ANALYSIS OF 3-D GRAIN BURNBACK OF SOLID PROPELLANT ROCKET MOTORS AND VERIFICATION WITH ROCKET MOTOR TESTS

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Prof. Dr. Canan ÖZGEN
Director

I certify that this thesis satisfies all the requirements as a thesis for the degree of Master of Science.

Prof. Dr. S. Kemal İDER
Head of Department

This is to certify that we have read this thesis and that in our opinion it is fully adequate, in scope and quality, as a thesis for the degree of Master of Science.

Asst. Prof. Dr. Abdullah ULAŞ
Supervisor

Examining Committee Members

Prof. Dr. Hüseyin VURAL (METU, ME)
Asst. Prof. Dr. Abdullah ULAŞ (METU, ME)
Assoc. Prof. Dr. Cemil YAMALI (METU, ME)
Assoc. Prof. Dr. Suat KADIOĞLU (METU, ME)
Dr. Mehmet Ali AK (TÜBİTAK-SAGE)
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Name, Last name: Gökay PÜSKÜLCÜ

Signature:
ABSTRACT

ANALYSIS OF 3-D GRAIN BURNBACK OF SOLID PROPELLANT ROCKET MOTORS AND VERIFICATION WITH ROCKET MOTOR TESTS

PÜSKÜLCÜ, Gökay
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Supervisor: Asst. Prof. Dr. Abdullah ULAŞ

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Solid propellant rocket motors are the most widely used propulsion systems for military applications that require high thrust to weight ratio for relatively short time intervals.

Very wide range of magnitude and duration of the thrust can be obtained from solid propellant rocket motors by making some small changes at the design of the rocket motor. The most effective of these design criteria is the geometry of the solid propellant grain. So the most important step in designing the solid propellant rocket motor is determination of the geometry of the solid propellant grain.

The performance prediction of the solid rocket motor can be achieved easily if the burnback steps of the rocket motor are known.

In this study, grain burnback analysis for some 3-D grain geometries is investigated. The method used is solid modeling of the propellant grain for some predefined intervals of burnback.

In this method, the initial grain geometry is modeled parametrically using commercial software. For every burn step, the parameters are adapted. So the new
grain geometry for every burnback step is modeled. By analyzing these geometries, burn area change of the grain geometry is obtained. Using this data and internal ballistics parameters, the performance of the solid propellant rocket motor is achieved.

To verify the outputs obtained from this study, rocket motor tests are performed.

The results obtained from this study shows that, the procedure that was developed, can be successfully used for the preliminary design of a solid propellant rocket motor where a lot of different geometries are examined.

Keywords: Solid Propellant Rocket Motors, Grain Geometry, Grain Burnback Analysis, Internal Ballistic
ÖZ

KATI YAKITLI ROKET MOTORU YAKIT GEOMETRİLERİNİN 3 BOYUTLU YANMA GERİLEMESİ ANALİZİNİN YAPILMASI VE ROKET MOTORU DENEMELERİ İLE DOĞRULANMASI

PÜSKÜLCÜ, Gökay
Yüksek Lisans Tezi, Makina Mühendisliği Ana Bilim Dalı
Tez Yöneticisi: Yard. Doç. Dr. Abdullah ULAŞ

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Katı yakıtlı roket motorları özellikle kısa süreli çalışan ve yüksek itki-ağırlık oranlı bir itki sisteme gereksinim duyan askeri uygulamalarda en sıkılıkla kullanılan itki sistemleridir. Bu tarz sistemlerde ihtiyaç duyulan itki miktarı ve süresi çok büyük değişimler göstermektedir.


Katı yakıtlı roket motorlarında yakıt çekirdeği yanma gerilemesi adımları bilindiği taktirde motor performansı kolaylıkla belirlenebilmektedir.

Bu çalışmada çeşitli 2 ve 3 boyutlu yakıt geometrileri için yakıt yanma gerilemesi çözümlemeleri yapılmıştır. Yakıt yanma gerilemesi çözümleri için yakıt çekirdeğinin her yanma adımı için katı modellenmesi yöntemi uygulanmıştır.
Bu yöntemde, yakıt çekirdeğinin ilk geometrisi ticari yazılımlar kullanılarak parametrik olarak modellenmektedir. Daha sonra her yanma adımı için değişime uğrayan parametreler yeniden girilmektedir. Bu sayede eşzamanlı olarak yeni oluşan yakıt geometrisi modellenebilmektedir. Son olarak her yanma adında oluşturulan yeni yakıt geometrileri inceленerek, yakıt geometrisinin yanma alanının değişimi elde edilmektedir. Bu veriler ve çeşitli iç balistik parametreleri kullanılarak roket motorunun performansına ulaşılabilinmektedir.

Bu çalışma sonucunda ulaşılan çıktıların doğrulanabilmesi için, küçük ölçekli roket motoru denemeleri yapılmıştır. Ayrıca elde bulunan bilgisayar kodları da doğrulama amacıyla kullanılmıştır.

Bu çalışmadan çıkan sonuçlar, geliştirilen yöntemin katı yakıtlı roket motoru tasarımının özellikle çok sayıda yakıt geometrisi alternatifinin incelendiği ön tasarım aşamalarında, yeterli hassasiyet ile kullanılabileceğini göstermektedir.

Anahtar Kelimeler: Katı yakıtlı roket motorları, yanma gerilemesi analizi, yakıt çekirdeği geometrisi, iç balistik
To My Family
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NOMENCLATURE

\( a \)  Linear burning rate coefficient, m/s
\( a_m \)  Modified linear burning rate coefficient, m/s
\( A \)  Upstream burning area, m²
\( A_c \)  Port area of given cross section, m²
\( A_b \)  Burning area, m²
\( A_e \)  Nozzle exit area, m²
\( A_t \)  Nozzle throat area, m²
\( c^* \)  Characteristic velocity, m/s
\( C_D \)  Nozzle discharge coefficient, s/m
\( C_f \)  Thrust coefficient
\( c_p \)  Specific heat, J/kg K
\( F \)  Thrust, N
\( g \)  Gravitational acceleration, m/s²
\( G \)  Mass flow rate per unit area, kg/m²
\( G_0 \)  Mass flow rate per unit area limit, kg/m²
\( I_{sp} \)  Specific impulse, s
\( I_{sp,real} \)  Real specific impulse, s
\( J \)  Erosive burning factor
\( K \)  Burning area to throat area ratio
\( K_{sp} \)  Upstream burning area to cross sectional area ratio
\( L_2 \)  Propellant slot length, mm
\( L_{S} \)  Propellant slotted section length, mm
\( m_d \)  Nozzle discharge mass flow rate, kg/s
\( M_w \)  Molecular Weight, kg/kmol
\( n \)  Burning rate pressure exponent
\( N \)  Number of slots on star geometry
\( P_a \)  Atmospheric pressure, Pa
\( P_c \)  
Chamber pressure, Pa

\( P_e \)  
Pressure at the nozzle exit, Pa

\( P_m \)  
Modified chamber pressure, Pa

\( r_b \)  
Burning rate of propellant, \( m/s \)

\( r_{b,e} \)  
Erosive burning rate of propellant, \( m/s \)

\( R \)  
Universal gas constant, \( J/kg \, K \)

\( R_m \)  
Gas constant of product gas, \( J/kg \, K \)

\( R_p \)  
Radius of propellant, m

\( R1 \)  
Propellant grain inner radius, mm

\( R2 \)  
Propellant grain slot tip radius, mm

\( t_b \)  
Burning time, s

\( T \)  
Temperature, K

\( T_0 \)  
Temperature the burning rate is measured, K

\( T_i \)  
Temperature the burning rate is to be calculated, K

\( T_c \)  
Chamber Temperature, K

\( V_a \)  
Available motor volume, \( m^3 \)

\( V_l \)  
Volumetric loading

\( V_p \)  
Propellant volume, \( m^3 \)

\( w_f \)  
Web fraction

\( \gamma \)  
Specific heat ratio

\( \sigma_p \)  
Temperature sensitivity of burning rate, \% \( ^\circ K^{-1} \)

\( \pi_K \)  
Temperature sensitivity of pressure, \% \( ^\circ K^{-1} \)

\( \rho_p \)  
Propellant density, \( kg/m^3 \)

\( \eta_{ia} \)  
Total motor efficiency
1.1 General

In today’s modern warfare, most of the weapon systems used needs some kind of propulsion system that helps them to move from one location to another location. For complex weapon systems like airplanes and so on, a very complex propulsion system is required. The propelling system must be able to run for hours of operation, refueling must be easy, and service life must be very long. For such applications very complex propulsion systems like internal combustion engines or gas turbine engines are used. These systems have a lot of moving parts with very low tolerances, low power to weight ratio and requires complex maintenance for their operation cycle.

For simple weapon systems like artillery rockets or surface to air rockets, cheaper, simpler and maintenance free propulsion system is required. These systems require a single-shot operating propulsion system. After the payload reaches the desired location, the propulsion system explodes with the payload. The rocket motors are the most common used propulsion system for such applications.

A rocket motor is simply an energy conversion system. The chemical energy stored inside the fuel is converted to thermal energy by oxidation, then the produced hot gasses accelerate through the nozzle, giving the rocket motor momentum, thus mechanical work is done [3, 23].

The rocket motor carries its own fuel and oxidizer with itself through the flight. The rocket motor does not need the surrounding air for oxidizing the propellant.
Thus the rocket motor is capable of operating in space and underwater, where there is no free oxygen for the propulsion system to use. This advantage makes the rocket motor, the only alternative for spacecraft propulsion.

1.2 Classification of Rocket Motors

Classifying rocket motors can be done in many ways. But in literature the most common way of classifying is by the state of the fuel and oxidizer used (Figure 1) [3, 23].

![Figure 1 - Rocket Motor Types](image)

The subcategories of rocket motors are as follows:
1.2.1 Solid Propellant Rocket Motors

In solid propellant rocket motors, both the fuel and the oxidizer are in solid state. The oxidizer and the fuel are premixed and usually cast inside the combustion chamber at the production stages. The solid propellant rocket motors don’t need any mechanism for mixing or supplying the mixture to the combustion chamber. Thus they are the simplest type of rocket motors. The solid propellant rocket motors can be used for a wide variety of applications requiring few Newtons to several million Newtons of thrust.

1.2.2 Liquid Propellant Rocket Motors

In liquid propellant rocket motors, both the fuel and the oxidizers are in liquid state. They are stored at different tanks. An injector is used to mix the fuel and the oxidizer at a specific amount for the most effective burning. The liquid propellant rocket motors usually use Hydrogen or small molecule size hydrocarbons which are easy to oxidize, as fuel. For the oxidizer, usually liquid Oxygen is used. The liquid propellant rocket motors give more thrust than the solid propellant rocket motor for the same amount of mass of propellant used. Also liquid propellant rocket motors are capable of on-off controlling or adjusting the thrust level during operation [3, 12]. But the process of mixing right amount of fuel and oxidizer requires very complex parts to be used. Thus liquid propellant rocket motors are much more complex and expensive than solid propellant rocket motors [3]. So the liquid propellant rocket motor is used at systems where more complex and expensive propulsion system is feasible, like space shuttle upper stage propulsion, or satellite orbital transfer propulsion.

1.2.3 Gaseous Propellant Rocket Motors

Gaseous propellant rocket motors are much like the liquid propellant rocket motors. The main difference is that, both the fuel and the oxidizer are at gaseous phase. Since the density of the gaseous stage is much lower than the liquid stage,
the tanks of the gaseous propellant rocket motors, which store the same mass of propellant, must be much bigger. This disadvantage makes gaseous propellant rocket motors the least used type of rocket motors.

1.2.4 Hybrid Rocket Motors

In hybrid rocket motors, usually a solid fuel grain with no or very little amount of oxidizer is placed inside the combustion chamber and the liquid oxidizer is placed at the tank. Since the fuel is very stable without the oxidizer, production, handling and storage of such rocket motors are much easier than solid propellant rocket motors. When the rocket motor is fired, the oxidizer is injected inside the combustion chamber, onto the solid fuel. Thus the oxidization of propellant occurs much controllably and smoothly [3]. Although the hybrid rocket motor concept is old, there are not many applications. But the interests on hybrid rocket motors are rising steadily these days.
CHAPTER 2

SOLID PROPELLANT ROCKET MOTORS

As mentioned before solid propellant rocket motors are the most widely used rocket motors. They are simple, easy to manufacture and cheap. Most solid propellant rocket motors require no maintenance for their entire shelf life. They are usually manufactured for single operation so their reliability is very high. They can reach enormous thrust levels like millions of Newtons. All these advantages make them superior to all other propulsion systems for most military application.

The Solid Propellant Rocket Motors are mainly composed of a combustion chamber, a converging-diverging nozzle, solid propellant, an igniter and if necessary, an insulator. Figure 2 shows the parts of a simple solid propellant rocket motor.

Figure 2 – Solid Propellant Rocket Motor
2.1. Parts of Solid Propellant Rocket Motors

2.1.1. Propellant

As the name implies, solid propellant rocket motors use a fuel-oxidizer mixture, which is called propellant, in solid state for the energy source. The propellants can be subcategorized into two groups by the production method; cast or free standing.

Cast propellants are prepared in mixing chambers in highly viscous liquid form and then poured inside the motor case. The motor is then “baked”, which helps the propellant to plasticise. The mandrel that is placed inside the motor case gives the propellant the desired shape. Then the mandrel is removed, leaving the cavity required for the motor to operate.

Free standing propellants are prepared by the same method as cast propellants, but instead of pouring directly inside the motor case, the propellants are poured inside some molds having the desired geometry. Then the propellants are “baked” like the cast propellants. After this process, the plasticised propellants are placed inside the motor case and fixed using mechanical or chemical methods. Usually this kind of production method is used to obtain large burning areas, therefore larger thrust for shorter time.

Another way of categorizing solid propellants is by chemical composition; double base or composite propellants.

Composite propellants are the most commonly used type of propellants in solid rocket motors. They have good burning stability, high energy and relatively low response to temperature changes. The term composite states that, the propellant is made of two or more kind of substance in a heterogeneous mixture, without any chemical binding between them. Usually an oxidizer in fine powder form is mixed with the fine powder metal particles which are used as energy source. Then this mixture is added to some plasticizing agent which is usually an energy source itself. Then the propellant is “baked” to gain hardness, thus good mechanical properties are obtained. Most commonly ammonium perchlorate for oxidizer, aluminum
powder for energy source and HTPB (Hydroxyl Terminated PolyButadiane) as the plastic fuel binder are used. Aluminum is used because of its high heat of combustion, cheapness and commercial availability. HTPB is a good plastic binder since it has good mechanical properties when it plasticizes (low thermal conductivity and high mechanical strength). By using these substances, large rocket motors with complex grain geometries, having large thrust to weight ratio can be produced.

Composite propellants are usually classified as cast propellants. For the production method, the oxidizer crystals are ground to proper size distribution, containing very fine, medium, and larger particles, and then it is mixed with metal particles of proper nominal size distribution. Next necessary agents for stabilizing the burning rate of the propellant are added to the mixture. Then the plastic binder is added to the mixture, obtaining very high viscous fluid. The thick dough is then poured inside the motor case at vacuum, to obtain bubble formation. Then the motor case is “baked”, which is sometimes called curing, for about a week. The propellant solidifies during this process, retaining the shape of the motor case and the mandrel inside it. After the curing is finished, the mandrel is taken out, leaving the propellant at the desired grain geometry.

Another type of solid propellant is double base propellants. The term double base represents that two kinds of base propellants are mixed to obtain this propellant. In double base propellants, both the oxidizer and the energy source material are the parts of the same molecule chain. Therefore a chemical binding is present between the oxidizer and fuel. The two kind of base propellants usually used are nitrocellulose and nitroglycerine. These two base propellants are mixed together in a solvent to perform a mixture. Generally nitrocellulose is the dominant substance in double base propellants. Also some stabilizing and plastisizing agents are added to the mixture at production stages. In some rare applications, metal additives like aluminum powder for extra energy and external oxidizers like ammonium perchlorate are added to the mixture.

Double base propellants are usually prepared in large mixing chambers with excess amount of solvents like acetone, and then extruded through the die to obtain
desired geometry. After the extrusion process, the propellant is dried to remove the excess solvent. The extruded propellant is then ready to be inserted inside the motor case. The propellant is usually inside the motor case as free standing.

Double base propellants are usually used in smaller rocket motors with less complex grain geometries, giving high thrust for short period of time. Double base propellants are also used widely at gun systems of different caliber. For these applications, usually fine hollow rods are produced and then chopped to obtain fine particles.

Obtaining large and complex grain geometries with double base propellants is not easy. First of all, the extrusion process for three dimensional geometries is nearly impossible. Also as the size increases, the equipment for production and handling of the propellant gets more complex. For these reasons, double base propellants are not used widely on larger rocket motors.

2.1.2. Motor Case

The motor case of solid propellant rocket motors has two main duties. First one is that the motor case holds all the other parts of the rocket motor, the propellant, the igniter, the nozzle, the insulating layers and the necessary apparatus for joining the rocket motor with the rest of the missile. The second one is, it is the combustion chamber, where the propellant burns to generate hot gasses.

The motor case is nearly always cylindrical. In some applications, spherical motor case is used. This has two main reasons. First of all, the best geometry for high pressure vessels is cylindrical or spherical geometries. And second is that the geometry of the rest of the missile is cylindrical.

Since the motor case is also the combustion chamber, the case is subjected to very high pressures like 100 bar for larger rocket motors, and even higher for smaller rocket motors. Also high temperatures in the absence of good thermal insulation are received at operation. These severe conditions require high strength materials to be used in production of motor cases. Usually high strength alloy
aluminum, heat treated high strength steels or fiber reinforced composite is used. Since aluminum has lower melting temperature, it is used in rocket motors with short burn times or there must be given extra attention to the insulation. Fiber reinforced composites have advantages of lowering weight, having good thermal insulation and easing to reach insensitivity. But for manufacturing costs and chemical stability problems, steel is the most common used material for motor cases.

The motor case is usually tested hydrostatically before the assembly. Usually 150 to 200% of the maximum expected working pressure is used for medium size motors. For larger motors, the number goes down to reduce cost and weight of the motor case.

The motor case is also designed for satisfying the requirements coming from high inertial forces at the operation.

2.1.3. Igniter

In order to reduce the risk of misfiring of rocket motors, the propellant used is usually chosen to be not very sensitive to external disturbances. Thus a more sensitive igniter is used to fire rocket motors. The igniter usually consists of an electrically ignited primary charge, and a more energetic secondary charge, which ignites the solid propellant.

The ignition process of a solid propellant rocket motor is a complex process that involves combustion, heat transfer and fluid flow [22]. The process starts with the electrical signal reaching the primary charge which is usually named as “squib”. Squibs are pyrotechnic elements composed of a resistive wire and very sensitive energetic material. The resistive wire gets hot as the electrical charge passes through. As the wire gets hot enough, the particles of energetic materials that are touching the wire ignite. These ignited particles produce enough heat to fire the rest of the energetic material. The heat generated by firing of the squib is then transferred to the secondary ignition charge that is less sensitive. When enough heat
is transferred, secondary charge ignites. The secondary charge can be in solid or powder form.

By the time the secondary charge is ignited, the flame, hot gasses and some burning particles are dispersed into the motor case, onto the solid propellant. When enough heat is transferred to the solid propellant surface, the surface of the propellant ignites.

The above explained process requires some time since combustion, flow and heat transfer processes all require some time to perform. The overall time required for the solid propellant to ignite after the ignition signal receiving the squib is called ignition lag or ignition delay.

The secondary (main) charge used at igniters can be a solid propellant which is more sensitive than the main solid propellant used in motor, or an energetic powder like black powder, nitrocellulose or nitroglycerine. To enhance the heat transfer from the igniter to the solid propellant, some energetic materials like MTV (Magnesium/Teflon/Viton) are also used as the secondary charge. These materials not only produce hot gasses, but also buildup some condensed-phase particles that are at elevated temperatures. These particles flow through the motor case and stuck onto the solid propellant. So the heat can be transferred much efficiently to the solid propellant by conduction.

2.1.4. Insulation and Liner

The temperature of the product gasses inside the motor case can reach above 3000°C when the solid propellant burns. These high temperature gasses are exhausted through the nozzle to the ambient atmosphere. But this process requires some time and during this process, some heat is transferred to the subcomponents of the rocket motor. In some cases the temperature of the motor case or any other subcomponent can reach the melting temperature of the material the component is made of. This situation can lead to catastrophic failure of the rocket motor. To prevent this, some kind of insulation is applied to rocket motors.
The primary place for insulation is usually between the solid propellant and the motor case. For long burning free standing grain geometries, overheating of motor case is very critical. So, more emphases are given to insulation in these motors.

EPDM (Ethylene Propylene Diene Monomer) is a typical insulator material used at rocket motors. It has low thermal conductivity, high heat capacity and is capable of ablative cooling.

For case bonded cast propellants, there is a need of good adhesive properties between the motor case and the solid propellant. This is usually obtained by installing some liner material between the motor case and the solid propellant. This layer also provides a barrier for the migration of chemicals from the solid propellant to the motor case, or vice versa.

In some cases, the liner can provide sufficient insulation that the need for extra insulation layer vanishes.

2.1.5. Nozzle

In a rocket motor, the thrust is obtained by discharging high energy product gasses through the nozzle. The performance of the rocket motor depends mostly on how much of the total energy of the product gasses are converted into useful work through the nozzle. So the nozzle is a very important subcomponent of a rocket motor.

In a converging-diverging nozzle, the high pressure product gas in the combustion chamber accelerates through the converging part of the nozzle. While accelerating, the pressure of the gas drops. Finally the velocity of the gas reaches the speed of sound at the throat of the nozzle. After this, the gas further accelerates through the diverging part, reducing the pressure. Finally the gas is discharged to atmosphere through the exit of the nozzle. The discharged gas has a velocity greater than speed of sound, thus this type of nozzles also called “supersonic nozzles”. The high velocity discharged gas gives momentum to the rocket motor.
Nozzles can be classified according to the shape of the contour, or their structural mounting technique.

The most common type of nozzles is the bell shaped fixed nozzle, which are efficient and easy to manufacture.

Another important subject about rocket motor nozzles is the material. The product gas of the solid propellant rocket motor is at very high temperatures, 3000 °K or higher, and contains some very hard particles like metal oxides. A material that can withstand high temperatures and erosive characteristic of exhaust gas is very hard to find. Usually carbon containing composites or graphite is used at the throat of the nozzle, where the risk is the highest. Also oxide ceramics and some hardened plastics are used in some applications. In some very short burn time rocket motors, even steel nozzles can be used without much performance loss. But for longer burning rocket motors, the deformation of the throat of the rocket motor exceeds 5% in terms of area, which performs a great performance loss [24]. So advanced materials mentioned above are used in these rocket motor nozzles.
CHAPTER 3

SOLID PROPELLANT ROCKET MOTOR DESIGN METHODOLOGY
AND GOVERNING EQUATIONS

Design of a solid propellant rocket motor starts with a mission requirement. The mission requirement is simply what is expected from the rocket motor. The time of operation, the thrust level, the operating environment, the geometrical constraints and so on are given to the designer. The designer’s duty is to buildup such a rocket motor to satisfy all the needs that are given to him.

The solid rocket motor designer plays with some parameters to obtain the requirements. These parameters are called ballistic parameters. The ballistic parameters are either the properties of the solid propellant; burning rate, temperature sensitivity, propellant density, specific impulse, temperature and ratio of specific heat of products; properties arising from the mission requirements; thrust, duration of operation, total impulse, length to diameter ratio and ambient pressure; grain geometry related; volumetric loading, web fraction, sliver fraction; nozzle geometry related; throat area; and some combined; pressure, port to throat ratio. Sometimes some of these parameters can not be changed. In literature these parameters are divided into more subcategories like dependent and independent parameters, but it is sometimes very difficult to decide this dependency [2, 8, 9]. So in this study, that kind of categorization is not used.
3.1 Ballistic Parameters

The above mentioned parameters will be explained in detail in the following section. More emphases will be given to the ones that will be used later in this study. For some of the parameters, only a brief definition will be given. To explain some parameters, other parameters will be used.

3.1.1 Thrust

The thrust of a solid propellant rocket motor is the magnitude of force that the motor applies to the rocket system. The thrust is the main design constraint of a propulsion system. It can be calculated by using:

\[ F = C_f P_c A_t \]  

(3.1)

Where \( C_f \) is the thrust coefficient, \( P_c \) is the chamber pressure and \( A_t \) is the throat area.

3.1.2. Thrust Coefficient

\( C_f \) is a nondimensional parameter that depends only on the combustion gasses specific heat ratio, \( \gamma \), expansion ratio of the nozzle, and the ratio of chamber and ambient pressures. \( C_f \) gives the efficiency of a nozzle for a given propellant and nozzle geometry.

It is expressed as;

\[
C_f = \sqrt{2 - \frac{\gamma^2}{(\gamma-1)(\gamma+1)}} \left[ \frac{2}{(\gamma+1)} \right]^\gamma \left[ 1 - \left( \frac{P_e}{P_c} \right)^{\frac{\gamma-1}{\gamma}} + \frac{(P_s - P_a)}{P_c} \left( \frac{A_s}{A_t} \right) \right]^{\frac{\gamma+1}{\gamma}}
\]

(3.2)
The thrust coefficient is highest when the exit pressure is equal to the ambient pressure. But this is not the case for most solid propellant rocket motors for some design constraints. The ambient and chamber pressure, thus the exit pressure changes during flight, so exit to ambient pressure ratio of one can not be maintained during the operation.

3.1.3. Chamber Pressure

Chamber pressure is the pressure inside the motor case during the operation of the solid propellant rocket motor. The chamber pressure is not constant during the whole operation for most rocket motors. Also for relatively long motors (large length to diameter ratio) the chamber pressure varies from forward end to aft end of the rocket motor due to flow effects. Since the chamber pressure is the main parameter affecting the thrust of the rocket motor, it is one of the most useful parameter that the designer modifies to reach the thrust level required. Usually the maximum operating pressure is fixed initially, and then the chamber pressure is left free to modifications.

The maximum operating pressure is fixed due to mechanical strength of the motor case and other components. It is the designer’s duty to stay below the limit during the whole operating cycle of the rocket motor.

Sometimes the term, average operating pressure of a rocket motor, is used at the early stages of the design. This term is the average of the chamber pressure during the full operating cycle. When a constant thrust level rocket motor is to be designed, the average pressure is calculated first. Then depending on the grain geometry, the real pressure curve is obtained. The curve must have an average equal to the average operating pressure.
3.1.4. Nozzle Throat Area

Another factor affecting the thrust is the nozzle throat area. The throat area is fixed at the design stages of the rocket motor. So during operation, thrust can not be altered using throat area. The nozzle throat area is used for calculation of many other parameters like port to throat ratio, expansion ratio and so on.

3.1.5. Expansion Ratio

The expansion ratio is the ratio of the area of the nozzle exit plane to the area of the nozzle throat. The expansion ratio defines how much the pressure of the product gas is reduced before it is exhausted. The expansion ratio is optimal when the exit pressure of the rocket motor, that is the pressure of the product gas at the nozzle exit plane, is equal to the ambient pressure. By satisfying this condition, maximum useful work can be obtained from the product gas before they are discharged to environment. If the expansion ratio is further increased, then the discharged gas has a lower pressure than the ambient pressure, causing a shock to be formed at the nozzle exit which is not wanted. This kind of nozzles is called over expanded nozzles. If the expansion ration is left smaller than the optimum value, the nozzle is called under expanded. Some of the energy is discharged to the environment before it is converted to useful work. For most rocket motor applications, a slightly under expanded nozzle is used to prevent over expansion to occur during the operation.

The expansion ratio for the optimum expansion can be calculated by using the following formula:

\[
\frac{A_t}{A_e} = \left[ \frac{\gamma + 1}{2} \right]^{\frac{1}{\gamma - 1}} \left[ \frac{P_e}{P_c} \right]^{\frac{1}{\gamma - 1}} \sqrt{\gamma + 1} \left[ 1 - \left( \frac{P_e}{P_c} \right)^{\frac{\gamma - 1}{\gamma}} \right] \tag{3.3}
\]
3.1.6. Nozzle Discharge Coefficient

The gas discharged from the nozzle exit to the ambient atmosphere is described by;

\[ \dot{m}_d = C_D A_r P_C \]  \hspace{1cm} (3.4)

Where, \( C_D \) is the nozzle discharge coefficient. In an ideal rocket motor, \( C_D \) depends on only the nature and the temperature of the combustion gases. Theoretically, \( C_D \) is given as;

\[ C_D = \sqrt{\gamma \left[ \frac{2}{\gamma+1} \frac{(\gamma+1)}{\gamma-1} \frac{1}{R} \frac{M_w}{T_C} \right]} \]  \hspace{1cm} (3.5)

Where, \( \gamma \) is the specific heat ratio, \( T_C \) is the chamber temperature and \( M_w \) is the molecular weight of the combustion gases, and \( R \) is the universal gas constant.

3.1.7. Characteristic Velocity

The reverse of the nozzle discharge coefficient is called the characteristic velocity of the nozzle. It can be formulized by;

\[ c^* = \frac{1}{C_D} = \frac{P_C A_r}{\dot{m}} \]  \hspace{1cm} (3.6)

It is simply used instead of nozzle discharge coefficient in some literature, but the main aspect is not different.
3.1.8. Burning Rate

The burning rate of a solid propellant is the distance the propellant surface regresses due to burning of the surface material in a unit time. The physics of solid propellant burning states that the burning action takes place only on the surface of the propellant. The process is very complex, but if we simplify it, the solid propellant particle at the surface of the propellant that is exposed to the atmosphere gasifies due to the heat transfer from the flame. Then the gasified propellant particles burn in a very short distance, giving more heat to the remained propellant surface. This process goes continuously until all the propellant is burned out. The speed of the burning surface regression is called burning rate of the propellant, $r_b$.

The burning rate of the propellant depends on several factors. The chamber pressure is the dominant factor affecting the burning rate. For a fixed solid propellant formulation the burning rate of the propellant is defined by the power law;

$$r_b = a P^n$$

(3.7)

Where “a” is the burning rate coefficient and “n” is the pressure exponent. The “a” and “n” are estimated by testing of the propellant at different pressures and then fitting a straight line on the log-log graph of burning rate versus pressure. These tests are usually performed by using test setups called strand burners [19]. In general, these setups record the time for a constant length propellant strand to burn at different pressures, reaching the burning rate. Some propellants require more than one line fitting for different pressure ranges. For this kind of propellants, several “a” and “n” values are used during the ballistic calculations, depending on the chamber pressure. Another way of introducing burning rate into ballistic calculations is to use the burning rate that is tested at each pressure level directly, without applying a curve fit. But this kind of approach both requires a lot of test data and makes the ballistic calculations more complex. Also the propellants with lower “n” value make the ballistic equations more stable.
The temperature is another factor affecting the burning rate of a propellant. This effect is usually expressed as temperature sensitivity of the solid propellant. Usually the propellants with lower temperature sensitivity are preferred in solid rocket motors but it is not possible all the time.

The effect of temperature cannot be expressed independent of the pressure. Their effect is coupled and can be expressed in two forms:

\[
\sigma_p = \frac{1}{r_b} \left[ \frac{\partial T_b}{\partial T} \right]_p \\
\pi_K = \frac{1}{P} \left[ \frac{\partial P}{\partial T} \right]_K
\]

The first one, \(\sigma_p\) is the sensitivity of burning rate expressed in percent change in burning rate per degree change in propellant temperature at a particular value of chamber pressure. The second one, \(\pi_K\) is the temperature sensitivity of pressure expressed in percent change of chamber pressure per degree change in propellant temperature at a particular value of K, which is the ratio of the burning surface area to the throat area.

By using the above equations, the modified burning rate coefficient and chamber pressure can be calculated as follows:

\[
a_m = a \left[ e^{\sigma_p (T_i - T_0)} \right] \\
P_m = P \left[ e^{\pi_K (T_i - T_0)} \right]
\]

Where \(T_i\) is the initial temperature at which the burning rate is to be calculated and \(T_0\) is the temperature at which the original burning rate calculations are made. The process of obtaining \(\sigma_p\) and \(\pi_K\) is not very easy, so usually the temperature sensitivities of propellants are obtained by performing full scale rocket motor tests.
The other factors affecting the burning rate are acceleration, manufacturing techniques and internal flow. The acceleration effects are important for acceleration levels higher than 10g directed into the propellant burning surface. Since the propellant grains are usually cylindrical in general form, the linear acceleration of the rocket system has very low effect on the burning rate. But for motors having high spin rate, acceleration effects should be considered.

The manufacturing effects arise mostly on composite propellants where the propellant and oxidizer are in a non-uniform mixture form. For such propellants, the distribution of oxidizer and fuel particles is very dependent on the manufacturing technique. Some parts of the propellant have higher oxidizer to fuel ratio than other parts. Also the oxidizer and fuel particles are combination of different sized particles. The distribution of different sized particles in the propellant grain causes the burning rate to vary from point to point. This kind of variation is usually unpredictable, so the manufacturing techniques of the solid propellant should be very standardized. Also sometimes migration of some particles from one location of the grain to another location occurs during long term storage of the rocket motors. Some precautions should be applied to rocket motors to prevent this. For double base propellants, this migration problem can lead to unstable burning of the propellant, causing a rapid destruction of the rocket motor.

The flow related effects arises from the changes the product gasses made to the combustion process of the solid propellant. The flow related effects are very significant in some cases. So this subject will be discussed in detailed in the erosive burning section.

3.1.9. Density

Density of a propellant is an important factor when the space available for the propellant is limited. The denser the propellant, the more propellant mass can be stored in the same volume. Or the same amount of denser propellant can be fit into the same chamber with more freedom of grain geometry. The designer has more grain geometry alternatives, and the port area, which will be discussed later on,
could be greater, resulting less erosive burning characteristics, which will be discussed later also. But the density of the propellant alone is not a very important parameter. The amount of energy that can be obtained from a unit volume is more important than the mass of unit volume itself. Usually the density is a positive parameter affecting this parameter since more mass can be stored in the same volume and the total impulse obtained from the unit mass of propellant, which is specific impulse and will be discussed on the next section.

3.1.10. Specific Impulse

Specific impulse is one of the most important properties of a rocket motor. Specific impulse is the amount of impulse that can be achieved by consuming unit amount of propellant. The total impulse that can be given to the rocket system by the rocket motor can be calculated by using the specific impulse and the propellant amount. It is also a measure of the rocket motors efficiency. \( I_{sp} \), the specific impulse of a rocket motor can be calculated as follows;

\[
I_{sp} = \sqrt{\frac{2\gamma}{\gamma - 1} R_m \frac{T_C}{M_w} \left[ 1 - \left( \frac{P_e}{P_C} \right)^\gamma \right] + \left[ \frac{\gamma + 1}{2(\gamma - 1)} \right] \frac{R_m T_C}{\gamma M_w} \left( \frac{P_e - P_a}{P_C} \right) A_e A_t} \]  
(3.12)

Where \( R_m \) is the gas constant, \( M_w \) is the molecular weight and \( \gamma \) is the specific heat ratio of the combustion products.

Since the above given formula is for ideal case, the real \( I_{sp} \) is always less than the one calculated by the above formula. The above formula assumes the product gasses to act like ideal gasses. Also some inefficiency arises from the nozzle flow. Also for metalized propellants the unburned metal particles and the produced metal oxides also reduces the \( I_{sp} \). Unburned particles are quiet important for relatively short rocket motors, since the propellant particles do not have enough time to burn up fully before they are discharged through the nozzle. If all the
inefficiencies are summed and the total efficiency is represented with $\eta_{\mu}$, the real $I_{sp}$ of the motor will be as follows:

$$I_{sp,\text{real}} = \eta_{\mu} I_{sp}$$ \hspace{2cm} (3.13)

The value of $\eta_{\mu}$ varies from 0.93 to 0.96 for modern solid propellant rocket motors [3].

3.1.11. Volumetric Loading Fraction

Solid propellant rocket motors can only function if there is a surface of propellant that is open to the internal cavity of motor case. This is needed in order to start and sustain the burning of the propellant. So, all solid propellant rocket motors have a void volume inside the motor case. The amount of void space is critical for a rocket motor since the dimensional constraints are usually very strict for aerial systems. To analyze this criterion, a parameter called volumetric loading fraction is used. Volumetric loading fraction, $V_l$, is defined as the ratio of the propellant volume to the total available chamber volume.

$$V_l = \frac{V_p}{V_s}$$ \hspace{2cm} (3.14)

The larger the volumetric loading fraction, the more propellant is stored in the same volume. Thus without changing the $I_{sp}$ of the system, the total impulse can be increased. But on the contrary, usually the burning area of the propellant decreases as the volumetric loading increase. So, volumetric loading fraction of 1 is not the best value. Usually 0.75 to 0.85 volumetric loading is used for tactical missile rocket motors [3].
3.1.12. Web Fraction

Web fraction is the ratio of the thickness of the propellant to the grain outer radius (Figure 3). It is the parameter that controls the burn time of rocket motor. Since the thickness of the propellant equals to the burning rate times the burning time, web fraction can be formulized as:

\[
w_f = \frac{\text{Web thickness}}{\text{Radius}} = \frac{r_w t_b}{R_p}
\]  

(3.15)

Figure 3 - Web Fraction

3.1.13. Sliver Fraction

A typical solid propellant rocket motor has a thrust versus time curve having some ignition hump at the beginning and a tail off region at the end of its operation. The tail off region is the region where the thrust level drops significantly in a short time, and then a very low level of thrust compared to the normal thrust of the motor lasts for a while, slowly but continuously decaying. In this tail off region the thrust
level is so little that it is usually referred as the lost thrust. The reason for such a behavior is the decrease of the burning area. In some grain geometries this decreased area burning lasts for a short time. In some grain configuration it can last remarkably long.

The measure of this lost thrust region is given by a parameter, named sliver fraction. Sliver fraction is defined as the ratio of the volume of the propellant that is left when the chamber pressure drops below some level to the initial volume of the propellant. Figure 4 explains the sliver fraction more significantly. The burning area drop is very rapid at some point of burnout, when the propellant on some parts of the motor case has finished, while on some regions there is still some propellant to burn.

Sliver fraction is important for two major reasons. The first one is the loss of thrust. Some useful propellant is burned in very low pressures that no useful thrust can be achieved. The second one is the thermal load increase on the motor case during the sliver burning. The propellant itself is a very good insulator. It protects the surrounding motor case from the thermal and corrosive effects of the hot
product gases. But during the sliver burn phase, some regions of the motor case are open to the product gases, resulting a rapid increase of the thermal and structural loads on the motor case.

3.1.14. Erosive Burning

The increase in the propellant burning rate due to axial gas flow inside the combustion chamber is known as erosive burning. It occurs when high velocity combustion gas flows parallel to the propellant burning surface. Erosive burning concept has a complex physical background. The increase of the heat transfer to the propellant surface due to increase of the flow velocity is one of the main causes. To handle erosive burning, two factors, J and $K_p$, are introduced. The factors are defined as;

$$J = \frac{K_p}{K}, \quad K_p = \frac{A}{A_c}, \quad K = \frac{A_b}{A_t}$$

(3.16)

Where $A_c$ is the area of a given cross section of the central port, $A$ is the propellant burning area upstream of the cross section, $A_b$ is the propellant burning area, and $A_t$ is the nozzle throat area.

By using the above two factors and the burning rate of the propellant with an empirical table, Table 1, the erosive burning probability is estimated.
Table 1 - Erosive Burning Aptness [4]

<table>
<thead>
<tr>
<th>J</th>
<th>K</th>
<th>Erosive Burning</th>
<th>Pressure Drop</th>
</tr>
</thead>
<tbody>
<tr>
<td>&lt;0.2</td>
<td>&lt;50</td>
<td>No</td>
<td>Low</td>
</tr>
<tr>
<td></td>
<td>50-100</td>
<td>Yes when ( r_b &lt; 10 \text{ mm/s} )</td>
<td>(Less than 5%)</td>
</tr>
<tr>
<td></td>
<td>100-150</td>
<td>Yes when ( r_b &lt; 20 \text{ mm/s} )</td>
<td></td>
</tr>
<tr>
<td></td>
<td>&gt;150</td>
<td>Yes, very important when ( r_b &lt; 10 \text{ mm/s} )</td>
<td></td>
</tr>
<tr>
<td>0.2 to 0.35</td>
<td>&lt;50</td>
<td>No</td>
<td>Intermediate</td>
</tr>
<tr>
<td></td>
<td>50-100</td>
<td>Yes when ( r_b &lt; 10 \text{ mm/s} )</td>
<td>(About 10%)</td>
</tr>
<tr>
<td></td>
<td>100-150</td>
<td>Yes when ( r_b &lt; 20 \text{ mm/s} )</td>
<td></td>
</tr>
<tr>
<td></td>
<td>&gt;150</td>
<td>Yes, very important when ( r_b &lt; 10 \text{ mm/s} )</td>
<td></td>
</tr>
<tr>
<td>0.35 to 0.5</td>
<td>&lt;50</td>
<td>Yes when ( r_b &lt; 10 \text{ mm/s} )</td>
<td>Intermediate</td>
</tr>
<tr>
<td></td>
<td>50-150</td>
<td>Yes when ( r_b &lt; 20 \text{ mm/s} )</td>
<td>(About 10%)</td>
</tr>
<tr>
<td></td>
<td>&gt;150</td>
<td>Yes, very important when ( r_b &lt; 10 \text{ mm/s} )</td>
<td></td>
</tr>
<tr>
<td>0.5 to 0.8</td>
<td>&lt;150</td>
<td>Yes, very important when ( r_b &lt; 20 \text{ mm/s} )</td>
<td>High</td>
</tr>
<tr>
<td></td>
<td>&gt;150</td>
<td>Yes, very important when ( r_b &lt; 10 \text{ mm/s} )</td>
<td>(Up to 40%)</td>
</tr>
<tr>
<td>0.8 to 1</td>
<td>Any value</td>
<td>Yes, very important when ( r_b &lt; 20 \text{ mm/s} )</td>
<td>Very High</td>
</tr>
</tbody>
</table>

Also there are some semi empiric erosive burning laws used to calculate the increase in the burning rate due to erosive burning. One of them is:

\[
r_{b,e} = r_b \left[ 1 + \left( G - G_0 \right) \right]
\]  

(3.17)

Where \( r_b \) is the propellant burning rate without erosive burning, \( r_{b,e} \) is the propellant burning rate with erosive burning effects, \( G \) is the mass flow rate per unit area for the port cross section, and \( G_0 \) is the mass flow rate per unit area limit before the erosive burning starts. The \( G_0 \) value can be obtained from experiments, or
previously designed and fired rocket motors [13, 20]. These semi empiric models require a lot of past knowledge on rocket motors.

There exist some more detailed studies on erosive burning characteristics of solid propellants, where more detailed analysis can be done with less past knowledge. These studies include both numerical analysis and experimental verification of the analysis. The study of Türker Güdü is an example of such detailed study in Turkey [20]. This study includes a numerical approach used to predict erosive characteristics, verified by experimental data. This study is very helpful for the designer to predict the amount of erosive burning, at a newly developed rocket motor.

Predicting the amount of erosive burning is very important at the final stages of the design. Erosive burning results in higher pressure and thrust than the expected ones. This increase can be damaging to the rocket motor itself or other components of the rocket system. Also, the increase of the burning rate with erosive burning causes the propellant at the aft end of the rocket motor to burn up before the head end, increasing the thermal loads at the aft end of the motor case. To overcome this problem, insulation at the aft end is usually thickened.

Since the erosive burning is maximum when the port area is minimum, the highest burning rate change is seen usually at the ignition period and early stages of operation. The port area increases as the propellant burns, resulting in the decrease of the erosive burning behavior. Rocket systems usually require high thrust at the very beginning of the firing. Erosive burning helps the designer in this aspect.

Generally to avoid unwanted erosive burning of propellant in a rocket motor, low length to diameter ratio, high port to throat area ratio motors are preferred. Also the larger the burning rate of the propellant, the less sensitive it is to erosive burning. But these conditions can not be satisfied all the time. The diameter is usually prefixed by the mission requirements, and the length is determined by the total impulse needed. The port area can not be increased very much, since the volumetric loading decreases. Using tapered geometries with larger port area at the aft end, where the erosive burning is critical, is a good solution for erosive burning
reduction without reducing the volumetric loading. Also fast burning propellants can not be used for long burning tactical solid propellant rocket motors.

The erosive burning characteristic of a composite propellant is given on Figure 5. As seen from the figure, the maximum and average pressure both increases as the motor length increases. But the area below the pressure time curve, therefore the total impulse does not change very much.

![Figure 5 - Pressure rise due to erosive burning](image)

3.2. Simplified Internal Ballistic Design of Solid Propellant Rocket Motors

The performance of a solid propellant rocket motor can be calculated easily if some necessary simplifications are done. Especially for the preliminary design stages of the rocket motor, these simplifications are acceptable. At the final analysis
of the selected design, internal ballistic calculations can be done without these
simplifications. But for early stages, the time consumption of no simplified analysis
is not acceptable.

The major assumptions taken are listed below.

- The product gas is assumed to be constant property, homogenous ideal gas.
  \( P = \rho RT, \quad C_p = \left[ \gamma / (\gamma - 1) \right] R \)
- Both the length to diameter ratio being less than 10 and the star shaped part
  having large port area at the nozzle exit region satisfies the non-erosive
  burning requirements. The pressure and burning rate is assumed to be
  constant everywhere at a given time.
- Steady state burning is assumed. For each burn step, the given thickness of
  propellant burns, then the whole chamber pressure stabilizes. The next burn
  step starts from the stabilized state.
- No frictional or heat transfer losses are assumed along the chamber and the
  nozzle contour.
- The volume increase inside the motor case due to solid propellant surface
  regression is neglected.

With the help of these assumptions, simplified zero dimensional internal
flow is solved.

The motor case is a closed chamber with one opening, the nozzle throat, to
the surrounding environment. Therefore the mass balance inside the motor case is
given by,

\[
\dot{m}_{\text{generated}} + \dot{m}_{\text{in}} - \dot{m}_{\text{out}} = \dot{m}_{\text{change}} \cong 0 \Rightarrow \dot{m}_{\text{generated}} + \dot{m}_{\text{in}} = \dot{m}_{\text{out}} \quad (3.18)
\]

The net change inside the motor case is zero. And since the motor case has
no opening for the inlet, the equation further simplifies to,

\[
\dot{m}_{\text{generated}} = \dot{m}_{\text{out}} \quad (3.19)
\]
The generated mass is the product gas of the burning solid propellant. The generation rate is simply the net burning area times the burning rate times the density of the propellant,

\[ m_{\text{generated}} = \dot{r}_b \rho_p A_h \]  

(3.20)

On the other side, the mass discharged to environment from the nozzle throat is the mass out,

\[ m_{\text{out}} = P_c A_{m} C_d \]  

(3.21)

If we put these terms into the mass balance,

\[ \dot{r}_b \rho_p A_h = P_c A_{m} C_d \]  

(3.22)

The burning rate of the propellant can be shown as,

\[ \dot{r}_b = a P_c^n \]  

(3.23)

The above equation is a semi-empirical equation, where the “a” and “n” are obtained from experiments. The “a” and “n” values for the propellant used for this study are given for the pressure unit of bar. But the rest of the equations are all use pressure unit of Pascal. So the burning rate is modified for Pascal unit of pressure.

\[ \dot{r}_b = a (P_c / 10^5)^n \]  

(3.24)

Introducing the burning rate into the mass balance equation, the below formula for the chamber pressure at a given burning area, propellant properties and nozzle parameters can be achieved.
\[ P_c = \left( \frac{a \rho \mu A_0 c^*}{A_n 10^{\text{bp}}} \right)^{\frac{1}{3}} \]  

(3.25)

The \( C_d \) is replaced by \( 1/c^* \) for harmony with the literature.

With the help of the above equation, the chamber pressure for every burn step can be calculated. The burn step is the change of the grain geometry after a constant thickness of propellant burns out. The time required for that burn step can be calculated easily by dividing the fixed thickness to the burning rate of the propellant at that burn step.

\[ \Delta t = \frac{\Delta x}{\dot{r}_b} \]  

(3.26)

When the time required for every burn step is obtained, the total time of burn can be achieved. The pressure versus time of operation is therefore obtained.
CHAPTER 4

GRAIN BURNBACK

Grain burnback analysis is the determination of the change in the solid propellant grain geometry during the operation of the rocket motor. The grain geometry changes due to the burning of the propellant from the surface. The change of the geometry causes the burning area, therefore the propellant amount that is exposed to the inner cavity, to change. As explained in the previous chapter, the pressure of the rocket motor can be calculated if the burning area is known.

The burnback analysis is a pure geometrical analysis. The internal ballistic calculations or parameters are not an input for the burnback analysis. The geometry deforms regardless of the flow inside the chamber or thermal effects. The obtained data from the burnback analysis is an input to the performance prediction analysis.

From the burnback analysis, except the pressure, some very useful data can be obtained. These are,

- Mass of the propellant remaining, therefore the mass of the rocket motor.
- The sliver fraction, therefore the tail off period.
- The place and amount of expected thermal loading.
- The fragmenting propellant particles, either from stress or geometric reasons.
- The port area for every burn step, therefore the erosive burning characteristic.

The above extra data is very helpful when designing a rocket motor. Not only performance wise but also stability and functionality wise.
Grain burnback analysis is usually done by using numerical methods [2, 4]. In these methods, grain surface regression is computed by using some numerical algorithms. These methods do not need to divide the grain geometry into simple solids. The complex geometry can be modeled as a piece and burnback analysis is performed. The main disadvantages of such methods are the numerical errors involved and time required for analysis.

Another method for burnback analysis is, using analytical methods [1]. In these methods, usually the dimensional parameters are adapted for every burn step. The most popular code using analytical methods is SPP, a code developed in USA [14, 15 and 16]. This code and most other analytical methods divide the grain geometry into multiple simple geometries like cube, sphere, torus or pyramid. By this way the dimensional parameters are easier to modify during analysis.

At TÜBİTAK-SAGE many grain burnback codes were developed in recent years. These codes were 1 or 2 dimensional codes. They were developed for specific grain geometries like star shape or wagon wheel. Some of the codes were capable of solving so called “quasi three dimensional” geometries. “Quasi three dimensional” term is used for 3 dimensional geometries that can be either divided into two or more 2-D sections or the change in the third dimension is very simple, like the taper angle of the grain. But none of these codes were capable of solving fully 3-D grain geometries.

The most recent study on grain burnback subject is done by Cengizhan Yıldırım, as his Master of Science study [2]. In this study, WAGON2D and STAR2D codes were developed for 2 dimensional grain geometries, wagon wheel and star. Also the quasi three dimensional CELLQ3D code and its variant FINOCYL code were developed for analyzing three dimensional grain geometries. These codes were limited to approximate solving of tapered two dimensional geometries and simple finocyl geometry.
4.1. Geometric Limitations

While the propellant burns, the surface of the grain regresses into the grain in the direction normal to the propellant surface. This phenomenon sometimes causes some problems. The most common one is the round edge formation on the concave cusp parts. Also the round edge termination on convex round edges is a problem on some geometry (Figure 6) [1, 2].

![Figure 6 - Cusp round edge formation](image)

4.2. Grain Geometries Used At Solid Propellant Rocket Motors

For different applications, different types of thrust-time profiles are required. If the properties of the propellant are fixed, the main parameter affecting the thrust-time profile is the grain geometry. The change in the grain geometry during operation of the rocket motor causes the burn area to change, therefore the thrust of the motor changes. This thrust change is negligible for neutral burning rocket motors. If the thrust level increases during the operation, the motor is named progressive burning. If the thrust level decreases during the operation, then the rocket motor is named regressive burning. If the thrust level is very high at the
beginning, then after some time, a rapid decrease of thrust level occurs, and the operation continues with the reduced thrust, the motor is named boost and sustain. This kind of thrust profile is very useful for applications requiring high thrust while accelerating at the beginning, then very low thrust to overcome the drag force and sustain the achieved velocity. Space shuttle launch rockets and guided airborne missiles are good examples.

4.2.1. End Burning Cylinder

The simplest grain geometry is the end burning cylinder. The propellant is either cast into a cylindrical cavity, the motor case or a propellant tube, or extruded through a cylindrical hole. Then the cylindrical side is inhibited. Head end flat side is also inhibited. After the inhibition, the propellant can only burn from the aft end flat side, which is left uninhibited. The burning area for such grain geometry remains constant along the operation of the rocket motor. The thrust time profile of such motors is given in Figure 7. This profile can sometimes lead to progressive side, since the wall affects on the sides of the propellant causes the propellants flat face to become conical. The main advantage of end burning motors is the high volumetric loading. But the thrust level is usually low compared to the size of the motor. Also the thermal loading on motor case walls is very high on such rocket motors.
4.2.2. Side Burning Cylinder

Side burning cylindrical grain geometry is very similar to the end burning geometry. The main difference is that, both the aft and head end of the geometry is inhibited instead of circular side. The propellant burns in the radial direction only. The burning area, therefore the thrust of the motor, decreases as the diameter of the propellant decreases. This kind of geometry can not be applied to cast composite propellants since the outer burning surface can not be achieved during the casting process. Usually multiple rods of side burning cylindrical grain made of double base propellants are used for high thrust, short operating motors.

Another alternative of side burning cylindrical grain geometry is the hollow cylinder geometry. This kind of geometry is widely used at composite propellant rocket motors. The propellant is cast into the chamber with a cylindrical mandrel placed concentrically. Then the mandrel is taken out, leaving a hollow cylinder. The burning area is the inside cylindrical wall of the grain. The burning area, therefore the thrust, is progressive on these kinds of grain geometries (Figure 8).
advantage of such grain geometry is high volumetric loading and ease of manufacture and analysis. The burnback analysis can be done analytically, without the need of any tools.

The third alternative of side burning cylindrical grain geometry is the hollow cylinder with both the inner and outer surfaces uninhibited. This grain geometry gives a neutral burning characteristic to the rocket motor, since the area increase at the inner surface is equal to the area decrease on the outer surface. Again these kinds of grain geometries can not be applied to cast composite propellants.

Figure 8 - Thrust-Time Profile of Circular Hallow Grain

4.2.3. Star Shape

The burning area that can be obtained from end or side burning circular grains is limited by the diameter of the rocket motor. In order to obtain higher thrust levels with slow burning propellants, the burning area should be increased. The simplest way of increasing the burn area is inserting some radial slots, either smooth
or sharp edged, to the cylindrical grain. The obtained geometry is called slotted cylindrical or star shaped. The parameters defining the geometry can be seen in Figure 9.

The thrust time profile of a star shaped grain can be regressive, neutral or progressive depending on the parameters defining the geometry. Star shaped grains are used very widely on rocket industry, especially on artillery rockets.

The volumetric loading is fairly good, but the sliver ratio is high on star shaped grains, so extra insulation should be considered.
4.2.4. Wagon Wheel, Dogbone and Dendrite

In order to further increase the burning area, the ends of the slots on star shaped grains are sometimes widened. These kinds of geometries are called wagon wheel, dendrite and dogbone. These geometries give larger burning area in the expense of lowering the volumetric loading. The most common used one is the wagon wheel (Figure 10). The dogbone and dendrite are the modified versions of the wagon wheel geometry, in order to achieve more structural stability or increased burn area. They are not used commonly since the volumetric loading decreases further.

![Figure 10 - Parameters of Wagon Wheel Grain Geometry](image)

4.2.5. Finocyl and Trumpet

All the above mentioned geometries were two dimensional geometries. The cross sectional shape of the grain is constant from head end to aft end on these kinds of geometries. These geometries are optimum for one level of thrust. It can be
progressive or regressive but the lower and upper levels of thrust are not very different. In order to obtain duel level of thrust, the three dimensional grain geometries are used. These geometries are simply composed of two different geometries attached together. The most common one of these geometries is the finocyl geometry (Figure 11). The word finocyl comes from the phrase “fins on a cylinder”. The grain geometry is a hollow cylinder with some radial slots on one side. The part with the fins on acts like the star shaped grain while the non-finned part acts like a side burning cylinder. The total geometry is therefore, behaves like the total of the two geometries. The finned part burns quickly, giving high thrust level, while the cylindrical part burns slower, giving low level of thrust.

![Figure 11 - Finocyl Grain Geometry](image)

Except from giving duel level thrust, the finocyl geometry reduces the erosive burning if the finned side is at the aft end. The fins help the port area to increase, reducing the erosive burning parameters. The main disadvantage is the production and analysis difficulties. Also increase of the thermal loading of the motor case at the finned side is a disadvantage.

Although having some disadvantages, the finocyl grain geometry is widely used on rocket motor industry. Almost all spacecraft launchers use finocyl grain geometry to obtain boost and sustain type of thrust [5].
Another type of three-dimensional grain geometry is the trumpet geometry (Figure 12). Trumpet geometry is much like the finocyl geometry. The only difference is the interface between cylindrical and finned parts. The crossing is much smoother on the trumpet geometry. But this smoothness decreases the advantageous duel thrust characteristics of the geometry.

![Figure 12 - Trumpet Grain Geometry](image)

4.3. Comparison of the Grain Geometries

As mentioned in the previous section, different applications require different types of thrust time profile. Usually there exists more than one geometrical solution for a design problem. The best one is the one that can be analyzed and manufactured.

As to compare the main characteristics of common grain geometries, Table 2 is given [4].
<table>
<thead>
<tr>
<th>Geometry</th>
<th>Volumetric Loading</th>
<th>Web thickness</th>
<th>Burn Area</th>
<th>Burn Area Neutrality</th>
<th>Sliver Area Fraction</th>
<th>Web Fraction</th>
<th>Extra Comment</th>
</tr>
</thead>
<tbody>
<tr>
<td>End Burning</td>
<td>0.98-1</td>
<td>Very Large</td>
<td>Small</td>
<td>Excellent</td>
<td>0</td>
<td>1</td>
<td>Low thrust</td>
</tr>
<tr>
<td>Star</td>
<td>0.75-0.84</td>
<td>Medium</td>
<td>Medium</td>
<td>Good</td>
<td>5-10%</td>
<td>3.5-5.5</td>
<td></td>
</tr>
<tr>
<td>Wagon Wheel</td>
<td>0.5-0.7</td>
<td>Small</td>
<td>Very Large</td>
<td>Excellent</td>
<td>5-10%</td>
<td>6-12</td>
<td></td>
</tr>
<tr>
<td>Trumpet</td>
<td>&gt;0.88</td>
<td>Large</td>
<td>Medium</td>
<td>Excellent</td>
<td>0</td>
<td>&gt;2</td>
<td></td>
</tr>
<tr>
<td>Finocyl</td>
<td>0.75-0.85</td>
<td>Large</td>
<td>Large</td>
<td>Good</td>
<td>0</td>
<td>3</td>
<td></td>
</tr>
</tbody>
</table>
CHAPTER 5

BURNBACK ANALYSIS USING SOLID MODELLING

The burnback methodology used in this thesis study is solid modeling of the propellant grain. In general, the solid propellant grain at the beginning of the operation is modeled parametrically. Then the parameters that change during the burnback process are decided and for every burn step they are modified accordingly. This parameterization can be done either to the complete grain geometry or the geometry can be divided into simpler geometries before the parameterization process.

Also in this thesis study two solid modeling programs were used to perform burnback analysis. The first one is the AutoCAD Mechanical Desktop software from Autodesk Inc., the second one is the Unigraphics NX software from EDS. The method of approach used for each program will be given in detail at the following sections.

On some of the analysis done, some simplifications are made. They will be stated at the appropriate positions. Also at the end of each section a brief flow chart of the analysis is given.

5.1 Burnback Analysis Using AutoCAD Mechanical Desktop Software

AutoCAD Mechanical Desktop (AMD) is a 3 dimensional solid modeling software based on the AutoCAD software of Autodesk Inc., which is a computer aided drawing software usually used for 2 dimensional drawings. The AutoCAD software is not a parametric software. The drawings done cannot be modified by
changing simple parameters. All the drawing has to be done from the beginning if some dimensions are changed. Although AMD being more capable of parameterization of the model, it is not a fully parametric software. For fully parameterization of AutoCAD drawings, a computer language, LISP is used. With the help of the LISP language, AutoCAD commands can be controlled externally. The commands used to draw the initial geometry are inserted into the code for one time. Then by changing the parameters inside the code, the geometry can be updated easily. If the parameters to be changed are obeying an order, then the process of obtaining geometries can be done automatically.

In the case of burnback analysis of 2 dimensional geometries, AutoCAD with the help of LISP language is used. A LISP code is written to draw the initial grain cross sectional geometry. Inside the code, the dimensions that are to be changed are assigned to some variables. Then the part that performs the drawing is placed inside a loop. The loop runs for necessary times, then stops. Inside the loop, the newly formed grain geometry is drawn, and then the necessary data is extracted from the drawn geometry. The data extracted can be the area or perimeter of the grain geometry.

In the case of 3 dimensional grain geometries, instead of AutoCAD, AMD is used. Again with the help of LISP language, AMD can be used to model 3 dimensional parametric geometries. The same methodology is applied to 3 dimensional geometries. Inside the code, the dimensions that are to be changed are assigned to some variables. Then the part that performs the drawing is placed inside a loop. The loop runs for necessary times, then stops, obtaining the necessary data. In the case of 3 dimensional geometries the data to be extracted is the volume of the grain. The burn area can be simply obtained by dividing the volume change between two consecutive burn steps by the burn thickness. The burn area obtained is of course the average burn area between two consecutive burn steps.

The burnback of the grain geometry can be performed in two ways, either the mandrel (the empty space inside the grain) of the given geometry is modeled or the solid propellant grain itself. The two methods are alike each other. The dimensions that should change during the burnback process are either reduced or
increased. For the ease of the process, the mandrel is modeled in this study. Therefore the volume increase of the void space inside the grain geometry is obtained. Since the volume increase of the void space is equal to the volume decrease of the grain, the data obtained can be used to calculate the burn area. From the burn area change data, pressure time data is obtained.

For purposes other than the burn area change, the grain volume change can be obtained easily by subtracting the void volume from the motor case inner volume for every burn step. Therefore, mass moment of inertia or mass change of the rocket motor at any burn step can be calculated.

The performance of the rocket motor is calculated by using the equation given in the simplified internal ballistic design of solid propellant rocket motors section. Then the burn area versus web burned, burn area versus time, and pressure versus time graphics are drawn. The burn area versus web burned graph is the main aim of this study. The other graphs are obtained using the burnback analysis data with the simplified internal ballistics equations. They are for verification purposes of the methodology with the test results.

The main disadvantage of AMD software is the compatibility problems with the LISP language. Although the LISP language is an integrated language of the AutoCAD software, there are some problems with the 3 dimensional commands of AMD. The LISP language is incapable of controlling some AMD commands. This incompatibility restricts some complex geometry to be analyzed by using AMD software.

The analysis made by using AMD software neglects the rounding of the sharp corners on the axial direction. This rounding effect on axial direction occurs on finocyl geometry, at the inner end of the fins. This brings some error to the analysis. The amount of error brought to analysis will be examined by comparing the results of AMD analysis with the UG analysis, where the rounding effect is taken into account.
5.2 Burnback Analysis Using Unigraphics NX Software

The Unigraphics NX software is a fully parametric computer aided design and manufacturing software. It is a new generation software that can be used extensively on all stages of the design. By using Unigraphics, modeling, structural analysis and manufacturing code generation can be done.

Being fully parametric, Unigraphics does not require an external code to perform burnback analysis. The grain geometry is modeled parametrically for one time. Then by changing the parameters, burnback of the initial geometry is obtained.

The part family command of Unigraphics software is used for the burnback analysis. This command builds a Microsoft Excel based spreadsheet, which includes a column for every parameter chosen (Figure 13). Every row of the spreadsheet is for a new solid model, which is a member of the family. The parameters that are to be changed during burnback of the geometry are modified for every solid model. Then Unigraphics creates all the models automatically. Using a simple macro written in Unigraphics, the properties, the volume and the port area, are obtained from the solid models of every burn step. Then the same methodology with AMD is used to perform performance prediction of the rocket motor.
In order to obtain the rounding effect on sharp corners during the burnback, the sharp corners are pre-rounded with very small radius fillets.

For the finocyl geometry, a parent solid model has been prepared. This parent model is modified initially for any finocyl geometry (Figure 14).

After the necessary modifications on the parameters, the solid models of the propellant grain after every fixed burnstep are modeled (Figure 15).

Figure 13 - Example of Part Family Spreadsheet

<table>
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<th>B</th>
<th>C</th>
<th>D</th>
<th>E</th>
<th>F</th>
<th>G</th>
<th>H</th>
<th>I</th>
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<td>66.7</td>
</tr>
</tbody>
</table>
Figure 14 - Solid Model of Initial Propellant Grain

Figure 15 - Solid Model of Propellant Grain after Several Burnsteps
For simplification of the parent grain geometry, only half of a slot is modeled for the burnback analysis. This means that for a 6 slotted star geometry, only 1/12th of the grain is modeled. In order to obtain the full grain volume change, resulting propellant volumes are multiplied by two times the slot number.

5.3 Complex 3-D Geometry Case Study: Experimental Rocket Motor

In the previous section, the analysis of finocyl grain geometry is given. For the test case, the aft and head end of the grain is chosen to be flat. But in real rocket motor applications, the head end of the grain is usually spherical, since the motor case head end is in spherical form for structural reasons. Also on some large grain geometries, there exists some stress relief voids. These voids are usually filled with some easy burning, foam material before the ignition. When the motor ignites, these foam materials burn away, leaving some extra burning area. Also the insulation on large motors is not uniform everywhere. It is usually thicker at the aft end of the motor. This thickness variation of the insulation affects the thickness of the propellant. In order to remove the mandrel easily after curing, the mandrel must be tapered for long motors. Although this taper angle is very small, for motors longer than 2 meters, the cross sectional area of the grain changes considerably.

All these effects form a complex 3 dimensional grain geometry. Such geometry is usually hard to analyze with conventional quasi 3-D methods. These methods can analyze geometries which can be divided into sub-grains having 2-D geometry. For example finocyl geometry can be divided into 2 parts. One cylindrical part, one star shaped part. By analyzing the 2 geometries, the total geometry’s burnback can be achieved. Of course the rounding effect is neglected in this method.

Experimental Rocket Motor of TÜBİTAK-SAGE has a 3 slotted Wagon Wheel grain configuration [7]. But the geometry cannot be divided into 2-D sections.
Especially the taper angle on the mandrel restricts such division. Also the stress relief is a complex 3-D geometry that can not be divided into 2-D geometries.

The burnback analysis of Experimental Rocket Motor is done using 3-D methodology on Unigraphics NX software. 1/6\textsuperscript{th} of the propellant grain is modeled only for simplification.
CHAPTER 6

SOLID PROPELLANT ROCKET MOTOR TESTS

One of the primary steps of developing a design tool is to verify it. Any design tool that is not verified by performing tests or using previously gained data cannot be used for designing.

In general, the tool can be verified by using 2 kinds of approaches. The first one is performing tests and comparing the results with the outputs of the tool. The second one is comparing the outputs with the outputs of previously verified tools.

The first method, testing of design, is usually very expensive and hard to perform. One must have a well prepared test setup for the specific design. The required data acquisition setup should be available. Also the errors of experiment should be handled with care. In addition to above mentioned difficulties, when the system to be designed is a solid propellant rocket motor, some extra difficulties arise. Manufacturing of a solid propellant rocket motor is a very complex and time consuming process. The infrastructure needed for solid propellant rocket motor manufacturing is available at few sites in Turkey. All of these sites are either owned by military or working for military. Also the materials used in the manufacturing of solid propellants are mostly very expensive and hard to obtain. And above all listed, the information about the composition of the solid propellant and manufacturing techniques are all confidential.

The second method, comparison of output with previously verified tools, also has some drawbacks. First of all, a valid tool should be available. For solid propellant rocket motors, the design tools are mostly confidential. The distribution of such confidential tools worldwide is forbidden by national security laws of tool
developing countries. Some preliminary design tools are valid for educational purposes but these are not enough for complete validification of a new tool. This drawback alone restricts the use of this method.

At TUBİTAK-SAGE, many static and dynamic firing of solid propellant rocket motors has been performed since 1980s. These tests are performed mostly to verify the final design of several types of solid propellant rocket motors. During these tests some very useful data about the performance of the rocket motor are obtained. The data acquired is used to optimize the design. Also TUBİTAK-SAGE has the necessary infrastructure to manufacture the desired solid propellant rocket motors for tests.

6.1. Static Firing of Solid Propellant Rocket Motors

The performance of a rocket motor can be tested most accurately by static firing the rocket motor on a test bench. The thrust, chamber pressure, nozzle discharge velocities and even the change of grain geometry can be acquired using appropriate equipment. The thrust and pressure versus time data of the rocket motor gives directly the performance of the rocket motor. The nozzle discharge velocities and grain geometry change are very hard to obtain compared to thrust and pressure and they are not used directly to give the performance of the rocket motor.
The test bench facility used for this study which is located at TÜBİTAK-SAGE’s Lalahan campus, near Ankara is shown in Figure 16. The test bench is designed for static testing of solid propellant rocket motors up to 300 mm of diameter and about 200,000 Newtons of thrust. The test bench facility also has an environmental conditioning chamber for testing of rocket motors at elevated or reduced temperatures and humidity levels. There is also a spin generating system for testing of rocket motors at rotation, the way they travel at dynamic firing.

Figure 16 - Test Bench Facility of TÜBİTAK-SAGE

Figure 17 - Data Acquisition System
The data acquisition system used during the static tests is a DaqBook 2000E data acquisition system integrated to a personal computer. The DBK-01 16 channel BNC input module and DBK-30A power management modules are also used during the tests. The system is capable of acquiring data at 200 kHz maximum. The whole data acquisition system is shown in Figure 17. The software used with the data acquisition system is DaqView. Both the system and the software are products of IOtech Incorporation, USA.

The pressure transducer used during the static tests is a Druck PMP-4015 High Frequency Piezoelectric Pressure Transducer.

For the tests performed during this study, thrust data was not measured due to lack of necessary force transducer suitable for low thrust levels. The pressure data alone is enough for performance measurement. The thrust data is used mostly for verifying pressure data or for cases where pressure data can not be obtained.

Also all tests are recorded by using a Sony TRV-75E digital camcorder which is capable of taking video at 25 frames per second. The camera is zoomed to the nozzle exit of the test motor to record the plume of the motor. Large instabilities accruing inside the rocket motor can be detected by the change of the plume geometry.

6.2. BAM (Ballistic Research Motor) Tests

Ballistic Research Motor (BAM) is the test motor used at TÜBİTAK-SAGE for verification of solid propellants produced (Figure 18). BAM is a small, simple solid propellant rocket motor. The outer diameter of the rocket motor is about 90 mm. The length of the motor is about 300 mm including the nozzle. The motor case is made of thick steel, suitable of multi firing. There is a pressure measuring opening at the rear end of the motor case. The nozzle has a graphite insert, reducing the throat opening during firing of motors. The propellant is cast into a cylindrical tube, lathed to the final geometry and inserted inside the motor case. The igniter used for BAM is MTV and black powder mixture ignited by an electrically
activated squib. The mixture and squib are pressed inside a steel case with holes on
sides.

The main purpose of BAM is to control the quality of the solid propellant
produced at each batch. The grain geometry of the solid propellant is pure cylinder
with tapered sides. The purpose of such geometry is that, it is possible to predict the
change of geometry during burning by using simple algorithms. Also the tapered
ends give a nearly neutral burning characteristic to the rocket motor. The only
difference between the tests is the propellant itself, so the quality of the propellant is
controlled. [6]

The BAM is chosen for verification of this study since the infrastructure for
static firing test of BAM is available at TÜBİTAK-SAGE. The motor cases,
nozzles, test bench connection equipment and pressure transducer connection
equipment are all ready to use for BAM, which makes the tests to be performed
with less work.

In order to use BAM for this study some changes were made in the design.
The major change is the geometry of the grain. Instead of cylindrical constant burn
area grain geometry, 3-D finocyl geometry is used, which is the purpose of this
study. Also the igniter of the motors is replaced with a new design. The new design uses a feed through to prevent product gas to exit through the rear entrance, where the ignition wires are entering. In the previous design, the ignition signal wire was leaving the motor case through the nozzle. The wire passing through the nozzle prevents blocking of the nozzle exit before the test. This blocking is necessary for reducing the ignition delay of the motor. The pressure inside the motor case is increased during the ignition of the propellant by blocking the nozzle exit. This pressure rise decreases the time required for the propellant to ignite. After the propellant ignites, the motor reaches to its operating pressure, opening the blockage at the throat.

In order to open the blockage of the nozzle at the operating pressure of the motor, a nozzle cap with shear pins is designed. The cap restricts the gas to escape from the nozzle exit until the pressure inside the motor case reaches 12 bars. At 12 bars the pins holding the cap shear-out. Then the cap is carried away by the gas flowing through the nozzle. Total of two 3.2 mm diameter aluminum pins with 90 MPa ultimate shear strength gives enough resistance for 12 bars ignition pressure.

Finally the steel propellant tubes are replaced with composite propellant tubes because the new grain geometries to be tested burn about 4 times longer than normal BAM (Figure 21). This brings more thermal loading to the propellant tube and the motor case. To reduce this load, a thermally more insulating material, composite tubes are used in this study.

6.3. Static Test Methodology

The static firing of solid propellant rocket motors for verification of the design tool starts with choosing the grain geometry to be analyzed. Then manufacturing of the right mandrel geometry is done. Then the solid propellant with desired geometry is manufactured. And finally the motors are fired on the static test bench, acquiring the necessary data. The steps will be explained in detail below.
As the first step, the chosen grain geometry is analyzed for mechanical constraints. When the solid propellant rocket motor is fired, the pressure rises inside the chamber. This pressure rise develops a normal force onto the surfaces of the solid propellant. The solid propellant is a pencil eraser like material, with very low strength. So, even the normal pressure distribution onto the grain geometry can sometime yield to stress values higher than the solid propellant can withstand. In these cases, cracking and fragmenting of solid propellant may occur. Cracking or fragmentation increases the burning area rapidly and significantly, causing a rapid pressure increase, which can lead to explosion of the rocket motor. To prevent this kind of problem, all the grain geometries are exposed to finite element stress analysis before they are manufactured and fired. The commercially available ANSYS Finite Element software was used for this analysis (Figure 19). The solid propellant was assumed as viscoelastic during the analysis. For the analysis, the solid model of the initial propellant grain is taken. For meshing of the geometry, smart mesh type 3 is used. The propellant is fixed from the outer surface, where it is attached to the tube. Constant 8 MPa pressure load is given to the inner, aft and rear sides of the propellant grain. After the analysis, Von Mises stresses are compared with the ultimate tensile strength of the propellant, which is taken as 0.5 MPa. The necessary constants for the analysis and UTS of the propellant were found from the recent studies on rocket motors and solid propellants at TÜBİTAK-SAGE [10, 11].
After the mechanical analysis, the mandrel geometries to obtain desired grain geometries are designed and manufactured (Figure 20). The mandrels are divided into two parts. One cylindrical part and one star shaped slotted part for ease of manufacturing. The cylindrical part is manufactured by lathing, and the slotted part is manufactured by using wire erosion. Then the two parts are assembled to obtain the finocyl geometry mandrel. The assembled mandrels are then coated with polytetrafluoroethylene (PTFE) to prevent the solid propellant to stick onto the mandrel while curing.
The next step is manufacturing the propellant tubes that the solid propellant is cast into. The previously used steel tubes were manufactured by lathing. For this study composite tubes are used as previously explained (Figure 21). The composite tubes used are manufactured from the scrap launch tubes of M-72 antitank missile. The tubes are made of glass fiber and epoxy resin. The inner diameter of the tubes was originally 66.7 mm, and left untouched. The outer diameter is machined using a lathe in order to fit inside the steel motor case. Then the tubes are cut to proper length.
After the tubes are manufactured, the manufacturing of solid propellant and liner material are performed. For this study, a much known type of propellant for TÜBİTAK-SAGE is used. This propellant is a composite propellant composed of ammonium perchloride, HTPB and aluminum. This propellant is chosen because a lot of data exists about the burning characteristics and manufacturing techniques at TÜBİTAK-SAGE. The liner is the material used for bonding the propellant to the tube, or motor case. The propellant was prepared at a five gallon mixer, then cast into the propellant tubes, which the mandrels where placed in before (Figure 22 and Figure 23).
Figure 22 - Five Gallon Propellant Mixer
• Then the propellant is exposed to vacuum for some period for the air to escape (Figure 24). The air bubbles inside the solid propellant act as burning area increasers and can cause rapid pressure increases during operation. So the air inside the propellant should be removed before it is cured. In vacuum the air inside the thick propellant dough moves upwards to the vacuumed cavity (Figure 25). For larger motors, the whole casting process is done in vacuum to prevent any bubbling. For smaller motors like BAM, the casting process is done at atmospheric conditions and then the motors are placed in vacuum.
Figure 24 - Vacuum Chamber

Figure 25 - Air bubbles moving to vacuumed cavity
After the casting and vacuuming of the propellants are finished, the tubes with mandrel and solid propellant are placed inside a temperature-controlled oven to be cured (Figure 26). After about 1 week, the tubes are taken out of the oven.

The next step is lathing of the propellant. The casting of propellant leaves some unwanted propellant at the sides of the propellant tube (Figure 27). These unwanted propellants are removed with the help of lathe. This way, all the propellants are assured to be at the same initial geometry. After the lathing is done, the aft and head end surfaces of the propellants are inhibited to prevent burning on these surfaces. For this process, liner material is used.
After the lathing process, the propellants are taken to quality control department for inspection. At this step two main criterions are checked. First the geometry of the propellant, second the quality of casting. The first one is checked by using ordinary geometrical measuring techniques. The second is checked by applying non destructive testing (NDT) methods. The most common one is X-ray photography of the propellant (Figure 28). This process shows us the inside homogeneity of the propellant without breaking it. Also the air bubbles, which act as area increasers during burning, are checked.
• If the propellant grains are suitable for testing, then the propellant grains are assembled with the motor cases and nozzles. At this stage, the necessary liner is applied to the motor case and nozzle interface.

• After the rocket motors to be tested are ready, they are taken to the test site. At the test site the pressure transducer is attached to the rocket motor case (Figure 29). Then the MTV based igniter is assembled to the rocket motor (Figure 30). Finally the motor to be tested is attached to the test bench (Figure 31).
Figure 29 - Pressure Transducer attached to the motor case
Figure 30 - Igniter Assembly

Figure 31 - Test Bench
• After all the data acquisition system is ready, the igniter is wired to the firing line. Then the firing signal is given to the rocket motor, while acquiring the pressure data.

• The test is also recorded by a camcorder (Figure 32). The recordings, especially the zoomed image of the nozzle section gives a detailed examination of the operation period. The instabilities or ignition delay like events can be detected from the changes in the plume formation.

[Image: Figure 32 - Static Firing of a Rocket Motor]

• After the test finishes, the data is processed for analysis.
The aim of this study is to develop a methodology to perform burnback analysis of 3 dimensional grain geometries. In the previous chapters, the methodologies were explained. In this chapter the results obtained from the analysis performed and comparison of some of these results with rocket motor test will be given.

7.1 Finocyl BAM Analysis

In this study, as a test case, finocyl grain geometry is chosen. The finocyl geometry is a 3 dimensional grain geometry composed of a cylindrical and star shaped parts attached together. The reason for finocyl geometry to be chosen is the widespread usage of such geometries. The main parameters of finocyl geometry are given on Figure 33.
During this study 25 different finocyl geometries are analyzed by using both AMD and Unigraphics software. The main parameters that were changed are finned part length, circular part diameter, fin width, fin length and the number of fins. The outer diameter of 66.7 mm and length of the whole grain of 150 mm are left constant.

The nozzle used for the analysis has a throat diameter of 11 mm and exit section diameter of 39 mm.

The propellant used for the analysis is a composite propellant composed of ammonium perchloride, HTPB and aluminum. The propellant has “a” value of 7.25E-5 and “n” value of 0.3179. Also the calculated $c^*$ value of the product gas, using NASA-Lewis program, is 1600 m/s [7].

The parameters of the analyzed geometries are given on Table 3.
Table 3 - Analyzed Geometries

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For the burnback analysis of finocyl geometries, finally the fixed burnback distance should be selected. For the selection of such fixed distance, the effects of burnback distance are examined first on finocyl geometry. The burnback analysis is done with several burnback distances. The results are compared in Figure 34.
The comparison of burn step shows that 1 mm of fixed burn distance gives acceptable accuracy with acceptable analysis time. 1 mm is about 5% of the web thickness for the analyzed finocyl geometries. Therefore the burnback analysis is completed in about 20 steps. Analyzing 20 steps lasts about ½ hour, including the model generation, analysis of burn steps and graphing of obtained data. When the burn step number increases, the required time increases significantly. For detailed analysis of final geometry, smaller burn steps like 1% or 2% can be used to obtain more accurate results.

From the analysis done, burn area versus web burned, burn area versus time and pressure versus time data are obtained and given in Appendix A (Figure 52 to Figure 76).

As seen from the burn area versus web burned graphs, the two methods used for the burnback analysis of finocyl geometries give very similar results. The main difference between the methods is the rounding effect of axial sharp edges. On
finocyl geometries this effect is valid at the inner edge of the slots at the finned part. The analysis using AMD software neglects this rounding effect. UG analysis includes such effect. So the difference at the burn area is because of the rounding effect. The rounding effect is not very significant at the first burn area calculation, then the rounding effect decreases the burn area slightly, compared to the not rounded one. After some burn steps, the rounding effect starts to increase the burn area.

5 of the analyzed finocyl grain geometries are chosen for rocket motor tests. The chosen grain geometries are manufactured and tested statically at the test bench. The chosen geometries are #1, #17, #18, #21 and #24. These geometries are chosen because they represent different pressure-time characteristics seen on the analyzed configurations. The method that was followed during the tests is given in Chapter 6. Total of 10 rocket motors, two for each configuration, were manufactured. One of the rocket motors produced could not pass the NDT. There were large air bubbles inside the grain, so it was not tested. Also from two tests, pressure data could not be taken because of transducer and data acquisition system failure. The pressure data taken from these 7 tests are compared with the analysis results in Figure 35 to Figure 38.
In Figure 35, pressure data from burnback simulations of configuration # 01 finocyl BAM are compared with the test results. The simulations were done by using both Unigraphics and AutoCAD softwares. The detailed dimensions of the grain geometry are given in Table 3. The simulations were done for inhibited aft and head end sides.
In Figure 36 pressure data from burnback simulations of configuration #17 finocyl BAM are compared with the test results. The simulations were done by using both Unigraphics and AutoCAD softwares. The detailed dimensions of the grain geometry are given in Table 3. The simulations were done for inhibited aft and head end sides.
In Figure 37 pressure data from burnback simulations of configuration #21 finocyl BAM are compared with the test result. The simulations were done by using both Unigraphics and AutoCAD softwares. The detailed dimensions of the grain geometry are given in Table 3. The simulations were done for inhibited aft and head end sides.
In Figure 38, pressure data from burnback simulations of configuration # 24 finocyl BAM are compared with the test results. The simulations were done by using both Unigraphics and AutoCAD softwares. The detailed dimensions of the grain geometry are given in Table 3. The simulations were done for inhibited aft and head end sides.

The pressure data taken from the tests do not intersect with the analysis done. The possible reason for such behavior may be erosive burning or experimental errors.

In order to check the erosive burning characteristic of test cases, the flow inside the motor case is analyzed. The first step of the analysis is obtaining the free volume solid model of the motor, at the time of ignition. Since the erosive burning characteristics decay as the propellant burn, the initial geometry is checked only. Then by using Fluent commercial software, the mesh is generated. Finally the flow along the cavity is solved by using Set-3D software. In Figure 39 and Figure 40,
pressure and velocity change along the cavity is given. The analysis results show that the pressure and velocity inside the grain cavity does not change significantly. Therefore erosive burning is not valid for the test case [5, 7 and 8].

Figure 39 - Pressure Variation along Motor Axis
Figure 40 - Velocity Variation along Motor Axis

The second possible reason was experimental errors. One of the most common errors that are seen on rocket motor test is insufficient inhibition of the surfaces which are not supposed to burn. The inhibited surface starts to burn if the flame finds an even small opening to pass through. To analyze the surface inhibition errors, the burnback analysis of the test cases are done again. This time the grain is modeled to burn from the inhibited sides also. This way all the surfaces that are not touching the propellant tube are assumed to burn. The analysis involving end burn are done using AMD software only, since the results of both software are very similar. The results of burnback analysis are given in Figure 41 to Figure 44.
Figure 41 – Comparison of #01 End Burn Simulation and Test Data

Figure 42 – Comparison of #17 End Burn Simulation and Test Data
Figure 43 – Comparison of #21 End Burn Simulation and Test Data

Figure 44 – Comparison of #24 End Burn Simulation and Test Data
As seen from the comparison of the test data and burnback analysis data with end burn, the general trends of the graphs are very similar. The inhibition of the ends was done by applying liner material on the surface after the lathing operation by using a brush. The performance of such inhibition is very dependent on the liner applying process. The test results show that the inhibition of the sides was not successful for this study.

The magnitude of the pressure at the test data is usually lower than the analyzed data. The main reason for such behavior is the properties of the propellant. The properties used for the burnback analysis were the ideal properties. Especially the $c^*$ value used for the simulations is the maximum value that the propellant can have. The $c^*$ depends on the temperature of the product gas. The used $c^*$ for the propellant, is calculated using adiabatic flame temperature. But in real operation, the flame temperature is lower than the adiabatic one, since heat loss is valid. Also the $c^*$ used is calculated for complete combustion of propellant. But in real operation, especially in short length rocket motors, some aluminum particles are exhausted without complete combustion. All these effects decrease the $c^*$ value significantly [8 and 18]. In literature, the $c^*$ decrease for small rocket motors is given about 5% to 10%, depending on the propellant mass, length of the motor, aluminum content and some other properties. Figure 45 and Figure 46 show the change of $I_{sp}$ efficiency, which is the total of $c^*$ efficiency and thrust coefficient with respect to residence time and flame temperature. The $I_{sp}$ efficiency for our case is between 80% and 90%.
Figure 45 - Change of $I_{sp}$ Efficiency with Residence Time [8]

Figure 46 - Change of $I_{sp}$ Efficiency with Flame Temperature [8]
On Figure 47 the burnback simulation of #17 with a \( c^* \) value of 1450, instead of previously used 1600, is given. This is a decrease of about 10% in \( c^* \) value.

![Pressure vs. Time](image.png)

Figure 47 - Comparison of #17 with \( c^* \) Modification

The final graph obtained is very similar to the test data. Some small differences at the pressure curve still exist. One reason for such difference is that the propellant properties change slightly from batch to batch. Also the manufacturing process has a great influence on the motor performance.

An important factor affecting the pressure inside the motor case is the throat area of the nozzle. The analysis done in this study assumes the throat area to be constant during the operation of the rocket motor. But in real rocket motor operation the throat area is not constant. For small throated nozzles like the one used in this
study, boundary layer formation affects the throat area [6 and 17]. Also the aluminum oxide particles that are condensed inside the motor or the metal particles from the melted igniter casing reduce the effective throat area when passing from the throat cross section. The sudden peaks at the test data are caused by large igniter case particles that are passing through the nozzle throat, reducing the effective area significantly. Slug formation at the throat is also an important factor. Especially at the early stages of operation, slug formation at the cold throat is very effective. Later on, when the throat is heated, the formed slug melts and the throat area increases. The decreased area at the early stages of operation makes the pressure curve steeper than expected [21]. Especially on configuration #17, this effect can be seen clearly.

Since the burnback analysis done in this study does not include unsteady effects at the ignition and tail-off, the results are generally different from the test data at the ignition and tail-off regions. This proves that, steady state assumption is valid at the steady state operation of the rocket motor only. The ignition and tail-off regions can not be assumed steady state as expected.

All these effects are changing the magnitude of the calculated pressure. But the trend of the pressure curve at the steady state operation is determined mainly by the burning area change. The scope of this study is to determine the burning area change of the propellant grain during operation therefore the pressure inside the motor case and the methods used give acceptable burn area versus web burned values.

As explained in the previous sections, there are plenty of burnback codes developed and used at TÜBİTAK-SAGE. The most recent study on this subject is done by Cengizhan Yıldırım, as his Master of Science study [2]. In this study, several codes were developed for various 2 dimensional grain geometries. The most important of these codes are CELLO3D code and its variant FINOCYL code. These codes were quasi three dimensional codes. Quasi three dimensional term, as explained before, stands for three dimensional codes that can be divided into several two dimensional sections. In Figure 48, the burnback simulation done by using the methodology introduced in this study is compared with the burnback simulation
done by using FINOCYL code, developed by Cengizhan Yıldırım [2]. Also the test results are shown in the graph.

![Pressure vs. Time](image_url)

Figure 48 - Comparison of #17 with FINOCYL Code

As seen from the figure the quasi three dimensional code also gives satisfactory results obtaining the general trend of the pressure curve. The tail-off and ignition periods are again different from the test data. This is expectable since FINOCYL code is also a stead state code. The FINOCYL code is less capable of predicting the sustain part, since 3 dimensional effects are more dominant in that region.
7.2 Experimental Rocket Motor Analysis

As a test case of complex 3-D grain geometry, Experimental Rocket Motor is analyzed. The burn area change of the grain geometry is given on Figure 49. The burn area change is compared with the analysis made by using 2-D analysis software. In this analysis, stress relief and the insulation layer was neglected in order to obtain 2-D grain geometry. In order to compare the effect of taper angle, UG analysis is made for tapered grain geometry, without any insulation, head dome or stress relief. The result is similar to the analysis without the taper angle for the initial burn steps. As the web burns, the effect of taper becomes more significant.

Figure 49 - Burn Area versus Web Burned of Experimental Rocket Motor

The effect of stress relief void, head end dome and insulation can be seen from the graph. The burn area at the initial burn steps is not very different from the
previous analysis. As the web burns, the progressive characteristic of burn area becomes neutral. Also from the graph, the decrease of propellant amount inside the motor case due to stress relief void, insulation and head end dome can be seen by analyzing the area under the burn area versus web burned curve, since the area gives the total volume of propellant inside the motor case.

In order to compare the burnback simulation with the real test cases, the static firing test data of Experimental Rocket Motor is used. The pressure obtained from previous test and from the analysis made with UG is given in Figure 50. The pressure and burn area from both the analysis and the test data are divided by reference values since the test data is confidential.

Figure 50 - Comparison of Experimental Rocket Motor
As seen from the graph, the two results are very different. The main reason for this is erosive burning characteristics of Experimental Rocket Motor. The analysis was made for non-erosive burning rocket motor. In the case of Experimental Rocket Motor, this assumption is not valid any more. The typical pressure change of an erosive burning rocket motor is given in Figure 51. The comparison of Experimental Rocket Motor test data and analysis using non-erosive burning assumption is very similar to this graph. The trends of the curves are very much alike. The magnitude of the pressure obtained by the analysis is again different from the test data. The uncertainties have some effects on this difference. But the graph given in literature has also some errors. The erosive burning motor should have lower pressure at the end of operation. The erosive burning at the beginning of operation causes the aft end of the propellant to burn-up faster than the head end. This faster burn-up causes the burn area to decrease before the tail-off region. But on Figure 51 this phenomena is neglected. Both the erosive and non-erosive burning motors are given to have equal pressures after some time. The non-erosive burning motor should have larger pressure before the tail-off began, as given in Figure 50.
Figure 51 - Comparison of Erosive and Non-erosive Burning Motors [4]
CHAPTER 8

CONCLUSION AND FUTURE WORK

In this thesis, brief information about the solid propellant rocket motors is given. Also the design methodology and governing equations are discussed briefly.

One of the most important steps of solid propellant rocket motor design, the grain burnback analysis was the main scope of this thesis. The methodology used for burnback analysis and verification is given in detail.

Two commercially available solid modeling softwares, Unigraphics and AutoCAD Mechanical Desktop are used to perform burnback analysis. The method used with both softwares is the same. The initial grain geometry is modeled parametrically with the software. Then the parameters that change during the burnback are adapted for every burn step. The change of the volume of the grain geometry gives the amount of propellant burned for that interval. By dividing this volume by the thickness, the burn area is acquired.

After receiving the burn area change for fixed burn thickness, by using simplified equations, the time required for every burn step is calculated. Then adapting this time data to the burn area change data, the burn area change, therefore the pressure, versus time data is obtained.

This pressure versus time data is used to verify the method. The pressure data calculated from the burnback analysis is compared with the pressure data obtained from rocket motor tests.

The rocket motor tests for the verification of burnback analysis are performed at TÜBİTAK-SAGE. The detailed method for the rocket motor production and test is given in chapter 6.
For the test case of burnback analyses, finocyl grain geometry is chosen. The finocyl geometry is very popular at modern rocket motors. Most of the modern rocket motors giving boost sustain type of thrust uses finocyl grain geometry. The analyses and the rocket motor tests done were very useful for obtaining information on finocyl grain geometry, which is very new to Turkish rocket industry.

Also for a test case of complex 3-D grain geometry, Experimental Rocket Motor is analyzed. The burnback analysis for the grain geometry of Experimental Rocket Motor is done. The burn area change data is compared with previous analysis, using 2-D approach. The pressure data from the burnback analysis is also compared with the test results of Experimental Rocket Motor. The pressure data from the analysis does not fit with the test data. The reason for this was erosive burning characteristic of Experimental Rocket Motor. The effects of erosive burning characteristic on rocket motor pressure change were given in literature. The general trend of test data and analysis data fits perfectly to the trend given in literature. This shows that the method developed can be used for the final burnback analysis of complex grain geometry, if the motor is not burning erosive.

The method developed in this study not only calculates the burn area, but generates the solid model of the propellant grain for every burn step. These solid models can be used easily for structural analysis without the need for a modification. Also a very important data, the mass properties of the grain geometry, thus the motor altogether, can be achieved from the solid models. These data are very important for flight mechanics at the stage of calculating the final flight path.

Finally the experimental part of this study supplied a large knowledge on static testing of rocket motors. The previously used test motor (BAM) is improved by adding extra features like advanced insulation, ignition delay reduction and sealing. The effect of metal igniter case on rocket motors with long burn time is also seen. The metal pieces that are formed by melting of the igniter case become a real threat for test motors with long burn time. For such motors, softer igniter case materials like plastics should be use.
8.1. Recommendation for Future Work

The method used in this study is applicable to non-erosive burning rocket motors. For erosive burning rocket motors, the burnback method has to be coupled with an internal flow solver, in order to include the erosive burning effects.

Another way of including erosive burning characteristic in the burnback analysis is by dividing the grain into fine slices, and comparing the erosive burning parameters with the experimental data. By this way the slices at which erosive burning occurs, are treated different.

By developing an algorithm, the method used in this study can handle erosive burning characteristic.

The complex 3-D test case, Burnback analysis of Experimental Rocket Motor was performed non-parametrically. This means that, in order to analyze a geometry similar to Experimental Rocket Motor but not having the same dimensions, the model has to be developed again. Parameterization of such geometry can help making fine tuning on the final geometry.

In this study a large number of finocyl geometries are examined. Also some very important test data is obtained for the analyzed geometries. By using these data and some statistical techniques, the effects of the geometrical parameters on performance of finocyl geometry can be studied. This kind of study may be very useful at the preliminary design stages of a new rocket motor.
REFERENCES

[8] Netzer D. W., “Propulsion Analysis for Tactical Solid Propellant Rocket Motors”
APPENDIX:

SIMULATION RESULTS
Figure 52 - Pressure and Burn Area Change of #1
Figure 53 - Pressure and Burn Area Change of #2
Figure 54 - Pressure and Burn Area Change of #3
Figure 55 - Pressure and Burn Area Change of #4
Figure 56 - Pressure and Burn Area Change of #5
Figure 57 - Pressure and Burn Area Change of #6
Figure 58 - Pressure and Burn Area Change of #7
Figure 59 - Pressure and Burn Area Change of #8
Figure 60 - Pressure and Burn Area Change of #9
Figure 61 - Pressure and Burn Area Change of #10
Figure 62 - Pressure and Burn Area Change of #11
Figure 63 - Pressure and Burn Area Change of #12
Figure 64 - Pressure and Burn Area Change of #13
Figure 65 - Pressure and Burn Area Change of #14
Figure 66 - Pressure and Burn Area Change of #15
Figure 67 - Pressure and Burn Area Change of #16
Figure 68 - Pressure and Burn Area Change of #17
Figure 69 - Pressure and Burn Area Change of #18
Figure 70 - Pressure and Burn Area Change of #19
Figure 71 - Pressure and Burn Area Change of #20
Figure 72 - Pressure and Burn Area Change of #21
Figure 73 - Pressure and Burn Area Change of #22
Figure 74 - Pressure and Burn Area Change of #23
Figure 75 - Pressure and Burn Area Change of #24
Figure 76 - Pressure and Burn Area Change of #25