Design and Fabrication of Vortex Cooled Liquid Rocket Engine

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***Abstract*-**In this project, I have introduced a swirl oxidizer flow shield inside a rocket engine in a fluid propellant rocket engine is to prevent convective and conductive forms of heating of the engine structural parts of the same to a maximum extend. The flow field includes an outer fluid vortex spiraling towards a closed end of the flow field with the help of a centrifugal force provided by the oxidizer port positioning and the introduction of oxidizer in a desired velocity to create an opposite vortex flow from the down end of the chamber to upward through specially designed and positioned oxidizer ports. This swirling flow field becomes an inner fluid vortex trough the core of the chamber wall diameter opposite to the outer vortex direction. This inner vortex will cause proper mixing of the fuel and oxidizer, intern cause proper burning. Outer vortex consisting of oxidizer will prevent heating of chamber wall and related assemblies. This does have enormous advantages than existing designs. In the first phase of this project I have completed the designing and material selection of the rocket engine with the help of the design values calculated using the design equations provided by rocket lab (NASA).I have compared some of the major material specification and selected EN22 as my material of fabrication for the rocket engine. And the fabrication process is carried out with proper steps, then the static testing is done

Index Terms – Rocket Propulsion, Design, Vortex cooling, Laboratory Experiments.

INTRODUCTION

All liquid rocket engines have tankage and pipes to store and transfer propellant, an injector system, a combustion chamber which is very typically cylindrical, and one (sometimes two or more) rocket nozzles. Liquid systems enable higher specific impulse than solids and hybrid rocket engines and can provide very high tankage efficiency. Unlike gases, a typical liquid propellant has a density similar to water, approximately 0.7-1.4g/cm³ (except liquid hydrogen which has a much lower density), while requiring only relatively modest pressure to prevent vaporization. This combination of density and low pressure permits very lightweight tankage; approximately 1% of the contents for dense propellants and around 10% for liquid hydrogen (due to its low density and the mass of the required insulation).

For injection into the combustion chamber the propellant pressure at the injectors needs to be greater than the chamber pressure; this can be achieved with a pump. Suitable pumps usually use centrifugal turbo pumps due to their high power and light weight, although reciprocating pumps have been employed in the past. Turbo pumps are usually extremely lightweight and can give excellent performance; with an on-Earth weight well under 1% of the thrust. Indeed, overall rocket engine thrust to weight ratios including a turbo pump have been as high as 133:1 with the Liquid-propellant rocket 3 Soviet NK-33 rocket engine.

Alternatively, instead of pumps, a heavy tank of a high-pressure inert gas such as helium can be used, and the pump forgone; but the delta-v that the stage can achieve is often much lower due to the extra mass of the tankage, reducing performance; but for high altitude or vacuum use the tankage mass can be acceptable. The major components of a rocket engine are therefore the combustion chamber (thrust chamber), pyrotechnic igniter, propellant feed system, valves, regulators, the propellant tanks, and the rocket engine nozzle. In terms of feeding propellants to the combustion chamber, liquid-propellant engines are either pressure-fed or pump-fed, and pump-fed engines work in either a gas-generator cycle, a staged-combustion cycle, or an expander cycle. A liquid rocket engine (LRE) can be tested prior to use, whereas for a solid rocket motor a rigorous quality management must be applied during manufacturing to ensure high reliability. A LRE can also usually be reused for several flights, as in the Space Shuttle.

Bipropellant liquid rockets are simple in concept but due to high temperatures and high speed moving parts, very complex in practice. Use of liquid propellants can be associated with a number of issues:

- Because the propellant is a very large proportion of the mass of the vehicle, the center of mass shifts significantly rearward as the propellant is used; one will typically lose control of the vehicle if its center mass gets too close to the center of drag.
- When operated within an atmosphere, pressurization of the typically very thin-walled propellant tanks must guarantee positive gauge pressure at all times to avoid catastrophic collapse of the tank.
- Liquid propellants are subject to slosh, which has frequently led to loss of control of the vehicle. This can be controlled with slosh baffles in the tanks as well as judicious control laws in the guidance system.
- They can suffer from pogo oscillation where the rocket suffers from uncommanded cycles of acceleration.
- Liquid propellants often need ullage motors in zero-gravity or during staging to avoid sucking gas into engines at start up.

They are also subject to vortexing within the tank, particularly towards the end of the burn, which can also result in gas being sucked into the engine or pump.

• Liquid propellants can leak, especially hydrogen, possibly leading to the formation of an explosive mixture.

• Turbo pumps to pump liquid propellants are complex to design, and can suffer serious failure modes, such as over speeding if they run dry or shedding fragments at high speed if metal particles from the manufacturing process enter the pump.

• Cryogenic propellants, such as liquid oxygen, freeze atmospheric water vapour into very hard crystals. This can damage or block seals and valves and can cause leaks and other failures. Avoiding this problem often requires lengthy chill down procedures which attempt to remove as much of the vapour from the system as possible. Ice can also form on the outside of the tank, and later fall and damage the vehicle. External foam insulation can cause issues as shown by the Space Shuttle Columbia disaster. Non-cryogenic propellants do not cause such problems.

• Non-storable liquid rockets require considerable preparation immediately before launch. This makes them less practical than solid rockets for most weapon systems.

VORTEX COOLED ROCKET ENGINE

The heat created during combustion in a rocket engine is contaminated within the exhaust gases. Most of the heat is expelled along with the gas that contains it. However, heat is transferred to the thrust chamber walls in quantities sufficient to require attention.

Thrust chamber designed are generally categorized or identified by the hot gas wall cooling method or the configuration of the coolant passages where the coolant pressure inside may be as high as 500 atmospheres. The high combustion temperatures (2500 to 3600 K) and the high heat transfer rates (up to 16 kj/cm2-s) encountered in a combustion chamber present a formidable challenge to the designer. To meet this challenge, several chamber cooling techniques have been utilized successfully. Selection of the optimum cooling method for a thrust chamber depends on many consideration, and combustion chamber material.

Regenerative cooling is the most widely used method of cooling a thrust chamber and is accomplished by flowing high velocity coolant over the back side of the chamber hot gas wall to convectively cool the hot gas liner. The coolant with the heat input from cooling the liner is then discharged into the injector and utilized as a propellant. Earlier thrust chamber designs, such as the V-2 and Redstone, had low chamber pressure, low heat flux and low coolant pressure requirements, which could be satisfied by a simplified “Double wall chamber” design with regenerative and film cooling. For subsequent rocket engine applications, however, chamber pressures were increased and the cooling requirements become more difficult to satisfy. It became necessary to design new coolant configurations that were more efficient structurally and had improved heat transfer characteristics.

dump cooling, which is similar to regenerative cooling because the coolant flows through small passages over the back side of the thrust chamber wall. the difference, however, is that after cooling the thrust chamber, the coolent is discharged overboard through openings at the aft end of the divergent nozzle. the method has limited application because of the performance loss resulting from dumping the coolant overboard. to date, dump cooling has not been used in an actual application.

Film cooling provides protection from excessive heat by introducing a thin film of coolant or propellant through orifices around the injector periphery or through manifolded orifices in the chamber wall near the injector or chamber throat region. This method is typically used in high heat flux regions and in combination with regenerative cooling.

Transpiration cooling provides coolant (either gaseous or liquid propellant) through a porous chamber wall at a rate sufficient to maintain the chamber hot gas wall to the desired temperature. The technique is really a special case of film cooling.

With ablative cooling, combustion gas-side wall material is sacrificed by melting, vaporization and chemical changes to dissipate heat. As a result, relatively cool gases flow over the wall surface, thus lowering the boundary-layer temperature and assisting the cooling process.

With radiation cooling, heat is radiated from the outer surface of the combustion chamber or nozzle extension wall. Radiation cooling is typically used for small thrust chambers with a high-temperature wall material (refractory) and in low-heat flux regions, such as a nozzle extension.

Liquid rocket engines may be run for repeatedly over extended periods of time, requiring active cooling. An efficient means of cooling LRE nozzles involves using the liquid fuel or oxidizer itself to provide the cooling, often referred to as regenerative cooling. SEA's liquid rocket combustion chamber and nozzle design code, TDK, can predict the heat transfer to the surface of the nozzle as a function of the nozzle geometry and fuel / oxidizer properties.

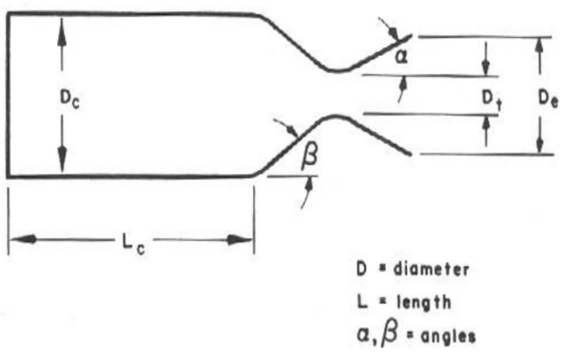
The standard RTE code predicts the amount of cooling produced by a regenerative cooling system, given the heat transfer to the cooling fluid. In this research effort, we are combining the two codes, so they may iteratively determine the optimal cooling design. In addition, we are extending the capabilities of TDK and RTE to evaluate novel nozzle designs and cooling concepts.

Introducing a swirl oxidizer flow shield inside a rocket engine in a fluid propellant rocket engine is to prevent convective and conductive forms of heating of the engine structural parts of the same to a maximum extend. The flow field includes an outer fluid vortex spiraling towards a closed end of the flow field with the help of a centrifugal force provided by the oxidizer port positioning and the introduction of oxidizer in a desired velocity to create an opposite vortex flow from the down end of the chamber to upward through specially designed and positioned oxidizer ports. This swirling flow field becomes an inner fluid vortex trough the core of the chamber wall diameter opposite to the outer vortex direction. This inner vortex will cause proper mixing of the fuel and oxidizer, intern cause proper burning. Outer vortex consisting of oxidizer will prevent heating of chamber wall and related assemblies. This does have enormous advantages than existing designs

MATHEMATICAL MODELLING OF THE ROCKET ENGINE

performance evaluation

The following section will detail simplified equations for the design of small liquid-fuel rocket motors. The nomenclature for the motor design is shown in Figure.



NOZZLE DESIGN

The nozzle throat cross-sectional area may be computed if the total propellant flow rate is known and the propellants and operating conditions have been chosen. Assuming perfect gas law theory:

$$A_t = w_t/P_t \sqrt{R\,T_t/\gamma\, g_c} \qquad (7)$$

where R = gas constant, given by R = R / M. R is the universal gas constant equal to 1545.32 ft-lb/lb° R, and M is the molecular weight of the gas. The molecular weight of the hot gaseous products of combustion of gaseous oxygen and hydrocarbon fuel is about 24, so that R is about 65 ft-lb/lb° R. Gamma, g, is the ratio of gas specific heats and is a thermodynamic variable which the reader is encouraged to read about elsewhere (see Bibliography). Gamma is about 1.2 for the products of combustion of gaseous oxygen and hydrocarbon fuel.

gc is a constant relating to the earth's gravitation and is equal to 32.2 ft/sec/sec. For further calculations the reader may consider the following as constants whenever gaseous oxygen and hydrocarbon fuel propellants are used:

$$R = 65 \text{ ft-lb/lb°R}$$

$$\gamma = 1.2$$

$$g_c = 32.2 \text{ ft/sec}^2$$

Tt is the temperature of the gases at the nozzle throat. The gas temperature at the nozzle throat is less than in the combustion chamber due to loss of thermal energy in accelerating the gas to local speed of sound (Mach number = 1) at the throat. Therefore

$$T_t = T_c \left[\frac{1}{1 + \frac{\gamma - 1}{2}}\right] \qquad (8)$$

For $\gamma = 1.2$

$$T_t = (.909)(T_c) \qquad (9)$$

Tc is the combustion chamber flame temperature in degrees Rankine (°R), given by

$$T\ (°R) = T\ (°F) + 460 \qquad (10)$$

Pt is gas pressure at the nozzle throat. The pressure at the nozzle throat is less than in the combustion chamber due to acceleration of the gas to the local speed of sound (Mach number =1) at the throat. Therefore

$$P_t = P_c \left[1 + \frac{\gamma - 1}{2}\right]^{-\frac{\gamma}{\gamma-1}} \qquad (11)$$

For $\gamma = 1.2$

$$P_t = (.564)(P_c) \qquad (12)$$

The hot gases must now be expanded in the diverging section of the nozzle to obtain maximum thrust. The pressure of these gases will decrease as energy is used to accelerate the gas and we must now find that area of the nozzle where the gas pressure is equal to atmospheric pressure. This area will then be the nozzle exit area.

Mach number is the ratio of the gas velocity to the local speed of sound. The mach number at the nozzle exit is given by a perfect gas law expansion expression

$$M_e^2 = \frac{2}{\gamma - 1}\left[\left(\frac{P_c}{P_{atm}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \qquad (13)$$

Pc is the pressure in the combustion chamber and Patm is atmospheric pressure, or 14.7 psi. Since g is fixed at 1.2 for gaseous oxygen and hydrocarbon propellant products, we can eliminate the parameters for future design use.

Table 1: Nozzle parameters for various chamber pressures

TABLE III Nozzle Parameters for Various Chamber Pressures, $\gamma = 1.2$, Patm = 14.7 psi

P_c	M_e	A_e/A_t	T_e/T_c
100	1.95	1.79	0.725
200	2.33	2.74	0.65
300	2.55	3.65	0.606
400	2.73	4.6	0.574
500	2.83	5.28	0.55

Therefore,

$$A_e = A_t\ (A_e/A_t) \qquad (15)$$

The temperature ratio between the chamber gases and those at the nozzle exit is given by

$$T_e = T_c \ (T_e/T_c) \quad (16)$$

The nozzle throat area diameter is given by

$$D_t = \sqrt{4A_t/\pi}\ , \quad \pi = 3.14 \quad (17)$$

and the exit diameter is given by

$$D_e = \sqrt{4A_e/\pi} \quad (18)$$

A good value for the nozzle convergence half-angle b (see Figure 3) is 60°. The nozzle divergence half-angle, a, should be no greater than 15° to prevent nozzle internal flow losses

COMBUSTION CHAMBER DESIGN

A parameter describing the chamber volume required for complete combustion is the characteristic chamber length, L*, which is given by

$$L^* = V_c/A_t \quad (19)$$

Where Vc is the chamber volume (including the converging section of the nozzle), in cubic inches, and At is the nozzle throat area (in2). For gaseous oxygen/hydrocarbon fuels, an L* of 50 to 100 inches is appropriate. L* is really a substitute for determining the chamber residence time of the reacting propellants.

To reduce losses due to flow velocity of gases within the chamber, the combustion chamber cross-sectional area should be at least three time the nozzle throat area. The combustion chamber cross-sectional area is given by

$$A_c = \pi D_c^2/4 \quad (20)$$

The chamber volume is given by

$$V_c = A_c L_c + \text{convergent volume}$$

For small combustion chambers the convergent volume is about 1/10 the volume of the cylindrical portion of the chamber, so that

$$V_c = 1.1(A_c L_c) \quad (21)$$

The chamber diameter for small combustion chambers (thrust levels less than 75 pounds) should be three to five times the nozzle throat diameter so the injector will have useable face area.

Chamber Wall

The combustion chamber must be able to withstand the internal pressure of the hot combustion gases. The combustion chamber must also be physically attached to the cooling jacket and, therefore, the chamber wall thickness must be sufficient for welding or brazing purposes. Since the chamber will be a cylindrical shell, the working stress in the wall is given by

$$S = PD/2t_w \quad (22)$$

Where P is the pressure in the combustion chamber (neglecting the effect of coolant pressure on the outside of the shell), D is the mean diameter of the cylinder, and tw is the thickness of the cylinder wall. A typical material for small water-cooled combustion chambers is copper, for which the allowable working stress is about 8,000 psi. The thickness of the combustion chamber wall is therefore given by

$$t_w = PD/16000 \quad (23)$$

This is the minimum thickness; actually the thickness should be somewhat greater to allow for welding, buckling, and stress concentration. The thickness of the chamber wall and nozzle are usually equal.

Equation can also be used to calculate the wall thickness of the water cooling jacket. Here again, the value of two will be the minimum thickness since welding factors and design considerations (such as O-ring grooves, etc.) will usually require walls thicker than those indication by the stress equation. A new allowable stress value must be used in Equation (22), dependent on the jacket material chosen.

MATERIALS

The combustion chamber and nozzle walls have to withstand relatively high temperature, high gas velocity, chemical erosion, and high stress. The wall material must be capable of high heat transfer rates (which means good thermal conductivity) yet, at the same time, have adequate strength to withstand the chamber combustion pressure. Material requirements are critical only in those parts which come into direct contact with propellant gases. Other motor components can be made of conventional materials

Exotic metals and difficult fabrication techniques are used in today's space and missile rocket engines, providing a lightweight structure absolutely required for efficient launch and flight vehicles. These are advanced metals and fabrication techniques are far outside the reach of the serious amateur builder. However, the use of more commonplace (and much less expensive!) metals and fabrication techniques is quite possible, except that a flight weight engine will not result. Since almost all amateur rocket firing should be conducted on a static test stand, this is not a severe restriction to the amateur builder.

Experience with a wide variety of rocket engine designs leads to the following recommendations for amateur rocket engines:

1. The combustion chamber and nozzle should be machined in one piece, from copper.
2. Those injector parts in contact with the hot chamber gases should also be machined from copper.
3. The cooling jacket and those injector parts not in contact with the hot propellant gases, should be fabricated from brass or stainless steel.
4. Expert machine and welding work is essential to produce a safe and useable rocket engine. Shoddy or careless workmanship, or poor welds, can easily cause engine failure.

INJECTORS

The function of the injector is to introduce the propellants into the combustion chamber in such a way that efficient combustion can occur. There are two types of injectors which the amateur builder can consider for small engine design. One of these is the impinging stream injector in which the oxidizer and fuel are injected through a number of separate holes so that the resulting streams intersect with each other. The fuel stream will impinge with the oxidizer stream and both will break up into small droplets.

When gaseous oxygen is used as the oxidizer, and a liquid hydrocarbon is used as the fuel, the impingement of the liquid stream with the high velocity gas stream results in diffusion and vaporization, causing good mixing and efficient combustion. A disadvantage of this type of injector is that extremely small holes are required for small engine flow rates and the hydraulic characteristics and equations normally used to predict injector parameters do not give good results for small orifices. The small holes are also difficult to drill, especially in soft copper.

However, to provide a complete picture of the equations used in rocket engine design, we present below the equation for the low of liquid through a simple orifice (a round drilled hole, for example)

$$w = C_d A \sqrt{2g\rho\Delta P} \tag{25}$$

where
w = propellant flow rate, lb/sec
A = area of orifice, ft2
DP = pressure drop across orifice, lb/ft2
∟ = density of propellant, lb/ft3
g = gravitational constant, 32.2 ft/sec2
Cd = orifice discharge coefficient

The discharge coefficient for a well-shaped simple orifice will usually have a value between 0.5 and 0.7. The injection velocity, or velocity of the liquid stream issuing from the orifice, is given by Injection pressure drops of 70 to 150 psi, or injection velocities of 50 to 100 ft/sec, are usually used in small liquid-fuel rocket engines. The injection pressure drop must be high enough to eliminate combustion instability inside the combustion chamber but must not be so high that the tankage and pressurization system used to supply fuel to the engine is penalized.

A second type of injector is the spray nozzle in which conical, solid cone, hollow cone, or other type of spray sheet can be obtained. When a liquid hydrocarbon fuel is forced through a spray nozzle (similar to those used in home oil burners) the resulting fuel droplets are easily mixed with gaseous oxygen and the resulting mixture readily vaporized and burned. Spray nozzles are especially attractive for the amateur builder since several companies manufacture them commercially for oil burners and other applications. The amateur need only determine the size and spray characteristics required for his engine design and the correct spray nozzle can then be purchased at a low cost. Figure illustrates the two types of injectors.

DESIGN VALUES

EQUATION REFERENCE : Rocket Lab (California) –NASA and US Navy -liquid propellant rockets.

COMBUSTION SECTION

Pc = 20.40 atm
Tc = 3172.22 o C
(Gaseous Oxygen and Butane standard mixture ratio at 2.5 from the performance values, pressure and temperature values are determined)
Ac = 3872.67 mm2
Length of combustion chamber, Lc = 144 mm
Diameter of the combustion chamber Dc = 70 mm

WALL THICKNESS

Thickness at inlet, twinlet = 30 mm
Thickness at outlet, tw outlet = 25 mm
(These are minimum thickness that calculated from design equations, for fabrication the values should be greater than this)

NOZZLE SECTION

At = 154.9 mm2
γ = 1.2
We are assuming the thrust that want to produce as 89 newton.
Wt = 0.342 N/sec
Tt= 2883.55 o C
Pt= 11.5056atm
At the exit portion,
Me = 2.55 mach
Ae/At = 3.65
Ae = 3870.97 mm2
De = 70.20 mm
Nozzle Convergence angle, β = 60o
(From the comparison of many existing design angles which giving maximum performance)
Nozzle Convergence angle, α = 15o
(This value is selected to prevent nozzle internal flow losses)

OXYGEN INPUT

For common engines, the pressure values are in between 7 to 15 psi ie, maximum 7atm .Here we want a swirling vortex flow at 45o angle. For proper swirling Pressure must be about 10 atm.

CATIA DESIGN

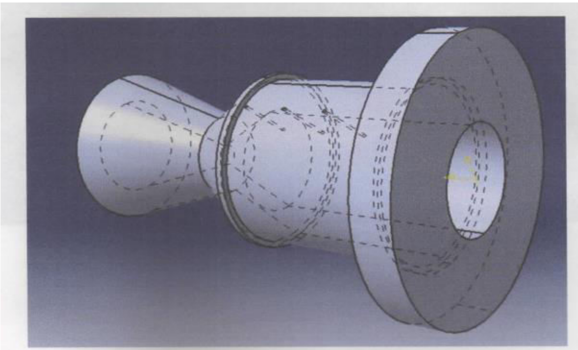


Figure 1: CatPart design of engine I

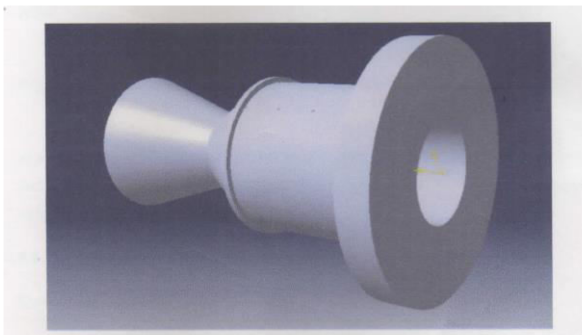


Figure 2: CatPart design of engine II

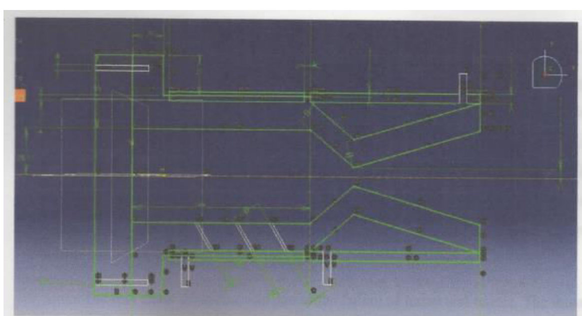


Figure 1: Sectional view.

MATERIAL SELECTION

Material selection may be categorized as one among the toughest task or stage while carrying out an engine manufacturing process. It is the most risky part of all stages and requires greater care and huge amount of evaluations and enquires. This is the stage that predicts the failure or success of an engine to certain extends. The material used for machine our rocket engine is being chosen after considering a lot of criterions, The basic idea behind our project work is that once an attempt was made by a group of students and was a failure.

The root cause of failure was due to the failure of throat and nozzle. The throat section was unable to withstand such a tremendous amount of heat and failed. So we investigated their work and found out that their engine was made using En8.

PROPERTIES OF En8

En8 is a medium strength steel, good tensile strength, also known as 080M40 and is an unalloyed medium carbon steel, It is normally suitable for stressed pins, shafts studs, etc. It is available as normalized or rolled and is supplied as plates, round hot rolled, hexagon, flat, square and round drawn or tuned. En8 have high carbon contents and are alloyed with Cr, Ni and Mo. This makes them highly hardenable steels which influences the rate at which they produce. We decided to give a try using the same material with better modifications so as to avoid throat failure. Henceforth, we used a better heat resistant material to machine the nozzle section. The material we opted was H 13 the maximum heat resistant material to be machined in a common lathe.

PROPERTIES OF H13

Composition in percentage
Chemical Composition

C	Si	Mn	P	S	Cr	Mo	V
0.32-0.45	0.80-1.20	0.20 - 0.50	<=0.30	<=0.30	4.75 - 5.50	1.10 - 1.75	0.80 - 1.20

H13 has high hardenability, excellent wear resistance, and hot toughness. It has good thermal shock,nitriding will improve hardness but diminishes shock resistance. H13 has greater homogeneity and an exceptionally fine structure, resulting in improved machinability, polishability and high temperature tensile strength.

Physical properties.

Physical Properties	Value
Density7.8G/CC	7.8g/cc
Heat Capacity	0.46J/g-c
Thermal Conductivity	24.3W/m-K

FABRICATION

LATHE

FIRST STEP:

The primary step involved in machining is that the required amount of raw material as per the catia design is brought.

SECOND STEP:

Outer and inner section of the combustion section was machined using En8 alloy. It
took about 3 hours of machining in lathe.

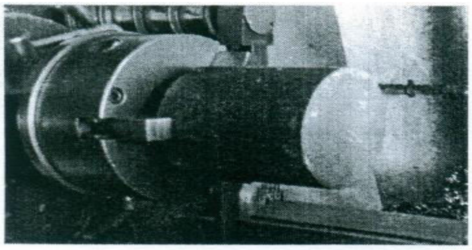


Figure 15: Machining the outer section

THIRD STEP:

A chamber to supply high pressure oxygen was machined using Mspipe.

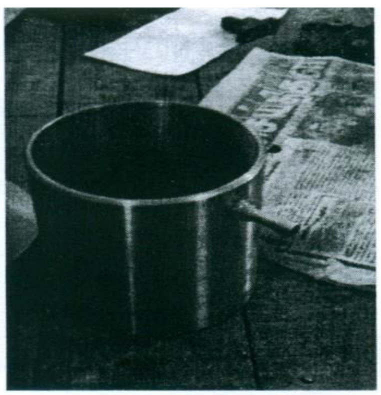


Figure 18: Camber for pressurization of oxygen supply

FOURTH STEP:
Inner and outer section of nozzle was machined using lathe. It was the most timeconsuming part and it took about 6 hours to complete it.

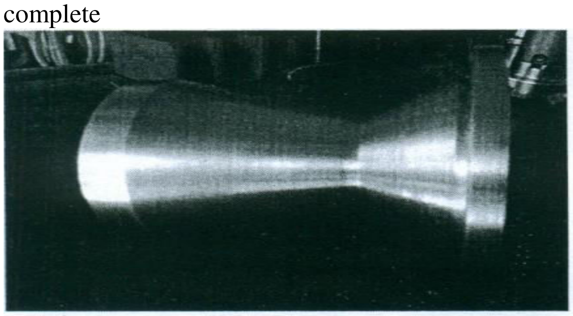


Figure 20: Outer section of nozzle section

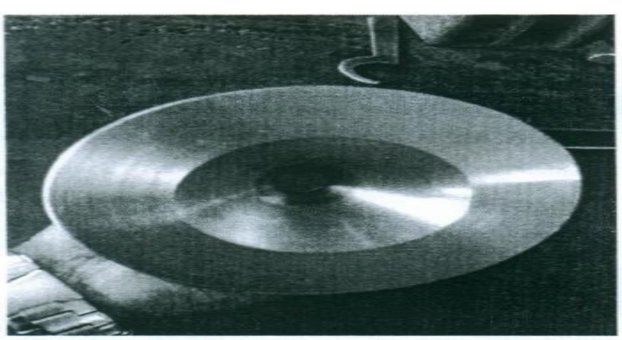


Figure 21: Inner section of nozzle

FIFTH STEP

Inner and outer nozzle cover to provide water cooling was machined using lathe.

SIXTH STEP

The final step in lathe was that the chamber head was made and a hole was made in the head so as to ensure the availability of fuel within the combustion section.

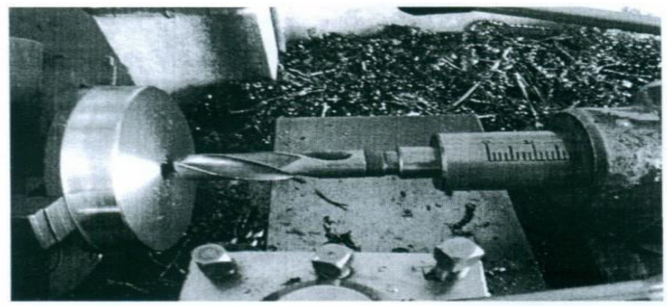


Figure 22: Making hole in the head.

WELDING

These individual sections were joint together using welding. Tig welding was predominantly used at those joints and brass welding was also used. Tig welding was used at joints that have to withstand tremendous amount of heat. Brass welding was utilized at portions that require to withstand high pressure. The cooling pipes were welded using normal welding process.
Gas tungsten arc welding (GTAW), also known as tungsten inert gas (TIG) welding, IS an arc welding process that uses a nonconsumable tungsten electrode to produce the weld. The weld area is protected from atmospheric contamination by an inert shielding gas (argon or helium), and a filler metal is normally used, though some welds, known as autogenous welds, do not require it. A constant-current welding power supplyproduces energy which is conducted across the arc through a column of highly ionized gas and metal vapors known as plasma.

DRILLIG
Drilling was used majorly at two portions. Firstly, tangential holes were drilled out at the inner walls of the combustion section to an outer oxidizer chamber(where it is stored) so as to provide oxidizer tangentially and to generate a vortex flow within the combustion section.

Figure 24: Drilling

In total it took about 18 hours for lathe and 2 hours for welding and in total 20
hours of machining was required. The clear picture of the complete engine after manufacturing is depicted in the figure below.

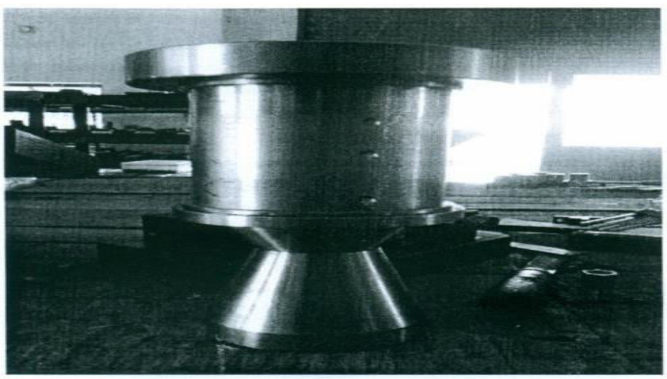


Figure 25: Engine

CONCLUSION AND DISCUSSION

We had used the basic ideologies theoretically presented before us to manufacture a swirling rocket engine. With the efficient use of a vortex flow we could reduce the heat transfer to the liners and it provides effective cooling; there by reduce the need of an effective cooling system. The recirculation present within the flow also allows it to undergo complete combustion. Conventionally, for an engine, ¼ of its total weight is of its cooling system. By the use of a swirling mechanism we could reduce the material used.

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