DESIGN, FABRICATE AND TESTING SMALL ROCKET MOTOR

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Report submitted in partial fulfillment of the requirements
for the award of the degree of
Bachelor of Mechanical Engineering

Faculty of Mechanical Engineering
UNIVERSITI MALAYSIA PAHANG

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I certify that the project entitled "*Design, Fabricate and Testing small rocket motor*" is written by *Mohamad Izwan Ghazali*. I have examined the final copy of this project and in my opinion; it is fully adequate in terms of scope and quality for the award of the degree of Bachelor of Engineering. I here with recommend that it be accepted in partial fulfillment of the requirements for the degree of Bachelor of Mechanical Engineering.

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DEDICATION

I specially dedicate to my beloved parents

and those who have guided

and motivated me for this project

ACKNOWLEDGEMENT

All praise to Allah for giving me a chance to complete my project without any difficulty. All the time taken in doing the experiment has taught me many things about the significance of case study to the whole world. I am grateful and would like to express my sincere gratitude to my supervisor Mr. Amir Aziz for his germinal ideas, invaluable guidance, continuous encouragement and constant support in making this research possible. They always impressed me with his professional conduct. They have been the most reliable person starting from the first day I applied to graduate program to these concluding moments. I am truly grateful for his progressive vision about my training in science, his tolerance of my naive mistakes, and his commitment to my future career. I also sincerely thanks for the time spent proofreading and correcting my many mistakes

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ABSTRACT

There was a lot of study on Solid Rocket Motor (SRM) based solid propellant. This project focus on and discusses the study of optimum design based SRM characteristics including the methods of the optimum design selection and fabrication, analysis using COSMOS and static thrust testing. Before that, the researcher has focus on the fundamental of solid rocket motor for designing and fabricating. There are two main factors need to be considered in the design selection and fabrication which are performance or processability and mechanical strength. The theoretical performance of the propellant was obtained by using CHEM program. Together with literature study and theoretical performance, three models or design of nozzle with different size throat were finalized with consideration of the mechanical and processability factors. The propellant was a mixture of Potassium nitrate and sucrose. The rocket motors were manufactured or fabricated using lathe and milling machine. Then three solid rocket motors were tested to get the thrust and performance. The results show that the increasing of thrust and combustion pressure lead to the decreasing the throat size and increasing the throat length. The highest thrust was 1260N and burning time about 4 Sec. Meanwhile, for the performance characteristics, the specific impulse, I_{sp} that obtained from static thrust testing for solid propellant was 4% lower than theoretically.

ABSTRAK

Terdapat banyak kajian mengenai Roket Enjin berdasarkan bahan bakar pepejal. Keutamaan projek ini untuk membincangkan serta membuat kajian mengenai cirri-ciri reka bentuk terbaik ataupun optimum yang berasaskan SRM termasuk pemilihan reka bentuk terbaik dan analisis menggunakan COSMOS sebelum proses pembuatan roket enjin dilakukan. Sebelum itu penyelidik telah member tumpuan kepada asas-asas pembinaan enjin roket. Terdapat dua factor yang harus diambil kira semasa proses pemilihan bahan untuk membina enjin roket ini iaitu kekuatan bahan dan daya ketahanan bahan dalam tekanan dan suhu yang tinggi. Daya tujah pada awalnya diambil kira setelah menggunakan CHEM. Secara teorinya kita telah mendapat hasil daya tujah roket tersebut dan dapat membuat kesimpulan awal untuk pemilihan reka bentuk yang terbaik. Perubahan luas tekak roket enjin dapat mengeluarkan hasil yang berbeza-beza dan adakalanya gagal disebabkan saiz tekak tidak sesuai dengan tekanan yang dikenakan. Tiga enjin telah diuji untuk dilihat serta dicatat hasilnya untuk dibuat kesimpulan dan pemilihan yang paling terbaik. Bahan bakar yang digunakan dalam projek ini adaqlah Pottasium Nitrat dan sukrosa. Hasil kajian menunjukkan bahawa peningkatan tekanan teras dan pembakaran membawa kepada panjang tekak. Daya tujah yang tertinggi yang Berjaya dihasilkan adalah 1260 newton dan terbakar dalam kira-kira 4 saat. Bahan bakar yang digunakan dalam projek inin adalah bahan bakar untuk pembuat roket amatur.

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LIST OF SYMBOLS

α	alpha ferrite
°C	Degree Celsius
%	Percentage
λ	correction factor
L_{cone}	length of cone
r_2	outer radius
$\dot{m}$	mass flow rate
P_c	Pressure chamber
A_t	Throat area
T_c	Chamber temperature
R	Ryberg constant
A_e	Exit area
P_e	Exit pressure
k	Gamma
V_t	volume of throat
mm	millimeter
m	Meter
$\in$	throat ratio
m/s	Meter per second
$P_\circ$	Initial pressure
R_p	Resistant potential
s	second
M	Mach number
v_t	volume of throat
T_t	temperature of throat
γ	Gamma ferrite

ρ	density
F	thrust
C_F	thrust coefficient
I_s	specific impulse
c^*	cee-star
g_o	gravity
a	speed of sound
γ	specific heat ratio

LIST OF ABBREVIATIONS

A	Area
AISI	American Iron and Steel Institute
C	Carbon
d	Density
KNO_3	Potassium nitrate
Fe^{2+}	Iron ion
Fe_3C	Cementite
H_2O	Water
L	Liquid
M	Metal
Mn	Manganese
m/s	metre per second
NaCl	Sodium Chloride
O_2	Oxygen gas
OH	Hydroxide
CHEM	Chemical
S	Sulphur
FKM	Faculty of Mechanical Engineering
FYP	Final Year Project
UMP	Universiti Malaysia Pahang

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CHAPTER 1

INTRODUCTION

1.1 Introduction

Rocket motor is one of the significant components in constructing amateur solid rocket and it comprises a lot of application theory of the nozzle. This component of nozzle and fluid flow related to the pressure, temperature and velocity. The prior knowledge about rocket motors' theory and nozzle must be studied in order to get the blue prints for the design. Another vital thing that needs to be considered while creating this rocket motor is, it must be design for optimum dimension and need to be analysed by using COSMOS or Fenite Element Analysis. An Optimum dimension can be defined as the best diameter of the nozzle, because in theoretical knowledge, there is a rule about exit diameter and throat diameter for nozzle. Besides that, for preventing any failure during test launcher, suitable angle also must be considered in this project because incorrect dimension for rocket motors will lead to failure during launcher. So, crucial things in getting the best result for analysis will depend on the correct dimension and angle design. Next, fabricate the rocket motors and test rig where the rocket motors was fabricated by using lathe machine and drilling machine. Finally, report writing with the real result of testing.

1.2 Problem Statement

This project is about our idea of designing the optimum rockets motor for a small launcher and conducting an analysis in the rocket motors. The rocket motor in this project functioned as a device producing thrust in rocket's launching. In the rocket's industry, the rocket engine usually built by using the theory of nozzle and fluid low where its design and structure become the key point in creating a good rocket engine. The correct design and size of rocket engine must be created in order to support rocket during launching and any failure will bring danger to the people in the rocket if the rocket explode.

1.3 Project Objectives

Developing an optimum performance for rocket motors by two primary objectives first to theoretically analyze the operation of small solid propellant rocket motor and to conduct testing with which to compare the theoretical result.

1.4 Scope

i. Design of a rocket motors (including the COSMOS's analysis)
ii. Fabricate the rocket motors
iii. Conduct experimental and test rig
iv. Analysis and report writing

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CHAPTER 2

LITERATURE REVIEW

2.1 Introduction

Solid motor rocket consists of nozzle, casing, propellant and igniter same as in figure 2.1. But it also comprises time delay and the charger which process the explosion of parachute. Generally, the rocket could be propelled by using liquid or solid propellant. In this study, only solid propellant for rocket motor will be discussed. The main elements for solid propellant are oxidizer, fuel and binder.

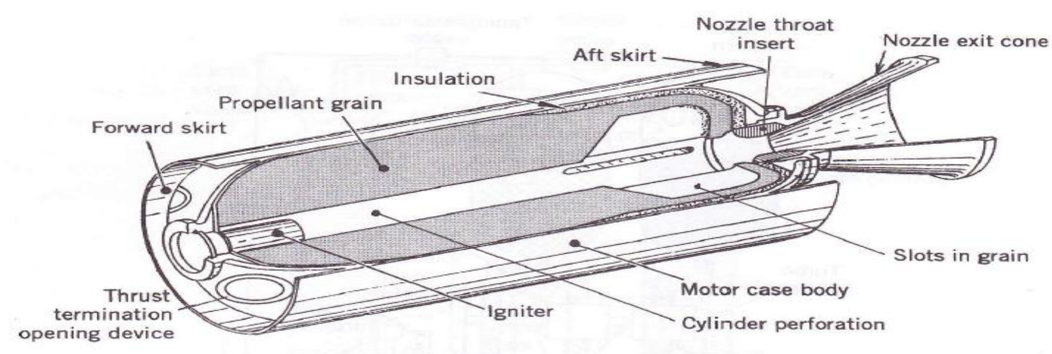


Figure 2.1: Solid Rocket Propellant

- Source : Refferences Book (P.R EVANS "Composite Motor Case Design")

Solid rocket motor consists of a solid propellant grain embedded into a stronger metallic or composite case with an insulator material and a liner between the case and the grain. The motors which are mainly utilized in defense and space technologies are generally for a long time and transported from one place to another before their ignition process. Mechanical properties of solid propellant are very sensitive to temperature changes. (H.C Yildrim, 2010)

2.2 Nozzle theory

A nozzle is a device design to control the pressure or characteristic of a fluid flow especially to increase velocity as it exists or enters an enclosed chamber or pipe by an orifice. A nozzle is often a pipe or tube of varying cross sectional area, and it can be used to direct or modify the flow of a fluid liquid or gas. Nozzles are frequently used to control the rate of flow, speed, direction, mass, shape, and the pressure of the stream that emerges from them. Increase the kinetic energy of the following medium at the expense of its pressure and internal energy. Nozzle typically involves no work and any change potential energy is negligible. But nozzle it experiences large changes in its velocity. The principal conservation of mass in a steady flow with a single inlet and outlet is expressed by equating the mass flow rate $\dot{m}$.

2.3 Type of nozzle & Correction Factor

The nozzle is a device that increases the velocity of a fluid at the expense of pressure. The cross sectional area of the nozzle decreases in the flow direction for subsonic flow and an increase in supersonic flow. The rate of heat transfer of fluid that flowing through a nozzle by the surroundings is very small since the fluid has high velocities, and thus it does not spend enough time in the device for any significant heat transfer to take place. In rocket applications, nozzle can be divided into two types which are conical and bell nozzle. Bell’s nozzle more efficiency than conical nozzle but for our design or amateur design, we consider the conical nozzle because it easier to fabricate compared to the bell nozzle.

2.4 Cone and Bell Shape Nozzle

The conical nozzle is the oldest and perhaps the simplest configuration. It is relatively easy to fabricate and still be used today in the many small nozzles. A theoretical correction factor λ can be applied to the nozzle exit momentum of an ideal rocket with a conical nozzle exhaust. This factor is the ratio between the momentum of the gases in a nozzle with a finite nozzle angle 2α and the momentum of an ideal nozzle with all gases flowing in an axial direction :

$$\lambda = \frac{1}{2}\left(1 + \cos\alpha\right) \tag{2.1}$$

Where :

λ = Correction factor
α = cone divergence half angle

For a rocket nozzle with a divergence cone angle of 30° (half angle = 15°) the exit momentum and therefore, the exhaust velocity will be 98.3% of the velocity calculated. A small nozzle divergence angle causes most of the momentum to be axial and thus gives a high specific impulse, but long nozzle has a penalty in the rocket propulsion system mass. A large divergence angle gives short and light weight design but performance is low. Below figure shown the optimum conical nozzle shape and length (between 12° and 18°) :

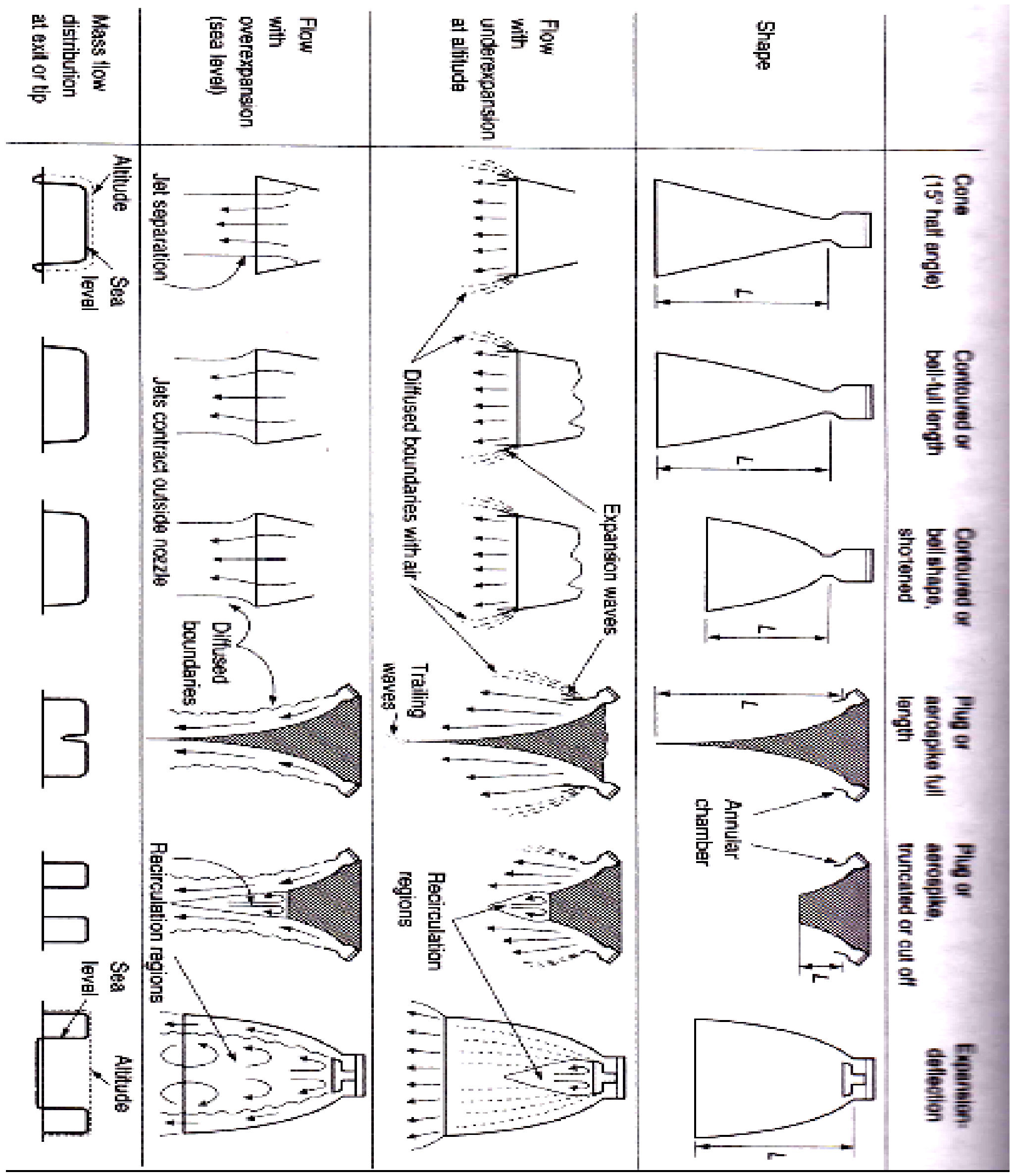


Figure 2.2 : Simplified diagram of several different nozzle configurations and their flow effect

Table 2.1 : Correction factor

Nozzle Cone Divergence Half Angle (degree)	Correction factor
0	1.0
2	0.9997
4	0.9988
6	0.9972
8	0.9951
10	0.9924
12	0.9890
14	0.9851
15	0.9830
16	0.9806
18	0.9755
20	0.9698
22	0.9636
24	0.9567

A change flow direction of a supersonic gas in an expanding wall geometry can only be achieved through expansion waves. Related formula for ratio length expansion nozzles with radius:

$$L_{cone} = \frac{r_2 - r_1}{\tan\alpha} \tag{2.2}$$

Where :

L_{cone} = Divergent length
r_1 = Throat radius
r_2 = Exit radius
$\tan\alpha$ = Cone divergence half angle

The theory has previously said there are differences in the fluid flowing through the nozzle. The properties of the fluid can be expressed in the figure below.

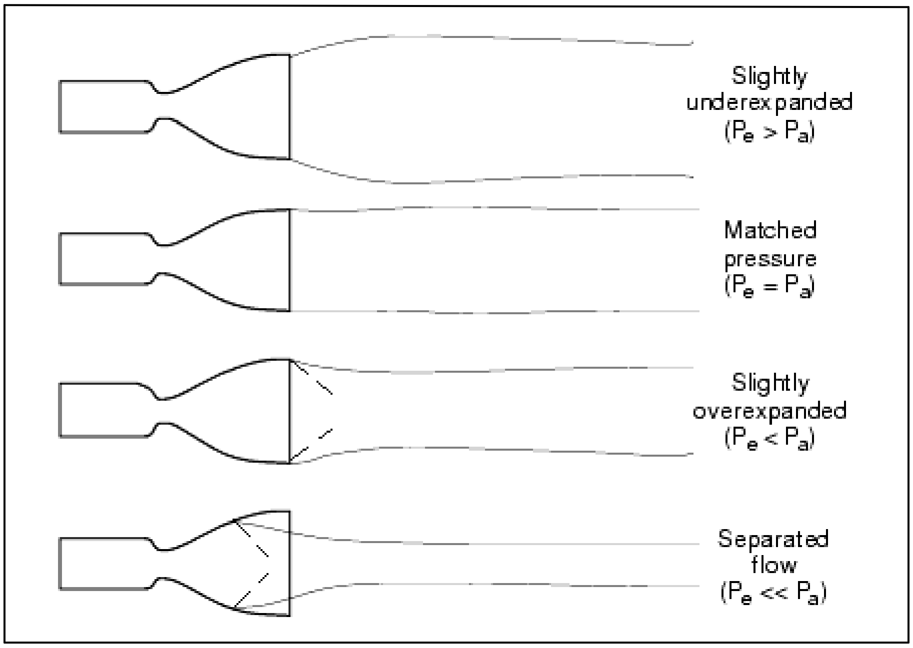


Figure 2.3 : Type Of Flow

- Source : Refferences Book(Rocket Propulsion Elements (Eighth Edition) by George P.Sutton & Oscar Biblarz)

2.5 Supersonic nozzle

For a rocket motor, the nozzle usually has a circular cross section. The combustion chamber radius R_c is obtained from the study of the chamber while the value for the throat area A_t and throat radius R_t is from equation:

$$\dot{m} = \frac{r P_c A_t}{\left(R T_c\right)^{\frac{1}{2}}} \qquad (2.3)$$

Where :

$\dot{m}$ = Mass flow rate
T_c = Chamber temperature
P_c = Chamber pressure
A_t = Throat area

Finally the radius and area of the nozzle exit is obtained from the equation:

$$\frac{A_e}{A_t} = \frac{r}{\left(P_e/P_c\right)^{1/k}\left[2k/(k-1)\cdot\left\{1-(P_e/P_c)^{(k-1)/k}\right\}\right]} \tag{2.4}$$

Where :

k = Specific heat ratio
T_c = Chamber temperature
P_c = Chamber pressure
A_t = Throat area

In actual the nozzle performance is not very sensitive to the geometric design which is selected for easy manufacturing. For a convergence conical half angle is around 30 degrees. The radius of curvature near the throat must be sufficient enough in order to ensure the progressive velocity increase. Finally, area increase in the divergence must be sufficiently progressive avoid boundary layer separation.

Supersonic nozzle the ratio between the throat and any downstream area at which a pressure prevails can be expressed as a function of the pressure ratio and the ratio of specific heats by using the equation below :

$$V_t = \sqrt{\frac{2k}{k+1}RT_1} \tag{2.5}$$

Where :

k = Specific heat ratio
T_c = Chamber temperature
V_t = Throat velocity

As we know, the function of the nozzle is converting the thermal energy in the propellant into kinetic energy as efficiently as possible, in order to obtain high exhaust velocity along the desired direction. The required nozzle area decreases to a minimum and then increases again. It consists of a convergent section followed by divergent section.

Throat pressure for isentropic flow called critical pressure ratio range between 0.53 and 0.57 of the inlet pressure. Flow for the inlet condition less than the maximum if the pressure ratio longer than that given. The equation of the critical pressure and throat pressure ratio at below:

$$\frac{P_t}{P_c} = \left[\frac{2}{(k+1)}\right]^{k/(k-1)} \tag{2.6}$$

Where :

k = Specific heat ratio
P_c = Chamber pressure
P_t = Throat pressure

$$P_{\circ} = P\left[1 + 1/2(k-1)M^2\right]^{k/(k-1)} \tag{2.7}$$

Where :

k = Specific heat ratio
$P_{\circ}$ = Critical pressure
P = Atmosphere pressure
M = Mach number

Besides that, at point critical pressure the values of the specific volume, temperature and velocity can be obtained :

$$v_t = v_c \left[\frac{(k+1)}{2} \right]^{1/(k-1)} \tag{2.8}$$

$$T_t = \frac{2T_c}{(k+1)} \tag{2.9}$$

$$V_t = \sqrt{\frac{2k}{k+1} RT_c} \tag{2.10}$$

Where :

k = Specific heat ratio
T_c = Chamber temperature
V_t = Throat velocity
v_t = Volume throat
v_c = Chamber volume

2.6 Nozzle flow and throat condition

Nozzle of this type consists of the convergent section followed by divergent section. From the continuity equation, the area is inversely proportional to the ratio velocity per volume. This quantity has been plotted in Figure 2.7. There is a maximum in the curve because at first the velocity increases at a finer than the precise volume. However, in divergent section the exact volume increased at a finer rate. The minimum nozzle area is called the throat area. The ratio of the nozzle exit area to the nozzle throat area is called the nozzle expansion area ratio. It is an important nozzle parameter:

$$\in = \frac{A_e}{A_t} \tag{2.11}$$

Where :

$\in$ = Expansion area ratio

A_t = Area of throat
A_e = Exit area

The pressure ratio means the ratio of exit area to throat area of fluid only. Besides that, the mass flow between the throat and exit area is written as follows:

$$(\rho VA)_e = (\rho VA)_t \tag{2.12}$$

By substituting,

$$A_e / A_t = \rho_t V_t A_t = \frac{r(P_c / P_e)^{1/k} \{RT_c\}^{1/2}}{V_e} \tag{2.13}$$

After that replacing the relation for as in equation,

So, the equation becomes:

$$\frac{A_e}{A_t} = \frac{r}{\left({P_e}/{P_c} \right)^{1/k} \left[{2k}/{(k-1)} \cdot \left\{ 1 - (P_e / P_c)^{(k-1)/k} \right\} \right]^{1/2}} \tag{2.14}$$

Conversely the value of pressure ratio between exit pressure and chamber pressure can be determined when there is no flow within the divergent section of the nozzle and area ratio is fixed.

2.7 Thrust and thrust coefficient

The thrust is the force produced by a rocket propulsion system. In a simplified way, it is the reaction experienced by its structure due to the ejection of matter at high velocity. All ship propeller and oars generate their forward push at the expense of the momentum of the water or air masses, which are accelerated toward the rear. In rocket propulsion, relatively small masses are involved that are ejected at high velocity. This force represents the total propulsion force when the nozzle exit pressure equals the ambient pressure. The figure will be shown the external pressure acted on the outer surface and changing pressure on the inside of a typical thermal rocket engine.

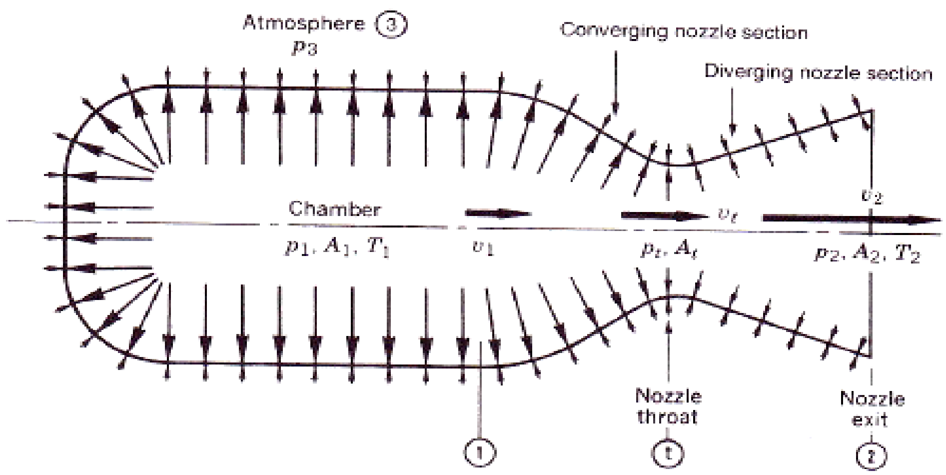


Figure 2.4: Pressure act on the nozzle

Pressure balance on chamber and nozzle interior wall is not uniform. The internal gas pressure indicates by length of arrows is highest in the chamber and decrease in the nozzle until it reaches the nozzle exit pressure. The superficial or atmospherical pressure is uniform. It's because of fixed nozzle geometry and changes in ambient pressure due to variations in altitude, there can be an imbalance of the apparent environment or atmospherical pressure and the local pressure of the hot gas jet at the exit plane of the nozzle. Equation at below shown the thrust:

$$F = \dot{m}V_2 + (p_e - p_t)A_2 \tag{2.15}$$

Where :

F = Thrust (N)
$\dot{m}$ = mass flow rate (kg/sec)
p_e = Exit pressure (Pa)
p_t = Throat pressure (Pa)

Value of thrust can maximize when atmospheric pressure equal to zero but for optimum condition's exit pressure equal to atmospheric pressure for giving chamber pressure. As an above equation shows that the ideal thrust equation. It is shown that the thrust proportional to the throat area. The thrust rocket depends on flight velocity. Changes in ambient pressure affect the pressure thrust. Atmospheric pressure decreases with increasing altitude. The thrust coefficient and thrust have peaked when exit pressured equal to ambient pressure. This peak value also knows optimum thrust coefficient and is important to design nozzle. Thrust coefficient can simplify to:

$$C_F = \frac{F}{p_1 A_t} \tag{2.16}$$

Where :

F = Thrust (N)
p_1 = Pressure Chamber (Pa)
A_t = Throat Area (cm^2)

The thrust coefficient has a value ranging from about 0.8 to 1.9. Its convenient parameters for seeing the effects of chamber pressure or altitude for flight condition. Process the ideal thrust coefficient and the theoretical specific impulse to the actual thrust coefficient and the delivered specific impulse. The summary equations are presented in the below:

$$I_s = (I_s)_{opt} + \frac{c^* \in}{g_o}\left(\frac{p_2}{p_1} - \frac{p_3}{p_1}\right) \tag{2.17}$$

Where :

F = Thrust (N)
p_1 = Pressure Chamber (Pa)
p_3 = Atmosphere Pressure (Pa)

2.8 Exhaust Velocity

When the atmospheric pressure equal to exit pressure the effective exhaust velocity c is equal to the average actual exhaust velocity of the propellant gases. The characteristic velocity has been used frequently in the rocket propulsion literature. Its symbol “Cee-star” is defined as :

$$c = V_2 + (p_2 - p_3)A_2 / \dot{m} \tag{2.18}$$

Where :

$\dot{m}$ = mass flow rate (kg/sec)
p_1 = Pressure Chamber (Pa)
p_3 = Atmosphere Pressure (Pa)
A_2 = Exit area
V_2 = Exit velocity

The characteristic velocity c* is used in comparing the relative performance of different chemical rocket propulsion system designs and propellants, it is easily determined from measured data. It relates to the efficiency of combustion and is essentially independent of nozzle characteristic.

2.9 Specific Impulse

The specific impulse is the total impulse per unit weight of propellant. It is an important figure of the performance rocket system. A higher number means better performance. In this project, discuss of the specific impulse for constant thrust propellant mass flow. All of specific impulse equation state at below:

$$I_s = I_t / \dot{m} g_0 \tag{2.19}$$

Where :

$\dot{m}$ = mass flow rate (kg/sec)
p_1 = Pressure Chamber (Pa)
p_3 = Atmosphere Pressure (Pa)
A_2 = Exit area
V_2 = Exit velocity

2.10 Ideal Rocket

For chemical rocket propulsion, the measured actual performance is usually between 1% and 6% below the calculated idea value. An ideal rocket unit must occupy the following assumptions:

i. The working substance is homogeneous.
ii. All working fluids are gaseous. Any condensed phases add a negligible amount of total mass.
iii. Perfect gas law ($pv = nRT$)
iv. No heat transfer crosses the nozzle wall. Flow is adiabatic.
v. All boundary layer effect is negligible. No friction.
vi. No shock waves in the nozzle.
vii. Propellant flow is steady and constant. The expansion of the working fluid is uniform and steady without vibration.

viii. Gas velocity, pressure, temperature, and density are all uniform across any section normal to the nozzle axis.

ix. All exhaust gases leaving the rocket have an axially directed velocity.

For a bipropellant rocket, the idealized theory postulates an injection system in which the fuel and an oxidizer perfect mixture so that homogeneous working substance results. For a solid propellant rocket, the propellant must be uniform and the burning rate must be steady. In well-designed supersonic nozzles, the convergent smoothly and without normal shocks or discontinuities thus the flow expansion losses are generally small. For item number 9 at above some use a conical exit nozzle with an angle 15° half angle their base configuration in their ideal nozzle.

2.11 Mach Number

The Mach Number M is a dimension flow parameter and used to define the ratio of the local flow velocity V to the local acoustic velocity a :

$$M = V / a \tag{2.20}$$

Mach number less than one correspond to subsonic flow and greater than one to supersonic flow. When the Mach number is equal to one then the flow is moving at precisely the velocity of sound. The throat of all supersonic nozzles must be equal to one.

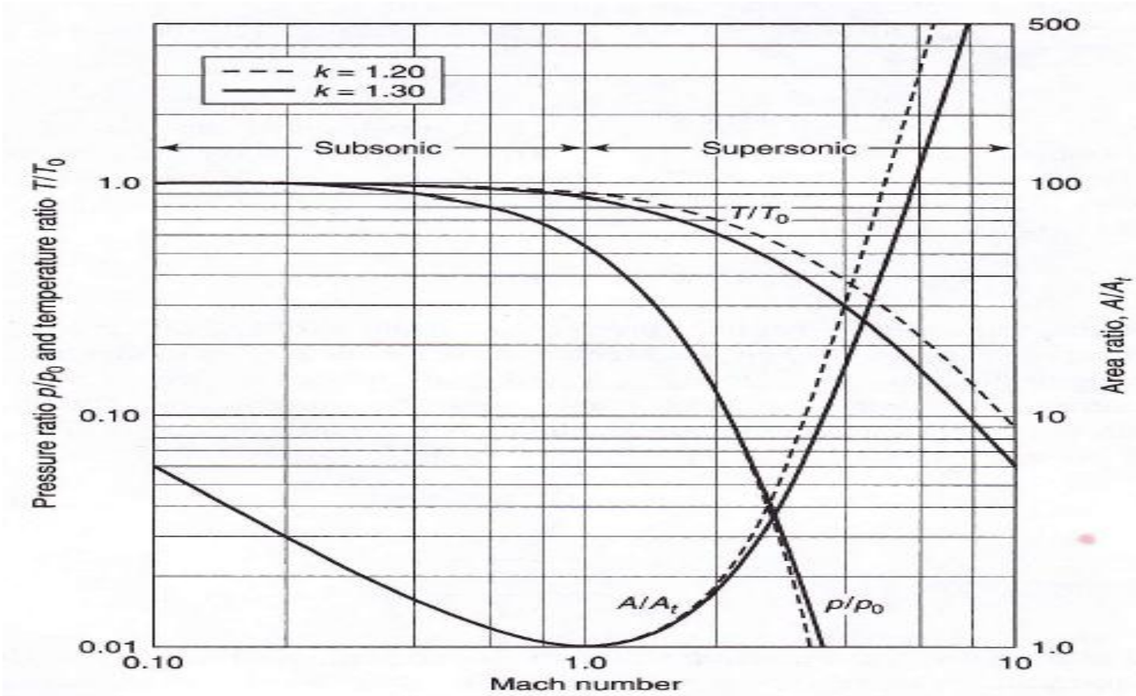


Figure 2.5 : Graph Pressure ratio and temperature versus Mach number (Relationship of area ratio, pressure ratio, and temperature ratio as functions of Mach number in a De Laval nozzle for the subsonic and supersonic nozzle regions)

- Source : Refferences Book (Rocket Propulsion Elementsmby George P. Sutton & Oscar Biblarz)

As can be seen from the figure above for subsonic flow the chamber contraction ratio A_1 / A_t can be small with values 3 to 6 and the passage is convergent. There is no noticeable effect of variation of the k. In solid rocket motors the chamber area A_1 refera port cavityow passage or port cavity in the virgin grain. With supersonic flow the nozzle section diverges and the area ratio becomes large quickly.

2.12 Load Cell

The load cell is a device to measure high thrust of lateral force such as in a turbine jet engine and rocket engine is normally prepared by using a simple block of load cell. A load cell is weighing equipment, which equips with force sensor to compute the force. Mechanical lever scales were widely used before the strain gauge. Most conventional load cells for loads up to 100kg contain a spring element made from steel, which is deformed under a load, and it is measured by a sensor.

The accuracy of this load cell is limited by hysteresis and creep. In 1843, English physicist Sir Charles Wheatstone invented a bridge circuit that could measure electric resistances, and it is ideal to measure resistance changes that occur in strain gages. Later, the first bonded resistance wire strain gage was developed in the 1940s. A strain gage load cell is converting the load acting on a system into electrical signals. The gauges themselves are bonded onto a beam or structural member who deforms when weight is applied. The strain changes the electrical resistance of the gauges in proportion to the load when weight is applied.

Strain-gage load cells convert the load acting on them into electrical signals. The gauges themselves are bonded onto a beam or structural member that deforms when weight is applied and in most cases, four strain gages are used to obtain maximum sensitivity and temperature compensation. Figure 2.6 show two of the gauges is in tension, and two in compression, and are wired with compensation adjustments. The strain changes the electrical resistance of the gauges in proportion to the load when weight is applied. Strain gage load cells continue to increase their accuracy and lower their unit costs and other load cells are fading into obscurity.

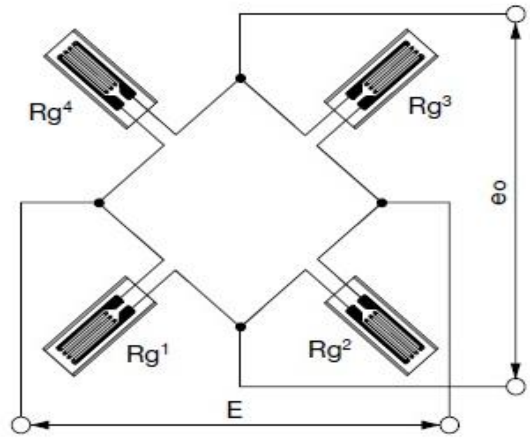


Figure 2.6: Winston bridge

- Source : Development load cell for high thrust measurement (Makeen Amin,Rizalman Mamat,Amir Aziz)

The strain gage is connected together. To measure the accurate value, the strain gage and adhesive should match the measuring material and operating conditions including temperature. Thus to obtain, is an output voltage that is proportional to a change in resistance. The microscopic output voltage is amplified in analogue or digital recording. The four gage system has four strain gages connected to each all four sides of the bridge.

CHAPTER 3

METHODOLOGY

3.1 Introduction

The main objective of this chapter is to discuss the method to obtain theoretical result, calculations and related propellant formulation. The theoretical result is obtained from calculations and software simulation in order to produce optimum design and to avoid failures. This chapter starts with making basic assumptions which is crucial in order to simplify the analysis of a rocket motor. This chapter also explain about the design and analysis of rocket nozzle using solidwork and theory calculation on how to create optimum dimension for converging and diverging nozzle.

3.2 Flow Chart of the Processing Method

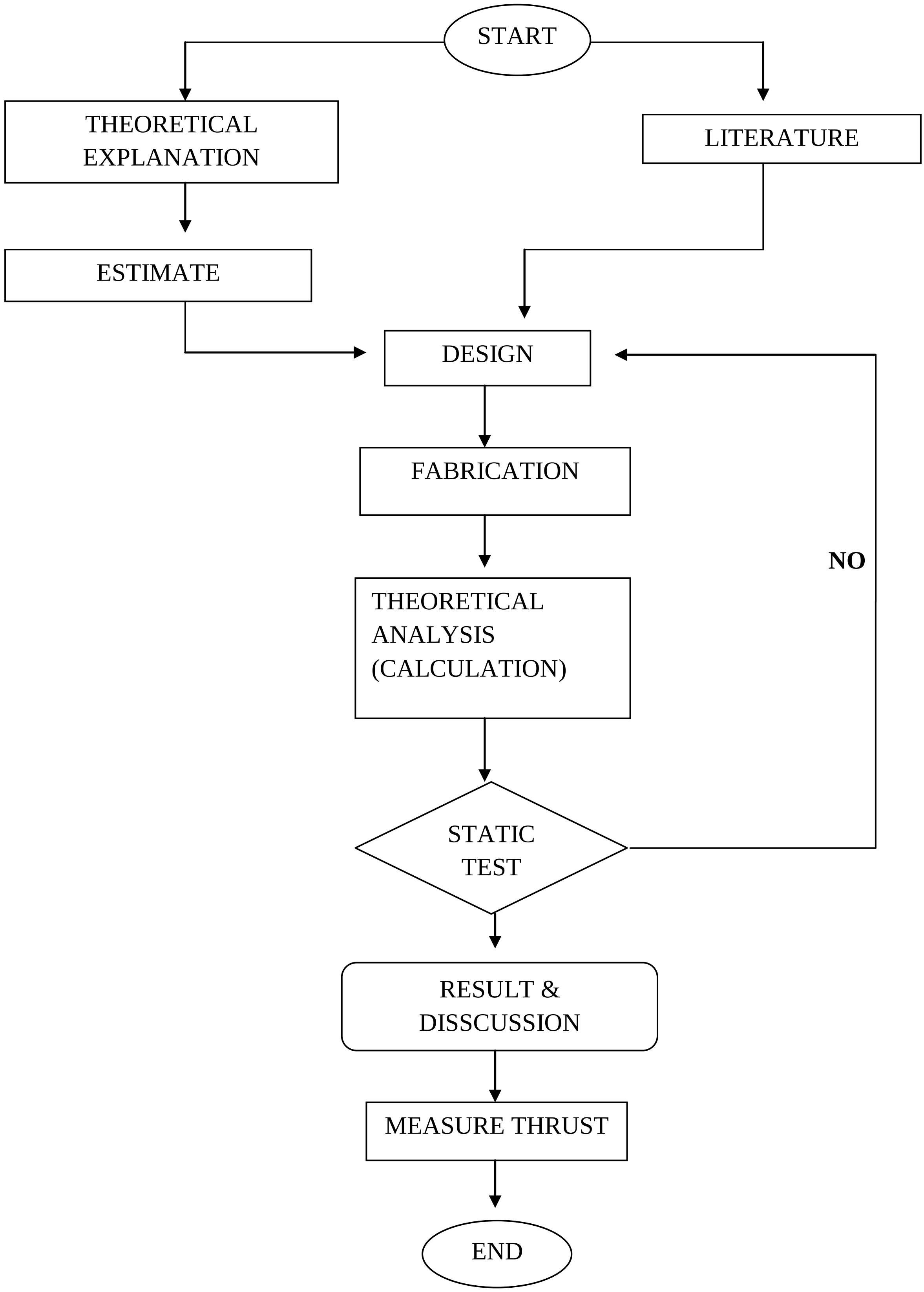


Figure 3.1 : Flow chart

3.3 Chemical Rocket Propellant Performance Analysis Using CHEM

In this chapter, the characteristic of nozzle based on theory calculation on how to find an optimum dimension for rocket motors are shown. The optimum dimension for rocket motor can be achieved by finding out the best percentage of mixed propellant. By using CHEM's software, the value of density, specific impulse and gamma (k) by a specific percentage of propellant can be determined. The result of Potassium nitrate and sucrose as a propellant from CHEM software are shown below:

a) 65% Potassium nitrate
b) 35% Sucrose

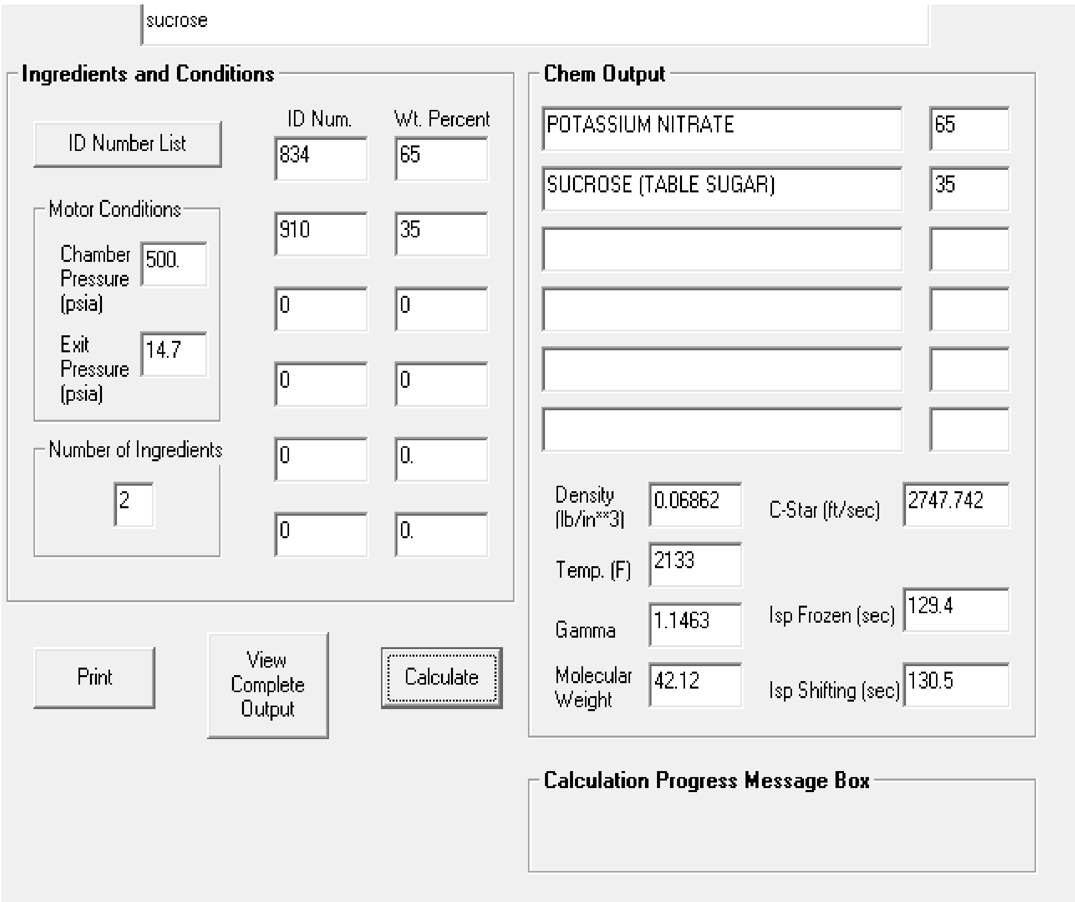


Figure 3.2 : Potassium Nitrate

Table 3.1: Parameters from CHEM

Characteristic	**Value**
Propellant density , ρ	0.06862 $\frac{lbf}{in^3}$
Specific Impulse, I_s	129.4 sec
Characteristic of velocity, c^*	2747.747 m/s
Specific heat ratio , γ	1.1463

Theoratically, the mass of oxidizer should be greater than the mass of fuel. These value will be used to determine the ideal composition of the propellant mixture. Figures 3.1 shows the values based on chemical equation that can be proven by using related equation.

3.4 Theoretical Calculation Parameter & Performance Of Nozzle

Based on related equation and specific value, we can calculate the theoretical result of the nozzle. An output data from CHEM software listed several parameters useful in the development of rocket motors. These parameters measured $c *, I_{sp}, C_F$, Mach no, and others use established equations for the characteristic performance parameter. They are recalculated and compare with the CHEM program output. Some basic data for example propellants, desired chamber pressure or likely reaction to be known or postulated. The three value list below :

Table 3.2: Parameters from CHEM

Characteristic	**Value**
Pressure Chamber,P_c	500 Psi
Burning time, t_b	129.4 sec
Exit diameter of Nozzle, D_e	2747.747 m/s

3.41 Theoretical Calculation using Potassium Nitrate (KNO_3)

Table 3.3: Throat 13 mm

Characteristic	**Value**
Propellant density , ρ	1899 3951 $\frac{kg}{m^3}$
Specific Impulse, I_s	129.4 sec
Characteristic of velocity, c^*	2747.747 m/s
Specific heat ratio , γ	1.1463

$2k^2 = 2(1.1463)^2 = 2.6280$

$k-1 = 0.1463$

$k+1 = 2.1463$

$\frac{P_2}{P_1} = 0.02939$

$(k+1)/(k-1) = 14.6705$

$k-1/k = 0.1276$

$\lambda = \frac{1}{2}(1+\cos\alpha) = 0.9829$

i. Thrust Ratio, C_F

$$C_F = \lambda\left[\frac{2k^2}{k-1}\left(\frac{2}{k+1}\right)^{(k+1)/(k-1)}\left(1-\frac{P_2}{P_1}\right)^{k-1/k}\right]^{1/2}$$

$$= 0.9829\left[\frac{2.6280}{0.1463}\left(\frac{2}{2.1463}\right)^{(2.1463)/(0.1463)}(1-0.02939)^{0.1276}\right]^{1/2}$$

$$= 0.9829\left[17.9631(0.3544)(0.9962)\right]^{1/2}$$

- $C_F = 6.0080$

Where :

C_F = Thrust Ratio
λ = Thrust Efficiency
P_2 = Exit Pressure
P_1 = Chamber Pressure
k = Specific Heat Ratio - chamber

ii. Pressure Throat, P_t

$$\frac{P_t}{P_1} = \left[2/k+1\right]^{k/(k-1)}$$

$$= \left[2/2.1463\right]^{7.8352}$$

- $P_t = 1982.62 kPa$

Where :

P_t = Throat Pressure
P_1 = Chamber Pressure
k = Specific Heat Ratio - chamber

iii. Nozzle Area Ratio, A_t / A_2

$$A_t / A_2 = \left(k+1/2\right)^{1/(k+1)} \left(\frac{P_2}{P_1}\right)^{1/k} \left[\frac{k+1}{k-1}\left[1-\left(\frac{P_2}{P_1}\right)^{(k-1)/k}\right]\right]^{1/2}$$

$$= (1.07315)^{0.4659}(0.02939)^{0.8723}\left[14.6705\left[1-0.02939^{0.1276}\right]\right]^{1/2}$$

$$= (1.0334)(0.0461)\left[14.6705-9.3537\right]^{1/2}$$

$A_t / A_2 = 0.1098$

- ❖ $A_2 / A_t = 9.1052$
- ❖ $\in$ = 9.1052

Where :

A_2 = Exit Area

A_t = Throat Area

P_2 = Exit Pressure

P_1 = Chmaber Pressure

k = Specific Heat Ratio – chamber

iv. Diameter And Area of Throat

$A_2 / A_t = 9.1052$

$A_t = A_2 / 9.1052$

- ❖ $A_t = 1.32\ cm^2$

$D_t = \left[4A_t / \eta\right]^{1/2}$

$= \left[4(3.5144)/\eta\right]^{1/2}$

- ❖ $D_t = 1.3\ cm$

v. Length of Throat

$l_t = D_t / 2$

= 1.3/2

- ❖ l_t = 0.65 cm

vi. Length of Diverging Nozzle

$(D_e - D_t) / 2l = \tan 15^o$

$4.2847 / 2l = 0.2679$

$2l = 15.9936$

- ❖ $l = 8$ cm

Where :

$l =$ Length of throat

$D_t =$ Diameter of throat

$\alpha = 15^o$

vii. Thrust

$F = C_F A_t P_1$

$= (6.0080)(1.32\times10^{-4})(3447378.5)$

❖ F= 2734 N

Where :

C_F = Thrust Ratio

P_1 = Chmaber Pressure

A_t = Throat Area

viii. Exit Velocity

$v_e = I_s g_o$

= (129.4 sec)(9.81 m/s^2)

❖ v_e = 1269.414 m/s

Where :

I_s = Specific Impulse (sec)

v_e = Exit velocity (m/s)

g_o = gravity (m/s^2)

ix. Propellant Mass Flow Rate

$\dot{m} = F / v_e$

$= 1127.1688 N / 1269.414 m/s$

❖ $\dot{m} = 0.8879\ kg/\sec$

Where :

F = Thrust (N)

v_e = Exit velocity (m/s)

$\dot{m}$ = Mass Flow Rate (kg/sec)

x. Thickness of Propellant

$L = r \times t_b$

$r = \sqrt{k}\left[2/(k+1)\right]^{(k+1)/2(k-1)}$

$$= (1.0706)\left[0.9318\right]^{(0.1570)}$$

$$r = 1.0588$$

$$L = 1.0588 \times 3\sec$$

❖ L = 3.1764 cm

Where :

L = Propellant Thickness

t_b = Burning Time

xi. Area of Burning

$$A_{bt} = \dot{m} \Big/ r\rho$$

$$= 0.8879kg/\sec \Big/ (1.0588 \times 10^{-2})(18994.27)$$

❖ $A_{bt} = 0.004415\ cm^2$

Where :

ρ = Propellant Density

$\dot{m}$ = Mass Flow Rate

xii. Length of Propellant

a) Initial Burning Surface

$$A_{bi} = L_P(D_i\pi) + 2\pi\left[\left(\frac{D_o^2}{4} - \frac{D_i^2}{4}\right)\right]$$

$$= L_P(0.4\pi) + 2\pi\left[3.0625 - 0.4225\right]$$

❖ $A_{bi} = 1.25L_P + 16.5876$ ....(1)

b) Final Burning Surface

$$A_{bf} = \left[L_P - 2t_p\right]\left(D_o\pi\right)$$

$$= \left[L_P - 2(3)\right]\left(3.5\pi\right)$$

$$A_{bf} = 10.9955L_P - 65.9734 \text{ ....(2)}$$

Equation (1) equal to equation (2) to get length of propellant

- ❖ L_P = 9 cm

Where :
A_{bi} = Initial Burning Surface
L_P = Propellant Length
D_i = Propellant Inner Diameter
D_o = Propellant Outer Diameter

xiii. Volume Of Propellant

$$V_P = \pi\left[\left(\frac{D_o^{\;2}}{4} - \frac{D_i^{\;2}}{4}\right)\right]L_P$$

$$= \pi\left[10.24 - 0.0004\right]L_P$$

$$= 32.1686(9)$$

- ❖ $V_P = 289.5178 \ cm^3$

Where :
V_P = Propellant Volume
L_P = Propellant Length
D_i = Propellant Inner Diameter
D_o = Propellant Outer Diameter

3.4.2 Theoretical Calculation using Potassium Nitrate (KNO_3)

[Throat 15 mm]

i. Thrust Coefficent, C_F

$$C_F = \lambda \left[\frac{2k^2}{k-1} \left(\frac{2}{k+1} \right)^{(k+1)/(k-1)} \left(1 - \frac{P_2}{P_1} \right)^{k-1/k} \right]^{1/2}$$

$$= 0.9829 \left[\frac{2.6280}{0.1463} \left(\frac{2}{2.1463} \right)^{(2.1463)/(0.1463)} \left(1 - 0.02939\right)^{0.1276} \right]^{1/2}$$

$$= 0.9829 \left[17.9631 (0.3544)(0.9962) \right]^{1/2}$$

❖ $C_F = 2.4770$

Where :

C_F = Thrust Ratio

λ = Thrust Efficiency

P_2 = Exit Pressure

P_1 = Chamber Pressure

ii. Throat Pressure, $\boldsymbol{P_t}$

$$\frac{P_t}{P_1} = \left[2/k+1 \right]^{k/(k-1)}$$

❖ $P_t = 1982.62 kPa$

Where :

P_t = Throat Pressure

P_1 = Chamber Pressure

k = Specific Heat Ratio – chamber

iii. Nozzle Area Ratio, A_t / A_2

$$A_t / A_2 = \left(k+1/2\right)^{1/(k+1)} \left(\frac{P_2}{P_1}\right)^{1/k} \left[\frac{k+1}{k-1}\left[1-\left(\frac{P_2}{P_1}\right)^{(k-1)/k}\right]\right]^{1/2}$$

$$= (1.10645)^{0.4519}(0.0239)^{0.8244}\left[10.3941(1-0.02939^{0.1755}\right]^{1/2}$$

$$= (0.05715)(2.1902)$$

- $\frac{A_2}{A_t} = 0.1251$
- $A_2 / A_t = 9.1052$

Where :

A_2 = Exit Area
A_t = Throat Area
P_2 = Exit Pressure
P_1 = Chamber Pressure
k = Specific Heat Ratio – chamber

iv. Diameter And Area of Throat
- Assume throat diameter 15 mm

$D_t = 15mm$
$A_t = 1.767cm^2$

v. Length of Throat

$l_t = D_t / 2$

= 0.75cm

- l_t = 0.75 cm

vi. Length of Diverging Nozzle

$(3.5-1.5)/2l = \tan 15^o$

$2/2l = 0.2679$

$2l = 7.4654$

$l = 3.7327$

- ❖ $l = 3.7327$ cm

Where :

$l =$ Length of throat

$D_t =$ Diameter of throat

$\alpha = 15^o$

vii. Thrust

$F = C_F A_t P_1$

$= (2.4770)(1.767\times10^{-4})(3447378.5)$

$= 2508.8689 N$

Where :

C_F = Thrust Ratio

P_1 = Chamber Pressure

A_t = Throat Area

viii. Exit Velocity

$v_e = I_s g_o$

= (129.4 sec)(9.81 m/s^2)

- ❖ v_e = 1269.414 m/s

Where :

I_s = Specific Impulse (sec)

v_e = Exit velocity (m/s)

g_o = gravity (m/s^2)

ix. Propellant Mass Flow Rate

$\dot{m} = F / v_e$

= 1.18863 kg/sec

Where :

F = Thrust (N)

v_e = Exit velocity (m/s)

$\dot{m}$ = Mass Flow Rate (kg/sec)

Thickness of Propellant

$$L = r \times t_b$$

$$r = \sqrt{k}\left[2/(k+1)\right]^{(k+1)/2(k-1)}$$

$$r = 0.6666$$

$$= 0.6666 \times 3\sec$$

❖ L = 2 cm

Where :

L = Propellant Thickness
t_b = Burning Time

x. Area of Burning

$$A_{bt} = \dot{m} / r\rho$$

$$= (1.18863) / \left[\left(0.6506 \times 10^{-2}\right)(1691.8242)\right]$$

❖ $A_{bt} = 1079.88 cm^2 \; cm^2$

Where :
ρ = Propellant Density

$\dot{m}$ = Mass Flow Rate

xi. Length of Propellant
a) Initial Burning Surface

$$A_{bi} = L_P(D_i\pi) + 2\pi\left[\left(\frac{D_o^{\ 2}}{4} - \frac{D_i^{\ 2}}{4}\right)\right]$$

$$= L_P(2.4\pi) + 2\pi[8.8]$$

$$= 7.5398L_P + 55.2922$$

- $A_{bi} = 7.5398L_P + 55.2920 \quad (1)$

b) Final Burning Surface

$$A_{bf} = \left[L_P - 2t_p\right]\left(D_o\pi\right)$$
$$= \left[L_P - 2(2)\right]\left(6.4\pi\right)$$
$$A_{bf} = 20.1062L_P - 80.4248 \qquad (2)$$

Equation (1) equal to equation (2) to get length of propellant

$$A_{bi} + A_{bf} = A_{bt}$$
$$= (20.1062L_P - 80.4248) + (7.5398L_P + 55.2920)$$
$$(27.646)L_P - 25.1328 = 789.5878$$

- L_P = 30 cm

Where :
A_{bi} = Initial Burning Surface
L_P = Propellant Length
D_i = Propellant Inner Diameter
D_o = Propellant Outer Diameter

xii. Volume Of Propellant

$$V_P = \pi\left[\left(\frac{D_o^{\,2}}{4} - \frac{D_i^{\,2}}{4}\right)\right]L_P$$

$$= \pi\left[10.24 - 1.44\right]L_P$$

$$= \pi(264)$$

- $V_P = 830 \;\; cm^3$

Where :
V_P = Propellant Volume
L_P = Propellant Length

3.5 Summarize From Theory Of Calculation

Table 3. 4 : Characteristic of Nozzle

Type of Throat Diameter	13 mm	15 mm	20 mm
Thrust Ratio	2.4770	2.4770	2.4770
Pressure Throat	4264 kPa	4010 kPa	1982.62 kPa
Exit Diameter	3.5 cm	3.5 cm	7 cm
Exit Area	9.6211 cm^2	9.6211 cm^2	38.48 cm^2
Nozzle Area Ratio	9.1052	9.1052	9.1052
Diameter and Area Throat	1.3 cm / 1.32 cm^2	1.5 cm/1.76 cm^2	2.1153cm/3.51 cm^2
Length of Throat	0.0675 cm	0.08 cm	1.05765 cm
Length of Diverging Nozzle	4.2 cm	4.2 cm	8 cm
Thrust	2245 N	2580 N	5000 N
Exit Velocity	1269.4 m/s	1056.3 m/s	1269.414 m/s
Material	Low carbon steel	Low carbon steel	Low carbon steel
Inner Angle	15 degree	15 degree	15 degree
Outer Angle	30 degree	30 degree	30 degree
Yield Stress (Mpa)	165	165	165
Ultimate Tensile Stress(Mpa)	296	296	296
Modulus Young (Mpa)	200100	200100	200100
Melting point	1510 celsius	1510 celsius	1510 celsius

Table 3.5: Physical characteristic for propellant

Propellant	**KNO-C**
Weight (gram)	$290cm^3$
Type	Hollow Cylindrical
Segment	1
Lenght (cm)	30 cm
Outside diameter, OD (cm)	3.4 cm
Diameter, ID (cm)	0.4 cm

Table 3.3: Casing Characteristic

Material	**Stainles steel (SAE30304)**
Thickness,t (mm)	2
Outer Diameter,OD (mm)	37.22
Yield Stress (Mpa)	517
Modulus Young,(E)	724
Melting point	186300

3.6 Design of Nozzle

The design of the nozzle are drawn using Solidwork 2010. Based on data obtained from theoretical calculations, we can design the optimum nozzle. Three different throat diameter are choose that is 13mm, 15mm and 22mm. Different throats diameter will result in different value of thrust. The experimental value is compared with the theoretical value to prove the theoretical calculation and to choose the most suitable characteristic to produce optimum nozzle which will be fabricated.

3.6.1 Angle 30 Degree / Throat 13 mm

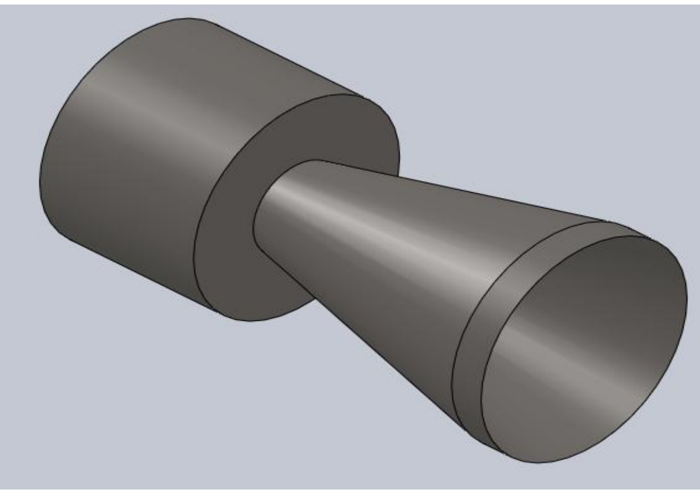


Figure 3.2 : Nozzle with throat 22 mm

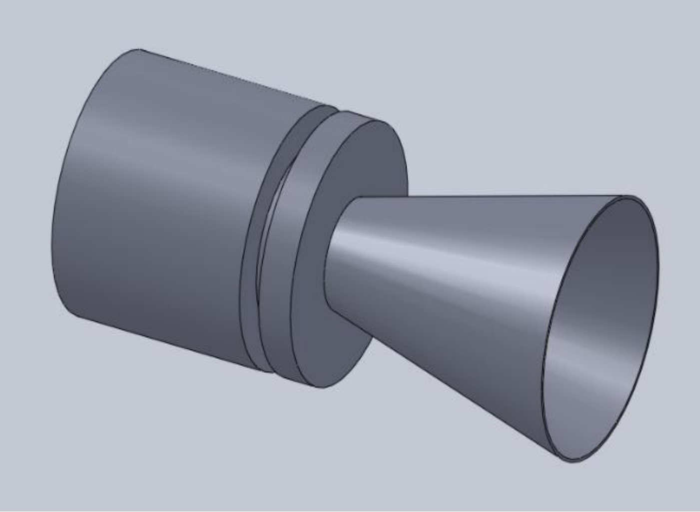


Figure 3.3 : Nozzle with throat 13 mm

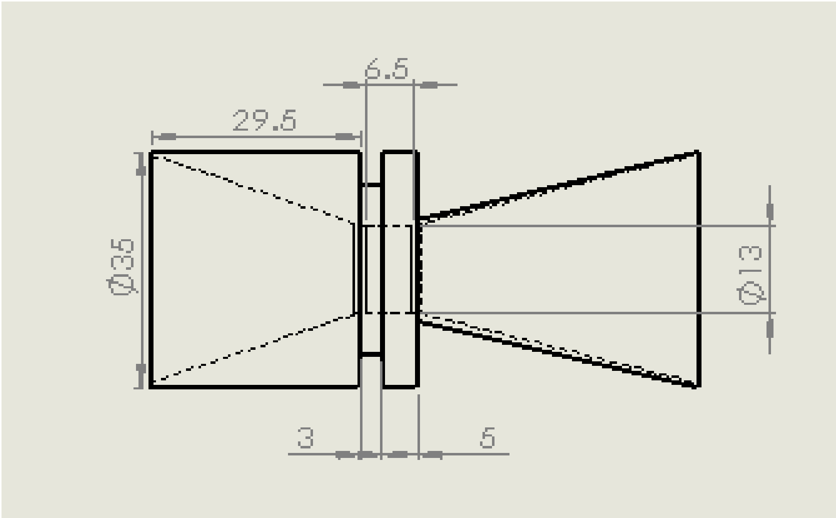


Figure 3.4 : Drawing of nozzle throat 13 mm with dimension

The design comprises of 30^0 angle of chamber and 15^0 for divergent section. This nozzle is 70 mm in length. It also had 35 mm for exit diameter and 13 mm throat diameter. For this design, the length of throat must not be less than 6.5 mm to ensure the flow can change from turbulent to laminar flow.

3.62 Nozzle with 15 mm diameter of throat

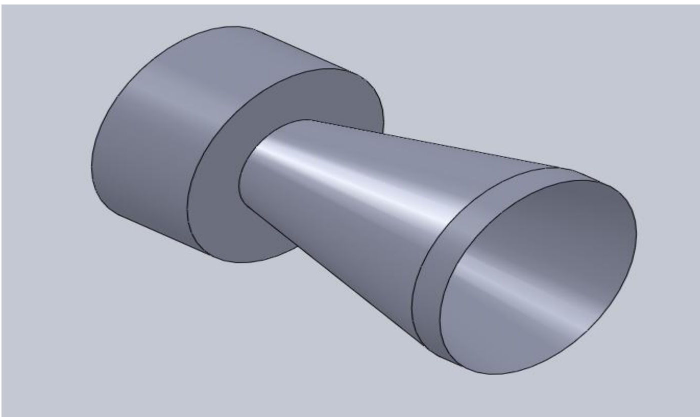


Figure 3.5 : Nozzle with 15 mm diameter of throat

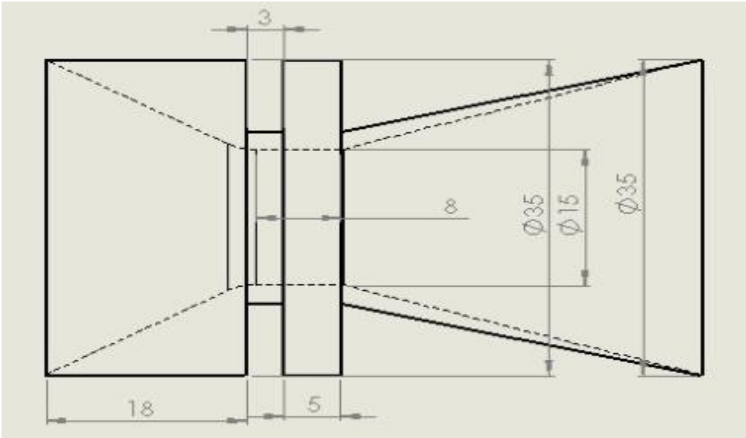


Figure 3.6: Drawing of nozzle with dimension

The design has 30^0 angle of chamber and 15^0 for divergent section. This nozzle is 70 mm in length. The dimension of exit diameter and throat diameter are 35 mm and 15 mm respectively. For this design, the length of throat can not be less than 7.5 mm to ensure the flow can change from turbulent to laminar flow (stagnation condition).

3.7 Casing Analysis

i. Casing circumferences :

$$s = j\theta = (3.5 \times 2\pi)$$
$$= 21.9911cm$$

ii. Gap between the pin :

$$q = sc - (D_{pin} \times n_{pin})$$
$$= 219.911 - (4.1656 \times 8)$$
$$= 186.5826mm$$

iii. Area each gap in hoop direction :

$$A_H = t_c \times q$$
$$= 2mm \times 186.5826$$
$$= 3.73 \times 10^{-4}$$

iv. Force on hoop direction from chamber pressure :

$$F_H = \sigma_H \times A_H$$
$$= (60.3291 Mpa)(3.73\times10^{-4})$$
$$= 22.5027 kN$$

v. The area that holds pressure is the casing :

$$A_L = \pi\left[\frac{OD^2}{4} - \frac{ID^2}{4}\right]$$
$$= \pi\left[2.01\times10^{-4}\right]$$
$$= 6.3146\times10^{-4} m^2$$

vi. Hoop stress

$$P = 3447.379\times10^3$$
$$r = 35\times10^{-3}$$
$$t = 2\times10^{-3}$$
$$\sigma_H = \Pr\!\big/ t$$
$$= (3447.379\times10^3)\left(35\times10^{-3}\right)/\left(2\times10^{-3}\right)$$
$$= 60.3291 Mpa$$

vii. Longitudinal stress

$$\sigma_L = \Pr\!\big/ 2t$$
$$= (3447.379\times10^3)\left(35\times10^{-3}\right)/\,2\left(2\times10^{-3}\right)$$
$$= 30.1645 Mpa$$

viii. Resultant stress

$$\sigma_r^{\ 2} = \sigma_H^{\ 2} + \sigma_L^{\ 2}$$
$$= (60.3291 Mpa)^2 + (30.1645 Mpa)^2$$
$$= 67.4158 Mpa$$

ix. Ultimate tensile stress aluminum & Safety factor

$$\sigma_{UL} = 159 MPa$$
$$S.F = \sigma_{UL} / \sigma_r$$
$$S.F = 2.3585$$

3.8 Force longitudinal direction

$$F_L = \sigma_L + A_L$$
$$= \left(30.1645 \times 10^6 Pa\right)\left(6.3146 \times 10^{-4}\right)$$
$$= 19.04 kN$$

x. Resultant force

$$F_r^{\ 2} = F_H^{\ 2} + F_L^{\ 2}$$
$$= \left(19.04\right)^2 + \left(22.5027\right)^2$$
$$= 19.04 kN$$

xi. Total area each gap

$$A = A_H \times n$$
$$= \left(3.73 \times 10^{-4} m^2\right)\left(8\right)$$
$$= 2.984 \times 10^{-3} m^2$$

xii. Stress act on casing

$$\sigma = {}^{F}\!/\!_{A}$$
$$= \left(19.04 kN\right) / \left(2.984 \times 10^{-3} m^2\right)$$
$$= 63.8070 MPa$$

xiii. Safety factor

$$S.F = \sigma_U / \sigma$$
$$= \left(159 \times 10^6\right)\left(63.8070 \times 10^6\right)$$
$$= 2.4920$$

3.9 Pin Analysis

Material = Stainless steel

i. Force on bulkhead

$$F_b = P_2 \times A_b$$
$$A_b = \pi D^2 / 4$$

$$= \pi\left(70\times10^{-3}\right)^2 \big/ 4$$

$$= 3.8484\times10^{-3}\,m^2$$

$$F_b = \left(3447.379\times10^3\right)\times\left(3.8484\times10^{-3}\right)$$

$$= 13.2671kN$$

ii. Use 16 pins. So force acted on each pin :

$$F_{pin} = 13.2671\times10^3 / 16$$

$$= 829.1937N$$

iii. Minimum pin radius

$$R_{\min} = \left[\frac{F_{pin}}{\pi\sigma_{USS}}\right]^{1/2}$$

$$= 1.2884mm$$

$$S.F = R_{pin} / R_{pin(\min)}$$

$$= 1.6170$$

= 1.6170

iv. Shear stress on the pin

$$\gamma_{pin} = \frac{F_{pin}}{A_{pin}}$$

$$= \frac{829.1937N}{1.3628\times10^{-5}}$$

$$= 60.8430MPa$$

v. Safety factor

$$\sigma_{UL} = 159MPa$$

$$S.F = \frac{159MPa}{60.8430MPa}$$

$$= 2.61328$$

3.10 Analysis (COSMOS Flow)

Simulation analysis using COSMOS flow is conducted on the rocket motor design to ensure that the design is sustainable under the design load. Solidwork

COSMOS flow analysis is used to study the possible deformations of the rocket motor components and to determine whether the design would fail or not under applied pressure and thermal environment. This analysis is also used to study the characteristic of the flow inside the rocket motor such as the flow temperature distribution, pressure distribution, velocity of the flow and the Mach number. The Solidworks Simulation is used to simulate the flow

3.10 .1 First Model (Throat 20 mm)

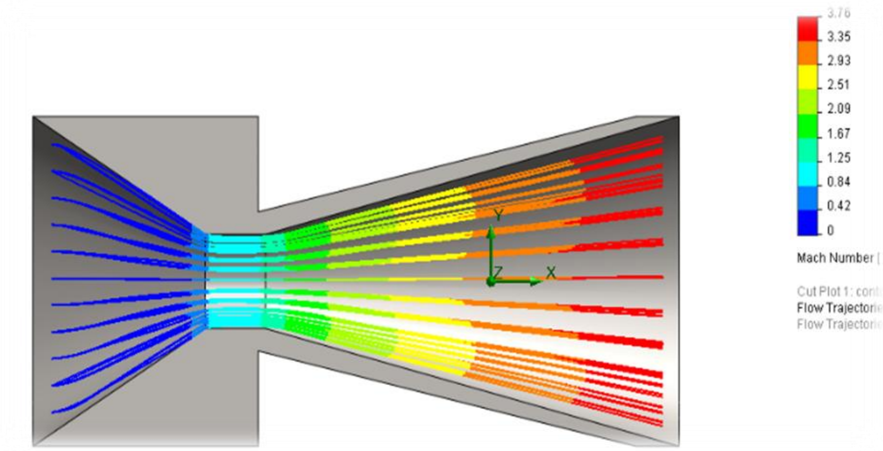


Figure 3.7 : Mach number of nozzle throat 20mm

The figure 3.7 shows the flow trajectories of Mach number for 20 mm throat diameter nozzle. From the theoretical calculation of optimum nozzle design, Mach number should be equal to 1 on throat and greater than 1 for convergence section. From figure 3.7, it is known that the value of Mach number on throat is between 0.84 and 1.25. This value is closed to 1. For converging section, the range of Mach number value is between 1.67 and 3.35.

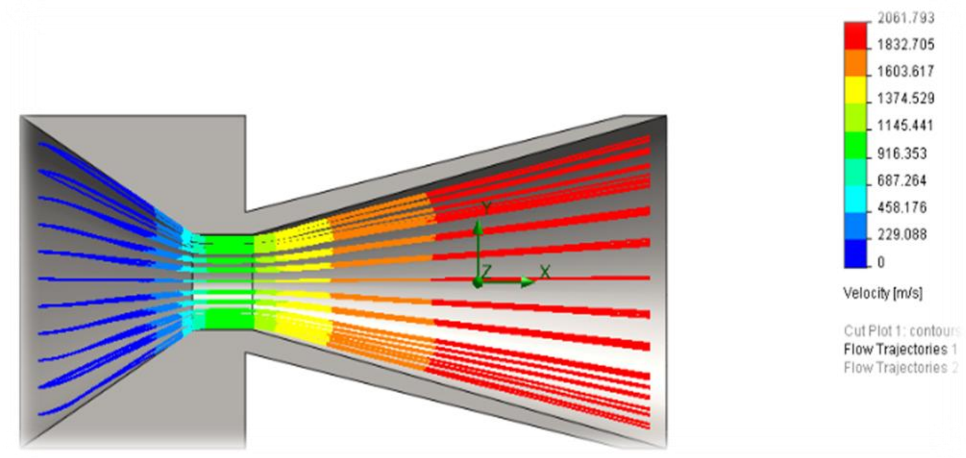


Figure 3.8 : Velocity

The figure 3.8 shows the flow trajectories of velocity in the nozzle with 20 mm throat. From the theoretical calculations of the optimum nozzle design is 1395 m/sec, the velocity should be closed with the simulation result using cosmos flow. From figure 3.8, the analysis shows the value of velocity at the throat are between 916.353 m/s and 1145.441 m/s. For converging section, the range of velocity are between 1374.529 m/s and 2061.793 m/s.

3.10 .2 Second Model (Throat 13 mm)

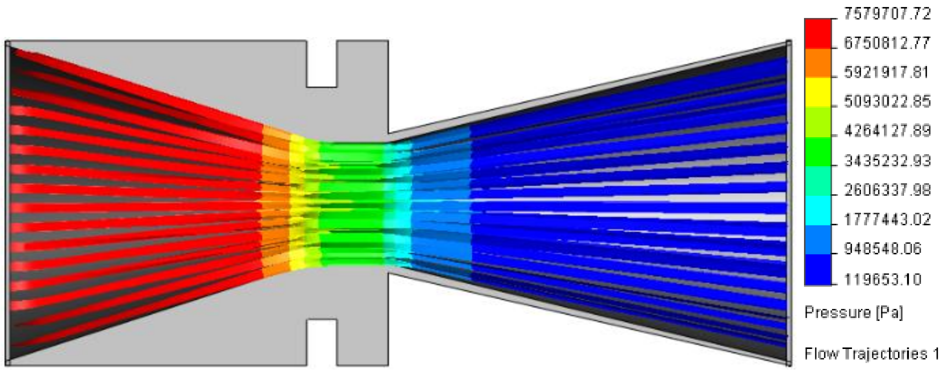


Figure 3.9 : Pressure

The figure 3.10 shows flow trajectories of pressure in the 13 mm diameter throat nozzle. From theoretical calculation of optimum design of nozzle, the value of pressure should be equal 7579 kPa. The analysis shows the pressure at the throat area are between 2606 kPa and 5093 kPa while the pressure at converging area will decrease to atmospheric pressure . The highest pressure in the chamber is 7579 kPa.

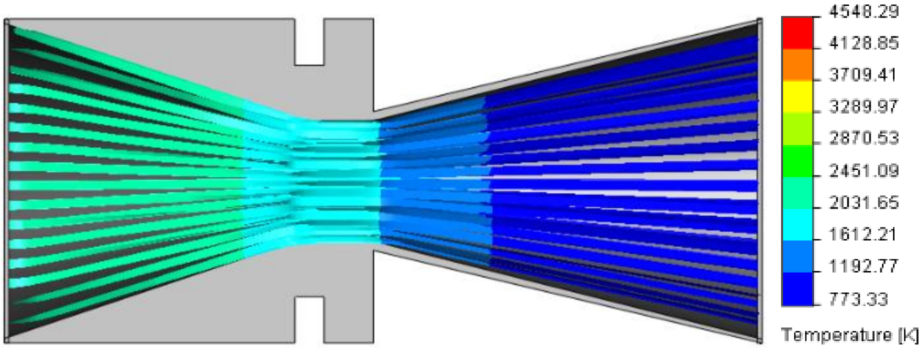


Figure 3.10 : Temperature

The figure 3.11 shows the flow trajectories of temperature of the nozzle. The temperature inside the nozzle should be equal to the theoretical value from the calculation 3578 celcius.From the analysis of flow trajectories of temperature, the value of temperature at the throat are between 1612.21 Celsius and 2031.65 Celsius. This value approximately to one. For converging section, the range of temperature are between 1612 Celsius and 773.33 Celsius.

3.10.3 Nozzle throat 15mm

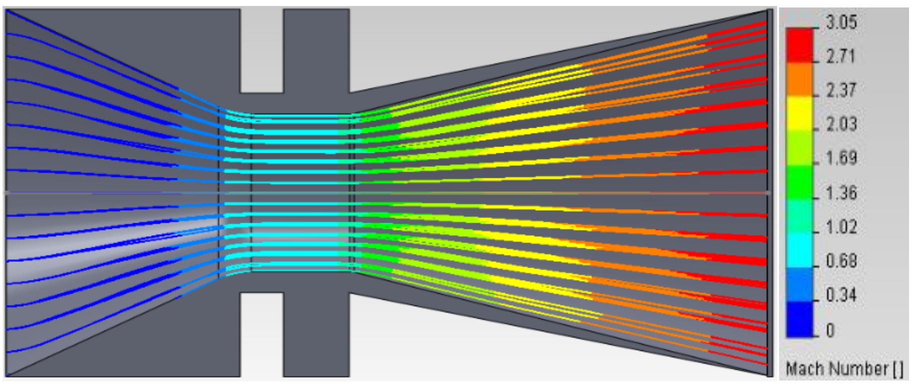


Figure 3.11: Mach Number

Based on figure 3.12, the flow trajectories of Mach number for 15 mm diameter throat nozzle is shown. From the theoretical calculation of the optimum design of the nozzle, Mach number should be equal to 1 at the throat and at the convergence section the Mach number should be greater than one. The value of Mach number from the simulation at the throat are between 0.68 and 1.02. This value is closed to one. For converging section the range of Mach number is between 1.69 and 3.05.

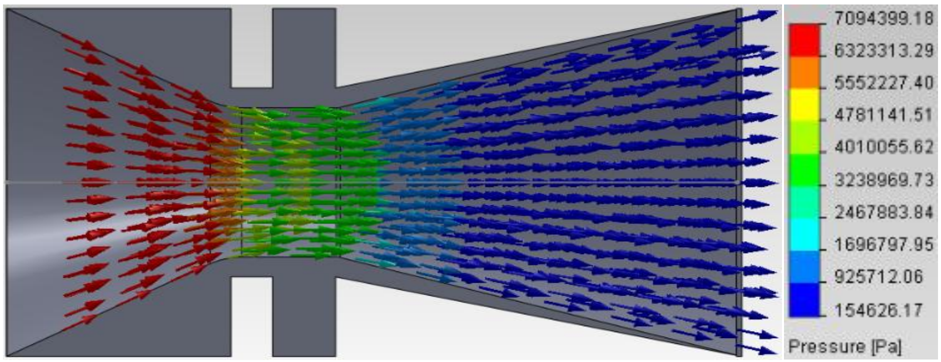


Figure 3.12 : Pressure

Based on figure 3.13, flow trajectories of pressure for 15 mm diameter nozzle. From theoretical calculation of optimum nozzle design, the value of pressure should be equal 7094 kPa. The result from the analysis shows the pressure at the throat are between 3238 kPa and 4781 kPa. Pressure at converging area is decreasing gradually until it reaches atmospheric pressure. The highest pressure inside the chamber is 7094 kPa.

3.11 Fabrication

The best material for fabricating the nozzle is stainless steel but in this project mild steel is used because it is easier to fabricate and the weight reduction between nozzle design can be seen more clearly. In this subtopic, the setups and procedures that was used to fabricate rocket motors (nozzle) is discussed. As stated earlier, there are three section in nozzle that is convergent section, throat section and divergence section. For convergence section, three different throat diameters is used that is 13mm 15mm and 20 mm. The diameter at the throat were chosen based on the theory of fluid flow, smaller diameters at throat will produce higher thrust.

Figure 3.13: Cutting the Mild Steel bar

In the fabricating process of the rocket motor, several machines such as CNC machine, wire cut and lathe machine are used. Figure 3.14 shows the mild steel bar being prepared and cut into designed size.

Figure 3.14: work-piece in the turning process

Based on figure 3.15, lathe machine is used for surface finishing. A suitable cutting speed was set before running the machine. Surface finishing is done to get correct diameter . The total length from the bottom of the body to the top of the cone was calculated between 65mm and 70mm. Constant cutting speed is used during fabricating process. For mild steel, the suitable RPM and cutting speed 560 Rev/min and 200 respectively. The cutter used is a carbide tool bit

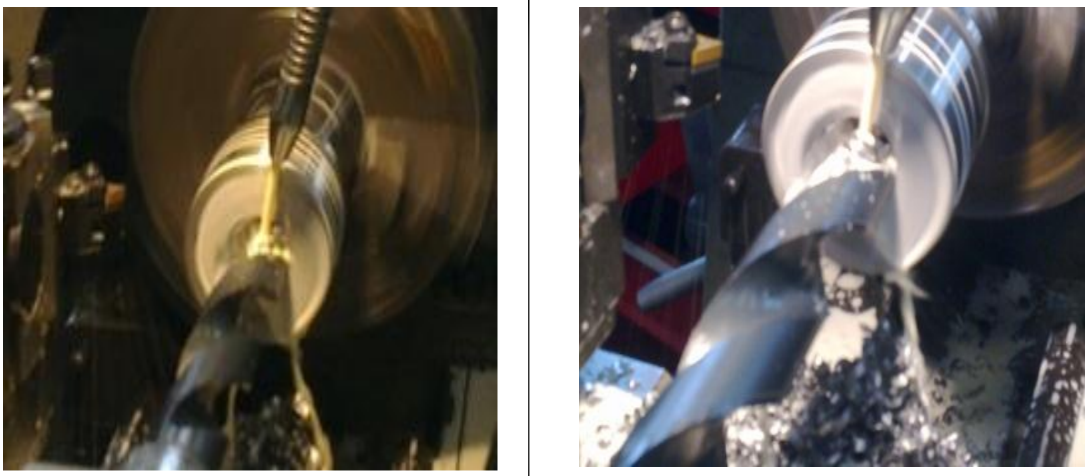


Figure 3.15 (a) & Figure 3.15 (b) : Fabrication of internal cone

A drill is used to fabricate the internal cone. The bore rim tool bit is used in the drilling process. The outer cone shape is done first and then drilled to make a hole. Diameter of step drill followed are 9.5mm, 12mm, 15mm, 18mm and 20mm. New cutting speed was used in drill process.

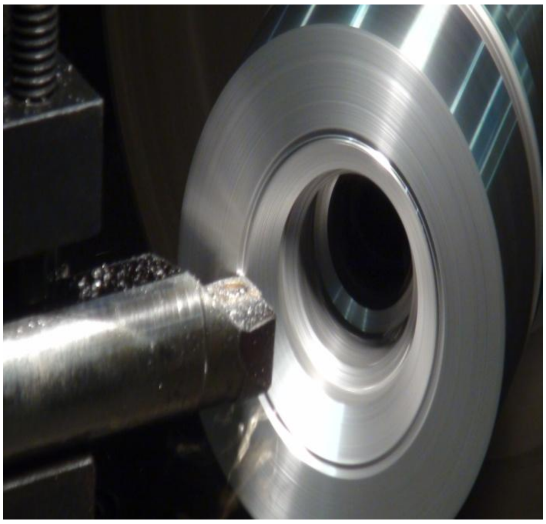



Figure 3.16 (a) & 3.16 (b): The fabrication of internal cone using bore rim tool bit

The angle of tool bits were fixed at 15 degree and 30 degree for convergent and divergent section respectively in order to get desired shape. The maximum allowable thickness of mild steel for each turning is 0.5 mm with rotation of 550 RPM to get a smooth surface. To precisicely measured the inner length, the thickness of the outer diameter is used as a reference.

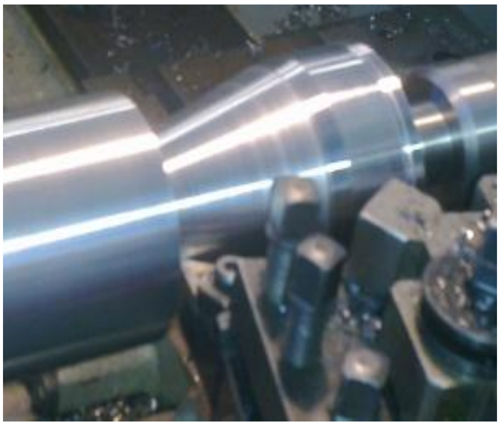



Figure 3.17 (a) & Figure 3.17 (b): External cone being fabricated

For divergent section the external cone was fabricated using carbide tool bit. Position of carbide tool was twisted 15 degrees. Make sure choosing a small value angle 15 degrees than internal cone angle to make an external cone to prevent the nozzle wall very thin.

Figure 3.18: Grinding process to get smooth surface

Air grinder and sandpaper were used to smoothen the inner surface of the nozzle. Smooth surface is required in convergent section to avoid chocking and stagnation pressure build up as fluid flow through the rocket motors. If thye pressure is to high, rocket motors may explode. The grinding process is done manually and must be done while the nozzle is rotating in order to get a fine surface

Figure 3.19: Threading process

Hole and thread for a screw are fabricated. The milling machine is used to make the hole because it is more precise. This process must be done with caution in order to prevent the tool bits from broken. All of the complete rocket motor will show at the appendix F.

Figure 3.20: Complete Nozzle

CHAPTER 4

TESTING

4.1 Static Thrust Testing

The static thrust test was conducted to measure the value of the thrust, average thrust, pressure and temperature using the static test rig.The standard nozzle has a conical nozzle profile with 15 degree divergence half angle. It was fabricared from mild steel bar stock turned and bored to the desired dimensions on a standard lathe. The nozzle is mounted to the combustion chamber by six screw.

Table 4. 1 : Different length propellant

PROPELLANT (% /total length)	TIME (sec)	REMARKS
30 %	1.5 – 2 sec	successful
50 %	3 sec	successful
70 %	4 sec	successful
100 %	6 sec	successful

The testing was conducted by preparing a solid rocket motor (SRM) and a propellant is inserted in the SRM. The SRM was held by a frictionless holder on the static test and these testing were conducted at a field to avoid accidents. Based on figure 4.1 it shows the locations for igniter installations and the overall layout of the testing setup. The igniter was connected to the 12V battery. A load cell was attached to the steel holder opposite the SRM to measure the thrust. A strain gauge was installed to convert the force acted on the load cell to voltage which is being measured by a data acquisition (DAQ) system.

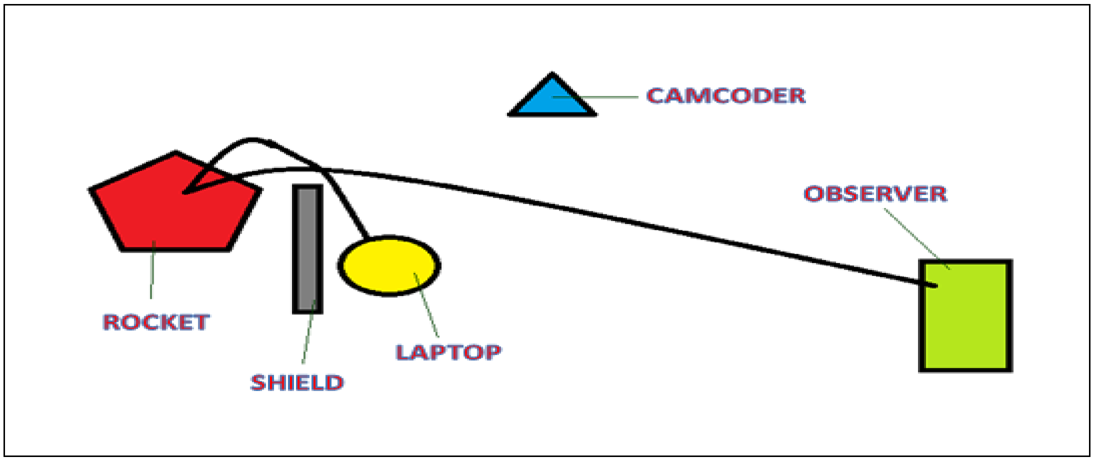


Figure 4.1 : Static thrust testing setup

4.2 Rocket motor

This motor has been used quite extensively for static thrust testing and has good reliability. The drawing of rocket motor can be referred in chapter 3. The rocket motor has 35mm diameter and 300mm length, including the nozzle.

The nozzle employed has a conical profile with 15 degree divergence angle and was made from mild steel. The nozzle contour is rounded at the throat to avoid sharp discontinuities, which could lead to shock losses as reported by Nakka . The nozzle was fabricated with well-rounded, especially at the throat to ensure the gas flow could pass through smoothly. Then, the entire surface inside the nozzle was polished as shown in the appendix (f).

4.3 Static thrust facilities

The static thrust test facilities are located in UMP Alternative Energy laboratory. The test or bunker was equipped with holder to hold the rocket motor and a 1000kg load cell for measuring the thrust during the test. Both equipments were installed in the bunker as shown in figure 4.2. It was made from 30mm thickness galvanic ion and prospect which has dimensions of 1.0m length and 0.6m width. The load cell was attached to a 60mm mild steel block. The purpose of employing this heavy and thick plate is to eliminate undesired deflections under high thrust and small detonations. It was found that, the test place could withstand repeated small detonations from the fail tests.

Figure 4. 2: Solid Rocket Motor attached at Bunker

The motor is held horizontally and attached to the load cell as shown in Figure 4.3 to allow small horizontal motion encountered by the motor during testing. However, this method produce frictions, which contribute to an error in the thrust measurement. The frictionless holder was used to hold the rocket motor.

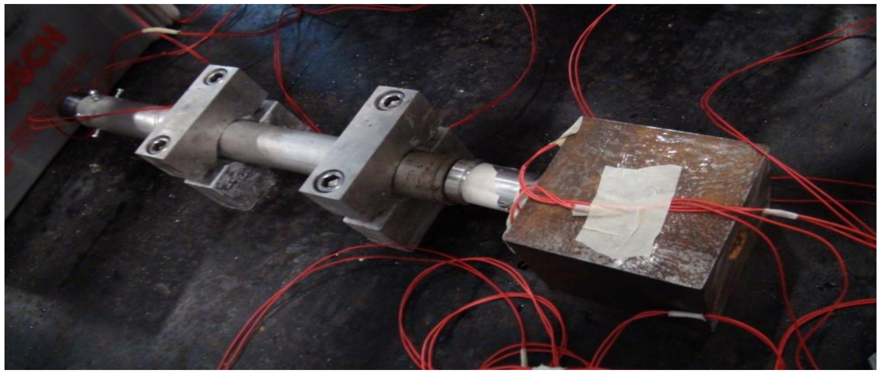


Figure 4. 3: SRM attached load cell

The attachment between load cell and motor was made from a modified aluminum hollow bar as shown figure 4.4. When the rocket motor is in operation, the thrust produced was measured by the load cell by the principal of a small deformation on the surface of the cell. The small change in dimension was measured by a strain gauge, which installed on the surface of the load cell. Finally, the force produce was converted into electrical signals in the form of voltage and measured by a data logger system. The output data were collected and saved in a database to be measured and analyzed.

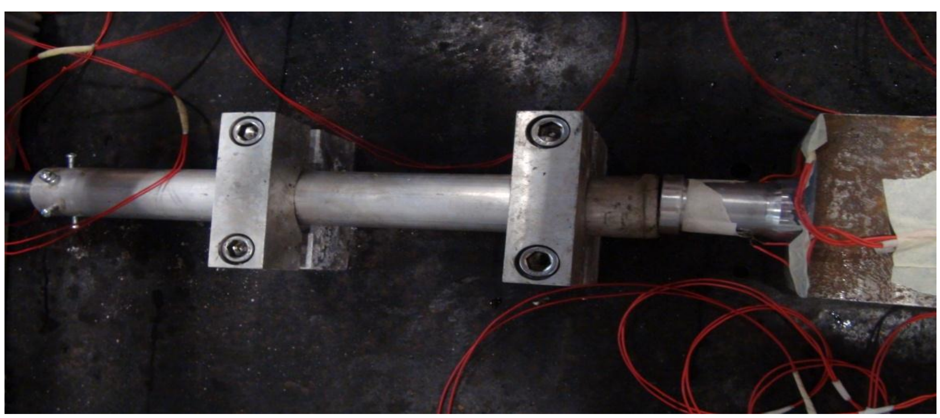


Figure 4.4 : Attachement load cell with data logger

4.4 Testing Procedure

The testing was conducted in an energy alternative laboratory. Before loading the propellant into the casing, it is cut to obtain the correct length and weight in order to ensure the reproducible results. Then, the grain was inspected and cleaned manually to remove the residue. The gap between the nozzle and the casing was sealed using white tape that was wrapped around the nozzle. Then, it was secured to the casing using six screws. Then an igniter is assembled to the rocket motor as shown in Figure 4.5. The connectivity and resistance of the igniter were checked before testing.

After all the installation and inspections are carried out, the rocket motor was placed onto the frictionless holder and all the apparatus is installed including the load cell. During firing, all data are collected and transfer to the computer to be analyzed. A typical result from static thrust testing will show in Figure 4.5. A mini video camera was used to record the firing as shown in Figure 4.5. For the safety reason, all observers must wear an ear plug and stay in a safe place. Normally, it took about 10 to 15 minutes to remove all the smoke after the test. For the safety reason, the observer cannot come closer till gases all out.

Figure 4.5: Static Thrust Testing

From the observations while conductinng the test, the temperature and propellant effect at the nozzle is shown in Figure 4.5. Not all of the testing conducted is successful. The defect and failure on the casing and the bulkhead was shown in Figure 4.6 and 4.7. The failure occurs at casing during testing using aluminum casing.

Figure 4.6: Nozzle After Testing

4.5 Failure During Static Thrust Testing

Figure 4.2: Casing Melting/Failure

Figure 4.3: Bulkhead failure

Failure occurs during static thrust testing because a trial and error concept with different material of casing and bulkhead is used. The failure may be because the thickness of casing and the bulkhead. From this test, we used casing from aluminum with the thickness of 2mm. From references the melting point of aluminum range between 650 degree Celsius till 685 degree Celsius, so the chamber temperature of this experiment it achieves up to 1800 degree Celsius. The suitable casing material for this testing is galvanic ion, low carbon steel and stainless steel.

CHAPTER 5

RESULT AND DISCUSSION

5.1 Introduction

In this chapter, the result of static thrust testing and discussion for each result.is shown. Only two out of three designs of nozzles were tested due to the safety precaution. In this study, the nozzle with throat diameter of 13mm and 15mm are tested to measure the thrust. Before the static thrust measurement test was done, the test is carry on without load cell first. This step is crucial for observing the performance of nozzle whether it will fails or not. The raw data and calibration is shown in Appendix D.

5.1 Testing without load cell

Table5.1: Testing without load cell

PROPELLANT (% /total length)	TIME (sec)	REMARKS
30 %	1.5 – 2 sec	Successful
50 %	3 sec	Successful
70 %	4 sec	Successful
100%	6 sec	Successful

i. The test for 13 mm and 15 mm throat diameter nozzle was conducted and the results are shown in table 5.1 Each nozzle was tested for four times to determine whether it will fails or not.

ii. The percentage of propellant is the length of the propeller per length of casing times hundred.

iii. The test for nozzle with 20mm throat diameter is not conducted because it present a high risk if it fails an explosion might occur during testing.

iv. Generally, this test was conducted in order to determine a suitable material for fabricating rocket motors. Rocket motor was one of the devices that enable the high pressure fluid flow through on it. Besides, it also has to endure high temperature. The designed nozzle can be used as combustion nozzle and steam nozzle

5.2 Testing with a load cell

Table 5.2: Testing with load cell and different material of the casing

CASING MATERIAL	DATE	TIME	THRUST	REMARKS
Aluminum	12/5/2012	6 sec	2690N	Successful
Aluminum	12/5/2012	6 sec	2692N	Successful
Aluminum	15/5/2012	6 sec	-	Failed
Galvanic ion	29/3/2012	6 sec	2523N	Successful
Galvanic ion	29/3/2012	6 sec	2543N	Successful
Galvanic ion	16/5/2012	6 sec	2532N	Successful
Aluminum	16/5/2012	6 sec	-	Failed
Aluminum	12/5/2012	6 sec	-	Failed

i. The test was conducted for 13mm diameter throat nozzle with a load cell and the result is shown in table 5.2.

ii. The nozzles with throat 20mm did not undergo any test because it has high risk if fails or explode during testing.

iii. This test was conducted eight times with different casing material and bulkhead. The casing is made from aluminum, it can last up to three times of testing or rocket launch. After that the casing and bulkhead will explode or fail. This happened due to the temperature of the combustion exceeding the melting point of aluminium. Besides that, the yield strength of aluminum was lower than galvanic ion. In this study, the failure usually occurs at casing and the bulkhead and the nozzle perfectly not failure.

iv. Table 5.2 shows the thrust force using aluminum nozzle is higher than thrust from galvanic ion nozzle. This is because the galvanic ion is heavier than aluminum and friction from galvanic ion is greater than aluminum. Aluminum has a smoother surface

5.3 Thrust from nozzle with different size of throat

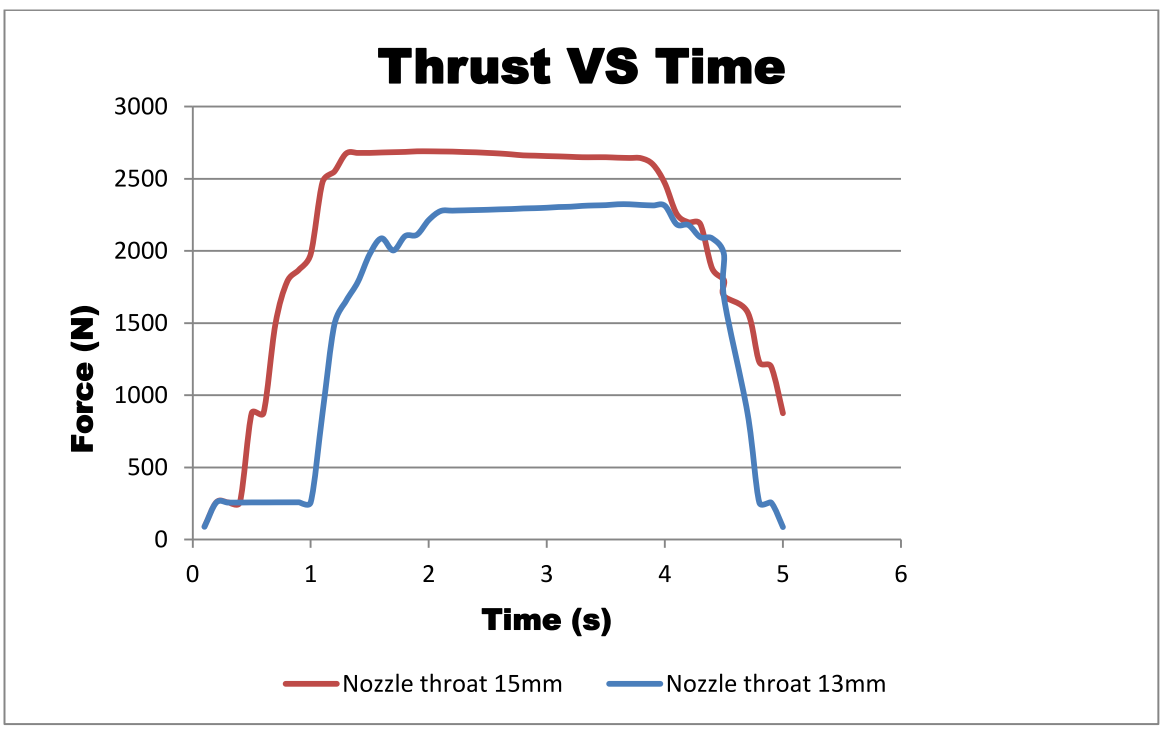


Figure 5.1: Graph Thrust versus Time

Figure 5.1 shows the result of static test for a rocket motor using load cell. For 15 mm diameter throat nozzle, it produces maximum or peak thrust of 2690 N at 1.9 sec and this rocket motor produce thrust for 0.7 sec till 4.9 sec, in this range rocket stay flight upward. Besides, for nozzle throat of 13 mm diameter the maximum thrust produce at 3.8 sec is 2318 N. The thrust is directly proportional to throat area. This means that if the throat size is increase the thrust will increase if the chamber pressure is maintained. Equation 2.15 is used to calculate the thrust.

5.4 Chamber temperature

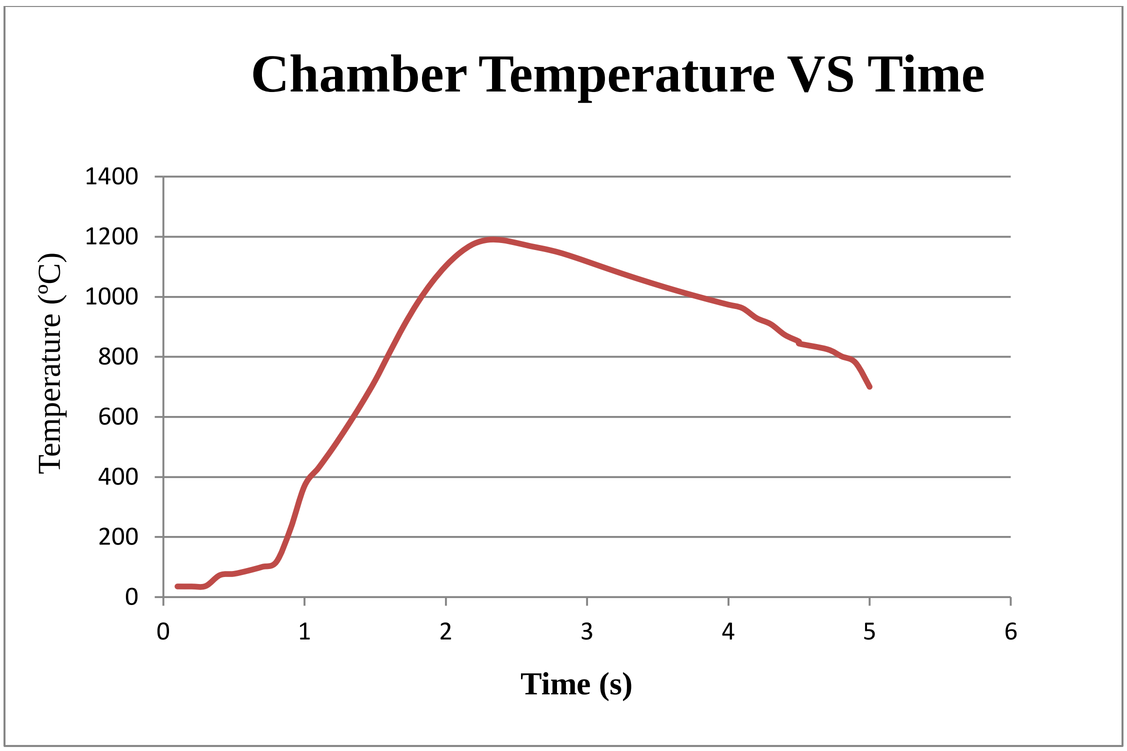


Figure 5.2: Graph Temperature versus Time

Figure 5.2 shows the temperature inside the chamber during static test for a rocket motor with time. This graph shown maximum temperature is 1189 Celsius at 2.3sec. The thrust and chamber pressure generates by rocket motor is proportional to the burning area at any particular time. The temperature sensitivity of burning rate is therefore directly proportional to the temperature sensitivity of thrust. The temperature sensitivity of burning rate is defined as the change in burning rate for a given change in propellant initial temperature at constant pressure which related to thrust. The increase in the propellant burning rate due to axial gas flow inside the combustion chamber is known as erosive burning. It occurs when high velocity combustion gas flow parallel to the propellant burning surface. The heat transfer at the propellant surface is increase due to increase flow of velocity.

Table 5.3: Rocket performance in two different throat diameters

Characteristic	Theoretical (13mm)	Experimental (13mm)	Theoretical (15mm)	Experimrntal (15mm)
Thrust Ratio	2.4770		2.4770	
Pressure Throat	4264 kPa		4010 kPa	
Exit Diameter	3.5 cm	3.5 cm	3.5cm	3.5cm
Exit Area	9.6211	9.6211	9.6211	9.6211
Nozzle Area Ratio	9.1052	9.1052	9.1052	9.1052
Diameter and Area Throat	1.3 cm / 1.32	1.5 cm/1.7671	1.5 cm/1.7671	1.5 cm/1.7671
Length of Throat	0.0675 cm	0.08 cm	0.9 cm	0.9 cm
Length of Diverging Nozzle	4.2 cm	4.2 cm	4.2 cm	4.2 cm
Thrust	2787 N	2690 N	2367 N	2318 N
Exit Velocity	1269.4 m/s		1056.3 m/s	

The rocket motor performance is analytically predicted using known geometry and propellant characteristics. All the equations used in this analysis is assume the be one-dimensional, steady, isentropic flow of a calorically perfect gas through the nozzles. The example of the calculations can be referred in chapter 3.

By comparing both results as shown in Table 5.3, it is known that the rocket motor with throat diameter of 13mm will have higher specific impulse compared to nozzle with 15mm.throat However, the burning rate is long and rise rapidly. By referring to both chamber pressures, some conclusion can be made:

1. The solid rocket motor was properly manufactured. There is no leak and failures at the bulkhead and casing. In order to get the best of solid rocket motors a lot of effort and skills were needed for operating the machine.

2. At the biggest size of the throat, the propellant deflagrates faster and produces short burning time.

3. Higher chamber pressure produces higher specific impulse. By increasing the throat diameter from 13mm to 15mm and retaining other parameters, the chamber pressure for 15mm throat has been halve for about 6% compared to the chamber pressure for the 13mm throat. Furthermore, by having smaller throat size the burning duration became longer.

4. Thrust from experiment is smaller compared to theoretical thrust because the value depends on the material of a rocket motor and weight of material used which can reduce the rocket thrust.

5. In addition, frictional force also affects the thrust produced by the rocket, so we must use a holder with less friction.

CHAPTER 6

CONCLUSION AND RECOMMENDATIONS

6.1 Conclusion

Theoretical analysis of a small solid propellant rocket motor was conducted, with the performance results presented in the form of two limiting models. The actual findings are expected to lie within this theoretical range, and experimental measurements found this to be the case. This result shows the importance of considering the effect of phase flow for small solid propellant rocket motors.

Theoretical analysis suggested that the propellant performance should remain essentially constant over a fairly wide range of oxidizer/fuel ratios, however, other factors such as burn rate and castability will reduced but still in acceptable range. The best overall ratios are 65/35. The maximum thrust produce was 2690 N which is 3% lower than theoretical design and analysis by the nozzle throat 15 mm. And maximum thrust produced by the nozzle with throat diameter of 13 mm is 2318 N which 2.5% lower than theoretical design analysis.

Significant improvement of the motor's performance does not appear to be achieved by variation of the nozzle design. The conical nozzle emerged as a highly satisfactory design when the manufacturing ability is considered. A thrust versus time curve of the rocket motor's performance was obtained, providing a base upon which to determine the expected acceleration, velocity, and altitude of the rocket in actual flight.

6.2 Recommendation

From the this study, there were several suggestions that needed to be considered to improve and advancing the design, fabricating and testing of rocket motor in the future,. The recommendation are as follows:

1. Stainless steel should be used in the design and fabrication of a new rocket motor because it has high yield strength and can prevent corrosion.
2. The load cell used in static thrust testing should be more sensitive and accurate and calibration also needed to be done to avoid reading errors during data collection.
3. In this study we have many potential errors, such as proportions of the components in the propellant could be skewed, setting the nozzle , calibration errors and uneven distribution of propellant in the rocket engine. So prevention steps must be taken early before do the tested and fabricated.
4. In fabrication process, it is crucial to ensure there is no leak in the attachment for bulkhead and nozzle. This can bbe done by carefully fabricating and inspecting the process.
5. Use nitrocellulose as the binder since it can also act as a fuel and would give a higher performance.
6. DAQ system needs refinement to ensure the system can calibrate the exact data from static test.
7. Valid control rooms for static thrust testing to prevent injuries.

REFFERENCES

1. Rizalman Mamat, "Ciri-Ciri Propelan Berasaskan Kalium," Faculty of Mechanical Engineering, Universiti Teknologi Malaysia, Johor Bharu, 2002.

2. Martin J.L.Turner, *Rocket and Spacecraft Propulsion. Principle, Practise and New Developments (Eigth Edition)*: Springer, 2010.

3. Numerical simulation of the dynamic thermostructural response of a composite rocket nozzle throat. E.V. Morozov, J.F.P. Pitot de la Beaujardiere School of Aerospace, Civil & Mechanical Engineering, University of New South Wales, Department of Mechanical Engineering, University of KwaZulu-Natal, Howard College Campus, Durban, South Africa.2009.

4. Coupled simulation of heat transfer and temperature of the composite rocket nozzle wall. Zhang Xiaoying, Aerospace Science and Tecnology.2010.

5. Numerical study of the start-up process in an optimized rocket nozzle. José Antonio Moríñigo and José Juan Salvá. Dpto. Programas Espaciales, Instituto Nacional de Técnica Aeroespacial, Spain.Dpto. Motopropulsión y Termofluidodinámica, ETSI Aeronáuticos, Universidad Politécnica de Madrid, Spain.2007.

6. 3-D grain burnback analysis of solid propellant rocket motors: Part 2 – modeling and Simulations. G. Püskülcü and A. Ulas. Department of Mechanical Engineering, Middle East Technical University, 06531 Ankara, Turkey.2008.

7. Spatial linear analysis of the flow in a solid rocket motor with burning walls. L. Massa. Department of Mechanical & Aerospace Engineering, The University of Texas.2008.

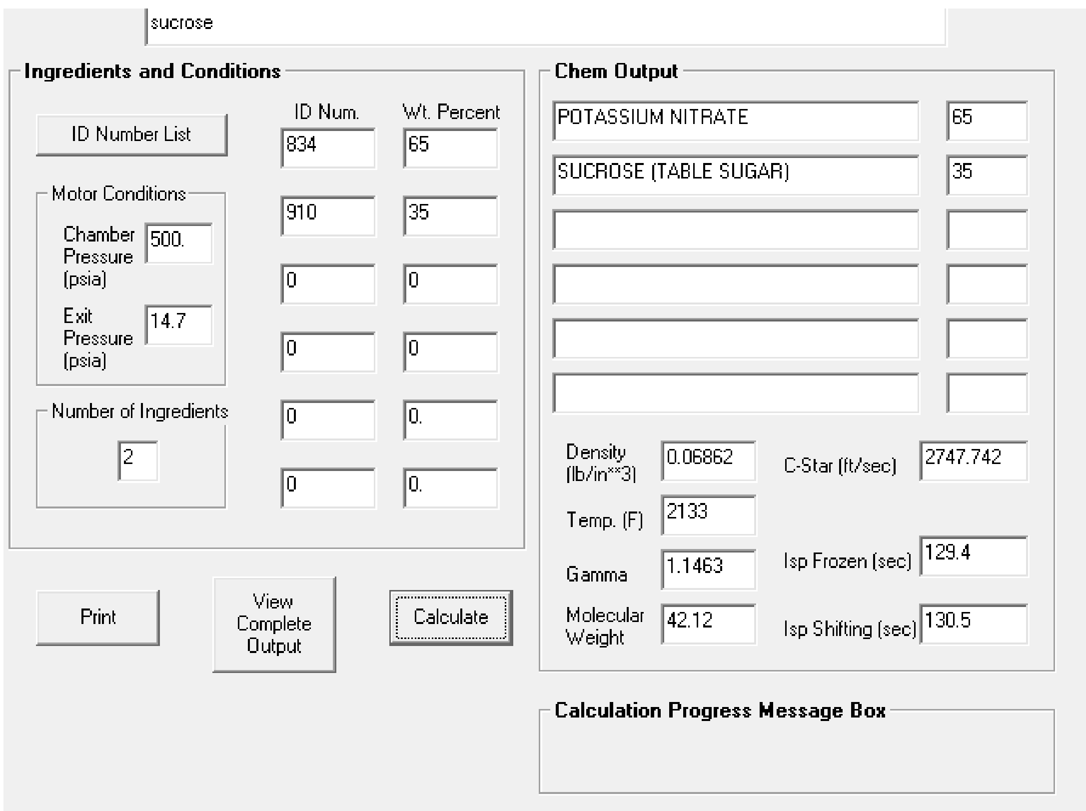
8. Forty Years Of Model Rocketry A Safety Report by G.Harry Stine.National of Rocketry.1997.

9. Easy PVC Rockets by Jason Smiley.2005

APPENDIX A

Guide for Using CHEM Program

Chemical Equilibrium of Application, CHEM is the program for chemical equilibrium calculation developed by Dr. S. Gordon, Dr. B. J. McBride, and co-workers in NASA Glenn Research Center for more than 40 years ago. It does regression and estimation of thermodynamic data, *etc*. The CEC71 program can be downloaded at http://www.grc.nasa.gov/WWW/CEAWeb/ceaguiDownload-win.htm .



FILE MENU

- **ID Number List** – Number for chemical composition.Each che,ical composition has different ID number
- **Chamber pressure**- Assumption for input at chamber pressure.
- **Exit pressure**- Assumption or from theory of calculation/ simulation.
- **Number of ingredients** - Allows used no of chemical composition.
- **Chem Output** – Name of ingredients based of list ID number.

A selection of several units is available for initial pressure P_{in}, namely bar (default); atm; psia; or mmHg. The selected unit applies to all the values listed. The estimated or assigned combustion temperature $\mathbf{T_c}$ may be selected as either Kelvin (default), Rankine, Celsius, or Fahrenheit. The default estimated value is 3800 K.

APPENDIX B

Output of CHEM for propellant

```
PROPELLANT THERMOCHEMISTRY PROGRAM
 CHEM - Version 2
 COPYRIGHT 1999 - CP TECHNOLOGIES

  Title Case        Case   1 of   1

                              DH   DENS COMPOSITION

 AMMONIUM PERCHLORATE (AP)        -602   .07040    1CL  4H   1N   4O
 R45                              40   .03250  661C 999H   1N   9O
 ALUMINUM OXIDE                 -4000   .06700    2AL  3O
 IPDI                            -501   .03840   12C  18H   2N   2O

 INGREDIENT WEIGHTS (IN ORDER) AND TOTAL WEIGHT        (LAST ITEM IN LIST)

   40.5000    16.0000    21.0000     3.5000    81.0000
0INGREDIENT VOLUME RATIOS
 AMMONIUM PERCHLORATE (AP)      39.077%   R45                        33.441%
 ALUMINUM OXIDE                21.291%   IPDI                        6.191%

 THE PROPELLENT DENSITY IS  .05502 LB/CU-IN OR  1.5230 GM/CC

 NUMBER OF GRAM ATOMS OF EACH ELEMENT PRESENT IN INGREDIENTS

   3.417829 H     1.350593 C      .377937 N     2.043955 O
    .411926 AL     .344690 CL
```

CHAMBER RESULTS FOLLOW:

T(K)	T(F)	P(ATM)	P(PSI)	ENTHALPY	ENTROPY	CP/CV	GAS	RT/V	MOL WT
1270.	1826.	34.01	500.00	-109.49	160.64	1.2253	3.108	10.944	26.06

DAMPED AND UNDAMPED SPEED OF SOUND= 1863.315 AND 2192.348 FT/SEC

1.11243 H2	1.07326 CO	.34467 HCl	.20595 Al2O3&
.18851 N2	.16241 H2O	.13018 CH4	.09520 CO2
.05173 C&	.00079 NH3	.00011 CNH	.00003 C2H4
2.09E-05 C2H6	1.28E-05 CH2O		

EXHAUST RESULTS FOLLOW:

T(K)	T(F)	P(ATM)	P(PSI)	ENTHALPY	ENTROPY	CP/CV	GAS	RT/V	MOL WT
929.	1214.	1.00	14.70	-131.98	160.64	1.2109	2.824	.354	28.68

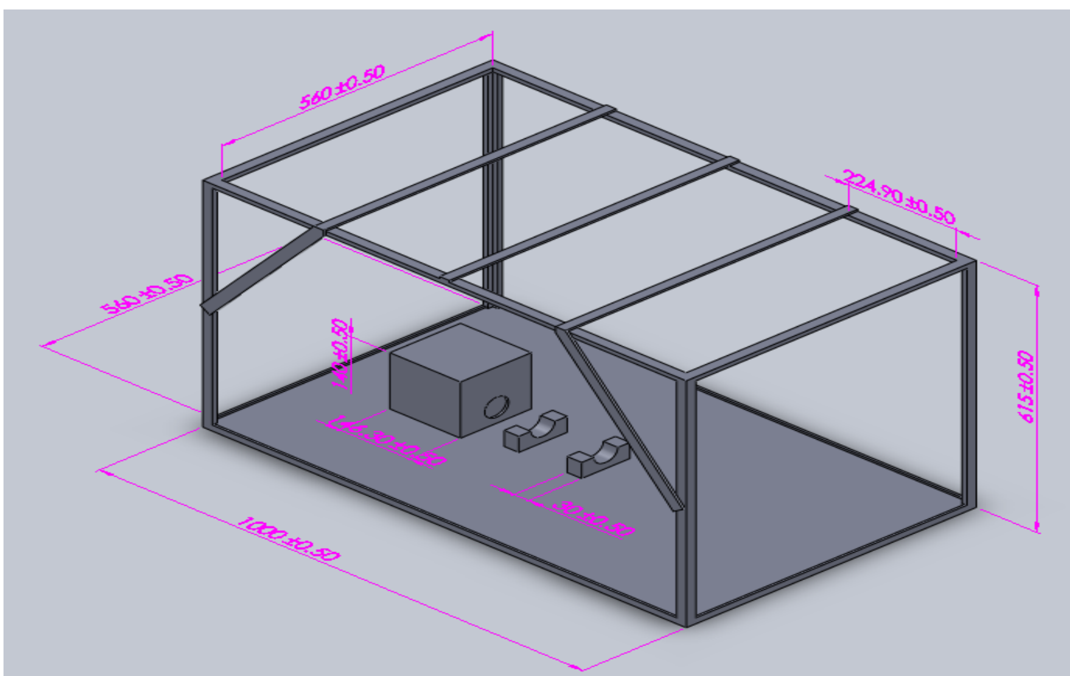
DAMPED AND UNDAMPED SPEED OF SOUND= 1519.076 AND 1864.563 FT/SEC

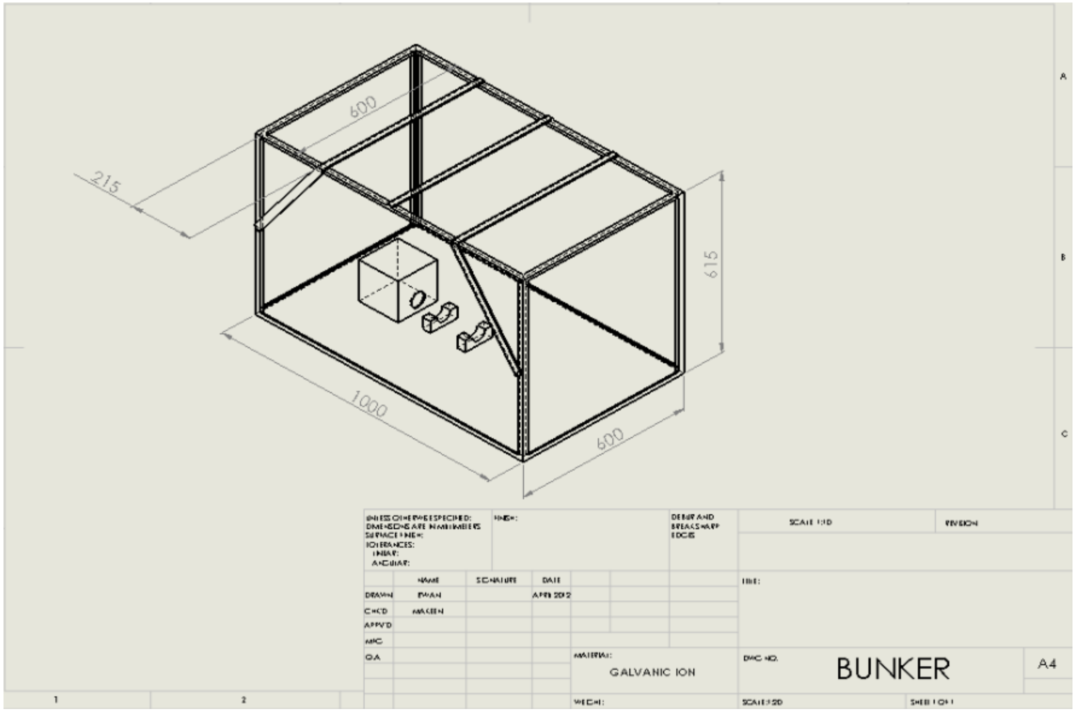
APPENDIX C

DRAWING

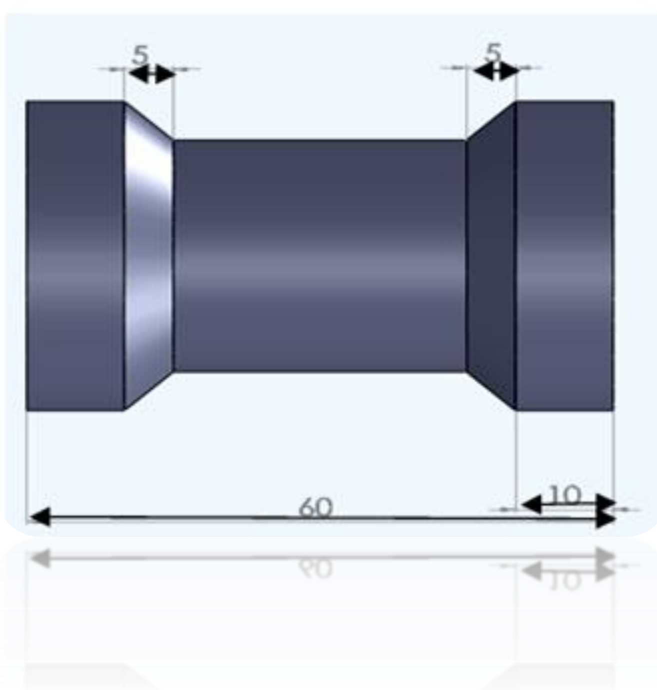
Bunker Design

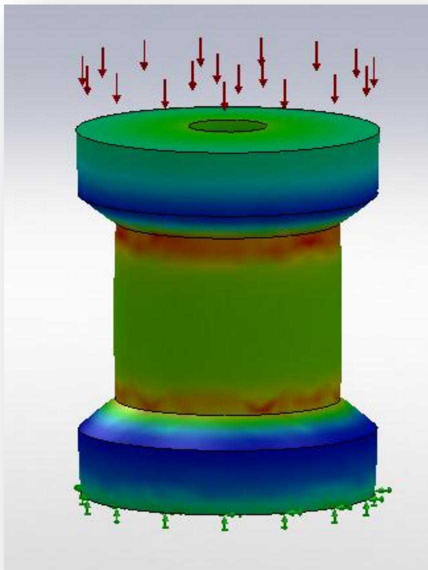
This type of bunker built for control of dangerous activities which is test of rocket engine. The bunker use as Perspex protection from outside people when the testing of rocket engine. Bunker will deflect wave from inside explosion to prevent series injuries to people. Basic plan is to provide a structure that very strong in physical compression. The dimension of bunker is 100 cm × 60 cm. The wall thickness is 1 inch with Perspex and steel net.

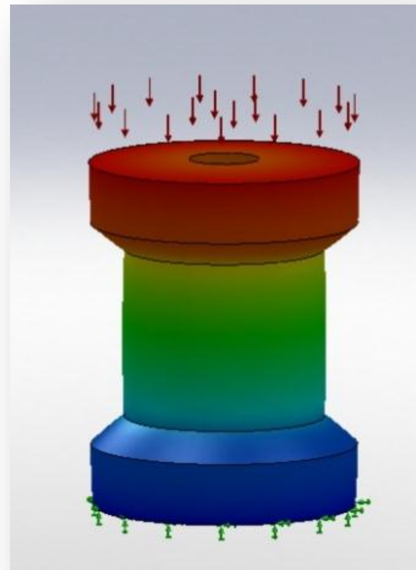




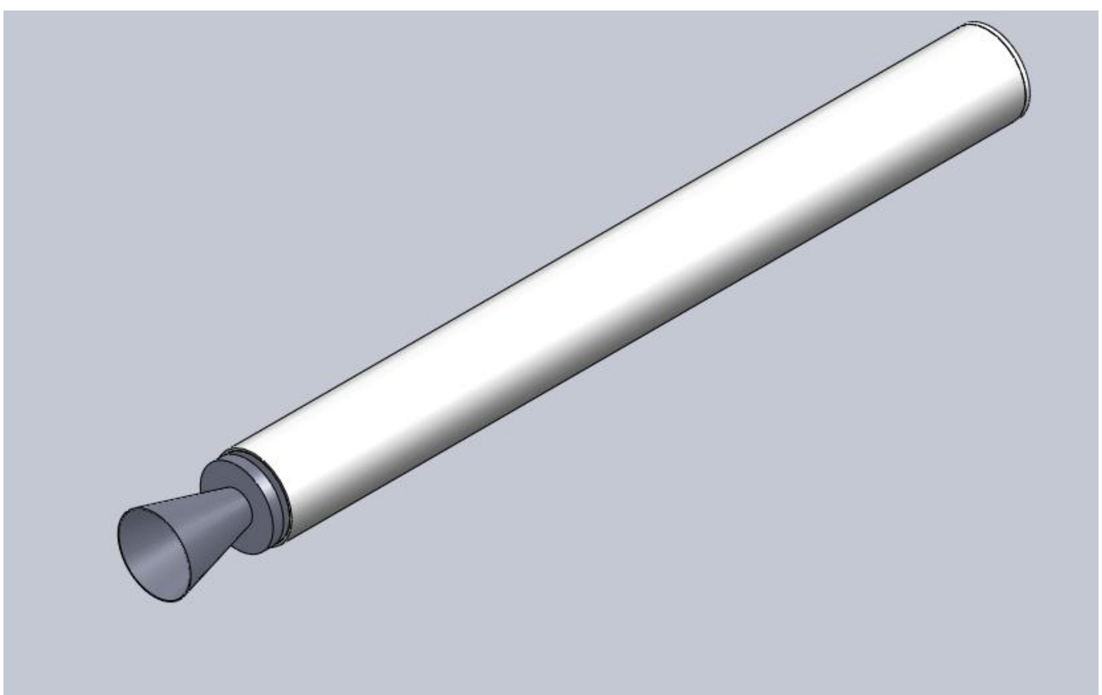
Design load cell







Solid Rocket Motor



APPENDIX D

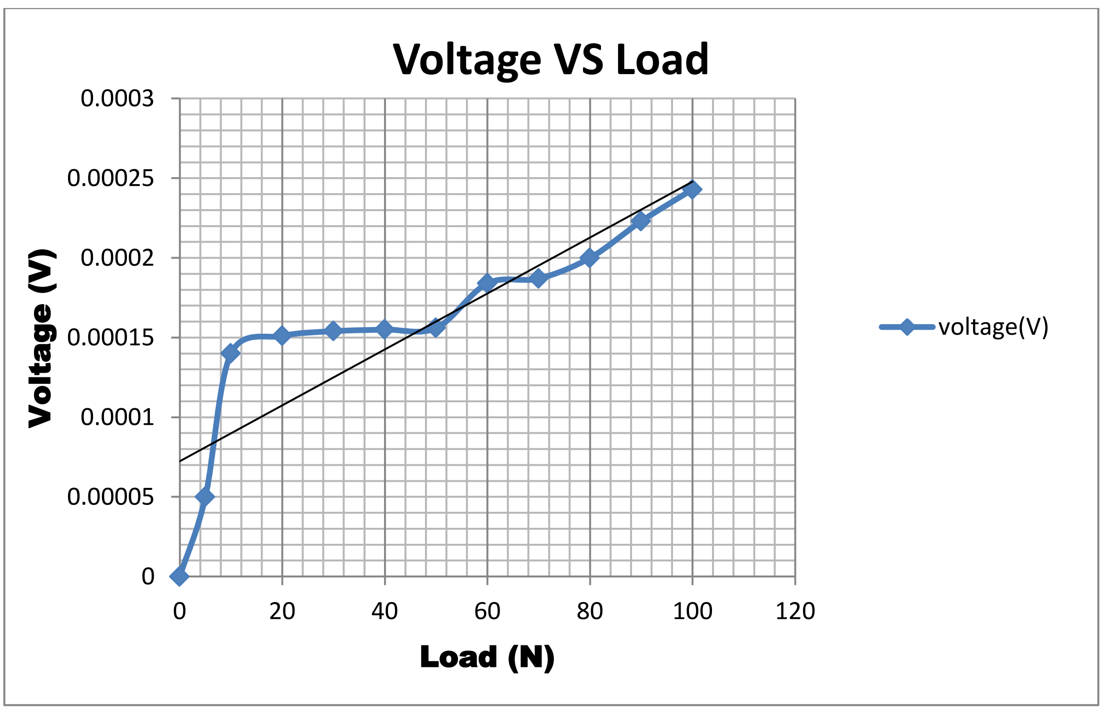
RAW DATA

0.1	0.005112	89	35.51
0.2	0.005112	258	35.45
0.3	0.005112	258	37
0.4	0.005112	258	73.09
0.5	0.005113	876	77.57
0.6	0.005113	876	88.02
0.7	0.005116	1488	100.71
0.8	0.005118	1786	117.21
0.9	0.005119	1869	226.92
1	0.00512	1978	370.62
1.1	0.005136	2476	431.08
1.2	0.005196	2549	496.29
1.3	0.005208	2675	566.66
1.4	0.005215	2679	641.21
1.5	0.005216	2679	721.46
1.6	0.005222	2682	812.49
1.7	0.005226	2684	901.34
1.8	0.005229	2686	980.22
1.9	0.005237	2690	1047.21
2	0.005237	2690	1102.88
2.1	0.005235	2689	1146.62
2.2	0.005234	2688	1176.84
2.3	0.005228	2685	1189.74
2.4	0.005224	2683	1188.45
2.5	0.005216	2679	1179.44
2.6	0.005209	2675	1168.61
2.7	0.005196	2669	1159.25
2.8	0.005184	2662	1147.85
2.9	0.005178	2660	1133.28
3	0.005173	2657	1117.25
3.1	0.005169	2655	1100.91
3.2	0.005164	2652	1084.76
3.3	0.005153	2649	1069.07
3.4	0.005157	2649	1053.93
3.5	0.005158	2649	1039.27
3.6	0.005152	2646	1025.16
3.7	0.005148	2644	1011.61
3.8	0.005145	2642	998.57
3.9	0.005141	2598	985.99
4	0.005135	2468	973.84
4.1	0.005139	2256	962.09
4.2	0.005137	2198	929.11
4.3	0.005132	2186	908.96
4.4	0.005129	1879	873.22
4.5	0.005126	1798	850.95
4.5	0.005115	1689	844.21
4.7	0.005112	1576	825.28
4.8	0.005112	1230	802.06
4.9	0.005112	1198	780.69
5	0.005112	875	700.2

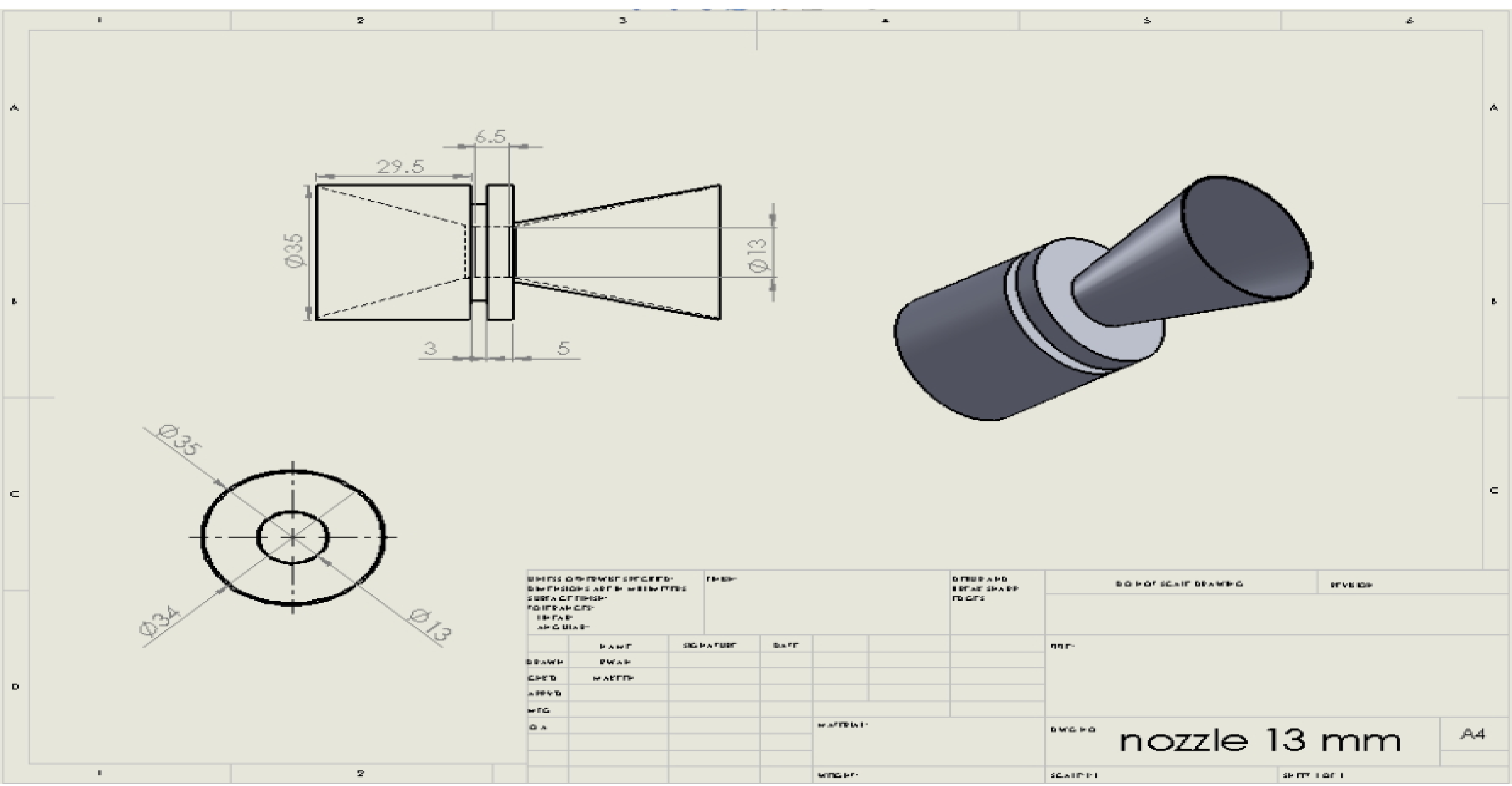
APPENDIX E

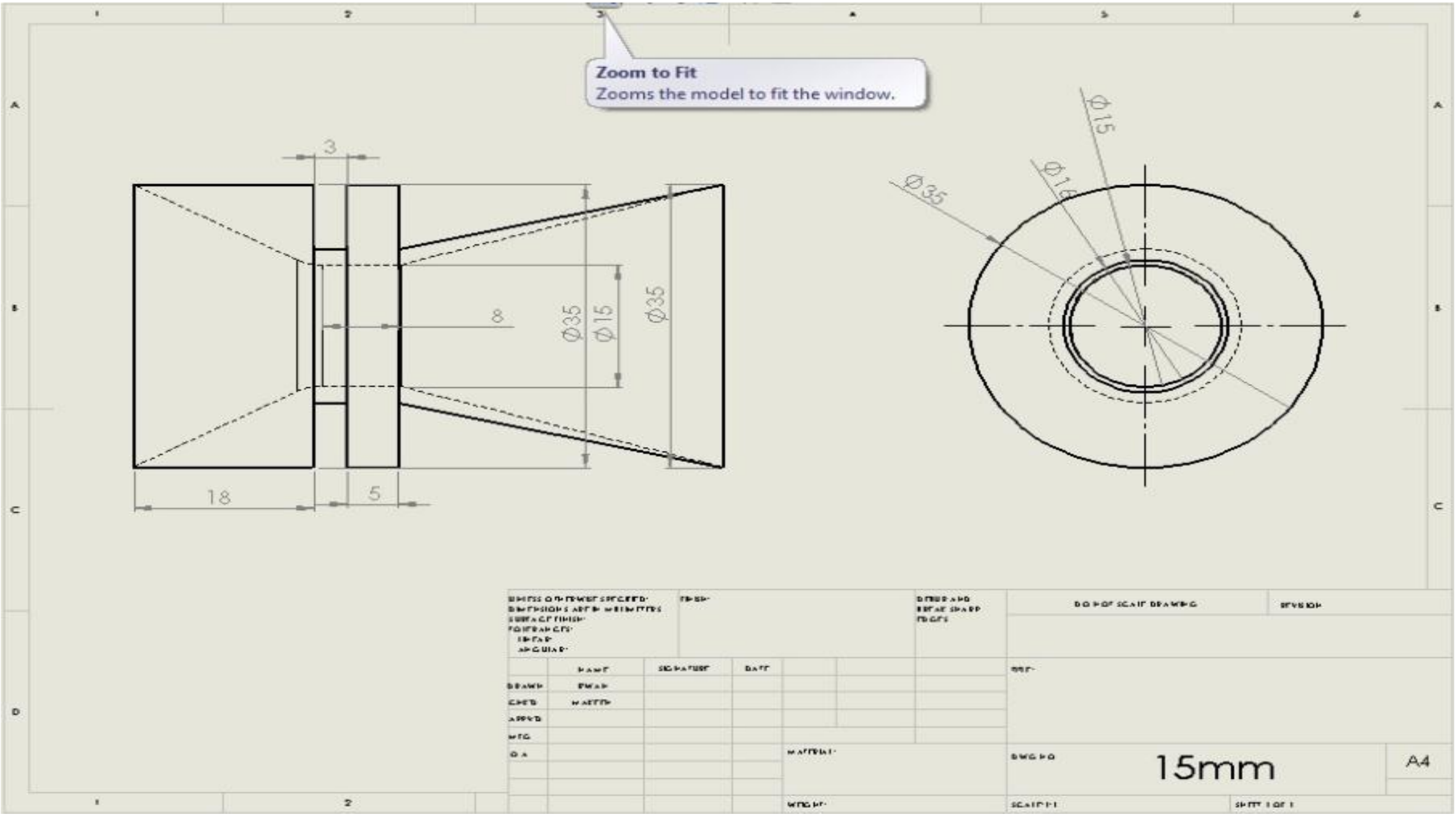
CALIBRATION RESULT

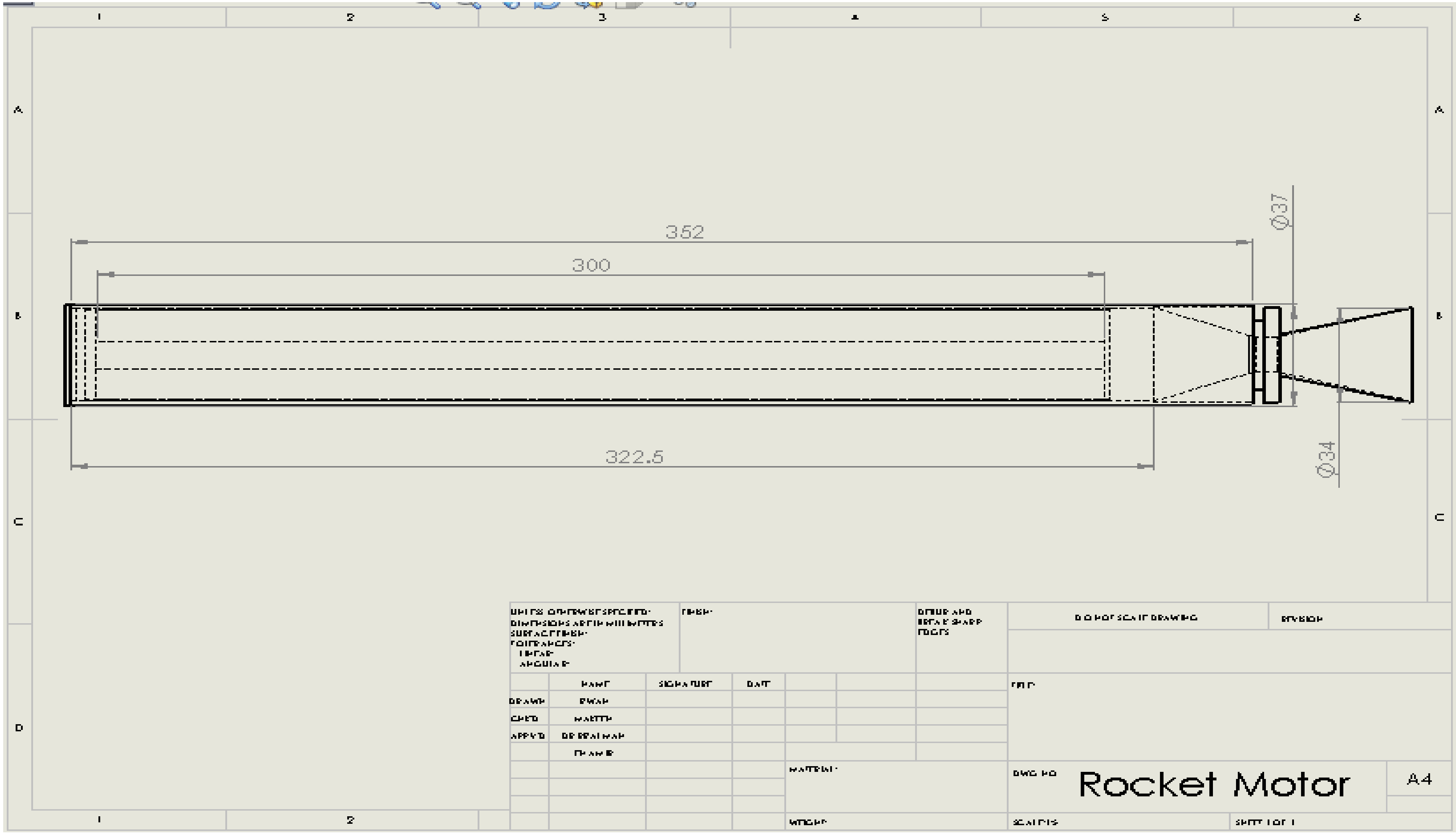
load(N)	voltage(V)
0	0
5	0.00005
10	0.00014
20	0.000151
30	0.000154
40	0.000155
50	0.000156
60	0.000184
70	0.000187
80	0.0002
90	0.000223
100	0.000243



APPENDIX F




Zoom to Fit
Zooms the model to fit the window.
3
18
5
8
Ø35
Ø15
Ø35
Ø15
Ø16
Ø35
DO NOT SCALE DRAWING
15mm
A4



352
300
322.5
Ø37
Ø34
Rocket Motor
A4