DESIGN, FABRICATION AND TESTING OF A SOLID PROPELLANT ROCKET MOTOR

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DESIGN , FABRICATION AND TESTING OF A SOLID PROPELLANT ROCKET MOTOR

by

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MENGKAJI DAN MEREKA BENTUK SEBUAH ROKET MOTOR JENIS BAHAN API PEPEJAL

oleh

ANG KIANG LONG

Tesis yang diserahkan untuk
memenuhi keperluan bagi
Ijazah Sarjana Sains

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Plans fail for lack of counsel, but with many advisers they succeed.

Proverb 15:22 (New International Version)

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LIST OF ABBREVIATIONS

AP	Ammonium perchlorate
Al	Aluminum
CAD	Computer-aided design
CDB	Cast double-base
CFD	Computational fluid dynamic
DB	Double-base
DIY	Do-It-Yourself
EDB	Extruded double base
EMCDM	Elastomeric-modified cast double-base
FEM	Finite element method
MEASAT	Malaysia East Asia Satellite
MSDS	Material safety data sheet
NC	Nitrocellulose
NCBI	National Center for Biotechnology Information
NG	Nitroglycerine
NDT	Non-Destructive Test
PVC	Polyvinyl Chloride
SEM	Scanning electron microscopy
UHF	Ultra High Frequency
USM	Universiti Sains Malaysia
2D	2-Dimensional
3D	3-Dimensional

LIST OF SYMBOLS

A	Cross sectional area
A_b or $A_{surface}$	Propellant grain burning area
A_t	Nozzle throat area
A_2	Cross-sectional area at exhaust area
a	Speed of sound (in calculation of Mach number)
a	Acceleration (in calculation of force)
α	Nozzle half angle
C	Carbon (chemical element)
C_F	Thrust coefficient
Cl	Chlorine (chemical element)
c	Effective exhaust velocity
c^*	Characteristic exhaust velocity
c_p	specific heat ratio for constant pressure
c_v	specific heat ratio at constant volume,
D	Diameter
D_{grain}	Grain diameter
D_t	Diameter for the throat area
D_2	Exhaust area
d	Thickness of a simple cylinder
E	Young’s modulus
ε	Nozzle expansion ratio
F	Thrust produced by rocket
Fe	Ferric (chemical element)
g_0	Standard acceleration of gravity
γ	Ratio of specific heat at constant pressure c_p and specific heat at constant volume c_v
H	Hydrogen (chemical element)
h	Enthalpy
h_0	Stagnation enthalpy per unit mass
I_s	Specific impulse
I_t	Total impulse
J	Mechanical equivalent of heat
K	Potassium (chemical element)
L	length
L_c	Length of the converging section
L_D	Length of the diverging section
L_{grain}	Grain
λ	Theoretical collection factor
M	Mach number
M_w	Molecular weight
m	Mass of the model rocket
m_{ic}	Mass of igniter charge

LIST OF SYMBOLS

m_p	Propellant mass
m_o	Initial mass
m_o	Mass of rocket before ignition
m_1	Mass of rocket at current time
$\dot{m}$	Mass flow rate
N	Nitrogen (chemical element)
O	Oxygen (chemical element)
P	Pressure
p	Chamber pressure
ρ_b	Propellant grain density
p_1	Chamber pressure
R	Gas constant
$R^{'}$	Universal gas constant
r	Burn rate
r	Radius of a simple cylinder
s	Distance
σ_θ	Hoop stress
σ_1	Longitudinal stress
T	Temperature
t or t_b	Burning time
υ	Poisson's ratio
V_{av}	Average velocity during the time period
V_F	Motor free volume
$V_{propellant}$	Volume of propellant burned per second
V_t	Specific volume at throat area
V_2	Specific volume at nozzle exhaust area
v_e	Effective exhaust velocity
v_t	Critical velocity
v	Velocity
$\dot{w}$	Weight to propellant burn per second
ξ	Propellant mass fraction
ξ_v	Velocity correction factor
ξ_d	Discharge correction factor
ξ_F	Thrust correction factor

MENGKAJI DAN MEREKA BENTUK

SEBUAH ROKET MOTOR JENIS BAHAN API PEPEJAL

ABSTRAK

Objectif tesis ini adalah mengkaji dan mereka bentuk sebuah roket motor jenis bahan api pepejal. Pertama, kajian dijalankan tentang rekabentuk asas yang wujud dalam dunia ini dan teori untuk menentukan rekabentuk roket motor. Kedua, penentuan bahan mentah untuk bahan api pepejal dan dimensi untuk roket motor dijalankan secara kaedah teori. Keputusan daripada simulasi perisian computer (SolidWork 2008) terhadap pengaliran udara in bahagian nozzle juga dilampirkan. Di samping itu, teknik-teknik fabrikasi dan ujian tanpa gagal (NDT) bahan api pepejal dimasukkan dalam laporan ini. Dua jenis rekabentuk bahan api pepejal telah disediakan dalam projek ini untuk penentuan prestasi roket melalui ujian statik. Ujian statik untuk kedua-dua rekabentuk bahan api pepejal telah dijalankan berulang kali supaya prestasi roket motor yang dianalisakan dengan lebih baik. Propellant yang berbentuk satu lubang silinder (single perforated) memberi jumlah impulse yang lebih tinggi daripada yang berbentuk tiga lubang silinder (tri-perforated). Propellant berbentuk tiga lubang silinder memberi daya tujah dua kali berganda berbanding dengan bahan api yang berbentuk satu lubang silinder.

DESIGN, FABRICATION AND TESTING OF A SOLID PROPELLANT ROCKET MOTOR

ABSTRACT

The objectives of this paper are to design, fabricate and test a solid propellant rocket motor. First, Literature studies on rocket motor fundamental which covered existing motor designs and theories for determining the properties of the rocket motor. Next, the numerical analysis in grain material and dimension for the rocket motor are included in this report. The simulation result by computer aids software (SolidWorks 2008) for the flow characteristics especially in the nozzle area is presented. The fabrication techniques and the Non-Destructive Test (NDT) on solid propellant also included in this thesis. Two solid propellant grain configurations were designed in this project to studies their effect on the rocket performance. Thrust profile of each grain configuration was determined through experiment measurements in static test. Repeatability test was done to determine the reliability of the experimental results. It was found that the single perforated grain profile given a higher total impulse compared to tri-perforated grain, although tri-perforated grain produced almost twice thrust value of single perforated grain in static test.

CHAPTER 1

INTRODUCTION

Rocket propulsion has become a comparatively mature field since last few decades, yet a new propellant has not been introduced for a rocket production application in the last 25 years. Research in most of the new propulsion systems are based upon the existing proven units in the chemical and electrical propulsion area [1].

Rocket that use chemical propellant to produce thrust can be further divided into two categories, which also known as solid propellant rocket and liquid propellant rocket [2]. The simplicity of a solid propellant rocket compared to liquid propellant rocket made it a popular choice for weapons and booster rockets that attached to huge orbiting rockets.

According to literature studies, rockets have been existed in human history close to a thousand years, they have been evolved into more sophisticated designs since the invention of gunpowder. However, rocket propulsion in the military missile application did not grow as fast as the guidance systems, warheads and reentry vehicles [3].The decision to obsolete the current well-tested rocket motor and replace with something completely new and untested is a no-go for military.

Inspired by the work of rocket pioneers like Tsiolkovsky in the years of 1925-1935, amateur rocket societies were founded in Unites States, USSR and Germany as testing ground for new technologies [4]. Since then, standards and procedures in launching a rocket were introduced for the reason of safety. Rockets are then defined in different classes according to the total impulse produced [5] [6].

In this report, literature review on existing solid rocket motor design including the method of analysis was presented. Properties of the selected material for the solid propellant and the rocket motor were carefully studied, in order to make sure the experiments conducted in a safe environment.

Computer aided software was used to simulate the internal flow in the chamber during combustion. Investigation on the rocket motor design through simulation also carried out in this project.

Two kind of circular perforated propellant grains were fabricated in this project. These solid propellant grains were fabricated in an innovative way rather than conventional method. Through experimental measurements, properties of the solid propellant were discovered. High performance instrument like X-ray was used for the examination on solid propellant as Non-destructive test (NDT).

One of the valuable data in this project was the thrust curve measurement for the solid propellant rocket motor through static test. The data obtained in the experiments were then be analyzed and discussed accordingly. Observation and detail discussions were enclosed within this report.

1.1 Problem statement

Space technology developments in Malaysia started in year of 1960s by the first communication satellite receiving station established in the nation [7], followed by the launching of MEASAT-1 and MEASAT-2 in the space in 1996 [8]. In Year 2007, we managed to send our first astronaut into the space [9].This shown that Malaysia is keen in developing the space industry.

However, all these missions were done in foreign countries due to lack of rocket launching facilities and expertise in this nation. The rockets used to send the payloads into the orbit were out-sourced and not locally made. Beside space exploration, military weapon used in this country [10] were originate from foreign countries especially the missile technologies. All these reveal the needs in rocket propulsion research within this nation.

There are a few national higher learning institutions that involved in rocket propulsion research and development. In year 2004, Universiti Teknologi Malaysia (UTM) has done a research on the potential of potassium nitrate and sucrose as solid fuel for rocket propulsion [11]. Similar research was done by Universiti Sains Malaysia (USM) in year 2008 who mainly emphasized experiments on sugar based propellant rocket [12]. However, this type of propellant is not used in professionally designed rockets due to their low performance compare to the typical used chemical like ammonium perchlorates. However, potassium nitrate is cheaper compare to ammonium perchlorates and potassium nitrate is yet been used as commercial rocket propellant. Thus, this creates the need to further investigate the use of potassium nitrate with other chemical to produce higher rocket performance, which will contribute to development in the nation rocket industry.

The invention of high speed computers and internet services have stimulates the amateur experimental rocketry activities around the globe. Information can be easily accessed with no restriction to design, fabricate and launch an experimental rocket. One of the main concerns in experimental rocketry is the new chemical propellant. Misuse of chemical compound and differences in experiment climate might cause accident. Thus, publication of a proven propellant not only can contribute to the knowledge in rocketry, but also to saves life of those who involved in experimental rocketry.

1.2 Aims and Objectives

Design a solid propellant rocket motor is the aim of this project. It can be further break down into two areas which consist of:

a) Solid propellant: To study the characteristic of the new formulated potassium nitrate/ferric oxide/epoxy solid composite propellant.
b) Rocket motor: To design a rocket motor that can operates with the new formulated solid composite propellant.

In order to achieve the aims of this project, here are the objectives that make this project a success.

i) Design the nozzle with nozzle theory and thermodynamic relations and simulate with computer aided software.
ii) Study the properties of the propellant especially the density and burn-rates and apply non-destructive test (NDT) in experiment.
iii) Study the rocket motor thrust profile through experimental result in static test and identifies the motor classification from the thrust profile.

1.3 Organization of the Thesis

This thesis has 8 chapters, including an introduction (Chapter 1) and conclusion (Chapter 8). Chapter 2 is the literature review on the solid propellant rocket motor. This chapter describes the basic component of a rocket motor and fundamental equations used in rocket designs. Chapter 3 describes the preliminary designs of the rocket motor. Selection of the grain configuration and material is included in this chapter. Nozzle theory and thermodynamic relations is used in nozzle design. Simulation on the preliminary design with computer aided software is discussed in Chapter 4. The fabrication techniques for propellant grain and igniter are discussed in Chapter 5 and Chapter 6 respectively. Chapter 7 discusses the experiment result in detail, which includes the non-destructive test (NDT) with X-ray.

CHAPTER 2

LITERATURE REVIEW ON SOLID PROPELLANT ROCKET MOTOR

2.1 Introduction to solid propellant rocket motor

The first generation of solid propellant rocket was invented by Chinese in A.D. 969 [13] which falls in period ruled by Song Dynasty. Song Dynasty [14] not only claimed to be the first government in the world to issue paper money, but also the first known use of gunpowder in the military warfare. Two military generals in Song Dynasty, Yue Yi Fang and Feng Ji Sheng created their revolutionary military weapon with arrows and tubes of gunpowder, in order to kill their enemies from greater distance in shorter time [15]. The Chinese words Huo Jian meaning 'Fire Arrow' or rocket has been pass down from generation to generation since then.

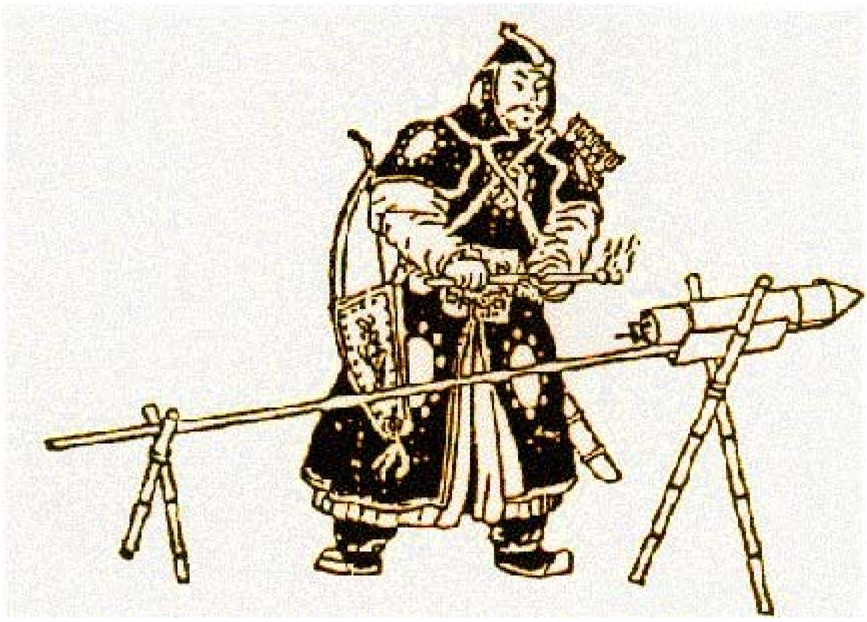

Figure 2.1 Chinese used the rocket in the battle. [16]

With the introduction and improvement in guns, rocket did not play an important role during World War time. The killing in the battle field during interwar years was mostly done with high speed bullets. It was the period where the amateur developed the experimental rocket. Tsiolkovsky [17] saw the possibilities to use rocket in space exploration instead as an artillery weapon at that time. He discovered the differences in term of dynamic force between rocket and gunnery. Rocket accelerates and able to carries payloads after firing while cannon ball decelerates and act as a projectile after firing. Tsiolkovsky introduced the rocket equation using Newton's third law, which defined in the following relationship [17].

$$F = \dot{m} v_e \tag{1}$$

where F is the thrust produced by the rocket is equal to opposite force in term of the mass flow rate $\dot{m}$ and the effective exhaust velocity v_e . He also introduced the calculation for the velocity of the vehicle v [17] in term of the effective exhaust velocity v_e , mass of rocket before ignition m_o and mass of rocket at current time m_1 . The equation is defined as below.

$$v = v_e \log_e \frac{m_0}{m_1} \tag{2}$$

Solid propellant rocketry continues make chapters in the history with discoveries in new material for greater performance. One of the examples was John Parsons created the first cast-able composite solid propellant in June of 1942 [18]. In Spain, research in ionospheric sounding rocket was limited by low specific impulse rocket propulsion in 1964 [19]. Double base propellant extruded in small diameter grains (<5cm) with simple grain configuration could not offer a solution to the Spain sounding rocket development. However, the requirement for this civilian use in sounding rockets was answered through technologies transfer from the UK, where the plastic composite propellant was introduced.

According to the literature finding [20], solid propellant rocket motors become the most popular propulsion source in United State (US) military missiles. Beside application as military missile, solid propellant rocket motor also served as boosters on the space shuttle. Since 1958, China involved in composite solid propellant rocket motor development for their satellites launch [21]. The reliability of solid propellant for space application was proven in a world wide space launches data recorded between 1st Jan 1980 until 31st July 2008 [22], where solid propulsion system makes 647 successful launches out of 662. Simple designs, high thrust levels and low cost in operations contributes to the 97.73% for successful launch.

With the enabling technologies such as improvement in grain and motor designs, case technologies, high performance components, thermo-chemical modeling and simulation development, high area ratio nozzle technologies and others factors listed in [23], solid rocket industry will continue to grow rapidly to meet the future propulsion needs in weapons and space boosters.

2.2 Components of a solid propellant rocket motor

A solid propellant rocket motor is designed to convert energy from the chemical propellant into supersonic exhaust gases through nozzle. As the result, thrust is created. Generally, solid propellant rocket motor consists of 5 main components (Figure 2.2). They are the case, nozzle, propellant grain, thermal insulation and ignition system.

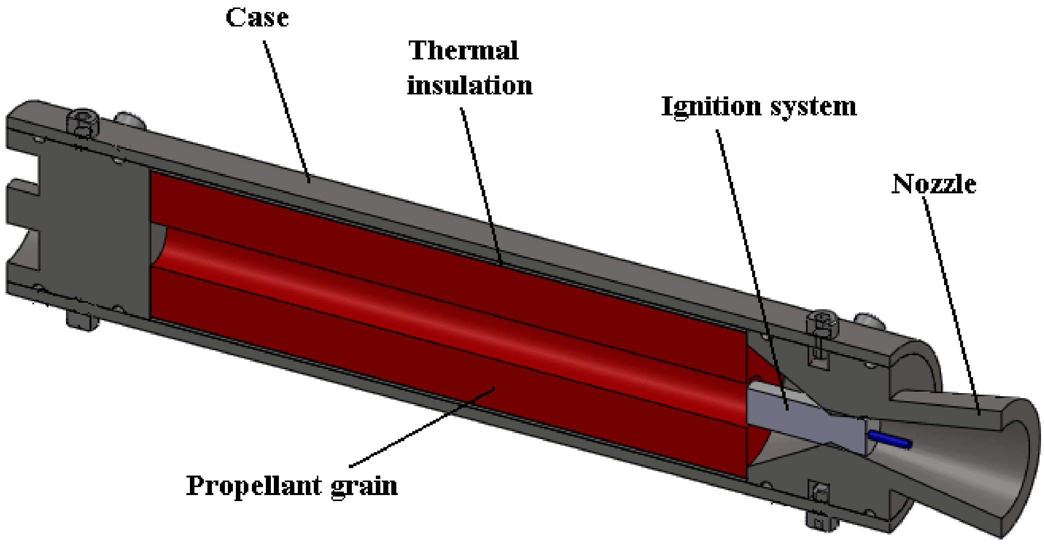

Figure 2.2 Typical solid propellant rocket motor.

2.2.1 The Case

The purpose of a solid propellant rocket motor case is to serves as a container for the solid propellant grain, also serves as vessel that hold up the combustion pressure during operation. It was found that the internal pressure could reach a value from 3MPa to 25MPa and recommended a safety factor of 1.4 for the design [24]. The material used for the case will be different according to the mission it carries.

The case design commonly limited by the motor and vehicle requirements, as the motor case served as main structure for missile and launch vehicle. Sutton [25] has listed a series of loads (appendix A) on the rocket motor case which must be considered during the design stage. He also mentioned that there are three type of materials used in the industry: high-strength metal (steel, aluminum, or titanium alloy), wound-filament reinforced plastic, and a mixture of these two for extra strength. Comparison between these three materials can be found in appendix B. Example of material used on ballistic missiles and space launches [24] is the AMS 6478 or AMS 6520 for their good mechanical strength (>1000MPa), while the tactical missile cases [26] are made of composite material like glass-epoxy, Kevlar-epoxy and carbon-epoxy with winding technique in production.

Rocket motor case configuration can be found in different shapes: long and thin cylinders (L/D of 10), spherical or nearly spherical shape. Since motor case is the primary structure for a typical rocket application, the mass of the case gives a value from 0.70 to 0.94 for the propellant mass fraction ξ. Propellant mass fraction is the ratio of propellant mass m_p to initial mass m_o [27].

$$\xi = m_p / m_0 \tag{3}$$

Assuming that there is no bending in the case wall and all the loads are taken in tension, then the simple membrane theory can be applied to estimate the stress in the motor cases. The following formula is used to calculate the hoop stress σ_θ, in term of longitudinal stress σ_1, radius of a simple cylinder r and thickness d, with the chamber pressure p [25].

$$\sigma_\theta = 2\sigma_1 = pr / d \tag{4}$$

The combustion pressure will create growth in length L and in diameter D, which can be defined in term of Young's modulus E and Poisson's ratio υ [25].

$$\Delta L = \frac{pLD}{4Ed}(1-2\upsilon) = \frac{\sigma_1 L}{E}(1-2\upsilon) \tag{5}$$

$$\Delta D = \frac{pD^2}{4Ed}(1-\frac{\upsilon}{2}) = \frac{\sigma_\theta D}{2E}(1-\frac{\upsilon}{2}) \tag{6}$$

Stress analysis nowadays can be done with the help of the computer aid software. SolidWorks 2008 is one of the examples used in this research. The theory of finite element was applied in the calculations to determine the case design with acceptable stress values.

2.2.2 The nozzle

The nozzle is a crucial part of the rocket motor, where the thrust is produced through the expansion and acceleration of the hot gases. In other words, this is the place where the chemical energy from combustion chamber transformed into kinetic energy. The size of a nozzle usually mentioned in throat diameter, and the value can be found from 0.05 inch to 54 inch [25]. The nozzle needs to be build with material that could withstand the high heat transfer during the operation period in the range of one second to a few minutes.

Nozzle can be divided into five categories according to the design and application. They are the fixed nozzle, moveable nozzle, submerged nozzle, extendible nozzle and blast-tube-mounted nozzle. These five categories of nozzle are self explainable by their name and illustrated in the figure 2.3. The construction of the nozzle can be as simple as the non-moveable nozzle to the complex multi-piece nozzle that could control the thrust vector.

The current research used the typical simpler and smaller nozzle design that usually used for the low chamber pressure and short operation time (< 10 seconds). Basic thermodynamic equations laid the foundation in determining the nozzle throat area, nozzle half angle and nozzle expansion ratio. The detail of this equation will be discussed in the section 2.3 of this report.

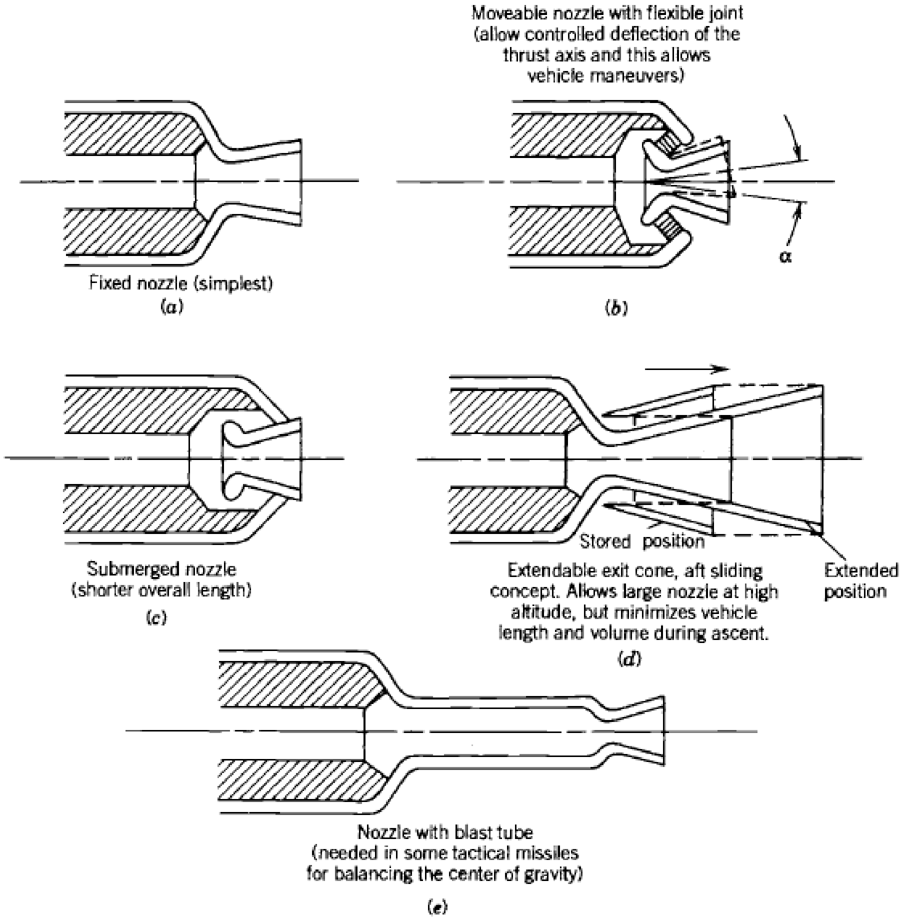

Figure 2.3 Categories of nozzle [25]

2.2.3 Propellant grain

Propellant grain is the propellant charge that produced high temperature gases in the combustion chamber during operation. Solid propellant is a product that contains the mixture of oxidizer, fuel and other chemical ingredients such as binder.

Propellants can be categorized into two major types according to the chemical compositions: double-base (DB) propellants and composite propellants.

Double-base propellants [28] mostly used in small tactical missiles, consists of nitrocellulose (NC), nitroglycerine (NG) and small portion of additives that form a homogeneous propellant grain. Nitrocellulose and nitroglycerine will gather the carbon, hydrogen and oxygen for the chemical reaction [29]. These two main ingredients will works as oxidizer and fuel at the same time. Within double base propellants, there are other names given according to the fabrication methods. For instance, cast double-base (CDB) and the extruded double base (EDB). In modern solid propellant, new element like the crystalline nitramines (HMX or RDX) is added in the double base propellants to form the cast-modified double-base propellant for better performance. When elastomeric binder is added to the previous one, it will become the elastomeric-modified cast double-base (EMCDM) propellant.

Density of double-base propellants are influenced by the density of their raw material and additives. Normally, EDB propellants have density value in between 1.55 to 1.66, while the CDB propellants run from 1.50 to 1.58 [30]. Specific heat capacity is approximately 0.350 calorie per gram per degree for all double-base propellants [30].

Composite propellants [28] are made of oxidizer, fuel and plastic binder to form a heterogeneous propellant grain. Oxidizer is the main ingredient of the propellant which might consist of 60-80% of the weight, while the fuel amount generally below 25% [31]. Typical composite propellants are made of Aluminum powder as fuel and ammonium perchlorate (AP) as oxidizer and HTPB as plastic binder. Fabrication of this kind of propellant usually involved the curing process of the binder. By adding other additive like RDX, HMX and elastomeric binder, they are named accordingly as double-base propellants. Most of the composite propellants produce lots of smoke during operation. Roger E. Lo introduced a novel kind of solid propellant namely cryogenic solid rocket motor where hydrogen and oxygen in solid form are used [32]. Ammonium nitrate (AN) is another kind of oxidizer is used in solid propellant that produced smokeless product compared to ammonium perchlorate [33]. Luigi De Luca and his team have done a research on the combustion mechanism of an RDX-based composite propellant [34].

There are a few factors that will influence the decision in selecting the desirable propellant in rocket motor design. One of them is high specific impulse produced by the propellant grain. Specific impulse of the propellants is often measured at standard condition where the combustion happened at internal pressure 1000 psi and exhaust at sea-level atmosphere through designed nozzle. The range of specific impulse for these propellants can refer to appendix C and appendix D, while the others influencing factor can be found in appendix E.

Detail about the burn rates and grain configurations for the solid propellant grain will be discussed in section 2.4.

2.2.4 Thermal insulation

Thermal insulation also known as thermal protection that applied on the wall of the case [35]. For solid propellant rocket in the industry, the manufacturer will design a layer of material called liner to bond the propellant with the wall. There is another kind of material referred as inhibitor that applied on the propellant grain surface to control the burning surface. All these three materials (thermal insulation, liner and inhibitor) are normally grouped as insulating materials.

Insulating materials will protect the components that exposed to extreme temperature especially the combustion chamber. Thermal protection is needed as the hot gases temperature could reach 2000 to 4000 K. Protection by ablation [36] usually used in practice as thermal protection where organic materials will protect the underlying surface through a decomposition mechanism.

2.2.5 Ignition system

Ignition system is very important as this is where the mechanism of initiating the combustion of the propellant grain took place. With the start of an electrical signal, heat is transfers from igniter to propellant surface, thus hot gases is generated by the burning of the grain surface. It is very important to make sure that the igniter used can generate sufficient heat and temperature to burn the grain surface. This is why the design of an igniter must compatible with the energy needed for a grain to ignite. Ignitibility of a propellant can be affected by the following factors [37]:

a) The formation of the propellant.

b) Propellant grain surface roughness.

c) Age of the propellant grain.

d) Propellant grain surface initial temperature.

e) Igniter propellant and Igniter initial temperature.

f) Environment temperature.

g) The mode of heat transfer.

Igniter propellant usually is small (<1% of motor propellant) [25], the detail of the igniter propellant will be discussed in Chapter 6 of this report. There are two types of igniter in the industry: pyrotechnic igniters and pyrogen igniters. Figure 2.4 below show the typical location of the igniter in the rocket motor.

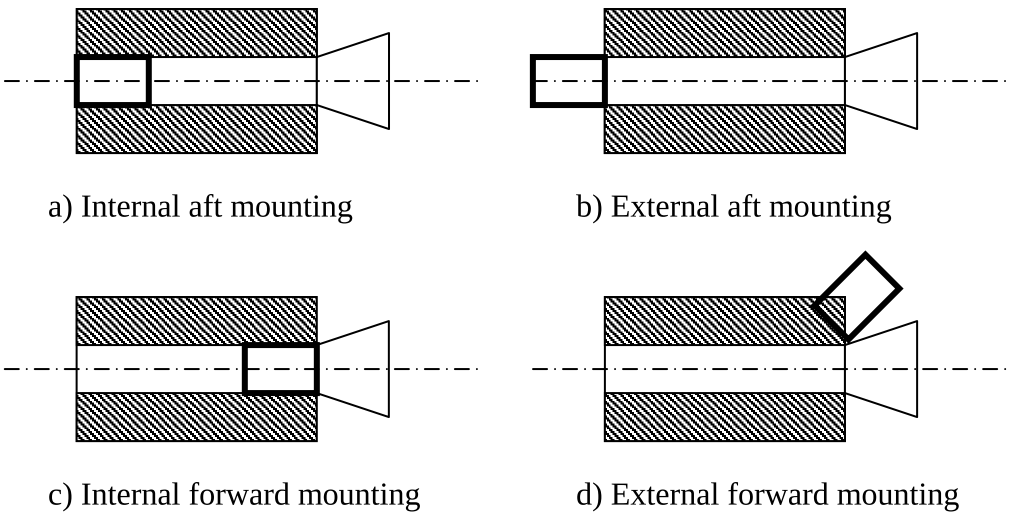

Figure 2.4 Igniter locations on a typical rocket motor

Pyrotechnic igniters made of powerful solid explosives material and usually packed in a small container. They have many names such as pellet basket, perforated tube and sheet igniters according to their designs. While the pyrogen igniters is usually design in the form of small rocket motor (that did not create thrust) and install inside larger rocket motor and the energy generated is usually higher than pyrotechnic igniters.

The design of igniters depends deeply on experimental results and testing. One example given by Sutton [25] is the mass of an igniter charge can be calculated by the following equation which developed from the AP/Al composite propellant rocket motor experiments.

$$m_{ic} = 0.12(V_F)^{0.7} \tag{7}$$

where m_{ic} is igniter charge in grams and V_F is the motor free volume in cubic inches.

2.2.6 Rocket motor design approach

It was found that there is no well-defined design approach for rocket motor [25] as the requirements change with the rocket application and the limited by designer resources such as background experiences, available data on rocket motor designs, and equipments for testing and analysis.

In preliminary design process, typical process in rocket motor design is started by considering the applications of the rocket motor, where the mission and the propulsion requirement are defined. The rocket motor applications will link with functional design parameter such as the total impulse, specific impulse and the rocket motor initial mass.

Selection of propellant and grain configuration is crucial in the preliminary design process. Sutton [25] mentioned that it is a challenge for the propellant to meet the three requirements: the performance (specific impulse), burning rates to suits the thrust-time curve and strength (maximum stress and strain). In order to reduce time for analyses and tests, a proven propellant will usually be used and modified to fit the new applications.

The drawings of the rocket motor with computer aid software will provide sufficient information for analyses. Volume for combustion chamber and propellant can be studied accordingly. From the layout, the rocket motor initial mass can be known. This helps the material selection for the components to meet the requirement in applications.

Cost is an important factor in rocket motor design, thus efforts in finding lower-cost materials and components, simpler fabrication process and fewer assembly procedures are part of the design process. Project plan is used to control the costing and the delivery schedule.

After the approval of the selected preliminary design, the final design of all the components can begin. Improvements in the design are expected during manufacturing testing and the detailed design is reviewed again before manufacturing can begin. The detail design is considered to be completed when the rocket motor successfully passes qualification tests and start production for deliveries [25]. An example of a simplified diagram in rocket motor design can be found in Appendix F in this report.

2.3 Fundamental equations in rocket designs

2.3.1 Total impulse and Specific impulse

Total impulse is the basic parameter used to define a rocket motor performance. Total impulse I_t can be defined by the graph area under the thrust curve obtained from the experiment (Figure 2.5). Total impulse can be calculated in term of thrust F and burning time t based on the following equation [27]:

$$I_t = \int_0^t F dt \tag{8}$$

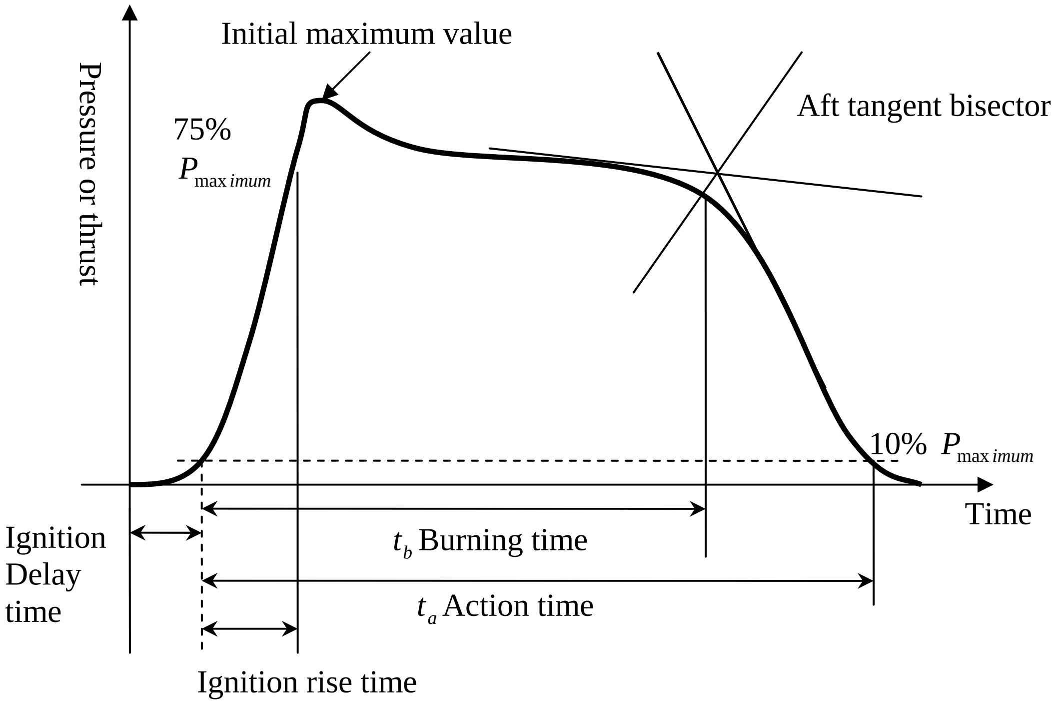

Figure 2.5 Burning time and action time definitions. [38]

Specific impulse I_s is another key parameter used to classify the rocket performance. Specific impulse is the total impulse per unit weight of propellant, thus given the total mass flow rate of propellant $\dot{m}$ and standard acceleration of gravity $g_0 (9.8066\, m/\sec^2)$ and total effective propellant mass m_p then it can be written as the following in the unit of seconds [27]:

$$I_s = \frac{\int_0^t F dt}{g_o \int \dot{m}\, dt} = \frac{I_t}{g_o \int \dot{m}\, dt} = \frac{I_t}{g_0 m_p} \tag{9}$$

In practice, the exhaust velocity is not uniform at the nozzle exit and it is assumed to be a uniform axial velocity c for theory analysis. Effective exhaust velocity c [27] is commonly used in Russian literature for rocket performance and it is related as below in unit of meter per second:

$$c = I_s g_0 = \frac{F}{\dot{m}} \tag{10}$$

2.3.2 Thrust

Hot gases produced by the propellant burning process will create changes in momentum and pressure through nozzle, a forward force called thrust is produced as the result. A simplified diagram of a rocket motor is illustrated in figure 2.6, where the chamber area denoted as 1, nozzle throat area denoted as t, nozzle exhaust area denoted as 2 and the atmosphere condition as 3. The pressure, cross sectional area, temperature and velocity are labeled as P, A, T and v. Total thrust consists of momentum thrust and pressure thrust. The first term in Equation 11 is the momentum thrust, while the second term is the pressure thrust.

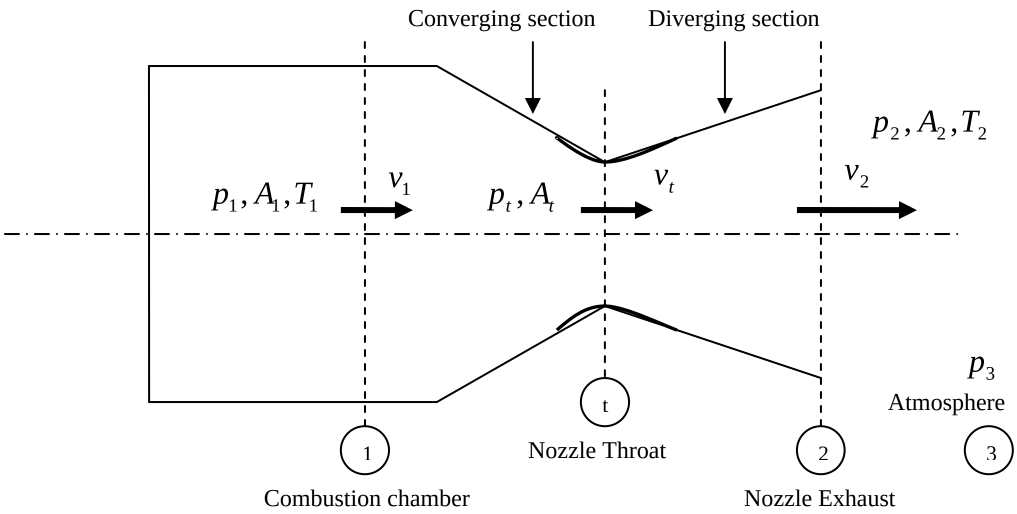

Figure 2.6 Simplified diagram for a rocket motor with De Laval nozzle.

Total thrust can be defined as [27]:

$$F = \dot{m} v_2 + (p_2 - p_3) A_2 \tag{11}$$

When the atmosphere pressure matched the exhaust pressure ($p_2 = p_3$), the nozzle is operating at optimum expansion ratio. In practice, nozzle is designed to exhaust gases at pressure equal or a little higher that atmosphere pressure.

Thrust F also can be calculated by relationship between thrust coefficient C_F, throat area A_t and chamber pressure p_1 [39] :

$$F = C_F A_t p_1 \tag{12}$$

Given that,

$$C_F = \sqrt{\frac{2\gamma^2}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1-\left(\frac{p_2}{p_1}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \frac{p_2 - p_3}{p_1}\frac{A_2}{A_t} \tag{13}$$

From Equation 10 and 11, the effective exhaust velocity c [27] can be written as below:

$$c = I_s g_0 = \frac{F}{\dot{m}} = \frac{\dot{m} v_2 + (p_2 - p_3)A_2}{\dot{m}} = v_2 + \frac{(p_2 - p_3)A_2}{\dot{m}} \tag{14}$$

Another term called characteristic exhaust velocity c^* [27] [39] provide a value for comparison between chemical rocket propulsion system designs and propellants.

$$c^* = \frac{p_1 A_t}{\dot{m}} = \frac{I_s g_0}{C_F} = \frac{c}{C_F} = \frac{\sqrt{\gamma R T_1}}{\gamma\sqrt{\left[\frac{2}{\gamma+1}\right]^{\frac{\gamma+1}{\gamma-1}}}} \tag{15}$$

2.3.3 Nozzle Theory

These are the assumptions [39] made upon an ideal rocket unit for simplifications in obtaining solutions for rocket propulsion system with simple, quasi-one-dimensional theory.

a) Homogeneous propellant is used.

b) All the working fluid is gases.

c) The perfect gas law is applied.

d) The flow is adiabatic.

e) There is no significant friction and all boundary layer effects are ignored.

f) No shock waves or discontinuities in the nozzle flow, thus flow entropy is zero.

g) The propellant flow is uniform, steady and constant.

h) All exhaust gases exit the nozzle with axially directed velocity.

i) Chemical equilibrium happened within the combustion chamber.

j) Stored propellants are at room temperature.

Using the following variables:

p, T, ρ	: Pressure, temperature and density of the gases;
v	: Gas flow velocity;
A	: Cross-section area of the nozzle;
R	: gas constant (R= $R^{'} / M_w$ **)**;
$R^{'}$	: Universal gas constant 8314.3 $J / kg - mole - K$ or 8.314 J/Kmol;
M_w	: Molecular weight of the flowing gas;
a	: Speed of sound 340.29 m/s ($a = \sqrt{\gamma RT}$);
γ	: Ratio of specific heat at constant pressure c_p and specific heat at constant volume c_v , ($\gamma = c_p / c_v$)
M	: Mach number ($M = v / a$);
V	: Specific volume

Base on assumptions above, the concept of enthalpy h can be used to constructs the thermodynamic relationship. The total enthalpy or stagnation enthalpy per unit mass h_0 is constant [39], given by

$$h_0 = h + \frac{v^2}{2J} = constant \qquad (16)$$

J is the mechanical equivalent of heat which equal to one in SI unit when there is a thermal unit (calorie) and mechanical unit (Joule) in the equation. The change of enthalpy within two point x and y can be transform into kinetic energy [39].

$$h_x - h_y = \frac{1}{2}\frac{\left(v_y^2 - v_x^2\right)}{J} = c_p\left(T_x - T_y\right) \qquad (17)$$

Taking the continuity of the mass flow from point x to point y, the mass flow rate can be written as [39]:

$$\dot{m} = \frac{Av}{V} = \rho Av \qquad (18)$$

The perfect gas law [39] defined that the pressure and the specific volume increase with the gas temperature and it is governed by the following relationship:

$$pV = RT \qquad (19)$$

In any section of the nozzle, isentropic flow process at any point x and y for the pressure, temperature and specific volume [39] [40] can be written as:

$$\frac{T_x}{T_y} = \left(\frac{P_x}{P_y}\right)^{\frac{\gamma-1}{\gamma}} = \left(\frac{V_y}{V_x}\right)^{\gamma-1} \quad \text{or} \tag{20}$$

$$\frac{P_y}{P_x} = \left(\frac{T_y}{T_x}\right)^{\frac{\gamma}{\gamma-1}} = \left(\frac{\rho_y}{\rho_x}\right)^{\gamma} \tag{21}$$

Considering stagnation conditions where the flow stopped isentropically, v_y in Equation 17 is zero, and the equation become this:

$$-\frac{1}{2}\frac{v_x^2}{J} = c_p\left(T_x - T_y\right) \tag{22}$$

Re-arrange Equation 22, by putting location y is the stagnation point,

$$T_0 = T + \frac{1}{2}\frac{v^2}{c_p J} \tag{23}$$

Substitute the specific heat ratio for constant pressure $c_p = \frac{\gamma R}{\gamma - 1}$ and Mach number $M = \frac{v}{\sqrt{\gamma RT}}$ into Equation 21,

$$T_0 = T\left[1 + \frac{1}{2}(\gamma - 1)M^2\right] \tag{24}$$