ROCKET PROPULSION

SUMMARY OF ROCKET PROPULSION TAUGHT ON AUTUM 2018
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1 Theory Of Rocket Propulsion

Work In Progress

This Section is Under Construction.

2 Nozzle Theory

The concept of an ideal rocket propulsion system is useful because the relevant basic thermodynamic principles can be expressed with relatively simple mathematical relationships, as shown in subsequent sections of this chapter. These equations describe quasi-one-dimensional nozzle flows, which represent an idealization and simplification of the full two- or three-dimensional equations of real aerothermo- chemical behavior. However, within the assumptions stated below, these descriptions are very adequate for obtaining useful solutions to many rocket propulsion systems and for preliminary design tasks. In chemical rocket propulsion, measured actual performances turn out to be usually between 1 and 6% below the calculated ideal values. In designing new rocket propulsion systems, it has become accepted practice to use such ideal rocket parameters, which can then be modified by appropriate corrections, such as those discussed in Section 3.5. An ideal rocket propulsion unit is defined as one for which the following assumptions are valid:

2.1 Performace Parameters

Now there are Two performance parameters That are Defined to asscess the Performance of the Nozzle geometry and the Propulsive Charecterstics of the Rocket.

- Cofficient of Thrust
- Charecteristic Vecocity

Now We can Easily Measure Properties like Throat Area (A_t) , Chamber Pressure (P_c) and Environmental Pressure (P_a). Now We have to define theese parameters in the Known or easily measurable Quntities.

2.1.1 Exit Jet velocity

Applying Energy Conservation equation of the Jet

$$\frac{V_J^2}{2} + h_e = \frac{V_c^2}{2} + h_c \tag{1}$$

we Know that the chamber velocity of Gas is negligible compared to the Jet thus we assume V_c is Zero.

$$\frac{V_J^2}{2} = h_c - h_e \tag{2}$$

$$\frac{V_J^2}{2} = C_p(T_c - T_e) \tag{3}$$

$$C_p = \frac{R_0\gamma}{mw(\gamma-1)} \tag{4}$$

$$V_J = \sqrt{2\frac{R_0\gamma}{mw(\gamma-1)}(T_c - T_e)} \tag{5}$$

Now as We cant measure the T_e we use iso-entropic Relation to Transform into measureble charecters like pressure

$$\Delta S = C_p \ln\frac{T_2}{T_1} - R\ln\frac{P_2}{P_1} \tag{6}$$

$$\frac{T_2}{T_1}^{\frac{\gamma}{\gamma-1}} = \frac{P_2}{P_1} \tag{7}$$

From the above Relation we can write that

$$V_J = \sqrt{2\frac{R_0 T_c \gamma}{mw(\gamma-1)}\left(1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right)} \tag{8}$$

2.1.2 Mass Flow Rate

Now Deriving the Equation of Mass Flow Rate:

$$\dot{m} = \rho_g A_t V_t \tag{9}$$

Now throat condition is chocked when local gas velocity is equal to sonic velocity in chocked condition which give supersonic flow in rocket nozzle.

$$V_t = \sqrt{\gamma R T_t} \tag{10}$$

$$\rho_t = \frac{P_t}{RT_t} \tag{11}$$

$$\dot{m} = \frac{P_t}{RT_t} A_t \sqrt{\gamma R T_t} \tag{12}$$

Nad we aslso have relation that

$$\frac{T_t}{T_c}^{\frac{\gamma}{\gamma-1}} = \frac{P_t}{P_c} \tag{13}$$

$$V_t = \sqrt{2\frac{R_0 \gamma}{mw(\gamma-1)}(T_c - T_e)} \tag{14}$$

$$\sqrt{\gamma R T_t} = V_t \tag{15}$$

$$\frac{T_c}{T_t} = 1 + \frac{\gamma-1}{2} \tag{16}$$

Now using the Abover realtions in the mass flow rate Derivation we get equation to form which contain measurable quantitites like P_c V_t T_c

$$\dot{m} = \frac{P_c A_t}{\sqrt{RT_c}} \Gamma \tag{17}$$

$$\Gamma = \sqrt{\gamma}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}} \tag{18}$$

Now the Chareteristic Velociy is defined as

$$C^* = \frac{P_c A_t}{\dot{m}} \tag{19}$$

2.1.3 Thrust

Now the Thrust calculations We have the formula

$$F = \dot{m} V_J + (P_c - P_a) A_e \tag{20}$$

We have already derived the equation of V_J and $\dot{m}$ earlier we can substitute that in this equation

$$F = P_c A_t \left(\frac{\dot{m} V_J}{P_c A_t} + \left(\frac{P_e}{P_c} - \frac{P_a}{P_c}\right)\frac{A_e}{A_t}\right) \tag{21}$$

We can expand and re-arrange to get full expression of the above form Now er define the New Terms in the Equations

Thrust coefficient is Defined as $C_F = \frac{F}{P_c A_t}$. The Optimum Expansion is occured at $P_e = P_a$ Where Perfect Expansion Takes Place Now The Thrust Cofficient at that situation is called **Optimum Thrust Cofficient**. Maximum value of the Thrust Cofficeint occurs when in vaccum $P_a = 0$.

Now I_{sp} is Defined interms of the C^* and C_F as follows

$$F = C_F P_c A_t \tag{22}$$

$$C^* = \frac{P_c A_t}{\dot{m}} \tag{23}$$

$$I_{sp} = \frac{F}{\dot{m} g} \tag{24}$$

$$I_{sp} = C^* C_F \tag{25}$$

3 Solid Rocket

This is the first of four chapters dealing exclusively with solid propellant rocket motors the word motor is as common to solid propellants as the word engine is to liquid propellants. In this chapter, we cover burning rates, grain configurations, rocket motor performance, and structural issues. In solid propellant rocket motors the pro- pellant is contained and stored directly within the combustion chamber. Solid Propellent Thrust cannot be randomly varied in flight.

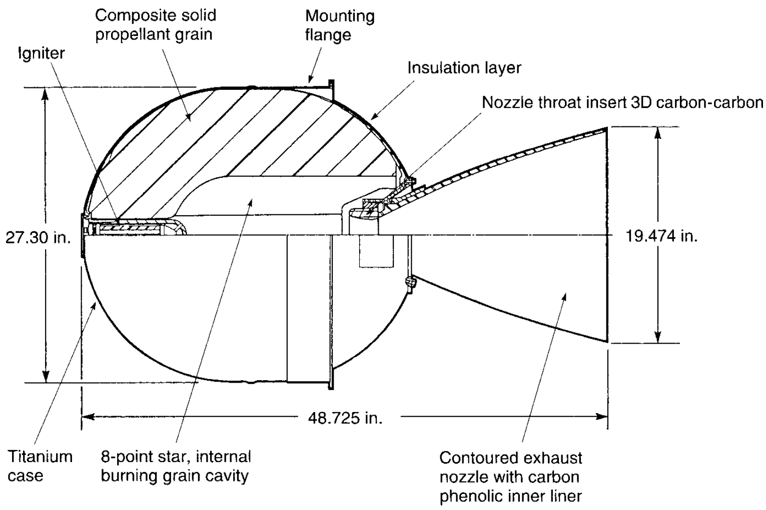

Figure 1: A Solid Rocket Motor

The grain is the solid body of the hardened propellant and typically accounts for 82 to 94% of the total rocket motor mass.In Fig.1 the grain configuration has a central cylindrical cavity with eight tapered slots, forming an eight-pointed star cross section. Many grains have slots, grooves, holes, or other geometric features that alter the initial burning surface and thus deter- mine the initial mass flow rate and the initial thrust.

The Atlas-V propellant consists of aluminized hydroxyl-terminated polybutadiene (HTPB) or ammonium perchlorate (AP).

In the Shuttle SRM the propellant mass fraction was 88.2%.

3.1 BASIC RELATIONS AND PROPELLANT BURNING RATE

A rocket motor's operation and its design depend on the propellant's combustion characteristics such as burning rate, burning surface, and grain geometry. The branch of applied science describing these is known as internal ballistics.

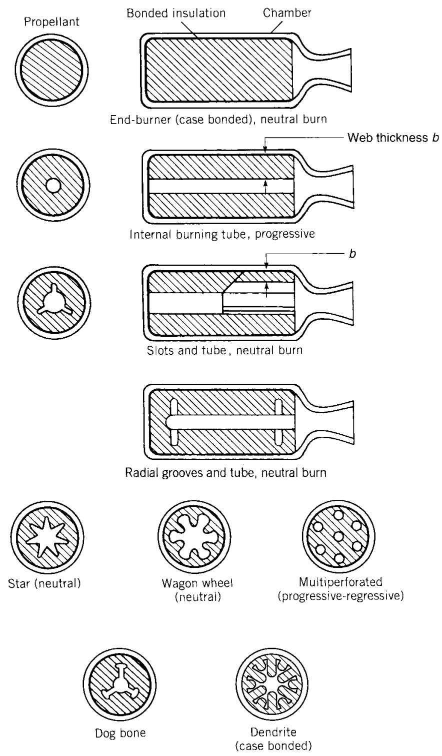

Figure 2: Grain Slot Configuration

The burning surface of a propellant grain recedes in a direction essentially perpen- dicular to it. The rate of regression, usually expressed in cm/sec, mm/sec, or in./sec, is the burning rate r.Burning rate is a function of the propellant composition. For composite propellants it can be increased by changing the propellant characteristics as follows:

- Add a burning rate catalyst, often called burning rate modifier (0.1 to 3.0% of propellant) or increase percentage of existing catalyst.
- Decrease the oxidizer particle size.
- Increase oxidizer percentage.
- Increase the heat of combustion of the binder and or the plasticizer.
- Imbed wires or metal staples in the propellant.

For any given propellant formulation the burning rate can be increased or otherwise modified by the following:

- Higher temperatures of solid propellant prior to start
- Higher combustion gas chamber pressures
- Higher combustion gas temperatures
- Higher gas flow velocities parallel to its burning surface
- Rocket motor motions (acceleration and spin-induced grain stress)

The burning rate of propellants in a motor is a function of several parameters. at any instant of time the mass flow rate $\dot{m}$ of the hot gases generated and flowing from the motor is given by:

$$\dot{m} = A_b r \rho_b \tag{26}$$

Here, A_b is the propellant grain burning area, r the burning rate, and ρ_b the solid propellant density prior to motor ignition. The total effective mass m of propellant burned is determined by integrating

$$\int \dot{m} dt = \rho_b \int A_b r dt \tag{27}$$

With many propellants it is possible to approximate the burning rate as a function of chamber pressure, at least over a limited range of chamber pressures. A log-log set of plots is shown in Fig. 12-6. For a majority of production-type propellants the most commonly used empirical equation is

$$r = a p_1^n \tag{28}$$

where r, the burning rate, is in mm/sec or inches/sec and the chamber operating pres- sure p_1 is in MPa or psia. Known as the temperature coefficient, a is an empirical constant influenced by the ambient grain temperature (T_b) the dimensions of a are defined by those of the other terms in above Equation. The burning rate exponent n (a pure number), sometimes also called the pressure exponent or the combustion index, is taken to be independent of the propellant initial temperature but influences the chamber operating pressure and the burning rate. For combustion stability n < 1.0 otherwise, when n > 1.0, any pressure disturbances present will be amplified in the chamber. Above Equation applies to all commonly used double-base,composite, or composite double-based propellants,

hile changes in ambient grain temperature (T_b) do not alter the propellant's chemical energy available for release during combustion, they do change the rate of reaction at which the energy is released and have a slight effect on c^* through changes in T_1.

To achieve such rates, combinations of very small sized ammonium perchlorate (AP), burning rate catalysts, additives, or embedded metal wire are needed.a propellant having a pressure exponent of zero displays essentially zero change in burning rate over a wide pressure range. Plateau propellants is the name given to those that exhibit nearly constant burning rate over a limited pressure range, and they are desirable for mini mizing effects of changes in initial temperature on motor operation

3.2 Classification Of Solid Propellents

Historically, the early rocket motor propellants used to be grouped into two classes:
double-base (DB) propellants were the first production propellants and subsequently the development of polymers as binders made the composite propellants feasible

3.2.1 Double Based Propellent

Double-base (DB) propellants form a homogeneous propellant grain, usually a nitrocellulose (NC) a solid ingredient that absorbs liquid nitroglycerine (NG), plus minor percentages of additives. The major ingredients are highly energetic materials and they contain both fuel and oxidizer. Both extruded double-base (EDB) and cast double-base (CDB) propellants have found extensive applications, mostly in small tactical missiles of older design. By adding crystalline nitramines (HMX or RDX) both performance and density can be improved; these are sometimes called cast-modified double-base propellants. Adding an elastomeric binder (rubber-like, such as crosslinked polybutadiene) further improves the physical properties and allows more nitramine and thus increasing performance slightly. The resulting propellant is called elastomeric-modified cast double-base (EMCDB). These four classes of double-base propellants have nearly smokeless exhausts. Adding some solid ammonium perchlorate (AP) and aluminum (Al) increases the density and the specific impulse slightly, but exhaust gases becomes smoky—such propellant is called composite-modified double-base propellant or CMDB.

3.2.2 Composite Propellent

Composite propellants form a heterogeneous propellant grain between oxidizer crystals and powdered fuel (usually aluminum) held together in a matrix of synthetic rubber (or plastic) binder, such as polybutadiene (HTPB). Composite propellants are cast from a mix of solid (AP crystals, Al powder) and liquid (HTPB, PPG) ingredients. The propellant is hardened by crosslinking or curing the liquid binder polymer with a small amount of curing agent, and curing it in an oven, where it becomes solid. In the past four decades composites have been the most commonly used class of propellant. Composites can be further subdivided:

- Conventional composite propellants, which usually contain between 60 and 72% AP as crystalline oxidizer, up to 22% Al powder as a metal fuel, and 8 to 16% of elastomeric binder (organic polymer) including its plasticizer.
- Modified composite propellant where an energetic nitramine (HMX or RDX) is added for obtaining some added performance and also a somewhat higher density.
- Modified composite propellant where an energetic plasticizer such as nitroglycerine (used in double-base propellants) is added to give increased performance. Sometimes HMX is also added.
- High-energy composite solid propellant (with added aluminum), where the organic elastomeric binder and the plasticizer are largely replaced by highly energetic materials and where some of the AP is replaced by HMX and RDX.
- Lower-energy composite propellant, where ammonium nitrate (AN) is the crystalline oxidizer (not AP). These are used for gas generator propellants. When large amounts of HMX are added, they become minimum smoke propellants with fair performance

3.2.3 Ignition Propellent

Propellants for igniters, a specialized field of propellant technology, are briefly described here. Requirements for igniter propellants include the following:

- Fast high heat release and high gas evolution per unit propellant mass to allow rapid filling of grain cavity with hot gas and to partially pressurize the chamber.
- Stable initiation and operation over a wide range of pressures (from subatmospheric to the high chamber pressures) and smooth burning at low pressures with no ignition overpressure surges.
- Rapid initiation of igniter propellant burning and low ignition time delays.
- Low sensitivity of burn rate to ambient temperature changes and low burning rate pressure exponent.
- Proper start, operation and storage over the required ambient temperature ranges.
- Safe and easy to manufacture, and safe to ship and handle.
- Satisfactory aging characteristics and long life.
- Minimal moisture absorption or degradation with time.
- Low cost of ingredients and fabrication.
- Low or no toxicity and low corrosive effects.

3.2.4 Gas Generator Propellent

Gas generator propellants are used to produce hot gases, not thrust. They generally have a low combustion temperature (800 to 1600 K), and most do not require internal insulators when used with metal cases

3.2.5 Desired Propellent Charecterstics

1. High performance or high specific impulse; this implies a high gas temperature and/or low exhaust gas molecular mass.
2. Predictable, reproducible, and initially adjustable burning rate to fit grain- design needs and thrust-time requirements.
3. For minimum variations in thrust or chamber pressure during burning, both the pressure or burning rate exponent and the temperature coefficient should be small.

4. Adequate physical properties (including bond strengths) over the intended operating temperature range with allowance for some degradation due to cumulative damage.
5. High density (resulting in a small-volume rocket motor).
6. Predictable, reproducible ignition qualities (such as acceptable ignition overpressures).
7. Desirable aging characteristics and long life. Aging and life predictions depend on the propellant's chemical and physical properties, cumulative damage criteria with load cycling (see Section 12.4) and thermal cycling, and from actual tests on propellant samples and test data from failed motors.
8. Low moisture absorption, because moisture often causes chemical deterioration.
9. Simple, reproducible, safe, low-cost, controllable, and low-hazard manufacturing.
10. Guaranteed availability of all raw materials and purchased components over the production and operating life of the propellant, and acceptable control over undesirable impurities.
11. Low technical risk, such as a favorable history of prior applications.
12. Relative insensitivity to certain external energy stimuli as described Section 13.3, the hazards section.
13. Nontoxic and noncorrosive exhaust gases, also called green exhausts.
14. Not prone to combustion instability (see Chapter 14).
15. Equivalent composition, performance and properties with every new propel- lant batch.
16. No slow or long-term chemical reactions or migrations between propellant ingredients or between propellant and insulator/liner.

Note

For Large Heat of Combustion of the Rocket Propellents We need The Heat of Formation of the Products must be large and Negative. And the Heat of Formation of the Reactants must be small and negative or Even preferably Positive.

3.3 Fuels

3.3.1 Aluminium

This section discusses solid fuels of which powdered spherical aluminum is the most common. It consists of small spherical particles (5 to 60 μm diameter) and is used with a wide variety of composite and composite-modified double-base propellant formulations, usually constituting 14 to 20% of the propellant by weight. Small aluminum particles can burn in air and aluminum powder is mildly toxic if inhaled. During rocket combustion this fuel is oxidized to aluminum oxide. Such oxide particles tend to agglomerate and form larger particles. Aluminum increases the heat of combustion, the propellant density, the combustion temperature, and thus the specific impulse. The oxide starts in liquid droplet form during combustion but solidifies in the nozzle as the gas temperature drops. When in the liquid state, the oxide can form a molten slag, which can accumulate in pockets (e.g., around an improperly designed submerged nozzle), thus adversely affecting the vehicle's mass ratio. It also can deposit on walls inside the combustion chamber

3.3.2 Boron

Boron is a high-energy fuel that is lighter than aluminum and has a high melting point (2304^oC). It is difficult to burn with high efficiency in combustion chambers of practical lengths. However, it can be efficiently oxidized if the boron particle size is sufficiently small. Boron has been used advantageously as a propellant in a rocket combined with an air-burning engine, where there is adequate combustion volume and atmospheric oxygen

3.3.3 Beryllium

Beryllium burns much more easily than boron, improving the specific impulse of a solid propellant motor by about 15 sec, but both beryllium and its highly toxic oxide powders are absorbed by animals and humans when inhaled. The technology with composite propellants using powdered beryllium fuel has been experimentally proven, but its severe toxicity makes any earth-bound application unlikely

> **Note**
>
> Now usually The Efficiency of the Propellent is Maximum when the Propellent and Oxidiser are in Stoichometric Ratio But Usually the Propellent is Kept Fuel Rich Due to the Fact that the fuel rich propellent Produces lower molecular weight burning Products like **CO** and **NO**. As we Know the I_{sp} is indirectly Proportional to Molecular Weight of the Working Fluid.

3.4 Binders

Binders are the Strcutural glue or the matrix that holds together the solid Preoppelent Grain particles. Liquid Polymers are Used as Binders . Binder Materials are also oxidised in the combustion process. Aliphatic Compounds are Preferred Due to small negative heat of Formation.

Common Binders Used are

- **HTPB**: Hydroxyl Terminated PolyButadiene
- **CTPB**: carbooxyl Terminated PolyButadiene
- **PBAN** : Polybutadiene Acrylic Acid or Acrylic Nitrile

Now HTPB is Preferred over the CTPB beacuase the HTPB contains Hydrogen which is lower Molecular weight and gives rise to low Molecular weight products and Thus give more performance than the Carboxyl Terminated CTPB. The Grade of the HTPB is determined by the no of butadiene Groups in the HTPB.

Now Poly-Butadiene Acrylic acid or Poly-Butadiene acrylonitrile is used in large rockets. The PBAN is hghly cross linked and have le

3.5 Oxidisers

Now there are Two Types of Oxidisers commonly used in the solid Rocket Propellents

- Percolate Based ($-ClO_4$)
- Nitrate Based ($-NO_3$)

Examples are

1. NH_4ClO_4 - Ammonium Percolate (AP)
2. $KClO_4$ - Potassium Percolate
3. $NaClO_4$ - Sodium Percolate
4. KNO_3 - Potassium Nitrate
5. $NaNO_3$ - Sodium Nitrate

Now Ammonium Percolate is Commnly Known as **AP** and most commonly used. Potassium Nitrate is Used as an Oxidizer in Gun powder.

3.5.1 Nitrate Based Oxidizers

- These are Relatively low performance Conpared to the Percolate based oridisers
- They are low cost Smoke-less and relatively non-toxic Exhaust.
- They are Usually used in the low performance Rockets and Gas generator Propellents.
- Ammonium Nitrate Even though having very good oxygen content undegoes phase transistion at different termparatures and thus changes volume.
- Repeated cycling throgh this transition phases may create voids and change the structure which is not desired in tigtly packed solid Rocket propellent.
- AN is HydroScopic and absorb mositure and Degrade propellent Also KNO_3.

3.5.2 Percolate Based Oxidizers

- Most Commonly Used.
- Only Slightly Dissolvable in Water Which is Benifitial.
- Undergoes Two Stage Decomposition. and supplied in white Crystals.
- Toxic Gases are Produced in Decomposition.

The AP particles comes in different sizes ranging From $5-60\mu m$. Smaller the Particles Faster the reactivity of the Particle. The burning rate of the composite propllent is controlled by using mixture of small and large particle.

- Coarse - $400-600\mu m$
- Medium - $50-200\mu m$
- Fine - $5-15\mu m$
- Ultra Fine - $1-5\mu m$

Ultra Fine and Fine Particles are highly explosive. We genrally use a combination of Sizes for Best Packing of the Particles.

3.6 Plasticizers

These are Low Viscocity Organic Liquids which are added to improve processing Properties Such as to lower the viscosity to improve the casting process and improve the shelf life of the propellent.

- Dioctyl adiphate
- Dioctyl Phalate

3.7 Curing Agents

This Causes Polymers to Form Long Chains of Larger Mlecular mass and interlocks between chains. The Percentage of these are about 0.2-0.3% in weight in the Solid Propellent.Minor change in the Percentage of the Curing agent may significantly affect the Physical Properties.It is used in the Composite Propellent and becomes the Binder to solodofy and become hard Eg **IPDI** , **TDI**.

3.8 Burn Rate Modifiers

Burn Rate Modifiers are USed to control the burning Rate either Decrasing or Increasing the Burning Rate.Wires are introduced in Grain to omproce heat Transfer and thus increading burning rate. The Grain is made in helical shape to increase the mould collapsible . Oxides of Copper , Iron and Lead are used as burning rate Enhancer while lithium Floride is used to Inhibit the Burning rate of th Propellent.

3.9 Burning Mechanism for Propellent

3.9.1 Double Based Propellent

- The Double Based Propellent Is based on NG and NC.
- The Propellent Degrades Orthonormally
- Fizz zone gases like NO NO_2 and aldehydes are formed
- If Pressure is Low $P_c < 1MPa$ then The Fizz zone is followd by a Dark zon in which the Chemical Recation between gases Take Place and No significatnt Rise in temperature is seen.
- if Pressure is Higher $P_c > 10MPa$ Then the Dark Zone is absent.
- The Extent of the Dark Zone is Pressure Dependent
- At Higher Pressure Burning Rate is Faster Since the Dark Zone is Absent There Will be efficinet Heat Transfer Too Buring Surface thus helps in combustion of the Propellent.

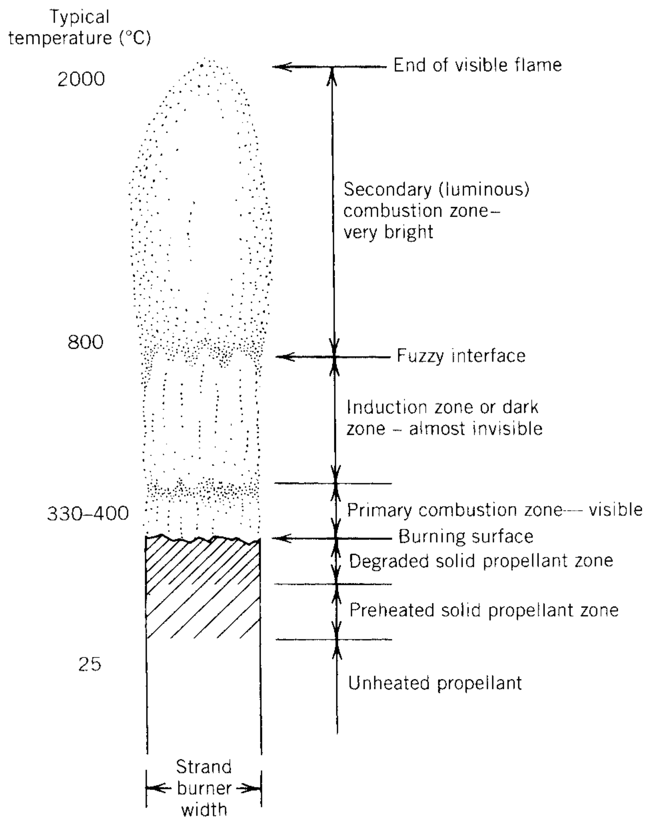

Figure 3: Double Based Combustion Process

$$-K_g\frac{dT}{dx} + \rho u_g cT + Q_{chem}dx = -K_g\frac{dT}{dx} + \frac{d\left(-K_g\frac{dT}{dx}dx\right)}{dx} + \rho u_g c(T + dT) \tag{29}$$

$$Q_{chem}dx = \frac{d\left(-K_g\frac{dT}{dx}dx\right)}{dx} + \rho u_g c(dT) \tag{30}$$

The Terms Used in the Above Situation are:

- Q - Heat Released From unit volume of Propellent
- k_g Thermal Conductivity of Gas

- c Specific Gas Constant
- u_g Gas velocity

Now the Boundary condtiionn are given by

$$x = 0; T = T_{surface} \tag{31}$$
$$x = L; T = T_{ignition} \tag{32}$$
$$Q = Ap^m e^{\frac{E}{R_0 T}} \tag{33}$$
$$\rho_g u_g = r\rho_p \tag{34}$$

3.9.2 Composite Propellent

Premixed Flame Produces Oxygen Rich gases. AP Can Burn itself in exothermic Reaction in Decomposition into Ammonium and $HClO_4$. Giving Rise to oxydizing gases. Combustion is Controlled by the Diffusion Process. The Final flame is made up of the diffusion and the Premixed Flame. At Lower Pressure Chemical Reactions are Slow and the Premixed Flame From the AP Dominates. At higher Pressure the Chemical Reactions are Faster and the Combustion is Controlled (Limited) by the Diffusion Process.

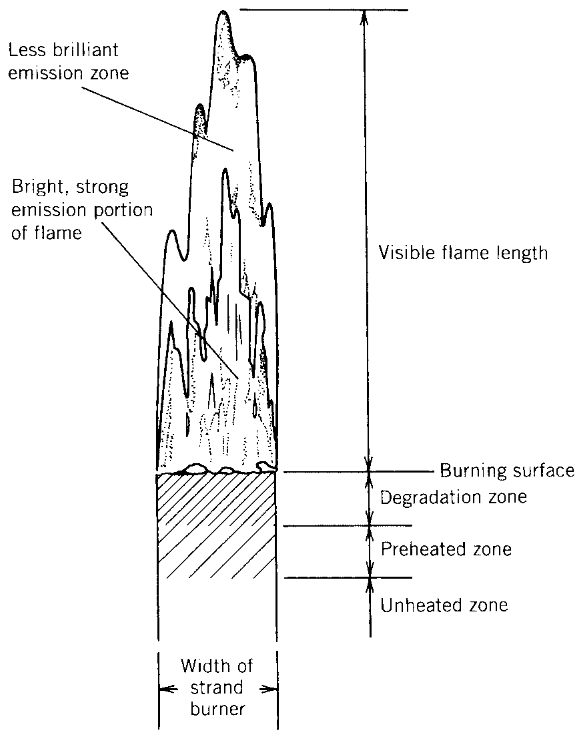

Figure 4: Composite Combustion Process

Modelling The Flame in Premixed Flame
Now the Pre mixed flame we apply the Energy concervation equation is given by

$$\dot{m}C(T_s - T_i) + \dot{m}q \tag{35}$$
$$\dot{m} = r\rho_g \tag{36}$$
$$\dot{m}C(T_s - T_i) + \dot{m}q = k_g \frac{T_f - T_s}{X^*} \tag{37}$$
$$r = \frac{k_g \left(\frac{T_f - T_s}{X^*}\right)}{\rho_g (C(T_s - T_i) + q)} \tag{38}$$

Modelling Flame

In case of modelleing flame in double Based and Compoiste Propellent We have taken different approaches. In case of double based appraoch we modelled fizz zone using eularean principles thus concentrating on region and not of specific particle thus we applied laws on specific region. In composite Propellent Appraoch we modelled buning of Propellent actually thus followd langrangian appraoch.

4 Glossary

- **SRB** - Solid Rocket Booster
- **SRM** - Solid Rocket Motor