Chapter 1

INTRODUCTION

1.1 History

Solid fuel rocket motors are used for many applications due to their high reliability and simplicity than liquid propellant rocket motors. Solid fuel used for rocket motors are known as propellant. The special shaped mass of propellant is known as propellant grain. Solid propellant grain is designed according to the mission profile of the system. The mission profile depends upon the burning characteristics of the propellant such as burn rate, pressure, thrust, specific impulse, characteristic velocity and the grain geometry etc.

Various grain shapes including star, cylindrical, wagon wheel, dog bone etc. are used to establish their characteristics. These single grain shapes are used to obtain one level of thrust like progressive, neutral, regressive thrust according to the mission requirement.

In order to obtain duel level of thrust, a single grain is design in which two shapes are combined like finocyl, conocyl etc. In finocyl grain a star shape at the aft end and a circular shape at the head end in a single grain is used to obtain first progressive thrust and then neutral thrust. So boost and sustain is obtained from single grain.

Configuration that combine both radial and longitudinal burning are referred to as "three dimensional grains" grains although all grains are geometrically 3-D. but the grains that burns only longitudinally or radially are known as "two dimensional grains". Grain configuration can be classified according to their web fraction, volumetric loading fraction and their length to diameter ratio [1].

Propellant can be classified according to their chemical composition and by their method of casting also.

By chemical composition propellants are mainly divided into two types, composite propellants and double base propellants. Double base propellants are homogeneous mixture of fuel and oxidizer; it is made up of two base propellants. Each base propellant can be used separately as fuel in rocket motors for example nitroglycerin (NG) and nitrocellulose (NC). Composite propellants are heterogeneous mixture of fuel and oxidizer with some binder like HTPB (hydrxi terminated polybutadiene). Plasticizers, curing agents and phase stablizers are also added and in some cases some metal contents are also used like Al or Mg. Additives control the burn rate of the propellants

Other two types of propellants are also very important nowadays such as modified double base (MDB) and composite modified double base (CMDB). MDB propellants contain double base propellant with some amount of nitro amines such as RDX, HMX etc. which increases its density and specific impulse and CMDB propellants contain are formed by adding composite materials like solid AP , Al in DB. It increases the specific impulse and density of DB. These four types of solid propellants will be under consideration in this study.

Two types of methods are suggested to cast propellant grains, first in which propellant is cast directly into the case having moulds of desired shape and left to cure for some time after curing moulds are removed known as case bonded grains, second in which it is cast into different moulds separately and then placed into the case for use known as free standing grains. First is preferred now a day due to better loading fraction.

Different work has been performed on solid rocket motor propellant grains as the results of 3-D grain burn back analysis of finocyl grain geometries and simulation are presented by Püskülcü and Ulas [12]. Internal ballistic simulation using minimum distance function approach was conducted by Michael et al. [13]. Ballistic evaluation of star port propellant grain is done by Himanshu Shekhar [14].

1.2 Performance Prediction

Performance of solid rocket motor can be predicted if the burn back analysis of propellant grain has been performed. There are various parameters which are used to predict the performance of solid rocket motors.

- Propellant mass fraction/mole fraction
- Specific impulse
- Thrust
- Adiabatic flame temperature
- Characteristic velocity, c*

- Ideal exhaust velocity
- Time

1.3 Research Contents and Technical approach

Following research methodology will be followed

- To study and comprehend all the 2-D and 3-D grain configuration and compare their burning characteristics
- Then choose one configuration according to mission requirement to obtain boost-sustain-boost (thrust-time) profile
- Solid modeling of desired propellant grain will be achieved using Pro Engineer Wild fire 4.0 software
- Using parametric modeling solid model with new configuration at each burn step will be created.
- This solid modeling of propellant grain will provide performance parameters such as propellant mass, thrust, burn area , pressure, burn rate etc.
- Ballistic parameters for all grains are calculated using four types of propellants such as DB, MDB, CMDB, Composite propellants
- Tec plot will be used for plotting different curves such as Thrust /time, Pressure /time, Burn Area / time, burn area/web burn
- Performance comparison among all these four types of propellants will be conducted.
- Propellant grain for JPL- nozzle is designed and performance results are compared for these four (DB, MDB, CMDB, Composite) propellants

Chapter 2

FANDAMENTALS OF ROCKET MOTORS

Space travel is amazing and complex, because there are so many obstacles to overcome. Modern war weapon systems require some propulsion system to travel. The propelling system must have long service life, refueling must be easy and must be able to run for the complete operation. For complex weapon system (like airplanes) gas turbine engines or internal combustion engine are used as propelling system having lot of moving parts, low power to weight ratio and are difficult to maintain. For simpler weapon system (like artillery rockets) simpler, cheaper and maintenance free propulsion system are required. They use single shot operating system, in which payload reaches to desired location where propulsion system (rocket motors) explodes.

A rocket motor is typically energy transfer system, in which the chemical energy inside the rocket motor is converted into thermal energy by combustion processes. The hot gaseous products exhaust through nozzle and produces thrust [4].

Rocket motors contain their fuel and oxidizer within itself during flight, as they need no surrounding air for oxidation of fuel. Rocket motors can work in space and below the surface so they are preferred for space craft propulsion systems.

2.1 Types of Rocket Motors

Rocket motors can be classified into three types depending upon their physical state of fuel

1. Solid rocket motors
2. Liquid rocket motors
3. Hybrid rocket motors
4. Gaseous rocket motors

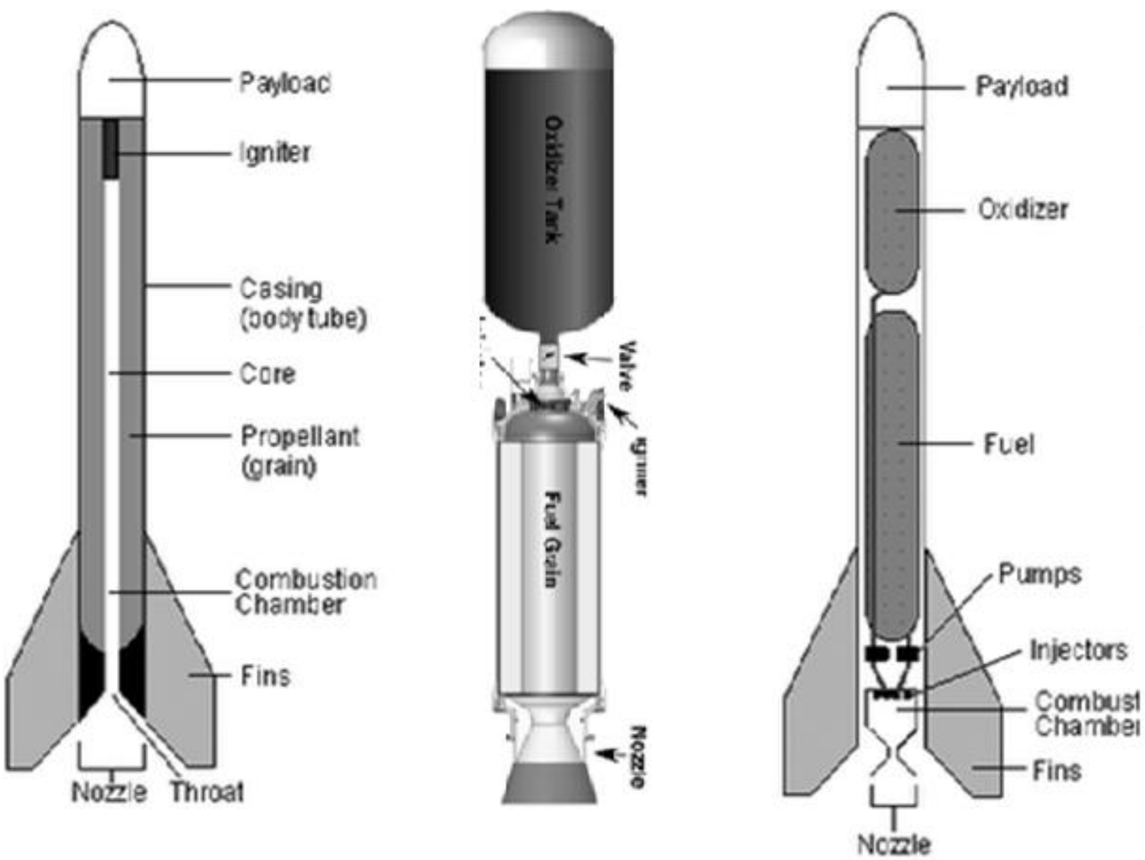


Figure 2.1 Types of rocket motors

2.1.1 Solid rocket motors

Solid rocket motors are the first type of rocket motors powered by gunpowder, which has wide range of operations as in 13th century Chinese used in warfare and after that Mongols, Arabs and Indians used it, they are used in military operations (missiles) [3]. As clear from the name in solid rocket motor fuel and oxidizer are in solid form. Solid rocket motors consist of chamber or casing, nozzle, insulation, igniters, propellant grain. Shaped mass of propellants is known as propellants grains and it contains all the chemical elements for complete combustion. Propellant grains are of different shapes such as star grain, wagon wheel, cylindrical grain etc. There are two methods in which propellant is cast in to the chamber; one is by pouring propellant material directly into the chamber having mandrel of desired shape, second by casting propellant separately out of the chamber and cast propellant is placed into the chamber. Nowadays first one method is preferred.

Premixed mixture of fuel and oxidizer are placed in chamber of rocket motor directly Solid material can be stored for long time (5 to 20 years) in the chamber. Once propellant is ignited in the combustion chamber, propellant burns usually smoothly in the at predetermined web steps from all its exposed surfaces and internal cavity of the propellant grain increases as the propellant mass is consumed through burning. During propellant combustion hot gases are produced and are exhaust through supersonic nozzle and produce thrust.

Motors are available in different types and sizes varying from 2 Newton to over few million Newton’s. In solid rocket motor no moving parts are required only some motors use moving nozzle and actuators for vectoring the line of thrust relative to motor axis. They are easy to use and require no maintenance.

2.1.2 Liquid Rocket Motors

Liquid rocket motors are known as liquid rocket engines. Liquid rocket engines use liquid propellants that are fed into the combustion chamber under pressure. Liquid rocket engines are of two types:

- Monopropellant rocket engines
- Bi-propellant rocket engines

Monopropellant rocket engines consist of a single liquid that contain both oxidizer and fuel in it and upon proper cartelization produce hot gases. Bi-propellants consist of separate liquid fuel and liquid oxidizer both are pumped in to the combustion chamber. In space launch vehicle pump feed liquid feed systems are used with larger amount of propellants.

Both oxidizer and fuel are fed into the combustion chamber where they react and produce hot gases and then they are accelerated and exhaust through supersonic nozzle with a high velocity, where nozzle consist of a converging section, diverging section and throat area is present between these two sections.

These hot gases exhaust through the nozzle and produce thrust. Liquid rocket engines produce higher thrust than solid rocket motors using the same amount of propellants. Liquid rocket engine has on/off thrust controlling system. Supplying the fixed amount of fuel and oxidizer to chamber is very expensive. Liquid rocket motors require moving parts such as fuel pump, oxidizer pump, cooling turbines. High maintenance is required for these parts so, they are mostly used for system where complex and high propulsion system is required.

2.1.3 Gaseous rocket motors

Least used type of rocket motor is gaseous rocket motors which are similar to liquid rocket motors. Gaseous rocket motors use gaseous fuel and oxidizer. Density of gaseous propellant is much lower than liquid propellants so storing same amount of propellant require bigger tanks which is the main disadvantage of this motor.

2.1.4 Hybrid Rocket motors

Hybrid rocket motors are combination of solid and liquid rocket motors. Hybrid rocket motors use solid fuel and liquid oxidizer. During motor operation the liquid oxidizer is injected on solid fuel into the combustion chamber and oxidizes the solid fuel and produces the hot gases which exhaust through nozzle and produce thrust. Hybrid rocket motors handling and storage is easier than solid rocket motors.

An example of hybrid rocket motor is a pressurized liquid propellant system that uses a solid propellant to generate hot gases for tank pressurization, flexible diaphragms are necessary to separate hot gases and the reactive liquid propellant in the tank. [1]

2.2 Components of Solid Rocket Motors

Solid rocket motors are the most widely used rocket motors because of their high reliability, simplicity, cheapness, low maintenance requirements and produces high level of thrust like millions of Newton’s.

Solid rocket motor consists of several typical components like combustion chamber, igniter, casing, insulation, converging diverging nozzle, burning material known as solid propellant. Simple solid rocket motor is shown in figure2.2 [3]

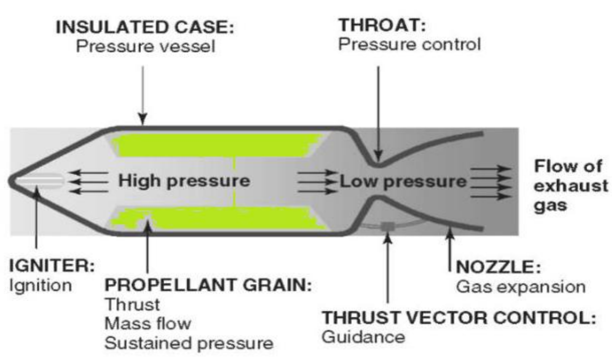


Figure 2.2 Components of solid rocket motor

Overview of major components is as follows:

2.2.1 Casing

Casing provide the basic structure of solid rocket motor, it carry out two purposes first: it holds all parts of solid rocket motor including nozzle insulation, propellant and payload together, second it act as a combustion chamber in which chemical reaction takes place as propellant ignites and the produced hot gases provide thrust to payload. Casing is internally insulated to prevent from heat loses due to combustion gases and detrimental heat affects. . Therefore it is made up of metal (high resistance steels or high strength aluminum alloys) or from composite materials (glass, carbon)

2.2.2 Nozzle

Nozzle design determines how much of the total energy is converted into kinetic energy, therefore it has critical importance in rocket motors. Rocket motors use the converging diverging nozzle in order to get greater thrust from combustion gases. The first section of the rocket nozzle is the converging section (subsonic section) through which the nozzle smoothly approaches the throat section where the velocity of hot gases becomes maximum having Mach # 1. After the throat the diverging section starts known as supersonic section in which the rapidly moving gases expands and the energy in the form of temperature and pressure is converted into kinematic energy. The gases leaving the divergent section have velocity much higher than the speed of sound.

In short rocket motors converts energy from high temperature and pressure to inertial energy in the form of extremely high velocities to propel rocket into space and beyond [5].

2.2.3 Igniter

Igniter provides high heat energy to the propellant surface to initiate combustion. It usually starts by electrical source. Igniters are made of high energy releasing materials mainly consist of black powder, pyrotechnic materials or a plasma charges. At the same time igniter is usually design to produce some initial pressure increase in the motor to assure more reproducible start up. Solid rocket motors are usually ignited by one shot pyrotechnic devices. Rocket chambers are self-sustaining so only once ignited. Overall time required by the solid propellant to ignite after receiving initiation signal is called ignition lag or ignition delay

2.2.4 Insulation

Insulating material is provided to prevent motor case, nozzle and igniter hardware material from exposure to hot gases. Insulating materials are basically temperature resistant and low conductivity materials [6]. Most commonly used insulating material is EPDM (Ethylene Propylene Diene Monomer) with addition of some reinforcing materials.

2.2.5 Propellant

In solid rocket motor fuel-oxidizer mixture is used as an energy producing source known as propellant which is in solid form. Specific shaped mass of propellant is known as propellant grain. In SRM the fuel and oxidizer together with little amount of binder or additives are combined in a mixture. Solid propellant is stable at ambient conditions and can be stored for long time.

Propellant can be categorized into two types one by their method of casting and the other by their chemical compositionCase bonded propellant are prepared in the combustion chamber by directly pouring the mixture with mandrel of desired shape in the case and then baked for some time after casting, the mandrel is removed and the propellant gets the desired shape leaving the cavity required for motor operation.

Free standing propellant are prepared as case bonded propellant but only difference is that instead of pouring into the cases they are poured into molds containing the desired shape and then finally installed in the combustion chamber. This type of production method is used to obtain larger thrust for shorter time [4].

Depending upon chemical composition the propellant are divided into two groups:

Homogeneous propellants and heterogeneous propellants.

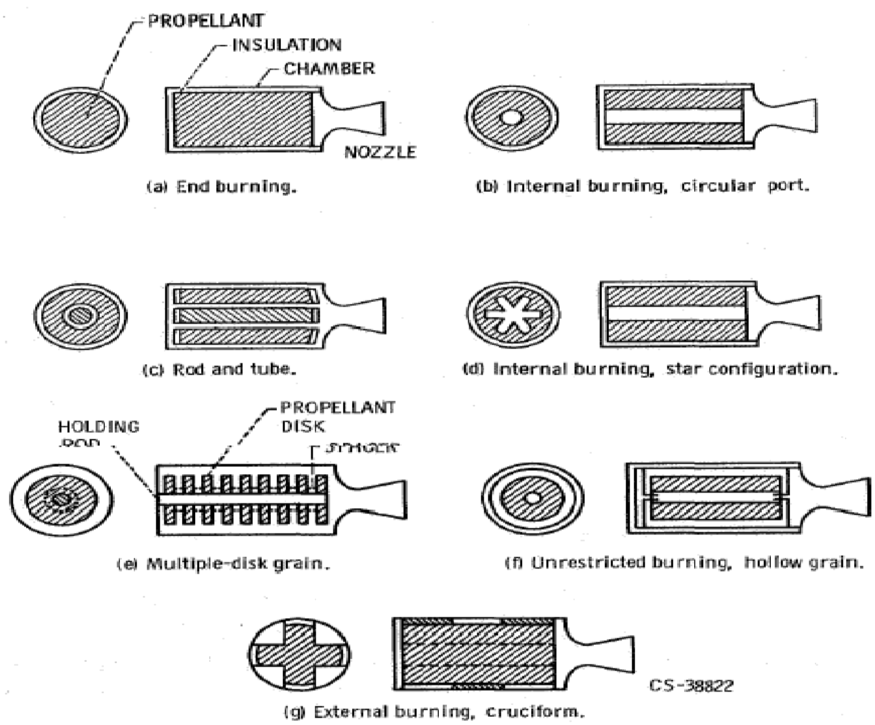
In homogeneous propellants fuel and oxidizer are part of the same molecular chain like hydrocarbon. Double base propellant are homogeneous propellants, it is made by the two energetic materials nitroglycerin (NG) and nitrocellulose (NC). Nitrocellulose is the dominant substance in double base propellants it provides high strength to grain and nitrocellulose gives high performance and fast burning. Both nitrocellulose and nitroglycerin have their fuel and oxidizer in it so they can be used alone as rocket propellant. Some binders and plasticizers are also added to the mixture during production stage. Some time some additives like aluminum powder (Al) to obtain extra energy and ammonium per chlorate (NH4ClO4) as external oxidizer is added [4].Double base propellants are used for small rockets motors with less complex grain geometries, giving high thrust for short time period.

Composite propellants are heterogeneous in nature as they consist of fuel and oxidizer of different substance. They are most commonly used type of propellant. They have good burning stability. Most commonly used composite propellant mixture consist of aluminum (Al) /hydroxyl terminated poly butadiene (HTPB) / ammonium perchalorate (NH4ClO4). None of the materials (AP, Al, HTPB etc.) in the composite propellant are used itself as propellant.

Composite modified double base is formed by adding composite propellants such as AP, Al etc. to double base propellants. In elasto-modified double base propellants an elastomeric binder (like rubber) is added to DB, which improves the physical properties and allows more nitramine and thus improves the performance of cast modified DB slightly.

Composite modified double base propellants are formed by adding composite materials like solid AP, Al in DB. It increases the specific impulse and density of DB. Modified composite propellants are formed from composite propellants by adding energetic nitramine. It increases the density and performance of composite propellants.

Propellant can also be classified by exposed burning surfaces shapes and by the method of burning. More attention has been devoted to grain shape design. Different shapes of grains are available like circular, star, wagon wheel, bone char, rod and tube, finocyl etc. Shapes of propellant grains are chosen according to the mission requirement as all have different thrusts vs. time profiles. Different grains with their burning method are shown in figure 2.3[1]



Figurer 2.3 Simplified diagrams of some grain configuration

Most solid rocket motors use single grain to obtain one level of thrust like progressive, regressive or neutral but according to mission requirement solid rocket motors use two shapes at a time to obtain dual level of thrust like finocyl grain in which two shapes, cylindrical and star shapes are used to obtain progressive and neutral thrust within the same motor.

Thrust/pressure vs. time profiles of grain are shown in Figure 2.4[6]

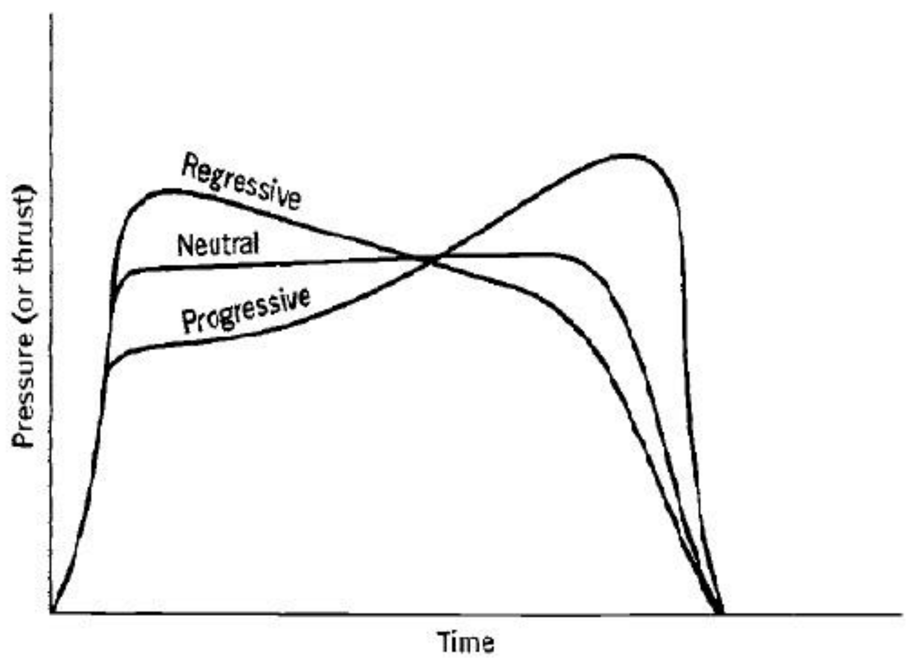


Figure 2.4: Classification of grain according to their pressure time curve

Chapter 3

BALLISTIC PARAMETER AND GRAIN DESIGN

3.1 Basic Terminologies

Perforation: The central cavity port or flow passage of a propellant grain

Slivers: Unburned mass of propellant is known as silvers

Progressive burning: In which thrust, pressure and burning surface area increases with time

Regressive burning: In which thrust, pressure and burning surface area decreases with time

Neutral burning: In which thrust, pressure and burning surface area remain approximately constant with time

Web: Largest distance that the burning surface will travel

Web thickness: the minimum thickness of the grain from the initial burning surface to the chamber insulated wall

Wed fraction: Ratio of the web thickness to the outer radius of the grain

Web steps: Burn distances are typically referred to as web steps

3.2 Ballistic Parameters

The rocket motor's operation and design depends on the combustion characteristics of the propellant, its chamber pressure, propellant burn rate, burning surface, grain geometry etc known as ballistic parameters and the branch of science that study these parameters are known as ballistic engineering [1]. Every kind of propellant has different ballistic behavior and mechanical properties. Brief introduction of each of these parameters is discussed here

3.2.1 Chamber pressure

The pressure inside the case of solid rocket motor due to the combustion of propellant is known as chamber pressure. Chamber pressure depends on the exposed surface area of the propellant, as surface area increases the chamber pressure increases. Chamber pressure is calculated by the relation given

$$P_c = \left[\frac{a\rho_p A_b C^*}{A_{th}}\right]^{\left(1/1-n\right)} \tag{1}$$

Where "a" and 'n' are empirical constants and remained almost constant for large pressure ranges. Characteristics velocity "C*", throat area "A_{th}" and propellant density remains constant for the desired propellant and chamber pressure depends directly on the burning surface area "A_b". Pressure decreases with the increase in altitude [1]

3.2.2 Burn rate

Burn rate of the propellant is the defined as the amount of material regresses per unit time or the rate at which combustion gases are generated per unit time. Propellant only burns through its exposed surfaces producing large amount of hot gases. Burn rate of propellant is calculated by the relation:

$$r_b = aP^n \tag{2}$$

Where 'a' and 'n' are empirical constants and can be calculated through strand burner tests. 'a' is known as temperature coefficient and 'n' is known as pressure exponent. Values of pressure exponent is 0.3 < n < 0.7. Pc is the chamber pressure. This equation shows that burn rate directly depends upon chamber pressure.

3.2.2.1 Burn rate relation with temperature

Temperature affect the chemical reaction rate and initial temperature of propellant grain, effect its burn rate, an uneven heating of grain also influences the burn rate of the propellant [1]. Temperature coefficient expresses the burn rate sensitivity to propellant temperature. Temperature coefficient can be calculated by the relation

$$\sigma_p = \left(\frac{\delta \ln r}{\delta T}\right)_p = \frac{1}{r}\left(\frac{\delta r}{\delta T}\right)_p \quad (3(a))$$

$$\pi_k = \left(\frac{\delta \ln p}{\delta T}\right)_k = \frac{1}{p_1}\left(\frac{\delta p}{\delta T}\right)_k \quad (3(b))$$

Equation (3(a)) describes the percent change of burning rate per degree change of propellant temperature at particular value of chamber pressure. Equation (3(b)) expresses percent change of chamber pressure per degree change in propellant temperature for particular value of K, as K is area ratio between burning surfaces to throat area [1].

3.2.3 Mass flow rate

Mass flow rate is calculated by using equation

$$m = \rho\, r_b A_b \quad (4)$$

Where ρ is the density of the propellant material and r_b is the propellant burn rate and A_b is the grain burn area. It is expressed in kg/s or lb/s.

3.2.4 Thrust

Thrust is the force that is produce by the rocket propulsion system [1]. As propulsion system works on the Newton third law of motion, as the propellant burns inside the rocket chamber the hot gases are injected through the nozzle with high velocity producing force/thrust that pushes the rocket motor upward. Thrust is calculated by the relation:

$$F = C_f P_c A_{th} \tag{5}$$

Where "C_f" is the thrust coefficient. Pc is the chamber pressure and A_{th} is the throat area of the nozzle.

Thrust coefficient can be calculated by the relation:

$$C_f = \sqrt{2\,\frac{\gamma^2}{(\gamma-1)}\left[\frac{2}{(\gamma+1)}\right]^{\frac{(\gamma+1)}{\gamma}}\left[1-\left[\frac{P_e}{P_c}\right]^{\frac{(\gamma-1)}{\gamma}}\right]} + \frac{(P_e-P_a)}{P_c}\left[\frac{A_e}{A_t}\right] \tag{6}$$

Thrust coefficient depends upon the specific heat ratio, nozzle expansion ratio and pressure ratios. Thrust coefficient gives the efficiency of nozzle for given propellant and nozzle geometry [4].

3.2.5 Characteristic velocity

Characteristic velocity depends upon propellant characteristics and combustion chamber design and is independent of nozzle characteristics. Characteristic velocity is used to compare the performance of two or more rocket system designs and also used to predict the propellant performance. It can be calculated by measured chamber pressure, throat area and mass flow rate of propellant as by the relation given.

$$C^* = \frac{P_c A_t}{m} \quad (7)$$

it can also be calculated by the relation

$$C^* = \frac{\sqrt{\gamma R T}}{\gamma \sqrt{\left[\frac{2}{\gamma+1}\right]^{\frac{(\gamma+1)}{(\gamma-1)}}}} \quad (8)$$

Where R is universal gas constant, gamma "γ" is ratio of specific heats and "T" is the absolute temperature. These all factors are properties of combustion gases.

3.2.6 Specific and total impulse

It represents the impulse (change in momentum) per unit amount of propellant used. If the "per unit amount of propellant" is given in terms of per unit mass (such as kilograms), then specific impulse has units of velocity. If the impulse is given per unit Earth-weight (such as kilopond's) of propellant, then the measure of specific impulse has units of time [7]. Specific impulse can be described as the ratio of motor thrust to mass flow rate. It is calculated by the relation

$$I_{sp} = \frac{C^* C_F}{g_0} = \frac{F}{m g_0} \quad (9)$$

Total impulse is the thrust integrated over burning time. This parameter is used to measure the efficiency of rocket motors. It is calculated by the relation

$$\text{Total Impulse} = \int F \, dt \quad (10)$$

3.2.7 Actual exhaust velocity

Exhaust velocity is defined as the velocity of the exhaust gases through nozzle after propellant combustion.

$$Ve^2 = \gamma \; R_g \; T_c \frac{\left[1 - \left(\frac{P_e}{P_c}\right)^{(\gamma-1)/\gamma}\right]}{(\gamma - 1)}$$

Where γ is ratio of specific heats ($^{C_p}/_{C_v}$), P_e is nozzle exit pressure, P_c is combustion chamber pressure , T_c is combustion chamber temperature, $R_g = \frac{RR}{MM}$ is exhaust flow specific gas constant as “RR” is universal gas constant and MM is exhaust gas molecular weight here k is typically between 1.21-1.26 for wide range of fuel and oxidizer [17].

3.3 Grain configurations

Shaped mass of propellant in combustion chamber is known as grain. Propellant grains are cast, molded and extruded and its appearance is hard rubber or plastic. Once propellant grain is ignited it will burn through all its exposed surfaces. Motor performance prediction depends upon the propellant material and grain configuration. There are two methods for holding the grain in the chamber. One is cartridge-loaded or free standing grains in which grains are manufactured separately from the case (by extrusion or by casting into cylindrical mold or cartridge) and then loaded into the case. The other one is known as case bonded grain in which propellant is cast directly into the case. Cartridge loaded grains are used in some small tactical missiles and a few medium sized motors. They have lower cost and are easier to impact. But the case bonded grains give somewhat better performance, no holding device and better loading fraction [1].

Some grain configurations are shown here and brief introduction of some grain is given below

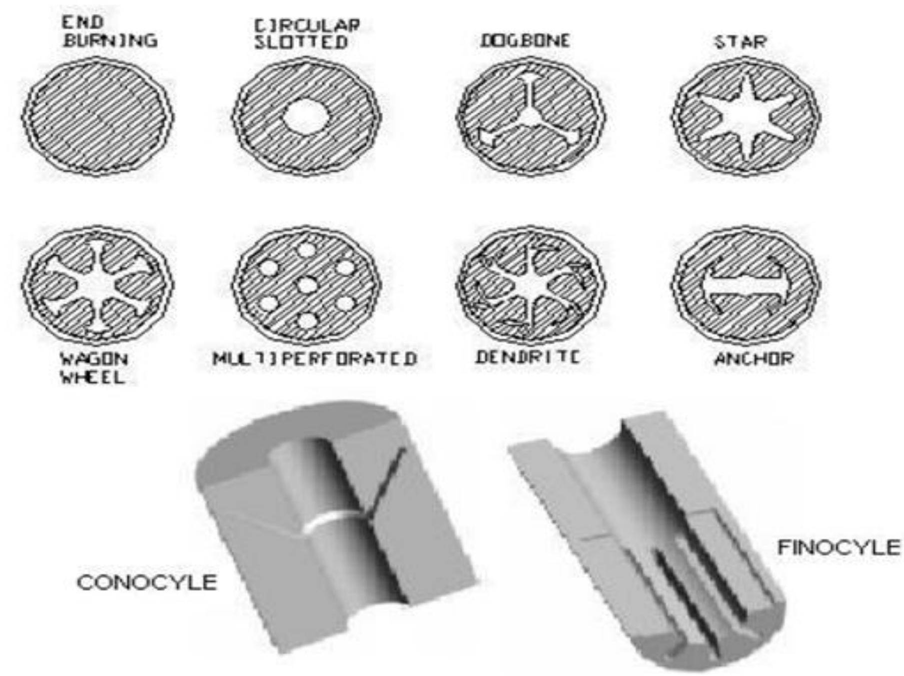


Figure 3.1 some grain configurations

3.3.1 Grain configuration with single level of thrust

3.3.1.1 End burner

In end burner the material is filled in the chamber and it attains the shape of the chamber, in which the area present at the end of the chamber is exposed for combustion and remaining all sides are restricted. In end burner the combustion proceeds longitudinally in the chamber, this is the main difference of end burner from other grain configurations. End burning grain is defined only by the two parameters such as length of the grain "L" and diameter of the grain "D". As the end burner exposed area is small and constant throughout so it produces neutral thrust. Burning surface of end burner is independent of web burned and is simply defined by cross sectional area of the grain. This type of grain is used for requiring long duration rocket motors where low level of thrust is required. [8]

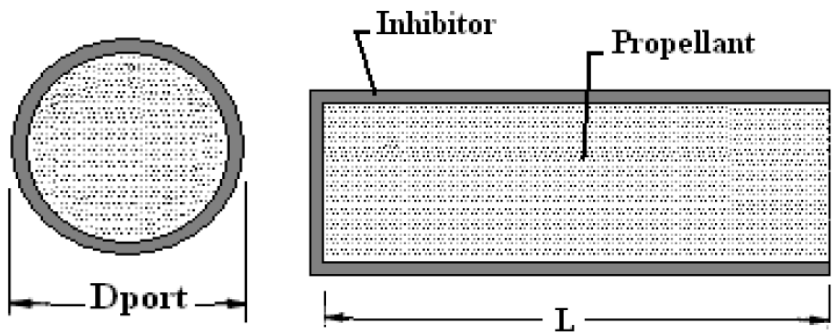


Figure 3.2 End burner

3.3.1.2 Tube grain

This is the most preferred configuration today. It is radial burning grain. In tube grain some time ends are restricted so only there is internal burning. This is known as internal burning tube. But in some cases tube grain ends are unrestricted and it burns both internally and at their end surfaces. This type of grain is known as internal and end burning grain. In tube grain configuration bore is added to the end burner grain. This grain produces progressive thrust as due to bore addition surface area changes as combustion proceeds. Tube grains are case bonded grain in which outer surface is inhibited. Tube grain is defined by the grain length “L” and two diameters outer diameter “D_{out}” and port diameter “D_{port}”.

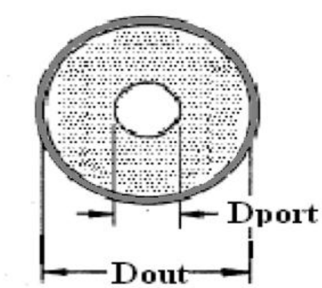
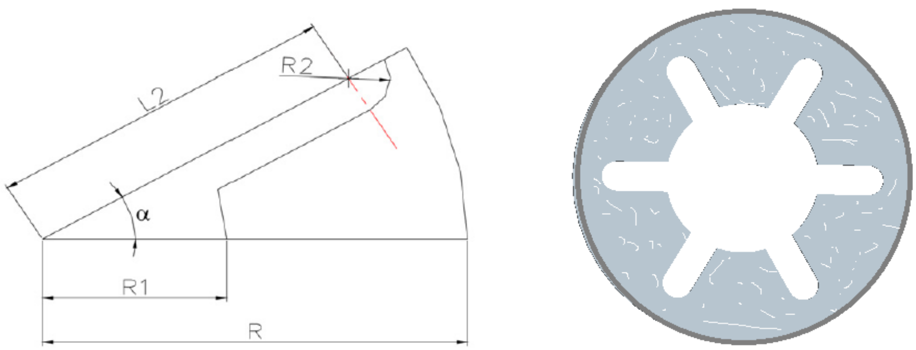


Figure 3.3 Cylindrical grain

3.3.1.3 Star grain

Star grain is radially burning cylindrical grain with distinctive geometric properties. Star grain is a neutral burning grain as star wedges produce regressive burning and cylindrical grain produces progressive burning. In star grain some variables are present that are changed to alter the geometry of star grain. As shown in star grain geometry in Figure. 3.4



1/12th part of star grain geometry showing parameters that controls the star shape

Figure 3.4 Star grain

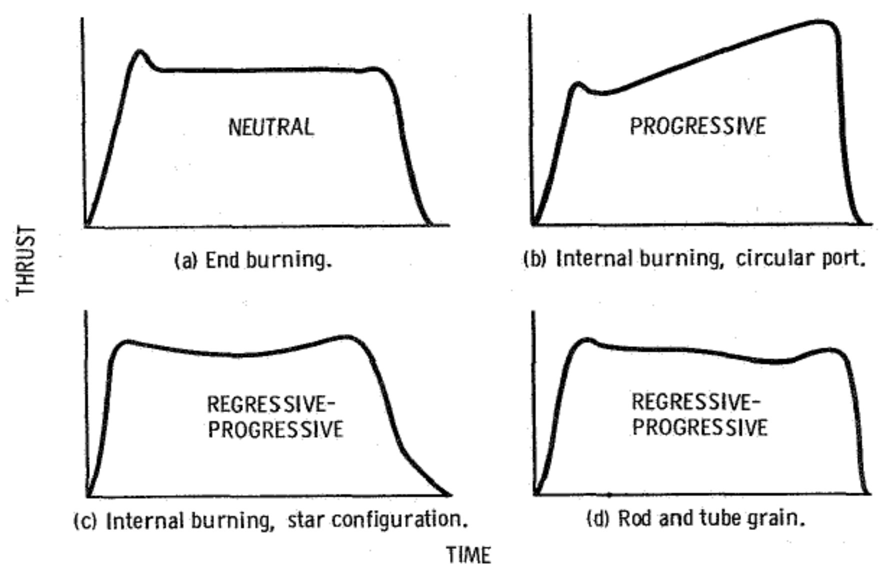


Figure 3.5 Single level-thrust produced by different grain configuration

3.3.2 Grain configuration with dual level of thrust

3.3.2.1 Finocyl grain

Finocyl grain is combination of two grains having different shapes, one is cylindrical grain and the other is star grain. This type of grain gives first progressive thrust and then neutral thrust. Here star grain used has different fin geometries such as fin length, number of fins, fin curve length and star angle. In star grains length of the grain remained constant and they possess only internal burning. Cylindrical grain used in finocyl grain is of decreasing length. Finocyl grain is used in booster system to obtain boost-sustain profile.

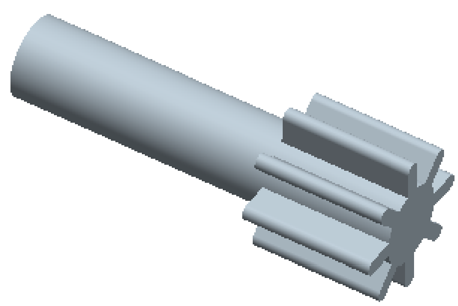


Figure 3.6 Finocyl grain

3.3.2.2 Wagon wheel

As its name indicates that its shape is like shape of wagon wheel. Wagon wheel is although a single grain but it produces dual level of thrust, first progressive and then neutral thrust. Other examples of wagon wheel are dendrite, dog bone and anchor configurations. Wagon wheel can also produce two step thrust by changing the propellant characteristics. Both thrust time curve are shown in Figure. 3.7

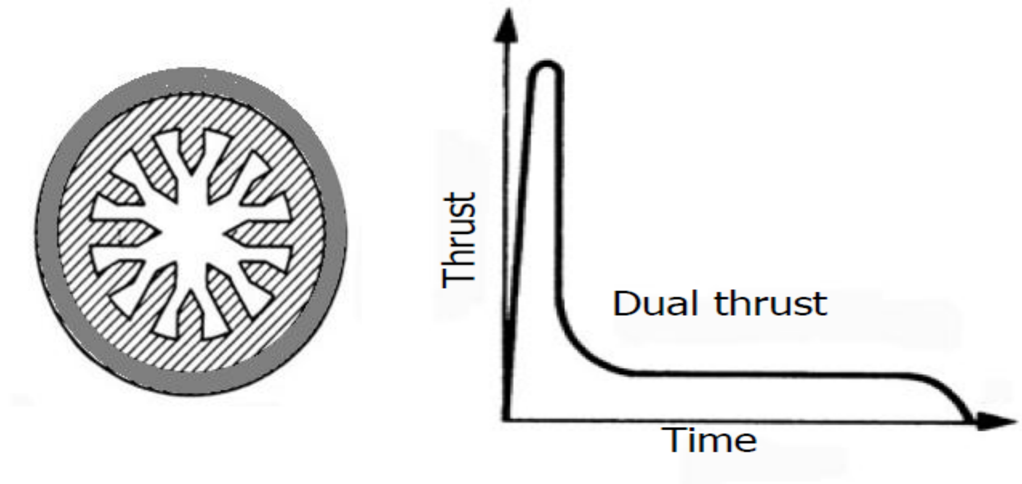


Figure 3.7(a) Wagon wheel grain producing dual thrust

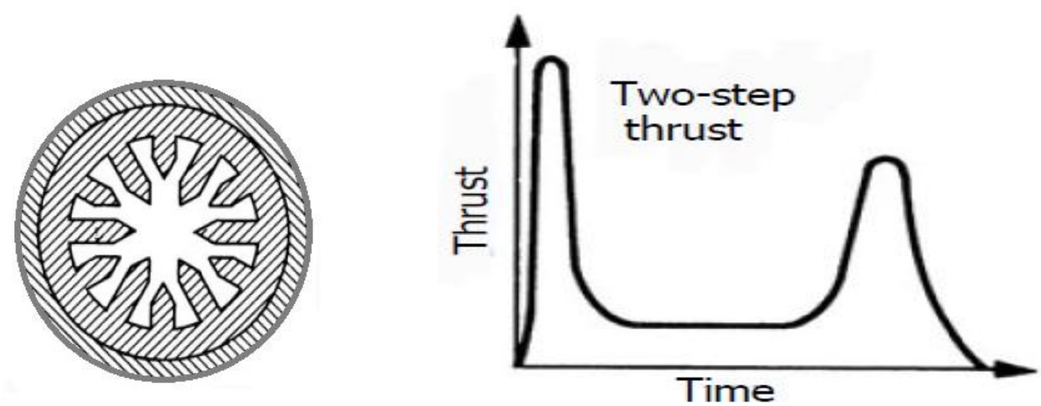


Figure 3.7(b) Wagon wheel grain with dual composition producing two-step thrust

Chapter 4

METHODOGOLOGY ADOPTED

Methodology adopted in this study consists of solid modeling of propellant grain in which initial propellant grain is modeled parametrically and then by changing the parameters (like length of grain, port diameter, fin length etc.), the propellant grain models at each burn step are created. Through this way grain burn back analysis of the propellant grain has been performed. It is necessary to predict the performance of solid rocket motor. Burn back analysis can be done by modeling complete grain geometry or just modeling some part of the grain geometry before parameterization.

Four different types of propellant materials are used in this study. Each propellant material has different ballistic properties. Properties of these four propellant materials are calculated in this study and their performance results are compared

Pro/ENGINEER Wildfire 4.0 (PTC) CAD software is used as solid modeling tool for this study. GUIPEP software is used to predict propellant material properties. TEC plot is used as post processing tool.

Methodology adopted in this study are discussed in detail in each section as given below in detail

4.1 Pro/ENGINEER Wildfire 4.0 modeling

Pro/ENGINEER Wildfire 4.0 is parametric, feature-based, associative solid modeling software which creates simple and complex designs. It is parametric and integrated 3-D CAD/CAM/CAE designing solution.

Propellant grain is modeled in Pro/Engineer. As grain geometry changes throughout the motor operation, so initial grain geometry is modeled in Pro/Engineer then, one by one, each successive web step of the receding grain is created and model is saved in Pro/Engineer.

In Pro/Engineer several tools are available which simplify the modeling method [9]. This tool simplifies each web step creation of the grain geometry. In this study 3-D grain geometry is modeled parametrically in Pro/Engineer. All of its dimensions like length of grain, port diameter, star fin length and star curve are the parameters that are controlled, thus by changing these parameters grain model at each burn step is created. At each web step Pro/Engineer creates new grain model as well as measures surface area of the grain.

In this tool family tables are used for the burn back analysis. Family tables creates a Microsoft excel spreadsheet, which correlate each web step with the changing parameters. Every row of the excel sheet is for new model. Against these parameters burn area comes as an output that is used for the calculation of ballistic parameters. The burn area used for calculations here is the average burn area between two consecutive burn steps. In this way 3-D grain burn back analysis is performed that is necessary for the performance prediction of SRM.

Two methods are used to perform grain burn back analysis: first one is to design a mandrel (cavity inside the grain) and second one is to model the solid grain itself. Both methods provide same results. In this study first method is used.

An excel spread sheet created in Pro/Engineer for finocyl grain is geometry [4] is shown in Table 4.1

Table 4.1 Excel spread sheet created in Pro/Engineer for finocyl grain

Pro/E Family Table
NEW_FINOCYL_DESIGN2

INST NAME	COMMON NAME	d13	d9	d1	d3	d12	d7	d0
!GENERIC	new_finocyl_design2.prt	3	30	100	30	10	50	66.7
NEW_FINOCYL_DESIGN_INST	new_finocyl_design.prt_INST	4	32	99	32	10	50	66.7
NEW_FINOCYL_DESIGN_INST1	new_finocyl_design.prt_INST1	5	34	98	34	10	50	66.7
NEW_FINOCYL_DESIGN_INST2	new_finocyl_design.prt_INST2	6	36	97	36	10	50	66.7
NEW_FINOCYL_DESIGN_INST3	new_finocyl_design.prt_INST3	7	38	96	38	10	50	66.7
NEW_FINOCYL_DESIGN_INST4	new_finocyl_design.prt_INST4	8	40	95	40	10	50	66.7
NEW_FINOCYL_DESIGN_INST5	new_finocyl_design.prt_INST5	9	42	94	42	10	50	66.7
NEW_FINOCYL_DESIGN_INST6	new_finocyl_design.prt_INST6	10	44	93	44	10	50	66.7
NEW_FINOCYL_DESIGN_INST7	new_finocyl_design.prt_INST6_INST7	11	46	92	46	10	50	66.7
NEW_FINOCYL_DESIGN_INST8	new_finocyl_design.prt_INST6_INST8	12	48	91	48	10	50	66.7
NEW_FINOCYL_DESIGN_INST9	new_finocyl_design.prt_INST6_INST9	13	50	90	50	10	50	66.7
NEW_FINOCYL_DESIGN_INST10	new_finocyl_design.prt_INST6_INST10	14	52	89	52	10	50	66.7
NEW_FINOCYL_DESIGN_INST11	new_finocyl_design.prt_INST6_INST11	14.14	54	88	54	10	50	66.7
NEW_FINOCYL_DESIGN2_INST	new_finocyl_design2.prt_INST	14.14	56	87	56	10	50	66.7
NEW_FINOCYL_DESIGN2_INST1	new_finocyl_design2.prt_INST1	14.14	58	86	58	10	50	66.7
NEW_FINOCYL_DESIGN2_INST2	new_finocyl_design2.prt_INST2	14.14	60	85	60	10	50	66.7
NEW_FINOCYL_DESIGN2_INST3	new_finocyl_design2.prt_INST3	14.14	62	84	62	10	50	66.7
NEW_FINOCYL_DESIGN2_INST4	new_finocyl_design2.prt_INST4	14.14	64	83	64	10	50	66.7
NEW_FINOCYL_DESIGN2_INST5	new_finocyl_design2.prt_INST5	14.14	66	82	66	10	50	66.7
NEW_FINOCYL_DESIGN2_INST6	new_finocyl_design2.prt_INST6	14.14	68	81	68	10	50	66.7

Here d0, d1, d3, d7, d9, d12, d13 are the parameters of the finocyl grain geometry. Where d0 is the outer diameter, d1 is the length of cylindrical grain, d3 is the cylindrical grain inner diameter, d7 is the star grain length, d9 is the star grain port diameter, d12 is the star fin length and d13 is the star curve radius. This by changing these parameters new grain model at each burn step is created. So, that why grain burn back analysis of propellant grain has been performed.

Propellant grain can be modeled completely or grain geometry can be divided into smaller parts and only single part is modeled before parameterization. So, in this study complete grain is modeled and Figure. 4.1 shows initial finocyl grain model and its models at different burn steps.

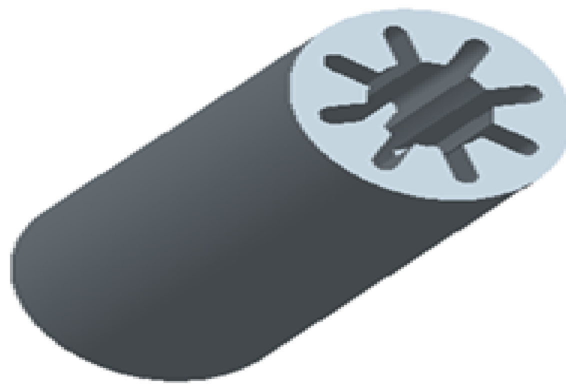


Figure 4.1 (a) Initial finocyl grain model

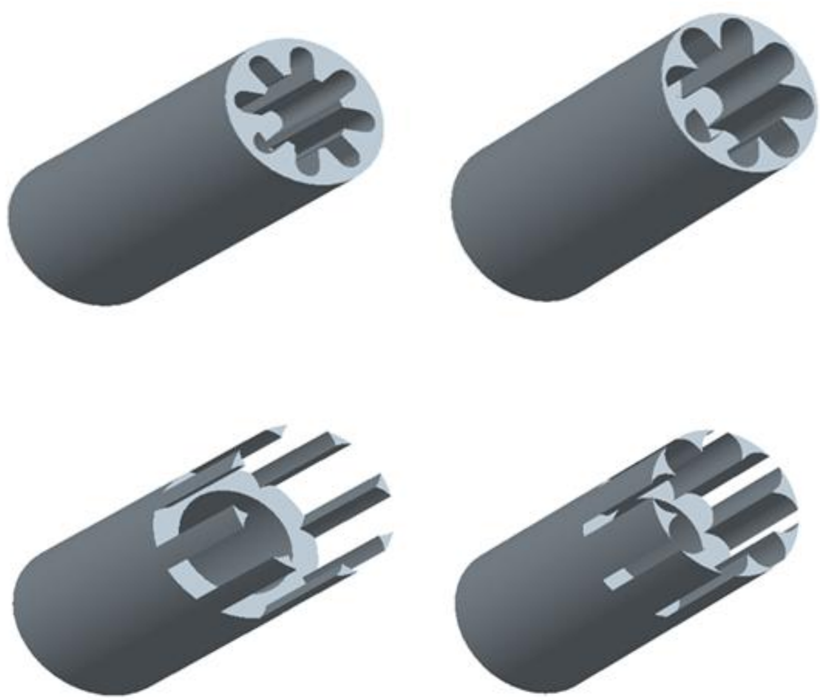


Figure 4.1 (b) Finocyl grain models at different burn steps

In this way propellant grain models for the required grain configuration are created in Pro/Engineer.

4.2 GUIPEP

GUIPEP is software written by Arthur J. Lekstutis. GUIPEP gives a graphical user interface to run the PROPEP propellant evaluation program. PROPEP is a program that evaluates the chemical equilibrium composition for the combustion of a solid or liquid rocket propellant. Moreover, it establishes rocket performance parameters such as I_{sp} and C*, and nozzle design parameters [11]. This software was used to calculate the properties of propellant materials used in this study such as values of pressure exponent "n" and temperature coefficient "a" and characteristic velocity value "c*".

Input for GUIPEP is the propellant material composition and operating conditions. Composition of propellant materials is entered in grams and total mass need not up to 100grams. Operating conditions are mostly used as default.

A screen shot outlook of GUIPEP input is shown in Figure. 2

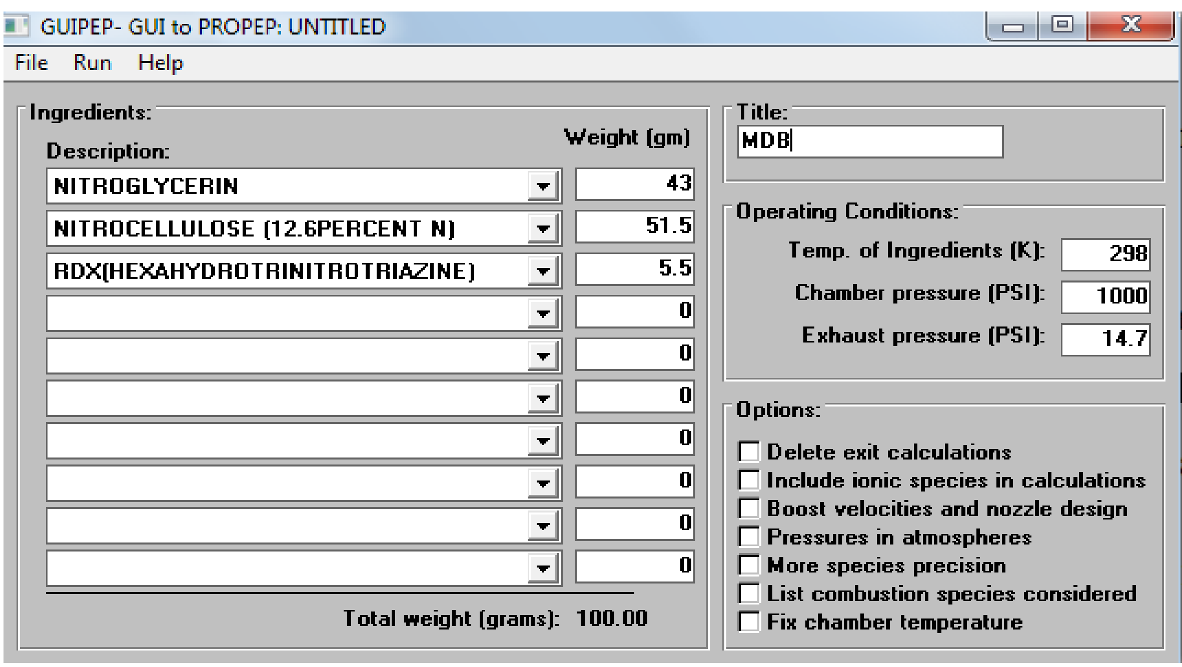


Figure 4.2 Screen shot of GUIPEP input

Then the final step is to run the program, after running the program by clicking single run output is displayed in a note pad file. Here output is displayed in three portions as shown in Figure 4.3

```
PRINT.TXT - Notepad
File  Edit  Format  View  Help
 MDB                    Run using June 1988 Version of PEP,
 Case  1 of  1           6 Sep 2011 at 11:54:24.95 pm

  CODE                                        WEIGHT    D-H   DENS       COMPOSITION
   693 NITROGLYCERIN                          43.000   -400  0.05780      3C    5H    3N    9C
   683 NITROCELLULOSE (12.6PERCENT N)         51.500   -617  0.05600    755H  600C  245N  990C
   847 RDX(HEXAHYDROTRINITROTRIAZINE)          5.500     66  0.06560      3C    6H    6N    6C

THE PROPELLANT DENSITY IS 0.05723 LB/CU-IN OR  1.5840 GM/CC
THE TOTAL PROPELLANT WEIGHT IS  100.0000 GRAMS

NUMBER OF GRAM ATOMS OF EACH ELEMENT PRESENT IN INGREDIENTS

   2.522720 H     1.776698 C     1.179806 N     3.724404 O

****************************CHAMBER RESULTS FOLLOW    *****************************

 T(K)  T(F)  P(ATM)   P(PSI) ENTHALPY  ENTROPY  CP/CV     GAS    RT/V
3191. 5285.   68.02  1000.00   -48.61   235.40  1.2099  3.700  18.386

SPECIFIC HEAT (MOLAR) OF GAS AND TOTAL=   11.449   11.452
NUMBER MOLS GAS AND CONDENSED=   3.6997   0.0000

  1.04540 H2O         0.98571 CO          0.79091 CO2         0.58298 N2
  0.17092 H2          0.06337 HO          0.02650 H           0.01413 O2
  0.01376 NO          0.00582 O           0.00008 HO2         0.00003 CHO
 1.14E-05 NHO        7.90E-06 N          6.81E-06 NO2        6.24E-06 NH3
 3.42E-06 NH2        2.59E-06 N2O        2.41E-06 NH         1.74E-06 CH2O
 1.66E-06 CNHO       1.64E-06 CNH        1.02E-06 NHO2

THE MOLECULAR WEIGHT OF THE MIXTURE IS   27.029

****************************EXHAUST RESULTS FOLLOW    *****************************

 T(K)  T(F)  P(ATM)   P(PSI) ENTHALPY  ENTROPY  CP/CV     GAS    RT/V
1656. 2521.    1.00    14.70  -121.96   235.40  1.2260  3.628   0.276

SPECIFIC HEAT (MOLAR) OF GAS AND TOTAL=   10.778   10.778
NUMBER MOLS GAS AND CONDENSED=   3.6280   0.0000

  0.99474 H2O         0.95294 CO2         0.82370 CO          0.58988 N2
  0.26654 H2          0.00009 H           0.00002 HO          0.00000 NH3

THE MOLECULAR WEIGHT OF THE MIXTURE IS   27.564

**********PERFORMANCE:  FROZEN ON FIRST LINE, SHIFTING ON SECOND LINE**********

IMPULSE   IS EX      T*      P*      C*     ISP*  OPT-EX  D-ISP     A*M    EX-T
  246.1  1.2221   2872.   38.11  4963.5           8.50  389.8  0.15431  1482.
  252.7  1.1706   2956.   38.79  5045.0   194.6    8.92  400.2  0.15684  1656.
```

Figure 4.3 Screen shot of GUIPEP output

Here first portion is an eco of complete input data and second portion shows the combustion chamber conditions and third/last portion presents the nozzle exhaust conditions.

4.3 Tec Plot

Tec plot is the name of a family of visualization software tools developed by Tec plot, Inc. Tec plot from Am tec is software allowing the user to display data in a very easy and attractive way. XY-Plots, 2D or 3D visualizations illustrate data created directly with Tec plot or imported from common industrial format [10]. In this study it is used as a post processing tool for simulation results.

4.4 Worked Example

An example of finocyl grain geometry is considered. As burn back analysis of the finocyl grain is shown in Figure. 1. The output of the Pro/Engineer software for finocyl grain is as shown in Table. 2.

Table 4.2 Pro/Engineer output

web step (m)	Burn Area, Ab (m^2)
0	0.026164
1	0.026489473
2	0.027090863
3	0.027642014
4	0.028191691
5	0.028494571
6	0.0271807
7	0.024356895
8	0.021784625
9	0.01986113
10	0.018290446
11	0.016851951
12	0.016142255
13	0.016234835
14	0.016429656
15	0.016605594
16	0.016762648
17	0.016900869
18	0.017020257
19	0.008537628
20	0

Burning characteristic values are calculated using equations discussed in chapter 3. Burning characteristics results are shown in table is shown in Table 4.3

Table 4.3 Values of burning characteristics

webstep(m)	Ab(m^2)	Pc (Pa)	Pc (Bar)	rb (m/s)	mb (kg/s)	del time(s)	t (s)	cf	F	Isp
0	0.02616	9184801	91.848	0.01226	0.513268	0	0	1.63	1422.27	266.12
0.001	0.02649	9353316	93.533	0.01233	0.522685	0.0810873	0.0811	1.635	1452.8	266.94
0.002	0.02709	9667255	96.673	0.01246	0.540229	0.0802352	0.1613	1.64	1506.16	267.76
0.003	0.02764	9957863	99.579	0.01258	0.556469	0.0794783	0.2408	1.643	1554.27	268.24
0.004	0.02819	10250424	102.5	0.0127	0.572818	0.0787453	0.3195	1.65	1606.75	269.39
0.005	0.02849	10412783	104.13	0.01276	0.581891	0.0783503	0.3979	1.652	1634.18	269.71
0.006	0.02718	9714435	97.144	0.01248	0.542866	0.0801103	0.478	1.64	1513.51	267.76
0.007	0.02436	8267242	82.672	0.01185	0.461993	0.0843542	0.5624	1.615	1268.4	263.67
0.008	0.02178	7015822	70.158	0.01125	0.392061	0.0889031	0.6513	1.599	1065.74	261.06
0.009	0.01986	6124069	61.241	0.01077	0.342227	0.0928558	0.7441	1.58	919.223	257.96
0.01	0.01829	5425290	54.253	0.01036	0.303178	0.0965265	0.8406	1.57	809.182	256.33
0.011	0.01685	4809590	48.096	0.00997	0.268771	0.10032	0.941	1.55	708.212	253.06
0.012	0.01614	4514698	45.147	0.00977	0.252292	0.1023719	1.0433	1.54	660.5	251.43
0.013	0.01623	4552827	45.528	0.00979	0.254423	0.1020968	1.1454	1.54	666.079	251.43
0.014	0.01643	4633398	46.334	0.00985	0.258925	0.1015253	1.247	1.545	680.067	252.24
0.015	0.01661	4706548	47.065	0.0099	0.263013	0.1010176	1.348	1.549	692.592	252.9
0.016	0.01676	4772156	47.722	0.00994	0.266679	0.1005711	1.4485	1.55	757.103	272.65
0.017	0.0169	4830136	48.301	0.00998	0.269919	0.1001832	1.5487	1.55	768.136	273.31
0.018	0.01702	4880396	48.804	0.01001	0.272728	0.0998519	1.6486	1.52	776.129	273.31
0.019	0.00854	1769393	17.694	0.00724	0.098878	0.1381523	1.7867	1.29	250.458	243.27
0.02	0	0	0	0	0	0	1.7867	0	0	0

These calculations are carried out using equation discussed in chapter 3 so calculation procedure using these equations for web step 9 is as follows.

As burn area, A_b obtained through Pro/Engineer is 0.01986 m^2. So chamber pressure P_c is calculated by using equation (1)

$$P_c = \left[\frac{a\rho_p A_b C^*}{A_{th}}\right]^{\left(1/_{1-n}\right)} = \left[\frac{0.0000725 * 1700 * 0.01986 * 1600}{0.000095}\right]^{\left(1/_{1-0.32}\right)} = 6.1MPa$$

Here values of "a", "n" and "C*" are calculated from GUIPEP software for composite propellant containing AP/Al/HTPB.

Burn rate is found using equation (2)

$$r_b = aP^n = 0.0000725 * (61.241)^{0.32} = 0.01077\ m/s$$

Mass flow rate is calculated using equation (4)

$$m = \rho\, r_b A_b = 1700 * 0.01077 * 0.01986 = 0.342227\ kg/s$$

After pressure, burn rate and mass flow rate calculation time is calculated by using

$$\Delta t_9 = \frac{\Delta x}{r_b} = \frac{0.009 - 0.008}{0.01077} = 0.0928558\ s$$

Here Δx is the difference between two web steps (present web step and previous web step). Time in second is calculated by adding delta time to the time calculated at previous web step as

$$t_9 = t_8 + \Delta t_9 = 0.06513 + 0.0928558 = 0.7441 \text{ s}$$

Thrust calculations are carried out using equation (5)

$$F = C_f P_c A_{th}$$

But before using this relation values of thrust coefficient C_f is calculated from graph shown in Figure 4.4, against values of pressure ratios$\left(\frac{P_c}{P_{atm}}\right)$.

So

$$F = 1.58 * 6124069 * 0.000095 = 919.223\ N$$

Last performance relation that is calculated is specific impulse that is calculated using equation (9)

$$I_{sp} = \frac{C^* C_F}{g_0} = \frac{F}{m g_0} = \frac{919.223}{0.342227 * 9.8} = 257.96\ s$$

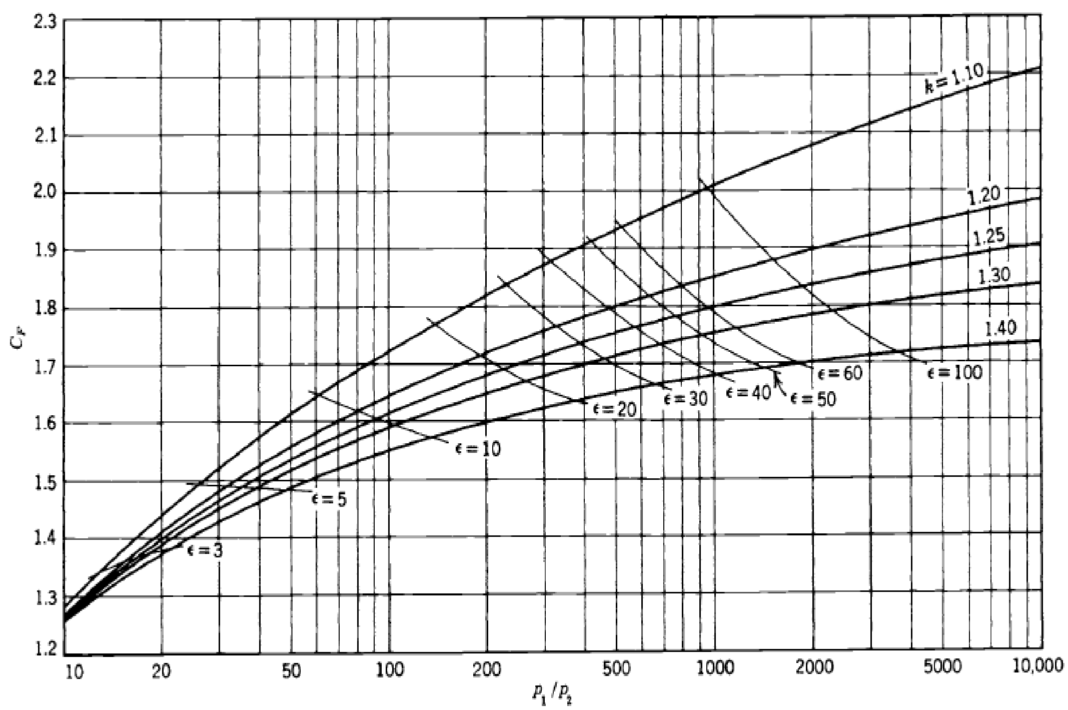


Figure 4.4 Thrust coefficient as a function of pressure ratio

In this way whole calculations are performed for four types of propellant and then using Tec plot pressure vs. time, thrust vs. time and specific impulse vs. time are plotted. Comparisons of the performance parameters and different propellants performed.

4.5 Adiabatic flame temperature calculation

In combustion study adiabatic flame temperature is described in two ways:

- Constant volume adiabatic flame temperature
- Constant pressure adiabatic flame temperature

Constant volume adiabatic flame temperature is the temperature that results from a complete combustion process that occurs without any work, heat transfer or changes in kinetic or potential energy. The constant pressure adiabatic flame temperature is the temperature that results from a complete combustion process that occurs without any heat transfer or changes in kinetic or potential energy. Its temperature is lower than the constant volume process because some of the energy is utilized to change the volume of the system (i.e., generate work). [15]

Adiabatic flame temperature is calculated manually and computationally for four types of propellants double base (DB), composite modified double base (CMDB), modified double base (MDB), composite. As double base propellants consist of only nitroglycerin (NG), nitrocellulose (NC) materials and modified double base consist of nitroglycerin (NG), nitrocellulose (NC) and RDX which is a nitramines, its addition to the double base propellants increase their ballistic properties. Composite modified double base propellants are combination of double base as well as composite propellants such as Al, AP (ammonium perchalorate) etc. and composite propellants consist of oxidizer as AP, polymeric binders as HTPB, metallic components such as aluminum or magnesium they are used as fuel. Curing agents, phase stabilizers, and solvents may be other additives included in small percentages.

4.5.1 Computational methodology:

An adiabatic flame temperature calculation code is developed in MATLAB. Here varying concentration of oxidizers and fuels is used for adiabatic flame temperature calculation. Flame temperature against oxidizer concentration is plotted for all four propellant materials.

For adiabatic flame temperature calculation first is to calculate oxygen balance (percentage by weight of oxygen positive or negative remaining after explosion assuming that all the carbon and hydrogen are converted into CO2 &H2O), it is calculated by using the formula for $C_aH_bN_cO_d$

$$\text{Oxygen Balance} = \left[\frac{(d - 2 * a - b/2) * 1600}{\text{molecular weight of compound}}\right]$$

The detonation of energetic materials will result in the formation of its decomposition products. These may be carbon monoxide, carbon dioxide, carbon, water, etc. In order to clarify the problems of decomposition products set of rules was developed by various researchers. [16].

4.5.1.1 Kistiakosky rule (K_W rules):

These rules should only be used for moderately oxygen deficient explosives with an **oxygen balance greater than – 40.0**.

i) Convert all carbon to CO
ii) Then remaining oxygen is used to convert hydrogen into H2O
iii) Then remaining oxygen is used to convert to CO2
iv) All nitrogen is converted into N2

4.5.1.2 Modified Kistiakosky rule:

The Kistiakowsky-Wilson concept cannot be used for explosive materials which have an **oxygen balance lower than – 40**. Under these circumstances the modified Kistiakowsky-Wilson concept has to be employed

i) Convert all hydrogen into H_2O
ii) Then remaining oxygen is used to convert to CO
iii) All nitrogen is converted into N_2

4.5.1.3 Springall-Roberts rules:

Springall-Roberts modified the Kistiakowsky-Wilson and modified Kistiakowsky-Wilson rules. He takes the Kistiakowsky-Wilson rules and adds two more conditions.

i) Convert all carbon to CO
ii) Then remaining oxygen is used to convert hydrogen into H_2O
iii) Then remaining oxygen is used to convert to CO_2
iv) All nitrogen is converted into N_2
v) One third of the carbon monoxide formed is converted to carbon and carbon dioxide
vi) One sixth of the original amount of carbon monoxide is converted to form carbon and water

For propellants which contains only some or all of the atoms: aluminum, boron, carbon, calcium, chlorine, fluorine, hydrogen, potassium, nitrogen, sodium and oxygen (with the formula $Al_{al}, B_b, C_c, Ca_{ca}, Cl_{cl}, F_f, O_o, N_n, Na_{na}$) the oxygen balance will be calculated by

$$OB = \frac{32\left[\frac{3}{4}al + \frac{3}{4}b + 1c + \frac{1}{2}ca - \frac{1}{4}cl - \frac{1}{4}f + \frac{1}{4}h + \frac{1}{4}k + 0n + \frac{1}{4}na - \frac{1}{2}o\right] * 100}{molecular\ weight\ of\ compound}$$

Except nitrogen all elements have a contribution to propellant composition.

4.5.1.4 Keshavarz rules:

Keshavarz proposed an approximation that all nitrogen go to N_2, fluorine to HF, chlorines to HCl, while a portion of the oxygen's form H_2O and carbons preferentially will be oxidized to CO rather than CO_2. The following pathways can be written to obtain detonation products of an energetic materials having composition CaHbNcOdFeClf: [16].

Case 1: d $\leq a$

$$eHF + fHCl + \frac{c}{2}\,N_2 + dCO_2 + (a-d)C_s + \left[\frac{b-e-f}{2}\right]H_2$$

Case 2: d > a and $\left[\frac{b-e-f}{2}\right] > d-a$

$$eHF + fHCl + \frac{c}{2}\,N_2 + aCO + (d-a)H_2O + \left[\frac{b-e-f}{2} - d + a\right]H_2$$

Case 3: $d \geq a + \left[\frac{b-e-f}{2}\right]$ and $d \leq 2a + \left[\frac{b-e-f}{2}\right]$

$$eHF + fHCl + \frac{c}{2}\,N_2 + \left[\frac{b-e-f}{2}\right]H_2O + \left[2a - d + \frac{b-e-f}{2}\right]CO + \left[d - a - \frac{b-e-f}{2}\right]CO_2$$

Case 4: $d > 2a + \left[\frac{b-e-f}{2}\right]$

$$eHF + fHCl + \frac{c}{2}\,N_2 + \left[\frac{b-e-f}{2}\right]H_2O + aCO_2 + \left[\frac{2d-b+e+f}{4} - a\right]O_2$$

Oxygen balance expresses the number of oxygen molecules remaining after oxidation of C, H, Al, F, Cl, to produce H_2O, CO, Al_2O_3, etc. If excess oxygen molecules are remaining after the oxidation reaction, the oxidizer is said to have a ‘positive’ oxygen balance. If the oxygen molecules are completely consumed and excess fuel molecules remain, the oxidizer is said to have a ‘negative’ oxygen balance.

Using these rules balanced chemical equation is made which can be used again to calculate the heat of explosion or heat of combustion using the formula

$$Q = \Delta Hc = \sum Hf\,(\text{products}) - \sum Hf\,(\text{reactants})$$

Heat of formation values of reactants and products are collected from JANAF TABLES. These values are used for calculation of adiabatic flame temperature through iteration using formula

$$\Delta Hc=\sum n\, Cp * (Td - 298.15)$$

n= number of moles of products

In this methodology in a loop values of temperature are checked against heat of combustion values and loop terminates when selected temperature value satisfy (1). This temperature value is known as "ADIABATIC FLAME TEMPERATURE". Code of adiabatic flame temperature calculation is given in Appendix. A.

Chapter 5

RESULTS AND DISCUSSIONS

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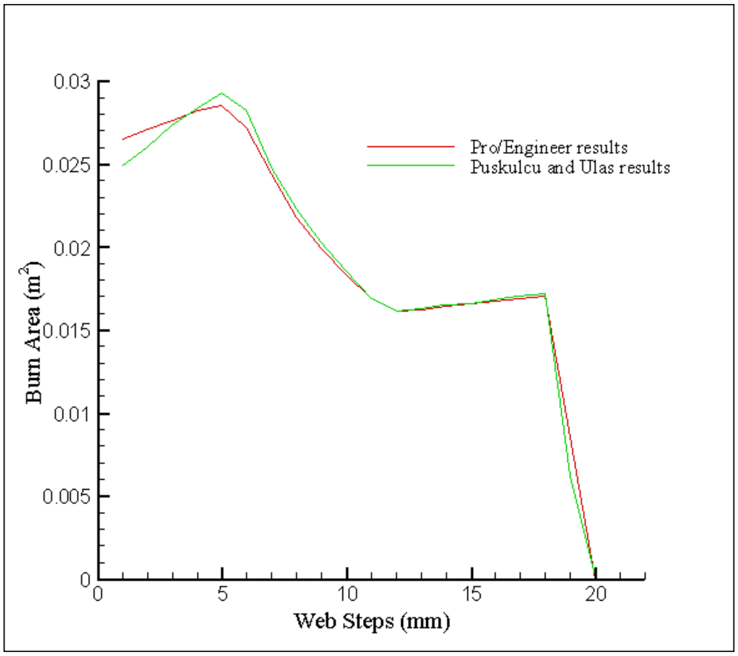
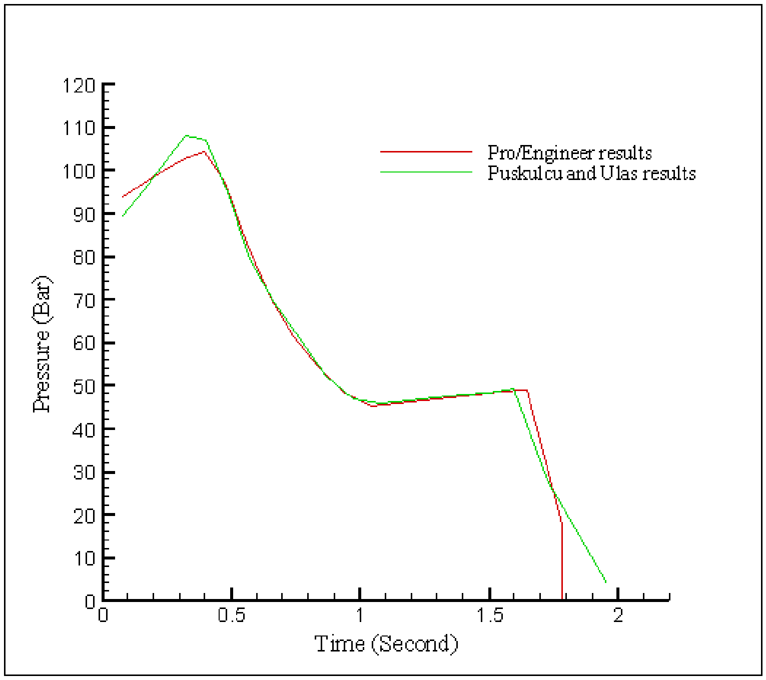
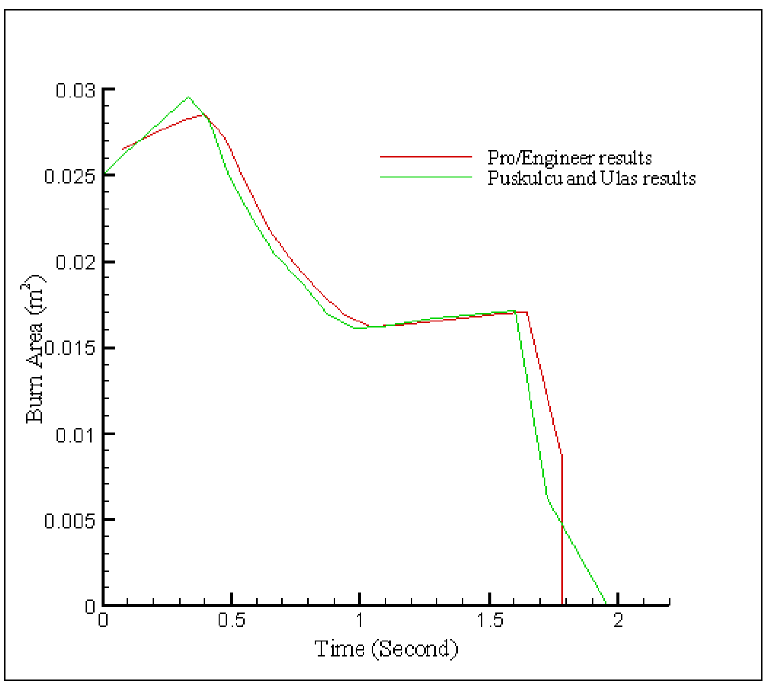
In this study grains are designed to produce boost-sustain-boost (thrust-time) profile. Modeling and simulations results of finocyl grain geometry [4], which produces boost-sustain profile, is chosen as reference. In the reference study 3D grain burn back of analysis of finocyl grain is performed using Unigraphics NX software and AutoCAD mechanical desktop software. Using the burn back analysis data and internal ballistic parameters, the performance of the solid propellant rocket motor in terms of motor pressure was achieved. Validation results of finocyl grain geometry using Pro/Engineer software are shown in Figure 5.1



(i)



(ii)



(iii)

Figure 5.1 Validation results

To produce boost-sustain-boost profile three different grain configurations are used and four type of propellant material composition are analyzed. Table 5.1 shows the composition of propellant materials used. Composition of propellant materials used mainly consist of nitroglycerin (NG), nitrocellulose (NC), cyclotrimethylenetrinitramine (RDX), aluminum metal (Al), hydroxi terminated polybutadiene binder (HTPB) and ammonium per chlorate (AP).

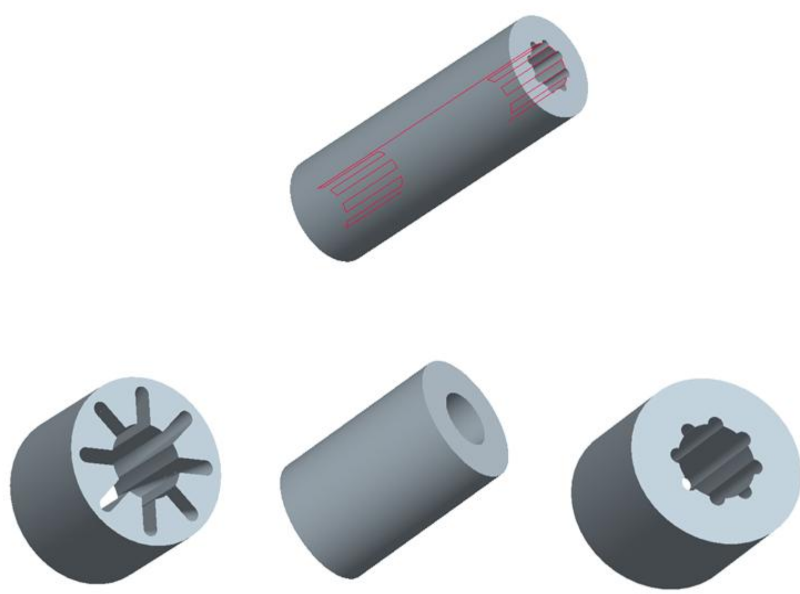
In all these configurations nozzle of throat area 0.000095 meter square is chosen and outer diameter of the grain is 66.7 millimeter. But grain length varies in all cases.

TABLE 5.1 Propellant materials properties

Propellants	Compositions	a (temperature co-efficient)	n (pressure exponent)
DB	NG/NC	0.0001128	0.3
MDB	NG/NC/RDX	0.00003035	0.4
CMDB	NG/NC/RDX/Al	0.00000905	0.49
Composite	HTPB/AP/Al	0.0000725	0.32

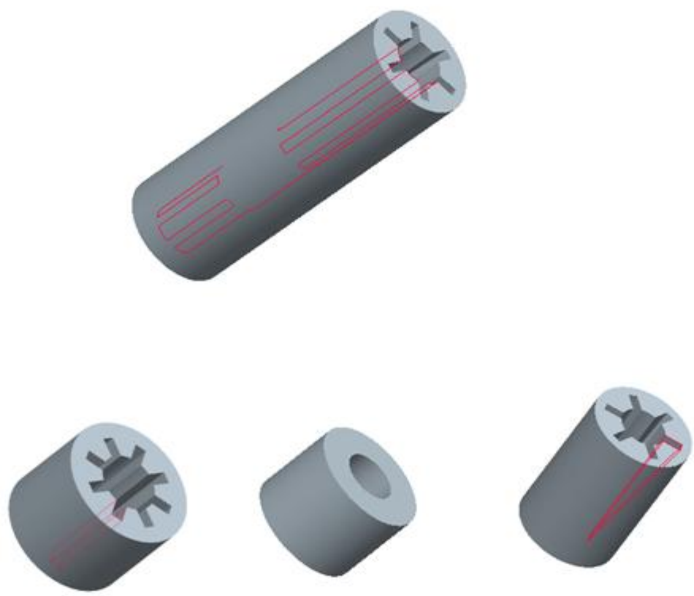
5.1 First configuration

A grain which is a combination of grains of different burning areas [1] is used to produce required (boost-sustain-boost) thrust-time profile. Different grain geometries are considered to produce boost-sustain-boost profile such as star-cylinder-star, cylinder-star-cylinder, star-cylinder-star of varying length, star-cylinder-petal, conocyl-star etc. Grains models of this configuration are shown in Figure 5.2 Simulation results of all these configurations are shown.



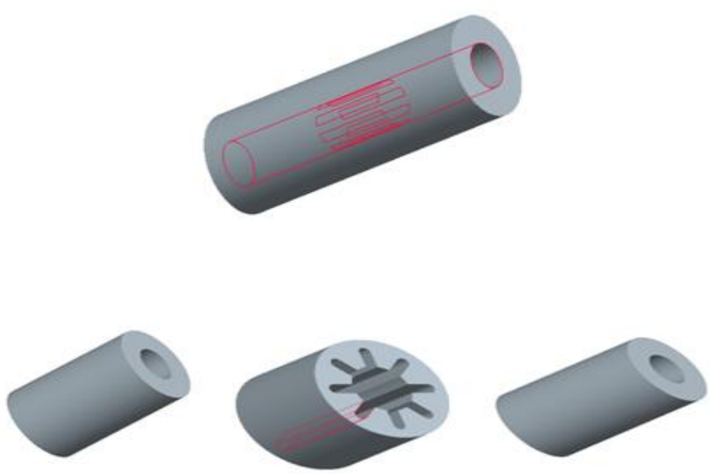
(a) Star of varying length-cylinder-petal grain

Grain length considered in this configuration is of 200mm. Both star and petal grains are of 50mm and cylindrical grain of 100mm.

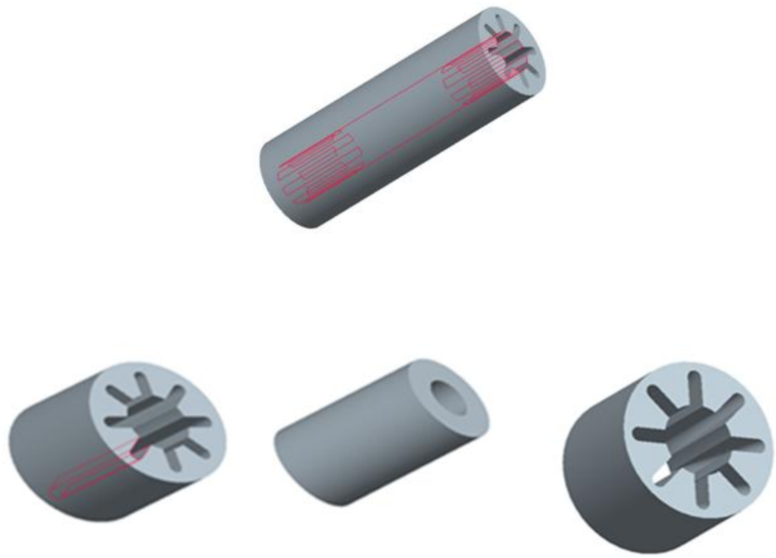


(b) Star of varying length-cylinder-star of constant length (Boost-sustain-boost)

Grain length considered in this configuration is of 200mm. star of varying length is of 100mm and cylinder grain and star grain of constant star fin length are 50, 50mm.



(c) Cylinder-star-cylinder



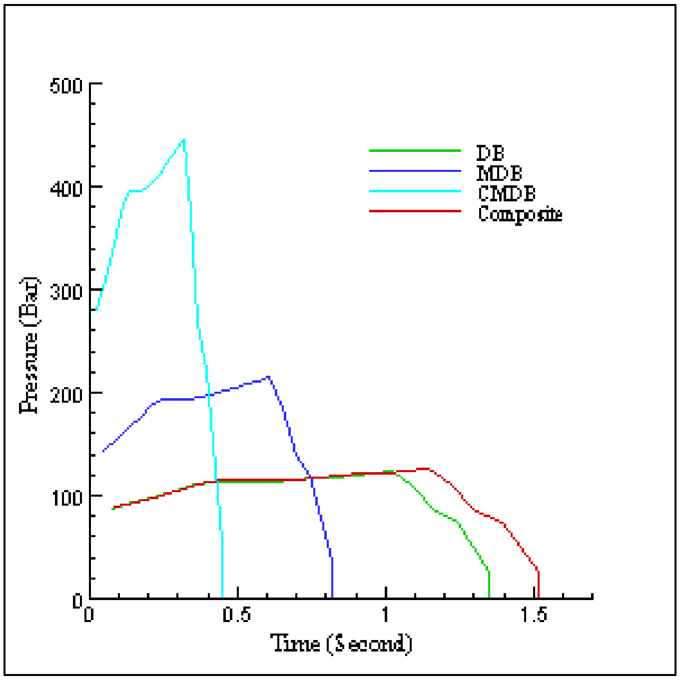
(d) star-cylinder-star of varing length

Figure 5.2 Examples of first grain configuration

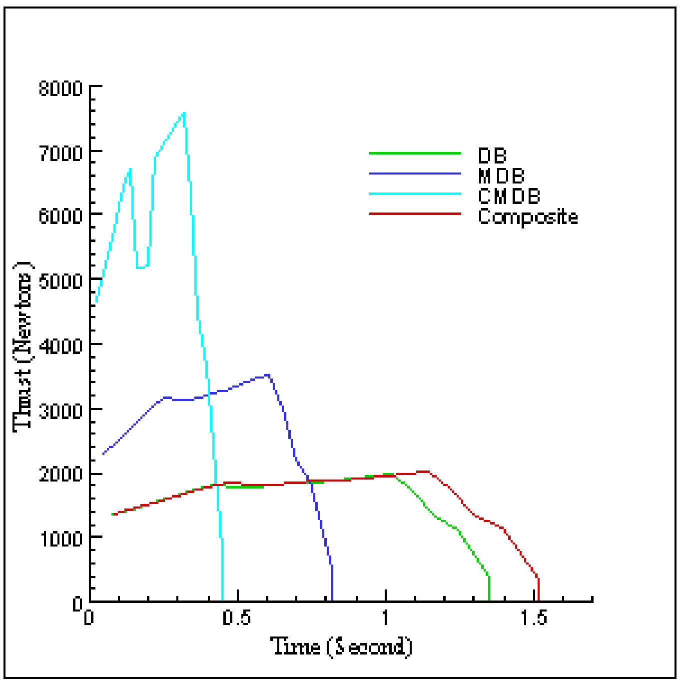
Grain length considered in this configuration is of 250mm. Both cylinders are of 100mm and star grain of 50mm.

This propellant grain configuration is designed to obtained boost-sustain-boost profile but the results obtained from this configuration are not according to the requirement. Simulation results of this configuration are presented in Figure. 2. Other two configurations are presented in the next section which produce the required boost-sustain-boost profile.

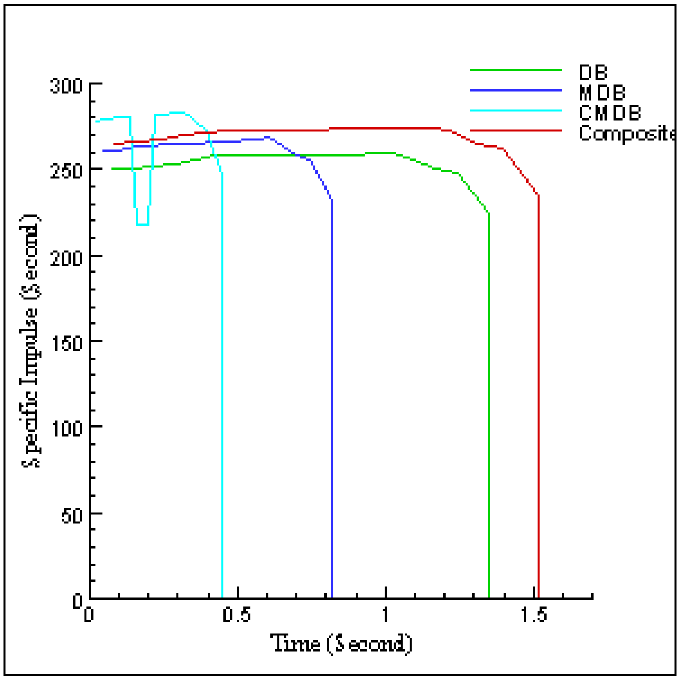
Results of first configuration Figure 5.2 (a)



(i)

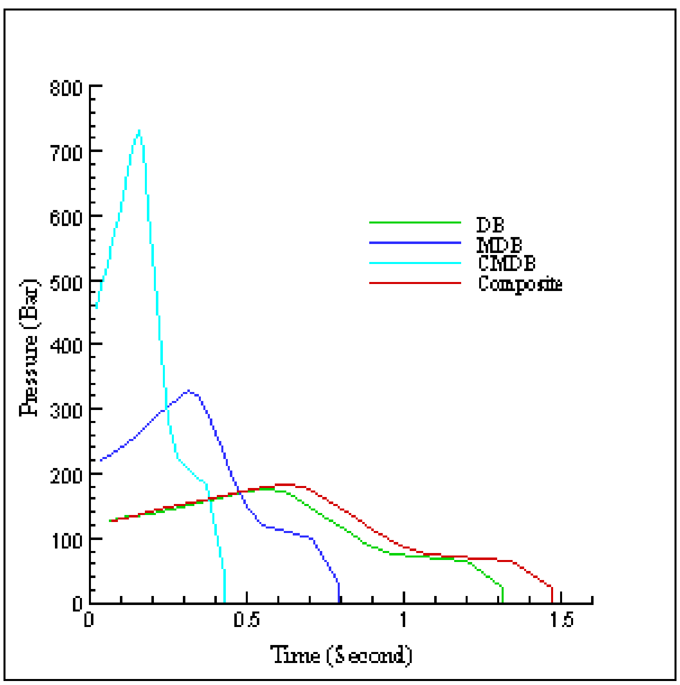


(ii)

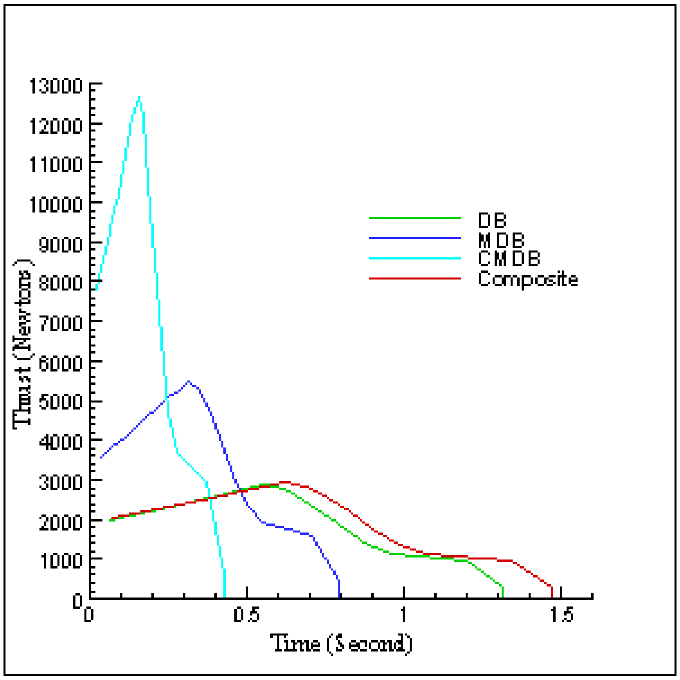


(iii)

Results of first configuration Figure 5.2 (b)

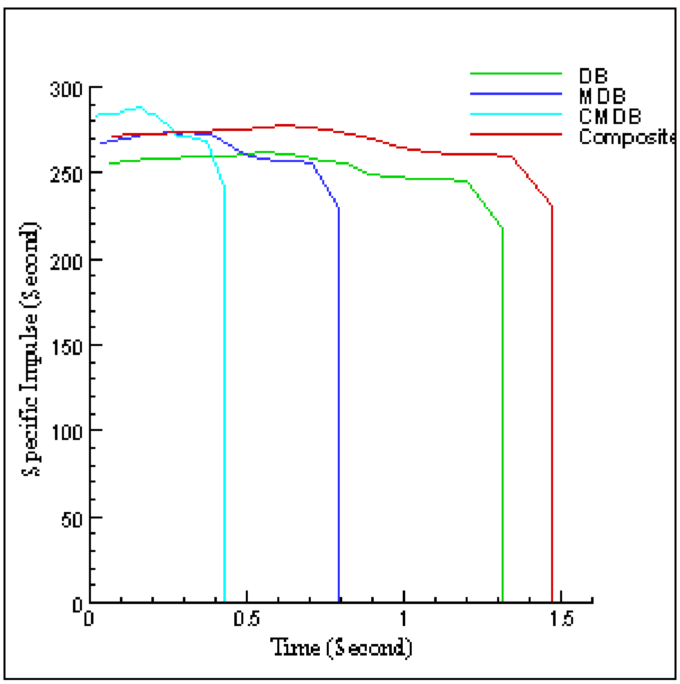


(i)



Thrust (Newtons)
Time (Second)
DB
MDB
CMDB
Composite

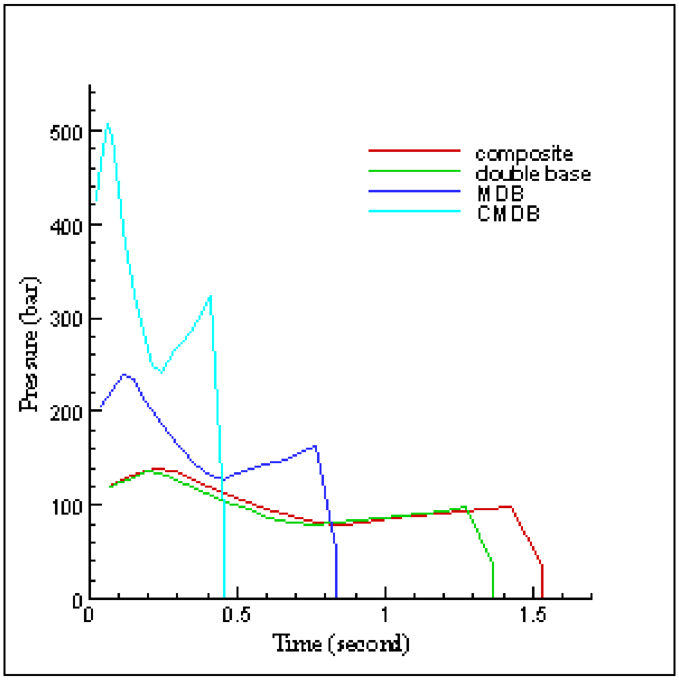
(ii)



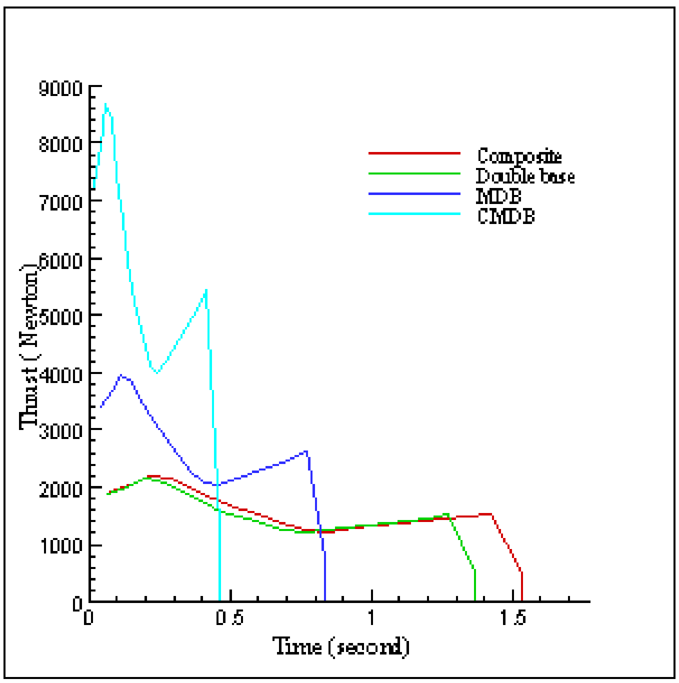
Specific Impulse (Second)
Time (Second)
DB
MDB
CMDB
Composite

(iii)

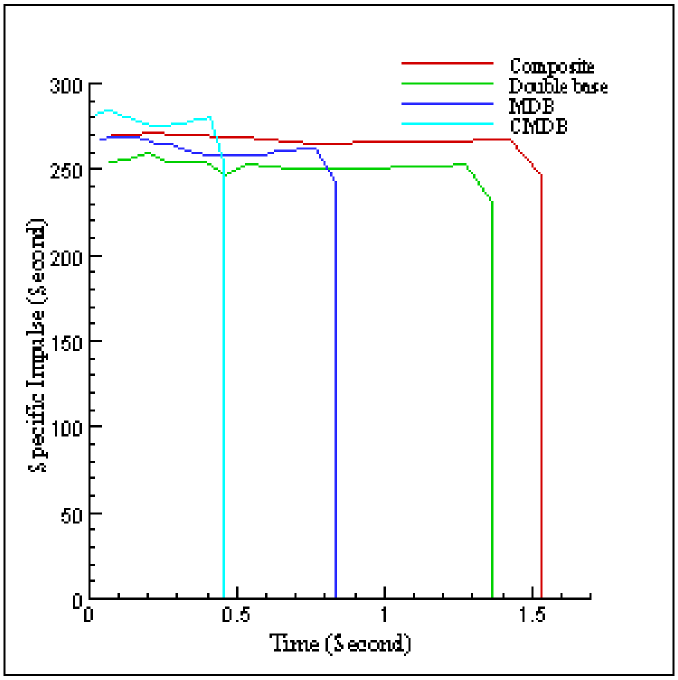
Results of first configuration Figure 5.2 (c)



(i)

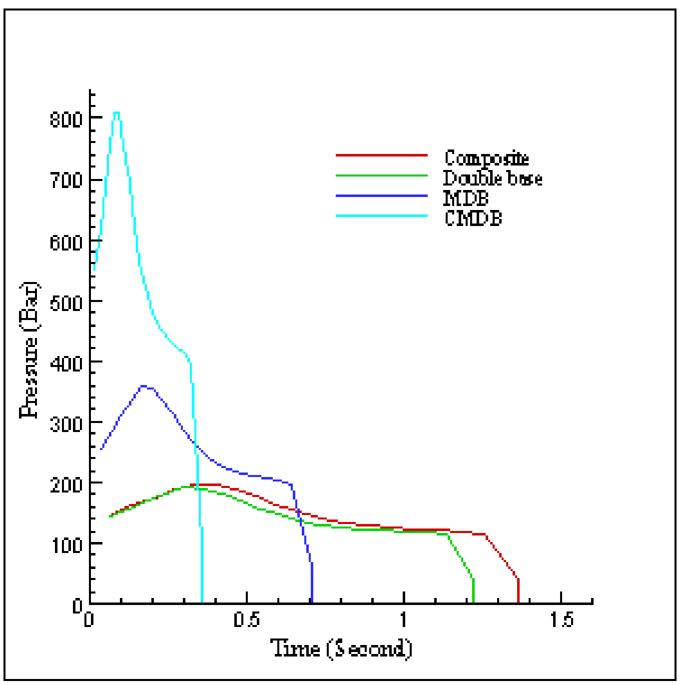


(ii)

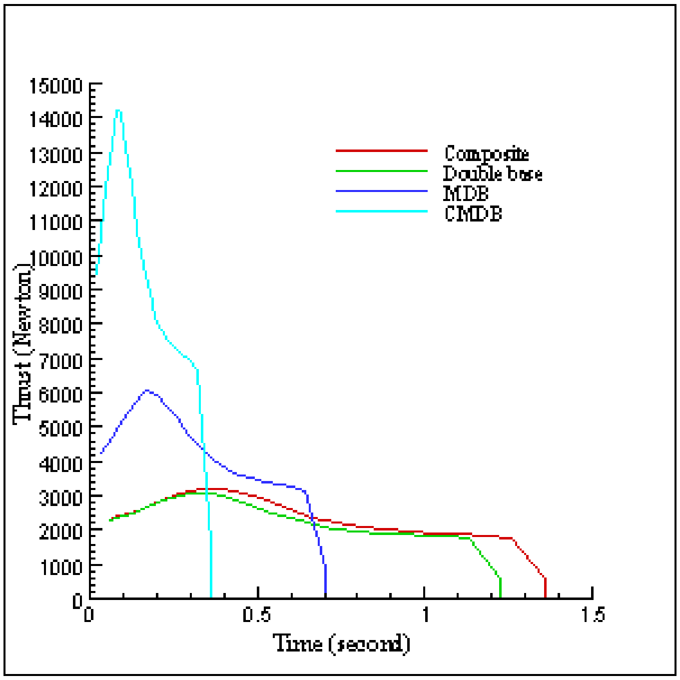


(iii)

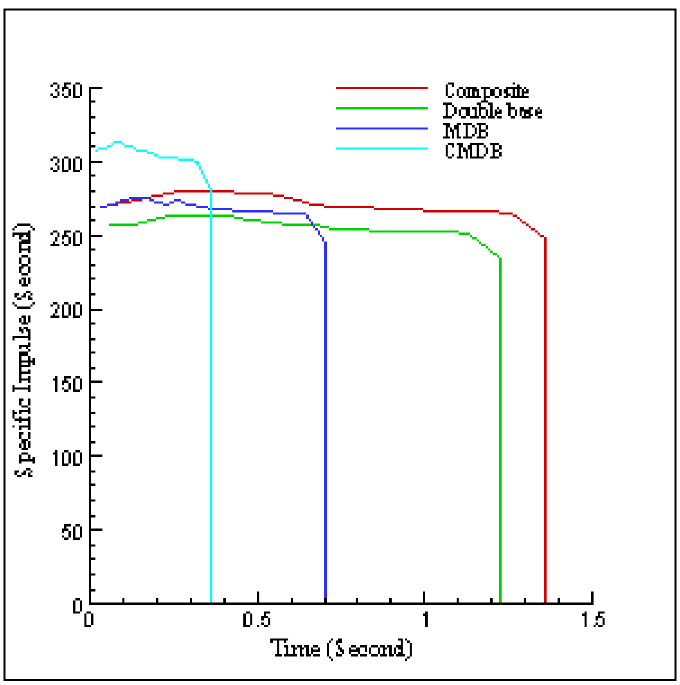
Results of first configuration Figure 5.2 (d)



(i)



(ii)



(iii)

Figure 5.3 Results of different designed grain configuration

These results shows that this propellant grains configuration produces only boost-sustain profile instead of boost-sustain-boost profile. All these grain geometries were simulated using four type of propellants such as DB, MDB, CMDB, Composite. Results comparison shows that composite modified double base propellants are more efficient than other propellants. CMDB propellants produce high thrust in short time compared to other propellants taking longer time and producing low level of thrust.

As results of this configuration are not according to the requirement so other grain configurations are considered to produce boost-sustain-boost profile.

5.2 Second Configuration

In this configuration grain is divided in two portions. In first portion a grain is a combination of two grain shapes which produce boost-sustain profile and second portion possesses a grain which have progressive profile. First portion of grain burns and produce boost-sustain profile, second portion of the grain starts burning when first portion finish burning. Fig. 5.4 is an example of this type of grain configuration. In which first portion possesses a finocyl grain and second portion possesses a cylindrical grain having larger port area. Here finocyl grain upon burning produce a boost-sustain profile and cylindrical grain produce boost profile. Complete combustion provides boost-sustain-boost (thrust-time) profile.

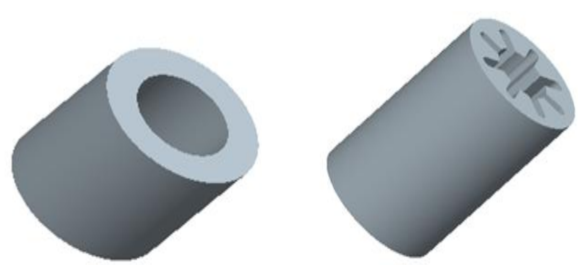
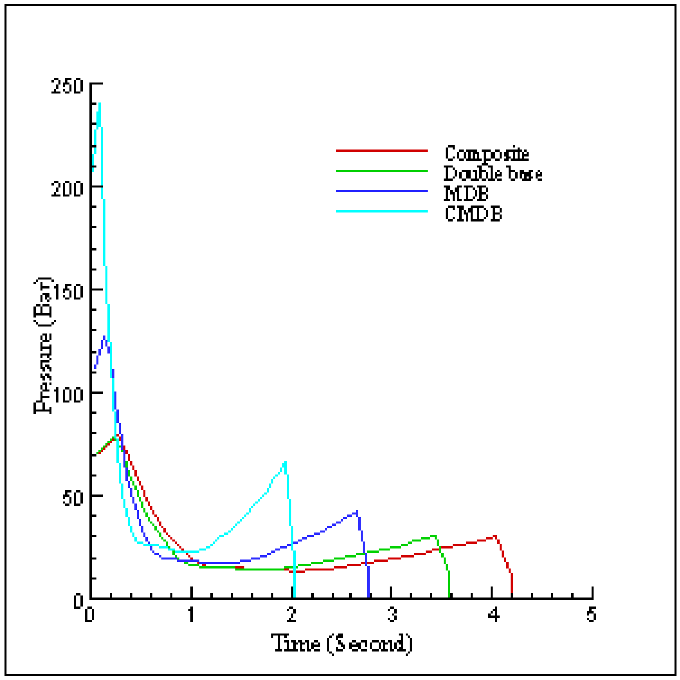
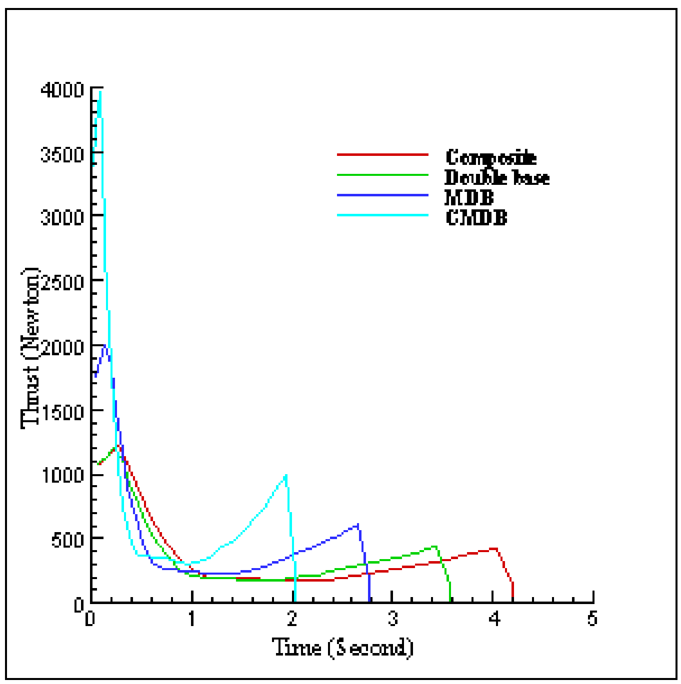


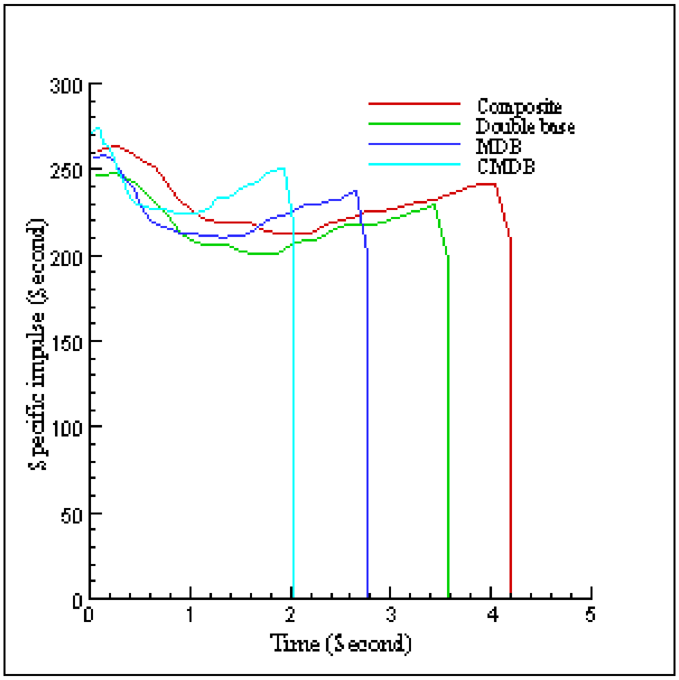
Figure 5.4 Finocyl grain and cylindrical grain


Composite
Double base
MDB
CMDB
Pressure (Bar)
Time (Second)

(i)


Composite
Double base
MDB
CMDB
Thrust (Newton)
Time (Second)

(ii)



(iii)

Figure 5.5 Results of second grain configuration

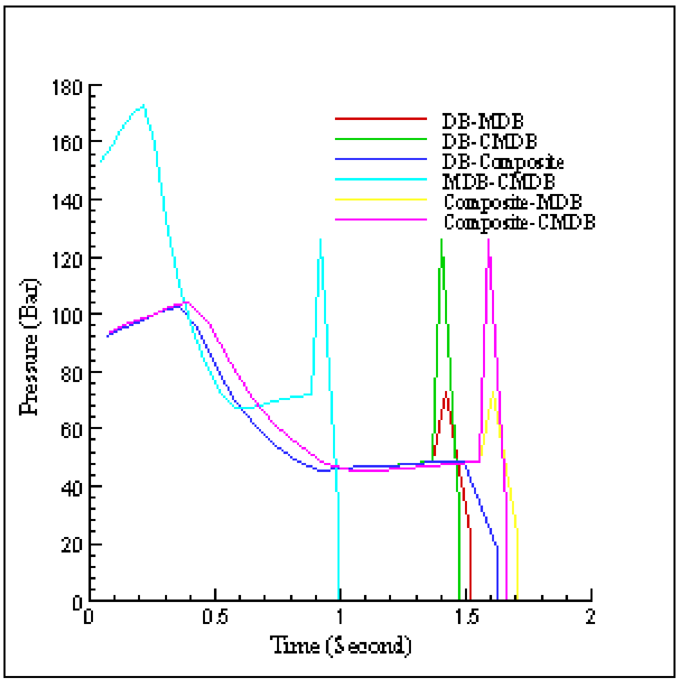
Figure.4 shows the results of this grain configuration. This configuration produces the required level of boost-sustain-boost profile. Comparison of the four types of propellant shows that composite and double base (DB) propellants produce very low level of thrust vs. time and pressure vs. time profile but CMDB show higher pressure and thrust vs. time profile than other three propellants. MDB produce low thrust and pressure vs. time profile which is higher than the DB and composite propellants. Performance parameter (specific impulse) profile shows that CMDB propellants are more efficient than other considered propellants and DB propellants are least efficient. Performance parameters have the almost same value for MDB and composite propellants.

5.3 Third Configuration

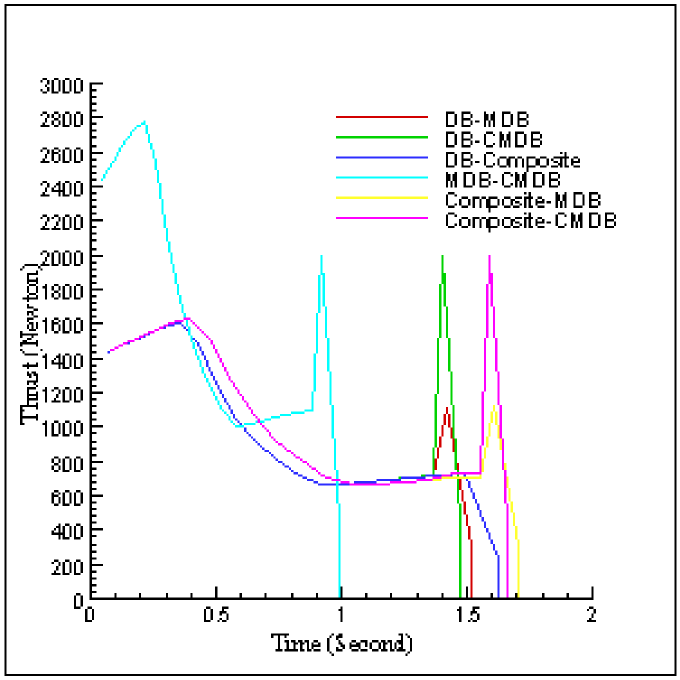
In this type of grain configuration, a grain having boost-sustain profile is used with dual propellant composition to produce boost-sustain-boost profile. In dual propellant composition first higher burn rate propellant and then lower burn rate propellant is used to obtain required level of thrust-time profile. This type of configuration is analyzed for six different propellants combination such as

- Double base and modified double base
- Double base and composite modified double base
- Double base and composite
- Modified double base and composite modified double base
- Composite and modified double base
- Composite and composite modified double base

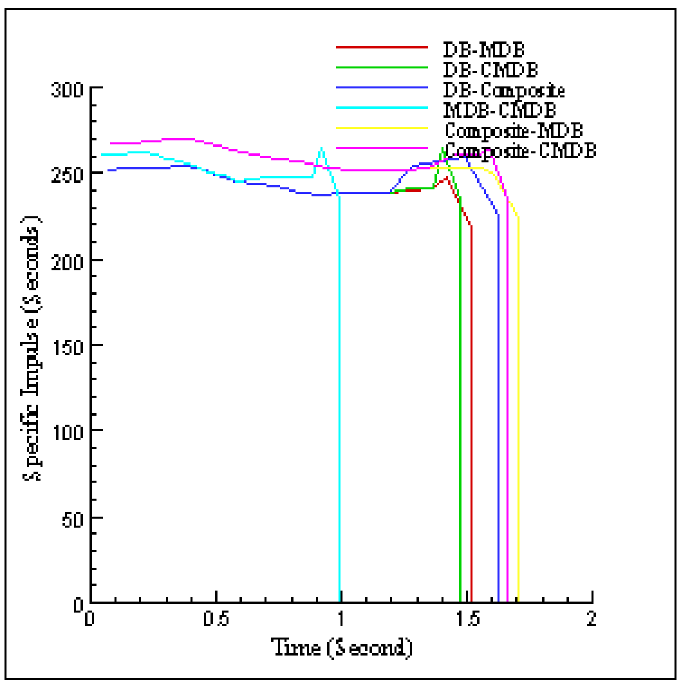
Finocyl grain is used to produce dual thrust. Fig.6 shows the burn back analysis of finocyl grain geometry.



(i)



(ii)



(iii)

Figure 5.6 Result of finocyl grain geometry having dual composition

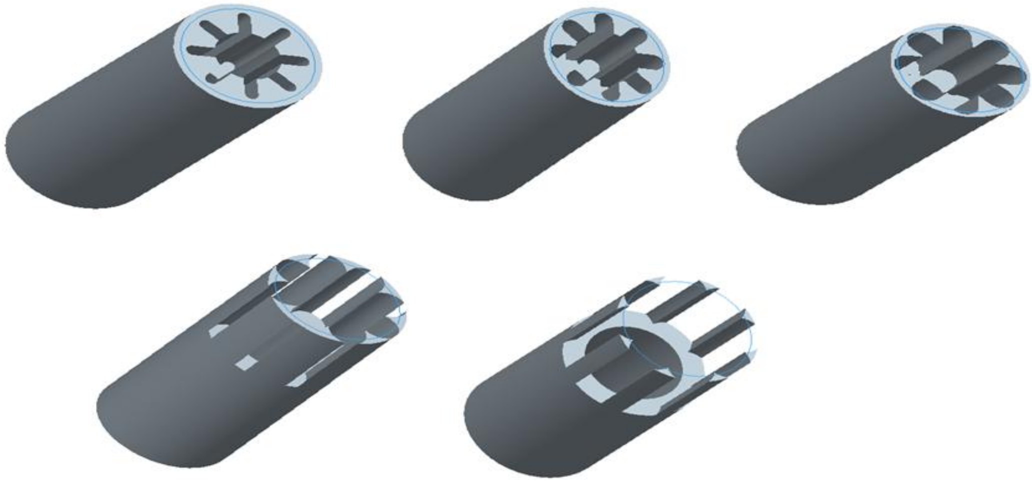


Figure 5.7 Grain burn back analysis for third configuration

Line between propellants grain represents the separation between two propellant compositions having different burning rates. Results shown in Fig. 5.6 are of finocyl grain geometry using six different propellant combinations. This shows that out of six propellant composition combinations MDB and CMDB combination produce best results in short period of time. But DB and CMDB, composite and CMDB combinations produce very low level of thrust-time profile as compared to other combinations. This type of grain configuration produces best results.

Composite modified double base (CMDB) propellants are used in booster, sustainers, space crafts, gas generators and submarine launched ballistic missiles. Double base propellants are used in short action rocket motors with burning time of few milliseconds to about 200 milliseconds [10]. Space shuttle solid rocket boosters use composite propellants also they are used in aerospace propulsion applications where specific impulse of 180-260seconds is desired.

These three grain configurations are combination of mainly two grain shapes one is the star shape grain and other one is the cylindrical shape grain. In general star grain produces neutral burn and cylindrical grain give progressive burn so by combining these two shapes in different ways produce the required thrust-time profile.

5.4 Boost-sustain-boost profile grain design for JPL nozzle

Chang in 1980's examined the effects of two phases with different particle sizes in Jet Propulsion Lab (JPL) nozzle for Titan IV and Titan III solid rocket motor (SRM) nozzle. Results obtained for two phase flow by chang [18-19] are validated by Miss Faiza Uzma MS student of Research center for Modeling and Simulations Nozzle inlet, throat, outlet parameters are shown in Table. 5.2

Table. 5.2 Nozzle different section areas

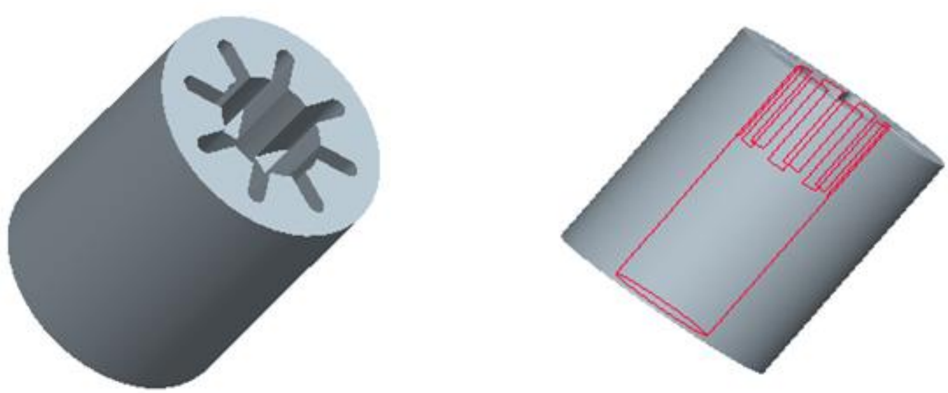
	Inlet	Throat	Outlet
Radius (m)	0.0642	0.020922	0.0531
Area (m^2)	0.01295	0.001375	0.00886

Using inlet radius of the JPL nozzle as the grain outer radius and using web thickness of 35.2mm for cylindrical grain and 17.2mm for star grain. Geometry parameters of finocyl grain are shown in Table. 5.3

Table. 5.3 Finocyl grain of JPL nozzle dimensions

Outer Diameter of grain (mm)	128.4
Port Diameter (mm)	58
Star fin length (mm)	18
Star fin radius of curvature(mm)	5
Star grain length (mm)	50
Cylindrical grain length (mm)	100
Total grain length (mm)	150

JPL nozzle grain design in shown in Figure 5.8



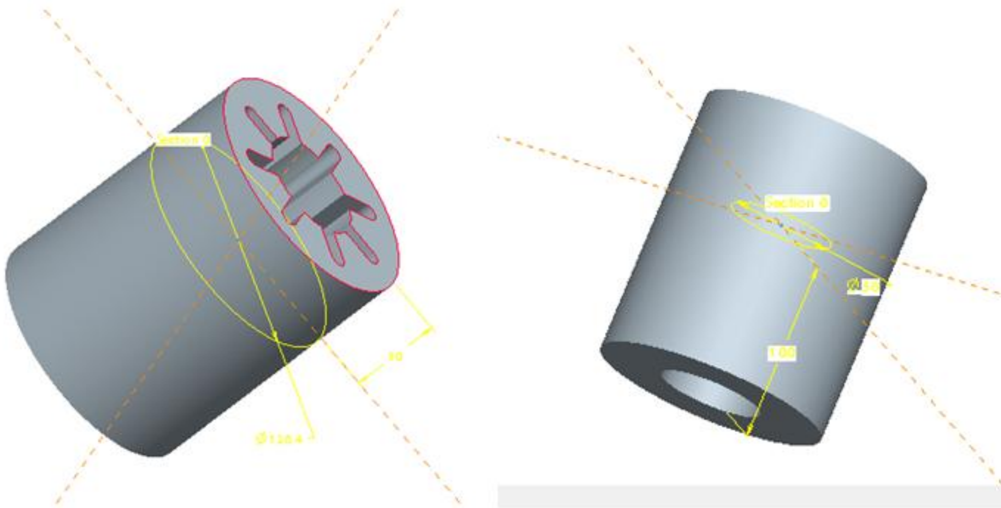
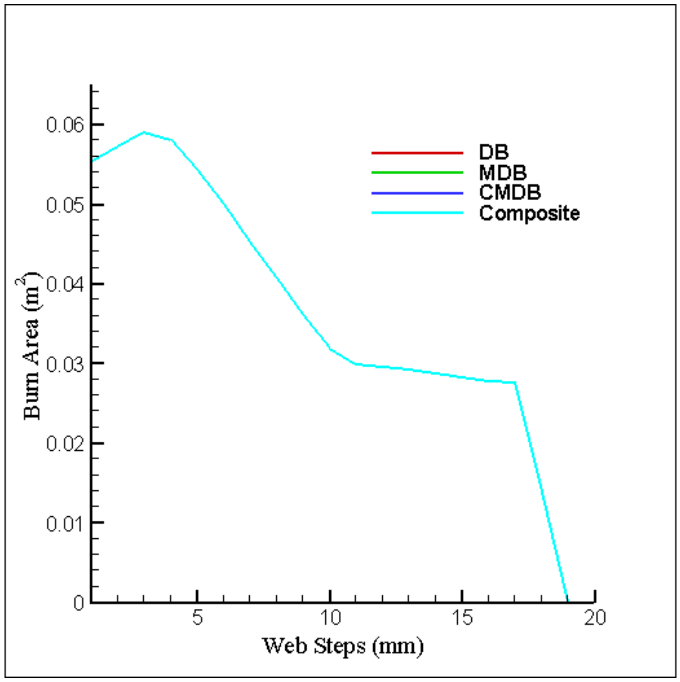


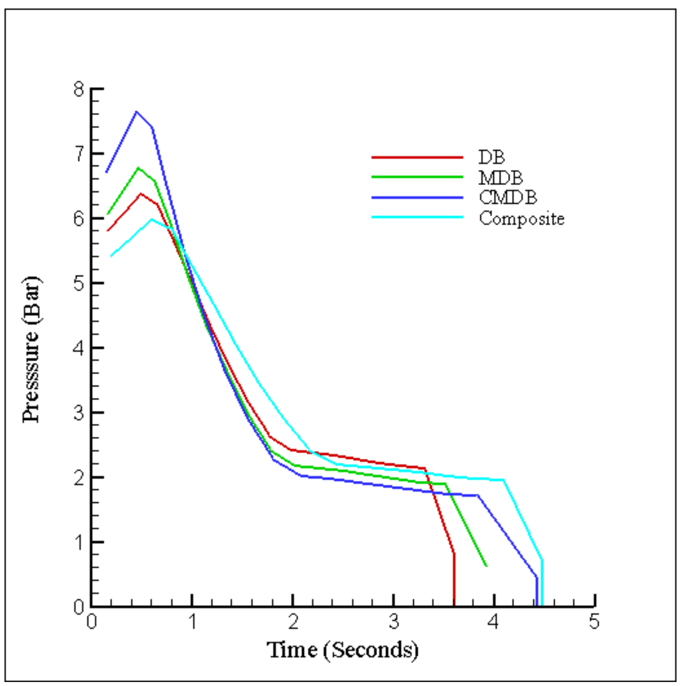
Figure. 5.8 Finocyl grain of JPL nozzle at different orientations

Burning characteristics are calculated using the procedure mention in chapter 5 for four types of propellants (DB, MDB, CMDB and Composite). Graphs are plotted against web step vs. burn area, time vs. Pressure, time vs. burn area, time vs. thrust and time vs. specific impulse. Results for four propellants are shown in Figure. 5.9



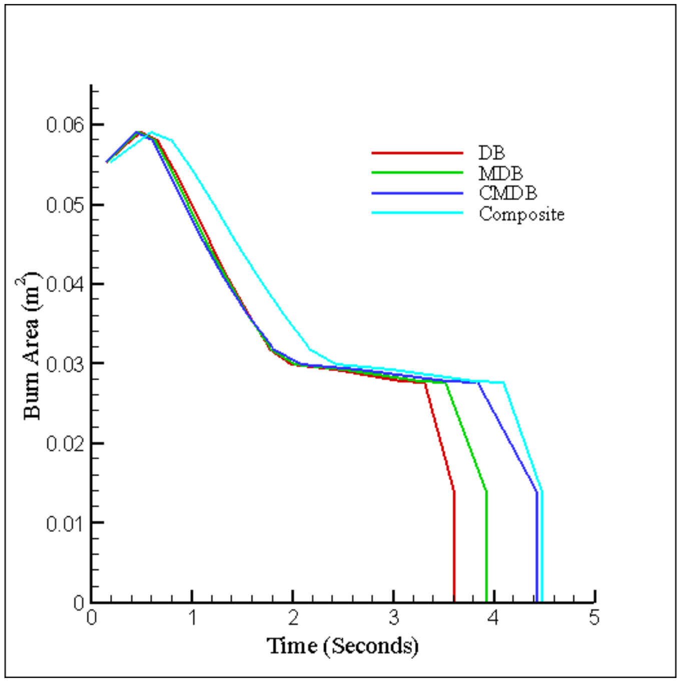
0.06
0.05
0.04
0.03
0.02
0.01
0
5
10
15
20
Burn Area (m^2)
Web Steps (mm)
DB
MDB
CMDB
Composite

(i)

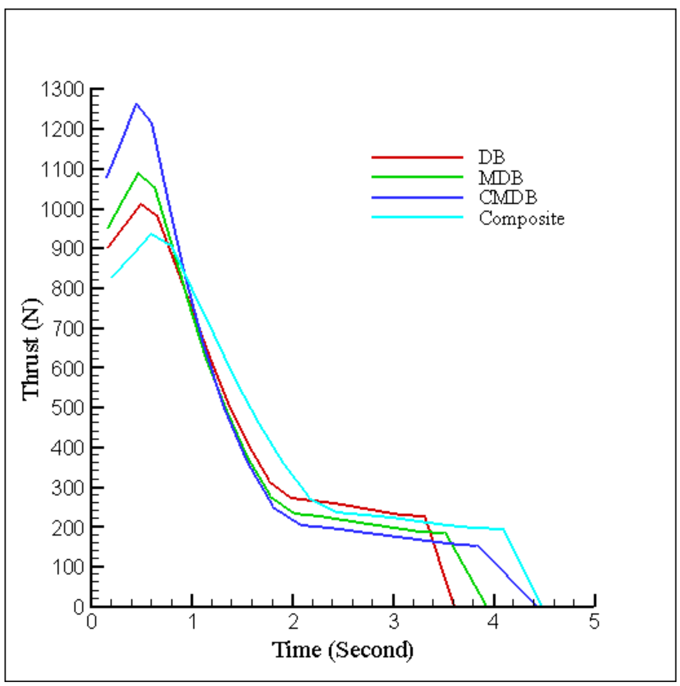


8
7
6
5
4
3
2
1
0
0
1
2
3
4
5
Presssure (Bar)
Time (Seconds)
DB
MDB
CMDB
Composite

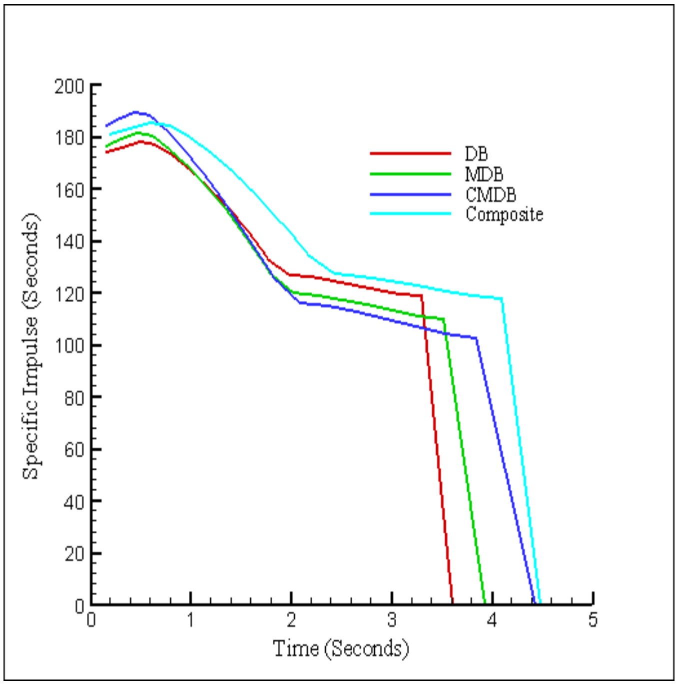
(ii)


0.06
0.05
0.04
0.03
0.02
0.01
0
Burn Area (m^2)
0
1
2
3
4
5
Time (Seconds)
DB
MDB
CMDB
Composite

(iii)


1300
1200
1100
1000
900
800
700
600
500
400
300
200
100
0
Thrust (N)
0
1
2
3
4
5
Time (Second)
DB
MDB
CMDB
Composite

(iv)



(v)

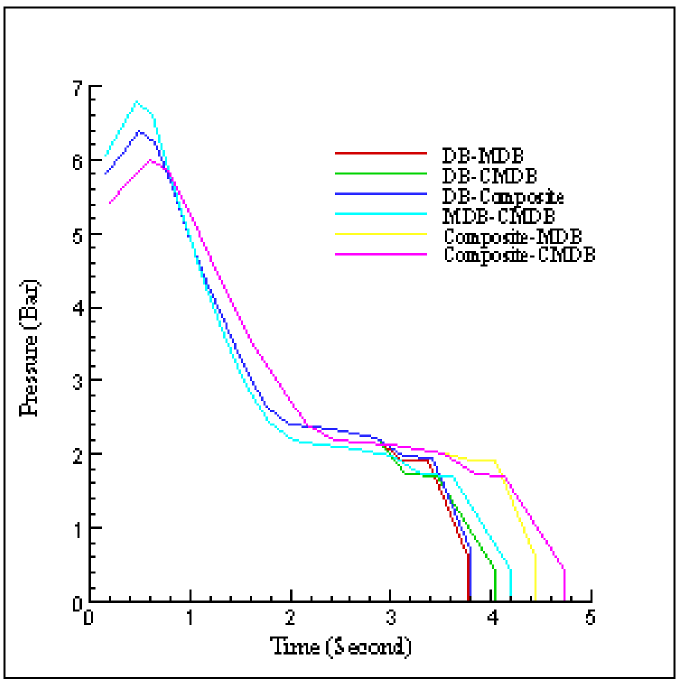
Figure. 5.9 Results of JPL nozzle Finocyl grain

These results shows clearly that using this grain produces a very low level of thrust, pressure and above all the performance parameter (Specific impulse) has a very low value. All propellant profiles are shown and it is clear from graph that composite modified double base propellant produce thrust value of 1270N, pressure value of 7.8bar and specific impulse of 195seconds which are the higher values obtained among all the propellants used. Modified double base produce thrust, pressure and specific impulse values of 6.7bar, 1950N, 185seconds which are higher than double base and composite propellants. From the performance parameters results it is clear that that the CMDB are more efficient propellant but composite propellants are less efficient than CMDB but more efficient than DB and MDB. DB propellant are least efficient than all propellants used.

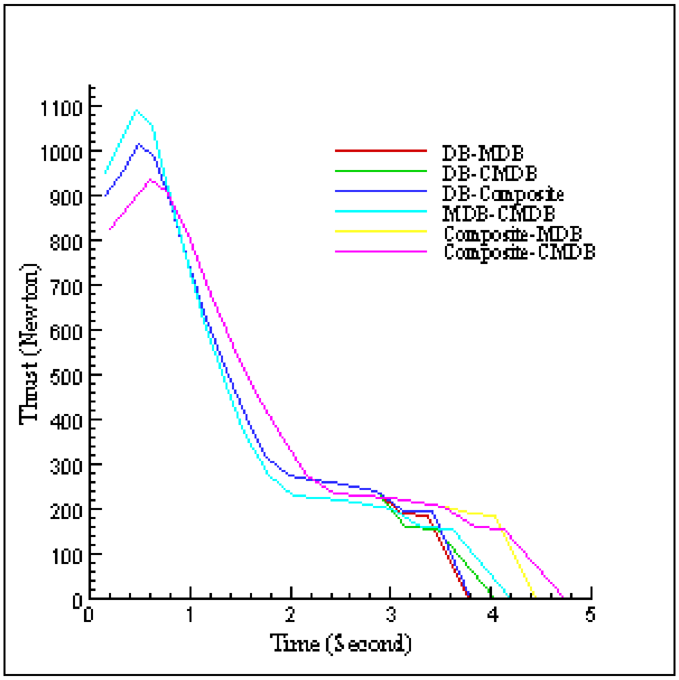
This grain is also tested for producing boost-sustain-boost profile using dual propellant compositions. Propellant combinations are as follows:

- Double base and modified double base
- Double base and composite modified double base
- Double base and composite
- Modified double base and composite modified double base
- Composite and modified double base
- Composite and composite modified double base

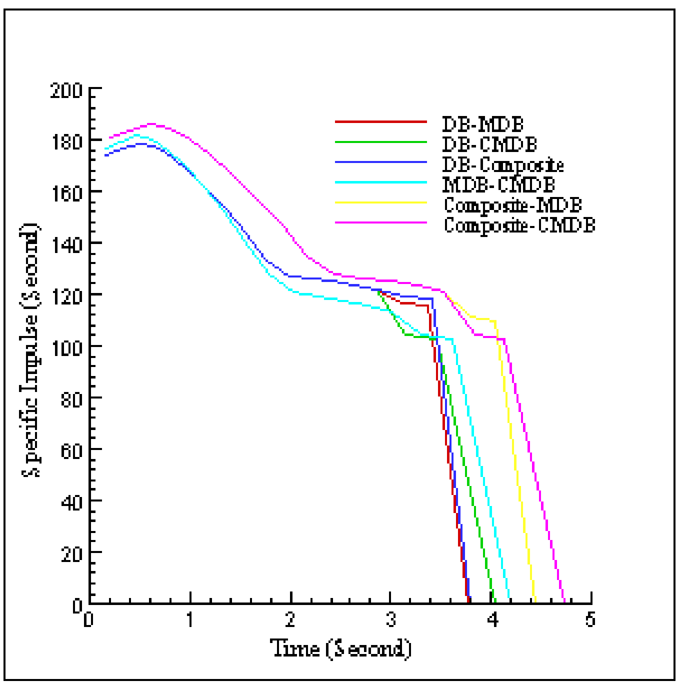
Sing these six combinations in a finocyl grain of JPL nozzle the obtained results fro boost-sustain-boost profile are shown in Figure. 5.10.



(i)



(ii)



(iii)

Figure. 5.10 Results of boost-sustain-boost grain configuration of JPL nozzle

Result obtained from these configurations show that no any one of the propellant combination provide the required boost-sustain-boost profile. This is because this nozzle is of very small size and throat area is very large as compared to the throat area taken from the reference [2] which is large than this nozzle area and provide best results.

5.5 Adiabatic flame temperature results

Results for adiabatic flame temperature calculated for double base, modified double base, composite modified double base using MATLAB are shown in Figure. 5.11 and Figure. 5.12. Code for their calculation are attached in appendix A.

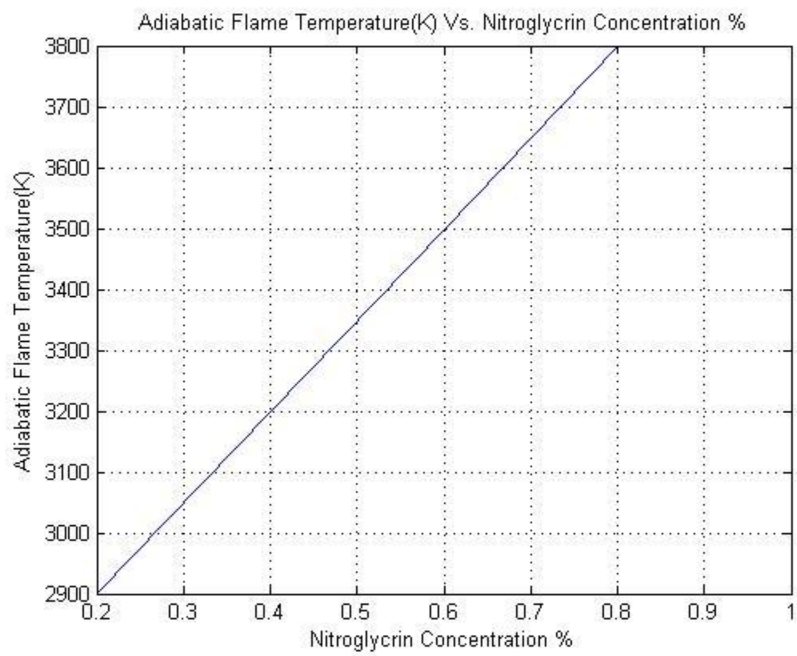


Figure. 5.11 Results for DB propellant

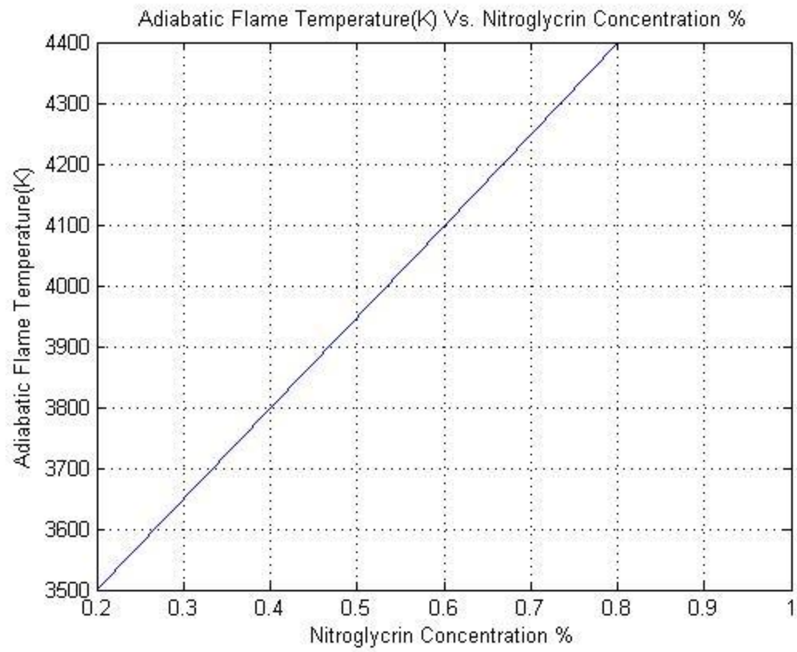


Figure. 5.12 Results for CMDB propellant

These results show that CMDB propellants have high adiabatic flame temperature as compared to double base propellants and MDB propellants the is that CMDB propellants contain metallic elements which are not presents in DB and MDB. This show that metal content presence in propellants increase their efficiency and their adiabatic flame temperature. So to achieve high flame temperature metal contents such as aluminum (Al) are added. In this study aluminum metal is added.

Chapter 6

CONCLUSIONS

Aim of this study is to produce boost-sustain-boost (thrust-time) profile using different grain configurations. Three grain configurations are considered. First one is produced by using single grain having three different burning areas like star-cylinder-star grain of varying length. In the second one, a grain is designed using restrictor which produces a boost-sustain-boost profile by burning in two steps; first step produces boost-sustain profile and second step produce a boost profile (first portion consist of finocyl garin and second portion consist of cylindrical grain). In the third case a grain is used which produces boost-sustain in single burn using one type of propellant. But it also produces boost-sustain-boost profile when dual propellant composition having different burning rates is used.

Four types of propellants are considered in this study double base (DB), composite, modified double base (MDB), composite modified double base (CMDB). Also dual composition of propellants are used all having different burning rates.

Mainly grains used in this study are cylindrical grain and star grain. These three configurations are combination of these grain shapes. Solid modeling of these grains is used as design methodology in Pro/ Engineer software. This software family table provides the relationship between different grain design parameters. By changing the parameters for burn back, new grain model at each web step is created.

Plots of performance parameters such as pressure vs. time, thrust vs time and specific impulse vs. time for these three configurations are compared which shows that the CMDB propellant produces higher thrust and pressure in all the three configurations. There are two factors on which SRM thrust-time profile depends; grain configuration and propellant composition. First two grain configurations are based on the grain configuration factor and third grain configuration is based on the propellant composition factor. Second and third type of grain configuration gives good results to achieve boost-sustain-boost profile.

Finocyl grain designed for JPL nozzle produce very low values of thrust, pressure. Its performance parameter show very low value. Results show that this nozzle is not suitable for producing boost-sustain-boost profile.

Adiabatic flame temperature calculation is done manually and computationally using MATLAB for four different types of propellants against different oxidizer concentrations. Results are compared. From which it is concluded that propellants containing metal contents have high adiabatic flame temperature than those having no metal content.

FUTURE RECOMMENDATIONS:

For future work wagon wheel shape is preferable to obtain boost-sustain-profile by both methods (by changing grain configuration or by dual composition). It can be done by changing grain configuration of wagon wheel, e.g., by changing its L/D ratio and curves ratios or by using dual composition of propellant material

REFERENCES

1. Sutton, G.P. and Biblarz, O., Rocket Propulsion Elements,7th ed., John Wiley and Sons,New York,2001.
2. Acik, S. “Internal Ballistics design Optimization of a Solid Rocket Motor”, May, 2010.
3. “Solid Rocket Motor.”http//en.wikipedia.org/wiki/solid_rocket_motor,en:User: Pbroks13,19 May 2008.
4. Puskulcu, G. “Analysis of 3-D Grain burnBack of Solid Propellant Rocket Motors and Verification with Rocket Motor tests”, August, 2010.
5. http://www.ehow.com/about_5305016-do-rocket-engines-nozzles.html
6. McBride, F.J. “Explosives in aerospace rocketry 6 Solid Propellant Rocket Systems”, 1968.
7. Wipipedia .free enpsyclopedia.
8. NASA., Solid propellant grain and internal ballistics, space vehicle design criteria, March 1972.
9. Pro Engineering, Wildfire 4.0, Parametric Technology Corp., Needham, MA, 2010.
10. http://hmf.enseeiht.fr/travaux/CD0405/optmfn/mci/manuals/tecplot/presenta.htm
11. Swiss propulsion labortary. Internet: www.spl.ch/software/index.html , Jun. 27,2004.
12. G. Püskülcü and A. Ulas, “3-D grain burnback analysis of solid propellant rocket motors: Part2-modeling and simulations ,” Aero Space and Technology, 12(2008), pp. 585–591.
13. Michael A. Willcox,∗ M. Quinn Brewster,† K. C. Tang,‡ D. Scott Stewart,§ and Igor Kuznetsov¶ “Solid rocket motor internal ballistic simulation using three-

dimensional grain burn back."Journal of propulsion and power, vol. 23, No. 3, May-june 2007.

14. H. Shekhar, "Parametric studies on star port propellant grain for ballistic evaluation ." Defence science Journal, vol. 55, No.4, pp.459-469, Oct. 2005.
15. http://en.wikipedia.org/wiki/Adiabatic_flame_temperature
16. H. Muthurajan* and ANG How Ghee, "Software Development for the Detonation Product, Analysis of High Energetic Materials - Part I," Central European Journal of Energetic Materials, **2008**, 5(3-4), 19-35.
17. http://www.qrg.northwestern.edu/projects/vss/docs/propulsion/3-how-you-calculate-specific-impulse.html
18. I.S. Chang, "*One- and two-phase nozzle flows*," AIAA J.vol. 18, pp. 1455-1461, Jan 31, 1980.
19. C. T. Crowe, "*Review-numerical models for dilute gas-particle flows*," ASME, Transactions, Jornal of Fluids Engineering, vol. 104, pp. 297-303, September 1982.

APPENDIX A

Double base propellants adiabatic flame temperature code:

```
clc
clear all
close all
NG=227;          %molecular weight of nitroglycerin
NC=297;          %molecular weight of nitrocellulose
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%concentration of fuel oxidiser and binder
massNC = [ 0.2 0.4 0.6 0.8];
massNG = [ 0.8 0.6 0.4 0.2];
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%heat of formation values of reactants and products
Hfco= -110320;
HfH2o = -241560;
HfN2 = 0;
Hfco2 = -393360;
HfNG = -385900;
HfNC = -772200;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%for loop for getting values of the array
for C = 1:4
molesNC(C) = massNC(C)/NC;    %moles of nitrocellulose
molesNG (C)= massNG(C)/NG;     %moles of nitroglycerin
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
CNC(C) = 6*molesNC(C);
CNG(C) = 3*molesNG(C);
totalC(C)  =   CNC(C)  +  CNG(C);   %total  amount  of  Carbon
contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
HNC(C) = 7*molesNC(C);
HNG(C) = 5*molesNG(C);
totalH(C)  =   HNC(C)  +  HNG(C); %total  amount  of  Hydrogen
contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
NNC(C) = 3*molesNC(C);
NNG(C) = 3*molesNG(C);
totalN(C)  =  NNC(C)  + NNG(C);    %total amount of nitrogen
contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
ONC(C) = 11*molesNC(C);
ONG(C) = 9*molesNG(C)
totalO(C)  =   ONC(C)  +  ONG(C);   %total  amount  of  oxygen
contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
```

```
OXYGEN_Balance(C)    =    (totalO(C)    -    2*totalC(C)    -
(totalH(C)/2))*1600;    %OXYGEN bALANCE

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
 %chemical compounds formed after oxygen balancing
if OXYGEN_Balance(C) >= -40
% using Kistiakowsky Wilson Rule
H2O(C) = totalH(C)/2;
N2(C) = totalN(C)/2;
CO2(C) = totalO(C) - (totalC(C) + H2O(C));
CO (C)= totalC(C) - CO2(C);
else
% using Modified Kistiakowsky Wilson Rule
H2O(C) = totalH(C)/2;
N2(C) = totalN(C)/2;
CO2(C) = 0;
CO(C) = totalC(C);
end
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Heat of combustion value calculation
deltaHc(C)  =(CO(C)*Hfco  +  H2O(C)*HfH2o  +  N2(C)*HfN2  +
CO2(C)* Hfco2)-(molesNG(C)*HfNG + molesNC(C)*HfNC);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Temperature and cp values table reading
A = xlsread('Book1.xls');
difference=150;
i =0;
error(1:61)=0;
x(1:61)=0;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%while  loop  for  the  iteration  of  adiabatic  flame
temperature calculation
while abs(difference) >90
i=i+1;
x(i)=   -((CO(C)*4.184*A(i,5)   +   H2O(C)*4.184*A(i,3)   +
N2(C)*4.184*A(i,4)      +      CO2(C)*4.184*A(i,2))*(A(i,1)-
298.15));
difference = x(i) - deltaHc(C);
error(i)= difference;
difference;
end
Anew(C)=A(i,1);
end
Anew
```

```
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Nitroglycrin Concentration and asiabatic flame temperature

    plot(massNG,Anew),title('ADIABATIC  FLAME  TEMPERATURE(K)
Vs. Nitroglycrin Concentration %')
    xlabel('Nitroglycrin Concentration %')
   ylabel('ADIABATIC FLAME TEMPERATURE(K)')
   grid
end
%%%%%%%%%%%%%%%%5555555555555%%%%%%%%%%%%%%%%%%%%%%%%%%%%
```

Modified Double Base propellants adiabatic flame temperature code:

```
clc
clear all
close all
NG=227;            %molecular weight of nitroglycerin
NC=297;            %molecular weight of nitrocellulose
RDX=222;           %molecular weight of RDX
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%concentration of fuel oxidiser and binder
massNC = [ 0.145 0.345 0.545 0.745];
massNG = [ 0.8 0.6 0.4 0.2];
massRDX =[ 0.055 0.055 0.055 0.055];
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%heat of formation values of reactants and products
Hfco= -110320;
HfH2o = -241560;
HfN2 = 0;
Hfco2 = -393360;
HfNG = -385900;
HfNC = -772200;
HfRDX = -61500;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%for loop for getting values of the array
for C = 1:4
molesNC(C) = massNC(C)/NC;     %moles of nitrocellulose
molesNG (C)= massNG(C)/NG;      %moles of nitroglycerin
molesRDX(C) = massRDX(C)/RDX;    %moles of RDX
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
CNC(C) = 6*molesNC(C);
CNG(C) = 3*molesNG(C);
CRDX(C) = 3*molesRDX(C);
totalC(C) =   CNC(C) + CNG(C) +CRDX(C);   %total amount of
Carbon contents
```

```
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
HNC(C) = 7*molesNC(C);
HNG(C) = 5*molesNG(C);
HRDX(C) = 6*molesRDX(C);
totalH(C) =  HNC(C) + HNG(C) +HRDX(C); %total amount of
Hydrogen contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
NNC(C) = 3*molesNC(C);
NNG(C) = 3*molesNG(C);
NRDX(C) = 6*molesRDX(C);
totalN(C) =  NNC(C) + NNG(C) +NRDX(C);   %total amount of
nitrogen contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
ONC(C) = 11*molesNC(C);
ONG(C) = 9*molesNG(C);
ORDX(C) = 6*molesRDX(C);
totalO(C) =  ONC(C) + ONG(C) +ORDX(C);  %total amount of
oxygen contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
OXYGEN_Balance(C)  =  (totalO(C)  -  2*totalC(C)  -
(totalH(C)/2))*1600;   %OXYGEN bALANCE

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%chemical compounds formed after oxygen balancing
if OXYGEN_Balance(C) >= -40
% using Kistiakowsky Wilson Rule
H2O(C) = totalH(C)/2;
N2(C) = totalN(C)/2;
CO2(C) = totalO(C) - (totalC(C) + H2O(C));
CO (C)= totalC(C) - CO2(C);
else
% using Modified Kistiakowsky Wilson Rule
H2O(C) = totalH(C)/2;
N2(C) = totalN(C)/2;
CO2(C) = 0;
CO(C) = totalC(C);
end
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Heat of combustion value calculation
deltaHc(C)  =(CO(C)*Hfco  +  H2O(C)*HfH2o  +  N2(C)*HfN2  +
CO2(C)*  Hfco2)-(molesNG(C)*HfNG  +  molesNC(C)*HfNC  +
molesRDX(C)*HfRDX);

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Temperature and cp values table reading
```

```
A = xlsread('Book1.xls');
difference=150;
i =0;
error(1:61)=0;
x(1:61)=0;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%while loop for the iteration of adiabatic flame
temperature calculation
while abs(difference) >90
i=i+1;
x(i)=   -((CO(C)*4.184*A(i,5)   +   H2O(C)*4.184*A(i,3)   +
N2(C)*4.184*A(i,4)      +      CO2(C)*4.184*A(i,2))*(A(i,1)-
298.15));
difference = x(i) - deltaHc(C);
error(i)= difference;
difference;
end
Anew(C)=A(i,1);
end
Anew
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

    plot(massNG,Anew),title('ADIABATIC FLAME TEMPERATURE(K)
Vs. Nitroglycrin Concentration %')
    xlabel('Nitroglycrin Concentration %')
   ylabel('ADIABATIC FLAME TEMPERATURE(K)')
   grid

%%%%%%%%%%%%%%%5555555555555%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
```

Composite Modified Double Base propellants adiabatic flame temperature code:

```
clc
clear all
close all
NG=227;          %molecular weight of nitroglycerin
NC=297;          %molecular weight of nitrocellulose
RDX=222;         %molecular weight of RDX
Al=27;         %molecular weight of aluminum
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%concentration of fuel oxidiser and binder
massNC = [ 0.045 0.245 0.445 0.645];
massNG = [ 0.8 0.6 0.4 0.2];
massRDX =[ 0.055 0.055 0.055 0.055];
```

```
massAl = [0.1 0.1 0.1 0.1];

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%heat of formation values of reactants and products
Hfco= -110320;
HfH2o = -241560;
HfN2 = 0;
Hfco2 = -393360;
HfNG = -385900;
HfNC = -772200;
HfRDX = -61500;
HfAl = 0;
HfAl2O3 = -1675700;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%for loop for getting values of the array
for C = 1:4
molesNC(C) = massNC(C)/NC;    %moles of nitrocellulose
molesNG (C)= massNG(C)/NG;     %moles of nitroglycerin
molesRDX(C) = massRDX(C)/RDX;   %moles of RDX
molesAl(C) = massAl(C)/Al;   %moles of Al
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
CNC(C) = 6*molesNC(C);
CNG(C) = 3*molesNG(C);
CRDX(C) = 3*molesRDX(C);
CAl(C) = 0*molesAl(C);
totalC(C)  =   CNC(C)  +  CNG(C)  +CRDX(C)+CAl(C);    %total
amount of Carbon contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
HNC(C) = 7*molesNC(C);
HNG(C) = 5*molesNG(C);
HRDX(C) = 6*molesRDX(C);
HAl(C) = 0*molesAl(C);
totalH(C)  =   HNC(C)  +  HNG(C)  +HRDX(C)+  HAl(C);  %total
amount of Hydrogen contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
NNC(C) = 3*molesNC(C);
NNG(C) = 3*molesNG(C);
NRDX(C) = 6*molesRDX(C);
NAl(C) = 0*molesAl(C);
totalN(C)  =   NNC(C)  +  NNG(C)  +NRDX(C)  +NAl(C);     %total
amount of nitrogen contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
ONC(C) = 11*molesNC(C);
ONG(C) = 9*molesNG(C);
ORDX(C) = 6*molesRDX(C);
```

```
OAl(C) = 0*molesAl(C);
totalO(C)  =   ONC(C)  +  ONG(C)  +ORDX(C)  +OAl(C);    %total
amount of oxygen contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
aluminumNC(C) = 0*molesNC(C);
aluminumNG(C) = 0*molesNG(C);
aluminumRDX(C) = 0*molesRDX(C);
aluminumAl(C) = 1*molesAl(C);
totalAluminum(C)     =aluminumNC(C)     +     aluminumNG(C)
+aluminumRDX(C)  +aluminumAl(C);    %total  amount  of  metal
contents
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
OXYGEN_Balance(C)  =  (totalC(C)-0.5*totalO(C)  +(totalH(C)/4)
+(3/4)*totalAluminum(C))*3200    %OXYGEN bALANCE
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
 %chemical compounds formed after oxygen balancing
if OXYGEN_Balance(C) >= -40
% using Kistiakowsky Wilson Rule
H2O(C) = totalH(C)/2
N2(C) = totalN(C)/2
Al2O3(C) = totalAluminum(C)/2
CO2(C) = totalO(C) - (totalC(C) + H2O(C) + 3*Al2O3(C))
CO (C)= totalC(C) - CO2(C)
else
% using Modified Kistiakowsky Wilson Rule
H2O(C) = totalH(C)/2
N2(C) = totalN(C)/2
CO2(C) = 0
CO(C) = totalC(C)
end
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Heat of combustion value calculation
deltaHc(C)  =((CO(C)*Hfco  +  H2O(C)*HfH2o  +  N2(C)*HfN2  +
CO2(C)*   Hfco2   +Al2O3(C)*HfAl2O3))-(molesNG(C)*HfNG    +
molesNC(C)*HfNC + molesRDX(C)*HfRDX+ molesAl(C)*HfAl);
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%Temperature and cp values table reading
A = xlsread('Book2.xls');
difference=180;
i =0;
error(1:61)=0;
x(1:61)=0;
Anew(C)=0;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
```

```
%while loop for the iteration of adiabatic flame
temperature calculation
while abs(difference) >90
i=i+1;
x(i)=  -((CO(C)*4.184*A(i,5)  +  H2O(C)*4.184*A(i,3)  +
N2(C)*4.184*A(i,4)     +     CO2(C)*4.184*A(i,2)     +
Al2O3(C)*4.184*A(i,6))*(A(i,1)- 298.15));
difference = x(i) - deltaHc(C);
error(i)= difference;
difference;
end
Anew(C)=A(i,1)
end
Anew
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

    plot(massNG,Anew),title('ADIABATIC FLAME TEMPERATURE(K)
Vs. Nitroglycrin Concentration %')
    xlabel('Nitroglycrin Concentration %')
   ylabel('ADIABATIC FLAME TEMPERATURE(K)')
   grid

%%%%%%%%%%%%%%%5555555555555%%%%%%%%%%%%%%%%%%%%%%////en
```