Abstract—Solid rocket motors (or SRMs) are simple devices comprising of bulk head, combustion chamber and nozzle in one single unit [1]. The solid rocket motor combustion chamber is made of an extruded mild steel metal with an adjoining nozzle screwed to the end of the pipe. It was designed and fabricated at the Center for Space Transport and Propulsion an activity center of the National Space Research and Development Agency (NASRDA). The combustion chamber of a solid rocket motor is the part that houses the propellant grains. Combustion takes place inside the chamber after ignition of the propellant grains and hot combustion gases are produced and forced out through the nozzle thereby generating high thrust for rocket lift off [4]. The simplicity of SRMs makes them a very suitable and an attractive choice for many rocket propulsion applications. Because there are few structural component parts, the SRM is efficient in that the vast majority of its weight is actually usable propellant [7].

Index Terms—Nozzle, Ignition, Combustion, Propellant, Thrust, Extrusion.

I. INTRODUCTION

In the design of combustion chamber for solid rocket motors, it is very critical to have adequate knowledge of the propellant characteristics such as the density, burn rate, maximum expected temperature, maximum expected pressure, burn time etc [3]. These characteristics are very paramount as they form the critical design input for material selection and sizing of the various component parts of the solid rocket motor. The propellant contains both fuel and oxidizer; therefore these devices can operate in the vacuum of space [4]. Thrust is developed as the high thermal energy of the combustion gases is converted to kinetic energy in the exhaust. Solid Rocket Motors can be ignited in a moment’s notice and don’t require external tanking of liquids prior to operation. Although, their efficiency (specific impulse) is generally lower than liquid rocket engine systems, and they cannot be readily throttled. Once they are ignited, the motor will burn to extinction unless special provisions are included to terminate thrust during the middle of a firing [4].

II. FLIGHT TRAJECTORY ANALYSIS

The design process started with a flight trajectory analysis for a 1Km altitude rocket using Potassium nitrate as oxidizer and Sucrose as the main fuel. The table below shows the flight trajectory simulation results for a 5.06Kg KNSU Propellant.

<table>
<thead>
<tr>
<th>PARAMETERS</th>
<th>SYMBOLS</th>
<th>VALUES</th>
<th>UNITS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Gravity Acceleration</td>
<td>g</td>
<td>9.69</td>
<td>m/s²</td>
</tr>
<tr>
<td>Air Density</td>
<td>ρ</td>
<td>1.223</td>
<td>kg/m³</td>
</tr>
<tr>
<td>Drag Coefficient</td>
<td>Cd</td>
<td>0.4</td>
<td></td>
</tr>
<tr>
<td>Burn Time</td>
<td>tb</td>
<td>7.0</td>
<td>s</td>
</tr>
<tr>
<td>Specific Impulse</td>
<td>I₀</td>
<td>150</td>
<td>s</td>
</tr>
<tr>
<td>Rocket Total Mass</td>
<td>mo</td>
<td>41.32998</td>
<td>kg</td>
</tr>
<tr>
<td>Propellant Mass Ratio</td>
<td>fp</td>
<td>0.122429</td>
<td></td>
</tr>
<tr>
<td>Mass Of Propellant</td>
<td>mp</td>
<td>5.06</td>
<td>kg</td>
</tr>
<tr>
<td>Propellant Mass Flow Rate</td>
<td>mp-dot</td>
<td>1.363881</td>
<td>kg/s</td>
</tr>
<tr>
<td>Mass Of Final Rocket</td>
<td>mf</td>
<td>36.26998</td>
<td>kg</td>
</tr>
<tr>
<td>Rocket Diameter</td>
<td>d</td>
<td>0.2</td>
<td>m</td>
</tr>
<tr>
<td>X-Section Area</td>
<td>A</td>
<td>0.031416</td>
<td>m²</td>
</tr>
<tr>
<td>Drag Factor</td>
<td>k</td>
<td>0.007684</td>
<td>kg/m</td>
</tr>
<tr>
<td>Fineness Ratio Of Rocket</td>
<td>λ</td>
<td>12</td>
<td></td>
</tr>
<tr>
<td>Length Of Rocket</td>
<td>L</td>
<td>2.4</td>
<td>m</td>
</tr>
<tr>
<td>Total Impulse</td>
<td>I</td>
<td>7354.71</td>
<td>N-s</td>
</tr>
<tr>
<td>Average Thrust</td>
<td>F</td>
<td>1982.402</td>
<td>N</td>
</tr>
<tr>
<td>Thrust-To-Weight Ratio</td>
<td>vo</td>
<td>4.949971</td>
<td></td>
</tr>
<tr>
<td>Number Of Iterations</td>
<td>N</td>
<td>300</td>
<td></td>
</tr>
<tr>
<td>Sampling Time</td>
<td>Δt</td>
<td>0.012367</td>
<td>s</td>
</tr>
<tr>
<td>Maximum Time</td>
<td>tmax</td>
<td>17.09553</td>
<td>s</td>
</tr>
<tr>
<td>Maximum Altitude</td>
<td>hmax</td>
<td>1200</td>
<td>m</td>
</tr>
</tbody>
</table>

Table 1.1 Flight trajectory parameters.

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Using ProPep Simulation software for the characterization of the sugar based propellant, the results generated can be seen in the table below.

### Table 1.2 KNSU PROPELLANT GRAIN PARAMETERS

<table>
<thead>
<tr>
<th>S/N</th>
<th>PARAMETERS</th>
<th>VALUES</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Propellant outer Diameter, D (m)</td>
<td>0.105</td>
</tr>
<tr>
<td>2</td>
<td>Propellant Core Diameter, D₀ (m)</td>
<td>0.038</td>
</tr>
<tr>
<td>3</td>
<td>Bate Length (m)</td>
<td>0.118</td>
</tr>
<tr>
<td>4</td>
<td>Mass of propellant (kg)</td>
<td>5.06</td>
</tr>
<tr>
<td>5</td>
<td>Number of bates, N</td>
<td>3</td>
</tr>
<tr>
<td>6</td>
<td>Throat diameter (m)</td>
<td>0.036</td>
</tr>
<tr>
<td>7</td>
<td>Total length of grain (m)</td>
<td>0.354659</td>
</tr>
<tr>
<td>8</td>
<td>Burn time, tₙ (s)</td>
<td>7.0</td>
</tr>
<tr>
<td>9</td>
<td>Propellant density (kg/m³)</td>
<td>1896</td>
</tr>
<tr>
<td>10</td>
<td>Max. Kn</td>
<td>88.199</td>
</tr>
<tr>
<td>11</td>
<td>Average thrust, F (N)</td>
<td>1925.907</td>
</tr>
<tr>
<td>12</td>
<td>Max pressure, Pₑ (MPa)</td>
<td>1.08</td>
</tr>
<tr>
<td>13</td>
<td>Max Temperature, Pₑ(ºC)</td>
<td>1376</td>
</tr>
</tbody>
</table>

### III. NOZZLE DESIGN PARAMETERS

NozzleDesign Analysis was carried out using critical input from the ProPep simulation software.

**Input Data from ProPep:**
- Max. Temperature inside chamber = 1650K
- Max. Pressure inside chamber = 1.08MPa= P₁
- Mass Flow Rate = 0.723Kg/s
- Specific Impulse = 150s
- Air pressure at 10KM= 0.02644MPa = P₂
- Burn time = 7s
- Pressure Ratio = Pₚ/P₁= 0.0245MPa

The critical throat pressure for which the isentropic mass flow is maximum Pᵣ is calculated next [1].

\[
Pᵣ = P₁ \left[ \frac{2}{K+1} \right] ^{\frac{K}{K-1}} \quad \text{(1)}
\]

Where k =1.3 is the specific heat ratio.

\[
Pᵣ = 0.6156 \text{MPa}
\]

The throat velocity vₜ is given by:

\[
vₜ = \sqrt{\left( \frac{2K}{K+1} \right) RT₁} \quad \text{(2)}
\]

\[
vₜ = 814.19 \text{ m/s}
\]

Where, R=355.4 JKg⁻¹K⁻¹ is gas constant,

\[T₁=1650K\text{ is the chamber temperature.}\]

Ideal exit velocity is calculated by:

\[
vₑ = \sqrt{\left( \frac{2K}{K+1} \right) RT₁ \left[ 1 - \left( \frac{P₂}{P₁} \right)^{\frac{k-1}{k}} \right]} \quad \text{(3)}
\]

\[
vₑ = 1710 \text{ m/s}
\]

Mach number, \(M = \frac{vₑ}{a} \quad \text{(4)}\)

where, a is the speed of sound = 340 m/s

\[M = 5.03\]

Specific volume (V₁) at nozzle entrance is:

\[V₁ = \frac{RT₁}{P₁} \quad \text{(5)}\]

\[V₁ = 0.543 \text{ m}^3/\text{kg}\]

Specific volume (Vₑ) at the throat is:

\[Vₑ = V₁ \left( \frac{K+1}{2} \right)^{\frac{1}{k-1}} \quad \text{(6)}\]

\[Vₑ = 0.865 \text{ m}^3/\text{kg}\]

Specific volume (Vₑ) at the exit is:

\[Vₑ = V₁ \left( \frac{P₂}{P₁} \right)^{\frac{1}{k}} \quad \text{(7)}\]

\[Vₑ = 9.42 \text{ m}^3/\text{kg}\]

Area (Aₑ) and Diameter (Dₑ) at the Throat Section:

\[Aₑ = \frac{mVₑ}{g} \quad \text{(8)}\]

where \(m\) is the propellant mass flow rate = 0.723Kg/s

\[Aₑ = 7.68 \times 10^{-4} \text{ m}^2\]

\[Dₑ = \sqrt{\frac{4Aₑ}{\pi}} \quad \text{(9)}\]

\[Dₑ = 0.03m\]
Area (Ae) and Diameter (De) at the Exit Section:
\[ A_e = \frac{\dot{m} \times V_2}{v_2} \]  
\[ A_e = 7.68 \times 10^{-4} \text{ m}^2 \]

\[ D_e = \sqrt{\frac{4A_e}{\pi}} \]  
\[ D_e = 0.07 \text{ m} \]

Length of Convergent (Lconv) Nozzle Section
\[ L_{\text{conv}} = \frac{D_{ch} - D_t}{2 \tan \alpha} \]  
Where: \( \alpha = 40^\circ \), \( D_{ch} \) is chamber diameter = 0.24 m
\( L_{\text{conv}} = 0.058 \text{ m} \)

Length of Divergent (Ldiv) Nozzle Section
\[ L_{\text{div}} = \frac{D_t - D_e}{2 \tan \beta} \]  
Where: \( \beta = 15^\circ \)
\[ L_{\text{div}} = 0.075 \text{ m} \]

The minimum wall thickness is calculated by [2], [6].
\[ t_{ch} = \frac{f \times P_{\text{max}} \times D_{ch}}{2 \times \delta_b} \]  
Where: 
\( P_{\text{max}} = 2.0 \text{ MPa} \) is maximum expected operating pressure.
\( D_{ch} = 122 \text{ mm} \) is maximum expected internal chamber diameter.
\( \delta_b = 400 \text{ MPa} \) is Ultimate tensile Strength of mild steel.
\( f = 8 \) is factor of safety.

The calculated chamber thickness is:
\[ t_{ch} = \frac{8 \times 2.0 \times 122}{2 \times 400} \]
\[ t_{ch} = 2.44 \text{ mm} \]
\( t_{ch} \) of 3 mm was selected.
Design Analysis of Mild Steel Combustion Chamber for a Sugar Base Propellant of a Solid Rocket Motor

Total Length of Combustion Chamber

\[ L_{th} = L_{conv} + L_{grain} + L_{free\_volume} \]

\[ L_{th} = 58 + 354 + 231 = 643 \text{mm} \]

VI. MATERIAL SELECTION

Material selection is also a key factor in the construction of rocket motors [7]. The following factors were considered before the final selection of the material used to fabricate the designed rocket motor.

- Weight of material
- Strength
- Availability of material
- Affordability/Cost effectiveness
- Ease of Fabrication
- Versatility of attachment options
- Fracture Toughness
- Thermal Conductivity
- Resistance to wear
- Melting point of material

Having exhaustively considered the above factors, mild steel was selected for the construction of the rocket motor.

REFERENCES

[2] Dennis Moss, Pressure vessel design manual (3rd edn), Gulf professional Publishing Burlington USA.