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Design And Analysis Of Thrust Chamber Of A Cryogenic Rocket Engine

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ABSTRACT

The basic concept of a rocket engine relays on the release of the internal energy of the propellant molecules in the combustion chamber, the acceleration of the reaction product and finally the release of the hot gases at the highest possible velocity in the convergent/divergent nozzle. Liquid rocket engines burn propellants, which undergo chemical reactions to convert the stored chemical energy to thermal energy which results in the generation of thrust. Thrust chamber of cryogenic engine is modeled at a chamber pressure of 40 bar and thrust of 50KN to reduce the high temperature and pressure in the combustion chamber. CFD analysis is done to show the pressure and temperature variation in the thrust chamber modeled for 50KN thrust and chamber pressure of 40 bar. The design of rocket engine should be such that it should withstand the high pressure and high temperature of the combustion chamber.

NOMENCLATURE

C*-characteristic velocity
C_f – thrust coefficient
A_t -throat area
D_t-throat diameter
F-thrust
R_{cc}-radius at the convergent cylindrical portion
R_{cd}-radius at the cylindrical divergent portion
D_e- nozzle exit diameter

1. INTRODUCTION

Payload capacity can be increased with the propulsion system having higher specific impulse, in general liquid propellant engines results in longer burning time than conventional solid rocket engines which result in higher specific impulse. Liquid propellants can be classified based on the storable conditions, namely earth storable, cryogenic, space storable etc. Among these cryogenic propellants are widely used because of their high specific impulse compared to other type of propellants. Cryogenics are high energetic propellant combination, also the future space mission needs a propulsion system with the high thrust and reusable capacities, like Space Shuttle Main Engine (SSME), which can also leads to reduction in launch cost per kg of payload with the highly improved payload capacity. As above said space shuttle main engine uses Liquid oxygen and liquid hydrogen as a propellant combination with high specific impulse over the other cryogenic propellant combinations. Apart from the LOX/ LH2 combination, LOX/ RP- 1 (Kerosene) semi-cryo combination is widely used in Russian launch vehicle engines of RD- 107, RD- 108and RD- 461 in the Soyuz launch vehicles, which offers slightly lower specific impulse than LOX/ LH2 combination with advantage of handling. The demand of increasing payloads and increasing size of satellite increases the need of more efficient rocket propulsion system. The cost of the payload can be decreased by implementing new reusable launching systems with the intentions to increase the efficiency of the launching system. For such condition high energetic cryogenic propellant combination provides suitable thrust to weight ratio with high specific impulse than current existed earth storable propellants.

The thrust chamber of a cryogenic rocket engine is exposed to extreme conditions during operation where pressures go as high as 20MPa and temperatures reach 3600K, this high pressure and high temperature leads to high heat transfer rates (100 to 160 MW/m^2) in thrust chamber. Under these harsh conditions, the combustor must incorporate active cooling to increase the thrust chamber life by depressing thermal stresses and preventing wall failure. Regenerative cooling is the most effective method used in cooling of liquid propellant rocket engines

2. DESIGN OF THRUST CHAMBER

The propellant combination of liquid hydrogen and liquid oxygen has high specific impulse compared to other propellant combinations have been used in many rocket engines. The thermal analysis is a major issue in at the channel exit. Those parameters are important for the design of injectors and of the coolant pump. The design of a liquid rocket engine, because the prediction of peak heat-flux from the combustion gases to the engine wall is necessary to ensure the structural integrity of the combustion chamber. The need for thermal analysis is essentially important to extend the engine life by effective and efficient cooling system. Moreover, the analysis of the cooling channel flow is essential to predict not only the efficiency of the coolant, but also the coolant temperature and pressure.The design of thrust chamber consists of many parameters and detail calculations, using basic geometric parameters are adequate to understand the regenerative cooling effect of the system. For the built-up of gas-dynamic profile of the combustion chamber, it is necessary to give some input data to the system such as thrust (at sea level), chamber pressure, ambient pressure and propellant components. Some of these parameters are listed below in the table.

Thrust	50KN
Fuel	Liquid hydrogen(LH_2)
Oxidizer	Liquid oxygen(LOX)
Ambient pressure	1 bar

Table 1 LPRE Requirements

Chamber pressure is very important in the designing of rocket engine. The thrust chamber performance increases with increase in chamber pressure and also higher chamber pressure reduces the performance losses due to kinetics. Thrust chamber size and weight decreases as the chamber pressure increases. Higher chamber pressure provides higher nozzle expansion ratio, which in turn reduces the chamber and nozzle envelope for a fixed thrust. For LOX/LH_2 combination of oxidiser and fuel, mixture mass ratio is found to be between 4.5 & 6 not at the stoichiometric value of 8. This is because a gas which is slightly richer in fuel tends to have a lower molecular weight. This results in a higher overall engine system performance. As area ratio (AR) increases, the specific impulse increases, due to higher expansion of hot gas which generates higher velocity at nozzle exit. But AR is primarily selected based on the application of engine in the booster stage or upper stage.

2.1 CALCULATION OF CHAMBER THROAT DIAMETER

The performance parameters are obtained through NASA-CEA code for the given input condition of the engine. Those values are listed in the following table.

PARAMETERS	VALUES
Characteristicvelocity(C^*)	2249 m/s
Specific impulse I_{sp}	376 s
Thrust coefficient, C_f	1.675
Molecular weight	13.37

Table 2 Parameters from NASA-CEA Code

Total mass flow rate can be calculated from the following equation

$$F = I_{sp\ act} \times \dot{m}_{total} \times g$$

$$\dot{m}_{total} = \frac{50000}{350 * 9.81}$$

$$= 14.5 \text{ Kg/s}$$

$$\dot{m}_{total} = \frac{P_c \times A_t}{C^*_{act}}$$

$$A_t = \frac{14.5 \times 2159}{40 \times 10^5}$$

$$= 7.83 \times 10^{-3} \text{ m}^2$$

$$D_t = 100 \text{ mm}$$

The chamber volume is the function of mass flow rate of the propellants and their average density and of the stay time needed for efficient combustion.

$$V_c = \dot{W}_{tc} * V * t_s$$

Combustion chamber volume (V_c)

$$V_c = L^* \times A_t$$

$$= 635 \times 7.83\text{E-3}$$

$$= 4987278 \text{ mm}^3$$

While designing the combustion chamber, proper value of L^* is to be considered because an increase in L^* beyond a certain limit results in decrease in overall engine system performance. In case of operating cryogenic engines the contraction ratio varies in the range from 1.58 to 5. Design of very high contraction area ratios (> 5) have difficult in maintaining stable boundary layer, adjacent to the throat.

2.2 SELECTION OF CHAMBER INTERNAL PROFILE

The configuration of the combustion chamber is cylindrical which provides sufficient surface area for cooling. For this engine configuration, semi convergent angle of 20 degrees has been selected, favoring a lower heat flux and thereby maximizing the life of the chamber. For current engine configuration, Rcd/ Dt were taken as 1. So Rcd = 1 * 100 = 100 mm**.** Usually the ratio of the radius of curvature at entry to the convergent of the diameter of the chamber Rcc/ Dc varies from 0.5 to 1.5. If the value is too low say < 0.5, then there is a possibility of flow separation along the convergent portion with associated heat transfer problems. Comparatively high value of Rcc/ Dc is taken for design. For the current configuration engine Rcc/ Dc has taken as 1.0, so the value of Rcc for the engine is 200 * 1 = 200 mm.

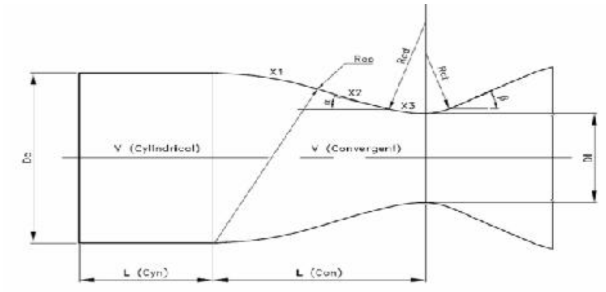

Fig 1 A Schematic of Thrust Chamber Profile

The convergent length of combustion chamber is the sum of the axial length of the portion between the cylindrical chamber & the Nozzle cone (X_1), length of the conical portion (X_2) & the length between conical potion and the throat (X_3), these three lengths are calculated as follows.

$$X_1 = R_{cc}\sin\alpha$$

$$= 200\sin 20$$

$$= 68.4 \text{ mm}$$

$$X_3 = R_{cd}\sin\alpha$$

$$= 100\sin 20$$

$$= 34.2 \text{ mm}$$

$$X_2 = \frac{\frac{D_C}{2} - \left[R_t + R_{cc}\left(1-\cos\alpha\right) + R_{cd}\left(1-\cos\alpha\right)\right]}{\tan\alpha}$$

$$= 87.67 \text{ mm}$$

$$L_{con} = X_1 + X_2 + X_3$$
$$= 190 \text{ mm}$$

The volume of the nozzle convergent portion (V_{con}) has been computed by assuming the convergent portion as the frustum of a cone as reported in literature by Huzel.

$$V_{con} = \frac{\pi}{3} \times L_{con} \times \left[R_c^{\,2} + R_t^{\,2} + R_c R_t\right]$$

Volume of the cone V_{con} = 3481931.8 mm^3

Volume of the cylindrical portion = V_c - V_{con}

= 4987278-3481931.8

V_{cyl} = 1505346 mm^3

Length of the cylindrical portion

$$L_{cyl} = \frac{V_{cyl}}{\frac{\pi}{4} d_c^{\,2}}$$

= 48 mm

2.3 SELECTION OF NOZZLE PROFILE

The use of bell type contour nozzle will give higher performance. The parabolic approximation methods suggested by G.V. R. Rao is a convenient tool for designing a near optimum bell nozzle contour. The nozzle contour immediately upstream of the throat is a circular arc and also the divergent section nozzle contour is made up of a circular entrance section with a short radius. From this point to exit, the parabola is fitted.

For this current configuration, divergent portion of the nozzle, a parabolic bell shaped contour has been chosen. The radius of curvature at the starting of the divergent (R_{ct}) decides the rate by which the expansion can be achieved in the divergent and also the rate of maximum heat flux. $R_{\theta ct}$/ D_t can vary from 0.25 to 1. Usually a low value is selected because it results in better thermal margin (heat flux is less) and smooth profile contour resulting in reduction of coolant pressure drop. For current engine design a ratio of 0.5 has been selected R_{ct} = 100 x 0.5 = 50 mm. The bell nozzle usually employs a higher initial wall angle for faster expansion in the initial divergent section and then a uniform axially directed flow at the nozzle exit. Axial length of nozzle from the throat section to the exit plane is calculated assuming the bell nozzle as an equivalent conical nozzle. Equivalent conical half angle is usually between 12 deg to 18 deg. The angle has been selected for this configuration as 20 deg with nozzle area ratio of 15.

Nozzle exit diameter is calculated based on the value of area ratio as in the literature by Schuff

$$D_e = \sqrt{\varepsilon} \times d_t$$
$$= \sqrt{15} \times 100$$
$$= 387.3 \text{ mm}$$

Substituting Rt = 50 mm, Rct = 25 mm and semi divergent angle θ=15 deg.

$$L_{div} = \frac{R_t\left(\sqrt{\varepsilon} - 1\right) + R_{ct}\left(\sec\theta - 1\right)}{Tan\theta}$$

$$L_{div} = \frac{50\left(\sqrt{15} - 1\right) + 25\left(\sec 15 - 1\right)}{Tan 15}$$

$$= 539.39 \text{ mm}$$

Length of the bell nozzle is to be 80% of that of an equivalent conical nozzle

L_{bell} = 0.8 L_{div}=434.4 mm

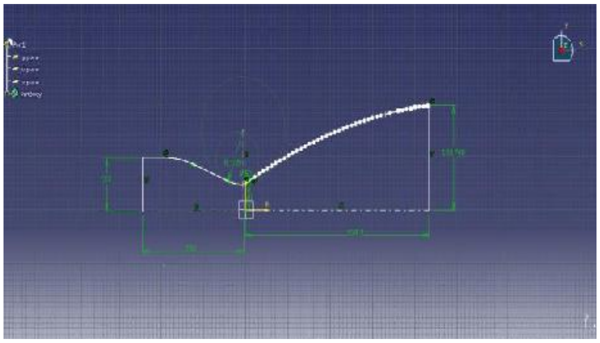

Fig 2 2-D Diagram of Thrust Chamber

3. INTRODUCTION TO COMPUTATION TECHNIQUE

In order to verify the design quality of the nozzle flow field investigation of the nozzle has to be studied. A 2-D axis symmetric computational simulation of the flow field is necessary, which may be either developing a code or by using commercial code. In the present study computation has been made to obtain flow through the nozzle using commercial software FLUENT. In the present study two dimensional structured grids were generated on the nozzle profile along with extended domain. For axis symmetric computation, quadrilateral cells with map scheme was used for grid generation

3.1 BOUNDARY CONDITIONS

The boundary conditions values are taken from NASA-CEA code.

Pressure inlet

Total pressure	=	40 bar
Total temperature	=	3420 K

Material properties (combustion gas)

Specific heat	=	3768 J/Kg-K
Viscosity	=	1.0455E-4
Kg/m-s		
Thermal conductivity	=	0.5590 W/mK
Molecular weight	=	13.37

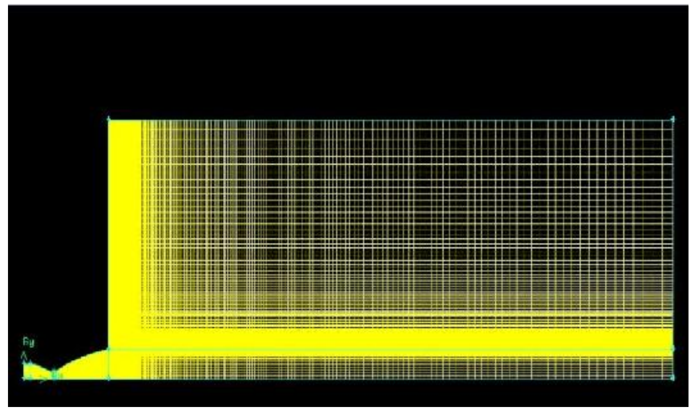

Fig 3 Mesh Domain of the Thrust Chamber

3.2 NOZZLE FLOW ANALYSIS RESULTS

The following results contains the information about the nozzle pressure, velocity, temperature and mach no variation along the nozzle and domain for LOX/LH_2 propellant combination.

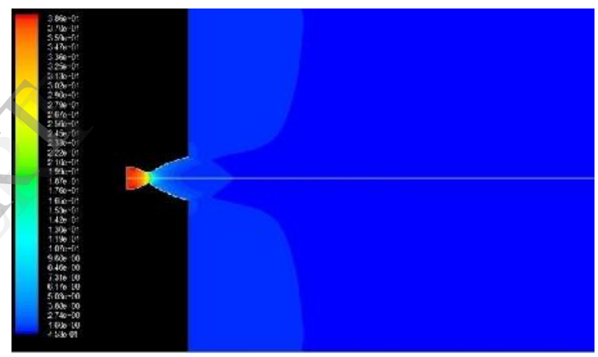

Fig 4 Contours of Static Pressure

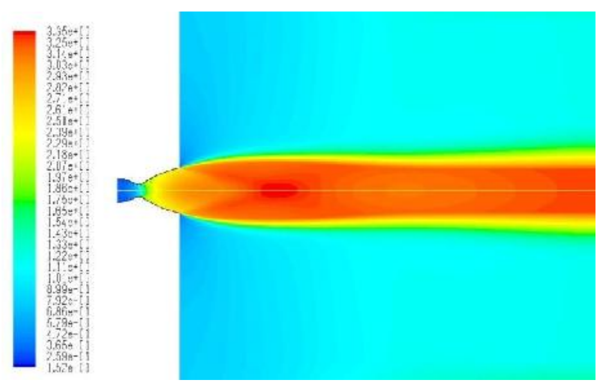

Fig 5 Contours of Mach Number

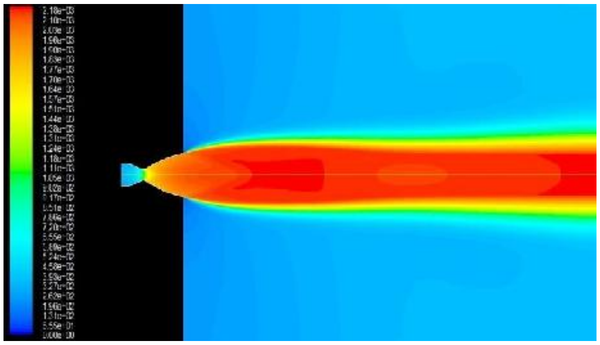

Fig 6 Contour of Velocity Magnitude (m/s)

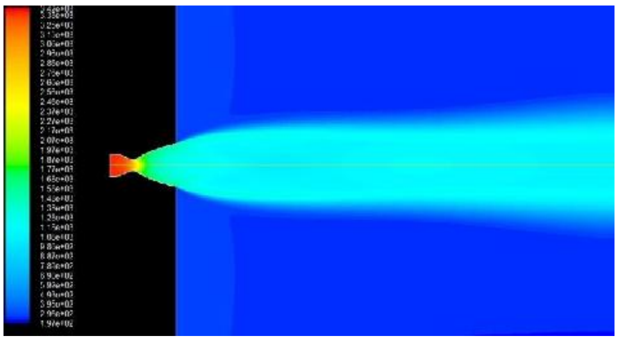

Fig 7 Contour of Static Temperature (K)

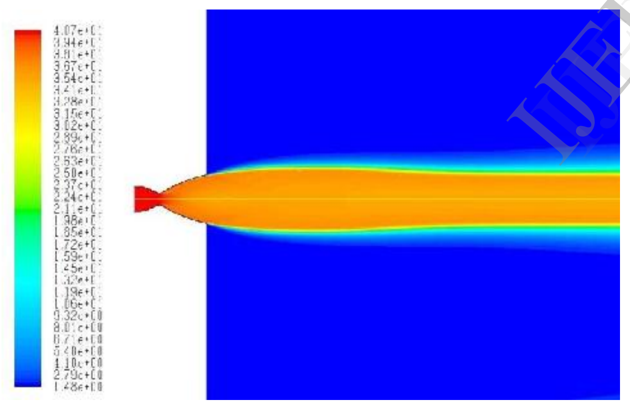

Fig 8 Contour of Total Pressure

Parameters	**Theoretical results**	**Computational results**
Totalmassflowrate (Kg/s)	14.7	14.92
Exit mach number	3.3	3.13
Total temperature(K)	3400	3400
Total pressure(bar)	40	40

Table 3 Nozzle Flow Analysis Results

4. CONCLUSION

Cryogenic propellants in liquid rocket engine provide high specific impulse which is suitable for use in rocket upper and booster stages. For the thrust chamber modeled at 50KN of thrust and chamber pressure of 40 bar with the propellant combination of LH_2/LOX shows a very good consistency when compared with computational results. From the analysis result it is found that cryogenic rocket engine with the propellant combination of LH_2/LOX is suitable for design of upper stage rocket.

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