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Fabrication and Analysis of Nan Composites By Varying Composition of AL - Al_2O_3

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ABSTRACT: Nowadays, demands for developing metal matrix composites for use in high performance applications, have been significantly increased. Among these composites, aluminum alloy matrix composites attract much attention due to their lightness, high thermal conductivity, moderate casting temperature, etc. Various kinds of ceramic materials, e.g. SiC, Al2O3, MgO and B4C, are extensively used to reinforce aluminum alloy matrices.

In this project Aluminium Alloy (LM4) is reinforced with 1, 1.5, 2.5 and 5% of nano particle of Aluminium oxide (Al2O3) and fabricated by using stir casting method. The fabricated Aluminium oxide (Al-Al2O3) nanocomposite materials are tested using tensile and hardness testing machines to determine its Mechanical properties. And the results are compared for choosing the best composite .A microstructure test was also conducted to determine the best results. Finally the composites are analyzed using Ansys version 10.0 Software.

I. INTRODUCTION

1.1 OVERVIEW:

A Liquid rocket engines employ liquid propellants which are fed under pressure from tanks into a combustion chamber. In the combustion the propellants chemically react (burn) to form hot gases which are then accelerated and ejected at high velocity through a nozzle, there by imparting momentum to the engine. Momentum is the product of mass and velocity. A thrust force of a rocket motor is the reaction experienced by the motor structure due to ejection to the high velocity.
According to the Newton's third law:
"For every action there is an equal and opposite reaction"
Based upon this law only the rocket works

1.2 Nozzle:

The function of nozzle is to convert the chemical - thermal energy generated in the combustion chamber into kinetic energy. The nozzle converts the slow moving, high pressure, high temperature gas in the combustion chamber into high velocity gas of low pressure and temperature. The nozzle is usually made a long enough (or the exit area is greater enough) such that the pressure in the combustion is reduced at the nozzle exit to the pressure existing outside the nozzle. If the rocket engine is being fired at sea level this pressure is about 14.7 Psi. If the engine is designed for the operation at high altitude the exit pressure is less than 14.7psi. The drop in the temperature of the combustion gases flowing through nozzles is high and can be as much as 2000-3000F. Since the gases in the combustion chamber may be at 5000-6000F. The gas temperature at the nozzle exit is still about 3000F.

1.3 DE-LAVAL (CONVERGENT &DIVERGENT):

A Rocket engine nozzle is a propelling nozzle usually of the DE-LAVAL type used in a rocket engine to expand and accelerate the combustion gases, from the burning propellants, so that the exhaust gases exit the nozzle at the supersonic velocities.

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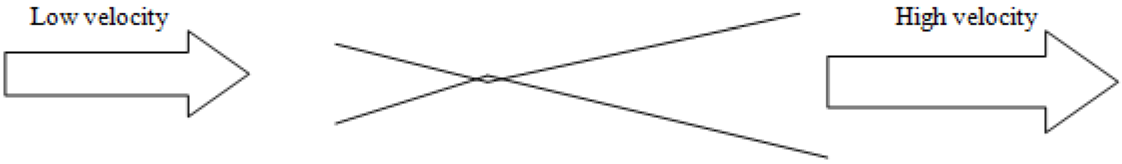

Fig: 1

As the above fig is mentioned that low pressure gas is enters into a divergent portion and leaves at in high velocity at divergent section of nozzle.

DIFFERENT TYPES OF NOZZLE:

1. **CONE (15° half angle)**
2. **CONTOURED OR BELL-FULL LENGTH NOZZLE**
3. **CONTOURED OR BELL SHORTENED**
4. **PLUG OR AEROSPIKE**
5. **EXPANSION DEFLECTION**

In that the contour (full length) is used for many operations. It has the better performance over all other nozzle. The plug nozzle and expansion deflection nozzles are under research conditions. And also it was high cost compare to contour nozzle.

1.4 PROPELLANT CHOICE:

Liquid rocket engine can burn variety of oxidizer and fuel combustion, some of which are tabulated Table I. Most of the propellants of the propellant combination listed are dangerous, toxic, and expensive. The amateur builder of rocket engines on the other hand, requires propellants that are readily available, reasonably safe and easy to handle, and inexpensive. The use of gaseous oxygen as the oxidizer and a hydrocarbon liquid as the fuel. They have good performance, the combustion flame is readily visible, and their combustion temperature presents an adequate design challenge to the amateur.

Gaseous oxygen can be readily and inexpensively obtained in pressurized cylinder in almost any community because of its use in oxy-acetylene welding. With reasonable precautions, to be detailed later, the gas (and cylinder) is safe to handle for test stand use. Gas pressures are easily regulated with commercial regulators and gas flow rate is easily controlled with commercially available valves.

Table: I

Propellant combination Oxidizer/fuel	Combustion pressure, Psi	Mixture ratio	Flame temp °F	Isp, sec
Liquid oxygen & gasoline	300	2.5	5470	242
Gaseous oxygen & gasoline	300	2.5	5742	261
Gaseous oxygen & gasoline	500	2.5	5862	279
Liquid oxygen & JP-4	500	2.2	5880	255
Liquid oxygen &methyl alcohol	300	1.25	5180	238
Gaseous oxygen & methyl alcohol	300	1.2	5220	248
Liquid oxygen & hydrogen	500	3.5	4500	363

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From the tables we can observe that Liquid oxygen & hydrogen is with standard high temperature and pressure.

II. LITERATURE REVIEW

1. **Scott Forde** .et.al. Has discussed that larger area ratio nozzle produces less thrust. The main chamber should be operated at constant pressure. By adding TAN specific-pumps would separate engine and TAN development. And most important benefit of TAN is increases engine system reliabilities resulting from operating the engine at chamber Pressure below the structural Design condition. And carried a two –hot fire tests on the TAN concept. The first tests were performed with single propellant injection into a 2000-lbf-thrust GOX/gases hydrogen rocket engine. Over 40% increase in sea-level thrust using a 25:1 expansion area ratio nozzle. The second tests carried by adding a water-cooled chamber. The TAN injector at 2:1 into a water-cooled nozzle [1].

2. **Taro Shimizu.et.al.** This study says that how the flow structures inside and outside a rocket nozzle. Due to over pressure which originates in shock waves, and improves high pressure load on the nozzle (or) rocket surface. Under clustering nozzle configuration, the interaction between the nozzles. As the NPR increases the overpressure arrives at the nozzle tip decreasing its amplitude. The function positive (or) negative based upon the pressure increases (or) decreases along the stream line. Due to clustering of nozzles the pressure disturbance induced by the neighbouring nozzle is 10% of the atmosphere pressure. The transition of the flow structure between free shock separation and restricted shock separation inside a nozzle, which would sometimes generate a destructive side-load[2]

3. **Macro Geron, et.al.** This work investigated the three –dimensional flow generated on this partitioning the primary nozzle into modules. A linear plug nozzle has been together with modules having two different geometries: rectangular cross section and Round –to-square module. The flow field of the test case R2S appears richer in the three dimensional effects than the test case REC. losses are happened due to the adjacent modules. The viscous stresses are high at nozzle exit. The gap is increased the plug contribution affected less. The partitioning of the primary nozzle with round-to-square modules causes a thrust reduction with respect to the two-dimensional model which can exceed1%. This paper says that to improve module efficiency and to reduce the maximum mechanical and thermal stresses[3]

4. **Lunciano Garelli et.al.** He analyzed the aero elastic process developed during the starting phase of a rocket engine. Better understanding of the behaviour of the structures as the shock waves propagate inside the engine nozzle. During the start up-phase the structure is deformed due to the advance shock wave that is highly detrimental to service life cycle of the rocket engine. Design of a nozzle it is important to predict the response of the structure under the thrust loads[4]

5. **Guobiao Cai, et.al.** He was optimized the liquid rocket engine nozzle using the CFD method. For solving the two-dimensional axis symmetric Navier- strokes equation (NS) and transport equations in an efficient number. this optimization is generates nearly 1.5% performance improvement over the initial one, approximately half of which is attributed to the decrease of friction loss, and the other half to the decrease of axial loss[5]

III. METHODOLOGY

The nozzle is important factor for rocket propulsion systems which exhaust gases at high velocity. So changing of the parameter it should carefully selected for this system.

3.1 ROCKET EQUATION:

$$F = m_a V_e + (P_e - P_a) A_e$$

Where,

Ma $\longrightarrow$ ***mass flow rate of the propellant***
Ve $\longrightarrow$ ***velocity at the exit condition***
Pe $\longrightarrow$ ***pressure at the exit condition***
Pa $\longrightarrow$ ***pressure at ambient condition***

Ae $\longrightarrow$ ***Area at exit condition***

In the above equation says that the Thrust can be increased by mass flow rate of the propellant (*Ma)* (or) increasing the exit velocity (Ve) (or) increases the exit area (Ae).

Here the mass flow rate depends upon the propellant mixture ratio and also it will produce a large thermal stresses on the walls of the nozzle. And the velocity is depends upon the pressure condition. And by increasing the area we can improve the Thrust of the rocket engine.

Area be modified by changing the exit dia of the nozzle with the exit Mach.no.and profile curve of the nozzle. Based upon the streamlines and the co-ordinate points of the nozzle profile curve may be obtained.

The best method is for the nozzle contour shape it gives the best performance

3.2 AN ACCURATE AND RAPID METHOD FOR THE DESIGN OF SUPERSONIC NOZZLE OVERVIEW:

This method is simple in all other methods. It gives the accurate and easy calculation of the nozzle contour from the derived equations of the stream lines flow region.

A procedure is given for designing two-dimensional nozzles in which the streamline co-ordinates are compared directly from tabulated flow parameters and appropriate equations. The method of characteristics is used to obtain the first part of flow which consist of a continuous expansion from a uniform sonic flow to a radial flow, the Folesch equation are then used for the transition from radial flow to the final uniform flow. Information is presented which enables the design to select and compute radially the wall contour for any nozzle (or) series of nozzle for a wide range of length-to-height ratio, Mach number and wall angle at the inflection point.

A nozzle is determined specifying by any two of these parameters

1. Length-To-height ratio.
2. MACH:NO:
3. Wall angle at the inflection point

IV. REGION OF NOZZLE

4.1 1. REGION I: (sub sonic region)

The calculation of the subsonic portion of a nozzle is simplified by the straight sonic line. A straight sonic line normal to the axis is always obtained when the velocity gradient along the x-axis vanishes at the sonic point.

$$Y= C\ (1+.1924X^6)$$

Where, c- constant.

2. REGION II: (THE INITIAL EXPANSION)

The computation for region II is based on the boundary conditions of a uniform parallel flow along sonic line with Y-axis.

3. REGION III: (THE SECONDARY EXPANSION)

The flow in region III was computed by the method characteristics by using the initial conditions which result in a continuous curvature and velocity gradient along the all stream lines between the sonic line and the Mach, line.

4. REGION IV: (THE RADIAL FLOW)

A plane radial flow (or source flow) is defined as out in which all dependent variables are a function only the radius from a fixed point in the plane. It is the same as the well-known area ratio equations in a one-dimensional flow. The stream lines within the radial flow are straight lines, which are extended, would get from the original of the radial flow.

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5. REGION V: (THE FINAL TRANSITION)

It is bounded by the radial flow Mach, line and the straight Mach, line. The co-ordinates for any stream line within this flow are by the equations.

V. MODEL CALULATION

5.1 DESIGN CALCULATION:

For design the nozzle standard conditions are to be assumed. Because of the test is made at in the sea-level condition. So there should be no vacuum. The pressure is at in ambient pressure.

1. Chamber pressure, Pc=1000**Psi** (or) $6.894*10^6$ **Pa**. at in the ground level the ignition temperature of the propellant may be taken.
2. Exit pressure , P_e
3. Ambient pressure, P_a=$1.01325*10^5$ **Pa** at in atmospheric pressure.
4. The chamber temperature T_c=2500k. This temperature is taken from the propellant Liquid oxygen & Liquid hydrogen. The temperature is 4500°F (Table: I)

Where Pe=Pa at in the sea level condition.

All the parameters are to consider when the nozzle Is to be designed it is suitable to get the values for exit Mach, no. conditions. As the temperature is must for this condition because it should at in high temperature. The material should be ability withstand at high temperature at nozzle walls.

The following values property is based upon the propellant mixture and initial values of the static thrust condition.

The formulas used for the obtaining these values are derived from the rocket propulsion system.

1. PRESSURE RATIO:

The pressure is ratio of the combustion pressure to the ambient pressure.

Pc/Pa = 68.038

Approximate value of ***k*** is taken as 1.3 for this condition.

K = specific heat ratio.

II. CRTICAL PRESSURE

The pressure which occurred at throat condition of nozzle for specific heat ratio & temperature.

Pc = $3.764*10^6$ Pa

III. THROAT VELOCITY

For the Mach, no=1 at the throat condition.

M=V_t/a

Where M = 1

V_t= velocity at the throat

, **a**= speed of the sound =1002.0 m/s.

V_t = 1074.73 m/s.

IV. THE EXIT VELOCITY

The velocity which is ejected from the nozzle exit is effective exhaust velocity.

V_e = 2770.42 m/s.

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V. EXIT MACH.NO

Based upon the exit velocity of the nozzle the mach, no is

Me=3.5

All the vales are obtained from the standard equations of rocket propulsion system.

As per the values the mach, no at the exit condition is 3.5.

Here the stream lines should be designed for the exit mach, no

From exit Mach, no. as 3.5.

As per the rapid method for design of a supersonic nozzle.

Length to height ratio is 6.3 from the graph .

Take initial angle, $\nu_B = 22^o$

Expansion angle, $\nu_D = 58.5^o$.

5.2 STREAM LINES

Calculation of stream co-ordinates value of the stream function:

If the final design Mach, no (M) & initial-expansion angle (ν_B) , length-to-height ratio (l/h) and wall at the inflection point (θ_R) have already been selected are available, The first step is to find out the stream function ψ from

$$\Psi=\theta_R/\theta_{max}.$$

Where θ_{max} depends on the ν_B

Stream lines in the region II and III:

A complete layout of the characteristics net for the region II and III with $\nu_{B=}22^o$ from the condition of exit Mach, no. The Mach lines are represents the limiting streamline is possible for ψ=1.00. Here the streamline should not be be greater than one.

The dimension Cartesian coordinates X/Ycr and Y/Ycr of any streamlines (particular values ($\theta_{R \text{ and }} \psi$).

Region II are to be calculated by linear interpolation method.

The first point on the streamline is always $X/Y_{cr}=0$ and $Y/Y_{cr} = \psi$.

Next point is to obtained at the intersection of the streamline with the right Mach line.

Continue this step until the point on the left Mach line is reached. This indicates that the streamline has crossed the first mach line.

When the given values of ψ are less than these limits, the stream line enters the portion of region which is downstream of the Mach line.

After the stream lines enters region a sufficient number of points in region may also be determined by interpolation along the left Mach line entry.

The values of θ along streamline in both regions II and III.

May be determined by interpolation between the values of θ at the value ν and η.

STREAMLINES IN REGION IV:

The flow with region IV is always in radial flow; that is, the stream line are straight lines which, if extended upstream, would all emanate from a common point. The stream line forming the wall contour of a nozzle may or may not enter the radial-flow region. On the one hand, the stream line may contact the radial flow at only one.

If the stream lines passes through the radial flow, this portion of the nozzle will be a straight line.

STREAMLINES IN REGION V:

Region v is bounded by the Mach line. Parallel and uniform flow at the design Mach number M is attained along the straight mach line. This stream lines within the region are computed from the Foelsch equation.

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The final nozzle co-ordinates are obtained to the desired scale by multiplying all dimensionless co-ordinates by a suitable factor.

A method is presented for computing flow which generates supersonic radial flow from a parallel and uniform sonic flow. The co-ordinates of each point in the characteristics nets and corresponding values of the stream function have been computed and tabulated for several flows of this type. The co-ordinates of any stream line in these flows may be obtained to a high degree of accuracy by simple linear interpolation between the tabulated points for the required value of the stream function. The local flow angle the streamlines may be obtained in the same way. The supersonic nozzle design is then completed by matching one of these streamlines to a stream line computed from the equation

5.3 STREAM LINE CALUCULATIONS
REGION:1 (SUBSONIC REGION)

As per the formula,

$Y=c(1+.1924X^6)$

C is the constant value may be taken as unity
The values may taken as x=-.175 based upon that the y values may be obtained
Until the values may be attained at X=0 and Y=1,
After that the throat section will be starts.

REGION: 2 (INITIAL EXPANSION REGION).

At in the exit mach, no 3.5
The corresponding values of length-to-height ratio is 6.3
Here the $\theta_R = \theta_{max}$

$$\theta_R = \frac{\left(\frac{r}{rcr}\right)D - \left(\frac{r}{rcr}\right)B + \theta \max\left(\frac{x}{ycr}\right)B}{\left(\frac{r}{rcr}\right)D\left(\frac{l}{h} - \cot\mu D\right)}$$

From the $\nu_D = 58.5^o$
The values may taken from the appropriate values from ref.5
$\theta_R = 18.093^o$
by these value the stream function may be as
$\psi = .4981$
$x/y_{cr} = 1.66347$
$y/y_{cr} = .1305$
This will be continued until the mach , no 1-3.5

REGION: 3 (THE SECONDARY EXPANSION)

In this region the value may be obtained from the linear interpolation method. From here we obtained the values there are 10 right angles be found in the
Characteristics net values.
$\Psi = .4981.$
Based upon this value there are 10 right angle lines are formed during this region And we should the level upto 10.

$$\theta_R = \frac{\left(\frac{r}{rcr}\right)D - \left(\frac{r}{rcr}\right)B + \theta \max\left(\frac{x}{ycr}\right)B}{\left(\frac{r}{rcr}\right)D\left(\frac{l}{h} - \cot\mu D\right)}$$

Here the values are taken from the characteristics net

1. $\left(\frac{x}{ycr}\right) = .875459$, $\left(\frac{y}{ycr}\right) = 1.3739.$
2. $\left(\frac{x}{ycr}\right) = .98088$, $\left(\frac{y}{ycr}\right) = 1.50182$
3. $\left(\frac{x}{ycr}\right) = 1.11288$ $\left(\frac{y}{ycr}\right) = 1.5247$

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4. $(\frac{x}{ycr})$ = 1.26046 $(\frac{y}{ycr})$ = 1.6017
5. $(\frac{x}{ycr})$ = 1.40074 $(\frac{y}{ycr})$ = 1.6817
6. $(\frac{x}{ycr})$ = 1.49929 $(\frac{y}{ycr})$ = 1.7247
7. $(\frac{x}{ycr})$ = 1.66347 $(\frac{y}{ycr})$ = 1.84916
8. $(\frac{x}{ycr})$ = 1.80403 $(\frac{y}{ycr})$ = 1.94144
9. $(\frac{x}{ycr})$ = 1.93184 $(\frac{y}{ycr})$ = 2.02594
10. $(\frac{x}{ycr})$ = 2.051 $(\frac{y}{ycr})$ = 2.10672

These are the values obtained in the region of regionIII

REGION: IV (RADIAL FLOW)

In this region the stream lines flows are at in the straight line, if extended upstream, would emanate from a common point.

Then the equation is,

$\nu_B = \nu_D - \theta_R$

= 58.5-18.093

= 40.407

So the values should be taken for 40.407

The equation may be written as

$$\frac{x}{ycr} = \frac{1}{\theta max}\left(\frac{r}{rcr}\right) R\cos\theta R + [\left(\frac{x}{ycr}\right) B - 1/\theta\max\left(\frac{r}{rcr}\right) B]$$

$$\frac{y}{ycr} = \frac{1}{\theta max}\left(\frac{r}{rcr}\right) R\sin\theta R$$

From these the values be

$\frac{x}{ycr} = 3.4247$

$\frac{y}{ycr}$ = 2.36,

REGION V: (PARALLEL AND UNIFORM FLOW)

This is the final region of flow is to be calculated by the value ranges at in the 0 to 18.093

θp = 0 to θp = θr

In between these degrees we have taken that With the two degrees each we can take as 8 points For that corresponding values should taken.

The equations for X and Y are

$$\frac{x}{ycr} = \frac{(\frac{r}{rcr})P}{\theta max}[\cos\theta P + (\theta R - \theta p)(\frac{cos\ \theta p}{\tan\mu p} - sin\theta p)] + [\left(\frac{x}{ycr}\right) B - \frac{\left(\frac{r}{rcr}\right) B}{\theta max})]$$

$$\frac{y}{ycr} = \frac{\left(\frac{r}{rcr}\right)P}{\theta max}[\sin\theta P + (\theta r - \theta p)(\frac{sin\theta p}{tan\mu p} + cos\theta p)]$$

From these equations finally we obtain the values for last region

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VI. RESULTS AND DISCUSSION

BASED UPON THE PROPELLANT PROPERTY:

LIQUID OXYGEN AND LIQUID HYDROGEN as propellant

1. Pressure ratio : 68.039
2. Specific heat : 1.3
3. Critical pressure : $3.764*10^6$ Pa
4. Velocity at the throat : 1002.0 m/s
5. Exit velocity : 2770.74 m/s
6. Exit mach. No : 3.5

Stream line function:

Length to height ratio = 6.3

Take initial angle, ν_B = 22°

Expansion angle, ν_D = 58.5°.

TABLE: II

s.no	$\frac{x}{ycr}$	$\frac{y}{ycr}$
1	-1.75	6.526
2.	-1.5	3.19
3.	-1.25	1.7339
4.	-1	1.1924
5.	-.75	1.0034
6.	-.5	1.003
7.	-.25	1.003
8.	0	1
9.	.87549	1.3739
10.	.98098	1.50182
11.	1.11288	1.5247
12.	1.26046	1.6107
13.	1.40074	1.68171
14.	1.499229	1.7241
15.	1.6634	1.84916
16.	1.8043	1.94144
17.	1.93138	2.02594
18.	2.051	2.10672
19.	3.4247	2.36
20.	4.157	2.419
21.	5.09	2.585
22.	6.2203	2.7378
23.	7.61065	2.886
24.	9.303	3.0265
25.	13.9456	3.2648
26.	17.695	3.44
27.	21.26	3.47

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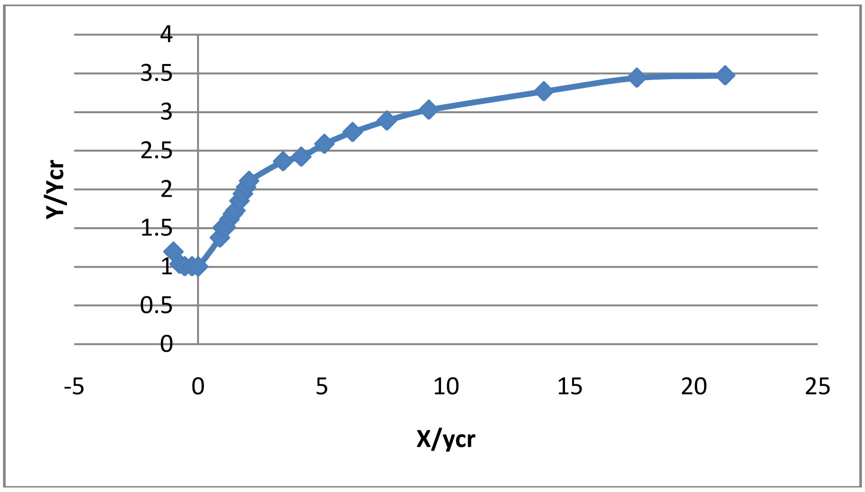
As per the points



This below figure shows that the modelling of 2D for the contour nozzle.

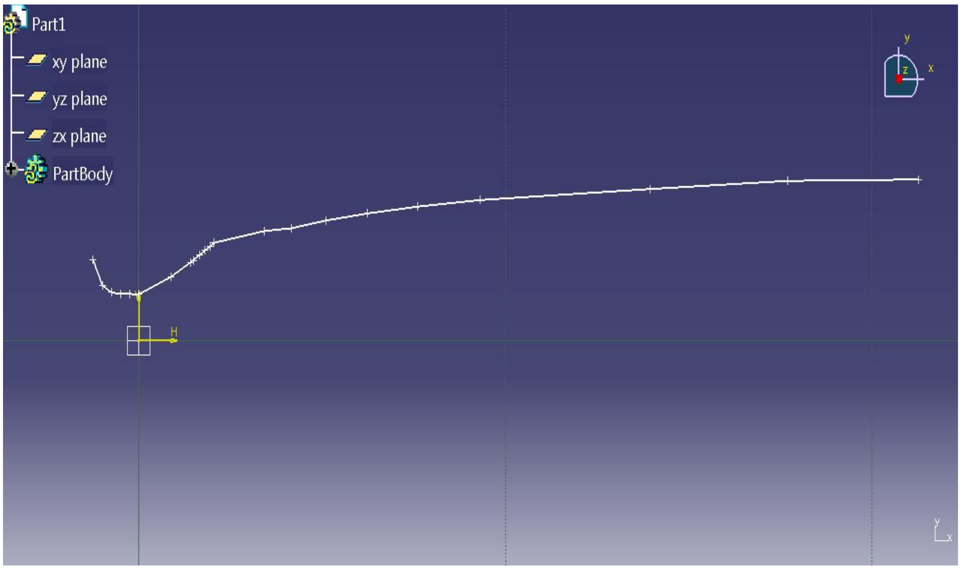


Fig :3

As per the table: 2 it represents the co-ordinate points for the profile .In those point from 1 to 8 is the subsonic region. And points from 9 are the initial expansion region From the points 10 to 20 is the secondary expansion region .From the points 21 is source flow region. From the points 22 to 27 is the last region.

Figure .2 represents the profile of the contour nozzle using excel format. And figure.3 represents the points of the line diagram

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VII. CONCULSION

In this project Phase-I the length and exit area of the rocket nozzle are optimized partially. For optimization, region to region method is used. The 2D model of nozzle is created by using CATIA software and it is to be converted into 3D model. Finally the optimized rocket nozzle efficiency will be compared with the existing one.

VIII. FURTURE WORK

Based on the calculations we have to design the nozzle in 2D form. For further work I have to covert this into 3D model. Also analyse the 3D model using the software CFD (Computational Fluid Dynamics) and to obtain the flow behaviour acting in the model.

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