Investigation into the Combustion and Performance of Small Solid Propellant Rocket Motors

Morgan G. Carter

UNSW@ADFA, Canberra, ACT

This thesis project aims to produce a thrust-measuring instrument to accurately measure the thrust-time relationship, hence impulse, of small solid propellant Estes rocket motors and compare with the accepted thrust-time relationships (Figure 3) and impulses provided in the Estes product catalogue. A temperature sensitivity analysis will also be conducted to determine how initial propellant temperature affects the performance of the Estes solid propellant rocket motors. This sort of analysis has many military applications. Military solid propellant missiles/rockets can potentially be launched over quite a large range of temperatures from -50°C at high altitudes on fighter aircraft to +50°C in hot and dry deserts. Test results from this experimentation will be qualified using rocket theory and quantified using calculations. Once the thrust-time-temperature relationships of the black-powder Estes rocket motors have been determined and interpreted; this thesis will focus on designing, constructing and testing a home-made solid propellant motor. Motor specifications such as grain geometry and propellant type will be varied to begin exploring the theory behind solid propellant rocket motor combustion and performance.

Nomenclature

\begin{align*}
I_t & = \text{total impulse [N s]} \\
F & = \text{thrust force [N]} \\
t_b & = \text{burning time of propellant grain [s]} \\
I_s & = \text{specific impulse [s]} \\
m & = \text{propellant mass flow rate [kg s}^{-1}] \\
g_0 & = \text{acceleration due to gravity [m s}^{-2}] \\
c & = \text{effective exhaust velocity [m s}^{-1}] \\
\zeta & = \text{propellant mass fraction} \\
m_p & = \text{propellant mass [kg]} \\
m_0 & = \text{vehicle/motor initial mass [kg]} \\
v_2 & = \text{instantaneous exhaust velocity [m s}^{-1}] \\
F_T & = \text{total thrust [N]} \\
p_2 & = \text{exhaust gas pressure at the nozzle exit [Pa]} \\
p_3 & = \text{ambient fluid pressure [Pa]} \\
A_2 & = \text{nozzle exit area [m}^2] \\
r & = \text{propellant burn rate [m s}^{-1}] \\
a & = \text{empirically determined constant of burn rate} \\
n & = \text{empirically determined constant of burn rate} \\
p & = \text{combustion chamber pressure [Pa]} \\
A_b & = \text{burning area of propellant grain [m}^2] \\
p_b & = \text{propellant density [kg m}^{-3}] \\
\sigma_p & = \text{temperature sensitivity of burning rate at constant pressure [K}^{-1}] \\
T_0 & = \text{propellant initial temperature [K]} \\
\pi_K & = \text{temperature sensitivity of pressure at constant motor geometry [% °C}^{-1}] \\
\end{align*}

1 Aeronautical Engineering: Project, Thesis and Practical Experience ZACM 4049/4050.
I. Introduction

This thesis represents the initial stages of a potentially ongoing investigation within UNSW@ADFA into the science and engineering behind solid propellant rocket motors. Solid propellant rocket motors are a very effective and relatively simple method of propulsion. They have many useful applications; ranging from amateur model rocketry to boosting payloads into space. Another useful application for solid propellant rocket motors is they provide a powerful, cheap and simple test bed for SCRAMJET motor research, such as that conducted by ADFA. They also have many useful military applications; such as for air-to-air missiles and small-arms shoulder-launched rockets. The advantage of solid propellant rocket motor research is that the same theory applies to motors of all scales. Small-scale ‘microrocket’ motors have been produced with a typical thrust of 0.001 Newtons (Rossi, Oriieux, Larangot, Do Conto & Esteve, 2002, p125) whilst the largest, most powerful solid propellant motor ever produced is the Space Shuttle’s Solid Rocket Boosters (SRB) with a maximum thrust of 14.7 million Newtons each (Heath & Dick, 2000, p24). This means that solid propellant rocket motor experiments and research can be conducted using small, scaled-down motors for convenience, safety and efficiency while the findings can be applied to large, full-scale motor applications (Sutton & Biblarz, 2001, p427).

Another reason for this thesis stems from the popular hobby of Amateur Model Rocketry. In the mid-1950s when the space age began, flying rockets became very popular. However, there were no safe propellants readily available for amateurs to launch their own rockets. In 1958 Vernon Estes developed the first safe, mass-produced model rocket motor. The Estes rocket motor was destined to make model rocketry one of the most popular outdoor activities enjoyed today (Estes Catalogue, 2006, p4). Commercially produced model rocket motors, such as Estes products, are very reliable; however, they are expensive to the model rocket enthusiast (refer to Appendix 10), especially considering the motor casing is non-reusable. If a reusable home-made solid propellant rocket motor could be designed, built and launched for a much lower cost than that of the commercially-produced motors while still being competitive in the areas of performance, reliability, safety and convenience, then this home-made motor would be a very attractive alternative to purchasing expensive motors from hobby shops.

II. Background

A. Introduction

In solid propellant rocket motors, the propellant to be burned is contained within the combustion chamber or case. The case is effectively a pressure vessel that is designed to contain the high gas pressures required for propellant combustion which for blackpowder propellant is in the range of 100 [psi] to 1000 [psi] or 0.69 [MPa] to 6.9 [MPa] (Zaehringer, 1955, p38). The solid propellant charge is called the grain and it contains all the chemical elements for complete burning. Sectional diagrams of a typical solid rocket motor design and an Estes motor design can be seen in Figures 1 and 2 respectively. Once ignited, the grain usually burns smoothly at a predetermined rate on all exposed surfaces of the grain. The exposed grain surfaces continue to recede in a direction perpendicular to the exposed surfaces until the propellant is burned and consumed. The resulting hot gas flows through the supersonic nozzle to impart thrust. For most solid motors, once they are ignited they cannot be extinguished and the thrust cannot be randomly throttled in any way. Almost all rocket motors are used only once. The hardware that remains after all the propellant has been burned and the mission completed—namely the nozzle, case, or thrust vector control device—is not reusable (Sutton & Biblarz, 2001, p6). In very rare applications, such as the Shuttle SRBs, is the hardware recovered, cleaned, refurbished, and reloaded. Reusability makes the design more complex, but if the hardware is reused often enough a major cost saving will result (Sutton & Biblarz, 2001, p419). Case reusability is a desired design requirement of the proposed home-made rocket motor in this thesis, albeit on a much smaller scale.
Rocket propulsion principles are essentially those of mechanics, thermodynamics, and chemistry. Propulsion is achieved by applying a force to a vehicle in the direction of intended travel. For rockets, this propulsive force is obtained by ejecting propellant at high velocity out of a nozzle. Rocket propulsion principles include; propulsive force, exhaust velocity, and the efficiencies of creating and converting system energies.

1. Impulse and Mass Fraction

The total impulse, \( I_t \) [N s], is the thrust force, \( F \) [N] (which can vary over time as seen in Figure 3), integrated over the burning time of the propellant grain, \( t_b \) [s]:

\[
I_t = \int F \, dt
\]  

(1)

\( I_t \) is proportional to the total energy released by all the propellant in a solid rocket motor. \( I_t \) for the C6-0 and D12-0 Estes rocket motors that will be used in this thesis are 10.00 [N s] and 20.00 [N s] respectively. The specific
impulse, $I_s$ [s], is the total impulse per unit weight of propellant. $I_s$ is an important performance parameter of a rocket motor and a higher value means better performance.

$$I_s = \left( \int F \, dt \right) / (g_0 \int m \, dt)$$  \hspace{1cm} (2)

Where $\dot{m}$ [kg s$^{-1}$] is the propellant mass flow rate and $g_0$ [m s$^{-2}$] is acceleration due to gravity. This equation will give a time-averaged specific impulse value for any rocket motor, particularly where the thrust varies with time (Sutton & Biblarz, 2001, pp27-28).

In a rocket nozzle the actual exhaust velocity is non-uniform over the entire exit cross-section and does not represent the entire thrust magnitude. The velocity profile is difficult to measure accurately and will not be an aim of this thesis. For convenience, a uniform axial velocity, $c$ is assumed which allows a one-dimensional description of the problem. This effective exhaust velocity, $c$ [m s$^{-1}$], is the average equivalent velocity relative to the motor case at which propellant is ejected from the nozzle over the total burn time, $t_b$. It is defined in the following way:

$$c = I_s g_0$$  \hspace{1cm} (3)

Since $c$ and $I_s$ differ only by an arbitrary constant, $g_0$, either one can be used as a measure of rocket performance. In solid propellant rockets it is difficult to measure the propellant flow rate accurately. Therefore, the specific impulse is often calculated from total impulse and the propellant weight (found using the difference between initial and final motor weights). In turn the total impulse is obtained from the integral of the measured thrust-time relationship (Sutton & Biblarz, 2001, p29).

The propellant mass fraction, $\zeta$, indicates the fraction of propellant mass, $m_p$ [kg], in a motor initial mass, $m_0$ [kg] (this fraction can also be applied to an entire vehicle or vehicle stage). When applied to a rocket motor, the value of the propellant mass fraction, $\zeta$, indicates the quality of the motor design, thus, will be a suitable comparison indicator when comparing the Estes motors with my home-made motor. A high value of $\zeta$ is desirable as it indicates a structurally efficient design (Sutton & Biblarz, 2001, p30).

$$\zeta = \frac{m_p}{m_0}$$  \hspace{1cm} (4)

$m_p$ for the Estes C6-0 and D12-0 motors is approx. 13 grams and 25 grams respectively. $m_0$ for the C6-0 and D12-0 motors is approx. 23 grams and 41 grams respectively. These values were determined from the Estes Catalogue, 2006, p35 and the initial motor tests detailed below.

2. **Thrust**

The thrust produced by a rocket motor can be thought of as the reaction experienced by its motor case due to the ejection of matter at high velocity. This ‘reaction’ phenomenon is more correctly stated as conservation of momentum. Momentum is a vector quantity and is defined as the product of mass times velocity. Rockets generate their forward momentum at the expense of the momentum of the ejected propellant products, which are accelerated towards the rear by the motor nozzle. Over a small amount of time, the mass of the ejected exhaust gases is relatively small compared with the instantaneous mass of the rocket; however, it is the shear magnitude of the exhaust gas velocity that imparts such a large change in momentum on the rocket, and therefore, such large accelerations. The thrust, due to change in momentum, is given below (Sutton & Biblarz, 2001, p32):

$$F = \dot{m} \cdot v_2$$  \hspace{1cm} (5)

Where $v_2$ [m s$^{-1}$] is the instantaneous exhaust velocity relative to the motor case. This force represents the total propulsion force when the nozzle exit pressure equals the ambient pressure. The pressure of the surrounding atmosphere gives rise to the second contribution that influences the thrust. Figure 4 shows schematically the external pressure acting uniformly on the outer surface of a rocket chamber/case and the gas pressures on the inside of a typical solid propellant rocket motor. The size of the arrows indicates relative magnitude of the pressures and velocities. A summation of all the pressure-surface vectors will yield an overall pressure thrust. The total thrust, $F_T$ [N], produced by the motor can be given by:
The first term on the RHS of equation (6) is the momentum thrust represented by the product of the propellant mass flow rate and its exhaust velocity relative to the motor case. The second term represents the pressure thrust consisting of the product of the cross-sectional area at the nozzle exit, \(A_2\) [m\(^2\)] (where the exhaust jet leaves the motor), and the difference between the exhaust gas pressure at the exit, \(p_2\) [Pa], and the ambient fluid pressure, \(p_3\) [Pa]. If the exhaust pressure is less than the surrounding fluid pressure, the pressure thrust is negative. Because this condition gives a lower thrust and is undesirable, the rocket nozzle is usually so designed that the exhaust pressure is equal (nozzle with ‘optimum expansion ratio’) or slightly higher (under-expanded nozzle) than the ambient atmospheric pressure (Sutton & Biblarz, 2001, pp32-33).

The above mentioned theory can quite easily be applied to small solid propellant rocket motors.

C. Propellant

As seen in Figures 1 and 2, the motor grain is the solid body of the hardened propellant and typically accounts for 82% to 94% of the total motor mass (55% to 60% of the total motor mass for the C6-0 and D12-0 Estes motors used in this thesis according to Estes Catalogue, 2006, p35). Grains can have many geometry types including slots, grooves, holes, or even no cavities at all which are known as end-burning grains such as the Estes motors (refer to Figure 2). End burning grains burn solely in the axial direction and maximises the amount of propellant that can be placed in a given cylindrical motor case (Sutton & Biblarz, 2001, p451). Various grain geometry types and associated thrust-time profiles can be seen in Figure 5. The various grain geometries alter the initial burning surface, which determines the initial mass flow rate and the initial thrust. The hot reaction gases of the burnt propellant flow along the perforation or grain cavity toward the nozzle (Sutton & Biblarz, 2001, pp 417-418). A high thrust is desired to apply initial acceleration, but, as propellant is consumed and the vehicle weight is reduced, a decrease in thrust is desirable; this often reduces the drag losses, and usually permits a more effective flight path (i.e. a greater height reached by the Estes rockets). Therefore, there is a benefit to vehicle mass, flight performance, and cost in having a higher initial thrust during the boost phase of the flight, followed by a lower thrust (often 10% to 30% of the boost thrust) during the sustaining phase of the powered flight (Sutton & Biblarz, 2001, p451). This principle has been applied to the C6-0 and D12-0 Estes motors as seen in Figure 3.

The Estes rocket motor propellant is tightly compacted blackpowder. Most blackpowders contain 75% of potassium nitrate (KNO\(_3\)) or sodium nitrate which acts as the oxidiser, 15% of charcoal or some form of carbon (C) as the fuel, and 10% sulphur (S) which acts as both a fuel and burn rate catalyst. Blackpowder can be ignited at a temperature of about 190°C (Zaehringer, 1955, p38) for the normal product and becomes hygroscopic (absorbs water) at humidities exceeding 90% and then rapidly ceases to burn (Fordham, 1980, p165). For this reason, Estes
rocket motors should be disposed of in water as this saturates the blackpowder and renders the motor useless (Estes Model Rocket Engine and Igniter Instructions, 2001-2003). The properties of blackpowder depend considerably on the charcoal used. It is important that the wood should be carbonised to the correct extent and this depends on the nature of the wood. For alderwood charcoal, the optimum carbon content is 74%, whereas for birchwood 82% gives the best results. In general, if the carbon content is too low a readily ignited blackpowder is obtained but it has slow burning properties. If the carbon content is too high, the blackpowder will be difficult to ignite and irregular in burning. The burning rate of the powder can also be varied by changing the potassium nitrate content. The maximum burning rate is usually obtained at a potassium nitrate content of rather less than 70%. The burning rate, \( r \) [m s\(^{-1}\)], of blackpowder depends exponentially on the chamber pressure, \( p \) [Pa], by the following equation, known as Vieille’s law or Saint Robert’s law:

\[
r = ap^n
\]

Where both \( a \) and \( n \) are empirically-determined constants (Fordham, 1980, p166). The constant, \( a \), is affected by the initial grain temperature and the constant \( n \) determines how sensitive the burn rate is to chamber pressure. A typical range of burn rates for blackpowder used in solid propellant rocket motors is approx. 0.1 to 0.5 inches per second or 0.00254 [m s\(^{-1}\)] to 0.0127 [m s\(^{-1}\)] (Zaehringer, 1955, p38). Factors affecting burn rate include; propellant type, oxidizer to fuel ratio (potassium nitrate to charcoal ratio), oxidizer (potassium nitrate) particle size and compaction, ratio of oxidizer particle sizes, burning rate catalysts (sulphur), heat of combustion of the binder, chamber pressure, propellant initial temperature, flame temperature, gas velocity through the grain cavity, and motor accelerations (Rochford, 1999, p7). Solid propellant rocket motors experiencing an acceleration will exhibit an increase in burning rate (Fuchs, Peretz & Timnat, 1982, p539).

The mechanism of combustion of blackpowder is extremely complex. The main process of combustion of blackpowder was studied exhaustively by Alfred Nobel and Frederick Augustus Abel. These experimental results were examined in much greater detail by Allen George Debus, who has provided a self-consistent account of the chemical reactions involved. Debus considers that the overall reaction can be divided into two distinct stages: (i) a rapid oxidation process and (ii) a slower reduction process. The oxidation process is responsible for the actual explosion. Whilst it is no doubt complex, it can be simplified to the following overall equation:

\[
10\text{KNO}_3 + 8\text{C} + 3\text{S} \rightarrow 2\text{K}_2\text{SO}_4 + 6\text{CO}_2 + 5\text{N}_2
\]

As the initial combustion of blackpowder contains, for each 10 molecules of potassium nitrate, 14 molecules of carbon and 4 molecules of sulphur, this equation does not account for 6 molecules of carbon and 1 molecule of sulphur. The excess carbon and sulphur take part in slower reduction reactions which are as follows:

\[
4\text{K}_2\text{CO}_3 + 7\text{S} \rightarrow 2\text{K}_2\text{SO}_4 + 3\text{K}_2\text{S}_2 + 4\text{CO}_2
\]

As the reduction is a slow process, it is not necessarily complete when the blackpowder has done its work. The reduction reactions are endothermic and lower the total heat evolution. On the other hand they increase the amount of gas evolved which increases mass flow rate and/or chamber pressure, hence, good for solid propellant in rocket motors. These equations only represent the overall reactions and that the actual paths are more complicated. As the reduction stage of the reaction does not necessarily go to completion the reaction products depend to some extent on the conditions of firing. In all cases, however, the chief products appear to be potassium carbonate (\( \text{K}_2\text{CO}_3 \)), potassium sulphate (\( \text{K}_2\text{SO}_4 \)), potassium disulphide (\( \text{K}_2\text{S}_2 \)), carbon dioxide (\( \text{CO}_2 \)), and nitrogen (\( \text{N}_2 \)). Side reactions give the by-products usually observed, namely, hydrogen (\( \text{H}_2 \)), hydrogen sulphide (\( \text{H}_2\text{S} \)), methane (\( \text{CH}_4 \)), ammonia (\( \text{NH}_3 \)), water (\( \text{H}_2\text{O} \)), and potassium thiocyanate (\( \text{KSCN} \)). In most analyses small amounts of unburnt powder have also been observed (Fordham, 1980, pp167-168).

D. **Temperature Sensitivity of Solid Propellant Grains**

Thrust is related to mass flow rate, \( \dot{m} \), by equation (5). Mass flow rate is related to the propellant burn rate by the following equation:
\[ \dot{m} = A_{b}r\rho_{b} \]  

(11)

Where \( r \) is propellant burn rate, \( A_{b} \) [m²] is the burning area of the propellant grain, and \( \rho_{b} \) [kg m⁻³] is the solid propellant density prior to ignition (Sutton & Biblarz, 2001, p427). 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. For a particular propellant and motor type, the relationship between chamber pressure, \( p \) (which is related to the thrust), and motor burning time, \( t_{b} \), varies with initial grain temperature as indicated in Figure 6 (Rochford, 1999, pp7-8).

There are two primary temperature sensitivity coefficients of interest: the temperature sensitivity of burning rate at constant pressure, \( \sigma_{p} \) [K⁻¹], and the temperature sensitivity of pressure at constant motor geometry, \( \pi_{K} \) [% °C⁻¹].

The definition of \( \sigma_{p} \) is as follows:

\[
\sigma_{p} = \frac{\delta \ln r}{\delta T_{0}}_{p} \quad \text{or} \quad \sigma_{p} = \frac{(\ln r_{2} - \ln r_{1})/(T_{02} - T_{01})}{p}
\]

(12)

Where \( r \) is the propellant burn rate, \( T_{0} \) [K] is the propellant initial temperature, and subscripts 1 and 2 indicate measurements taken at different initial temperatures. The subscript \( p \) indicates at constant chamber pressure (Rochford, 1999, p9). Typical values of \( \sigma_{p} \) are 0.001 to 0.009 per degree Kelvin [K⁻¹] (Sutton & Biblarz, 2001, p432). For a given pressure-time variation such as in Figure 6, dynamic burning is enhanced by larger values of \( \sigma_{p} \). Propellants with high \( \sigma_{p} \) tend to extinguish easily, ignite with greater difficulty and are prone to combustion instabilities (Kuo & Coates, 1977, p1187). The definition of \( \pi_{K} \) is as follows, where typical values of \( \pi_{K} \) are 0.067 to 0.278 percent per degree Celsius [% °C⁻¹] (Sutton & Biblarz, 2001, p432):

\[
\pi_{K} = \frac{\delta \ln p}{\delta T_{0}}_{K} \quad \text{or} \quad \pi_{K} = \frac{(\ln p_{2} - \ln p_{1})/(T_{02} - T_{01})}{p}
\]

(13)

Where the subscript \( K \) indicates at constant ratio of burning surface area, \( A_{b} \), to nozzle throat area, \( A_{t} \) [m²], by the following relation (Rochford, 1999, p9):

\[ K = A_{b}/A_{t} \]

(14)

The relationship between \( \sigma_{p} \) and \( \pi_{K} \) is given by the following relation:

\[ \pi_{K} = \sigma_{p}/(1-n) \]

(15)

Figure 6. Temperature sensitivity of burning rate at various initial propellant temperatures (Rochford, 1999, p8). The figure shows that the propellant burn time, \( t_{b} \), decreases (or burn rate, \( r \), increases) with increasing initial propellant temperature. The average chamber pressure increases with increasing initial propellant temperature.
Where $n$ is the constant pressure exponent as found in equation (7) (Rochford, 1999, p9).

E. Rocket Performance Testing

Before solid propellant rocket motors are put into operational use, they are subjected to several different types of tests. These tests can include; manufacturing inspection and fabrication, component tests, static rocket system tests, static vehicle tests, and flight tests, and are generally conducted in that order (Sutton & Biblarz, 2001, p711). In terms of performance testing of a rocket motor, it is obvious that flight tests would give the most meaningful performance results; rocket motors are designed and built to experience accelerations during flight in constantly changing environmental conditions (temperature, pressure, humidity, atmospheric gas composition etc.). Sometimes it is not feasible to conduct a flight test and static tests must be implemented instead. Flight testing requires special launch support equipment, means for observing, monitoring, and recording data (cameras, radar, telemetering, etc.), equipment for assuring range safety and for reducing data and evaluating test performance, and specially trained personnel. The launch equipment has to have provisions for loading or placing the vehicle into a launch position, for allowing access of various equipment and connections to launch support equipment, for aligning or aiming the vehicle, or for withstanding the exposure to the hot rocket plume at launch. During experimental flights extensive measurements are often made on the behaviour of the various vehicle subsystems; for example, rocket propulsion parameters, such as chamber pressures, temperatures, and so on, are measured and the data are telemetered and transmitted to a ground receiving station for recording and monitoring. Some flight tests rely on salvaging and examining the test vehicle (Sutton & Biblarz, 2001, pp 724-725). Other methods of flight evaluation include photography and electronic tracking. Inboard cameras enable the study of aerodynamic effects at high speeds. Optical tracking in conjunction with cameras and radar are extensively used to provide an outboard flight record from which position, velocity and acceleration may be determined (Zaehringer, 1955, pp119). Due to the limited storage space and payload of the Estes rockets, only a minimum of basic onboard electronic components may be carried by the rocket such as an altimeter or speedometer and associated power source and circuitry (Estes Catalogue, 2006, p6). This fact suggests that flight tests are only suitable for rockets of a much larger scale than the Estes products. When it is not feasible to conduct a flight test of a given vehicle or motor, a static performance test can conducted.

A static test is defined as testing or operating a motor in a fixed, non-mobile position. Thus, the motor is usually fastened to a test stand and not allowed to move during the test. The forces, pressures, temperatures, etc., are then measured and recorded during firing. Of all the fundamental quantities of a rocket motor, thrust is rather easily and accurately produced and controlled for calibration purposes. During a thrust-measuring test the motor is held fast to the test stand, although the stand itself is free to move to allow transfer of thrust from the motor to the measuring or thrust-sensitive element (Zaehringer, 1955, p93). Force or thrust sensitive elements or transducers are of three types; electrical, hydraulic, and mechanical. The electrical type is the most versatile and widely used for rocket motor tests. In the electrical force transducer, force or thrust on a load sensitive element causes some change in its electrical properties (i.e. resistance, capacitance etc.). Because of their excellent reliability, the strain gauge type of transducer (e.g. load cells) has become very popular in rocket testing. In a hydraulic system, the thrust acts through a piston of known area and produces a known pressure which is displayed or recorded. Since pressure is force per unit area, the thrust can be determined. This type of system is characterised by simplicity, ruggedness, and low cost. However, the frequency response is poor and operating pressure and thrust-alignment are critical factors. Mechanical force systems employ the use of springs, scales, etc. For any system where the thrust changes rapidly, the use of springs and scales is questionable as the frequency response of mechanical systems is poor. A spring-type mechanical system, however, is simple and can be used to advantage for measurements not requiring high accuracy. The calibration of thrust transducers is merely a matter of applying a known force or weight to the system and noting a change in electrical, hydraulic, or mechanical response (Zaehringer, 1955, p95).

F. Instrumentation and Measurement

A thrust measuring system usually requires one or more sensing elements (transducers), a device for recording, displaying and/or indicating the thrust-time relationship, and often another device for conditioning, amplifying, correcting, or transforming the transducer signal into a form suitable for recording, indicating, display, or analysis. Recording of rocket test data has been performed in several ways, such as on magnetic tapes or disks or by computers (Sutton & Biblarz, 2001, p720). Range of a thrust measuring system refers to the region extending from the minimum to the maximum thrust the system can measure. The range is usually limited by instrument yield strength and/or the linear operating range. The linear operating range implies that the system will become
Increasingly nonlinear beyond the maximum and minimum limits. Usually an additional margin is provided to permit temporary overloads without damage to the system or need for recalibration and to ensure measurements lie in the system linear range (Tse & Morse, 1989, p54). Resolution refers to the minimum change in the measured thrust that can be detected with a given instrument.

Errors in measurements are usually of two types: (i) human errors of improperly reading the instrument, chart, or record and of improperly interpreting or correcting these data, and (ii) instrument or system errors, which usually fall into four classifications: static errors, drift errors, dynamic response errors, and hysteresis errors. Static errors are usually fixed errors due to fabrication and installation variations; these errors can usually be detected by careful calibration, and an appropriate correction can then be applied to the reading (Sutton & Biblarz, 2001, p271). Drift error is the change in output over a period of time, usually caused by random wander and environmental conditions. All instruments will drift to a degree. The long-time drift of an instrument is a part of its specifications (Tse & Morse, 1989, p47). To avoid drift error the instrument has to be calibrated at frequent intervals at standard environmental conditions against known standard reference values of its whole range (Sutton & Biblarz, 2001, p271).

Dynamic response errors occur when the measuring system fails to register the true value of the measured quantity while this quantity is changing, particularly when it is changing rapidly. For example, the thrust force has a dynamic component due to vibrations, combustion oscillations, interactions with the support structure, etc. These dynamic changes can distort or amplify the thrust reading unless the test stand structure, the rocket mounting structure, and the thrust measuring and recording system are properly designed to avoid harmonic excitation or excessive energy damping. To obtain a good dynamic response requires a careful analysis and design of the total system (Sutton & Biblarz, 2001, pp721-722). A maximum frequency response refers to the maximum frequency, usually in cycles per second or Hertz [Hz], at which the instrument system will measure true values. The natural frequency of the measuring system is usually above the maximum response frequency. Generally, a high-frequency response requires more complex and expensive instrumentation. All of the instrument system (sensing elements, modulators and recorders) must be capable of a fast response. Most of the measurements in rocket testing are made with one of two types of instruments: those made under nearly steady static conditions, where only relatively gradual changes in the quantities occur, and those made with fast transient conditions, such as rocket starting, stopping, or vibrations. This latter type of instrument has frequency responses above 200 Hz, sometimes as high as 20,000 Hz. These fast measurements are necessary to evaluate the physical phenomena of rapid transients. Linearity of the instrument refers to the ratio of the input (usually pressure, temperature, force, etc.) to the output (usually voltage, output display change, etc.) over the range of the instrument. Very often the static calibration error indicates a deviation from a truly linear response. A nonlinear response can cause appreciable errors in dynamic measurements (Sutton & Biblarz, 2001, pp721-722). Dynamic errors cannot be subtracted from the measured data because a quantitative dynamic error for measurements cannot be deduced. This is because knowing how much each type of system error contributes to the overall measurement error is impossible. The dynamic error can only be observed by applying a known test signal to the instrument and noting the deviations at the output or by comparing the output of a test instrument with that of a reference in much the same way as this thesis plans to do with the measured thrust-time plots against the thrust-time plots provided by Estes (refer to Figure 3). The observations may show the deviations but do not give a numerical value for the transient error (Tse & Morse, 1989, pp198-199).

Dead zone or hysteresis errors are often caused by energy absorption within the instrument system or play in the instrument mechanism; in part, they limit the resolution of the instrument. Sensitivity refers to the change in response or reading caused by special influences. For example, the temperature sensitivity and the acceleration sensitivity refer to the change in measured value caused by temperature and acceleration. These are usually expressed in percent change of measured value per unit of temperature or acceleration. This information can serve to correct readings to reference or standard conditions (Sutton & Biblarz, 2001, p722). Electrical interference or noise within an instrumentation system, including the power supply, transmission lines, amplifiers, and recorders, can affect the accuracy of the recorded data, especially when low-output transducers are in use (Sutton & Biblarz, 2001, p722). Electrical noise occurs as a result of magnetic fields generated by current flow in wires in close proximity to the small wires inside a transducer or the lead wires, which are wires used to connect a transducer to the instrumentation system (Dally, Riley & McConnell, 1993, p18). In some instances, the voltage induced by the magnetic field is so large that the signal-to-noise ratio becomes problematic and it is difficult to separate the noise from the transducer signal. There are three ways of maximising the signal-to-noise ratio. These are: (i) all lead wires should be twisted, (ii) only shielded cables should be used and the shields should be grounded, and (iii) differential...
amplifiers should be used to reduce noise by destructive interference of the noise signal (Dally, Riley & McConnell, 1993, pp234, 236).

III. Thesis Process and Methodology

It has been decided that the best way to achieve the thesis aims is to effectively separate the thesis into three complimentary phases. These phases are classified as follows; (1) determination and analysis of Estes rocket performance, (2) determination and analysis of the temperature sensitivity of Estes rocket performance, (3) design, construction and test of home-made rocket motor. By focusing on one phase at a time, it will be easier to manage the thesis resources effectively and to channel the efforts of the project manager to produce a higher-quality end product. Phases (2) and (3) will be highly dependent on knowledge obtained and work done in phase (1), especially the design and construction of the thrust-measuring instrument and the literature review. The instrument must be fully operational before any of the phases can commence/continue according to the thesis task timeline (Appendix 9). At this early stage of the thesis project, success is based primarily on the successful construction and operation of the thrust-measuring instrument. At the time of writing, the instrument has been designed and an initial test has been conducted to determine whether the Estes rocket motors will ignite and operate normally whilst oriented upside-down (nozzle facing up). Details of this initial work are found below.

G. Design of the Thrust-Measuring Instrument

The primary design requirements of the thrust-measuring instrument are simplicity and accuracy. The life of the thesis project is relatively short, hence the need for a simple, easy-to-construct design. A simple design will assist in the project running on schedule. The thrust produced by the Estes rocket motors is relatively small compared with typical, everyday forces, hence the need for an accurate, sensitive design. An accurate design will ensure that the required resolution in thrust measurements will be achieved.

Initial discussions with the thesis supervisors, Dr. John Milthorpe and Dr. Neil Mudford, yielded four physics principles/methods that could be applied to the instrument design to determine thrust. These methods were; (1) measuring strain in a deflecting material, (2) measuring angular displacement or angular velocity, (3) measuring the transmission of pressure, and (4) measuring axial acceleration of an unrestrained rocket motor. Table 1 displays the basic principle equations behind how the thrust (F) could be deduced from known measured values. Further to these discussions, a literature review of previous and existing rocket thrust-measuring instruments was conducted to begin formulating some ideas on the potential solution.

<table>
<thead>
<tr>
<th>Method</th>
<th>Quantities involved</th>
<th>Basic equation/s involved</th>
<th>Quantity to be measured directly</th>
<th>Transducer type</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>σ (normal stress), E (Young’s Modulus of material), ε (normal strain), A (cross-sectional area)</td>
<td>σ=εE where σ=F/A</td>
<td>ε (normal strain)</td>
<td>strain gauges, load cells</td>
</tr>
<tr>
<td>2</td>
<td>ΣM (sum of moments), I (moment of inertia), ˙θ (angular acceleration), Δx (moment arm)</td>
<td>ΣM=I ˙θ where ΣM=Σ(FΔx)</td>
<td>˙θ (angular acceleration)</td>
<td>angular displacement sensor, angular velocity sensor, light sensor, camera</td>
</tr>
<tr>
<td>3</td>
<td>P (pressure), A (area of applied pressure)</td>
<td>P=F/A</td>
<td>P (pressure)</td>
<td>pressure cell, analogue pressure gauge</td>
</tr>
<tr>
<td>4</td>
<td>m (mass), a (axial acceleration), ΣF (sum of axial forces), D (drag force)</td>
<td>ΣF=ma where ΣF=F–D</td>
<td>a (axial acceleration)</td>
<td>accelerometer, camera</td>
</tr>
</tbody>
</table>

Table 1. Basic physics principles of instrument designs. This table displays how thrust (F) can be found from other directly measurable quantities. It also displays the type of transducers required to measure these quantities.

Method (1) in Table 1 seemed the most attractive design method for the instrument due to its accuracy and relative simplicity. The idea of using strain gauges on a cantilever beam to measure beam deflection was prompted by Richard Nakka’s Amateur Experimental Rocketry website (Nakka, viewed 21 March 08). Nakka maintained that a small cantilever beam fitted with four strain gauges would be deflected by the thrust from a rocket motor and the
strain gauges would undergo a change in electrical resistance. This deflection would be converted into an electronic signal that would be collected and processed by a computer. Providing the electrical signal was sampled at a high enough rate over the burn time of the rocket motor, the thrust-time relationship could be calculated using the principles listed in Table 1. Another deflection/strain type instrument that was reviewed was similar but slightly more complicated than Nakka’s design. Mungas, Das and Kulkarni used a deforming-rectangle type instrument rigged with multiple strain gages as seen in Figure 7. This design was quickly rejected due to its relatively complicated manufacture given the tight schedule of this thesis project.

In an attempt to keep the design of the instrument very simple, the idea of a simple cantilever beam with two strain gages, one on top of the beam measuring tension and one on the underside of the beam measuring compression, with the rocket motor mounted at the tip of the beam was explored further. The design was analysed mathematically in Appendix 1 to determine whether this design idea could possibly work. Many different beam dimensions and properties were analysed to determine the optimum beam design. After much analysis and discussion it was determined that, due to the fact that the beam has to deflect a measurable amount, there would be too great a lag-time in the beam response to keep up with the quick motor thrust variations and that the beam would vibrate close to its natural frequency giving erratic results (as seen in Figure 8). Damping was identified as a possible solution to these problems; however, a damping system would add much unwanted complexity to the instrument. It is also likely that the strain gauges would take too much effort and time to calibrate according to advice from Mr. Andrew Roberts, ACME Electronic Support.

Method (2) in Table 1 was explored in greater detail to determine whether it could be applied to a viable instrument design. The idea of measuring angular displacement or angular velocity of a pendulum or spinning wheel and differentiating the results to find angular acceleration, hence thrust, was prompted by a journal article on double-base propellant rockets (Kentgens, Mackowiak & Schoffl, 1996, pp122-123). In this article, a pendulum test rig was used to test a 1000 kN double-base, short-action, solid propellant rocket motor (as seen in Figure 9). The motor was mounted at the end of a 2500 kg pendulum used to provide enough inertia against the thrust force to ensure the pendulum did not over swing. Since the burning time of the motor was very short (approx. 10 milliseconds), the kinetic energy of the pendulum at burnout was almost entirely converted into potential energy being represented by the maximum amplitude of the first pendulum swing. The thrust-time relationship was derived from this principle (Kentgens, Mackowiak & Schoffl, 1996, pp122-123). After much discussion and consideration, this type of instrument could not be directly applied to the Estes motors due to the fact that the Estes motors burn for a much longer time.
longer time (approx. 2 seconds as seen in Figure 3) with a non-constant, decreasing thrust. This would cause the pendulum to oscillate; not only due to the thrust variation over time, but also due to the natural frequency of the pendulum (Inman, 2001, pp31-34). A variation of the pendulum design that was considered and analysed (refer to Appendix 2) consisted of mounting the motor tangentially on the rim of a free-spinning wheel and measuring the angular displacement or angular velocity using a sensor. As described in Appendix 2, many issues were identified for this design; however, a foreseeable solution could be identified for most issues. The issues that caused us to neglect this instrument design were that; (i) there would be moving parts, hence, a large potential for errors, frictional losses, and safety violations, (ii) the design would be relatively complicated and difficult to manufacture, and (iii) the results would be noisy due to the need to differentiate angular measurements at least once to obtain thrust.

Method (3) in Table 1 was explored in greater detail to determine whether it could be applied to a viable instrument design. The idea was prompted by Dr. Neil Mudford in one of our thesis discussions. He suggested that the relatively incompressible nature of a fluid could be utilized to transmit the pressure applied to the fluid by the rocket motor to a pressure transducer or pressure gauge without any response delays. An internet search was conducted to explore this idea further and a simple yet effective instrument design was discovered. The instrument was developed by an anonymous amateur rocket enthusiast (texnet, viewed 20 March 08). This enthusiast developed a design that utilizes a syringe to transmit the pressure applied by the rocket motor to an analogue pressure gauge. During the firing of the rocket motor, the gauge is filmed using a video camera with a sufficient recording rate (image frames per second) to determine the thrust-time relationship using the principles detailed in method (2) of Table 1. A basic representation of the instrument can be seen in Figure 10. A basic analysis using calculations of the syringe size required for the C6-0 and D12-0 Estes motors can be found in Appendix 3. This design idea was eventually neglected due to its crudeness and potential for inaccuracy. Friction in the rubber seal of the syringe’s plunger and friction between the motor casing and the motor holder would certainly be a problem. The analogue pressure gauge may also produce some errors as there may be some fluid leakage.

Method (4) in Table 1 was explored in greater detail to determine whether it could be applied to a viable instrument design. The thrust-time relationship of a rocket motor can be determined by measuring the axial acceleration of the rocket over its flight trajectory using accelerometers or optical tracking. Many rocket tests are conducted in this fashion, albeit mainly for large scale rockets only. The advantages of such a test method are that it exposes the motor propellant grain to the accelerations and ambient temperatures and pressures experienced by the motor in actual flight. These factors do affect the performance of the motor as detailed in the background section. The disadvantages of this method include the need for an appropriate firing range, the need to calculate/measure the drag force on the rocket and the complication in measuring the axial acceleration of the rocket created by the thrust. One method of measuring the acceleration would be to install an accelerometer and associated equipment inside the rocket. This would not be a practical method for the Estes rockets due to the significant limitations in the rocket’s payload as displayed in the Estes Catalogue, 2006, p35. As discussed with Dr. John Milthorpe, when using optical

Figure 9. Pendulum test rig (Kentgens, Mackowiak & Schoffl, 1996, p123). The amplitude of the first pendulum swing is measured to determine thrust. This works for short-action motors only because the propellant has burnt out entirely before the first swing has peaked.

Figure 10. Pressure instrument (texnet, viewed 20 March 08). This instrument converts the motor thrust into fluid pressure that is measured and recorded at an analogue pressure gauge using a video camera.
tracking at least three optical tracking devices, such as high-resolution cameras with known elevation angles (radar is too expensive for this thesis project), spaced known distances and angles apart would be required to triangulate the position or velocity of the rocket as it streaks across the sky. The position/velocity data would be differentiated to obtain the axial acceleration of the rocket. Method (4) for the instrument design was neglected due to the inaccuracy and complication in measuring the axial acceleration of a fast-moving rocket.

Method (1) in Table 1 was revisited once it was discovered that suitable load cells (Xtran) were available for this thesis project from SACME. The advantage of using a load cell over strain gauges directly applied to a deflecting cantilever beam is that the load cells are designed to deflect a minimal amount, such that beam excitation and delayed response would not be an issue. Given that the two load cells available to the thesis are rated at 100 Newtons and 250 Newtons maximum and that the average thrust of the C6-0 motor is 6 Newtons, it was identified that the load cell may not have enough resolution to capture the exact detail of the Estes thrust-time relationships. According to the Xtran load cell manuals, the load cells have a resolution of 0.1% of the load applied. This should be sufficient; however, to be certain that the required resolution is achieved; the instrument design utilizes a hinged, rigid, steel cantilever beam to amplify the thrust of the rocket motor at the load cell position along the beam by creating a moment arm. The cantilever beam was designed to be very stiff and rigid to prevent significant deflection in the instrument (refer to Appendix 4) but also to ensure that the natural frequency of the beam is much larger than the frequency of the motor (refer to Appendix 1 Spreadsheet). The mass of the beam was monitored during its design to ensure that the weight of the beam resting on the load cell plus the maximum thrust of the larger D12-0 motor did not exceed the design load of the load cell. As can be seen in Appendix 4, a factor of safety of 1.5 was factored into the calculations to ensure the load cell is not overstressed even if the thrust spikes at a larger than expected maximum thrust. Another advantage of the beam is that it separates the motor and load cell such that if the motor exploded, the load cell should be undamaged. It was deemed that the friction in the hinge of the beam would be negligible relative to the forces to be measured by the load cell. Because there were no major disadvantages for this design identified and that the design is easy to construct and operate, it was selected as the best thrust-measuring instrument design after many discussions with the thesis supervisors. As mentioned above, a rough mock-up construction of the instrument will be constructed to ensure that there are no unexpected design flaws before the final instrument is produced. A sketch of the rough mock-up design can be found in Appendix 4.

H. Initial Rocket Motor Tests

The Estes motor types selected for the first two phases of this thesis are the C6-0 and D12-0 motors. These motors were chosen because they; (i) produce a sufficiently high enough thrust to ensure that typical transducers could measure the required resolution in order to produce accurate thrust-time plots, (ii) do not have a time delay or ejection charge that produces a burst of thrust in the opposite direction to the nozzle (Estes Catalogue, 2006, p34), (iii) are relatively cheap allowing for many test firings of the motors to achieve valid results (refer to Appendix 10), and (iv) are small enough to mitigate addressing certain risks that are usually associated with explosives and large rocket motors (refer to Appendix 11). Test firings of two C6-0 motors were conducted to ensure that the motors would ignite and burn correctly in the downwards-facing orientation as they are to be located on the thrust-measuring instrument design selected for construction. This test was considered appropriate because Estes only designed the motors to be ignited and operated in an upwards-facing orientation. The test was conducted at 1545 h on Tuesday, 15 April 2008. An aluminium case with one capped-end was constructed for the motor to sit in while firing such that the motor was unrestrained at the nozzle end (as seen in Figure 11). The aluminium case was held in the vice located outside the ACME main workshop. The aluminium case provided a barrier between the vice jaws and the motor to ensure that the propellant grain was not cracked due to the stress applied by the vice. Cracks in the grain of a solid propellant motor cause the burning rate to increase dramatically due to the larger exposed grain surface. If the burning rate increases too dramatically, the motor will explode (Rao, 1992, p456).
The motors ignited and operated correctly (as seen in Figure 12). It was expected that, even though the motors lacked an ejection charge, they would remain in the aluminium case during the entire test as the capped-end of the aluminium case provided a reaction force to the motor thrust to prevent any acceleration in the downward direction. Unfortunately, this was not the case. Unexpectedly, once the burn wave propagated all the way through the grain to the clay retaining cap protecting the top surface of the grain inside the motor case, the high chamber gas pressure burst through the clay retaining cap creating a sudden imbalance of pressure above and below the motor case. Because there was nothing retaining the nozzle end of the motor, the higher pressure acting on the underside surfaces of the motor case caused the motor to accelerate upwards, out of the aluminium case as seen in Figure 13. The motor shot up approx 20 metres and landed, without incident, close to the vice. Assuming that the use of the C6-0 and D12-0 motors is continued during this thesis, provided the experiments are conducted outside with nothing located directly above the motor, this phenomenon should not be a safety concern. The device could be designed to restrain the motor in the upwards direction as well. Estes also produce rocket motors that have a fully plugged end to prevent this phenomenon from happening. These motors are designated D11-P and E9-P (Estes Catalogue, 2006, p35). Their use in this thesis might be considered. After the motor firing, a ‘rotten-egg’ smell was noticeable. This suggests that hydrogen sulfide (H$_2$S) is indeed a product of the blackpowder combustion as detailed in the background section of this report.

I. Future Thesis Development

Phase (1) of the thesis project will continue with the construction of a rough mock-up of the selected instrument design. At least one rocket motor thrust-time measurement will be conducted to determine whether the instrument will give reasonable results to compare with the accepted results provided by Estes. It is likely that during the analysis of the selected instrument design, potential design flaws were not concerned. If the instrument produces acceptable results that represent those provided by Estes, a refined version of the instrument will be constructed and calibrated for further experiments. Once the refined instrument is built, an experiment procedure will be designed to obtain valid performance data for the C6-0 and D12-0 Estes motors at ambient temperatures. This experiment procedure will detail information such as; (i) the method to be used, (ii) how many of each type of motor must be fired to obtain valid results, and (iii) how the results should be interpreted. Once the procedure has been written, timings to conduct the rocket motor tests will be organised with the appropriate stakeholders. Once the results from these tests have been gathered, interpreted, and compared with the data provided by Estes, phase (2) of the thesis will commence. The temperature sensitivity analysis procedure will be designed and written.

The temperature sensitivity analysis procedure will detail any expectations on the grain initial temperature dependence of motor performance, as well as a structured plan designed to give the best indication of the affect of initial grain temperature on propellant burning-rates; hence, the thrust-time relationship (impulse) of the motors. Individual motor samples will be either cooled or heated to temperatures within the recommended storage temperature range provided by Estes (Estes Model Rocket Engine and Igniter Instructions, 2001-2003). The motors will be given sufficient time in the heated/cooled environment to ensure that the propellant has reached the desired temperature. The motors will then be fired on the thrust-measuring instrument immediately after they have been removed from the controlled environment. The results will be interpreted and hopefully a thrust-time-temperature relationship can be determined for blackpowder type rocket motors. Phase (3) of the thesis is to design, construct and build a home-made rocket motor using instructions from Sleeter (Sleeter, 2004) and Nakka (Nakka, viewed 21 March 08). The rocket motor will be designed to produce a similar maximum thrust to the Estes products used for
this thesis; however, the home-made motor case is to be reusable in order to make it an economically viable option to the commercially-produced motors. Everything learnt in the two previous phases of the thesis will be applied to produce this rocket motor. If the measured performance of the rocket motor matches the theoretical performance of the rocket design, success will have been achieved.

IV. Summary

Most of the required solid propellant rocket motor theory and theory on instrumentation for the thesis project is contained in this initial thesis report. Applying this theory to the three phases of the thesis project should yield interesting and significant results. The thesis has its compulsory aims and its desirable aims as detailed in the Client Brief (Appendix 5). The chosen topic allows for much scope in formulating new ideas/aims further into the project’s life. It is flexible enough to delve further into the engineering behind solid propellant rocketry than the already-proposed phases of the thesis project.

References

School of Aeronautics and Astronautics, Purdue University, “Solid Rocket Propulsion,” Google Search, URL: https://engineering.purdue.edu/AE/Research/Propulsion/Info/rockets/solids/ [viewed 27 April 2008].

Appendices

Appendix 1 — Analysis of Cantilever Beam with Strain Gauges Design
Appendix 2 — Analysis of Spinning Wheel and Pendulum Design
Appendix 3 — Analysis of Pressure Instrument Design
Appendix 4 — Analysis of Hinged-Cantilever Beam Design

Initial Thesis Report 2008, UNSW@ADFA
Appendix 5 — Client Brief
Appendix 6 — Task Breakdown Structure
Appendix 7 — Task Outlines
Appendix 8 — Milestone Chart
Appendix 9 — Gantt Chart
Appendix 10 — Costing
Appendix 11 — Risk Assessment (Phase 1)