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Dual Diverging Annular Rocket Nozzle (Focusing Flow Separation and Minimizing Overexpansion)

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Abstract: ***Nozzle is a device designed to control the rate of flow, speed, direction, mass, shape, and the pressure of the stream that exhaust from them. Nozzles come in a variety of shapes and sizes depending on the mission of the rocket. This is very important for the understanding of the performance characteristics of rocket. The increasing demand for higher performance in rocket launches promotes the development of nozzles with higher performance, which is basically achieved by increasing the expansion ratio. However, this may lead to flow separation due to altitude variations and thus resulting in asymmetric forces, so-called side-loads, which may present life-limiting constraints on both the nozzle itself and other engine components. By the proper geometrical design of the nozzle, the exhaust of the propellant gases will be regulated in such a way that maximum effective rocket velocity can be reached without flow separation. Convergent divergent nozzle is the most commonly used nozzle.The present paper presents a comprehensive, review of supersonic flow separation in internal rocket nozzle due to altitude variations and hence a different method of separation control has been suggested by redesigning the bell-shaped rocket nozzle. In this work a comprehensive simulation of a flow in a typical supersonic converging-diverging bell shaped rocket nozzle has been reported. In the respective nozzle, flow suddenly contracts at a certain point and then expands after throat. All the simulation endeavors have been carried out by ANSYS FLUENT. The simulations have been conducted in 3D domains to provide better comparative platform. Also, a comprehensive simulation of a flow in the proposed supersonic converging-diverging bell shaped rocket nozzle has been conducted. Furthermore, comparison between CFD modeling results and corresponding available measured data has been presented.***

Keywords: Shock wave, over-expanded, rocket nozzle, flow separation, ANSYS FLUENT

1. Introduction

Nozzle produces thrust. It is used to convert thermal energy of hot chamber gases into kinetic energy and direct that energy along nozzle axis. Exhaust gases from combustion are pushed into throat region of nozzle. Throat is smaller cross-sectional area than rest of engine. The nozzle converts the low velocity, high pressure, high temperature gas in the combustion chamber into high velocity gas of lower pressure and temperature. The general range of exhaust velocity is 2 to 4.5 kilometer per second. The convergent and divergent (also known as C-D nozzle) type of nozzle is known as DE-LAVAL nozzle is shown in **Figure 1.** The inlet Mach number is less than one, Convergent section accelerates it to sonic velocity at the throat and further accelerated to supersonic velocities by the diverging section.

2. Nozzle Geometry Analysis

Rocket nozzle equations. The function of the nozzle is to accelerate gases produced by the propellant to maximum velocity in order to obtain maximum thrust. 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 are all determined by the rocket nozzle design. For steadily operating rocket propulsion system moving through a homogeneous atmosphere **total thrust and specific impulse** are:

$$F = \dot{m}.v_e + (p_e - p_a).A_e \quad (1)$$

Where,

$\dot{m}$ = mass flow rate

v_e = exit velocity of the exhaust gases

p_e = exit pressure

p_a = ambient pressure

A_e = Exit Area of the nozzle

$$I_{sp} = \frac{F}{\dot{m}g_0} \quad (2)$$

The first term is the momentum thrust and the second term represents the pressure thrust. The rocket nozzle is usually so designed that the exhaust pressure is equal or slightly higher than the ambient fluid pressure. Because changes in ambient pressure affect the pressure thrust, there is a variation of the rocket thrust with altitude (between 10% and 30%).

Velocity of sound and Mach number

$$a = \sqrt{\gamma.R.T} \quad (3)$$

$$M = v/a \quad (4)$$

The stagnation properties of a flow are those properties which would result if the flow is isentropic. Stagnation properties are constant in an isentropic flow. Thus, properties along the nozzle are best referenced against the stagnation properties. With these assumptions of ideal gas and isentropic flow, ratios of pressure, density and temperature can be related to the **stagnation pressure, density and temperature** at a given Mach number.

$$\frac{T_0}{T} = \left[1 + \frac{(\gamma-1)}{2}.M^2\right] \quad (5)$$

$$\frac{p_0}{p} = \left[1 + \frac{(\gamma-1)}{2}.M^2\right]^{\frac{\gamma}{\gamma-1}} \quad (6)$$

$$\frac{\rho_0}{\rho} = \left[1 + \frac{(\gamma-1)}{2}.M^2\right]^{\frac{1}{\gamma-1}} \quad (7)$$

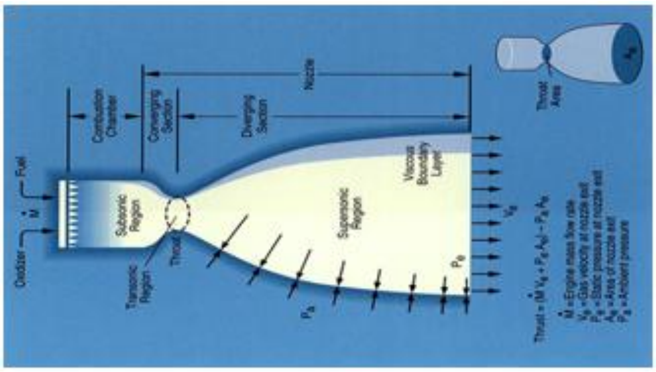

Figure 1: Basic flow sections of a conventional bell shaped nozzle

Additionally, the ratio of the local area to the **throat area** can be specified by the Mach number:

$$\frac{A}{A_t} = \frac{1}{M}\left[\frac{2}{\gamma+1}\left(1+\frac{(\gamma-1)}{2}.M^2\right)\right]^{\frac{\gamma}{2.(\gamma-1)}} \tag{8}$$

In a converging-diverging nozzle a large fraction of the thermal energy of the gases in the chamber is converted into kinetic energy. The flow velocity can be obtained from the conservation of total enthalpy:

$$v_e = \sqrt{2.(h_0 - h_e)} \tag{9}$$

From the isentropic relation for **exit velocity** becomes:

$$v_e = \sqrt{\frac{2\gamma}{\gamma-1}.\left(\frac{RT_0}{MolMass}\right)\left[1-\left(\frac{p_e}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} \tag{10}$$

Here,

$$\gamma = \frac{C_p}{C_v} = 1.2 \tag{11}$$

$$R = \text{Universal gas constant} = 8314.5(\text{J/kmol.K})$$
$$\text{MolMass} = \text{Molecular mass of the exhaust gases} = 14\text{kg/mol}$$
$$T_0 = \text{Inlet temperature} = 3000\text{K}$$
$$p_e = \text{absolute exit pressure} = .1\text{MPa}$$
$$p_0 = \text{absolute inlet pressure} = 12\text{MPa}$$

Thus, the **exit velocity of gases**:

$$v_e \approx 3500\text{m/s}$$

An increase of the ratio T_0/M increases the performance of the rocket. The influences of the pressure ratio p_0/p_e and of the specific heat ratio γ are less pronounced.

The nozzle **area expansion ratio (ε)** is an important nozzle design parameter:

$$\varepsilon = \frac{A_e}{A_t} \tag{12}$$

The maximum gas flow per unit area occurs at the throat (critical values):

$$\frac{p_t}{p_0} = \left[\frac{2}{(\gamma+1)}\right]^{\frac{\gamma}{\gamma-1}} \tag{13}$$

$$\frac{\rho_t}{\rho_0} = \left[\frac{2}{(\gamma+1)}\right]^{\frac{1}{\gamma-1}} \tag{14}$$

$$\frac{T_t}{T_0} = \left[\frac{2}{(\gamma+1)}\right] \tag{15}$$

Throat velocity is:

$$a = \sqrt{\frac{2\gamma}{\gamma+1}.R.T_0} = \sqrt{\gamma.R.T_t} \tag{16}$$

To attain sonic/supersonic flow:

$$\frac{p_0}{p_e} \geq \left[\frac{(\gamma+1)}{2}\right]^{\frac{\gamma}{\gamma-1}} \tag{17}$$

The mass flow rate as a function of nozzle geometry and fluid properties can be found from basic continuity where v the average velocity is, A is the nozzle area, and ρ is the density and ṁ is mass flowrate:

$$\dot{m} = \rho v A = \text{constant}$$

After substitutions, we have De Saint Venant's Equation:

$$\frac{\dot{m}}{A} = p_0\sqrt{\frac{2\gamma}{\gamma-1}\frac{1}{RT_0}}\left(\frac{p}{p_0}\right)^{\frac{2}{\gamma}}.\sqrt{\left[1-\left(\frac{p}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} \tag{18}$$

When sonic velocity is reached at the throat, it is not possible to increase the throat velocity or the flow rate in the nozzle by further lowering the exit pressure (choking the flow). Choking is a compressible flow effect that obstructs the flow, setting a limit to fluid velocity because the flow becomes supersonic and perturbations cannot moveupstream; in gas flow, choking takes place when a subsonic flow reaches M=1.

Mass flow rate:

$$\dot{m} = \rho_t v_t A_t = \frac{p_0 A_t\sqrt{\gamma}}{\sqrt{RT_0}}.\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \tag{19}$$

The **area ratio** is:

$$\frac{A_e}{A_t} = \frac{\sqrt{\gamma}.\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}}{\sqrt{\frac{2\gamma}{\gamma-1}\left(\frac{p_e}{p_0}\right)^{\frac{2}{\gamma}}\left[1-\left(\frac{p_e}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]}} \tag{20}$$

The **velocity ratio** is:

$$\frac{v_e}{v_t} = \sqrt{\frac{\gamma+1}{\gamma-1}\left[1-\left(\frac{p_e}{p_0}\right)^{\frac{\gamma-1}{\gamma}}\right]} \tag{21}$$

Calculated values for nozzle dimensions and geometry.
The dimensions of the convergent-divergent nozzle geometry are obtained through the following equations which are used in every spacecraft available during the present day.
Mass flow in rocket is calculated by:

$$\dot{m} = \frac{F_{thrust}}{v_e}$$

Where,
F_{thrust}=1.2MN and v_e=3500m/s

Putting the given values in the equation we obtain:

$$\dot{m} = 342.86\text{kg/s}$$

For high altitude (100 km or higher) expansion ratio in nozzle, given by above equations are between 40 and 200.

Area of the nozzle throat:

$$A_t = \frac{\dot{m}}{P_c\sqrt{\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left(\frac{MolMass}{R_u T_c}\right)}} \tag{22}$$

Given:
MolMass=14kg/mol; R_u=287(J/kg.K); T_c=3000K;
γ=1.2; P_c=12MPa and putting these values
$A_t \approx 0.0129m^2$
Now by using equation we can get the value of for ε=74:

A_e=0.95m^2
Also, the exit area of nozzle is given by:

$$A_e = \pi . r_e^2$$

Thus:
r_e=0.55m
Convergence area in Nozzle is:

$$A_c = 3. A_t$$
$$A_c = 0.0387m^2$$

Radius of the throat:

$$r_t = \sqrt{\frac{A_t}{\pi}} \quad (23)$$
$$r_t = 0.064m$$

Given, **diverging angle θ=15^0, and converging angle β=60^0**we can calculate the **converging and the diverging length,**

$$L_{cn} = \sqrt{\frac{A_c}{\pi}} . \frac{1}{\tan \beta} \quad (24)$$
$$L_{cn} = 0.081m$$
$$L_{dn} = \sqrt{\frac{A_e}{\pi}} . \frac{1}{\tan \theta} \quad (25)$$
$$L_{dn} = 2.04m$$

3. Conventional C-D Nozzle (DE-LAVAL NOZZLE)

3.1 Operation

Gas flows through nozzle from region of high pressure (chamber) to low pressure (ambient).Gas flows from combustion chamber into converging portion of nozzle, past the throat, through the diverging portion and then exhausts into the ambient as a jet. Pressure of ambient is referred to as back pressure. The variation of the pressure ratio with the length of the nozzle at different back pressure conditions is shown in **Figure 2.**

P_c : Chamber pressure
P_b: Back pressure or the atmospheric pressure
P_e : Pressure of the flow streams at nozzle exit

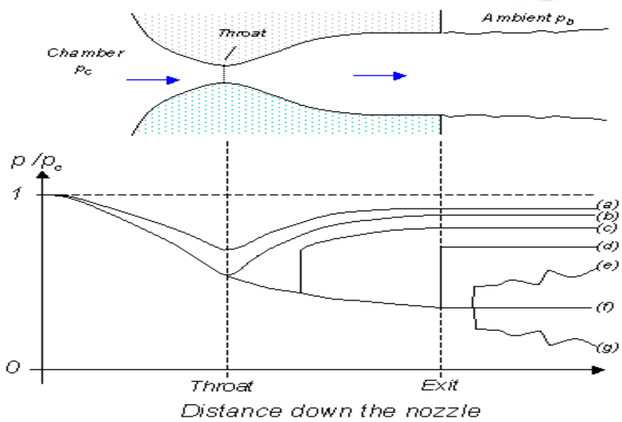

Figure 2: Pressure variation with the length of the Convergent-Divergent Nozzle

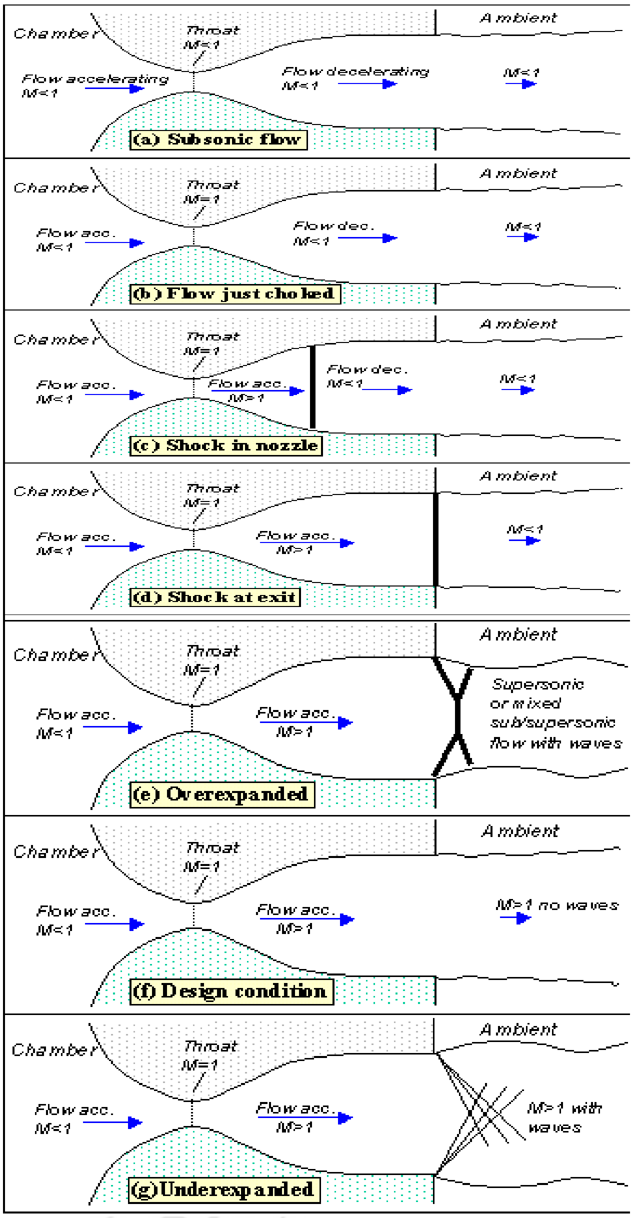

Figure 3: Flow regimes in a Convergent-Divergent Nozzle for different back pressures

The flow accelerates out of the chamber through the converging section, reaching its maximum (subsonic) speed at the throat. The flow then decelerates through the diverging section and exhausts into the ambient as a subsonic jet shown in **Figure 3. (a)**.

Lowering the back pressure in this state increases the flow speed everywhere in the nozzle. Further lowering P_b results in **Figure 3. (b)**. The flow pattern is exactly the same as in subsonic flow, except that the flow speed at the throat has just reached Mach 1. Flow through the nozzle is now choked since further reductions in the back pressure can't move the point of M=1 away from the throat. However, the flow pattern in the diverging section does change as the back pressure is lowered further.

As P_b is lowered below that needed to just choke the flow a region of supersonic flow forms just downstream of the throat. Unlike a subsonic flow, the supersonic flow accelerates as the area gets bigger. This region of supersonic

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acceleration is terminated by a normal shock wave. The shock wave produces a near-instantaneous deceleration of the flow to subsonic speed. This subsonic flow then decelerates through the remainder of the diverging section and exhausts as a subsonic jet. In **Figure 3. (c)** the shock wave is produced downstream the throat but inside the nozzle. In this regime if the back pressure is lowered or raised the length of supersonic flow in the diverging section before the shock wave increases or decreases, respectively.

If P_b is lowered enough the supersonic region may be extended all the way down the nozzle until the shock is sitting at the nozzle exit, **Figure 3. (d)**. Because of the very long region of acceleration (the entire nozzle length) the flow speed just before the shock will be very large. However, after the shock the flow in the jet will still be subsonic.

Lowering the back pressure further causes the shock to bend out into the jet and a complex pattern of shocks and reflections is set up in the jet which will now involve a mixture of subsonic and supersonic flow, or (if the back pressure is low enough) just supersonic flow. Because the shock is no longer perpendicular to the flow near the nozzle walls, it deflects it inward as it leaves the exit producing an initially contracting jet. We refer to this as **over-expanded** flow because in this case the pressure at the nozzle exit is lower than that in the ambient (the back pressure) i.e. the flow has been expanded by the nozzle too much, shown in **Figure 3. (e)**.

A further lowering of the back-pressure changes and weakens the wave pattern in the jet. Eventually, the back pressure will be lowered enough so that it is now equal to the pressure at the nozzle exit. In this case, the waves in the jet disappear altogether, **Figure 3. (f)**, and the jet will be uniformly supersonic. This situation, since it is often desirable, is referred to as the 'design condition', P_e=P_b.

Finally, if the back pressure is lowered even further we will create a new imbalance between the exit and back pressures (exit pressure greater than back pressure), **Figure 3. (g)**. In this situation, called **under-expanded**, expansion waves that produce gradual turning and acceleration in the jet form at the nozzle exit, initially turning the flow at the jet edges outward in a plume and setting up a different type of complex wave pattern.

3.2 Altitude Behavior

The Space Rockets mainly work in the regions shown in **Figure 3. (e)** to **Figure 3. (g)**. Altitude behavior of the rocket nozzle in these regions is shown in **Figure 4.**

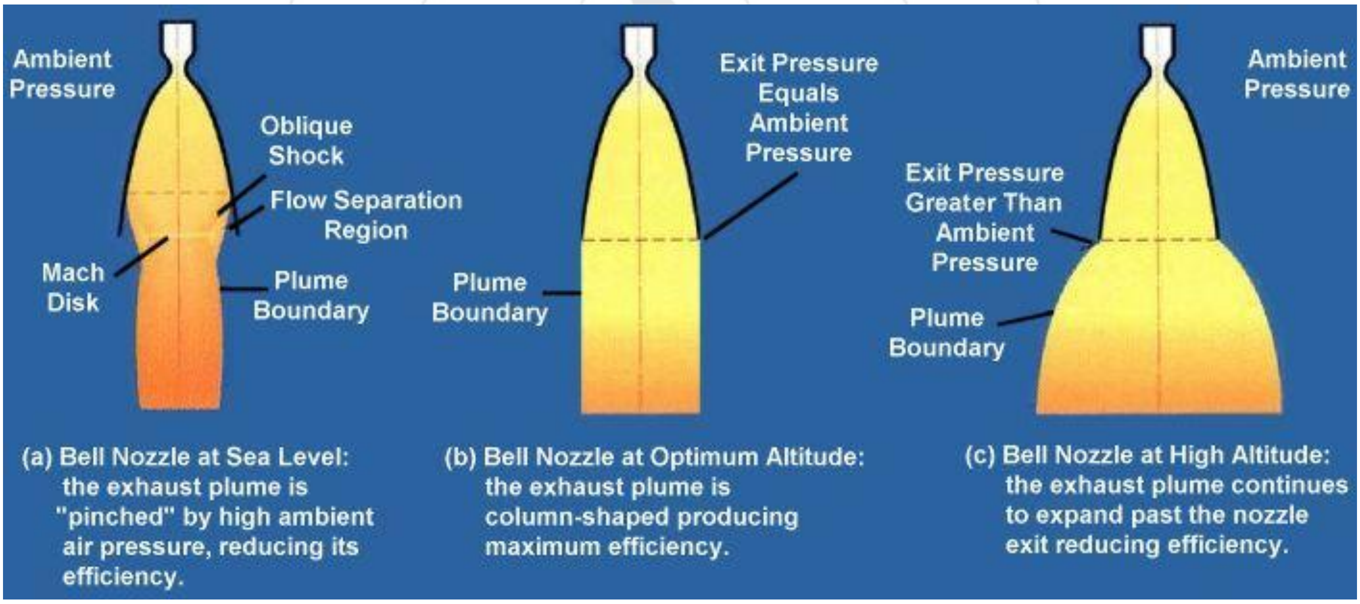

Figure 4: Velocity Streamlines for different altitudes

3.3 Disadvantages

- Over-Expansion of the nozzle thus causing a considerable thrust loss at lower altitudes
- Flow instability when over-expanded. May lead to uncertainty or unsteadiness of the thrust direction and dangerous high frequency wobble.

4. Proposed Idea (Dual Diverging Annular Nozzle)

Due to the above-mentioned disadvantages of the conventionally used bell-nozzle, a new design is proposed which is an annular shaped nozzle. It has a central divergent tube in the conventional bell shaped nozzle, shown in **Figure 5**. The purpose of the central divergent section is to force flow to remain attached to the nozzle walls. This behavior is desirable at low altitudes because the atmospheric pressure is high and may be greater than pressure of exhaust gases. When this occurs, the exhaust is forced inward and no longer exerts force on the nozzle walls, so thrust is decreased and the rocket becomes less efficient.

The central divergent section, however, increases the pressure of the exhaust gases by squeezing the gases into a smaller area thereby virtually eliminating any loss in thrust at low altitude. As altitude or back-pressure varies, flow is free to expand into the central section. This expansion into the central section allows the nozzle to compensate for altitude, i.e. exit pressure P_e adjusts to outside pressure P_b within nozzle.

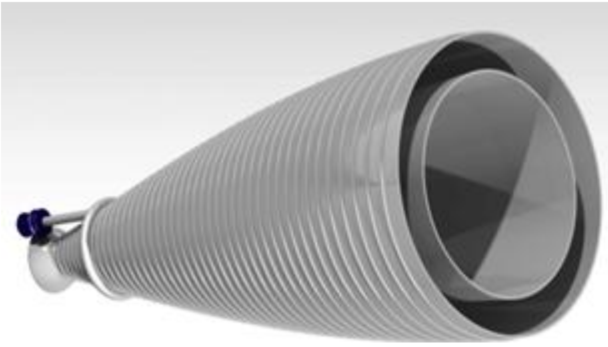

Figure 5: Dual Diverging Annular Nozzle

4.1 Advantages of Dual Diverging Annular Nozzle

- Smaller nozzle can be used.
- Exit velocity increases and hence thrust increases.
- Altitude compensation results in greater performance (no separation at over-expanded).
- Side thrust on the nozzle acting due to oblique shocks is reduced.

5. Modelling and Analysis

5.1 Modelling

The values used for geometric designing of nozzle are used to model the conventional bell shaped nozzle and the dual diverging annular nozzles in CATIA V5 shown in (**Figure 6-Figure 9**). The analysis of the flow through the both these nozzles is done in ANSYS FLUENT.

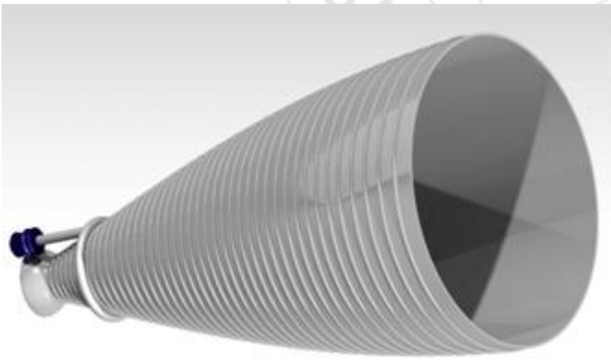

Figure 6: Conventional Bell Shaped Nozzle (CAD MODEL)

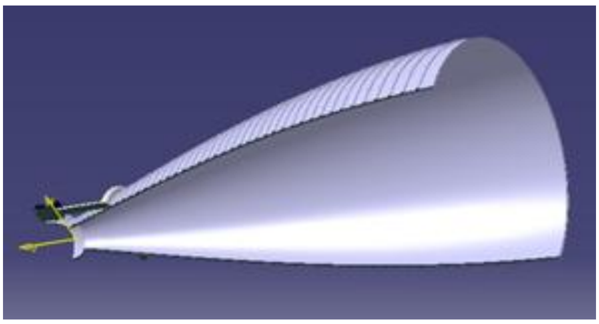

Figure 7: Cross Sectional View of Bell Nozzle

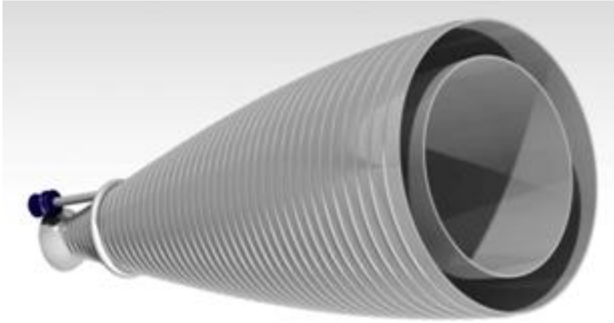

Figure 8: Dual Diverging Annular Nozzle (Cad Model)

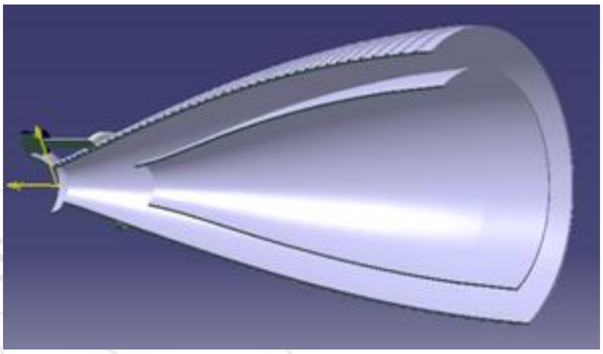

Figure 9: Cross Sectional View of Dual Diverging Annular Nozzle

The Cross-Sectional View Of The Annular Nozzle Showing The Inner Diverging Section Incorporated For Minimizing The Flow Reversal.

5.2 Analysis

5.2.1 Conventional Bell Shaped Nozzle

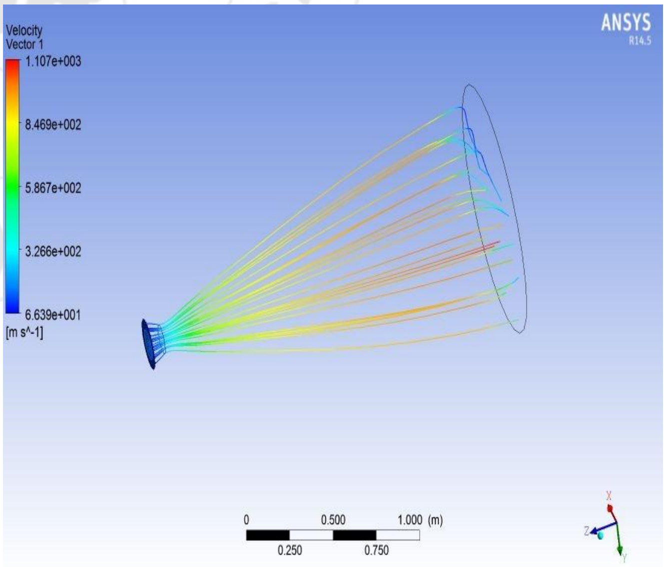

Figure 10: Velocity Streamlines for the Conventional Bell shaped Nozzle

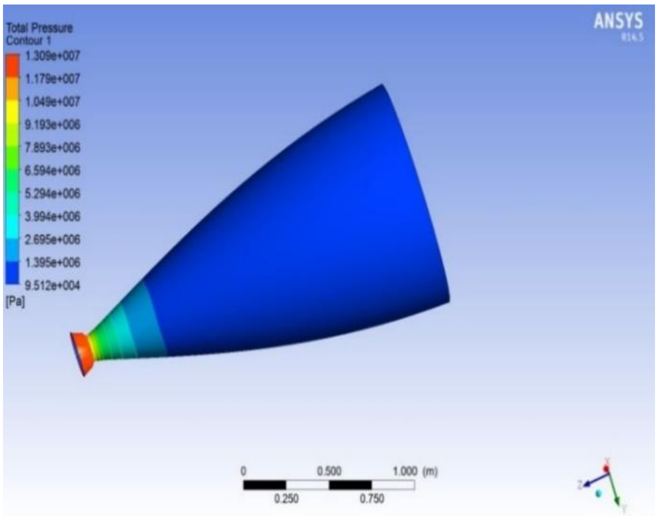

Figure 11: Pressure variation for the Conventional Bell shaped Nozzle

CFD is used for the analysis of the conventional Bell shaped nozzle. The boundary conditions used were the ambient pressure (p_a) as 0.1 MPa and the nozzle inlet pressure which is taken equal to 12 MPa as taken for the theoretical calculations. The boundary conditions given were Inlet Pressure and Ambient Pressure. The analysis shows the flow reversal in **Figure 10**. The streamlines are leaving the nozzle surface prior to the exit of the nozzle since the exit pressure is less than the ambient pressure as shown in **Figure 11**and thus many streamlines lose their velocity and hence reducing the thrust produced. On an average the exit velocity of the streamlines is 1107 m/s, using the thrust equation: $F = \dot{m}.v_e + (p_e - p_a).A_e$

$\dot{m} = 342.86\text{kg/s}, v_e = 1107\text{m/s}$,

$p_e = 0.09512$ MPa,

$p_a = 0.1$ MPa,

A_e=0.95m^2

This gives:

$$F = .375 \text{ MN}$$

5.2.2 Dual Diverging Annular Nozzle

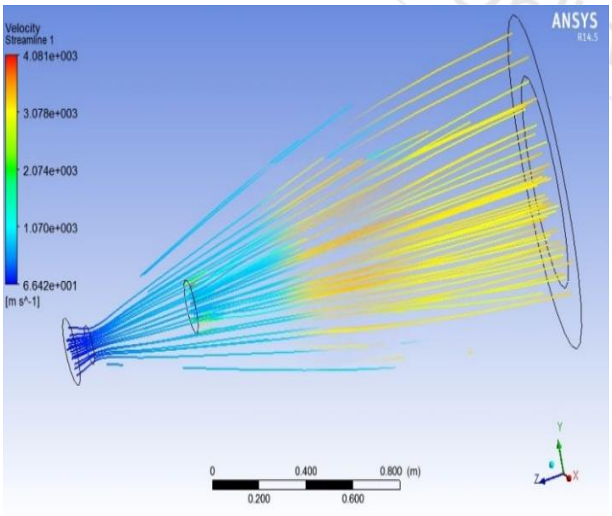

Figure 12: Velocity Streamlines for the Dual Diverging Annular Nozzle

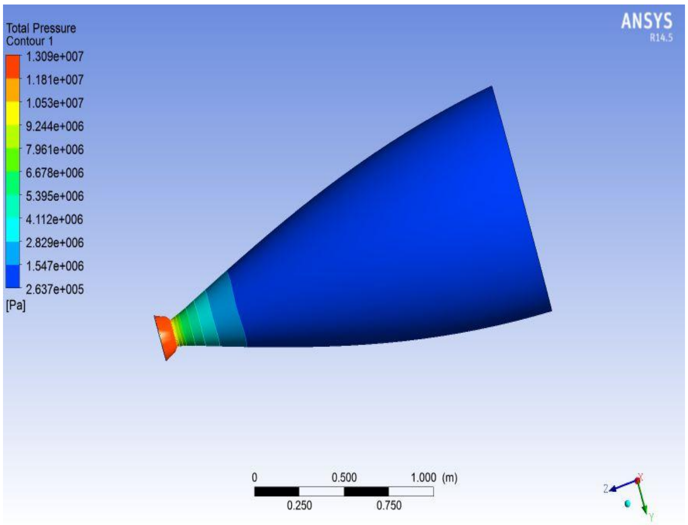

Figure 13: Pressure variation for the Dual Diverging Annular Nozzle

Similarly, analysis is done on the proposed Annular Nozzle using the same boundary conditions. The analysis shows a rather smooth streamline and no flow reversal in **Figure 12**as the exit pressure is more than the ambient pressure as shown in **Figure 13**. Due to no flow reversal, the amount of stream particles having the maximum velocity increases and hence, thrust force increases. Shock waves are reduced and the unwanted side thrusts are also reduced giving a smooth rocket flight.

$\dot{m} = 342.86\text{kg/s}$,

$v_e = 2900\text{m/s}$,

$p_e = 0.2637$ MPa,

$p_a = 0.1$ MPa,

A_e=0.95m^2

This gives:

$$F = 1.149\text{MN}$$

6. Comparison Graphs of Thrust Force, Exit Pressure and Velocity of Streamlines

Similar analysis was made for different back pressure conditions:

$\dot{m}$ = mass flow rate(kg/s)

v_e = exit velocity of the exhaust gases (m/s)

p_e = exit pressure(MPa)

p_a = ambient pressure(MPa)

A_e = Exit Area of the nozzle(m^2)

F = Thrust Force(MN)

	Bell Shaped Nozzle			Annular Nozzle		
p_a	p_e	v_e	F	p_e	v_e	F
0.1	0.09512	1107	0.375	0.2637	2900	1.149
0.05	0.08447	1821	0.657	0.2048	3046	1.191
0.025	0.07537	2218	0.808	0.1623	3496	1.329

Various analysis shows that the exit pressure for the conventional bell shaped nozzle is less than the ambient pressures at lower altitudes and thus causing overexpansion and reduction in the velocity at the exit due to the oblique shocks at these altitudes. Whereas in the Dual Diverging Annular Nozzle the exit pressure is always greater than the ambient pressure at all altitudes thus there is no flow reversal and the velocity at the exit is greater than that in the conventional bell shaped nozzle (**Figure14-Figure16**).

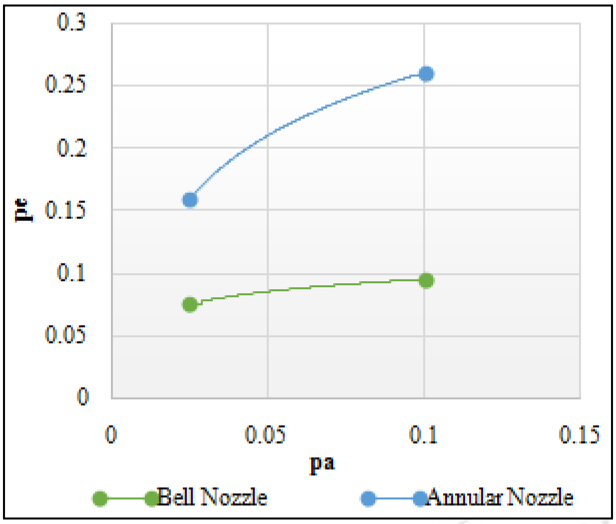

Figure 14: Comparison of exit pressure (p_e) vs ambient pressure (p_a)

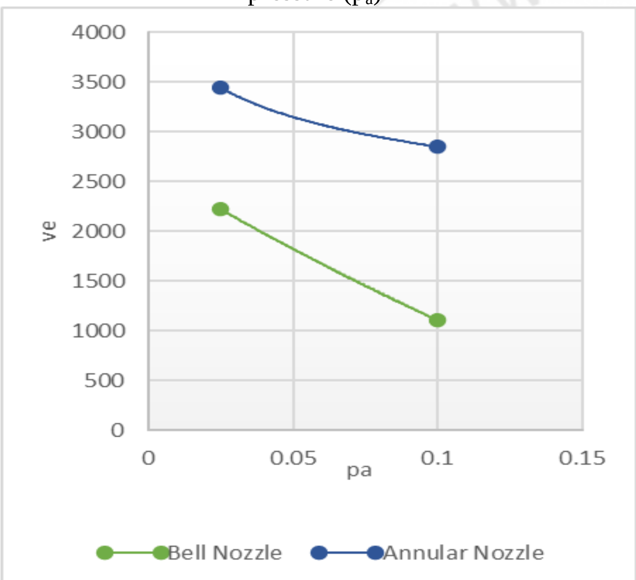

Figure 15: Comparison of exit velocity (v_e) vs ambient pressure (p_a)

The Thrust Force for a conventional bell shaped nozzle has an irregular variation throughout the flight due to overexpansion at the initial stages of the flight, whereas the analysis shows a gradual rise in the Thrust Force for the Dual Diverging Annular Nozzle as there is no case of overexpansion and thus the variation is smooth giving a smooth flight, shown in **Figure 16.**

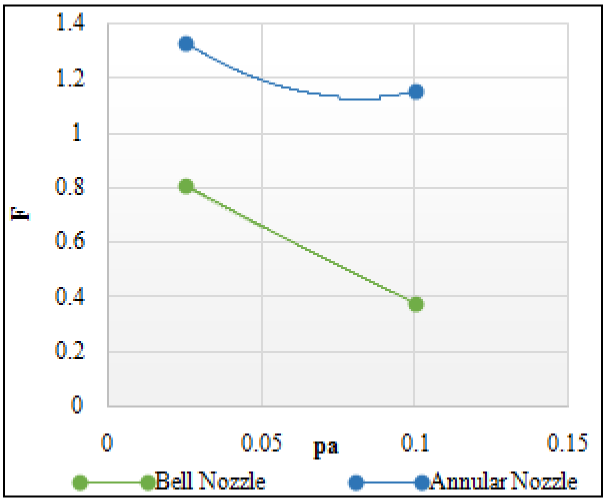

Figure 16: Comparison of Thrust (F) vs ambient pressure (p_a)

7. Thermal Stress/Strain Analysis of the inner Divergent section of Dual Divergent Annular Nozzle

The material used for the nozzle is Refractory Material **(4 Parts Tantalum carbide** and **1 part zirconium carbide** with **graphite)**
Having,
Melting Point = 3850-3880^{0}C (6960-7020^{0}F)
Density=1900kg/m3
Thermal Conductivity =18.5W/mK
Specific Heat = 750 J/kgK at 200^{0}C
Ambient Temperature= 15^{0}C at 0.1 MPa (International Standard Atmosphere)
Film coefficient of the gases=2.13*10^8W/m^{2}K
Tensile Ultimate Strength = 874MPa
Strain to Fail (%) = 0.15
Young Modulus = 430GPa
Bulk Modulus= 270GPa
Tantalum carbide and zirconium carbide are candidate for ultrahigh temperature applications because of their high melting point, good thermal shock resistance and absence of phase changes in the solid state. The flight data analysis shows that the maximum temperature of 2803.6^{0}C (5703^{0}F) is expected in the C-D nozzle. The melting point of Tantalum Carbide is higher than this temperature and hence the material does not fail due to melting. Using this boundary condition, the thermal stress and deformations of the diverging section are plotted, shown in **Figure 17**.

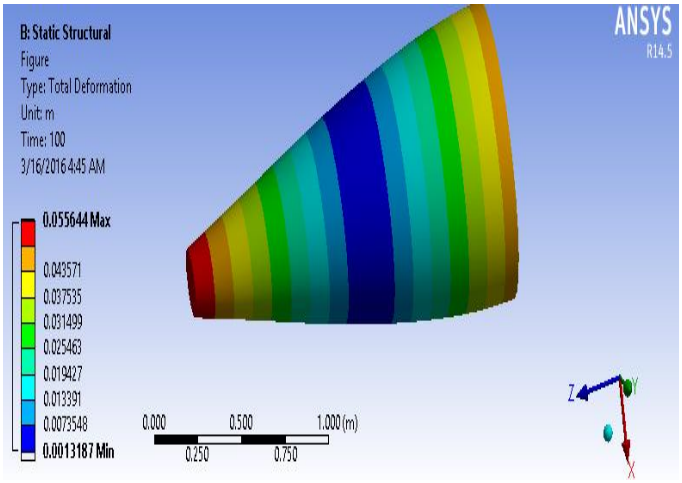

Figure 17: Thermal Stress/Strain Analysis of the Diverging Section of Dual Diverging Annular Nozzle

The figure is plotted for 3000 seconds, shows that the maximum thermal strain is 0.055m which is less than the failure strain for the Refractory Material, and hence the diverging part would not fail during the flight.

8. Conclusion

Space missions are a million dollar programs. Failure of one such mission leads to a drastic economical setback to any country. Incorporating new ideas in the field of shock waves in the C-D nozzle and flow reversals may reduce many rocket mission failures and can save country's economy and at the same time make way to new technologies in this field.

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