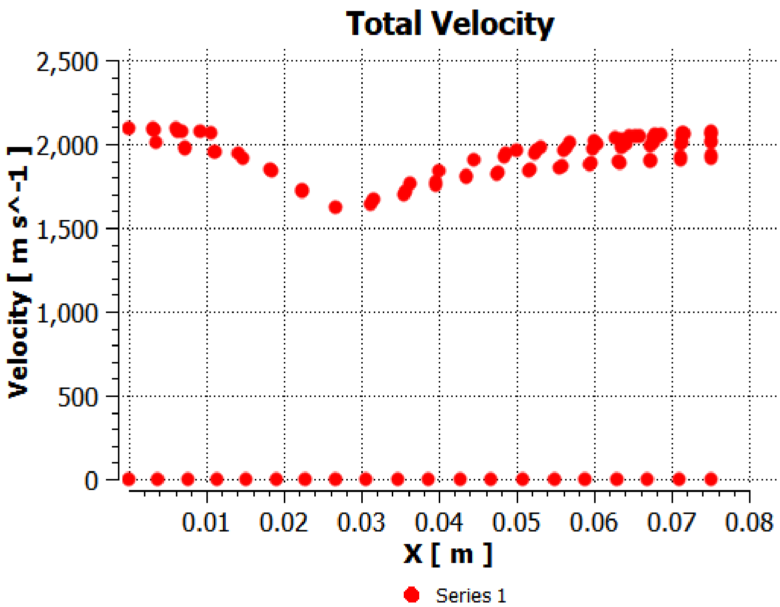


Fig 5.10 Total velocity variation along X-direction at Mach 1 for single inlet nozzle

The minimum velocity 0 m/s and the maximum velocity of 2.05 e^03 m/s is experiencedin the single inlet rocket nozzle at Mach 1.

5.1.1.3 At Mach 2.5

5.1.1.3.1 Pressure

The Combustion chamber pressure is taken to be between 1-20 MPa, with a gas constantof 0.287 KJ/Kg-K. The output of the analysis is shown in the figure below.

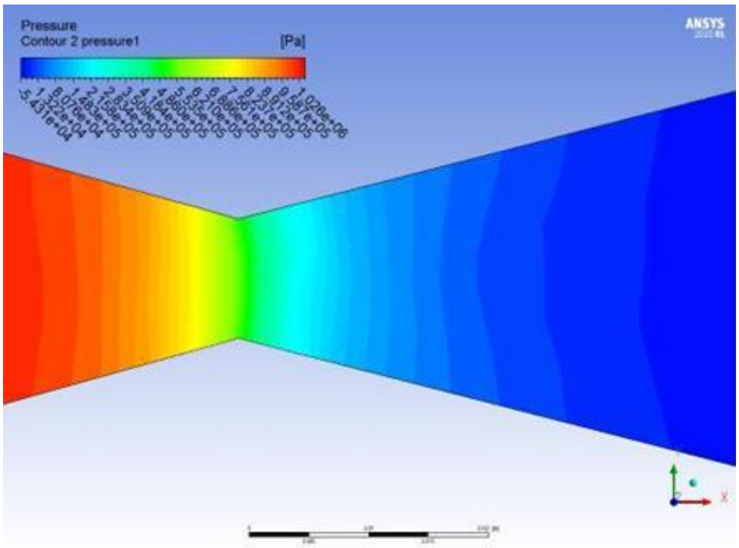


Fig 5.11 Pressure variation at Mach 2.5 for single inlet nozzle

The minimum pressure -5.431 e^04 Pa and the maximum pressure 1.026 e^06 Pa isexperienced in the single inlet rocket nozzle at Mach 2.5.

5.1.1.3.2 Temperature

The Ideal temperature for a combustion chamber is taken as 3200K and the specific heatratio as 1.2. The fluid considered is Air. The output of the analysis is shown below.

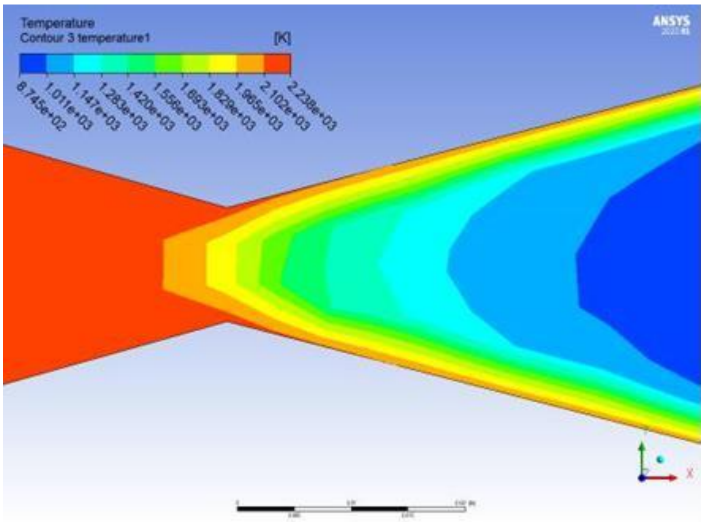
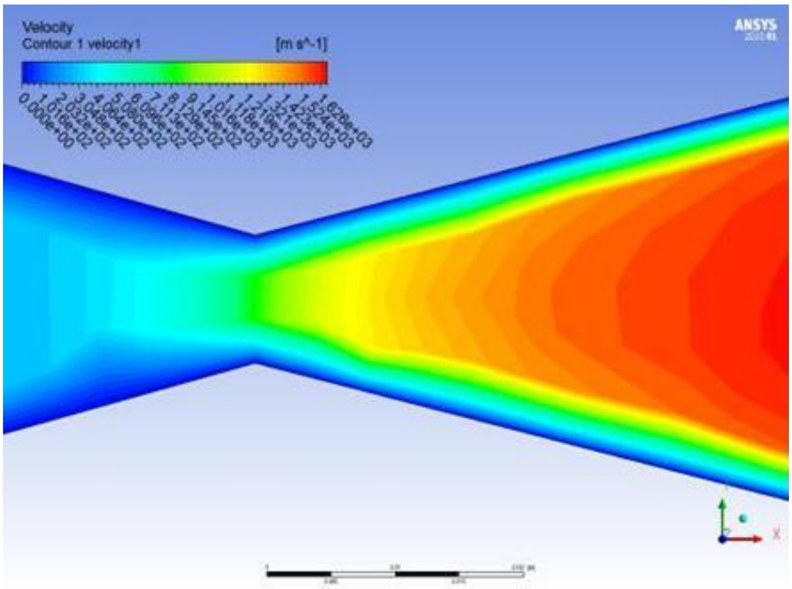


Fig 5.12 Temperature variation at Mach 2.5 for single inlet nozzle

The minimum temperature of 8.745 e^02 K and the maximum temperature 2.238 e^03 Kis experienced in the single inlet rocket nozzle at Mach 2.5.

5.1.1.3.3 Velocity

The variation of velocity at Mach 2.5 for a single inlet nozzle is shown.



single inlet rocket nozzle

Fig 5.13 Velocity variation at Mach 2.5 for single inlet rocket nozzle

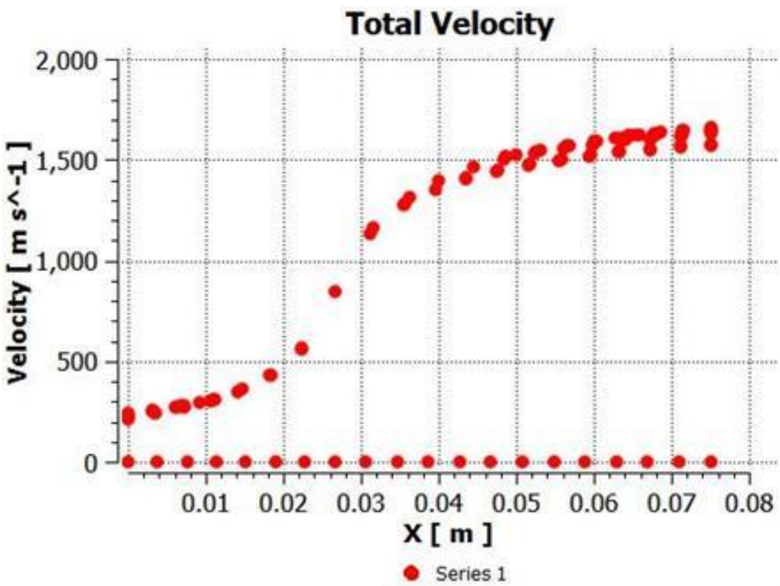


Fig 5.14 Total velocity variation along X-direction at Mach 2.5 for single inletnozzle

The minimum velocity of 0 m/s and the maximum velocity of 1.626 e^03 m/s is experienced in the single inlet rocket nozzle at Mach 2.5.

5.2 Four Inlet Nozzle Analysis

This section deals with the procedure of analysis of the four-inlet rocket nozzle givingoutput for temperature and pressure.

5.2.1 Modelling of Four Inlet Nozzle

The geometry of the nozzle is created using NX CAD and then transferred to ANSYS.

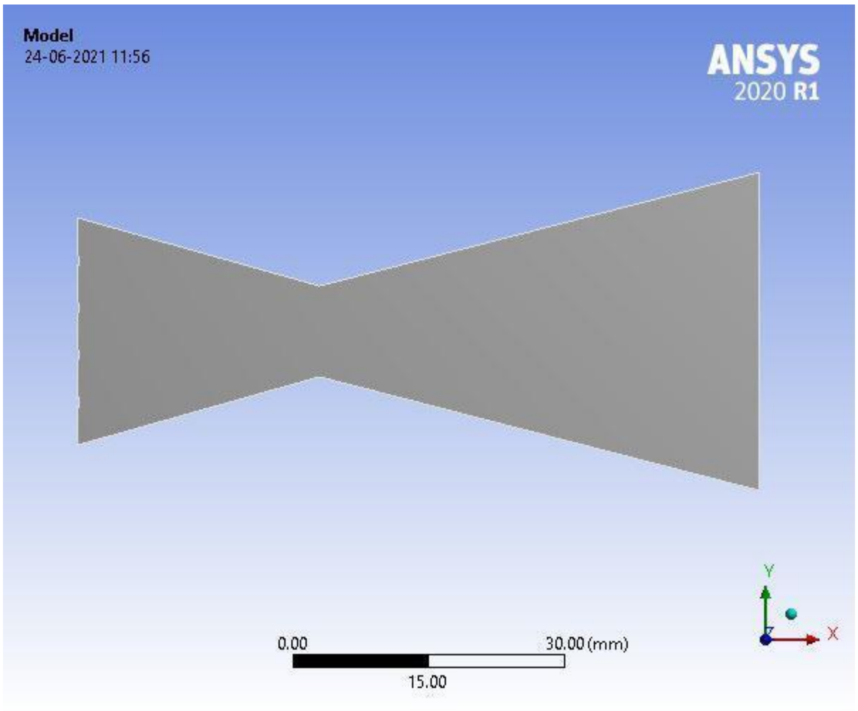


Fig 5.13 Model of four inlet rocket nozzle

2-Dimensional modeling of the nozzle is done using ANSYS. It is then converted to a surface model and split into different sections. The dimensions of the Nozzle are stated below.

Table 5.4 Dimensions of the rocket nozzle

Inlet Diameter	25mm
Exit Diameter	35mm
Throat	10mm
Length of the nozzle	75mm

5.2.2 Meshing of the Four Inlet Nozzle

The meshing is done using the ANSYS Software.

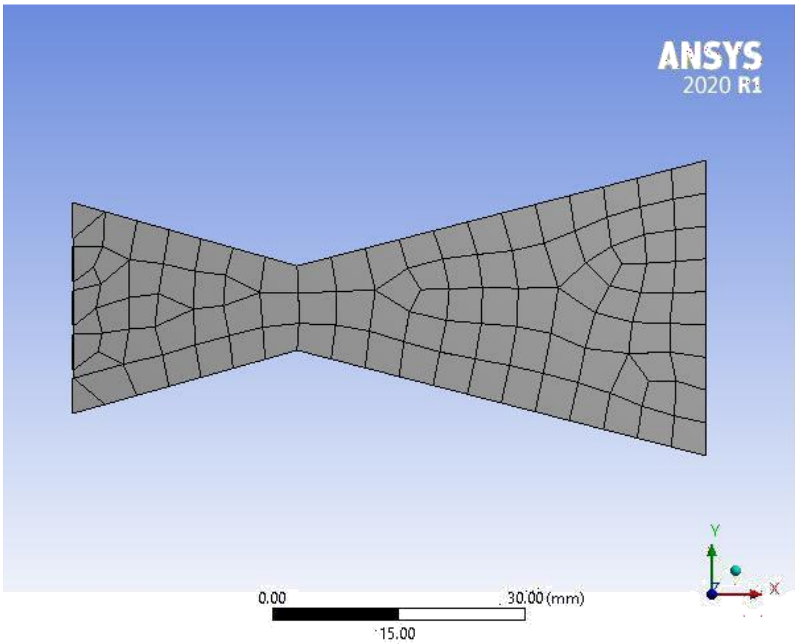
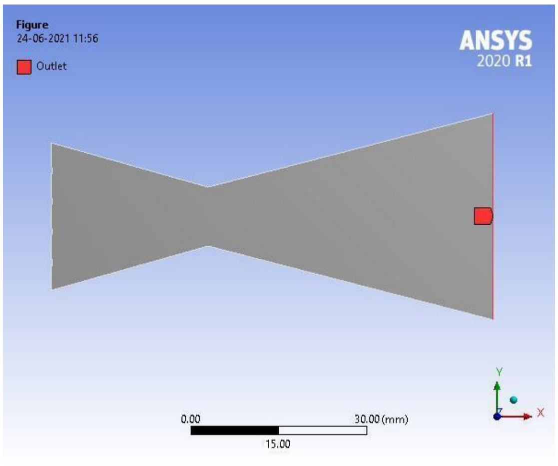


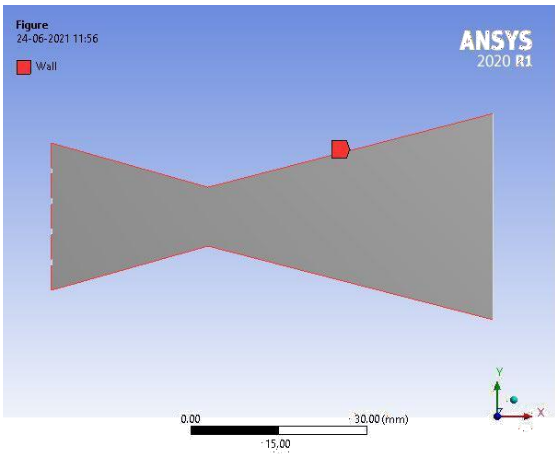
Fig 5.14 Meshing of four inlet rocket nozzle

Table 5.5 Mesh details of the four inlet rocket nozzle

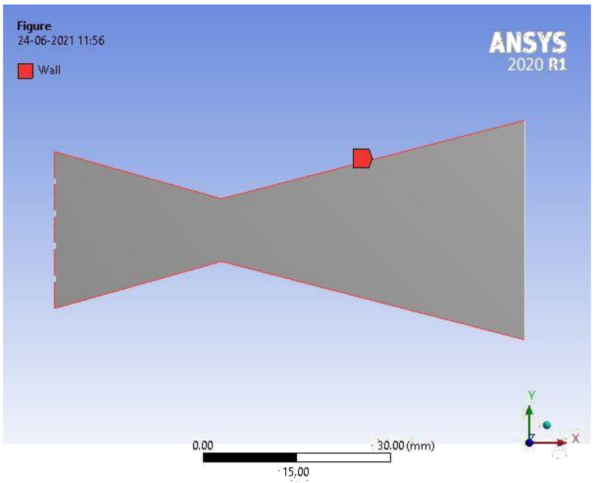
Domain	Nodes	Elements
Single Inlet Nozzle	138	106

The Automatic Meshing is used to perform meshing for the four-inlet nozzle. The mappedface meshing is used to have fine divisions over the throat of the nozzle. The four-inlet nozzle is divided into sections during meshing, outlet, wall and inlet.





(a) (b)



(c)

Fig 5.15 Meshing of four inlet rocket nozzle

5.2.3 Analysis of the Four Inlet Nozzle

The meshed nozzle is then used for analysis to find the exit temperatures and pressures under the effect of different Mach Numbers. The parameters setup is discussed in the tablebelow.

Table 5.6 Problem Setup for four inlet nozzle

General	Type: Density-based Velocity: Absolute Time: Steady 2D space: Planar
Models	Energy: On Viscous: Inviscid
Materials	Fluid: Air Density: Ideal Gas Viscosity: Sutherland
Boundary Conditions	Inlet: Pressure Inlet Outlet: Pressure Outlet Gauge Pressure: 0 Type: Symmetry
Initialization	Standard Initialization Compute from Inlet

5.2.4 Solutions for Four Inlet Nozzle Analysis

Considering that the four inlet rocket nozzle functions under the ideal conditions and the flow are Inviscid, the following analysis has been done.

The pressure for a Four Inlet Rocket Nozzle is four folds that of a Single Inlet Nozzle.Hence, the combustion chamber pressure is between 4-80 MPa.

5.2.4.1 At Mach 0.6

5.2.4.1.1 Pressure

The Combustion chamber pressure is taken to be 80 MPa, with a gas constant of 0.287KJ/Kg-K. The output of the analysis is shown in the figure below.

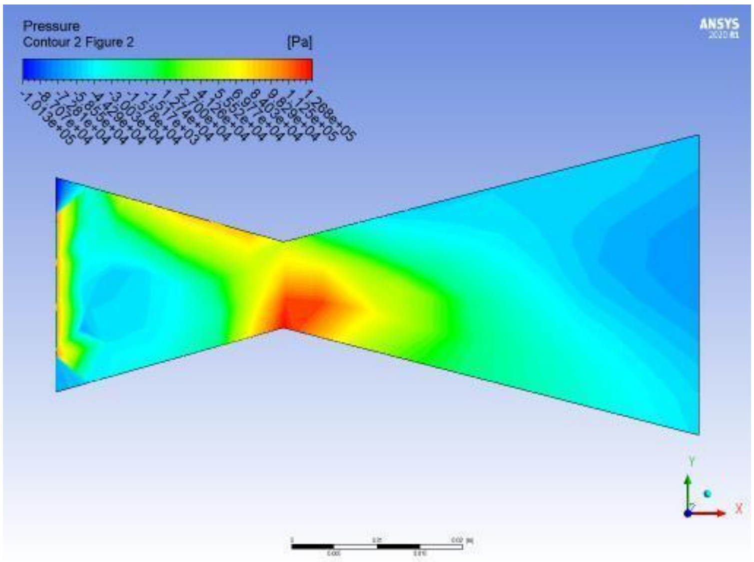


Fig 5.16 Pressure variation at Mach 0.6 for four inlet nozzle

The minimum pressure -1.013e^05 Pa and the maximum pressure 1.268 e^05 Pa isexperienced in the four inlet rocket nozzle at Mach 0.6.

5.2.4.1.2 Temperature

The Ideal temperature for a combustion chamber is taken as 3500K and the specific heatratio as 1.2. The fluid considered is Air. The output of the analysis is shown below.

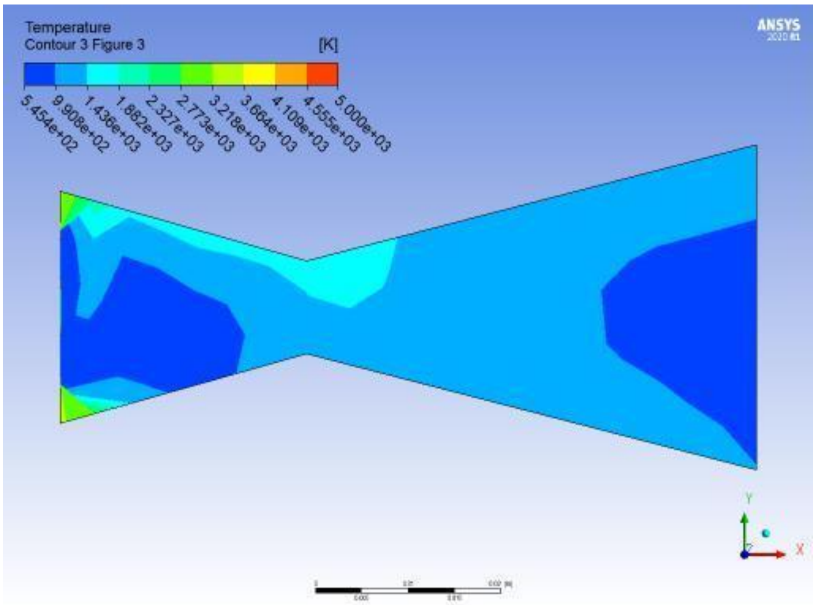


Fig 5.17 Temperature variation at Mach 0.6 for four inlet nozzle

The minimum temperature of 5.454 e^02 K and the maximum temperature of 5.000e^03 K is experienced in the four inlet rocket nozzle at Mach 0.6.

5.2.4.1.3 Velocity

The variation of velocity at Mach 0.6 for a Four Inlet Nozzle is shown.

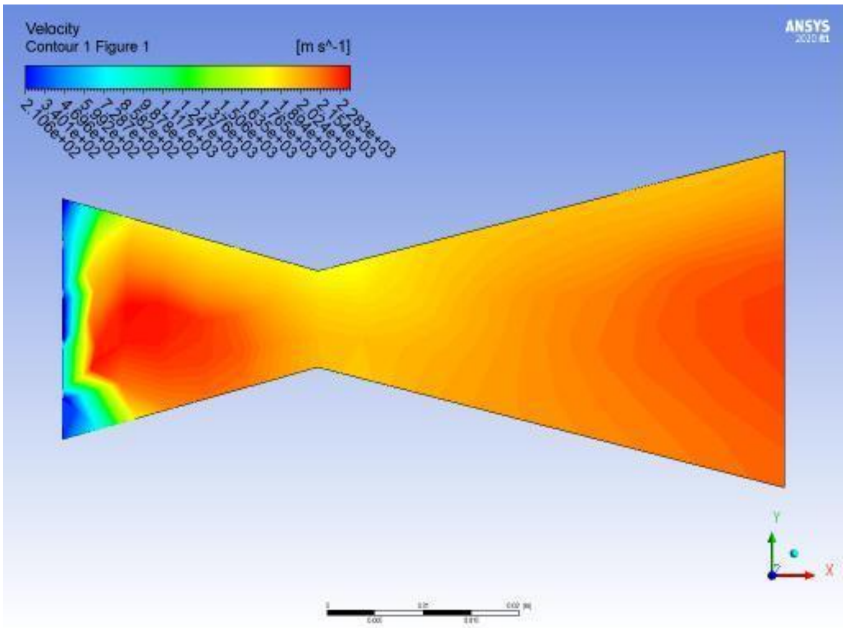


Fig 5.18 Velocity variation at Mach 0.6 for four inlet nozzle

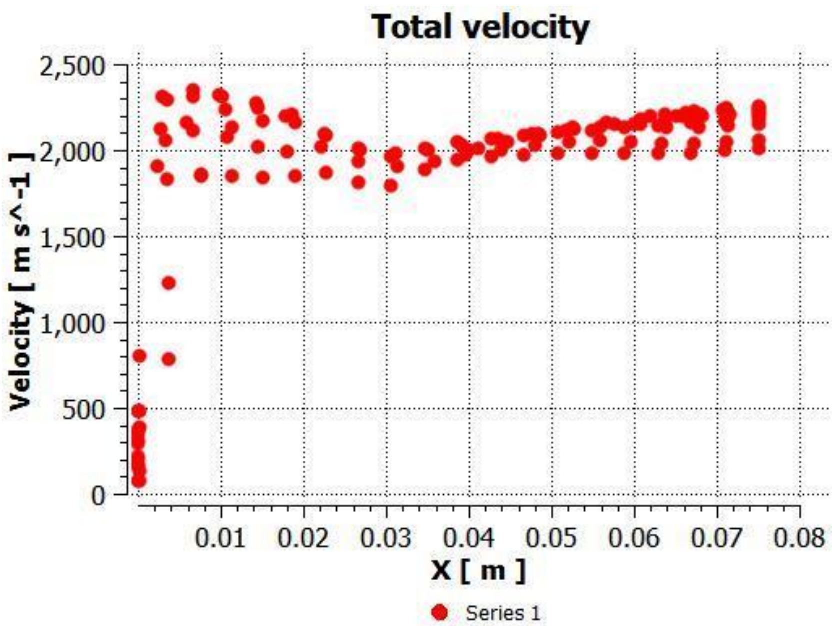


Fig 5.6 Total velocity variation along X-direction at Mach 0.6 for four inlet nozzle

The minimum velocity of 2.106 e^02 m/s and the maximum velocity of 2.283 e^03 m/s is experienced in the four inlet rocket nozzle at Mach 0.6.

5.2.4.2 At Mach 1

5.2.4.2.1 Pressure

The Combustion chamber pressure is taken to be 80 MPa, with a gas constant of 0.287KJ/Kg-K. The output of the analysis is shown in the figure below.

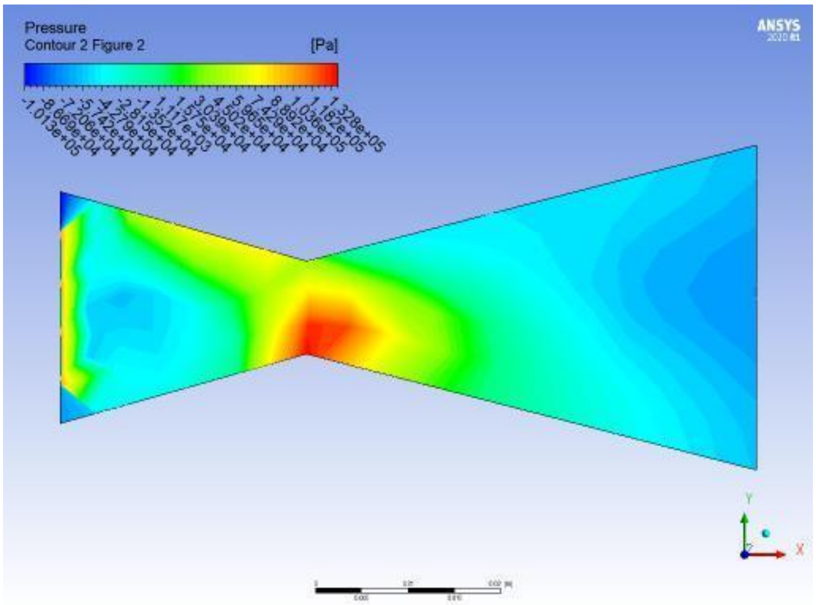


Fig 5.19 Pressure variation at Mach 1 for four inlet nozzle

The minimum pressure -1.013e^05 Pa and the maximum pressure 1.328 e^05 Pa isexperienced in the four inlet rocket nozzle at Mach 1.

5.2.4.2.2 Temperature

The Ideal temperature for a combustion chamber is taken as 3500K and the specific heatratio as 1.2. The fluid considered is Air. The output of the analysis is shownbelow.

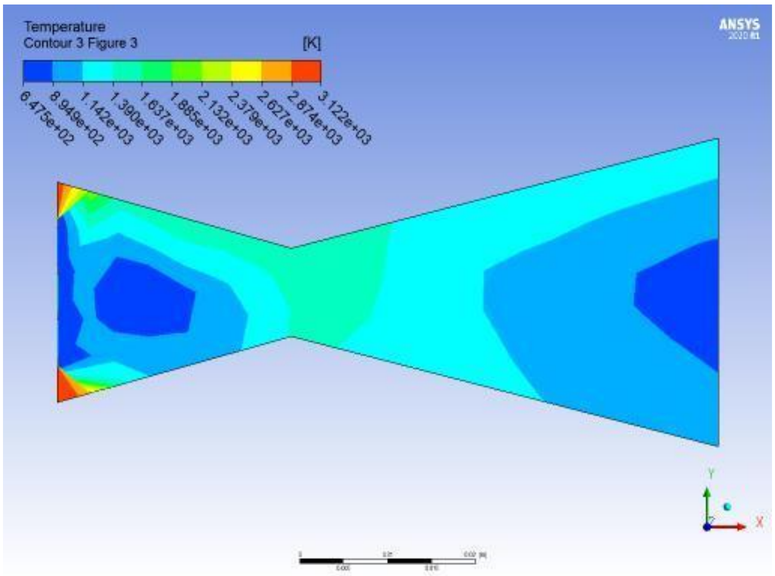


Fig 5.20 Temperature variation at Mach 1 for four inlet nozzle

The minimum temperature of 6.475 e^02 K and the maximum temperature of 3.122e^03 Kis experienced in the Four Inlet Rocket Nozzle at Mach 1.

5.2.4.2.3 Velocity

The variation of velocity at Mach 1 for a four inlet rocket nozzle is shown.

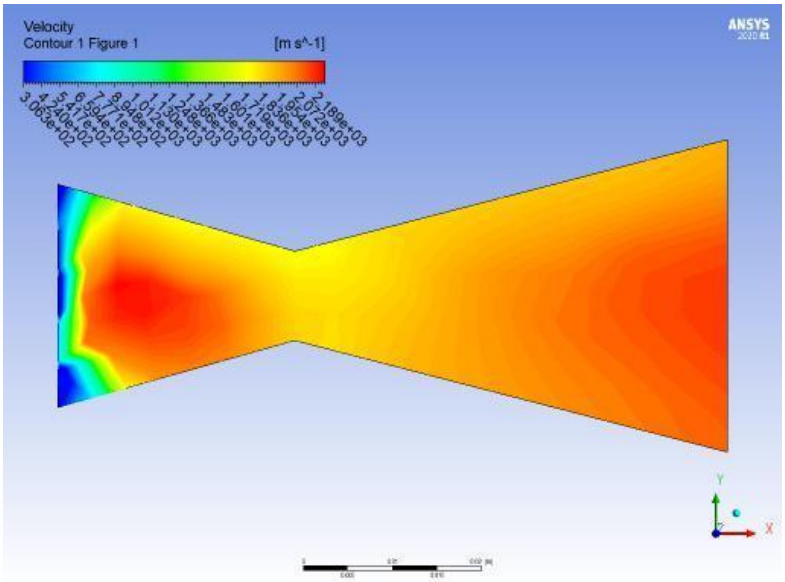


Fig 5.21 Velocity variation at Mach 1 for four inlet nozzle

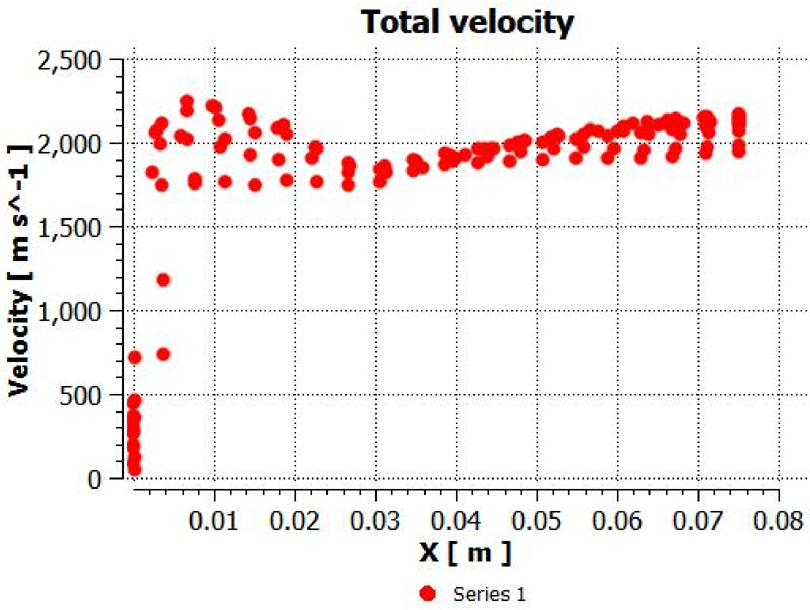


Fig 5.10 Total velocity variation along X-direction at Mach 1 for four inlet nozzle

The minimum velocity of 3.063 e^02 m/s and the maximum velocity of 2.189 e^03 m/s isexperienced in the four inlet rocket nozzle at Mach 1.

5.2.4.3 At Mach 2.5

5.2.4.3.1 Pressure

The Combustion chamber pressure is taken to be between 80 MPa, with a gas constant of 0.287 KJ/Kg-K. The output of the analysis is shown in the figure below.

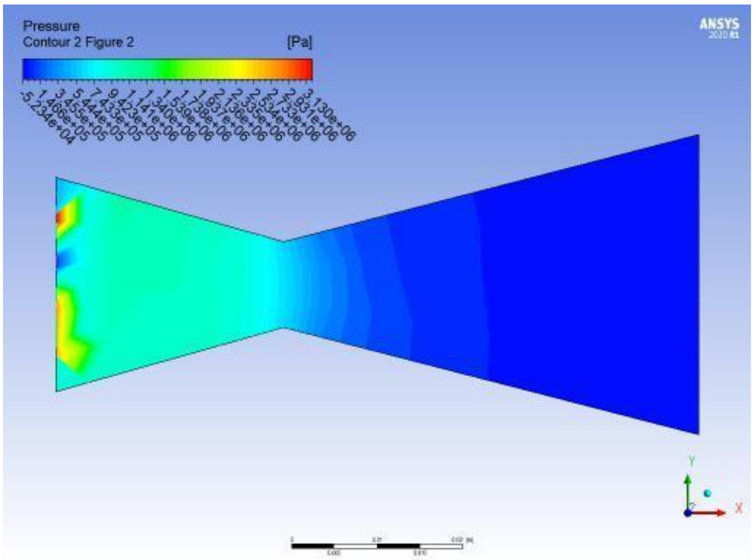


Fig 5.22 Pressure variation at Mach 2.5 for four inlet nozzle

The minimum pressure -5.234 e^04 Pa and the maximum pressure 3.130 e^06 Pa isexperienced in the Four Inlet Rocket Nozzle at Mach 2.5.

5.2.4.3.2 Temperature

The Ideal temperature for a combustion chamber is taken as 3500K and the specific heat ratio as 1.2. The fluid considered is Air. The output of the analysis is shown below.

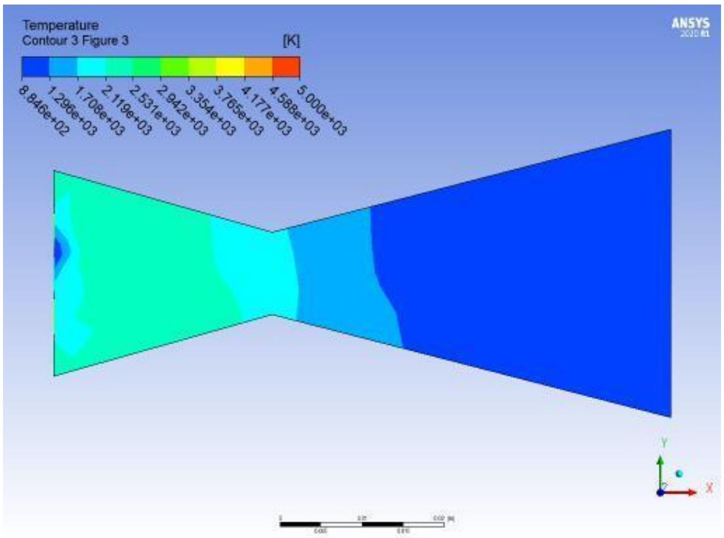


Fig 5.23 Temperature variation at Mach 2.5 for four inlet nozzle

The minimum temperature of 8.846 e^02 K and the maximum temperature of 5.00 e^03 Kis experienced in the Four Inlet Rocket Nozzle at Mach 2.5.

5.2.4.3.3 Velocity

The variation of velocity at Mach 2.5 for a Four Inlet Nozzle is shown.

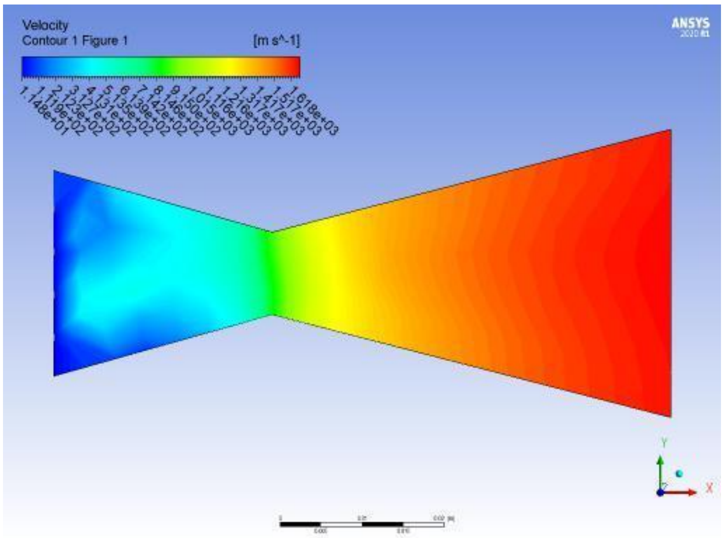


Fig 5.24 Velocity variation at Mach 2.5 for four inlet nozzle

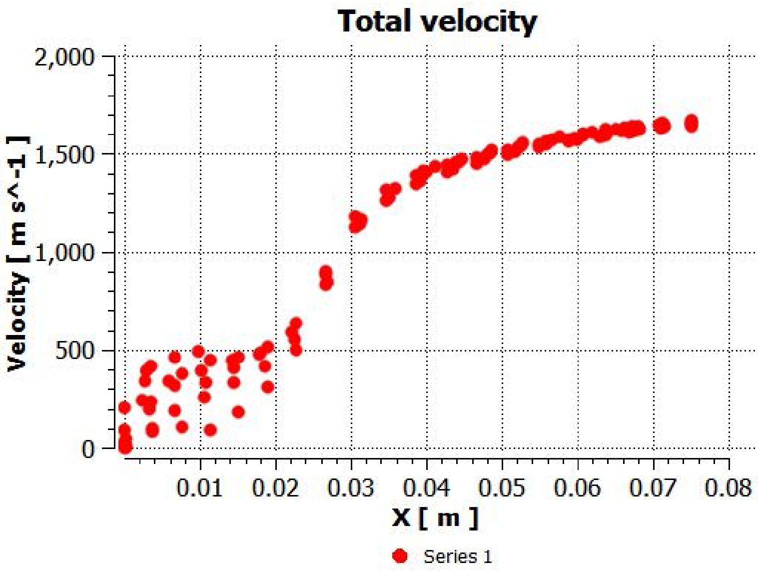


Fig 5.25 Total Velocity variation along X-direction at Mach 2.5 for four inlet nozzle

The minimum velocity is 1.148 e^01 m/s and the maximum velocity of 1.618 e^03 m/s isexperienced in the Four Inlet Rocket Nozzle at Mach 2.5.

5.3 Final Result

The results of the analysis carried on the Single Inlet Rocket Nozzle at various MachNumbers are listed:

Table 5.7 Final result of single inlet rocket nozzle analysis

TERMS	MACH 0.6	MACH 1	MACH 2.5
Minimum Pressure (Pa)	7.547 e^04	6.541 e^04	-5.431 e^04
Maximum Pressure (Pa)	1.045 e^06	1.066 e^06	1.026 e^06
Minimum Temperature (K)	9.597 e^02	1.006 e^03	8.745 e^02
Maximum Temperature (K)	3.105e^03	2.955 e^03	2.238 e^03
Minimum Velocity (m/s)	0	0	0
Maximum Velocity (m/s)	2.154 e^03	2.05 e^03	1.626 e^03

The results of the analysis carried out on the Four Inlet Rocket Nozzle for the different Mach Numbers are shown.

Table 5.8 Final result of single inlet rocket nozzle analysis

TERMS	MACH 0.6	MACH 1	MACH 2.5
Minimum Pressure (Pa)	-1.013e^05	-1.013e^05	-5.234 e^04
Maximum Pressure (Pa)	1.268 e^05	1.328 e^05	3.130 e^06
Minimum Temperature (K)	5.454 e^02	6.475 e^02	8.846 e^02
Maximum Temperature (K)	5.000e^03	3.122e^03	5.000 e^03
Minimum Velocity (m/s)	2.106 e^02	3.063 e^02	1.148 e^01
Maximum Velocity (m/s)	2.283 e^03	2.189 e^03	1.618 e^03

5.4 Conclusion

Nozzle models were developed considering the number of inlets (one & four) to evaluate the pressure, temperature and velocity. The parameters were compared theoretically and computationally with that of the numerical results found in the literature review. The results reveals that as the fuel and air entering in the combustion chamber according to the x and y plot, its burn due to high velocity and high temperatures. As temperature increases rapidly in the combustion chamber results in decrease in exit temperature at the convergent part of the nozzle.

In the real world, choosing the appropriate nozzle for rockets to contribute to the outer space missions has become necessary, not only to factor in safety and sustainability but also the costs. In this project work an attempt is made to compare the two types of nozzles, namely, Single Inlet Rocket Nozzle and Four Inlet Rocket Nozzle, theoretically and computationally.

On comparing the theoretical and computational analysis results of the Rocket Nozzles, concur that the output of Four Inlet Nozzles, for all the parameters, i.e., temperature, pressure, and velocity, are higher than that of the Single Inlet Rocket Nozzle. Maximum velocity output of the Four Inlet Nozzle is higher than that of the Single Inlet Rocket Nozzle, providing a higher take-off velocity, hence making it more preferable for use in Rockets. The Four Inlet Rocket Nozzle in the supersonic zones tend to provide the highest pressure, temperature and velocity outputs, giving desirable values for the use in Rockets. The results obtained in this study are closely validated with research study from the literature survey [**4**].

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