

# Investigation of Jet Noise Reduction in the Hybrid Rocket Nozzle

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## ABSTRACT:

This project presents an attempt to use the hybrid rocket motor for missile applications. The gas released with high pressure causes noise above 130 db. According to recent studies, loud noise above 120 dB has a strong vibrational energy that leads to a negative influence on electronic equipment with a high degree of integration. In this study, a basic fire-extinguishing nozzle with absorbent was selected as the reference model and numerical analysis was conducted using the commercial software, ANSYS FLUENT ver. 18.1 and comparing with materials of aluminums and titanium in nozzle section. The calculation results show that the jet noise is with obvious directivity. The total sound pressure levels are obviously much higher in 10°to 30°direction than that in other direction. Results show that the vibrational energy of the noise was reduced by minimizing the turbulence kinetic energy. Pressure and velocity distributions were calculated and graphically depicted with various absorbent configurations.

## INTRODUCTION;

- Hybrid rockets are simply a type of rocket that utilize their fuel and oxidizer in two separate phases. They have been studied since the early 1930s but have yet to reach the same level of adoption as solid fuel rockets and liquid fuel rockets. While relatively old, research in hybrid rocketry is still a developing field with many avenues for improvement and better understanding.

### Figure 1-1 Basic Hybrid Rocket

- The key feature as previously stated was that hybrid rockets utilize the fuel and oxidizer in two separate phases. This method is in contrast to liquid rockets which utilize the fuel and oxidizer both in liquid or gas form and also in contrast to solid rockets which mix oxidizer with fuel into a single solid fuel. In general there is some sort of pressurized oxidizer tank that feeds gas or liquid into the chamber that contains the solid fuel. Other variations exist with opposite phases or other more exotic. Rocket engines are a class of jet propulsion systems in which two (or more) propellants are reacted in an open volume. The heat evolved causes rapid expansion of gases through a convergent-divergent nozzle and the momentum of the resulting supersonic jet imparts a force on the rocket chamber. All rockets, by definition,

carry both the fuel and oxidizer required for the combustion reaction differentiating them from air-breathing jet engines. There exist three general classes of chemical rocket systems liquids, solids and hybrids. In a liquid system, both the fuel and oxidizer consist of liquids which are pumped separately into the combustion chamber and reacted. In a solid system the fuel and oxidizer both exist in solid form and are either molecularly bonded or intimately mixed physically. A hybrid system, as its name suggests, consists of one solid propellant and one liquid propellant. The fuel can be either the liquid or the solid and the same goes for the oxidizer. However, useful oxidizers tend to be liquids and so the typical configuration of a hybrid rocket consists of a liquid oxidizer reacted with a solid fuel.

## 1.2 HYBRID ROCKETS;

Rocket propulsion concepts in which one component of the propellant is stored in liquid phase while the other is stored in solid phase are called hybrid propulsion systems. Such systems most commonly employ a liquid oxidizer and solid fuel. Various combinations of solid fuels and liquid oxidizers as well as liquid fuels and solid oxidizers have been experimentally evaluated for use in hybrid rocket motors. Most common is the liquid oxidizer-solid fuel concept. The oxidizer can be either a non-cryogenic (storable) or a cryogenic liquid, depending on the application requirements. In this hybrid motor concept, oxidizer is injected into a precombustion or vaporization chamber upstream of the primary fuel grain. The fuel grain contains numerous axial combustion ports that generate fuel vapor to react with the injected oxidizer. An aft mixing chamber is employed to ensure that all fuel and oxidizer are burned before exiting the nozzle.

### 1.2.2 HYBRID ROCKET OXIDISER;

In high power rocketry and amateur rocketry by far the most common oxidizer and with rocket motors is nitrous oxide. Nitrous oxide is an oxidizing liquefied gas and is clear and colorless. It has a slightly sweet odor. At room temperature nitrous oxide is stable and inert. It is classified as a non-flammable gas. Nitrous oxide supports combustion and can detonate at temperatures in excess of 650°C.

- Nitrous oxide (N<sub>2</sub>O) –also known as dinitrogen monoxide NO<sub>x</sub> or laughing gas has a boiling point of -89.5 °C. at 1 atm and is normally maintained as a liquid at a pressure of 54 bar. Nitrous oxide has a molecular weight of 44.0 and a density of 1222 kg/m<sup>3</sup> at 20 °C. The theoretical and actual decomposition temperature of nitrous oxide is 7.27 MPa and 36.6 °C.

- Gaseous oxygen
- Hydrogen peroxide

Liquid oxygen (O<sub>2</sub>) –liquid oxygen has also chemical name AS LOX having boiling point -183°C. at 1 atm liquid oxygen has a molecular weight of 32.0 and a density of 1265 kg/m<sup>3</sup> at 20 degrees

- Nitrogen peroxide (N<sub>2</sub>O<sub>4</sub>)-nitrogen tetroxide has a boiling point of 21.2 degree Celsius and 1 atm nitrogen tetroxide has a molecular weight of 46.01 and a density of 1903 kg/m<sup>3</sup> at 20 degrees
- Nitric acid (HNO<sub>3</sub>)

## 1.6 HYBRID ROCKET MOTOR REGRESSION RATE;

In a hybrid motor, liquid oxidizer is fed into the combustion chamber from the oxidizer tank where it is ignited by an ignition source such as a pyrotechnic igniter. The fuel then ignites and burns in the presence of the oxidizer, where it vaporizes and burns along the length of the fuel grain. The rate at which the fuel burns is called regression rate and it is measured in meter per second (m/sec). The combination of fuel burn rate and oxidizer flow rate is called the mass flux and is measured as kilogram per meter squared second, the oxidizer flow rate affects the rate of regression of solid fuel and enables the following equation for solid fuel regression rate to be derived.

The regression rate is controlled by chamber pressure and follows the well established law of

$$r = a G^m \text{ Where,}$$

$r$  = fuel regression rate (m/sec)

$G$  = propellant mass flux ( $\text{Kg/m}^2\text{s}$ )  $X$  = length along the fuel grain port

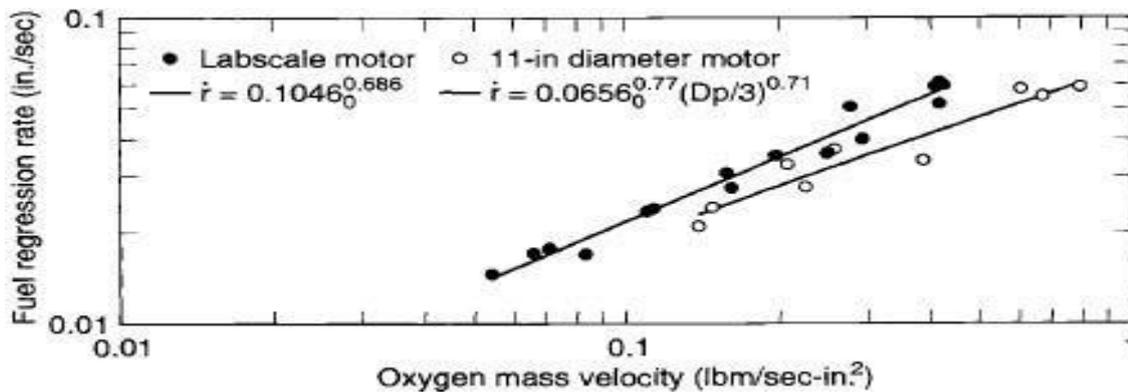
While  $a$  and  $m$  are empirical coefficients derived experimentally for specific propellant formulations. Since the rate of propellant gasification per unit area in a solid rocket motor, at a given propellant bulk temperature and in the absence of erosive burning, is determined only by chamber pressure, motor thrust is predetermined by the initial propellant grain surface area and grain geometrical characteristics. Throttling or extinguishment is very difficult to achieve in practical solid rocket motor configurations since the fuel and oxidizer cannot be separated. As the fuel grain of a hybrid typically contains no oxidizer, the combustion process and hence the regression of the fuel surface is markedly different from that of a solid rocket motor. Because the solid fuel must be vaporized before combustion can occur, the fuel surface regression is intrinsically related to the coupling of combustion port aerodynamics and heat transfer to the fuel grain surface. The primary combustion region over the fuel grain surface has been shown to be limited to a relatively narrow flame zone occurring within the fuel grain boundary layer. Factors affecting the development of the fuel grain boundary layer and, hence, fuel regression characteristics include pressure, gas temperature, grain composition, combustion port oxidizer mass flow rate, and combustion port length.

In hybrids with metallized fuel grains, radiation from the metal oxide particle cloud in the combustion port contributes a major portion of the total heat flux to the fuel grain.

The local regression rate of the fuel is also quite sensitive to the general turbulence level of the combustion port gas flow, localized combustion gas eddies or recirculation zones adjacent to the fuel surface act to significantly enhance the regression rate. Hybrid fuel regression rate is thought to be insensitive to fuel grain bulk temperatures over the range in which solid rocket motors may operate ( $-65^\circ\text{F}$  to  $165^\circ\text{F}$ ). This is due to the absence of heterogeneous fuel/oxidizer reactions at the fuel surface and because, over the above temperature range, the change in heat content of the solid fuel is small compared to the heat necessary to initiate vaporization of the fuel surface. Selection of fuel ingredients can also

have a significant impact on the grain regression rate, which is largely a function of the energy required to convert the fuel from solid to vapor phase. This energy is called the heat of gasification and, for polymeric fuels, includes the energy required to break polymer chains (heat of de polymerization) and the heat required to convert polymer fragments to gaseous phase. The term "heat of vaporization" is often used as a catchall phrase to include all decomposition mechanisms in hybrid fuels. In non-metallized fuels, low heats of gasification tend to produce higher regression rates.

Fuel is vaporized as a result of heat transferred from the flame zone to the fuel mass. Vaporized fuel is convected upward toward the flame zone while oxidizer from the free stream is transported to the flame zone by diffusion and flow turbulence. The flame is established at a location within the boundary layer determined by the stoichiometric conditions under which combustion can occur. The thickness of the flame is determined primarily by the rate at which the oxidation reaction occurs. This rate is largely dependent on pressure and typically follows an Arrhenius relationship. Hybrid regression rate has been observed to decrease as (combustion port diameter) increases.



Graph no: 1.7 Oxygen mass velocity vs. regression rate

### 1.15 PERFORMANCE ANALYSIS AND GRAIN CONFIGURATION;

A characteristic operating feature of hybrids is that the fuel regression rate is typically less than one-third that of composite solid rocket propellants. It is very difficult to obtain fuel regression rates comparable to propellant burn rates in solid rocket motors. Consequently, practical high-thrust hybrid motor designs must have multiple perforations (combustion ports) in the fuel grain to produce the required fuel surface area. The performance of a hybrid motor (defined in terms of delivered specific impulse) depends critically on the degree of flow mixing attained in the combustion chamber. High performance stems from high The solid propellant burning rate is controlled by chamber pressure and follows the well-established law of

$$\dot{r}^0 = a p_1^n$$

Where, a and n are empirical coefficients derived experimentally for specific propellant formulations. Since the rate of propellant gasification per unit area in a solid rocket motor, at a given propellant bulk temperature and in the absence of erosive burning, is determined only by chamber pressure, motor thrust is predetermined by the initial propellant grain surface area and grain geometrical characteristics. Throttling or extinguishment is very

difficult to achieve in practical solid rocket motor configurations since the fuel and oxidizer cannot be separated. As the fuel grain of a hybrid typically contains no oxidizer, the combustion process and hence the regression of the fuel surface is markedly different from that of a solid rocket motor. Because the solid fuel must be vaporized before combustion can occur, the fuel surface regression is intrinsically related to the coupling of combustion port aerodynamics and heat transfer to the fuel grain surface. The primary combustion region over the fuel grain surface has been shown to be limited to a relatively narrow flame zone occurring within the fuel grain boundary layer.

Factors affecting the development of the fuel grain boundary layer and, hence, fuel regression characteristics include pressure, gas temperature, grain composition, combustion port oxidizer mass flow rate, and combustion port length. The heat transfer relationships between the gas and solid phase depend on whether the boundary layer is laminar or turbulent. In a typical hybrid using oxygen as the oxidizer, the Reynolds number per unit length is on the order of  $1$  to  $2 \times 10^5$  per inch of grain length for flux levels between  $0.3$  and  $0.6$  lb. /sec/in. Thus, the properties of a turbulent boundary layer govern the convective heat transfer processes to non-metallized fuel grains. In hybrids with metallized fuel grains, radiation from the metal oxide particle cloud in the combustion port contributes a major portion of the total heat flux to the fuel grain.

The local regression rate of the fuel is also quite sensitive to the general turbulence level of the combustion port gas flow. Localized combustion gas eddies or recirculation zones adjacent to the fuel surface act to significantly enhance the regression rate. Hybrid fuel regression rate is thought to be insensitive to fuel grain bulk temperatures over the range in which solid rocket motors may operate ( $-65^\circ\text{F}$  to  $165^\circ\text{F}$ ). This is due to the absence of heterogeneous fuel/oxidizer reactions at the fuel surface and because, over the above temperature range, the change in heat content of the solid fuel is small compared to the heat necessary to initiate vaporization of the fuel surface.

Selection of fuel ingredients can also have a significant impact.

### **Methodology:**

Solid fuels include a rubberized material incorporating the oxidizer and other additives such as aluminum or ammonium perchlorate. The fuel-oxidizer composite is dangerous, as it may explode even during shipping and installation. The fuel burns very rapidly in the rocket combustion chamber to generate the rocket propulsion force known as thrust. Once the solid fuel is ignited, the rocket motor cannot be shut off. This is dangerous because there is no chance to ensure adequate thrust build-up before take-off. Although hybrids have been in development over the last 50 years, they have not made it into mainstream commercial applications because they did not produce as much thrust as liquid and solid systems. "Hybrid rockets tend to be sort of anemic in their ability to produce thrust," Cantwell said. This is because the fuel burns too slowly, relying on a process limited by the rate at which fuel evaporates and mixes with oxidizer. By contrast, the fuel and oxidizer are forced together in liquid systems and pre-mixed in solid systems.

## 2.1 C25 H52(n-Pentacosane);

n-pentacosane is one of the paraffin wax family .it has the n value 25 .we using this as the solid fuel in our hybrid rocket motor testing project .now we will discuss about its properties.

Table No 2.1 properties of n-pentacosane

A	7.478 A <sup>0</sup>
B	67.410 A <sup>0</sup>
C	4.970 A <sup>0</sup>
A	90.00 <sup>0</sup>
B	90.00 <sup>0</sup>
Γ	90.00 <sup>0</sup>
Volume	2505.34 A <sup>0</sup>
Temperature	22 <sup>0</sup> c
Common name	n-pentacosane
Formula weight	352.688 g/mole

### 2.1.1 LAYMAN'EXPLANATION;

Pentacosane is a circular hydrocarbon found in most insect .it is contact pheromone produce by female insects.it is part of an attractive blend in *orgyialeucostigma* and a defensive secretive gland in *rhinotermitidae*. *Capparidaceae*, a plant with insecticidal properties, contains 9 % pentacosane.

### 2.1.1 PHYSICALPROPERTIES

Table No 2.2 Liquid Oxygen Properties

PROPERTIES	VALUE
Molecular Weight	32
Boiling Point @ 1 atm	-297°F (-183°C)
Melting Point @ 1 atm	-362°F (-219°C)
Critical Temperature –	182°F (-118C)

Critical Pressure 729 psia	(50 atm)
Density, Gas @ 68°F (20°C), 1 atm	0.08 lb/scf
Specific Volume @ 68°F (20°C), 1 atm	12.08 scf/lb
Density, Liquid, @ Boiling Point, 1 atm	71 lb/scf
Heat of Vaporization	92 Btu/lb
Expansion Ratio, Liquid to Gas, BP to	68°F (20°C) 1 to 861

### 3.1 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  $T_0 = 3000\text{k}$
- We have to 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:

$$A_2/A_1 = (M_1/M_2) \left( \frac{1 + (\gamma - 1)/2 \times M_2^2}{1 + (\gamma - 1)/2 \times M_1^2} \right)^{3.833}$$

$M_1 = 0.2$

1

- Temperature at Nozzleinlet

$$T_0/T_1 = 1 + (\gamma - 1)/2 \times M_1^2$$

$$T_1 = 2980.32k$$

- Pressure at Nozzleinlet

$$P_0/P_1 = 1 + (\gamma - 1)/2 \times M_1^2$$

$$P_1 = 9.270bar$$

- Density at Nozzleinlet

$$\rho = p/(RT),$$

$$\rho_1 = 1.420Kg/m^3$$

- Velocity and free stream air velocity at Nozzleinlet

$$M = v/a$$

$$a = (\gamma RT)^{0.5}$$

$$v_1 = 99.824m/sec$$

- Mass flow rate atinlet

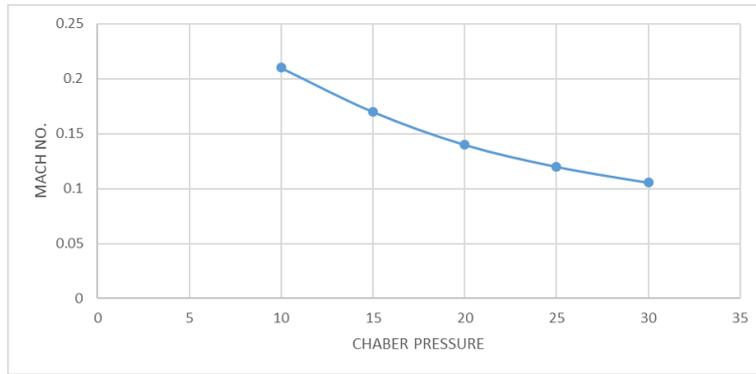
$$M^* = \rho AV$$

$$m^* = 22 Gm./sec$$

**Tableno:3.1 (Pressure Variation at Nozzle inletcondition)**

Variable stagnation pressure (N/m <sup>2</sup> )x10 <sup>5</sup>	Stagnation temp(k)	Nozzl e exit Mach	Nozzl e inlet Mach	Pressur e (N/m <sup>2</sup> )	Temp (K)	Density (Kg/m <sup>2</sup> )	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

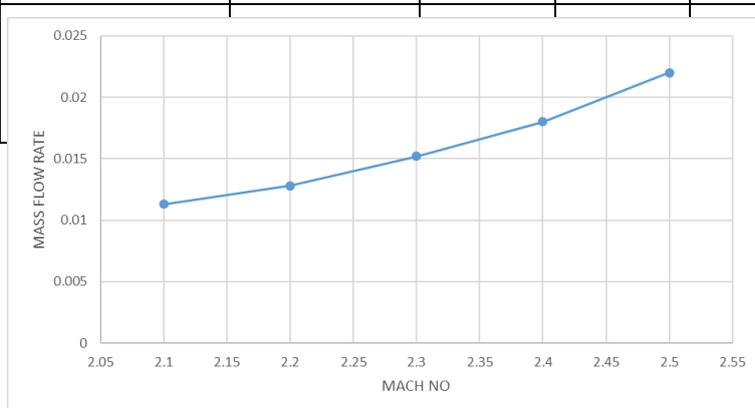
15	3000	2.4	0.170 0	14.723	2987.0 6	2.149	0.02707
10	3000	2.5	0.210 0	9.720	2980.3 2	1.420	0.0220



**Graph no: 3.1 Chamber pressure vs. inlet Mach no**

**Tableno:3.2 (At constant pressure 10bar variation of Machnumber)**

Stagnation pressure (N/mm <sup>2</sup> )x10 <sup>5</sup>	Stagnation temp(K)	Nozzle exit Mach no	Nozzle inlet Mach no	Pressure (1x10 <sup>5</sup> ) N/m <sup>2</sup>	Tempe(k)	Density (kg/m <sup>3</sup> )	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
					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 no: 3.2 Mach no vs. mass flow rate

### 3.2 THROATCONDITION;

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 Machno=0.21  
 Inlet temperature=2980.32k Inlet pressure=9.720 x10<sup>5</sup>N/mm<sup>2</sup> Inlet density=1.420kg/mm  
 flow rate =22gm/sec
- Now we to calculate
- For finding Temperature at nozzle throat  $(T_0/T_t) = 1 + (\gamma - 1)/2 \times M^2$   $T_t = 2608k$
- From temperature, pressure & density relation  $(T_0/T_t)^{\gamma/\gamma-1} = (\rho/\rho_t)^\gamma = P/P_t$
- For finding pressure nozzle throat

$$(T_0/T_t)^{\gamma/\gamma-1} = P/P_t \quad P_t = 5.464bar$$

- For finding Density at nozzle throat

$$(P/\rho_t)^\gamma = P/P_t \quad \rho_t = 0.912Kg/m^3$$

- For finding out throat diameter

$$a = (\gamma RT)^{0.5} \quad M = V/a \quad \text{where } M = 1$$

### 3.3 NOZZLE EXITCONDITION

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 Machno=1 Exit Machno=2.5  
 Throat diameter=14mm Throat pressure=5.464bar  
 Throat temperature=2698.695K  
 density=0.912kg/m<sup>3</sup>

- Using area ratio formula

$$A_e/A_t = (m_t/m_e) \times \left( \frac{(1 + (\gamma-1)0.5M^2)}{(1 + (\gamma-1)0.5M_e^2)} \right)^{3.833} \quad D_e=28\text{mm}$$

- Exit temperature

$$T_0/T_e = 1 + 0.15M_e^2 \quad T_e = 1548.38\text{k}$$

- Exit pressure

$$P_0/P_e = (1 + 0.15M_e^2)^{4.33} \quad P_e = 0.57\text{bar}$$

- Exit density

$$(P_0/\rho_e)^\gamma = (P_0/p_e) \quad \rho_e = 0.1604\text{Kg/m}^3$$

- Exit velocity calculation

$$M = V/a \quad a = (\gamma RT)^{0.5} \quad V_e = 1698.78\text{m/sec}$$

- Mass flow rate calculation

$$M^{\circ}/A^* = (P_0/T_0) \times \left( \frac{\sqrt{\gamma/R}}{2(\gamma+1)} \right)^{3.83} \quad M^{\circ} = 123.31\text{Gm/Sec}$$

- Thrust calculation

$$F = M^{\circ} V_e \quad F = 208.73\text{N}$$

- Discharge coefficient calculation

$$C_F = F/A_t \times P = 1.356 \quad \text{where } p \text{ denotes chamber pressure}$$

### 3.4 NOZZLE LENGTH CALCULATION;

Where Area ratio  $A_t/A_e = 0.25$

#### 3.4.1 CONVERGENT SECTION;

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

#### 3.4.2 DIVERGENT SECTION;

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

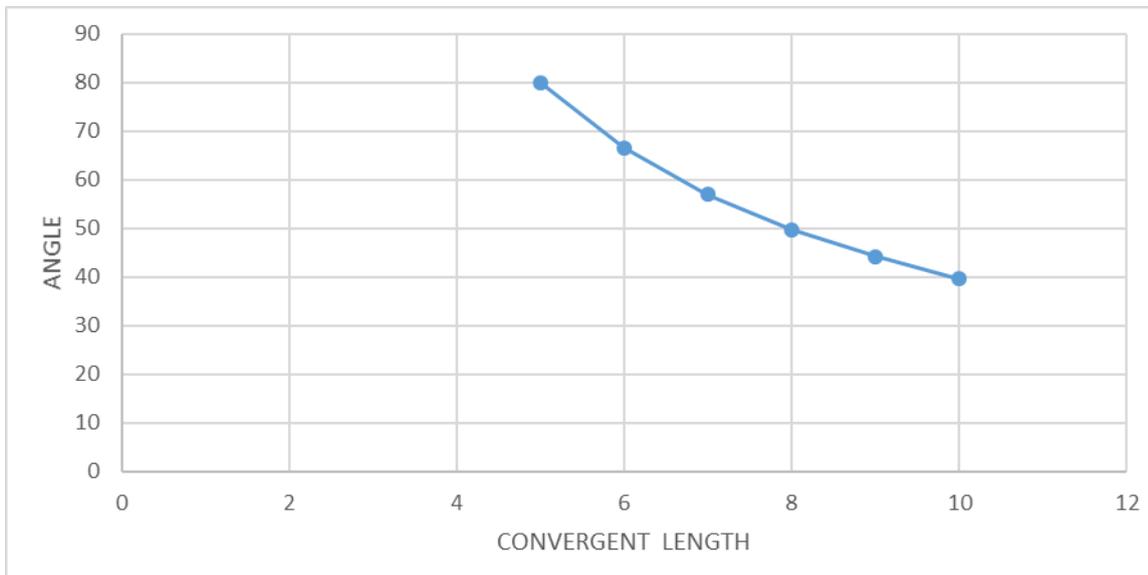
ANGLE( $\theta^\circ$ )	LENGTH(mm)
5	80.01
6	66.60
7	57.01
8	49.80
9	44.19
10	39.69

ANGLE( $\theta^\circ$ )	LENGTH(mm)
45	21
46	20.28
47	19.59
48	18.90
49	18.25
50	17.62

Convergent	Convergent
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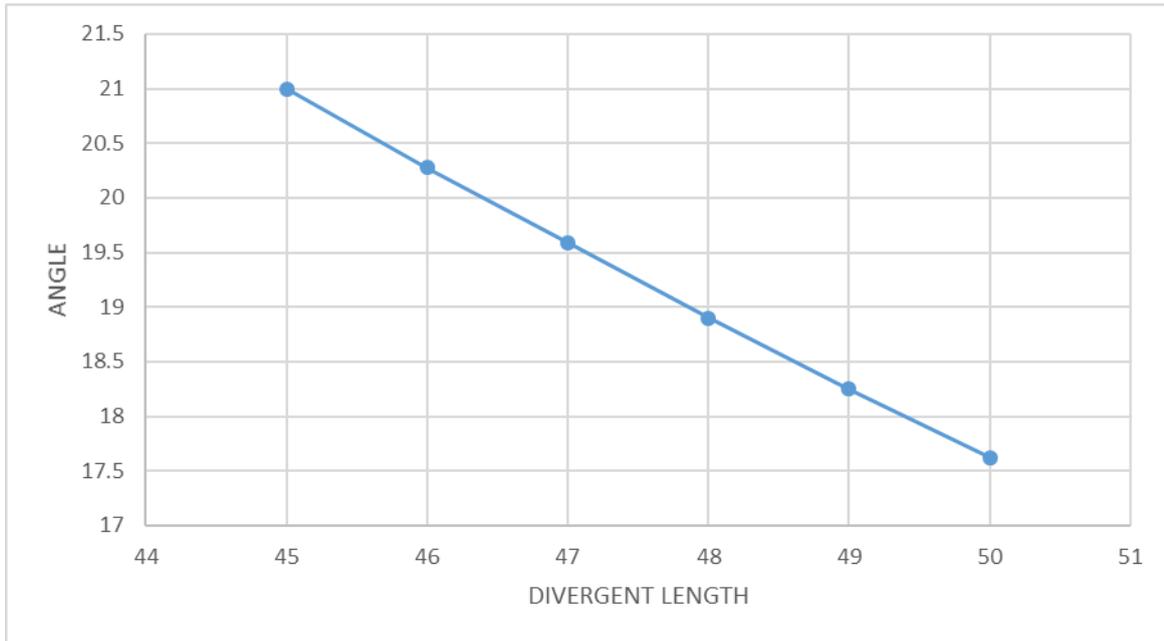
Table no 3.4.1 convergent section  
**For convergent section**

table no 3.4.2 divergent section



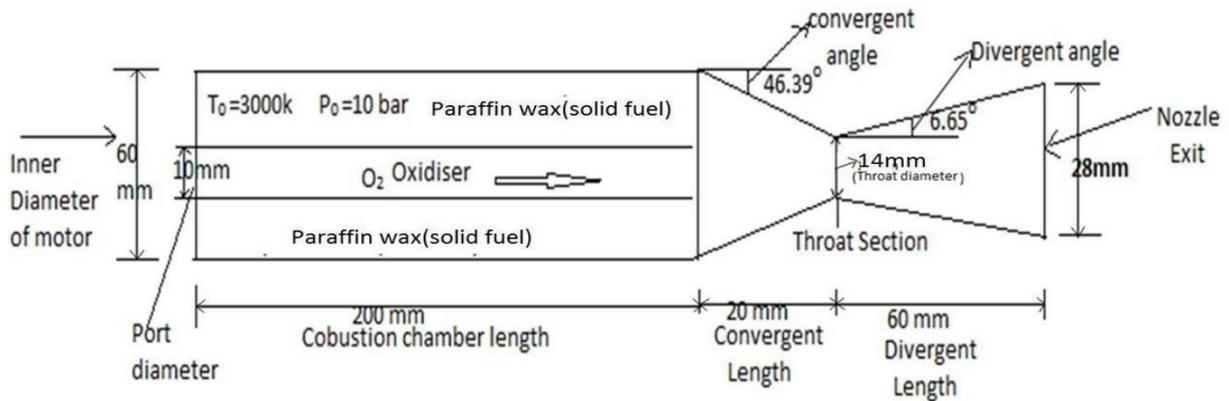
Graph no: 3.4.1 (convergent length Vs. Angle)

**For divergent shape:**



**Graph no: 3.4.2 Divergent length VS Angle**

### 3.5 2D ROCKET MOTOR DESIGN;



**Figure no 3.5 2d Rocket model**

### 3.6 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.

#### 3.6.1 CHAMBER;

The maximum yield stress is 200 Mpa

Chamber pressure=10bar

inner diameter of motor = 60 mm  $\sigma_{max}$

$$= (Pd / 2t)$$

$$t = 0.149 \text{ mm}$$

So thickness of chamber (t) = 0.149 mm should not be less than of 0.149 mm. For the safety purpose we take t=10mm, then the allowable maximum pressure Inthechamberiscalculatedasfollows, $\sigma_{max}=(Pd/2t)$ or, $200 \times 10^6=(px60/2 \times 10) = 66.6$  Mpa so the maximum allowable pressure in the combustion chamber is 80 Mpa .but in our case the maximum pressure in combustion chamber would not exceed this range, hence no safety problem .

### 3.6.2 C-DNOZZLE;

$$\text{At inlet} \quad \sigma_{max} = (Pd / 2t)$$

$$= 0.149$$

So thickness should not be less than 0.149 at inlet of cd nozzle .for the additional safety and attachment of cd nozzle with the chamber 40 mm thickness is used.

At the exit  $D_e = 28$  mm

$$\sigma_{max} = (Pd / 2t)$$

$$200 \times 10^6 = (60 \times 28) / (2 \times t) t$$

$$= 0.042 \text{ mm}$$

For the additional safety and need to attach the cd nozzle attach to the chamber here we used the thickness is 22 mm.

### 3.7 FUEL TO AIR RATIOCALCULATION;

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

(1) Paraffin wax( $C_{25}H_{52}$ )---fuel

(2)  $O_2$ (liquidoxygen) -oxidizer

The general chemical reaction in the combustion chamber as follow  $C_{25} H_{52} + 38 O_2 = 25 CO_2 + 26 H_2O$

Now ration of (oxidizer/fuel)  $stio = (38 \times 32) / (25 \times 12 + 52 \times 1) = 3.45$   $O_2$  mass flow rate available (steady rate) in market =  $3200 \times 10^{-3} \text{ m}^3/\text{min}$

Density  $O_2 = 1.429 \text{ kg/m}^3$

$$= 3200 \times 1.429 \text{ kg/sec}$$

$$= 0.076 \text{ kg/sec For that mass flow rate}$$

$$(Mf)_{stio} = 0.076 / 3.45 = 0.022 \text{ kg/sec}$$

And we are all know that

$$\frac{(\text{Fuel} / \text{oxidizer})_{\text{act}}}{(\text{fuel} / \text{oxidizer})_{\text{stio}}} = 0.8 (\text{Fuel} / \text{oxidizer})_{\text{act}} = (0.8 / 3.45) = 0.231$$

### **HEAT VALUE CALCULATION Heat of combustion;**

The heat of combustion is the energy released as heat when a compound undergoes complete combustion with oxygen under standard conditions. The chemical reaction is typically a hydrocarbon reacting with oxygen to form carbon dioxide, water and heat. It may be expressed with the quantities:

energy/mole of fuel (kJ/mole)

### **Heating value;**

The heating value (or energy value or calorific value) of a substance, usually a fuel or food (see food energy), is the amount of heat released during the combustion of a specified amount of it. The energy value is a characteristic for each substance. It is measured in units of energy per unit of the substance, usually mass, such as: kJ/kg, kJ/mole, kcal/kg, Btu/lb. Heating value is commonly determined by use of a bomb calorimeter.

### **Standard enthalpy of formation;**

The standard enthalpy of formation or standard heat of formation of a compound is the change of enthalpy that accompanies the formation of 1 mole of the compound from its elements, with all substances in their standard states.

Its symbol is  $\Delta H_f^\circ$  or  $\Delta_f H^\circ$ . The super script theta(0) on this symbol indicates that the process has been carried out under standard conditions.

Standard States are as follows: For gases: standard state is a pressure of exactly 1 bar For a substance present in a solution: a concentration of exactly 1 M at a pressure of 1 bar For a pure substance in a condensed state (a liquid or a solid): the pure liquid or solid under a pressure of 1 bar For an element: the form in which the element is most stable under 1 bar of pressure and the specified temperature. (Usually 25 degrees Celsius or 298.15 K) One exception is phosphorus: most stable under 1 bar is black phosphorus, but white phosphorus is used as the reference for zero enthalpy of formation For example, the standard enthalpy of formation of carbon dioxide would be the enthalpy of the following reaction under the conditions above:



Note that all elements are written in their standard states, and one mole of product is formed. This is true for all enthalpies of formation.

The standard enthalpy of formation is measured in units of energy per amount of substance. Most are defined in kilojoules per mole ( $\text{kJ mol}^{-1}$ ), but can also be measured in calories per mole, joules per mole or kilocalories per gram (any combination of these units conforming to the energy per mass or amount guideline). In physics the energy per particle is often expressed in electron volts which corresponds to about  $100 \text{ kJ mol}^{-1}$ .

All elements in their standard states (oxygen gas, solid carbon in the form of graphite, etc.) have a standard enthalpy of formation of zero, as there is no change involved in their formation.

### Standard enthalpy of reaction;

The standard enthalpy of formation is used in thermo chemistry to find the standard enthalpy change of reaction. This is done by subtracting the sum of the standard enthalpies of formation of the reactants (each being multiplied by its respective stoichiometric coefficient,  $v$ ) from the sum of the standard enthalpies of formation of the products (each also multiplied by its respective stoichiometric coefficient), as shown in the equation below:

$\Delta H^\circ = \sum (v \times \Delta H_f^\circ) (\text{products}) - \sum (v \times \Delta H_f^\circ) (\text{reactants})$  For example, for the reaction  $\text{CH}_4 + 2 \text{O}_2 \rightarrow \text{CO}_2 + 2\text{H}_2\text{O}$ :

$$\Delta H_r^\circ = [(1 \times \Delta H_f^\circ (\text{CO}_2)) + (2 \times \Delta H_f^\circ (\text{H}_2\text{O}))] (\text{products}) - [(1 \times \Delta H_f^\circ (\text{CH}_4)) + (2 \times \Delta H_f^\circ (\text{O}_2))] (\text{reactants})$$

If the standard enthalpy of the products is less than the standard enthalpy of the reactants, the standard enthalpy of reaction will be negative. This implies that the reaction is exothermic. The converse is also true; the standard enthalpy of reaction will be positive for an endothermic reaction.

### Key concepts for doing enthalpy calculations;

When a reaction is reversed, the magnitude of  $\Delta H$  stays the same, but the sign changes. When the balanced equation for a reaction is multiplied by an integer, the corresponding value of  $\Delta H$  must be multiplied by that integer as well. The change in enthalpy for a reaction can be calculated from the enthalpies of formation of the reactants and the products. Elements in their standard states are not included in the enthalpy calculations for these reactions since the enthalpy of an element in its standard state is zero.

**Entropy** is a mathematically-defined thermodynamic quantity that helps to account for the flow of energy through a thermodynamic process. Entropy was originally defined for a thermodynamically reversible process as

**Enthalpy** is a measure of the total energy of a thermodynamic system. It includes the internal energy, which is the energy required to create a system, and the amount of energy required to make room for it by displacing its environment and establishing its volume and pressure.

Enthalpy is a thermodynamic potential. It is a state function and an extensive quantity. **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. Since enthalpy is derived from pressure, volume, and internal energy, all of which are state functions, enthalpy is also a state function.

Paraffin wax is  $\text{C}_{25}\text{H}_{52}$

Molecular weight = 352 gm. /mole 1 mole = 25 mole carbon and 26 mole  $\text{H}_2$  1 gm. paraffin = 0.0028 mole paraffin

1 molecule of carbon =  $0.071 \times 12 \text{ gm/mole}$  1 molecule of  $\text{H}_2 = 0.0732 \times 2 = 0.146 \text{ gms}$

Now heat of combustion of carbon = 32.8 mj/kg  $H_c = 32.8 \text{ mj/kg} \times 1000 \text{ gm.} = 0.0279 \text{ mj}$

$H_c = 27.9 \text{ Kjoules}$

Heat of combustion for  $\text{H}_2 = 121 \text{ mj/kg}$  Heat of combustion = 45.6 Kjoules

Change in enthalpy= enthalpy of product- enthalpy of reactant

### 3.8.1 COMBUSTION WITH OXYGEN;

Complete combustion of paraffin wax As  $C_{25}H_{52} + 38O_2 = 25CO_2 + 26H_2O$

Heat of combustion (Hc) =  $-1.605 \times 10^7$  kJoule Heat of reactant =  $\{N_i h_i\} = -1.605 \times 10^7$  kJoule

Heat of product =  $3.29 \times 10^7$  kJoule

Flame temperature =  $\{N_i (h_{f,i} + C_p (T_{ad} - 298))\} = T_{ad} = 4409$  k

### 3.8.2 COMBUSTION WITH AIR;

$C_{25}H_{52} + 38(O_2 + 3.67 N_2) = 25 CO_2 + 26 H_2O + 142.88 N_2$

Here combustion takes place with air that is mixture of nitrogen plus oxygen and after combustion gives some amount of nitrogen molecule in rocket combustion testing procedure.

Adiabatic flame temperature = 4037.33 k

Table no 3.8. Thermochemical property hydrocarbon

Species	Enthalpy of formation (Kjoules/mole)	Specific heat capacity (joule/kelvin)
CH <sub>4</sub>	-74.831	---
CO <sub>2</sub>	393.546	56.21
H <sub>2</sub> O	241.845	43.87
N <sub>2</sub>	0	33.71
O <sub>2</sub>	0	0

### 3.8 MOLAR MASS CALCULATION

Molar masses of chemical compounds are equal to the sums of the molar masses of all the atoms in one molecule of that compound.

#### COMPLETE COMBUSTION;

As chemical equation  $C_{22}H_{52} = 25CO_2 + 26H_2O$  Moles of combustion products as 26 mole H<sub>2</sub>O as well as 25 Mole CO<sub>2</sub>

Molar masses of the individual combustion products:

Molar mass ( $H_2O$ ) =  $(2 \times 1.0079) + (1 + 15.009) = 18.0148$  gm. /mole

Molar mass of ( $CO_2$ ) =  $(1 \times 12.011) + (2 \times 15.999) = 44.009$  gm. /mole Mass fraction of combustion products:

$F(H_2O) = 0.289$                        $F(CO_2) = 0.7132$

Molecular weight of paraffin wax = 36.24 gm. /mole Universal gas constant =  $R = 8.314/36.24 = 229.39$  j/kgk

### 3.9.2 INCOMPLETE COMBUSTION;

As chemical equation                       $C_{22}H_{52} + 33 O_2 = 20 CO_2 + 26 H_2O + 5C$

Here we can see that reaction takes place with incomplete because after combustion give some amount of carbon molecule which is remaining after burning with oxygen.

Mole of combustion products

26 mole  $H_2O$ , 20 Mole  $CO_2$  and 5 mole of carbon molecule Molar mass of individual combustion products

Molar mass of  $H_2O = 18.0148$  gm/mole

Molar mass of  $CO_2 = 44.01$  gm/mole, Molar mass of  $C = 12.011$  gm/mole Mass fraction of combustion products

$F(H_2O) = 0.332$ ,  $F(CO_2) = 0.624$  &  $F(C) = 0.042$

Molecular weight of paraffin wax = 33.946 gm. /mole Gas constant value =  $R = 245$  j/kgk

### 3.9 MASS OF PROPELLANT;

In a chemical rocket propellants undergo exothermic chemical reactions to produce hot gas. There may be a single propellant, or multiple propellants; in the latter case one can distinguish fuel and oxidizer. The gases produced expand and push on a nozzle, which accelerates them until they rush out of the back of the rocket at extremely high speed. In pulse propulsion, a heavy, metallic base acquires the force from an explosion behind it, for example from an atomic bomb, and transfers it to a dampening system that reduces the shock to the payload.

We can calculate propellant mass using simple formula  $M = \text{volume} \times \text{density}$

Volume Area of cross section  $\times$  length of combustion chamber

Here we are taken density as density of paraffin wax .while volume is chamber motor volume.

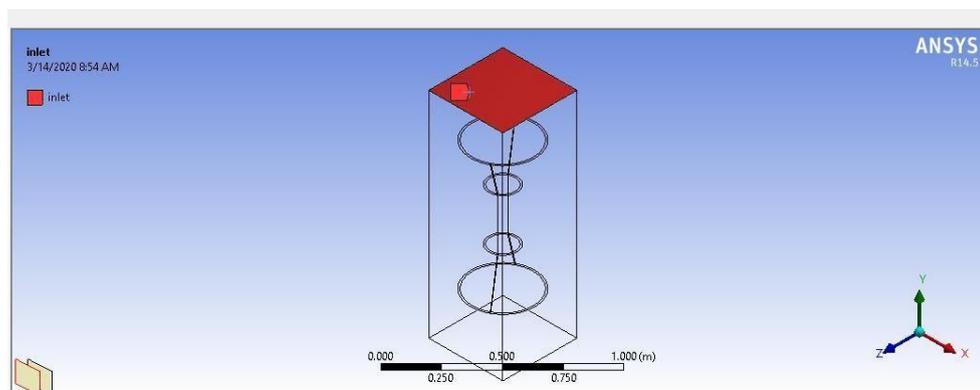
Area of the solid state fuel grain is

$$= \pi/4 \times (60^2 - 10^2) = 2748.89 \times 10^{-6} \text{ m}^2$$

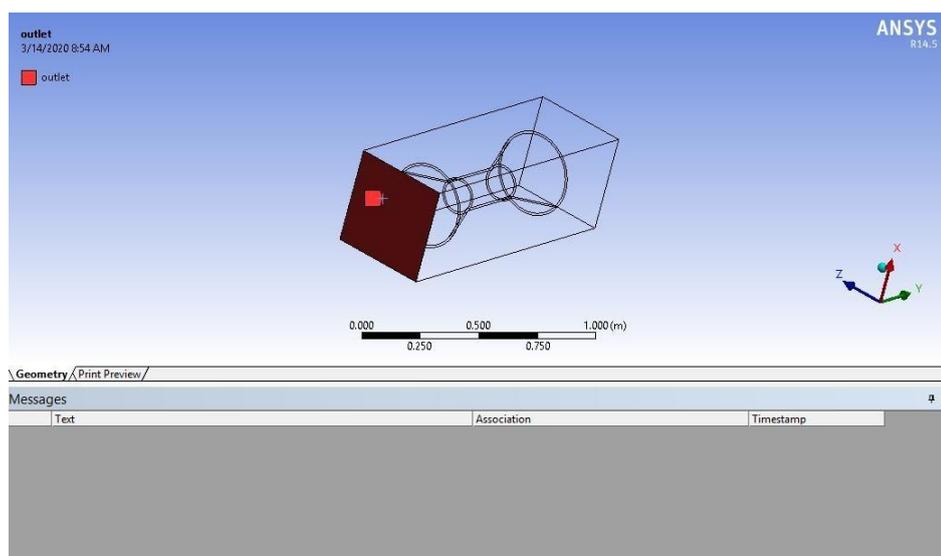
Density of the  $C_{25}H_{52}$  (n- pentacosane) =  $799 \text{ kg/m}^3$  We know that  $\rho = m/v$

$$m = \rho \times v = 799 \times (\text{Area} \times \text{Length}) = 439.29 \text{ gm.}$$

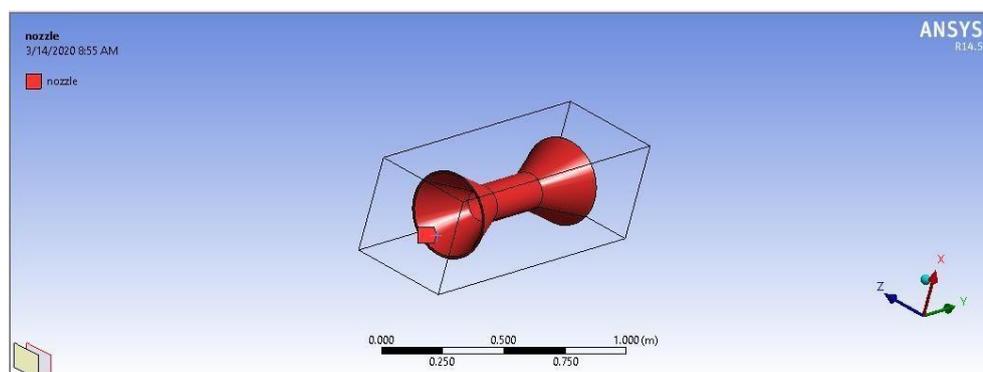
## ANALYSIS RESULTS; BOUNDARY CONDITION (INLET)



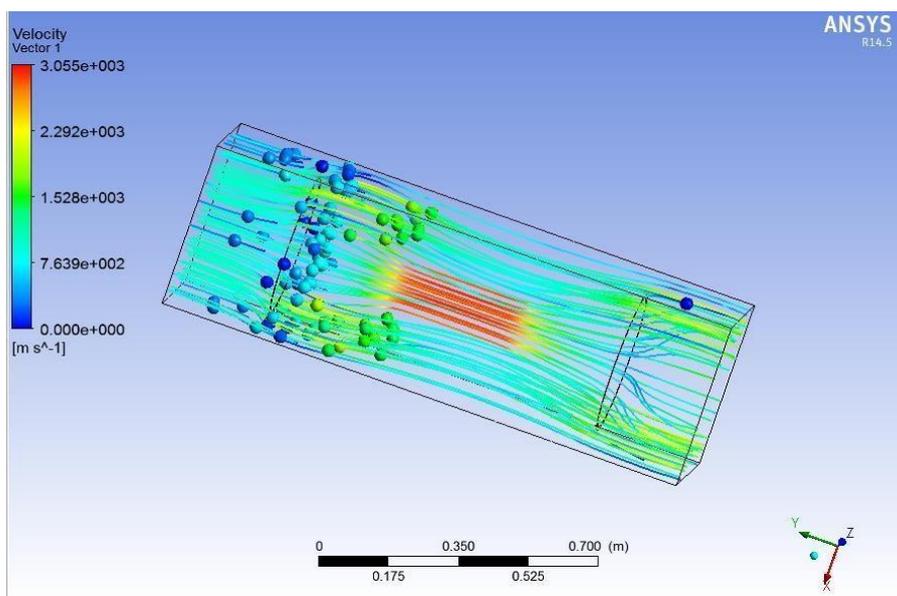
## BOUNDARY CONDITION (OUTLET)



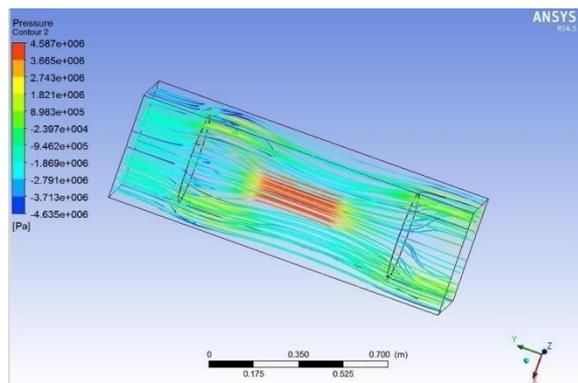
## BOUNDARY CONDITION (NOZZLE);



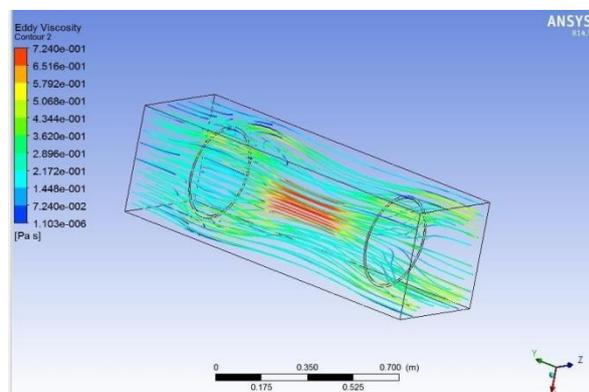
### ALUMINIUM (ANGLE 20) – VELOCITY



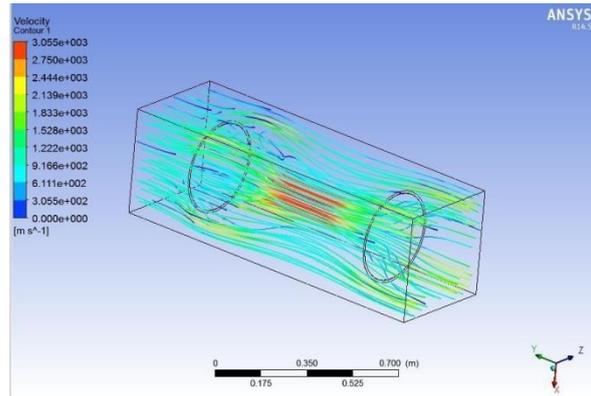
### ALUMINIUM (ANGLE 20) – PRESSURE



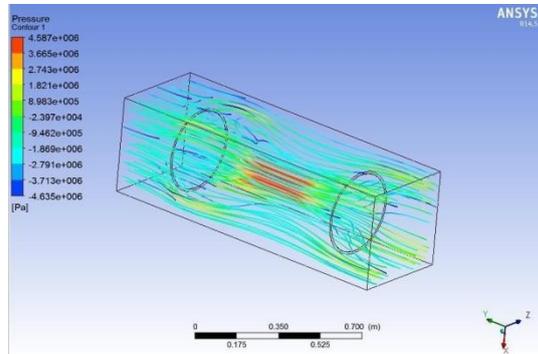
### ALUMINIUM (ANGLE 20) – EDDY VELOCITY



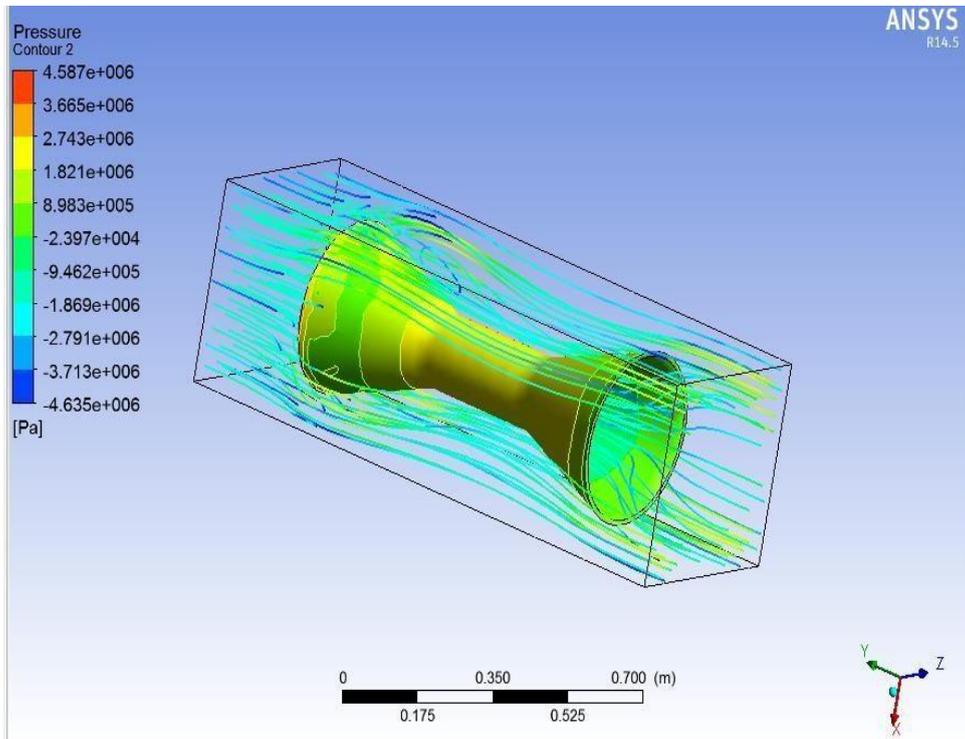
### STEEL (ANGLE 20) -VELOCITY



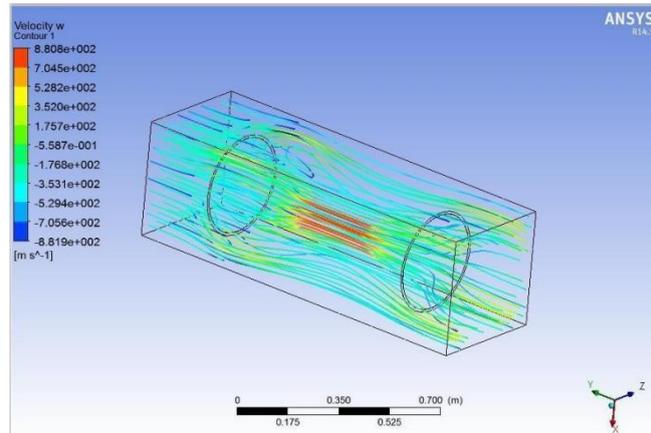
### STEEL (ANGLE 20) -PRESSURE



### TITANIUM (ANGLE 20) -VELOCITY

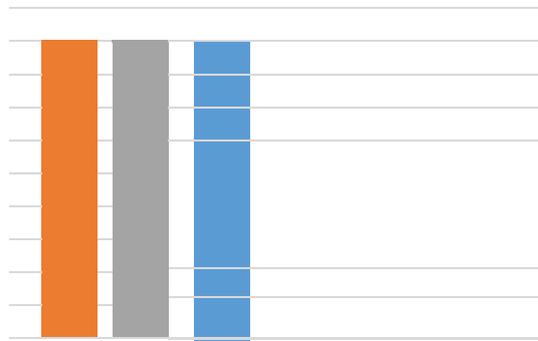


### TITANIUM (ANGLE 20) –PRESSURE

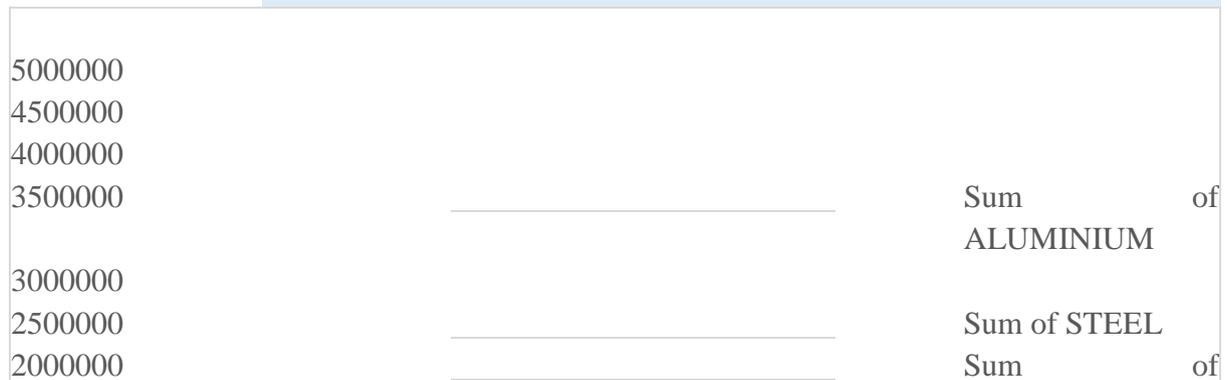


### COMPARISON OF RESULTS; COMPARISION OF ALUMINIUM, STEEL, AND TITANIUM IN ANGLE 20,25,30.

	Sum of Sum of Sum of			
ANGLE 20	ALUMINIUM	STEEL	TITANIUM	
PRESSURE	4587000	4587000	4587000	
VELOCITY		3055	305.5	880.88



<b>GrandTotal</b>	<b>4590055</b>	<b>4587305.5</b>	<b>4587880.</b>
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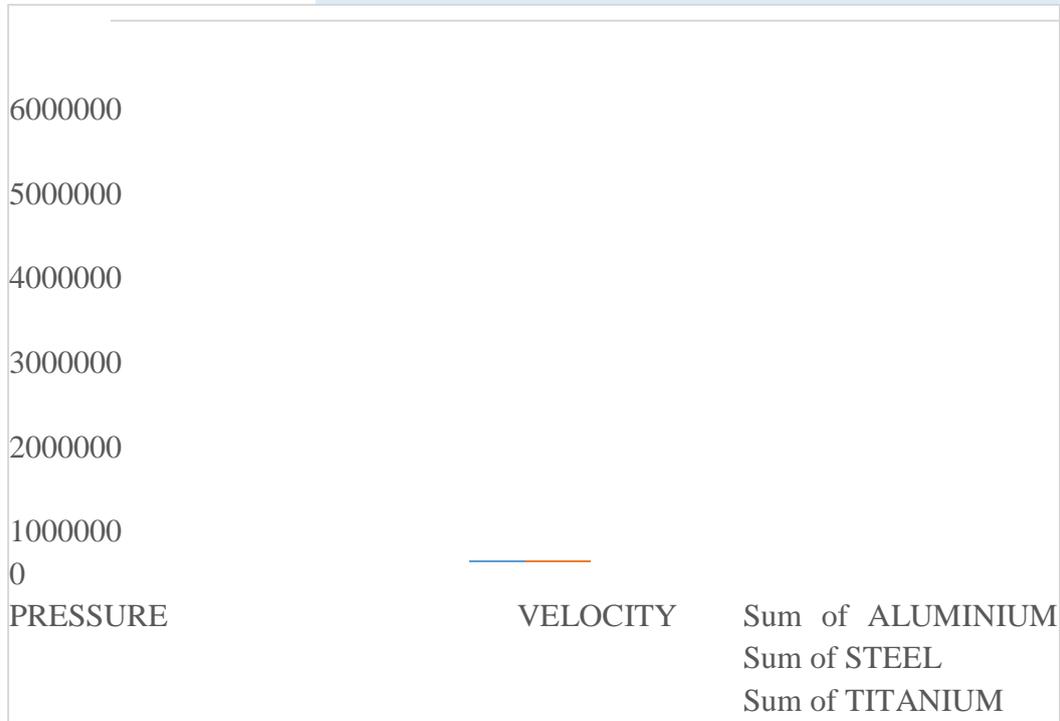




	Sum of ALUMINIUM	Sum of STEEL	Sum of TITANIUM
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PRESSURE 508000 508000 508000

VELOCITY	508000	850.3	2986	2986
<b>GrandTotal</b>	<b>508000</b>	<b>508000.3</b>	<b>5010986</b>	<b>5010986</b>



	Sum of ALUMINIUM	Sum of STEEL	Sum of TITANIUM
ANGLE 25	4691000	4987000	4691000
VELOCITY	3044	840.5	840.5
<b>Grand Total</b>	<b>4694044</b>	<b>4987840.5</b>	<b>4691840.5</b>

## 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 Aluminium.

## REFERENCES;

- [1] XuYueResearchProgressonAeroenginejetnoisereductionbymicrojet, *Aero. Sci. & Tech.*, (2): 52-54 (2011)
- [2] QiaoWeiyang, *Aeroacoustics of Flight Vehicle*. Beihang University Press, Beijing(2010).
- [3] Eric Zoppellari, Daniel Juve. Reduction of jet noise by water injection. AIAA-97-1622 (1997).
- [4] Thomas D Norum. Reduction in multi- component jet noise by water injection. AIAA-04-2976,(2004).
- [5] S.Sankaran,JopaulK.Ignatiusetc.Suppressionofhighmachnumberrocket jet noise by water injection. *Journal of spacecraft and rockets*, (6):1164- 1170 (2009).
- [6] AnjaneyuluKrothapalli,BRentonGreskaetc,Highspeedjetnoisereduction using microjets[R]. AIAA-02-2450(2002).