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Reduction of Noise in Aircraft Nozzle

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ABSTRACT: *The project mainly concentrates on the suppression of noise at the exit of the nozzle of the turbofan engine of the aircraft. The new concept was introduced in the convergent nozzle of the aircraft to reduce the noise emission. A nozzle is used to give the direction to the gases coming out of the combustion chamber. Nozzle is a tube with variable cross-sectional area. Nozzles are generally used to control the rate of flow, speed, direction, mass, shape, and/or the pressure of the exhaust stream that emerges from them. The nozzle is used to convert the chemical thermal energy generated in the combustion chamber into kinetic energy. The nozzle converts the low velocity, high pressure, high temperature gas in the combustion chamber into high velocitygas of lower pressure and low temperature. My study is to carried using software's like Catia for designing of the nozzle and Ansys fluent for analyzing the flows in the nozzle with different convergent and divergent angles. The Numerical study will be conducted to understand the air flows in a conical nozzle at different divergence degrees of angle using two-dimensional axisymmetric models, which solves the governing equations by a control volume method. The nozzle geometry co-ordinates are taken by using of method of characteristics which usually designed for nozzle. The present study is aimed at investigating the supersonic flow in conical nozzle for Mach number 3 at various divergence degree of angle. The throat diameter and inlet diameter is same for all nozzles with various divergence degree of angles. The flow is simulated using CFD fluent software. The flow parameters like pressure, Area of nozzle at exit are defined prior to the simulation. The result shows the variation in the pressure, temperature distribution and turbulence intensity.*
KEYWORDS *– Combustion chamber, CFD Analysis, Different divergent, Degree of angles.*

I. INTRODUCTION

Swedish engineer of French descent who, in trying to develop a more efficient steam engine, designed a turbine that was turned by jets of steam. The critical component is the one in which heat energy of the hot high-pressure steam from the boiler was converted into kinetic energy – was the nozzle from which the jet blew onto the wheel. De Laval found that the most efficient conversion occurred when the nozzle first narrowed, increasing the speed of the jet to the speed ofsound, and then expanded again. Above the speed of sound (but not below it) this expansion caused a further increase in the speed of the jet and led to a very efficient conversion of heat energy to motion. The theory of air resistance was first proposed by Sir Isaac Newton in 1726. Newton's theory was soon followed by other theoretical solution of fluid motion problems. All these were restricted to flow under idealized conditions, i.e. air was assumed to possess constant density and to move in response to pressure and inertia. Nowadays steam turbines are the preferred power source of electric power stations and large ships, although theyusually have a different design-to make best use of the fast steam jet, de Laval's turbine had to run at an impractically high speed. But for rockets the de Laval nozzle was just what was needed.

IMPORTANT FUNCTIONS OF NOZZLE:

- With minimum total pressure loss, it accelerates the flow with very high velocity.
- It can be used in thrust vectoring, by changing the direction of exhaust.
- Use to suppress infrared radiation.
- It allows the application of the thrust reversal; generally, clamshell and cascade types are used.
- In turbofan engine it mixes the mainstream and the bypass air.
- One of the most important function of the nozzle is to match exit and the atmosphere pressure that isdenoted by Pe=Pa,which is an optimum condition for the flow expansion from the nozzle.
- Thrust augmentation can be performed by the use of nozzle, especially in afterburner when additional fuelis burn in the tail pipe then the mass flow rate of the hot jet increases and for the expansion of this additionhot jet variable area nozzle is required.Generally, for subsonic flow, subsonic nozzles (convergent type) and for supersonic flow, De Laval nozzleare used this selection is generally based upon the Area-Mach number relationship.

INLET CONDITION:

The condition that one just after combustion chamber as well as the initial part of the nozzle at this condition area going to small just after the chamber so all the flow property going to change, here we to find out all the Mach no, velocity ,mass flow rate, thrust factor etc.

Given data at stagnation temperature as T_0 =3000k

- We must find out such pressure value in combustion chamber that have maximum Mach no, velocity so further we can get maximum thrust at nozzle exit condition.
- Pressure variation in combustion chamber (10-35) bar
- where we take variation of nozzle exit Mach no =(2.1 to 2.5)
- Inner diameter of chamber =60mm,port diameter=10mm
- For finding Mach number at Nozzle inlet
- Using area ratio formula:

$A_2/A_1 = (M_1/M_2)\ ((1+ (\Upsilon-1)/2\times M_2^2)/ (1+ (\Upsilon-1)/2\times M_1^2))^{\mathbf{3.833}}$ M_1=0.21

- Temperature at Nozzle inlet

$T_0/T_1 = 1+ (\Upsilon-1)/2\times M_1^2$ T_1 =2980.32k

- Pressure at Nozzle inlet

$P_0/P_1= 1+ (\Upsilon-1)/2\times M_1^2$ P_1 =9.270 bar

Density at Nozzle inlet

$\rho=p/ (RT)$, ρ_1 =1.420Kg/m^3

- Velocity and free stream air velocity at Nozzle inlet

M=v/a a= $(\Upsilon RT)^{0.5}$ v_1=99.824m/sec

INLET AND OUTLET:

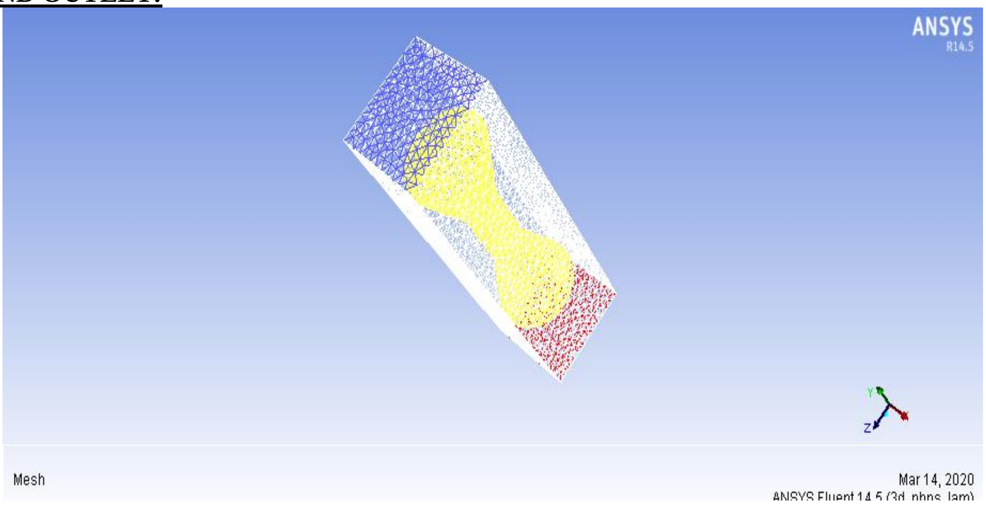
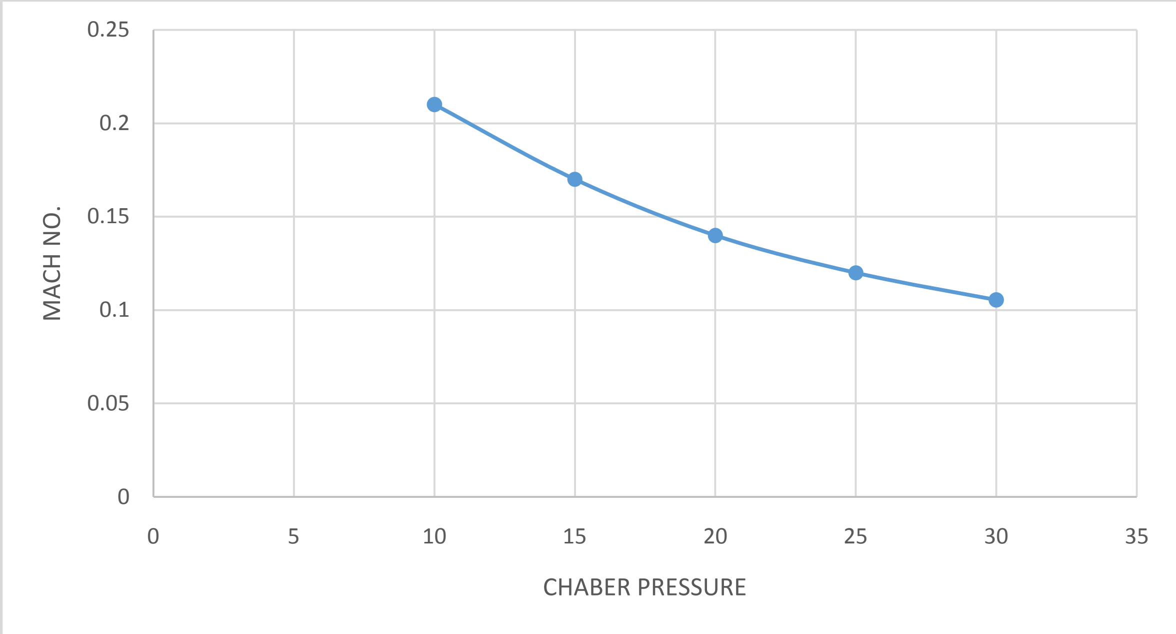


Table 1

(Pressure Variation at Nozzle inlet condition)

Variable stagnation pressure (N/m^2)x10^5	**Stagnation temp(k)**	**Nozzle exit Mach**	**Nozzle inlet Mach**	**Pressure (N/m^2)**	**Temp (K)**	**Density (Kg/m^2)**	**Mass flow rate (Kg/sec)**
30	3000	2.1	0.1055	29.792	2995.20	4.336	0.033
25	3000	2.2	0.1200	24.768	2993.41	3.607	0.03211
20	3000	2.3	0.1413	19.742	2991.02	2.877	0.03014
15	3000	2.4	0.1700	14.723	2987.06	2.149	0.02707
10	3000	2.5	0.2100	9.720	2980.32	1.420	0.0220

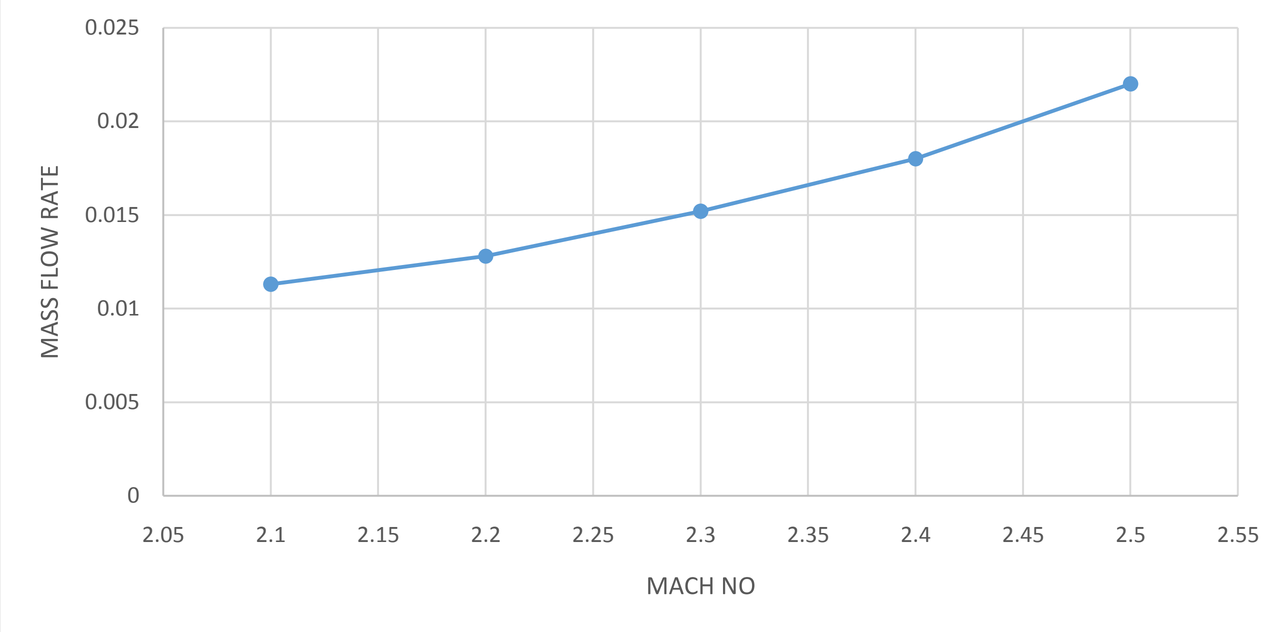


Graph 1: Chamberpressure vs. Inlet Mach no

Table 2

(At constant pressure 10bar variation of Mach number)

Stagnation pressure $(N/mm^2)x10^5$	Stagnation temp(K)	Nozzle exit Mach no	Nozzle inlet Mach no	Pressure $(1x10^5)$ N/m^2	Tempe(k)	Density (kg/m^3)	Mass flow rate (kg/sec)
10	3000	2.1	0.1055	9.93	2995.20	1.445	0.0113
10	3000	2.2	0.1200	9.90	2993.41	1.440	0.0128
10	3000	2.3	0.1413	9.87	2991.02	1.438	0.0152
10	3000	2.4	0.1700	9.82	2987.06	1.433	0.0180
10	3000	2.5	0.2100	9.72	2980.32	1.420	0.0220



Graph 2 Mach no vs. mass flow rate

THROAT CONDITION:

Throat is an important part of nozzle section at this condition nozzle have minimum area and other physical value of flow property will change like velocity will be large than inlet condition.

- As we have data from nozzle inlet condition: -

Inlet Mach no=o.21
Inlet temperature=2980.32k
Inlet pressure=9.720 x10^5N/mm^2
Inlet density=1.420kg/mm^3
Mass flow rate =22gm/sec

- Now we to calculate
- For finding Temperature at nozzle throat
  $(T_0/T_t)= 1+ (\Upsilon-1)/2\times M_*^2$ T_t =2608k
- From temperature, pressure & density relation
  $(T_0/T_t)^{\Upsilon/\Upsilon-1}= (\rho/\rho_t)^{\Upsilon}= P/P_{t.}$
- For finding pressure nozzle throat
  $(T_0/T_t)^{\Upsilon/\Upsilon-1} = P/P_t P_t$=5.464bar
- For finding Density at nozzle throat
  $(P/\rho t)^{\Upsilon}= P/P_t \rho_t$ = 0.912Kg/m^3
- For finding out throat diameter
  $a= (\Upsilon RT)^{0.5}$ M=V/a where M=1

Mass flow rate calculation at nozzle throat
$M^*= \rho_t A_t V_t D_T$ =14mm

NOZZLE EXIT CONDITION

The nozzle exit is the end point of nozzle section at which testing is carried out for all the final successful design of our rocket motor, we have checked out all physical variable value like thrust factor, mass flow rate, exit velocity, specific impulse etc.

- From above calculation at nozzle throat section

Nozzle throat Mach no=1
Exit Mach no=2.5
Throat diameter=14mm
Throat pressure=5.464bar
Throat temperature=2698.695k
Throat density=0.912kg/m^3

- Using area ratio formula

$A_e/A_t = (m_t/m_e) \times ((1+ (\Upsilon\text{-}1)0.5M_e^2)/ (1+ (\Upsilon\text{-}1)0.5M_t^2))^{\mathbf{3.833}}$ $D_{e=}$28mm

- Exit temperature
  $T_0/T_e=1+0.15M_e^2$ T_e = 1548.38k

Exit pressure

- $P_0/P_e= (1+0.15M_e^2)^{4.33}$ P_e= 0.57bar
- Exit density
  $(P0/\rho e_)^{\Upsilon}=(P_0/p_e)$ ρ_e = 0.1604Kg/m^3
- Exit velocity calculation
  M=V/a a= $(\Upsilon RT)^{0.5}$ V_e= 1698.78m/sec
- Mass flow rate calculation
  $M^{®}/A^{*}= (P_0/T_0) \times (\sqrt{\Upsilon/R}) \times (2/ (\Upsilon+1))^{3.83} M^{®}$ =123.31Gm/Sec
- Thrust calculation
  $F=M^{®}V_e$ F =208.73N
- Discharge coefficient calculation
  $C_F = F/A_t \times P$ = 1.356 where p denotes chamber pressure

NOZZLE LENGTH CALCULATION

Where Area ratio A_t/A_e= 0.25

CONVERGENT SECTION **DIVERGENT SECTION**

$L_c = (D_e+D_{th})/ (2\tan\theta)$ $L_D= (D_e- D_{th})/ (2\tan\theta)$

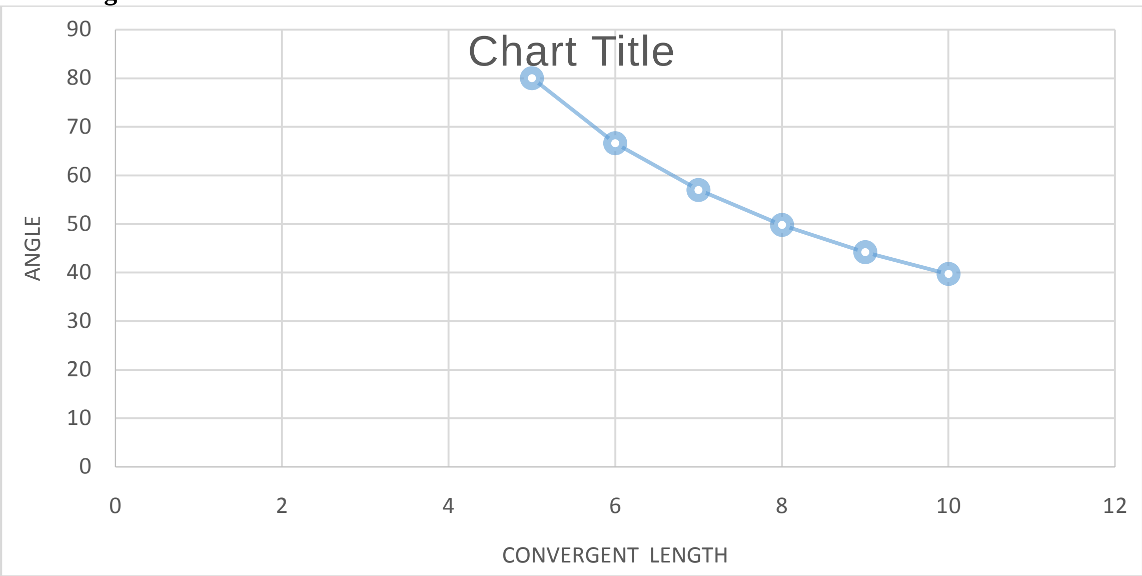
Table no 3.4.1 convergent section

ANGLE(θ®)	LENGTH(mm)
5	80.01
6	66.60
7	57.01
8	49.80
9	44.19
10	39.69

Table no 3.4.2 divergent section

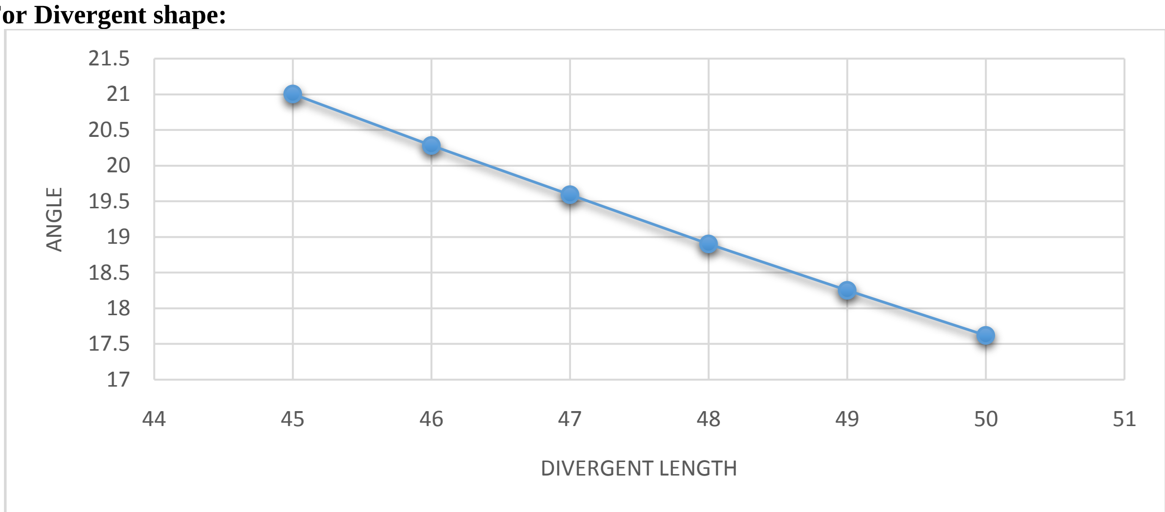
Convergent ANGLE(θ®)	Convergent LENGTH(mm
45	21
46	20.28
47	19.59
48	18.90
49	18.25
50	

For Convergent section



Graph 3 (convergent length Vs. Angle)

For Divergent shape:



Graph 4 Divergent length VS Angle

THICKNESS CALCULATION

For the fabrication of hybrid rocket motor material which we using is stainless steel .the yield stress steel is 205 Mpa .but presently in market only in the range of 200 Mpa is available.

FUEL TO AIR RATIO CALCULATION

Here we are using paraffin wax as solid fuel and liquid oxygen as oxidizer.

(1) Paraffin wax (c25H52)---fuel

(2) O2 (liquid oxygen) -oxidizer

The general chemical reaction in the combustion chamber as follow

C25 H52 + 38 O2 = 25 CO2 + 26 H2O

HEAT OF REACTION:

The Heat of Reaction is the change in the enthalpy of a chemical reaction that occurs at a constant pressure. It is a thermodynamic unit of measurement useful for calculating the amount of energy per mole either released or produced in a reaction.

Paraffin wax is $C_{25}H_{52}$

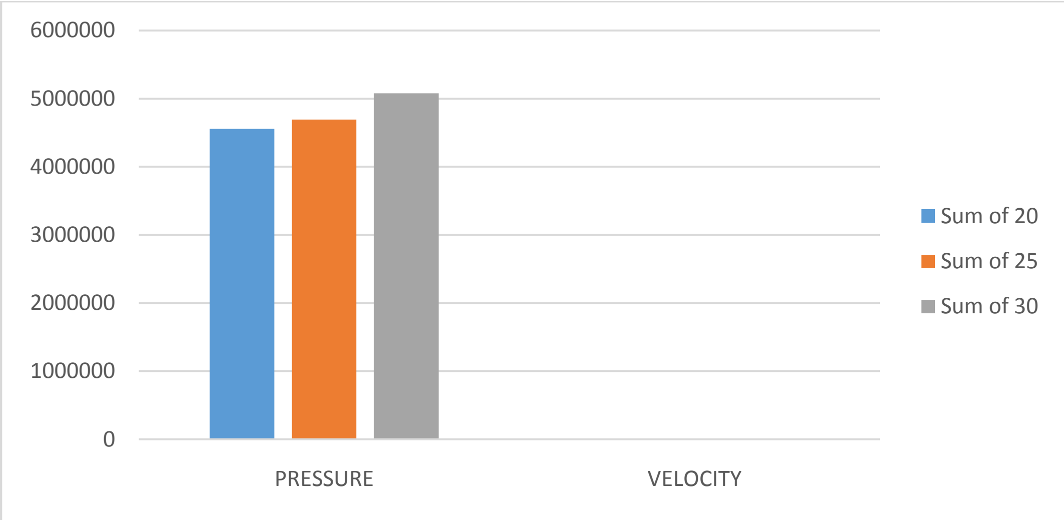
Molecular weight =352 gm. /mole

1mole =25 mole carbon and 26 mole H_2

1 gm. paraffin =0.0028 mole paraffin

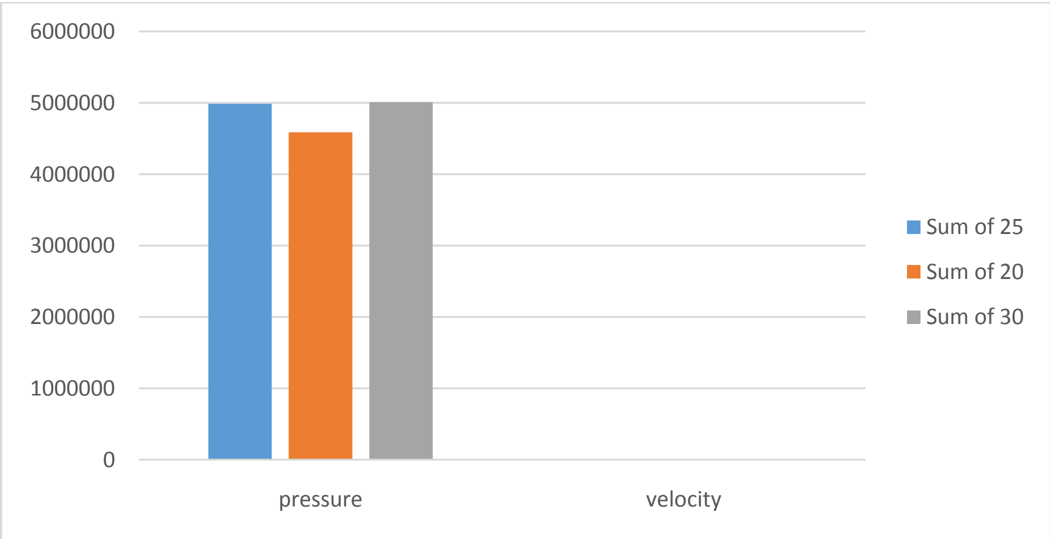
ALUMINIUM

ALUMINIUM	Sum of 20	Sum of 25	Sum of 30
Pressure	4557000	4690000	5080000
Velocity	3055	3044	850.3
Grand Total	**4560055**	**4693044**	**5080850.3**



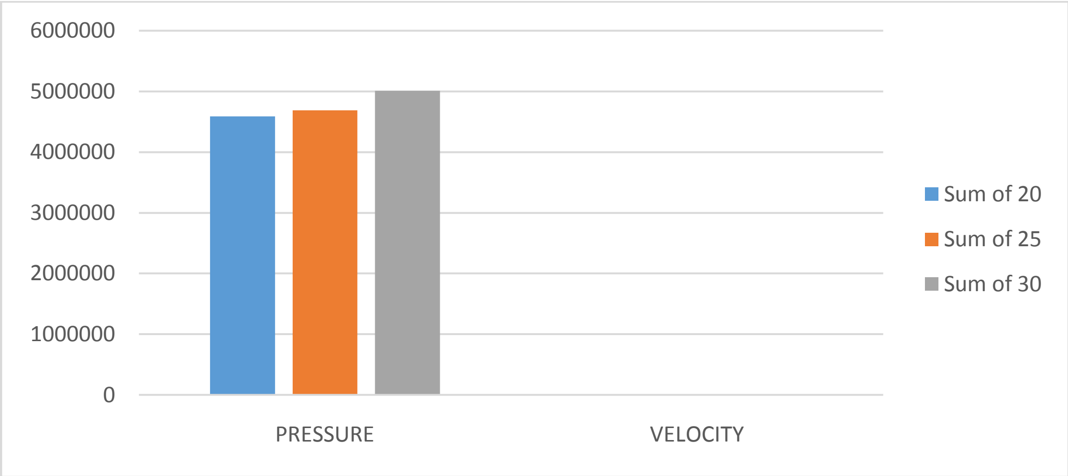
STEEL

STEEL	Sum of 25	Sum of 20	Sum of 30
pressure	4987000	4587000	5008000
velocity	840.5	3055	2986
Grand Total	**4987840.5**	**4590055**	**5010986**



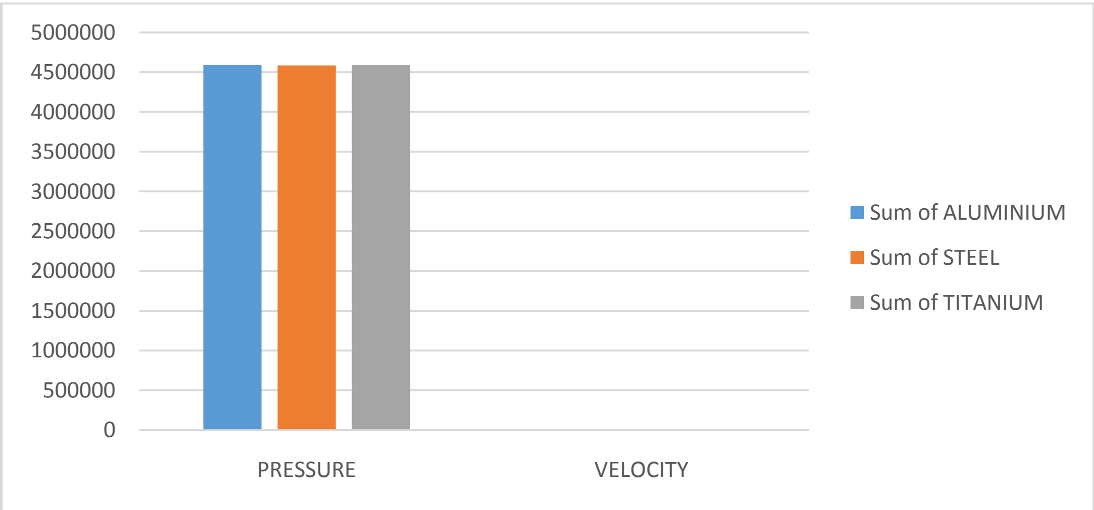
TITANIUM

TITANIUM	Sum of 20	Sum of 25	Sum of 30
PRESSURE	4587000	4691000	5008000
VELOCITY	880.8	840.5	2986
Grand Total	**4587880.8**	**4691840.5**	**5010986**

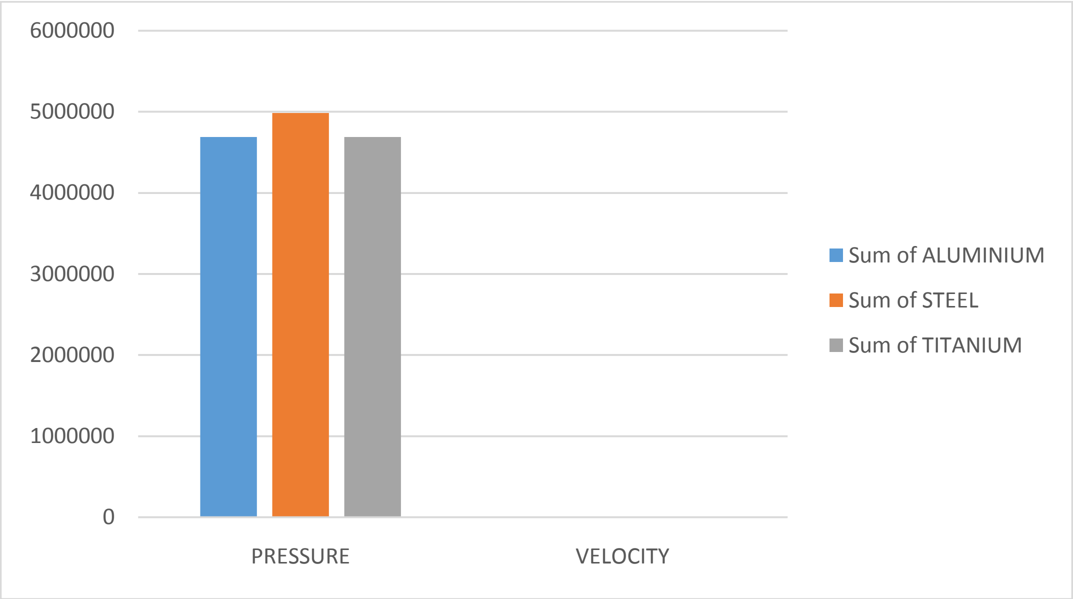


COMPARISION OF ALUMINIUM, STEEL, AND TITANIUM IN ANGLE 20,25,30.

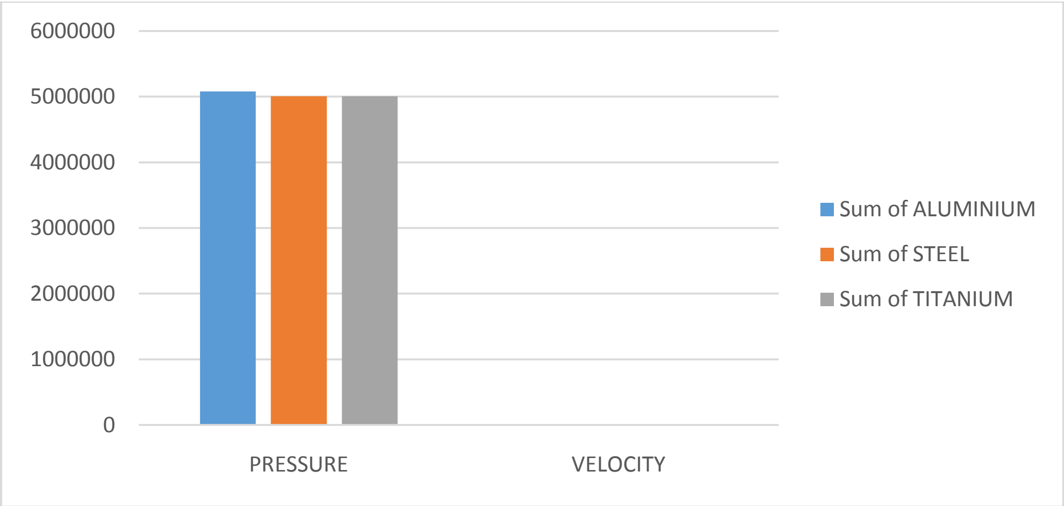
ANGLE 20	Sum of ALUMINIUM	Sum of STEEL	Sum of TITANIUM
PRESSURE	4587000	4587000	4587000
VELOCITY	3055	305.5	880.8
Grand Total	**4590055**	**4587305.5**	**4587880.8**



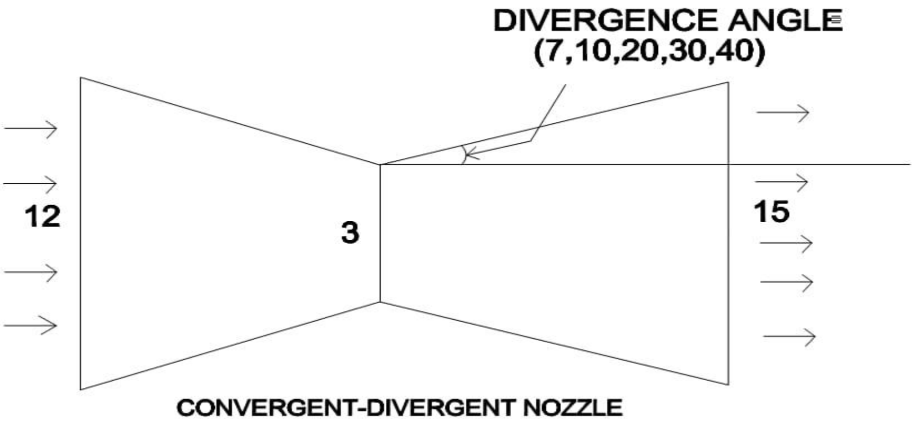
ANGLE 30	Sum of ALUMINIUM	Sum of STEEL	Sum of TITANIUM
PRESSURE	5080000	5008000	5008000
VELOCITY	850.3	2986	2986
Grand Total	**5080850.3**	**5010986**	**5010986**



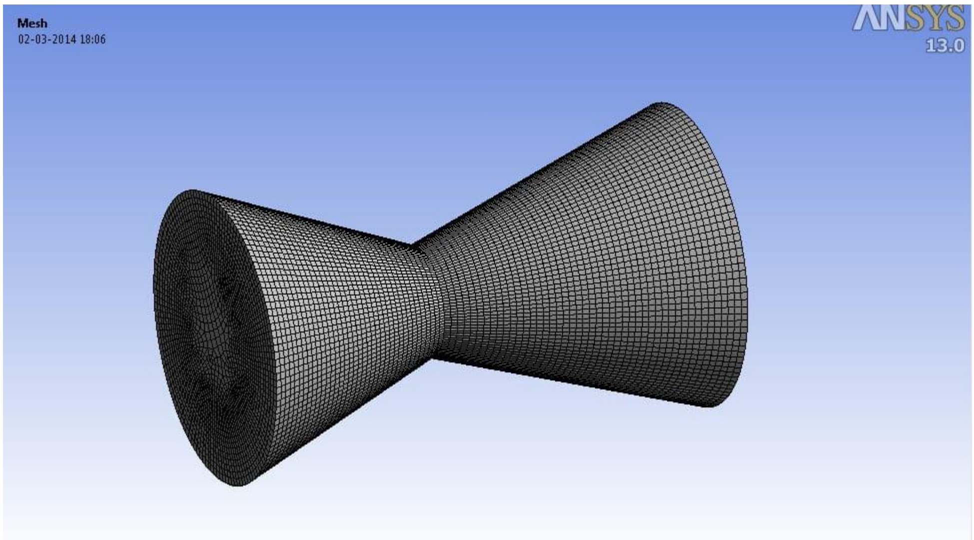
ANGLE 25	Sum of ALUMINIUM	Sum of STEEL	Sum of TITANIUM
PRESSURE	4691000	4987000	4691000
VELOCITY	3044	840.5	840.5
Grand Total	**4694044**	**4987840.5**	**4691840.5**



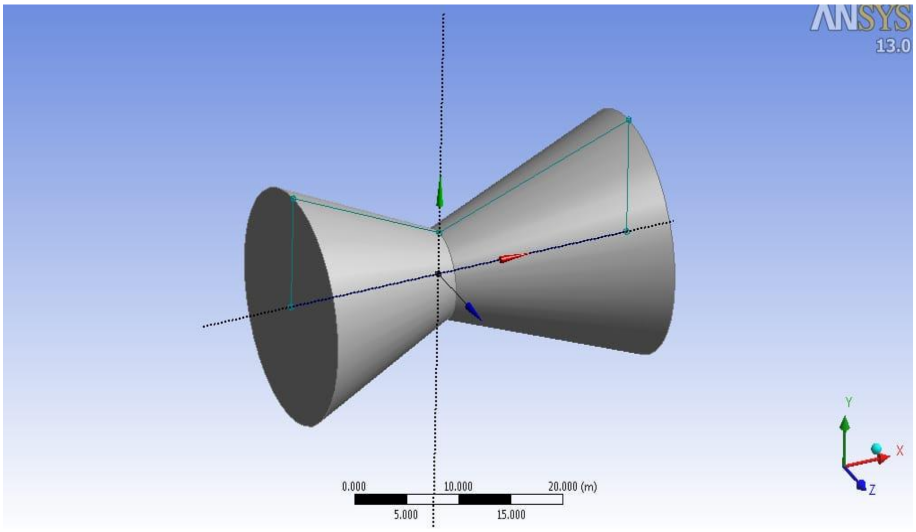
2D GEOMETRY



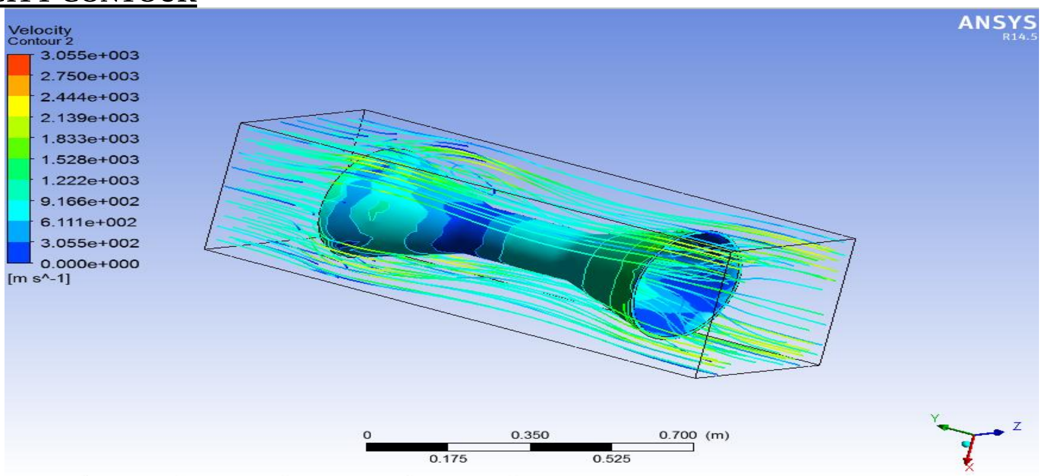
MESH



3D MODEL



VELOCITY CONTOUR



II. CONCLUSION

From the obtained numerical results it was concluded the followings. At the throat section also, the Mach number goes on increasing with increase in divergent angle. The static pressure decreases with increased divergent angle.

It was observed that oblique shocks are formed during flow through the nozzle. When the divergent angle was 30 °, the first shock occurred at 1m from the inlet and this wave reflected from the walls of the nozzle. It was found that the decrease in divergent angle displaces the shock towards the exit of the nozzle. The strengthen of nozzle was increased with the reduced angle. When the angle was reduced the strengthen of the nozzle was reduced. Also, from the obtained cfd results it was found that at different angles of investigation the 15 degree which means the lesser angle will provide better flow characterization. Also found the suitable material was Aluminum.

REFERENCES

[1]. Altman, D, "Hybrid Rocket Development History," 27th Joint PropulsionConference, 1991-2515, AIAA, Sacramento, CA, 1991, pp. 1-19.

[2]. J. P. Arvis, H. S. Jones, K. Kline, K. Smith, T. Slack, and T. Bales, "Development of a N20/HTPB Hybrid Rocket Motor," 33rd JointPropulsion Conference, 1997-2802, AIAA, Seattle, WA, 1997, pp. 1-8.

[3]. J. P. Arvis, H. S. Jones, K. Kline, K. Smith, T. Slack, and T. Bales,"Overview of Hybrid Sounding Rocket Program," 33rd Joint PropulsionConference, 1997-2799, AIAA, Seattle, WA, 1997, pp. 1-5.

[4]. J. P. Arvis and H. S. Jones, "A Standardized Technique for EvaluatingHybrid Rocket Motor Performance," 33rd Joint Propulsion Conference,1997-2933 AIAA, Seattle, WA, 1997, pp. 1-13.

[5]. T. A. Boardman, T. M Abel, S. E. Claflin, and C. W. Shaeffer, "Designand Test Planning for a 250-KLBF Thrust Hybrid Rocket Motor under theHybrid Propulsion Demonstration Program," 33rd Joint PropulsionConference, 1997-2804, AIAA, Seattle, WA, 1997, pp. 1-15.

[6]. C. Carmicino, and A. Russo Sorge, "Investigation of the Fuel RegressionRate Dependence on Oxidizer Injection and Chamber Pressure in a HybridRocket, " 39th Joint Propulsion Conference, 2003-4591, AIAA, Huntsville, AL, 2003, pp. 1-11.

[7]. Hollman, S., L. Frederick, R. A., Jr.," Lab scale Testing Techniques forHybrid Rockets," AIAA 1993-2409, 1993.

[8]. W. T. Horton, P. J. Mueller, and F. J. Redd," UTAH-X – Utah's ThirdAdvanced Hybrid Experiment," 32nd Joint Propulsion Conference, 1996-2695, AIAA, Lake Buena Vista, FL, 1996, pp. 1-5.

[9]. R. W. Humble, M. P. Bettner, and R. A. Sand fry," Hyster – Hybrid RocketProgram at The United States Airforce Academy," 33rd Joint Propulsion Conference, 1997-2797, AIAA, Seattle, WA, 1997, pp. 1-10.

[10]. R. Kannalath, A. Kuznetsov and B. Natan," Design of a Lab-ScaleHydrogen Peroxide/Hydroxyl Terminated Polybutadiene Hybrid Rocket Motor," 39th Joint Propulsion Conference, 2003-4744, AIAA, Huntsville,AL, 2003, pp. 1-11.