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Design of a Solid Rocket Propulsion System

Case Report

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Abstract

Rocket Propulsion is a class of jet propulsion that produces thrust by ejecting stored matter called propellant. A solid-propellant rocket is a rocket with a rocket engine that uses solid propellants (fuel/oxidizer). A simple solid rocket motor consists of a chamber/casing, nozzle, grain (propellant charge), and igniter. The earliest rockets were solid-fuel rockets powered by gunpowder; they were used in warfare by the Chinese, Indians, Mongols and Persians, as early as the 13th century. The latest rockets use Aluminum perchlorate, as oxidizer, Aluminum as a fuel and HTPB as a fuel binder and Isoforondisocyanat as a curative catalyst. Design of solid rocket propulsion system begins with the total impulse required, which determines the fuel/oxidizer mass. Grain geometry and chemistry are then chosen to satisfy the required motor characteristics. This paper presents design of solid rocket propulsion system for a sounding rocket. Different materials were selected for different parts of the propulsion system based on the factors like density, cost and availability. The design result shows that the mass of propellant is 960 kg with volume of 0.546176383 m^3 in the length of 1.944787875 m.

Nomenclature

A_T = nozzle throat area
A_B = Burning surface area
Pt = motor chamber pressure
K_n = Ratio of Burning surface area to the nozzle throat cross-section area
a = Burn rate pressure coefficient
a = Burn rate pressure conversion factor, MPa to Pa units (a =1 000 000^n)
ρ = propellant mass density
c* = propellant characteristic exhaust velocity
n = Burn rate pressure exponent
m = mass of propellant
V = volume of propellant
Ac = area of circle
A_{tr} = area of triangle
Ag = segment area
At = total front area of propellant
ΔL = expansion longitudinally in inches,
ΔD = expansion diametrically,
L = length of the case in inches,
D = Inner diameter of the case in inches,
E = Young's Modulus of Elasticity measured in kilo pound/in2 (or 1000 lb/in2),
D = Casing thickness in inches,
V = Poisson's ratio
HTPB- Hydroxyl Terminated Polybutadiene

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Introduction

PROPULSION a broad sense is the act of changing the motion of a body. Propulsion mechanisms provides a force that moves bodies that are initially at rest, changes a velocity or overcomes retarding forces when a body is propelled through a medium [1]. Rocket propulsion is a class of jet Propulsion that products thrust by ejecting stored matter called propellant. Rocket propulsion system can be classification according to the types of energy source, according to basic function, and according to types of function. According to the energy source rocket propulsion can be classified as chemical, nuclear, and solar. According to the basic function we can classify it as booster staging, sustainer, attitude control, orbit station keeping, etc. According to types of function rocket propulsion can be classified as aircraft, issile, assisted take-off, space vehicle, etc. A solid propellant rocket, a kind of chemical rocket propulsion, is a simple propulsion system that consists of a high-pressure vessel that contains all the solid components needed. The fuel and oxidizer are intimately mixed together and cast into a solid mass, called grain. The propellant grain usually has a hole down the center of the chamber, which is called perforation, and may be shaped in various ways.

The functioning starts with an ignition system whose firing causes the beginning of a chemical reaction over the solid surface in the perforation. Once ignited, a simple solid rocket motor cannot be shut off, as it contains all the propellants needed for combustion all together in the chamber where they are burned. After the ignition, the propellant grain burns on the entire inner surface of the perforation, until the end of the propellant. The heated gases generated during the solid combustion pressurize the inside of the chamber and are finally expelled through a nozzle which accelerates them producing the reaction force needed to move, known as thrust [2].

Solid Propellant Compositions

Black Powder

By the wording 'solid', it may imply to us that the propellant composition is in some sort of hardened state. In the old days (but still around today) black powder, a mechanical mix consisting of potassium nitrate (KNO_3), sulfur (S) and charcoal (C) would either be compressed mechanically or by adding rubber Arabicum (natural rubber) to solidify the otherwise powdery state of the ingredients. Both methods were essential for a good functioning of a rocket in order to ensure smooth combustion with time whereas a propellant in just powdered state would more likely lead to unpredictable burning with the increased risk of motor rupture (explosion) [3]. When black powder burns, it does this with a high temperature flame producing huge amount of smoke and solid products.

Figure 1. Solid propellant with their different perforation and thrust Vs time graph for each perforation.

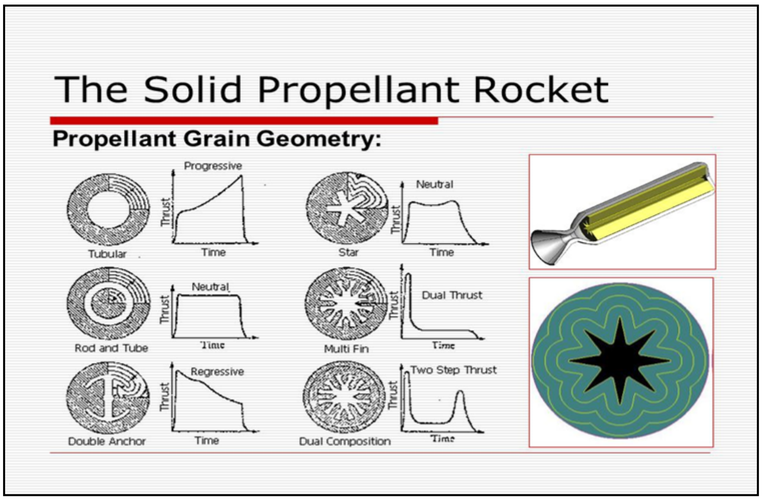


Table 1. Good balanced mixture for rocket use when using black powder [3].

Ingredient	Function	Formula	% per mass
Potassium Nitrate	Oxidizer	KNO_3	75
Sulfur	Fuel	S	10
Carbon	Fuel	C	15

Table 2. Characteristicsof Black Powder [3].

Black Powder	Value
Specific impulse	830 m/s
Reaction temperature	3070 K
Solid products	56%
Gaseous products	43%
Water	1%

Black powder is still in use as a rocket propellant, but only found in fireworks and model rocket motors. Black powder may occasionally be used for igniting solid propellant rocket motors. Due to its low specific impulse, hygroscopic and powdery nature, the propellant has for long been replace by far more energetic, mechanical strong and stable compounds.

Modern Solid Propellant

Modern solid propellants consist of ingredients that are in a solid and liquid state during the mixing process. All solids are typically based on an oxygen rich salt like ammonium perchlorate (NH_4ClO_4) as the oxidizer and aluminum powder (Al) as fuel. Ammonium perchlorate contains 54.5% per weight oxygen. The liquid part is often an industrial long chain polymer. The purpose of the polymer is to harden and thereby fixate the powered based part of the mix.

A commonly used polymer is Hydroxyl-Terminated Polybutadiene or HTPB ($C_xH_yO_z$). The hardening process is triggered by adding a hardener agent like Isoforondiisocyanat ($C_{12}H_{18}N_2O_2$) in order to start the cross-polymerization process. After mixing the propellant typically has a viscous state. The mixture is then poured by gravity or by pressure over to the rocket chamber where a casting mandrel is placed in the center of the engine. The geometry of the mandrel will vary from program to program. Often it can have a star shape. When the propellant has filled the allowable volume, the motor will be transported to a curing oven.

Typically, the mix will be allowed cure for a weekly long period at an elevated temperature (60-70°C). When cured, the propellant grain can be compared to the sensation of touching an ink rubber. A modern composite propellant charge can contain eighteen different ingredients. The main important ones are the molecule holding the oxygen (oxygen rich salt) and a fuel, e.g. aluminum, beryllium or the fuel binder itself. Other minor per mass ingredients are additives for chemical stability, pot life plasticizer, burn rate catalysts, anti-softening, curing additives, and more [3].

A modern composite propellant can give 2500 m/s (255 s) specific impulse.

Let take thrust 60 kN= 60,000 N.

mass of propellant consumed per second = thrust/ Specific impulse (I_{sp}) ----- (1)

mass of propellant consumed per second =60,000 (N) /2500(m/s) = 24 (*kg/sec*)

Mass of expelled propellant =mass of propellant consumed per secondEffective firing duration ----- (2)

Effective firing duration = 100,000 (m) / 2500 (m/s) = 40 sec

Mass of expelled propellant = 24 (*kg/sec*) * 40sec = 960 *kg*.

Total impulse = thrust * effective firing duration -----(3)

Total impulse=60,000 N * 40 sec =2,400,000 N-sec

Composite propellants are used in everything from small tactical rocket motors for missiles to the big and very powerful solid rocket boosters for launch vehicles. To balance the amount of oxygen with the amount of fuel is important in rocketry. In chemistry we know that a stoichiometric reaction is a complete chemical reaction between the amounts of reactants (oxidizer + fuel) and products. This is not always practically possible or wanted based on performance. As an example: in solid propellant mixtures you may find that theoretically you should have above 88% oxidizer concentration in your mix, since this will give you higher performance (specific impulse). That can sometimes be difficult to achieve since you may have other requirements on your propellant like mechanical strength and chemical bonding properties.

In modern solid propellant mixes we have to assess the balance of three major parts, the amount of oxidizer, the amount of fuel and the amount of binder. All solid propellants are potentially dangerous mixtures. This is so since they contain oxidizer and fuel already mixed together. The composition may be fairly robust but still it may just need a spark to start the combustion. As soon as the composition has started to burn, it will be very difficult or impractical to stop it.

Depending on the chemical composition used, a solid propellant

Table 3. Typical modern composite solid propellant mixture [3].

Ingredient	Function	Formula	% per mass
Ammonium Perchlorate	Oxidizer	NH_4ClO_4	68
Aluminum	Fuel	Al	20
HTPB	Fuel-Binder	$C_xH_yO_z$	11.5
Isoforondiisocyanat	Curative	$C_{12}H_{18}N_2O_2$	0.5

Table 4. Modern composite solid propellant mixture with corresponding mass.

Ingredient	Function	Formula	Mass (kg)
Ammonium Perchlorate	Oxidizer	NH_4ClO_4	652.8
Aluminum	Fuel	Al	192
HTPB	Fuel-Binder	$C_xH_yO_z$	110.4
Isoforondiisocyanat	Curative	$C_{12}H_{18}N_2O_2$	4.8

can burn slowly or not at all in at ambient pressure. However, if you ignite the same propellant in a confined volume, you may observe an opposite result, a quick reaction. Most solid propellants burn at a higher rate at a higher pressure. Sometimes a rocket engineer wants to increase the propellant burn rate withoutincreasing the pressure so much. For such cases we often add in iron oxides (e.g. Fe_2O_3) in amounts of 0.5 – 3.0% all depending on the chemicals used and the needs required. For such cases iron oxides behaves as a burn rate catalyst [3].

Determination of Area Ratio (K_n)

Kn is the ratio of the burn area of the propellant to the area of the nozzle throat. Add up all the exposed (uninhibited) surfaces of the propellant to get the burn area (the inside core area, plus the top & bottom faces of all the grains, for example). Call this area AB for burn area. Then, find the area of the nozzle's throat: AT = ¼ pi * D_T^2 where D_T is the throat diameter. If it has multiple throat openings, add up all the throat areas.

$K_n = A_B / A_T$ ----- (4)

Keep in mind that K_n is an instantaneous, time-varying value it is continually changing as the propellant burns. Depending on the grain geometry, the K_n may increase or decrease (or both) during the total burn time of the motor. The intial K_n is important because it affects how easily the motor will ignite. The maximum K_n or peak K_n is important because it is directly related to the peak chamber pressure. During the start-up phase of the motor burn, just after ignition, a sufficient initial K_n provides a transition to the equilibrium phase of combustion. If the initial K_n is too low, the motor may not reach steady-state burning. As the motor begins to come up to pressure, the combustion gases begin to flow. When there is gas flow there is reduced pressure.

If the pressure drop is high enough, the motor will cease (stop) to ignite (a "chuff"). If there is sufficient residual heat and pressure, the motor will ignite again and continue chuffing until the K_n increases enough to transition to steady combustion. As a practical guideline, an initial K_n of 220 is sufficient to produce reliable ignition. If the propellant is more energetic (finer AP or catalyzed), a lower initial K_n (180 to 200) could be enough. If the propellant has low-energy additives (fuel rich or low-energy effects chemicals), a higher K_n (240 or higher) may be required [4].

The term steady-state infers the operating condition whereby chamber pressure is solely a function of grain burning-surface area. In other words, the generation of combustion gases, and outflow of gases through the nozzle, are in a state of equilibrium (balance). Therefore, this excludes the initial pressure build-up as well as the pressure tail-off at burnout [5].

Table 5. Three common proportion modern propellant with corresponding specification [9].

Composition	Propellant 1	Propellant 2	Propellant 3
NH_4CLO_4(%)	70	68	72
Aluminum	16	18	16
Binder and additives	14	14	12
Density (lbm/in^3)	0.0636	0.0635	0.0641
Burning rate at 1000 psi (in/sec)	0.349	0.276	0.280
Burning rate exponent	0.21	0.3-0.45	0.28
Temperature coefficient of pressure (% 0F)	0.102	0.09	0.10
Adiabatic flame temperature(0F)	5790	6150	5909
Characteristic velocity (ft/sec)	5180	5200	5180

Table 6. Aluminized Ammonium perchlorate as a function of chamber pressure for expansion to sea level [9].

Chamber pressure(psia)	1500	1000	750	500	200
Chamber pressure (atm) or pressure ratio p1/p2	102.07	68.046	51.034	34.023	13.609
Chamber temperature(K)	3346.9	3322.7	3304.2	3276.6	3207.7
Nozzle exit temperature (K)	2007.7	2135.6	2226.8	2327.0	2433.6
Chamber enthalpy(cal/g)	-572.17	-572.17	-572.17	-572.17	-572.17
Exit enthalpy(cal/g)	-1382.19	-1325.15	-1282.42	-1219.8	-1071.2
Entropy((cal/g-K)	2.1826	2.2101	2.2597	2.2574	2.320
Chamber molecular mass(kg/mol)	29.303	29.215	29.149	29.050	28.908
Exit molecular mass(kg/mol)	29.879	29.853	29.820	29.763	29.668
Exit mach number	3.20	3.00	2.86	2.89	2.32
Specific heat ratio-chamber, K	1.1369	1.1351	1.1337	1.1318	1.1272
Specific impulse, vacuum (sec)	287.4	280.1	274.6	265.7	242.4
Specific impulse, sea level expansion (sec)	265.5	256.0	248.6	237.3	208.4
Characteristic velocity c* (m/sec)	1532	1529	1527	1525	1517
Nozzle area ration, A2/Ata	14.297	10.541	8.507	8.531	6.300
Thrust coefficient, cfa	1.700	1.641	1.596	1.597	1.529

$$p_t = \left[K_n * \frac{a}{\alpha} * \rho * C^* \right]^{\frac{1}{1-n}} \text{ ----- (5)}$$

initial K_n =195.7 and let take nozzle throat area A_T = 0.00397503904 m²

A_B = 0.77791514 m².

Chamber Sizing

A pressure vessel is a container designed to hold gases or liquids at a pressure substantially different from the ambient pressure. Design involves parameters such as maximum safe operating pressure and temperature, safety factor, corrosion allowance and minimum design temperature (for brittle fracture). Many pressure vessels are made of steel, Some are made of composite materials, such as filament wound composite using carbon fiber held in place with a polymer due to the very high tensile strength of carbon fiber these vessels can be very light, but are much more difficult to manufacture [6]. Choosing the grain geometry is the primary way to control the overall K_n curve for a motor design. The way the surface area of the propellant changes as the burn progresses is how the K_n curve will be shaped. Also, the nozzle throat diameter will scale the overall K_n curve inversely proportional to the square of the throat diameter [4].

$A_B = C * L$ ----- (6)

C = 0.4 m
L = 1.944787875 m

$\rho = m/v$ ---- (7)

V = 0.546176383 m³

$A = V/L$ ----- (8)

At = 0.280841108 m²

Atr =(1/2) at * bt *sin ct* ----- (9)

Atr = 0.002490529833 m²

Ag=θ/(360) πr²-(1/2) ab *sin c* ------ (10)

Ag = 0.003380903844 m²

Ac = At - (Atr + Ag) -----(11)

Ac = 0.274969675 m²

$Ac = \pi (r_2^2 - r_1^2)$ ----- (12)

r_2 = 0.302976831 m

Different materials were selected for different parts of the motor. Major factors like density, cost and availability affected the selection process [7].

Aluminum 6061-T6

It is an alloy of aluminum with magnesium and silicone as ma-

Figure 2. Propellant perforation and segment area portion with dimension.

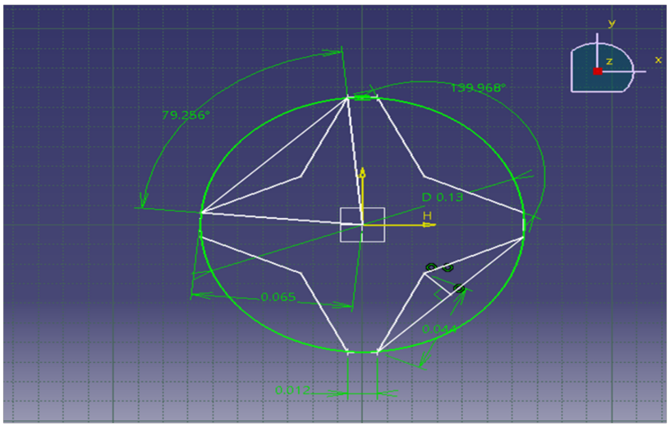
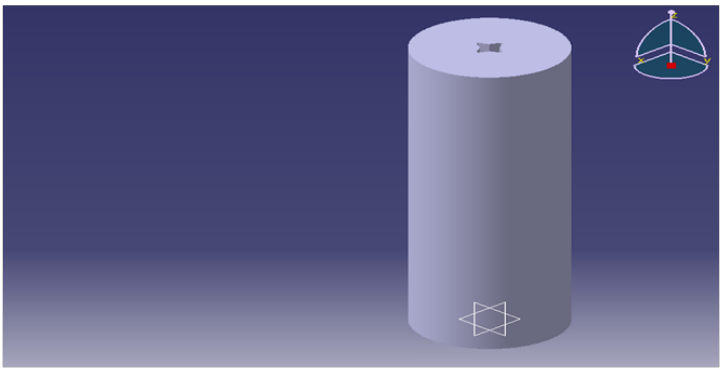


Figure 3. Symmetrical view of total propellant.



jor alloying elements and is widely used in aircraft construction industry. With a density of 2.7 g/cm^3 it is best suited for experimental projects in aerospace and its ease of machining is advantageous. This type of aluminum is incredibly strong, relativelylight-weight and is commonly used as motor casings for commercially produced amateur rocket motors. A minimum diameter rocket is a type of rocket which uses the walls of the rocket airframe as the casing for the motor, whereas traditional collegiate and amateur building methods use a motor mount to fit a motor which has a smaller diameter than the airframe. High-altitude rockets are incredibly efficient when they are built minimum diameter, therefore maximizing the potential amount of solid propellant.

The following properties of the alloy were observed for calculation of design pressure and burst pressure of the casing.

Two equations can be used to approximate the expansion of a motor casing, for both length and diameter, during a propellant burn due to chamber pressure. For 6061-T6 Aluminum, the elongation at break is 12%, the Young's Modulus is 10,000 ksi, Poisson's Ratio is 0.33 [8].

$$\Delta L = \frac{PLD}{4Ed}\left(1-2\nu\right) \text{ ----- (13)}$$

d = 0.04147165987 mm

Full-scale nozzle surface temperature history can be approximated in a small-scale nozzle by appropriate selection of wall thickness [9] take factor of safety 120 and d = 5 mm.

$$\Delta D = \frac{PD^2}{4Ed}\left(1-\frac{\nu}{2}\right) \text{ ----- (14)}$$

ΔD = 0.001488150906 mm

Figure 4. Front, top and side view of propellant.

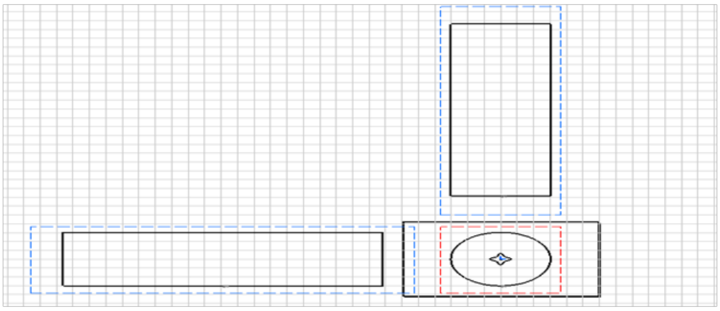


Table 7. Properties of Al 6061-T6 [8].

S/N	Properties	Values
1	Yield Strength (MPa)	241
2	Ultimate Strength (MPa)	290
3	Modulus of Elasticity (Mpa)	68,310
4	Poisson Ratio	0.33
5	Strength Ratio (Fty/Ftu)	0.831
6	Burst Factor	1.337

Conclusion

Design of solid rocket propulsion system includes determination of the total impulse required, the weight of fuel/oxidizer, grain geometry, and grain chemistry that satisfy the required motor characteristics. The paper presented selection of modern composite solid propellant mixture with a mass 960kg, volume of 0.546176383 m^3 and 1.944787875 m in length. The main chamber area portion is divided in to three segments: Segment area = 0.003380903844 m^2, triangle area = 0.002490529833 m^2 and area of circle=0.274969675 m^2. The values of burning area and initial area ratio (K_n) are 0.77791514 m^2 and 195.7 m^2 respectively.

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