International Journal of Advances in Engineering and Management (IJAEM)
Volume 4, Issue 1, Jan 2022 pp: 1097-1111 www.ijaem.net ISSN: 2395-5252

DESIGN OF A 2KN LIQUID-FUEL ROCKET ENGINE COMBUSTION CHAMBER – PART 1

Abdulkadir A. Bage [1]; Abdulrahman Abdulazeez [2]; Dr. Olusegun S. Sholiyi [3]
[1]Assistant Chief Engineer, Centre for Space Transport and Propulsion Epe, National Space Research and Development Agency, NIGERIA
[2]Research Fellow, National Centre for Petroleum Research & Development Bauchi, NIGERIA
[3]Director/CEO, Centre for Space Transport and Propulsion Epe, National Space Research and Development Agency, NIGERIA

Submission: 18-01-2022 Revised: 28-01-2022 Acceptance: 31-01-2022

ABSTRACT
This paper presents a preliminary design and simulation for a small combustion chamber designed using SolidWorks. The liquid rocket engine motor is designed as a pressure fed (blow-down) system, and is expected to operate at a chamber pressure of 1MPa, whilst providing a thrust of 2KN, the operating time was designed not to exceed 10s. The expansion ratio of combustion chamber is evaluated at 2.24 with a target combustion efficiency of 95% and nozzle efficiency of between 80-90% all of these corresponded to a specific impulse of (I_{sp}) of 215.15s, while the combustion chamber was designed to utilize a combination of gaseous oxygen (GOX) and Kerosene ($C_{12}H_{26(liq)}$) as its propellant. The design was intended for a semi static test with steel (AISI 1035 SS) specified as the material for the design. The combustion chamber weighed 6.4132kg with a designed thrust-to-weight ratio on the chamber of 31.8; the maximum resultant displacement was 0.0176436mm with a maximum Von Mises stress of 38.67N/mm^2.
Keywords: *Liquid fuel rocket engine, combustion chamber,*

I. INTRODUCTION

A liquid rocket engine converts the chemical energy (propellants) contained in the tanks into a propulsive force that will propel the vehicle from one place to another (Rocketlab, 2003). The most encompassing design goal for the LRE was to meet all of the mission requirements at the least mission life cycle cost. Desired characteristics of the LRE included high thrust per unit flow-rate (specific impulse, I_{sp}), simplicity, robustness, reliability of the power-cycle and components. The tasks of engine preliminary design and optimization as well as the judicious selection of components and component arrangements so that all the desired qualities enumerated above could be fully refined and mission goals satisfied within minimum investment and risk.

Advantages of liquid propellants
- High specific impulse
- High thrust to weight ratio of the rockets
- Ease to control; thrust could be controlled by dosing propellant flow ratio with valves

Disadvantages of liquid propellants
- Complexity of construction
- They are difficult to scale, as complexity grows quickly with growing thrust
- Some propellants are toxic (hydrazine and its variants) or cryogenic (hydrogen)

Normally liquid-fuel rocket engines comprise of an injector, a combustion chamber and a nozzle, as shown in Figure 1.

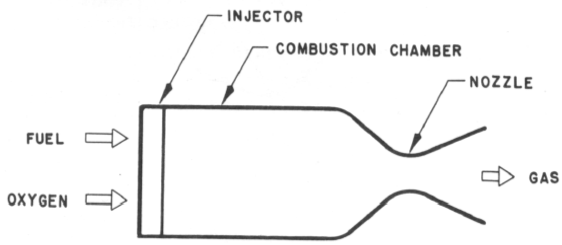


Figure 1: Typical Rocket Motor (Rocketlab, 2003)

The combustion chamber is where propellants are burned under high pressure. The primary function of the rocket combustion chamber is to confine the immense pressure and high

International Journal of Advances in Engineering and Management (IJAEM)
Volume 4, Issue 1, Jan 2022 pp: 1097-1111 www.ijaem.net ISSN: 2395-5252

temperature created during the combustion process. Rocket combustion chambers and nozzles are frequently cooled due to the high temperature generated by combustion process. Another crucial property of combustion chambers is that they must be long enough to ensure that the propellants are completely consumed before the exhaust gases enter the nozzle. The rocket nozzles' job is to transform chemical and thermal energy produced in the combustion chamber into kinetic energy. The nozzle converts the combustion chamber's slow-moving, high-pressure, high-temperature gas into a high-velocity, lower-pressure, lower temperature gas. A very high gas velocity is desirable because thrust is the product of mass (the amount of gas flowing through the nozzle) and velocity. As indicated in Figure 2, nozzles have a convergent and divergent section. The nozzle throat is the smallest flow area between the convergent and divergent sections. The flow area at the end of the diverging part of the nozzle is known as the exit area. The nozzle exit area is normally large enough that the pressure in the combustion chamber is decreased to the pressure outside the nozzle at the nozzle exit. At sea level, the exit pressure of the rocket engine is about 14.7 pounds per square inch (psi).

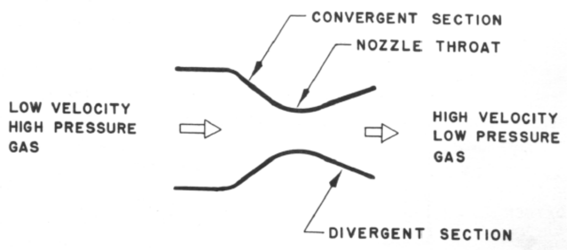


Figure 2: De-Laval Nozzle (Rocketlab, 2003)

II. MATERIALS AND METHODS

Fundamentals

Rocket propulsion is based on fundamental concepts of mechanics, thermodynamics, and chemistry. Propulsion is accomplished by applying a force to a vehicle, either to accelerate it or to maintain a constant velocity against a resisting force. Ejecting propellant at a high velocity produces this propulsive force. As with all designs, it is crucial that parameters related with the established objectives are properly defined; as a result, this part will focus on definitions and basic relationships in rocket propulsion, as well as other fundamental parameters.

Definitions

The *total impulse* $\boldsymbol{I_t}$, is the sum of the *thrust force* $\boldsymbol{F}$ (which varies with time) integrated over the *burning time* $\boldsymbol{t}$:

$$I_t = \int_0^t F dt \ldots\ldots (1)$$

For constant thrust and negligible start and stop transients this reduces to:

$$I_t = Ft \ldots\ldots (2)$$

$\boldsymbol{I_t}$ is proportionate to the total energy released in a propulsion system by all propellants. The overall impulse per unit weight of propellant is measured by the *specific impulse* $\boldsymbol{I_s}$. it is a key parameter for a rocket propulsion system's performance:

$$I_{sp} = \frac{\int_0^t F dt}{g_0 \int \dot{m} dt} \ldots\ldots (3)$$

Where

$g_0 =$ acceleration due to gravity g0 is 9.8066m $/\mathrm{sec}^2$

$\dot{m} = total\ mass\ flow\ rate\ of\ propellant$

For any rocket propulsion system, Equation (3) relates the time-averaged specific impulse value. When the thrust is constant and the propellant flow is constant, Equation (3) can be reduced to:

$$I_s = \frac{I_t}{(M_p g_0)} \ldots\ldots (4)$$

$$I_s = \frac{F}{\dot{m} g_0} = \frac{F}{\dot{w}} \ldots\ldots (5)$$

$$I_s = \frac{I_t}{M_p g_0} = \frac{I_t}{w} \ldots\ldots (6)$$

Where

$M_p g_0$ $= the\ total\ effective\ propellant\ weight\ w$

$\dot{w} = the\ propellant\ weight\ flow\ rate$

The effective exhaust velocity, abbreviated as c, is the average equivalent velocity at which propellant is ejected from the vehicle and is defined as:

$$C = I_{sp} g_0 = \frac{F}{\dot{m}} \ldots\ldots (7)$$

The *mass ratio* **MR** of a rocket vehicle is defined as the final mass $\boldsymbol{M_f}$ divided by the initial mass $\boldsymbol{M_0}$ of the rocket:

$$MR = M_f / M_0 \ldots\ldots (8)$$

When all of the useable propellant mass $\boldsymbol{M_p}$ has been spent and ejected from the rocket, the final mass $\boldsymbol{M_f}$ is the mass of the rocket vehicle. The final rocket vehicle mass $\boldsymbol{M_f}$ includes all non-fuel components such as guidance devices, navigation gear and payload. ***MR*** can range from 60% for tactical missiles to less than 10% for some unmanned launch vehicles. This mass ratio is a crucial factor to consider when evaluating flight performance. A full propulsion system's impulse to weight ratio is defined as the total impulse divided by the whole weight I_t and is divided by w_0, the vehicle's initial or propellant-loaded weight. A high value suggests a well-designed product. This can be represented in steady state terms as:

$$\frac{I_t}{w_0} = \frac{I_t}{(M_f + M_p)g_0} \dots\dots (9)$$

$$\frac{I_t}{w_0} = \frac{I_s}{\frac{M_f}{M_p} + 1} \dots\dots (10)$$

The thrust to weight ratio ***F/w₀*** expresses the acceleration that the engine is capable of giving to its own loaded propulsion system mass, at constant thrust the highest value of thrust to weight ratio occurs right before termination or burn out since the vehicle mass has been reduced by the mass of useable propellant. The thrust is the force exerted on a vehicle by a rocket propulsion system, or the reaction felt by its structure as a result of the expulsion of mass at high velocity. The thrust, due to a change in momentum, is given as:

$$F = \frac{dm}{dt} v_2 = \dot{m} v_2 = \frac{\dot{w}}{g_0} v_2 \dots\dots (11)$$

The gas exit velocity is uniform and axial, and the thrust and mass flow are constant. When the nozzle exit pressure equals the ambient pressure, this force represents the overall propulsion force.

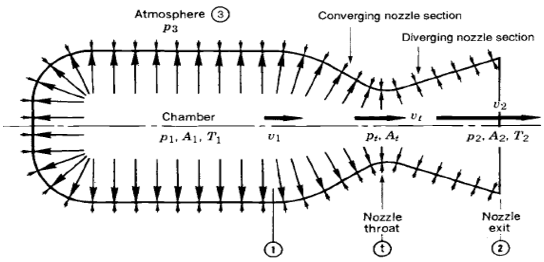


Figure 3: Schematics of pressure balance on chamber and nozzle wall (George, et al., 2001)

The second factor that influences thrust is the pressure of the surrounding fluid (local atmosphere). The external pressure acting uniformly on the exterior surface of a rocket chamber and the gas pressures on the inside of a typical thermal rocket engine are shown in Figure 3. The relative magnitude of the pressure forces is shown by the size and magnitude. There can be an imbalance between the external environment or atmospheric pressure P3 and the local pressure P2 of the hot gas jet at the nozzle's exit plane due to fixed nozzle geometry and changes in ambient pressure due to differences in altitude. Thus, the overall thrust of a continuously working rocket propulsion system travelling through a homogenous atmosphere is equal to:

$$F = \dot{m} v_2 + (P_2 - P_3)A_2 \dots\dots (12)$$

The first term is the momentum thrust, which is calculated by multiplying the propellant mass flow rate by its exhaust velocity relative to the vehicle. The pressure thrust is represented by the product of the cross-sectional area at nozzle exit A_2 (where the exhaust jet leaves the vehicle) and the difference between the exhaust gas pressure at the exit and the ambient fluid pressure. The pressure term is 0 and thrust is the same as in equation when the ambient atmospheric pressure equals the exhaust pressure in Equation (11). $P_3=0$ in vacuum and the thrust is:

$$F = \dot{m} v_2 + P_2 A_2 \dots\dots (13)$$

The rocket nozzle with optimum expansion ratio is defined as the pressure state in which the exhaust pressure is exactly matched to the surrounding fluid pressure (P2=P3). Because any change in ambient pressure affects the pressure thrust, Equation (12) shows that the thrust of a rocket unit is independent of the flight velocity; similarly the thrust of a rocket unit varies with altitude because atmospheric pressure decreases as altitude increases; as a result, the thrust and specific impulse will increase as the vehicle is propelled to higher altitudes. The change in pressure thrust caused by altitude fluctuations can be as much as 10% to 30% of the overall thrust. The effective exhaust velocity as defined by equation (7) applies to all rockets that thermodynamically expand hot gases in a nozzle, consequently from Equation (12) and at constant propellant mass flow the effective exhaust velocity can be modified to:

$$c = v_2 + (p_2 - p_3) A_2 / \dot{m} \dots\dots (14)$$

The effective exhaust velocity c equals the average actual exhaust velocity of the propellant gases v2 when P2=P3. When $P_2 \neq P_3$ then $c \neq v_2$, Because the second term of Equation (14) is relatively small in relation to v2, the effective exhaust velocity is usually close to the actual exhaust velocity when, as a result, when $c=v_2$ the thrust from equation (12) is given as:

$$F = (\dot{w}/g_0) v_2 = \dot{m} c \dots\dots (15)$$

The characteristic velocity ***c****, pronounced "cee-star", is defined as:

$$c^* = p_1 A_t / \dot{m} \dots\dots (16)$$

When comparing the relative performance of different chemical rocket propulsion systems designs and propellants, the characteristic velocity is used.

Nozzle Theory and Thermodynamic Relations

The thermodynamic relationship between the processes inside a rocket nozzle and chamber gives the mathematical tolls needed to calculate the

International Journal of Advances in Engineering and Management (IJAEM)
Volume 4, Issue 1, Jan 2022 pp: 1097-1111 www.ijaem.net ISSN: 2395-5252

performance of rocket propulsion systems and establish numerous critical design parameters.

The Ideal Rocket

The concept of an ideal rocket propulsion system is useful because the underlying basic thermodynamic principles can be expressed as mathematically. The equations theoretically describe a quasi-one-dimensional nozzle flow, which is an idealization and simplification of the full two- or three-dimensional equations and the real aero-thermo-chemical behavior. With the assumptions and simplifications listed below it is very adequate for obtaining useful solutions to many rocket propulsion systems, with actual measured performance for chemical rocket propulsion being between 1 and 6 percent below the calculated ideal value (George, et al., 2001).

Ideal Rocket Assumptions

An ideal rocket is one for which the following assumptions are valid:

1. The working substance (or chemical reaction products) is homogeneous.
2. All the species of the working fluid are gaseous.
3. The working substance obeys the perfect gas law.
4. There is no heat transfer across the rocket walls; therefore, the flow is adiabatic.
5. There is no appreciable friction and all boundary layer effects are neglected.
6. There are no shock waves or discontinuities in the nozzle flow.
7. The propellant flow is steady and constant.
8. All exhaust gases leaving the rocket have an axially directed velocity.
9. The gas velocity, pressure, temperature, and density are all uniform across any section normal to the nozzle axis.
10. Chemical equilibrium is established within the rocket chamber and the gas composition does not change in the nozzle (frozen flow) (George, et al., 2001).

The idealized theory for a liquid propellant rocket assumes an injection system in which the fuel and oxidizer are perfectly blended, resulting in a homogenous working substance. Postulates 4, 5, and 6 allow for the employment of isentropic expansion relations in the rocket nozzle, representing the highest conversion of heat to kinetic energy; this also indicates that the nozzle flow is thermodynamically reversible (George, et al., 2001).

Summary of Thermodynamic Relations

Basic relationships required for the development of the nozzle flow equations is summarized below.

The notion of energy conservation can easily be applied to the adiabatic, no-shaft-work process inside the nozzle. Inflow systems, the concept of enthalpy is useful; the enthalpy consist of internal thermal energy plus the flow work. Enthalpy can be stated for ideal gases as the product of specific heat c_p and the absolute temperature T. the stagnation enthalpy per unit mass h_0 is constant under the above assumptions, as illustrated:

$$h_0 = h + v^2/2 = constant \ldots\ldots (17)$$

Where:

$h_0 = stagnation\ enthalpy$

$h = enthalpy$

$v = gas\ velocity$

The conservation of energy for isentropic flow between any two sections x and y demonstrates that a drop in enthalpy or thermal content of the flow appears as an increase in kinetic energy and that any changes in potential energy can be ignored.

$$h_x - h_y = \frac{1}{2}\left(v_y^2 - v_x^2\right) = c_p\left(T_x - T_y\right) \ldots\ldots (18)$$

The continuity equation expresses the idea of mass conservation in a constant flow with a single input and outflow by equating the mass flow rate at any segment x to that at every other section y.

$$\dot{m}_x = \dot{m}_y \equiv \dot{m} = Av/V \ldots\ldots (19)$$

Where:

$A = cross\ sectional\ area$

$V = specific\ volume$

The perfect gas law (Smith, et al., 2001) is given as:

$$p_x V_x = RT_x \ldots\ldots (20)$$

The gas constant R is calculated by dividing the universal gas constant R' by the molecular mass $\mathfrak{M}$ of the flowing gas mixture. For ideal gases, the specific heat at constant pressure "c_p", constant volume "c_v", and their ratio constant "k" are constant throughout a wide range of temperatures and are connected as follows:

$$k = c_p/c_v \ldots\ldots (21)$$

$$c_p - c_v = R \ldots\ldots (22)$$

$$c_p = kR/(k-1) \ldots\ldots (23)$$

For any isentropic process the following relations hold between any two points x and y (Smith, et al., 2001):

$$T_x/T_y = \left(p_x/p_y\right)^{(k-1)/k} = \left(V_y/V_x\right)^{k-1} \ldots\ldots (24)$$

In an isentropic nozzle expansion, the pressure decreases significantly, while the absolute temperature decreases somewhat and the specific volume increases. The prevailing conditions when a flow is stopped isentropically are known as stagnation conditions, which are denoted by

International Journal of Advances in Engineering and Management (IJAEM)
Volume 4, Issue 1, Jan 2022 pp: 1097-1111 www.ijaem.net ISSN: 2395-5252

subscript "0". The energy equation yields the stagnation temperature T_0 as:

$$T_0 = T + v^2/2c_p \dots\dots (25)$$

Where:

$T = absolute\ fluid\ static\ temperature$

The following is the relationship between the stagnation pressure and the local pressure in the flow:

$$p_0/p = \left[1 + v^2/(2c_pT)\right]^{k/(k-1)} = \left(V/V_0\right)^k \dots\dots (26)$$

(George, et al., 2001) (Dieter, et al., 1992) (Bianchi)Defined the velocity of sound "α" or the acoustic velocity in ideal gases is independent of pressure and is defined as:

$$a = \sqrt{kRT} \dots\dots (27)$$

The Mach number M is a dimensionless flow parameter and is used to define the ratio of the flow velocity "v" to the local acoustic velocity "α" (George, et al., 2001) (Smith, et al., 2001):

$$M = v/a = v/\sqrt{kRT} \dots\dots (28)$$

A Mach number less than one correspond to subsonic flow and greater than one to supersonic flow, while at Mach number of one the flow is moving at precisely the velocity of sound often referred to as speed of sound (George, et al., 2001) (Dieter, et al., 1992) (Smith, et al., 2001). The relation between stagnation temperature and Mach number is given as:

$$T_0 = T\left[1 + \frac{1}{2}(k-1)M^2\right] \dots\dots (29)$$

$$M = \sqrt{\frac{2}{k-1}\left(\frac{T_0}{T} - 1\right)} \dots\dots (30)$$

The temperature and pressure stagnation values are designated as T_0 and p_0. The stagnation pressure during an adiabatic nozzle expansion, unlike the temperature, remains constant only for isentropic flows and may be calculated using the equation below:

$$p_0 = p\left[1 + \frac{1}{2}(k-1)M^2\right]^{k/(k-1)} \dots\dots (31)$$

(George, et al., 2001) Stated that the area ratio of a nozzle with isentropic flow can be expressed in terms of Mach number for any points x and y within the nozzle, this relationship, along with those for the ratios T/T_0 and p/p_0, is shown in the figure below for $A_x=A_t$ and $M_x=1.0$, otherwise:

$$\frac{A_y}{A_x} = \frac{M_x}{M_y}\sqrt{\left\{\frac{1 + \left[\frac{(k-1)}{2}\right]M_y^2}{1 + \left[\frac{(k-1)}{2}\right]M_x^2}\right\}^{(k+1)/(k-1)}} \dots\dots (32)$$

From Figure 4; it is obvious that for subsonic flow the chamber ratio $\boldsymbol{A_1/A_t}$ can be small, with values ranging between 3 and 6; and the passage is convergent; with no noticeable effects from the variation of $\boldsymbol{k}$. With supersonic flow the nozzle section diverges and the area ratio becomes large very quickly and is significantly influenced by the value of $\boldsymbol{k}$ (George, et al., 2001) (Dieter, et al., 1992) (Turner, 2005)

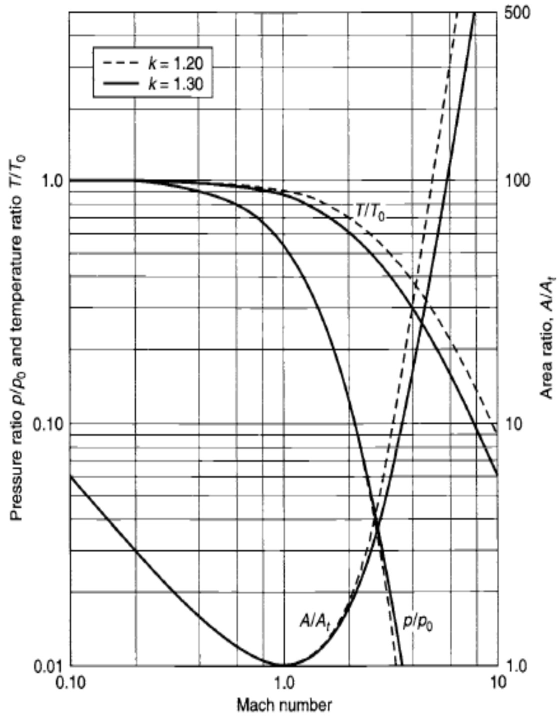


Figure 4: Relationship of area ratio; pressure ratio and temperature ratio as functions of Mach number in a De-Laval nozzle for the subsonic and supersonic nozzle regions (George, et al., 2001).

Isentropic Flow through Nozzles

In a converging-diverging nozzle a considerable portion of the gases in the chamber is transformed into kinetic energy which causes the gas pressure and temperature to decrease dramatically and the gas velocity to exceed two miles per second. Isentropic flow is a reversible process. The conservation of total or stagnation

International Journal of Advances in Engineering and Management (IJAEM)
Volume 4, Issue 1, Jan 2022 pp: 1097-1111 www.ijaem.net ISSN: 2395-5252

enthalpy h0 can be used to calculate flow velocity (Smith, et al., 2001):

$$v_2 = \sqrt{2(h_1 - h_2) + v_1^2} \ldots\ldots (33)$$

$$v_2 = \sqrt{\frac{2k}{k-1} RT_1 \left[1 - \left(\frac{p_2}{p_1}\right)^{\frac{(k-1)}{k}}\right] + v_1^2} \ldots\ldots (34)$$

This equation holds for any two positions within the nozzle and applies to both ideal and non-ideal rockets, with the subscripts 1 and 2 representing the nozzle inlet and exit conditions, respectively. The chamber velocity is small for a reasonably large chamber section compared to the nozzle throat; hence the term v_1 can be ignored (George, et al., 2001):

$$v_2 = \sqrt{\frac{2k}{k-1} RT_1 \left[1 - \left(\frac{p_2}{p_1}\right)^{\frac{(k-1)}{k}}\right]} \ldots\ldots (35)$$

The chamber temperature T_1 at the nozzle inlet under isentropic conditions, differ a little from stagnation temperature (George, et al., 2001) thus equation (35) may be re-written as:

$$v_2 = \sqrt{\frac{2k}{k-1} \frac{R'T_0}{\mathfrak{M}} \left[1 - \left(\frac{p_2}{p_1}\right)^{\frac{(k-1)}{k}}\right]} \ldots\ldots (36)$$

An increase in the ratio of T_0/M will increase the performance of the rocket and plays an important role in optimizing the mixture ratio in chemical rockets. The influence of the pressure ratio p_0/p_e and of the specific heat ratio is less pronounced.

Nozzle Flow and Throat Condition

The De-Laval nozzles consist of a convergent section followed by a divergent section, and the area is inversely proportional to the ratio v/V, according to the continuity equation. This may be observed in Figure 5, which exhibits a maximum in the v/V curve because the velocity initially grows at a faster rate than the specific volume, but the specific volume increases at a faster rate in the divergent section (George, et al., 2001) (Bianchi) (Turner, 2005). The throat region is the smallest nozzle area. The nozzle area expansion ratio (ε) is a critical nozzle design statistic that compares the nozzle exit area A_2 to the throat area A_t:

$$\varepsilon = \frac{A_2}{A_t} \ldots\ldots (37)$$

At the throat, where there is a unique gas pressure ratio that is only a function of the ratio of specific heats k, the maximum gas flow per unit area occurs; consequently, the pressure ratio is calculated by setting M=1 in equation (31) (Smith, et al., 2001) (Rocketlab, 2003) (Turner, 2005):

$$p_t/p_1 = [2/(k+1)]^{k/(k-1)} \ldots\ldots (38)$$

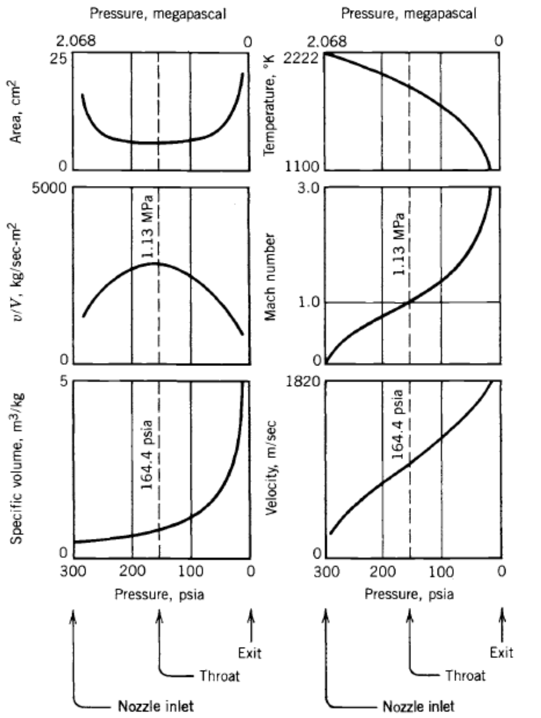


Figure 5: Typical values of cross-sectional area, temperature, specific volume and velocity with pressure in a rocket nozzle (Turner, 2005).

It is vital to remember that the throat (critical values) has the highest gas flow per unit area (Bianchi) (Dieter, et al., 1992), and the ratio of critical values to stagnation values is represented below:

$$\frac{p_t}{p_0} = \left[\frac{2}{(k+1)}\right]^{\frac{k}{k-1}} \quad (0.53 - 0.57)$$

$$\frac{T_t}{T_0} = \left[\frac{2}{(k+1)}\right] \quad (0.83 - 0.91)$$

$$\frac{\rho_t}{\rho_0} = \left[\frac{2}{(k+1)}\right]^{\frac{1}{k-1}} \quad (0.62 - 0.63)$$

The critical throat velocity v_t is:

$$v_t = \sqrt{\frac{2k}{k+1} RT_0} = a_t = \sqrt{kRT_t} \ldots\ldots (39)$$

This equation allows for quick evaluation of the throat velocity from nozzle inlet conditions.
To attain sonic/supersonic flow:

$$\frac{p_0}{p_e} \geq \left[\frac{(k+1)}{2}\right]^{\frac{k}{k-1}} \quad (1.75 - 1.89)$$

Mass Flow Rate through Nozzle

The steady state mass flow through the nozzle is:

$$\dot{m} = \rho v A = constant \dots\dots (40)$$

This is re-written as:

$$\frac{\dot{m}}{A} = p_0 \sqrt{\frac{2k}{k-1}\frac{1}{RT_0}\left(\frac{p}{p_0}\right)^{\frac{2}{k}}\left[1-\left(\frac{p}{p_0}\right)^{\frac{k-1}{k}}\right]} \dots\dots (41)$$

Equation (41) is referred to as the De Saint Venant's equation (Bianchi), and in terms of Mach number we have:

$$\frac{\dot{m}}{A} = p_0 \sqrt{\frac{k}{RT_0}} M(1+\delta M^2)^{-\frac{k+1}{(k-1)}} \dots\dots (42)$$

At the throat M=1

$$\frac{p_t}{p_0} = \left(\frac{2}{k+1}\right)^{\frac{k}{k-1}} \dots\dots (43)$$

Thus

$$\frac{\dot{m}}{A_t} = \Gamma \frac{p_0}{\sqrt{RT_0}} \dots\dots (44)$$

Note:

For a pressure ratio of 1, the mass flow rate per unit area is zero (no flow)

For a pressure ratio of 0, the mass flow rate per unit area is zero (expanded to vacuum)

Nozzle Configuration

There are a variety of proven nozzle configurations; common nozzles and chambers have a circular cross-section, a converging portion, a throat at the narrowest point, and a diverging part. The following are some highly interesting aspects to consider when it comes to nozzle configurations:

A. The converging section between the chamber and the nozzle throat is not critical in achieving performance; thus the subsonic flow in this section can be easily tuned at low pressure drop within varying radius without performance losses (George, et al., 2001).

B. The throat contour also is not very critical to performance and any radius is usually acceptable (George, et al., 2001).

C. The pressure gradients within the converging section and the throat section are high as such the flow will adhere to the walls (George, et al., 2001).

D. The divergent supersonic flow-section wall surface of the nozzle must be smooth and shiny throughout to minimize friction, radiation, absorption and convective heat transfers due to surface roughness, gaps, holes, sharp edges or protrusions must be avoided (George, et al., 2001).

Below is a list of six (6) different major nozzle configurations:

1. Cone (15˚ half angle)
2. Contoured or bell shape full length
3. Contoured or bell shaped shortened
4. Plug or aero-spike full length
5. Plug or aero-spike truncated or shortened
6. Expansion and deflection

However for the purpose of this study the cone half angle configuration was adopted.

DESIGN EQUATIONS

This section shall focus on the detailed simplified equations for the design of a small liquid fuel rocket motor. The nomenclature for the design is as follows:

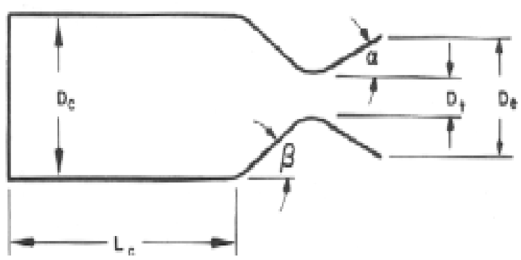


Figure 6: Motor Design Configuration (Rocketlab, 2003)

Where:
D=diameter
L=length
α,β=angles

Nozzle

Where the total propellant flow-rate is known and the propellant operating conditions have been specified, the nozzle throat cross-sectional area may be calculated using the following equations:

$$A_t = \frac{w_t}{P_t}\sqrt{\frac{RT_t}{k}} \dots\dots (49)$$

Where:

R = gas constant given by R = R'/M.

R is the universal gas constant, equal to 8314J/kmol.K and M is the molecular weight of the gas, k is the ratio of the specific heats and represents a thermodynamic variable for the combustion of hydrocarbon fuel and gaseous oxygen, k is set at 1.2. It is assumed that the perfect gas law theorem is followed. T_t being the temperature of gases at the nozzle throat, the gas temperature at the nozzle throat is less than that of the combustion chamber from losses due to thermal energy used in accelerating the gas to local speed of sound (Mach Number = 1) at the throat. Therefore from equation (29) where k = 1.2, T_t is approximately equal to:

$$T_t = 0.909T_0 \dots\dots (50)$$

Where

T_0 is the chamber flame temperature.

P_t being the pressure of gases at the nozzle throat

The gas pressure at the nozzle throat is equally less than that in the combustion chamber from losses and due to acceleration of the gas to the local speed

International Journal of Advances in Engineering and Management (IJAEM)
Volume 4, Issue 1, Jan 2022 pp: 1097-1111 www.ijaem.net ISSN: 2395-5252

of sound (Mach Number =1) at the throat. Therefore from equation (31) where k = 1.2, P_t is approximately equal to:

$$P_t = 0.564P_0 \dots\dots (51)$$

These hot gases will now be expanded in the diverging section of the nozzle to obtain maximum thrust. The pressure of these gases will decrease from losses in energy used to accelerate the gases; we must now find that area of the nozzle where the gas pressure is equal to the atmospheric pressure, this area will then be the nozzle exit area. Still from equation (31) the Mach number at the nozzle exit can be expressed in terms of the perfect gas law as (George, et al., 2001) (Rocketlab, 2003) (Turner, 2005):

$$M_e^2 = \frac{2}{k-1}\left[\left(\frac{P_0}{P_{atm}}\right)^{\frac{k-1}{k}} - 1\right] \dots\dots (52)$$

Where P_0 is the combustion chamber pressure and P_{atm} is the atmospheric pressure. From equation (32) the nozzle exit area corresponding to the exit Mach number where the Mach number at the throat is 1, is given as:

$$\frac{A_e}{A_t} = \frac{1}{M_e}\sqrt{\left\{\frac{1+\left[\frac{(k-1)}{2}\right]M_e^2}{1+\left[\frac{(k-1)}{2}\right]}\right\}^{(k+1)/(k-1)}} \dots\dots (53)$$

Since k is specified at 1.2 for gaseous oxygen and hydrocarbon propellants the Area ratio can be easily evaluated at atmospheric pressure given:

$$A_e = A_t\left(\frac{A_e}{A_t}\right) \dots\dots (54)$$

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

$$T_e = T_0\left(\frac{T_e}{T_0}\right) \dots\dots (55)$$

The nozzle throat and exit diameters are given by:

$$D_t = \sqrt{4A_t/\pi} \dots\dots (56)$$

$$D_e = \sqrt{4A_e/\pi} \dots\dots (57)$$

Where π = 3.142

Previous studies showed that a good value for the nozzle convergence half-angle β is 60, while the nozzle divergent half-angle α should be not more than 15 to prevent nozzle internal flow losses (Rocketlab, 2003) (Vigor, et al., 2004) (Turner, 2005).

Combustion Chamber

The characteristic chamber length L*, which is the parameter that describes the chamber volume necessary for full combustion is given as:

$$L^* = \frac{V_c}{A_t} \dots\dots (58)$$

Where:

V_c is the chamber volume (including the converging volume)

A_t is the nozzle throat area

An L* of 1270mm to 2540mm is adequate for gaseous oxygen and hydrocarbon fuels. L* is actually a replacement for determining the reacting propellants' residence time. The combustion chamber cross-sectional area should be at least three times the nozzle throat area to reduce losses due to gas flow velocity within the chamber.

The combustion chamber cross-sectional area is given by:

$$A_c = \frac{\pi D_c^2}{4} \dots\dots (59)$$

The chamber volume is given by:

$$V_c = A_c L_c + \mathit{convergent\ Volume} \dots\dots (60)$$

For small combustion chambers the convergent volume is about 1/10th the volume of the cylindrical portion of the chamber (George, et al., 2001) (Rocketlab, 2003) (Turner, 2005), so that:

$$V_c = 1.1(A_c L_c) \dots\dots (61)$$

In order for the injector to have useable face area, the chamber diameter for small combustion chambers (thrust levels less than 75 pounds) should be 3 to 5 times the nozzle throat diameter.

Chamber Wall Thickness

The combustion chamber must be able to withstand the internal pressure of the hot combustion gases, and since the combustion chamber must be physically attached to the cooling jacket the chamber wall thickness must be sufficient for welding or brazing purposes (Rocketlab, 2003). The working stress in the wall is provided by, because the chamber will be a cylindrical shell:

$$s = PD/2t_w \dots\dots (62)$$

Where

P= pressure in the combustion chamber (neglecting the effect of coolant pressure on the outside of the shell),

D= mean diameter of the cylinder, and

t_w= 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

$$t_w = PD/16000 \dots\dots (63)$$

This is the minimum thickness; actually the thickness should be somewhat greater to allow for welding, buckling, and stress concentration (Rocketlab, 2003). The chamber wall and nozzle are usually the same thickness. The water cooling jacket's wall thickness is similarly calculated using equation (61). Welding factors and design considerations will normally necessitate walls thicker than those indicated by stress calculations, the value of tw will be minimum thickness. If the jacket material differs from the nozzle material, a new allowable stress value will be utilized in equation (63) (Rocketlab, 2003).

Engine Cooling

There are basically two cooling methods, namely:

1. Steady state method
2. Transient heat transfer or unsteady heat transfer -heat sink

This study however considered the steady state method. In the steady state method, the heat transfer rate and the temperatures of the chambers reach thermal equilibrium; the steady state method can further be sub-divided into regenerative cooling and radiation cooling. Regenerative cooling is done by building a cooling jacket around the thrust chamber and circulating one of the propellants (usually the fuel) through it before it is fed to injector. This cooling method is used primarily with bi-propellant chambers of medium to large thrust; it has been effective in applications with high chamber pressure and high heat transfer rates, also most injectors use regenerative cooling. In radiation cooling the chamber and/or nozzle have only a single wall made of high temperature material. When it reaches thermal equilibrium, this wall usually glows red or white hot and radiates heat away to the surroundings or to empty space.

Materials

As with all rockets the combustion chamber and nozzle must be able to withstand relatively high temperature, high velocity, chemical erosion and high stress. The wall material must be capable of high heat transfer rates and at the same time have adequate strength to withstand the chamber combustion pressure (Rocketlab, 2003) (Turner, 2005). Material requirements are critical only to those parts which come into direct contact with propellant gases while other motor components can be made of conventional materials.

The following recommendations should be followed when building small rocket engines:

1. The combustion chamber and nozzle should be machined in one piece (copper is highly recommended).

2. The injector parts in contact with the hot chamber gases should be machined from copper if possible.

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

Propellants

The propellants which are the working substance of rocket engines, constitutes the fluid that undergoes chemical and thermodynamic changes (Rocketlab, 2003) and considering the ease of handling, a bi-propellant was selected for this particular project with the fuel being Kerosene (Dodecane) $C_{12}H_{26(liq)}$ and gaseous Oxygen $O_{2(gas)}$ as the oxidizer. The combustion equation for kerosene and gaseous oxygen is given as:

$$C_{12}H_{26(liq)} + 17.6O_{2(gas)} \rightarrow 12CO_{2(gas)} + 13H_2O_{(vap)}$$

For a given thrust F and a given effective exhaust velocity c, from equation (7) the total propellant mass flow m is given by:

$$\dot{m} = \frac{F}{c} \dots\dots (64)$$

Propellant Mixture Ratio

The propellant mixture ratio is defined as the ratio of the relative molecular mass of the oxidizer to that of the fuel, given mathematically as:

$$r = \frac{RMM_{oxidizer}}{RMM_{fuel}} \dots\dots (65)$$

The relations between the mixture ratio, the oxidizer and the fuel flow rates are as below:

$$\dot{m} = \dot{m}_o + \dot{m}_f \dots\dots (66)$$

$$\dot{m}_o = r\,\dot{m}/(r+1) \dots\dots (67)$$

$$\dot{m}_f = \dot{m}/(r+1) \dots\dots (68)$$

At optimum mixture ratio a stoichiometric ratio is achieved; that is; just enough oxygen is present to chemically react with all of the fuel and the highest flame temperature is achieved under this condition, however for safety purposes, the system was designed to operate at 95% efficiency "fuel rich" (having more fuel present than oxidizer), this condition is less severe on the rocket engine than burning at stoichiometric conditions.

The propellants were chosen for two reasons;

1. The propellants are readily available
2. The propellants are relatively cheap.

The combustion equation for Kerosene and gaseous Oxygen at 95% efficiency

Table 1: Propellant Mixture Ratio

Component	Oxygen	Kerosene
Chemical Formula	O2	C12H26
Relative Molecular Mass (g/mol)	563018	170.33
Density (kg/mol)	1142	810
Optimum Mixture ratio r	3.31	

DESIGN PARAMETERS

Table 2: Preliminary Design Parameters

Parameter	Requirements	Performance Characteristics
Operating Pressure Low pressure Assemblies High Pressure Assemblies	 1-2MPa 2-4MPa	 1-2MPa 2-4Mpa
Proof Pressure	Range Safety	1.25x MEOP of propellant tanks 1.50x MEOP of components, lines, fittings
Burst Pressure	Range Safety	1.5:1 propellant tanks 2.0:1 propellant control components 4.0:1 other components, lines and fittings
Propellants Kerosene Oxygen		 RMM 170.33gmol, Density 810kg/m3 RMM 563.18gmol, Density 1142kg/m3
Propellant Capacity	96% full	
Determination of Remaining Propellant		
System Mixture Ratio	3.31 Nominal	3.31 Nominal
LRE 2KN-Engine Thrust, N Isp, s Total Impulse, N-s	 2000 215.15 20000	 2000 215.15 20000
Leakage Requirements	$1x10^{-4}$ Pa-L/s	$1x10^{-4}$ Pa-L/s
Power		
Mass		

III. RESULTS AND DISCUSSION

Table 3: Parameters from PROPEP

PARAMETERS FROM PROPEP			
Parameters	**Symbol**	**Value**	**Unit**
Density of Propellant	ρ(Propellant)	0.2767	kg/m^3
Temperature of Chamber	T_c	3611.49	k
Pressure of Chamber	P_c	1000000	N/m^2
Ratio of Specific Heats	K	1.20	-
Molecular Mass of Propellant	MM	25	kg/kmol
Atmospheric Pressure	Patm	101352.9	N/m^2

Calculated parameters are given as:

Table 4: Calculated Parameters

CALCULATED PARAMETERS			
Parameters			
Total Impulse	I_t	20000.00	Ns
Mass Flow rate of Propellant	M_p	9.48	kg
Weight Flow rate of Propellant	W_p	9.30	N/s
Weight Flow rate of Kerosene	W_k	2.16	N/s
Weight Flow rate of Oxygen	W_o	7.14	N/s
Total Mass of Kerosene	M_k	2.20	kg
Total Mass of Oxygen	M_o	7.28	kg
Mass of Kerosene + Oxygen	M_k+M_o	9.48	kg
	m^3	**liter**	**US gal**
Volume of Kerosene	0.002715224	2.715224106	0.717286
Volume of Oxygen	0.006374595	6.374594877	1.683990
	1	**T**	**2**
Parameter	**Chamber**	**Throat**	**Exit**
Temperature (K)	3611.49	3283.17	2853.08
Pressure (N/m^2)	1000000.00	564473.93	243087.46
Mach Number	0.22	1.00	2.16
Area (m^2)	0.0225	0.0050	0.0112
Diameter (m)	0.1694	0.0799	0.1196
Radius (m)	0.0847	0.0400	0.0598
Velocity (m/s)	-	1144.65	1739.72
Density	0.28	0.17	0.09
Specific Volume (m^3/kg)	3.61	5.82	11.75
Temperature Ratio	Te/Tc	0.79	
	Tt/Tc	0.91	
Critical Pressure Ratio	Pt/Pc	0.56	

Nozzle Expansion Ratio (ε)	Ae/At	2.24
Nozzle Contraction Ratio (εc)	Ac/At	4.49
Density Ratio	ρt/ρc	0.62
Velocity Ratio	Ve/Vt	1.52

Table 5: Iteration for L* (El-Star)

Iteration For Determining L*					
L*	v_c	t_s	L_c	L_{conv}	L_c+L_{conv}
0.8	0.00401	0.00117	0.16182	0.02865	0.1905
0.9	0.00451	0.00132	0.18204	0.03224	0.2143
1	0.00502	0.00146	0.20227	0.03582	0.2381
1.1	0.00552	0.00161	0.22250	0.03940	0.2619
1.2	0.00602	0.00176	0.24273	0.04298	0.2857
1.3	0.00652	0.00190	0.26295	0.04656	0.3095
1.4	0.00702	0.00205	0.28318	0.05014	0.3333
1.5	0.00752	0.00220	0.30341	0.05373	0.3571
1.6	0.00803	0.00234	0.32364	0.05731	0.3809
1.7	0.00853	0.00249	0.34386	0.06089	0.4048
1.8	0.00903	0.00264	0.36409	0.06447	0.4286
1.9	0.00953	0.00278	0.38432	0.06805	0.4524
2	0.01003	0.00293	0.40454	0.07163	0.4762
2.1	0.01053	0.00307	0.42477	0.07522	0.5000
2.2	0.01104	0.00322	0.44500	0.07880	0.5238
2.3	0.01154	0.00337	0.46523	0.08238	0.5476
2.4	0.01204	0.00351	0.48545	0.08596	0.5714
2.5	0.01254	0.00366	0.50568	0.08954	0.5952
2.6	0.01304	0.00381	0.52591	0.09312	0.6190
2.7	0.01354	0.00395	0.54613	0.09671	0.6428
2.8	0.01405	0.00410	0.56636	0.10029	0.6666
2.9	0.01455	0.00425	0.58659	0.10387	0.6905
3	0.01505	0.00439	0.60682	0.10745	0.7143

From Table 4, 1.7 was selected as the optimal range for El-Star.

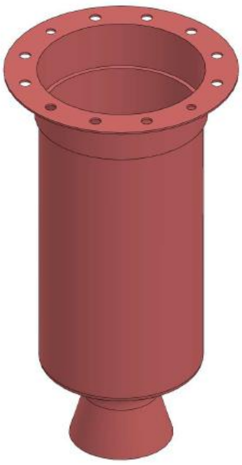
International Journal of Advances in Engineering and Management (IJAEM)
Volume 4, Issue 1, Jan 2022 pp: 1097-1111 www.ijaem.net ISSN: 2395-5252

Table 6: El-Star Parameters

L* PARAMETERS			
El-Star	L*	1.70	m
Chamber Length	L_c	0.34	m
Chamber Convergent Length	L_{conv}	0.06	m
Chamber Divergent Length	L_{div}	0.07	m
Chamber Divergent Length 80%	$L_{div\ 80\%}$	0.06	m
Volume of Chamber	V_c	0.009	m^3
Volume of Cylinder and Frustum	V_b	0.009	m^3
Conical Nozzle Contour	R	0.024	m

Table 2, specified the preliminary design parameters; these information were then used alongside the design equations to calculate the key combustion chamber parameters. Propep was used to evaluate the propellant thermodynamic properties, it can be seen from

Table 4, that the diameters of the combustion chamber, throat and exit area which are key parameters in the design of a typical De-Laval nozzle



Mass properties of New Chamber 2021
Configuration: Default
Coordinate system: -- default --
Density = 7850 kg/m^3
Mass = 6.41329 kg
Total weld mass = 0.00 grams
Volume = 0.000816979 m^3
Surface area = 568481.71 mm^2
Center of mass: (mm)
X = 0.00
Y = -160.36
Z = 0.00
Principal axes of inertia and principal moments of inertia:
(g * mm^2)
Taken at the center of mass
Ix = (0.00, 1.00, 0.00) Px = 39712394.34
Iy = (-1.00, 0.00, 0.00) Py = 140656496.46
Iz = (0.00, 0.00, 1.00) Pz = 140669611.90
Moments of inertia:

(g * mm^2)
Taken at the center of mass and aligned with the output coordinate system.

Lxx = 140656496.46	Lxy = -0.00	Lxz = 0.00	
Lyx = -0.00	Lyy = 39712394.34	Lyz = 0.00	
Lzx = 0.00	Lzy = 0.00		Lzz = 140669611.90

Moments of inertia:
(g * mm^2)
Taken at the output coordinate system

Ixx = 283113427.14	Ixy = -0.00	Ixz = 0.00	
Iyx = -0.00	Iyy = 39712394.34	Iyz = 0.00	
Izx = 0.00	Izy = 0.00		Izz = 283126542.58

were evaluated; an iterative method was used to determine El-Star (L*) which in turn was the basis for determining the combustion chamber length, chamber convergent length and chamber divergent length as well as conical nozzle contour. The values of these key parameters were then used to develop a 3D-CAD representation on SolidWorks.

The proposed material specified for the combustion chamber was specified as AISI 1035 Steel (SS) with a yield strength of 282.68 N/mm^2 and a tensile strength of 585 N/mm^2, the estimated mass of the combustion chamber was given as 6.41329kg, for a 2KN design specification, the parameters translated to a ***Thrust-to-Weight*** ratio of 31.80 on the chamber only, by convention this is a good starting point as most combustion chambers tend to be the heaviest part of the rocket which also doubles as a dead weight upon exhausting the usable propellant. The maximum Von Mises stress were 38.67 N/mm^2 which showed that the Von Mises stress were well below the limits for the chamber to yield or fracture, this might be due to the chamber wall thickness of 2.52mm. The design also had a maximum resultant displacement of 0.0176436mm, which was sufficient for the design to temporarily change shape (elastic deformation), however this shape change is believed to be self reversing since the Von Miss stress were well below the yield strength.

IV. RECOMMENDATION

It is recommended that further simulations are carried out using this initial design as a bench mark for comparison to further ensure that only the best and economical designs are put into service as well as ensure that the possibilities of failure are negated to the minimum, a factor of safety of 2 and above using different high strength materials should also be explored. An iterative method may be used to optimize the chamber design by establishing the minimum wall thickness required to withstand the designed thrust and maximum operating pressure and by extension reducing the chamber weight.

V. CONCLUSION

In conclusion, although the combustion chamber met the minimum design criteria there is the need for improvement; as the simulation had revealed that the Von Mises stress were within limits; however it is imperative that the possibility of further weight shedding are taken into account and that all possibility of failure are identified and appropriately catered for.

BIBLIOGRAPHY

[1]. Bianchi, D. Summary of Rocket Nozzle Relationships. Rome,Italy : Sapienza Universita Di Roma.

[2]. Bird, Byron R., Stewart, Warren E. and Lightfoot, Edwin N. 2012. Transport Phenomena. Singapore : John Wiley & Sons, 2012. ISBN: 978-81-265-0808-2.

[3]. Clayton, Chun K. S. 2006. Thunder Over The Horizon; From V2 Rockets to Ballistic Missiles. United States of America : Praeger Security International, 2006. ISBN: 0-275-98577-6.

[4]. Deborah, Shearer A., Gregory, Vogt L. and Ed, D. Rockets; A Teacher's Guide with Activities in Science, Mathematics, and Technology. Houston, Texas : National Aeronautics and Space Administration. EG-1999-06-108-HQ.

[5]. Dieter, Huzel K and David, Huang H. 1992. Modern Engineering For Design Of Liquid-Propellant Rocket Engines. Washington, DC : American Institute of Aeronautics and Astronautics, Inc, 1992. ISBN 1-56347-013-6.

[6]. George, Sutton P. and Oscar, Biblarz. 2001. Rocket Propulsion Elements. New York / Chicester / Weinheim / Brisbane / Singapore / Toronto : John Wiley & Sons, INC, 2001. ISBN 0-471-32642-9.

[7]. Hougen, Olaf A, Watson, Kenneth M and Ragatz, Roland A. 1976. Chemical Process Principles, Part -I, Materials and Energy

Balances. Calcutta : Asia Publishing House, 1976.

[8]. John, Coleman J. 1960. Optimum Stage-Weight Distribution of Multistage Rockets. Los Angeles 45, Carliformia : Space Technology Laboratories, Inc, 1960. STL/TN-60-0000-09036.

[9]. Rakshit, P C. 2001. Physical Chemistry. India : Sarat Book Distributers, 2001.

[10]. Rocketlab. 2003. How to Design, Build and Test Small Liquid-Fuel Rocket Engines. China Lake, California : Leroy J. Krzycki, 2003. ISBN 9600-1980-4.

[11]. Schoenheit, W. 1951. Propellant Feed Valves for Starting and Control of Rocket Motors. Farnborough, Hants : Royal Aircraft Establishment, 1951. U.D.C No: 621.455-441:621-33.

[12]. Smith, J. M., Van Ness, H. C. and Abbott, M. M. 2001. Introduction to Chemical Engineering Thermodynamics. New York : McGraw-Hill, 2001. ISBN 0-07-240296-2.

[13]. Turner, Martin J.L. 2005. Rocket and Spacecraft Propulsion Principles, Practice and New Developments. Chichester, UK : Springer-Praxis, 2005. ISBN 3-540-22190-5.

[14]. Vic, Bekarian, et al. 2013. The Design and Building of a Level 3 Rocket. California, San Diego : University of Carlifornia, 2013.

[15]. Vigor, Yang, et al. 2004. Liquid Rocket Thrust Chambers: Aspects of Modelling, Analysis, and Design. Reston, Virginia : American Institute of Aeronautics and Astronautics, Inc, 2004. ISBN 1-56347-223-6.