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STUDY OF NOZZLE FLOW SEPARATIONS IN DIVERGENT SECTION OF NOZZLE

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ABSTRACT

In this paper study of Nozzle flow Separation is carried out by simulation of Rocket Nozzle Design made through Autodesk Fusion 360 in ANSYS 2021R1 to investigate the Laminar Regime and Turbulent Regime in the divergent section of nozzle. Flow is analysed throughout the nozzle contour from convergent to divergent sections for its separation from nozzle walls due to boundary layer phenomena to visualize it for Laminar and Turbulent regimes. Nozzle design is made for design Mach No of 2.955 and the flow separation simulations are carried out to investigate the flow regimes.

Keywords: *Nozzle Flow Separation, Nozzle Pressure Ratios, Turbulent Regime, Laminar Regime, Axial Flow Direction.*

INTRODUCTION

The nozzle flow under certain circumstances generates critical side loads on the walls. This mainly occurs when the flow is separated due to an overexpansion and during the start-up transition processes. Different types of flow separation are possible with the different nozzle types as well within the same nozzle. At largely over-expanded regime boundary-layer is detached from the nozzle wall due to an adverse pressure gradient generating the separation shock. This supersonic and separated flow continues downstream and it may interact with the recompression shocks, asymmetric jet portions and/or possibly present internal shock, which can further lead to the severe lateral loads. These lateral or side-loads of uncontrolled flow separation are frequently present during the transient processes of engine start-up and shut-down operation. Under expanded nozzle produce more Isp than over expanded nozzles.

NOZZLE

A nozzle is a device designed with varying cross-sectional area especially in the form of a tube through which hot gas or liquid flows to generate thrust following Newton’s third law of motion. The nozzle is often used to control the characteristics of fluid flow, pressure, and the direction of flow, and to enhance the velocity of a gaseous substance exiting the nozzle to greater velocities. In the area of compressible flow, the nozzles are typically categorized as a convergent nozzle and a Convergent–divergent (CD) nozzle. Both types of nozzles have principled applications in aerospace industry and technology. A Convergent-Divergent nozzle is used to convert chemical energy into kinetic energy in a thermal chamber and vice versa. [11]

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Nozzle Flow Separation

In Rocket Engine Design and its optimum performance, it is important to have the knowledge of flow separation in rocket nozzles. Basically, separation of flow is studied under conditions of sea level. However, during the launcher or rocket ascent the change of ambient density is disregarded. In the design of altitude-adaptive dual bell nozzle the important factor of concern is the ambient flow properties. For this reason, within conventional convergent-divergent nozzles influence of ambient density on nozzle flow separation is to be studied. [1]

In a conventional convergent-divergent rocket nozzle the flow can withstand only a certain degree of overexpansion. Beyond this point the separation of boundary layer occurs, nozzle walls are lifted off and the ambient air is sucked into the remaining separated backflow section of the nozzle. For a given geometry of rocket nozzle, the flow separation position is a function of the total pressure, the ambient pressure, and the gas properties. This separation of flow leads to additional undesired loads on side walls of nozzle, stressing the nozzle, structure of launcher, the rocket engine, and the payload. The prediction of the position of flow separation is very important in rocket nozzle design and determines maximum possible area ratio of nozzle, which is engine performance deciding factor. [1]

On basic fluid dynamic phenomena flow separation in supersonic nozzles occurs at a certain pressure ratio of chamber to ambient pressure. This results in formation of shock and turbulent-boundary layer interaction inside the nozzle walls of divergent section. [2]

When the exit pressure to ambient pressure ratio of a nozzle is reduced to about 0.4–0.8, ambient air penetrates through the viscous layer. Thus, due to the adverse pressure gradient, separation of the boundary layer from the nozzle wall will get initiated, and this phenomenon is known as Nozzle flow separation. [3] Due to adverse pressure gradient, Separation of flow in nozzle occurs as the flow expands, making separated flows extended region. The part of the flow that separates the flow recirculating and the flow through the central region of the duct is called the dividing streamline. [4] Due to loss of energy in the flow separation in the boundary layer adjacent to nozzle wall initiates, an alternate method to prevent such separation of boundary layer is to reenergize the air by adding thin high-speed jet to it. [5]

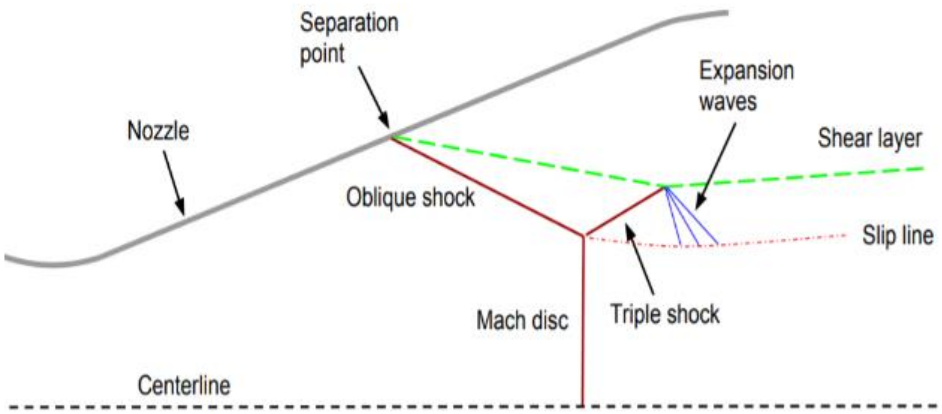

Fig 1.1 Shock Pattern of Supersonic Flow with Free Shock Separation inside an Axisymmetric Convergent- Divergent Nozzle [7]

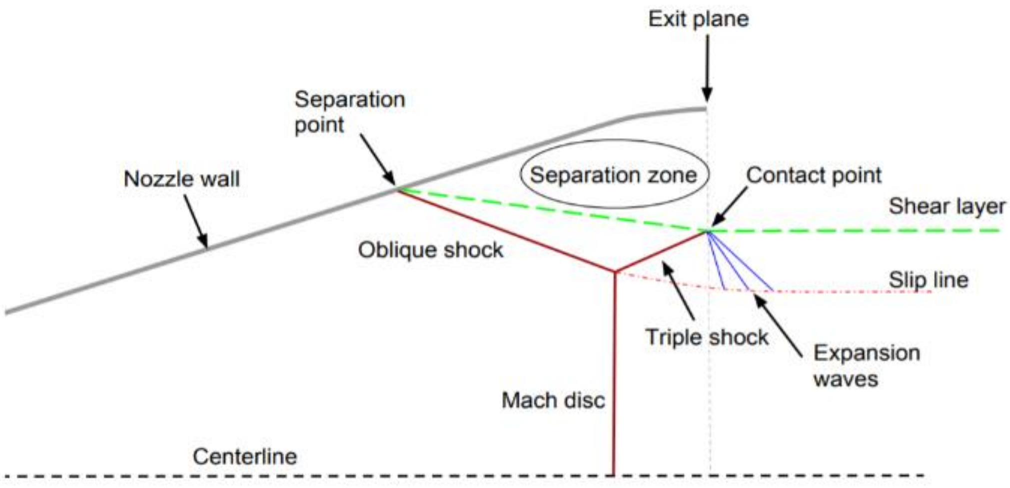

Fig 1.2 Schematic Diagram of Free shock separation [8]

Reflections inside different rocket nozzles with their generation of shock is studied and its impact of mainly two modes of separation, namely Restricted Shock Separation (RSS) and Free Shock Separation (FSS) is explored. [6]

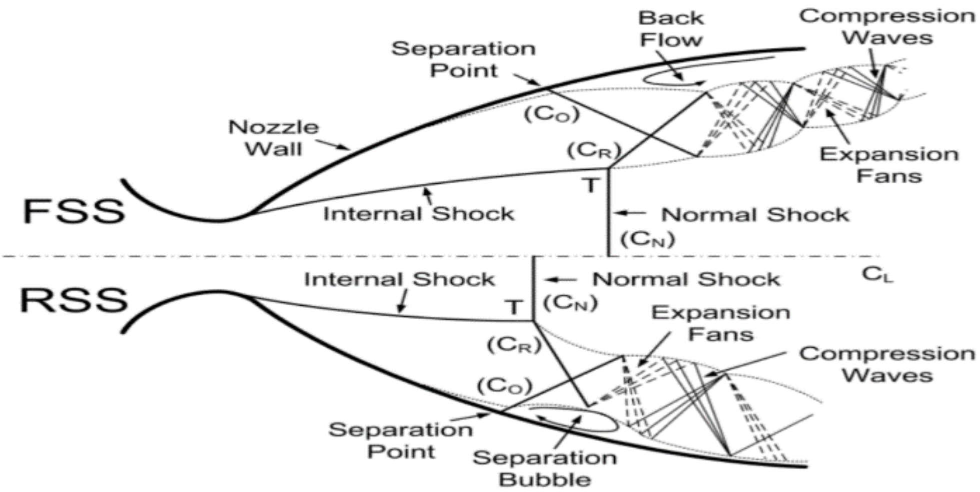

Fig 1.3 Phenomenological sketch of RSS and FSS [9]

Nozzle Design Parameters

Diameter of Convergence, Dc = 75 mm
Diameter of Throat, D t = 17.40 mm
Diameter of Exit, De = 49.21 mm
Length of Convergence, L c = 40.13 mm
Length of Divergence, L d = 78.96 mm
Total Length of Nozzle, L = 118.96 mm
Angle of Convergence = 28 o
Angle of Divergence = 15 0

Impulse, I = 3602 N-s
Thrust, T = 1669.6 N
Burn Time, t= 2.158 s
Gamma = 1.043
Nozzle Efficiency = 0.95
Exit Mach No, Me = 2.955
Throat Area, A t = 237.7 mm2
Exit Area, A e = 1902 mm2
Stagnation Temperature, To = 1625 K
A e/ At = 8

Po / P = 64.9254
To / T = 1.18774
A / A* = 12.16275
Velocity at Exit, V = 1940 m/s
Mass Flow Rate, m. = 9.5443 Kg/s

$$\frac{T_o}{T} = 1 + \frac{k-1}{2} M^2$$ [10]

$$\frac{P_o}{P} = \left(1 + \frac{k-1}{2} M^2\right)^{\frac{k}{k-1}}$$ [10]

$$\frac{A}{A^*} = \frac{1}{M}\left(\frac{1 + \frac{k-1}{2} M^2}{1 + \frac{k-1}{2}}\right)^{\frac{k+1}{2(k-1)}}$$ [10]

$$\dot{m} = p_o M A \sqrt{\frac{k}{RT_o}} \left(1 + \frac{k-1}{2} M^2\right)^{\frac{k+1}{-2(k-1)}}$$

[10]

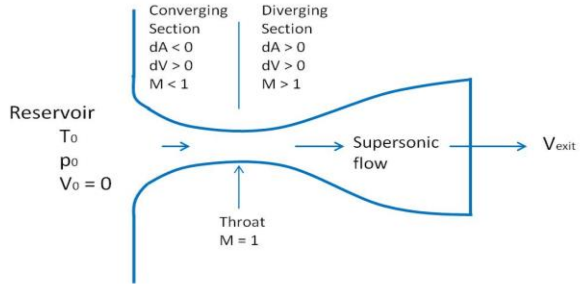

Fig 2.1 Subsonic to Supersonic Flow through Nozzle [12]

Nozzle Fusion 360 Designing Views

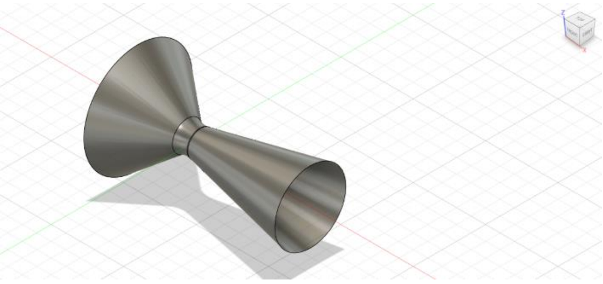

Fig 3.1 Front view of Nozzle Design

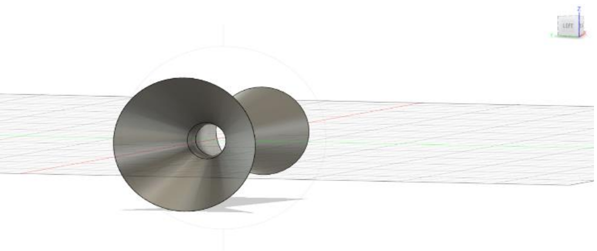

Fig 3.2 Isometric View of Nozzle Design

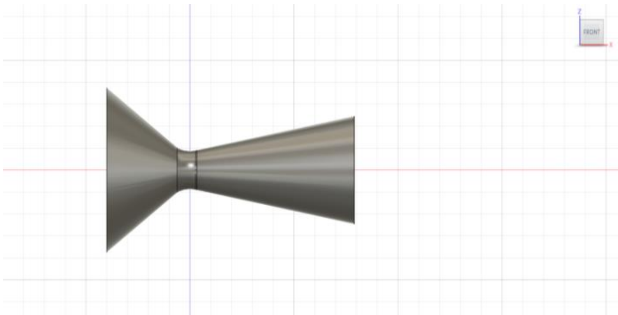

Fig 3.3 Side View of Nozzle Design

Nozzle Simulations Results

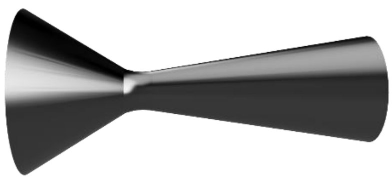

Fig 4.1 Nozzle Rendered Design

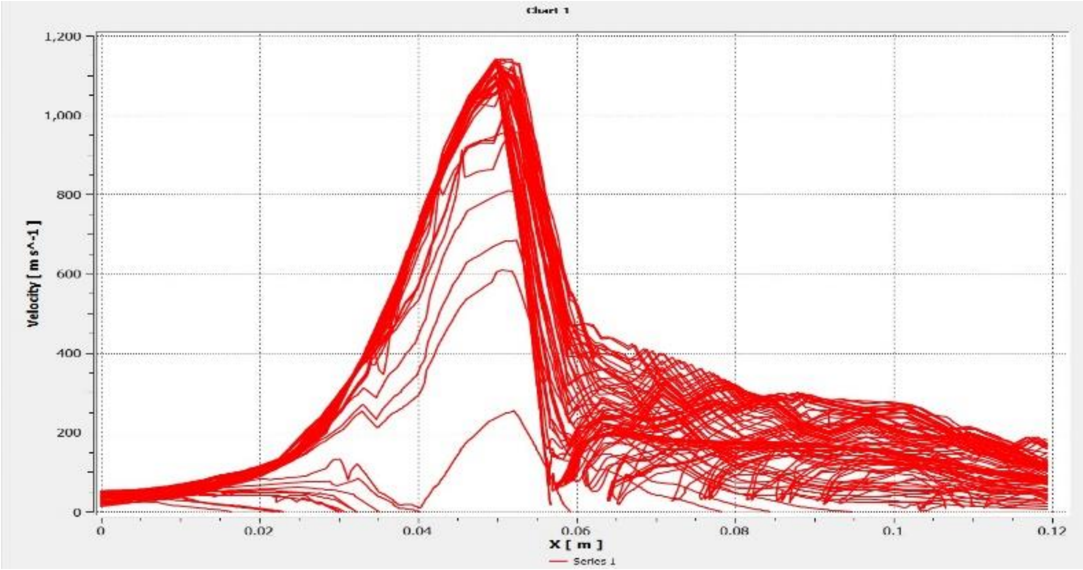

Fig 4.2 Graph of Velocity vs Axial Length of Nozzle

We can see that Velocity at Throat of Nozzle of increased between axial length of 0.4-0.6 m approaching to M=1 as design condition and we can see the Turbulent behaviour of Velocity at Divergent Section with curl of velocity flowing at rear end of nozzle.

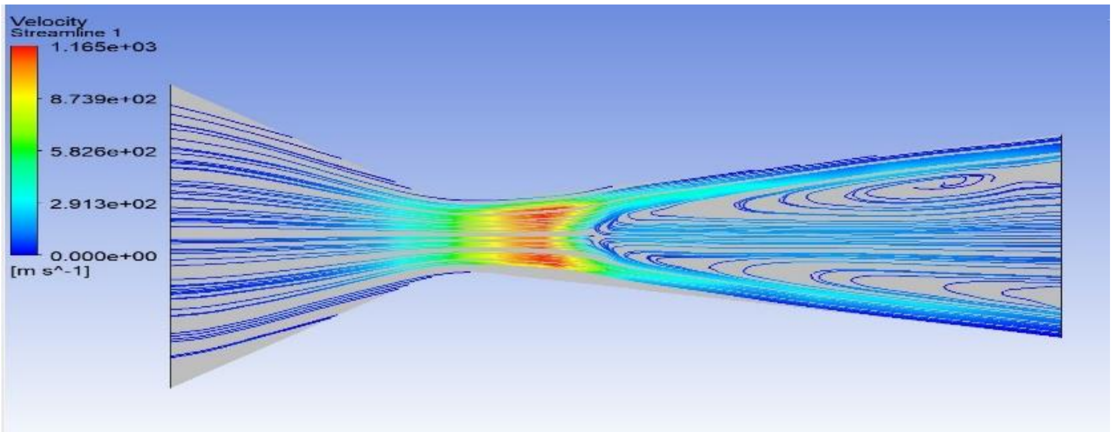

Fig 4.3 Simulation of Velocity flow across Nozzle Axial Length

We can see Velocity Streamlines across the axial direction of the nozzle. Here just after Nozzle Throat Velocity is approaching nearly 1200 m/s and we can visualize the flow turbulence due to Nozzle flow separation at nozzle exit in divergence portion. It can be seen that flow in mean half of nozzle is laminar and even at nozzle walls the flow is laminar but in between we have curl of flow due to which there is a reduction in flow exit velocity.

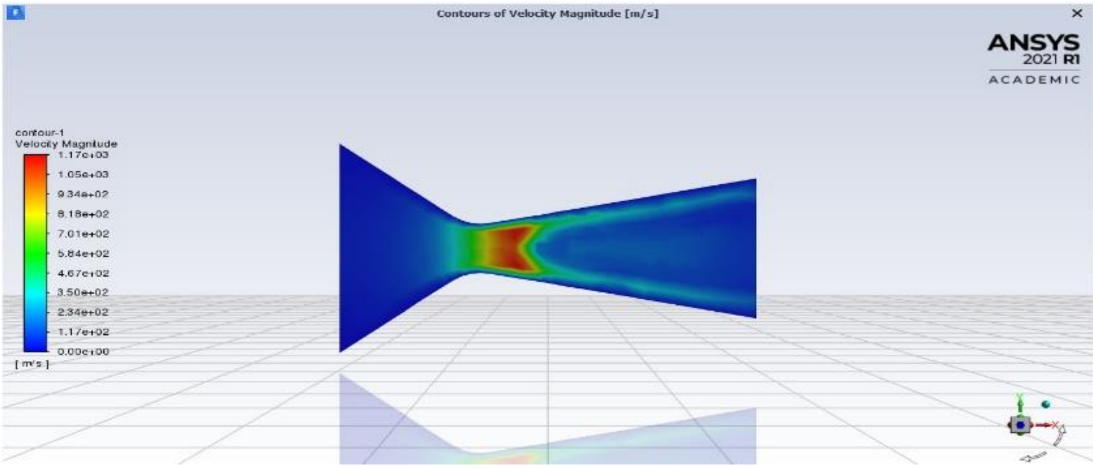

Fig 4.4 Simulation of Velocity Magnitude Contour

We can see the flow of velocity magnitude contour in the above figure. At Throat the magnitude of velocity is increased marking in Red Zone and at Convergent and Divergent portion of nozzle the magnitude is seen to be in laminar regime.

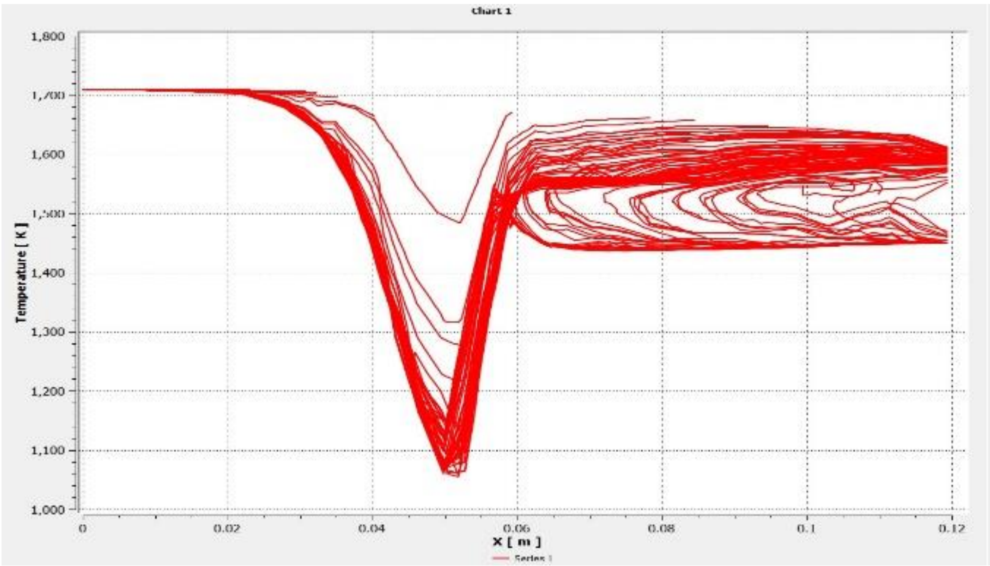

Fig 4.5 Temperature Chart across Nozzle Axial Direction

Plot of Temperature vs Axial Direction of Nozzle is depicted here. The plot is seen exact inverse mirror of velocity profile and therefore proves Basic thermodynamic relation of gases in flow. Temperature is directly proportional to Velocity.

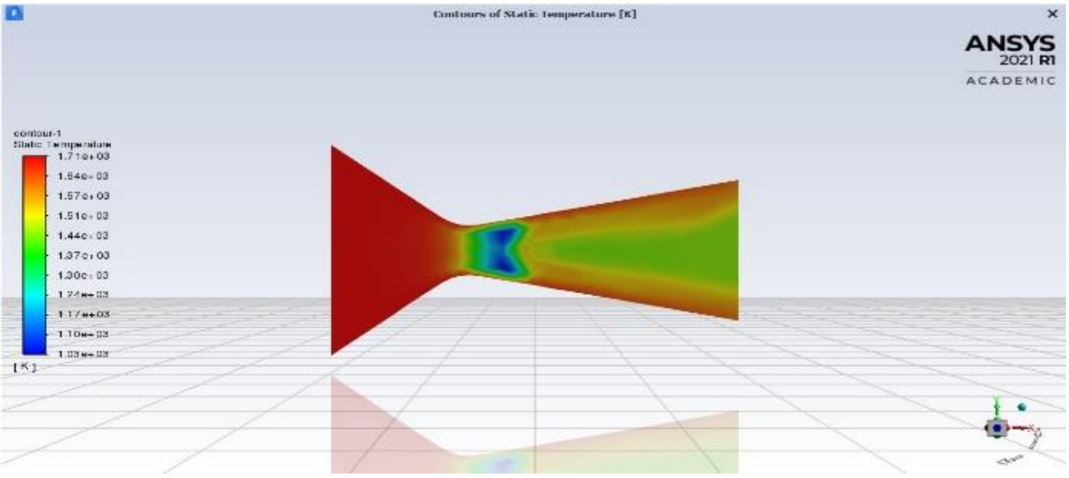

Fig 4.6 Contour of Static Temperature across Nozzle Axial Direction

Static temperature variation with respect to the nozzle axial direction is being depicted in the figure above. We can see static temperature is at its peak of the system in convergent section of nozzle while at the divergent section of nozzle, the dynamic portion increases and to maintain the energy balance of the system the Static contour is seen reduced.

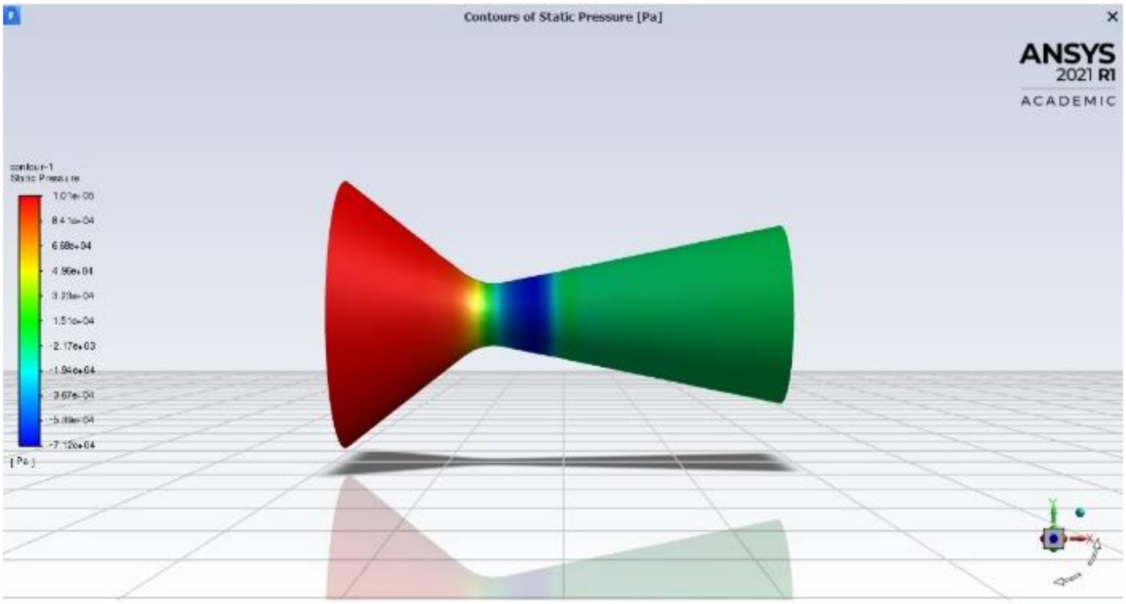

Fig 4.7 Contour of Static Pressure across Nozzle Axial Direction

Static Pressure variation with respect to the nozzle axial direction is being depicted in the figure above. We can see static pressure is at its peak of the system in convergent section of nozzle while at the divergent section of nozzle, the dynamic portion increases and to maintain the energy balance of the system the Static contour is seen reduced of the pressure. At throat we can see the Static pressure is at its minimum values.

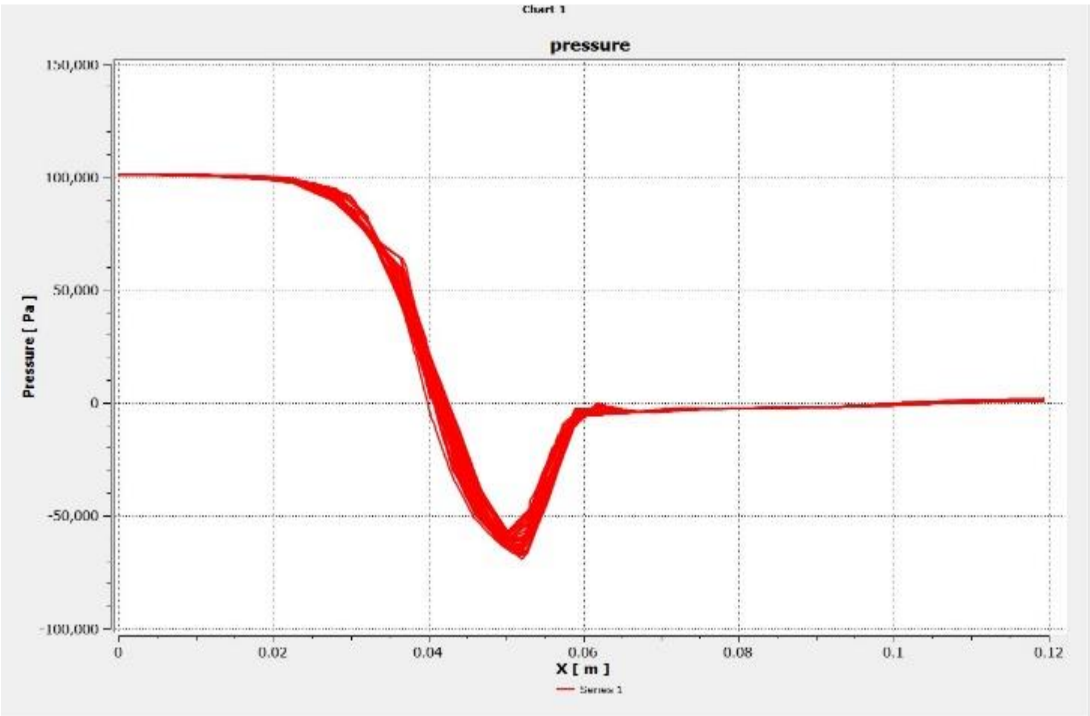

Fig 4.8 Variation of Pressure vs Axial Direction of Nozzle

Above figure shows the variation of Pressure across the nozzle axial direction. At the nozzle convergence we have pressure same as reservoir condition of 1.101325 bar and then with velocity increasing towards Nozzle throat we have drop in pressure to follow Momentum Equation.

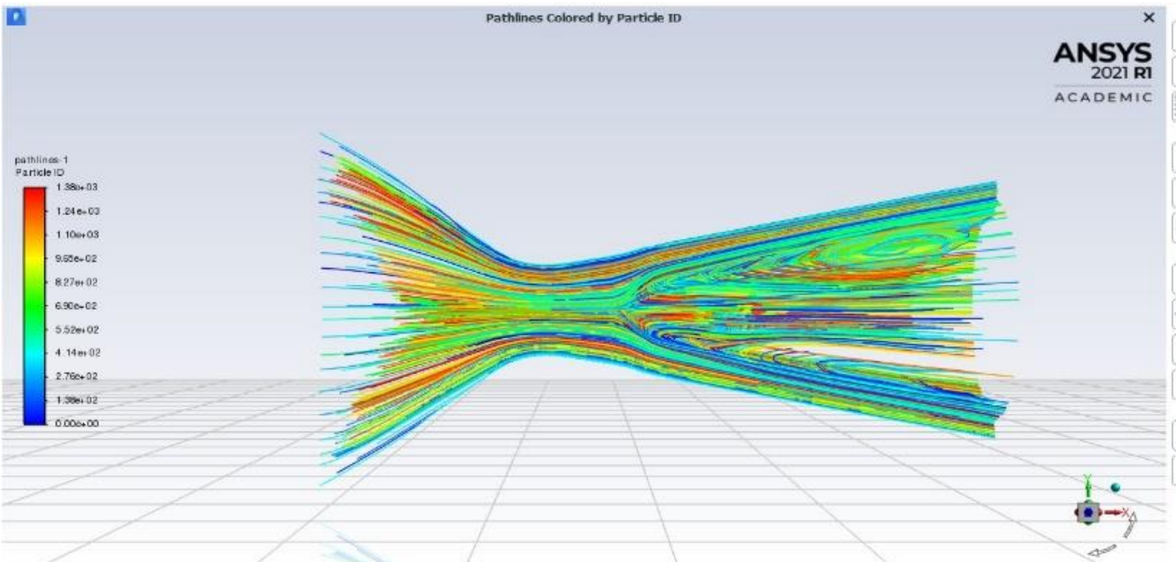

Fig 4.9 Path lines Contour across Nozzle Axial Direction

Across the Nozzle axial flow direction contour of coloured Path lines of particles is depicted for the Nozzle Convergent- Divergent Section. Here coloured path lines are used to make the flow separation across the nozzle be visualized in clear detail.

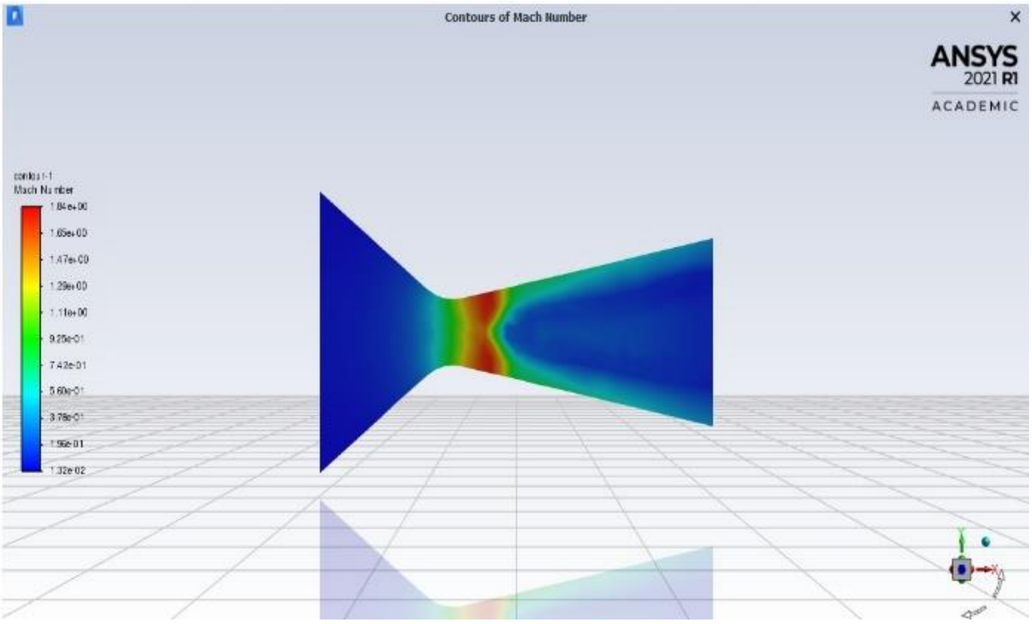

Fig 4.10 Contour of Mach No across Nozzle Axial Direction

In the above simulation figure Contour of Mach No across Nozzle Axial Direction is depicted. I t can be clearly seen that Mach No in convergent section is increasing because it follows continuity equation and after Throat Mach No is seen increasing . At Throat we have Minimum Area with maximum mass flow rate passing and having a Mach no as 1.

CONCLUSION

Rocket Nozzle with design exit Mach No of 2.955 was designed in Fusion 360 Software and then Simulated in ANSYS 2021R1 with above stated Axial flow Separation achieved. In this paper Nozzle Design was studied to its flow separation after throat section till exit in divergent section of nozzle with slight Turbulent regime achieved in rear exit frame of nozzle and also laminar flow regime was followed in mean axial Half diameter portion. The flow separation is seen from Nozzle wall with flow being curl making flow in that regime Turbulent and then again coming back to laminar regime at exit. Nozzle Design is successfully verified for Nozzle Flow separation in the Divergent section.

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