Available online www.ejaet.com

European Journal of Advances in Engineering and Technology, 2016, 3(5): 23-28

Research Article ISSN: 2394 - 658X

Analytical Computation of Bell Cone Nozzle with AP+AL+HTPB Propellant Rocket Motor for Variable Percentage of Oxidizer Fuel Mixture

Laeeq Ahmed Khan Aqil and Md. Abdul Raheem Junaidi

Department of Mechanical Engineering
Muffakham Jah College of Engineering and Technology, Hyderabad, Telangana, India
raheem.junaidi@mjcollege.ac.in

ABSTRACT

A rocket motor is designed based on the burning rate of 10mm/sec with the burning index of 0.4, which shall be useful to fire at sea level to perform its objective at the maximum altitude of 2 km. Thickness of the shell will be calculated based on the maximum operating pressure of the rocket motor and its throat area, based on the mass flow rate, chamber pressure, and characteristic velocity of propulsion. Propellant composition shall provide its characteristic velocity. Based on the burning rate, chamber pressure shall be evaluated and also operational range will be helpful in evaluating its maximum thrust generated at the time of its operation. A suitable material shall be selected to withstand internal pressure of the rocket motor generated during operation.

Key words: Solid Propellants, oxidizer, nozzle, Rocket motor

INTRODUCTION

A solid-fuel rocket uses solid propellants (fuel/oxidizer). Olden days rockets were solid-fuel by gunpowder and were used by Chinese ,Indus, Mongolians and Arabians in combat field in early 13th century(AD) All rockets used some form of solid powdered called propellant till 20th century, before liquid & hybrid rockets came in existence [1] . Solid rockets are still used today larger applications for their simplicity and reliability as it can remain in storage for long periods, and then reliably launch on short notice, they have been frequently used in military applications such as missiles. The lower performance of solid propellants (as compared to liquids) does not favor their use as primary propulsion in modern medium-to-large launch vehicles customarily used to orbit commercial satellites and launch major space probes. Solid rockets are used as light launch vehicles for low Earth orbit (LEO) payloads under 2 tons or escape payloads up to 1000 pounds. Solid propellant rockets discuss the burning rates, motor performance, grain configurations, and structural analysis. The propellant is contained and stored directly in the combustion chamber, sometimes hermetically sealed in the chamber for long-time storage (5- 20 years). Motors come in many different types and sizes, varying in thrust from about 2 N to over 4 million N.

DESIGN REQUIREMENTS

A solid propellant rocket is formed by following main components, a case containing the solid propellant and withstanding internal pressure when the rocket is operating, the solid propellant charge (or grain), which is usually bonded to the inner wall of the case and occupies before ignition the greater part of its volume. When burning, the solid propellant is transformed into hot combustion products. The volume occupied by the combustion products is called combustion chamber. The nozzle channels the discharge of the combustion products and because of its shape accelerates them to supersonic velocity. The igniter, which can be a pyrotechnic device or a small rocket, starts the rocket operating when an electrical signal is received. One can consider that the solid propellant after manufacturing is in a metastable state. It can remain inert when stored (in appropriate conditions) or it can support after ignition its continuous transformation into hot combustion products (self-combustion).The solid rocket is therefore inherently simple and therefore can possess high intrinsic reliability. After ignition, a solid rocket motor normally operates in accordance with a preset thrust program until all the propellant is consumed. All the efforts should be directed to the accurate prediction of the thrust (and pressure) programs to get the benefit of solid rocket motor concept

PERFORMANCE CHARACTERISTICS

Design of SRM involves multiple important parameters to define its performance are thrust, which is a measure of the total force delivered by a rocket motor for each second of operation, Essentially, thrust is the product of mass times acceleration also of course, gravity, pressure, and other consideration must be taken into relation. After thrust another important parameter, specific impulse (I_{sp}) which provides a comparative index to measure the number of pounds of thrust each pound of propellant will produce To compute specific impulse, the thrust is divided by the mass flow, or weight of the gas flowing through the nozzle throat per second. Also total impulse, which is abbreviated I_t. Computing this value is a simple matter of multiplying the motors thrust by its period of operation in seconds. Mass fraction is also a basic parameter for motor design. A high mass fraction indicates that more of the motor mass is propellant than is involved in the case, nozzle, and other components which do not produce thrust [2]. A high mass fraction can be achieved by optimizing motor and structural design for minimum weight and or by using more dense propellants, which then require smaller combustion chambers. The mass fraction may be defined as the ratio of the propellant

Solid Propellants

Solid propellants are mono and form single systems in which the oxidizer and fuel components are combined in a single mixture with a liquid material which holds them in suspension. And composition of it cast (poured) into the case, where it is cured to a solid state [3]. Propellant specialists divide solid propellants into two classifications - double-base and composite. These classifications refer to the physical and chemical characteristics of the propellants, as well as to the types of materials used in their manufacture [4]. Most composite propellants also may use a variety of chemicals to increase or decrease the burning rate (to control) hot gas production rate), provide better physical properties than are obtainable with the basic binder, or to regulate chemical reactions for better control during the manufacturing process.

Binders

Both natural and man-made materials have been used as binder-fuels for solid propellant grains, including asphalt, and synthetic liquid prepolymers. All of the organic prepolymers have rubbery properties following cure and form a strong matrix (binding structure) within which the inorganic fuels and chemical oxidizers are solidly bound.

The most commonly used liquid prepolymers cure up much like rubber.The prepolymers used as binders in composite propellants are initially liquid, so that the fuel and oxidizer can be blended in more easily prior to the start of polymerization (cure). The polymers cure to an irreversible solid form by cross linking (formation of long chain-like molecules linked together).

Additives

To obtain specific properties, several special ingredients may be added to composite propellants, including: oxidants, antioxidants, and curing and burning rate catalysts.

In other instances, an increase in the burning rate may be desirable, and this can be obtained by adding chemical agents known as burning rate catalysts which cause an increase in the combustion reactions.

Oxidizers

Number of inorganic chemicals is used to supply the oxygen required to support the combustion of metallic and other fuels in solid propellants. The amount of oxygen provided by each oxidizer depends upon its molecular structure, and certain performance characteristics may restrict applicability to various degrees.In general, the perchlorates: potassium, ammonium, lithium, sodium, and nitronium - have more oxygen in their structure than do the nitrates: ammonium, potassium, and sodium, although availability of oxygen is not the only consideration when choosing the oxidizer to be used.

STEADY-STATE REGIMES

Modern SRM design is characterized by a tridimensional geometry, generally associating axisymmetric and star-shaped patterns or intersegments (segmented grains) generate noticeable pressure variation which cannot be predicted by too crude computation [5]. Another trend is to use composite (filament winding) case, more deformable than a metallic one; the case deformation and the grain deformation should be taken into account in this situation and it is particularly important for the first phase of operation following ignition. So a natural trend is to couple the internal aerodynamics and the case/grain deformation, by taking into account the visco-elastic feature of the grain.

Aerodynamics at $t + \delta t$: new aerodynamic computation, A totally coupled computation of aerodynamics and geometry evolution requires use of variable meshes in the combustion chamber and in the grain and is very expensive in terms of computation time. In the classical approach previously mentioned, where accuracy is, a priori, of first order, δt can be chosen to maintain the computation time in a reasonable range. It should be also mentioned that the modern codes do not distinguish the flow zones at low velocity and the flow zones at high velocity. The whole flow from the combustion surface to the nozzle exit is described and computed with the same compressible equations and the same numerical methods [6]. This way of computing avoids difficult and unsolved issues of matching between domains (combustion chamber, nozzle).

Combustion of Solid Propellants

The combustion process in rocket propulsion systems is very efficient, when compared to other power plants, because the combustion temperatures are very high; this accelerates the rate of chemical reaction, helping to achieve nearly complete combustion [7]. The energy released in the combustion is between 95 and 99.5% of the possible maximum. This is difficult to improve. There has been much interesting research on rocket combustion and we have now a better understanding of the phenomena and of the behavior of burning propellants. This combustion area is still the domain of specialists. The rocket motor designers have been concerned not so much with the burning process as with controlling the combustion (start, stop, heat effects) and with preventing the occurrence of combustion instability [8].

Design Specifications

Design 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. The following are chosen or solved simultaneously. The results are exact dimensions for grain, nozzle, and case geometries: The grain burns at a predictable rate, given its surface area and chamber pressure. The chamber pressure is determined by the nozzle orifice diameter and grain burn rate. Allowable chamber pressure is a function of casing design. The length of burn time is determined by the grain web thickness [9].

The grain may or may not be bonded to the casing. Case-bonded motors are much more difficult to design, since the deformation, under operating conditions, of the case and the grain must be compatible. Common modes of failure in solid rocket motors include fracture of the grain, failure of case bonding, and air pockets in the grain. All of these produce an instantaneous increase in burn surface area and a corresponding increase in exhaust gas and pressure, which may potentially induce rupture of the casing. Another failure mode is casing seal design. Seals are required in casings that have to be opened to load the grain. Once a seal fails, hot gas will erode the escape path and result in failure. This was the cause of the Space Shuttle Challenger disaster.

METHODOLOGY

For designing solid propellant rocket motors, there is no single, well-defined procedure or design method. Each class of application has some different requirements. Individual designers and their organizations have different approaches, background experiences, sequences of steps, or emphasis. The approach also varies with the amount of available data on design issues, propellants, grains, hardware, or materials, with the degree of novelty (many 'new' motors are actually modifications of proven existing motors), or the available proven computer programs for analysis. Usually the following items are part of the preliminary design process. We start with the requirements for the flight vehicle and the motor. If the motor to be designed has some similarities to proven existing motors, their parameters and flight experience will be helpful in reducing the design effort and enhancing the confidence in the design. The selection of the propellant and the grain configuration are usually made early in the preliminary design; is not always easy for the propellant to satisfy its three key requirements, namely the performance (I_s), burning rate to suit the thrust-time curve, and strength (maximum stress and strain). A well-characterized propellant, a proven grain configuration, or a well-tested piece of hardware will usually be preferred and is often modified somewhat to fit the new application.

Compared to a new development, the use of proven propellant, grain design, or hardware components avoids many analyses and tests. An analysis of the structural integrity should be undertaken, at least in a few of the likely places, where stresses or strains might exceed those that can be tolerated by the grain or the other key components at the limits of loading or environmental conditions. An analysis of the nozzle should be done, particularly if the nozzle is complex or includes thrust vector control. Such a nozzle analysis was described briefly in an earlier section of this chapter. If gas flow analysis shows that erosive burning is likely to happen during a portion of the burning duration, it must be decided whether it can be tolerated, or whether it is excessive and a modification of the propellant, the nozzle material, or the grain geometry needs to be made. Usually a preliminary evaluation is also one of the resonances of the grain cavity with the aim of identifying possible combustion instability modes. Motor performance analysis, heat transfer, and stress analyses in critical locations will usually be done [10].

There is considerable interdependence and feedback between the propellant formulation, grain geometry/design, stress analysis, thermal analysis, major hardware component designs, and their manufacturing processes. It is difficult to finalize one of these without considering all others, and there may be several iterations of each. Data tests of laboratory samples, subscale motors, and full-scale motors have a strong influence on these steps. Preliminary layout drawings or CAD (computer-aided design) images of the motor with its key components will be made in sufficient detail to provide sizes and reasonably accurate dimensions. For example, a preliminary design of the thermal insulation (often with a heat transfer analysis) will provide preliminary dimensions for that insulator. The layout is used to estimate volumes, inert masses, or propellant masses, and thus the propellant mass fraction. If any of these analyses or layouts shows a potential problem or a possible failure to meet the initial requirements or constraints, then usually a modification of the design, possibly of the propellant, or of the grain configuration may need to be made. The design process needs to be repeated with the changed motor design. If the proposed changes are too complex or not effective, then a change in the motor requirements may be the cure to a particular problem of noncompliance with the requirements [11]. If the performance requirements are narrow and ambitious, it will be necessary to study the cumulative tolerances of the performance or of various other parameters. For example, practical tolerances may be assigned to the propellant density, nozzle throat diameter (erosion), burn rate scale factor, initial burning surface area, propellant mass, or pressure exponent. These, in turn, reflect themselves into tolerances in process specifications, specific inspections, dimensional tolerances, or accuracy of propellant ingredient weighing [12].

Cost is always a major factor and a portion of the design effort will be spent looking for lower-cost materials, simpler manufacturing processes, fewer assembly steps, or lower-cost component designs. For example, tooling for casting, mandrels for case winding, and tooling for insulator molding can be expensive [13]. The time needed for completing a design can be shortened when there is good communication and a cooperative spirit between designers, propellant specialists, analysts, customer representatives, manufacturing people, test personnel, or vendors concerned with this effort.

ANALYTICAL SOLUTION

This paper shows one method for making a preliminary determination of the design parameters of a solid rocket using a composite propellant. The rocket is launched at altitude from sea level:

Table -1 Design Parameters of a Solid Rocket using a Composite Propellant

Specific impulse (actual)	I_s	= 163 sec
Burning rate	**r**	= 10mm/sec
Propellant density	ρ_b	= 2355 kg/m^3
Specific heat ratio	**k**	= c_p/c_v = 1.25
Chamber pressure, nominal	**P**	= **97.6** MPa
Desired average thrust	**F**	= 4 tons
Maximum vehicle diameter	**D**	= 25 cm
Desired duration	t_b	= 14.25 sec
Pressure index	**n**	= 0.4
Yield stress of material	**σ**	=1327 MPa
Ambient pressure	**p**	0.2MPa

Table -2 Vehicle Payload (includes structure) Composition of Propellant

Fuel	5-22% Powdered Aluminum
Oxidizer	65-70% Ammonium Perchlorate (NH_4ClO_4 or AP)
Binder	8-14% Hydroxyl- Terminated Polybutadiene (HTPB)

Approximately neutral burning is desired.

Basic Design

- The total impulse I_t and propellant weight at sea level
- W_b is obtained from Eqs. $\mathbf{I_t = F_t\, t_b = I_s w_b}$: W_b=*75 kg*
- The propellant weight is W_b=*75 kg*. Allowing for a loss of 2% manufacturing tolerances
- The total propellant weight is 1.02 * *75* = 76.5.
- The volume required for this propellant V_b is given by $\mathbf{Vb = w_b/\rho_b}$ = .0324 m^3
- The web thickness b = rt_b = 0.05 cm.

Case Dimensions

- The outside diameter is fixed at 25cm. Heat-treated steel with an ultimate tensile strength of σ =1758 is to be used.
- The wall thickness $t = (p*D)/(2\sigma)$. t = 6.9*10^{-3} A spherical head end and a spherical segment at the nozzle end is consider

Grain Configuration

The grain will be cast into the case but will be thermally isolated from the case with an elastomeric insulator with an average thickness of 0.25 cm inside the case; the actual thickness will be less than 0.25 cm the cylindrical and forward closure regions, but thicker in the nozzle entry area. The outside diameter D for the grain is determined from the case thickness and liner to be 23.12 cm. The inside diameter Di of a simple hollow cylinder grain would be the outside diameter Do minus twice the web thickness or 13.12 cm. For a simple cylindrical grain, the volume determines the effective length, which can be determined from the equation $V_b = (\pi/4)L(Do^2 - D_i^2)$ where, L = 114 cm

The web fraction would be *2b/Do* = 0.43. The *L/Do* is (approximately) 4.9 for grains with this web fraction and this *L/Do* ratio. use of an internal burning tube. A tubular configuration is selected, although a slotted tube or fins would also be satisfactory. The initial or average burning area will be found from $F = \dot{w} I_s = P_b A_b r l_s, A_b.$

The actual grain now has to be designed into the case with spherical ends, so it will not be a simple cylindrical grain. The approximate volume occupied by the grain is found by subtracting the perforation volume from the chamber volume. There is a full hemisphere at the head end and a partial hemisphere of propellant at the nozzle end (0.6 volume of a full hemisphere).

$$V_b = 0.5 \{(\frac{\pi}{6})D_o^3(1=0.6)+(\pi/4)D_i^2(L+D_i/2+0.3D_i/2)\}$$

This is solved for L, with D = 25 cm. and the inside diameter *Di* = 23.12. The answer is L = 144 cm. The initial internal hollow tube burn area is about $\pi Di(L + Di/2 + 0.3Di/2)$ = 1.15 m^2

Nozzle Design

The thrust coefficient C_F can be found from k= 1.20 and a pressure ratio of Pl/P2 = 488. Then CF =1.73.
The throat area: $A_t = F/pC_F = 23.8\ cm^2$
The throat diameter is Dt = 5.5 cm.
The nozzle area ratio for optimum expansion) A_2/A_t is about 27. The exit area and diameter are therefore about $A_e=595\ cm^2$ and De = 27.52. However, this is larger than the maximum vehicle diameter of 25 cm, which is the maximum for the outside of the nozzle exit. Allowing for an exit cone thickness of 0.25 cm.
This nozzle can have a thin wall in the exit cone, but requires heavy ablative materials, probably in several layers near the throat and convergent nozzle regions.

Weight Estimate

The steel case weight (assume a cylinder with two spherical ends and that steel weight density is 0.3 lbf/in^3) is

$$t\pi L\rho + (\pi/4)tD^2\rho = 241 \text{ kg}$$

With attachments, flanges, igniter, and pressure tap bosses, this is increased to 253. The nozzle weight is composed of the weights of the individual parts, estimated for their densities and geometries.

Performance

The total impulse-to-weight ratio is 1.02 Comparison with *Is* shows this to be an acceptable value, indicating a good performance. The total launch weight is 253 kg, and the weight at burnout or thrust termination is 64. The initial and final thrust-to-weight ratios and accelerations are *F/w = 7.9 and F/w*= 31.25

RESULTS AND DISCUSSION

Among different composition of propellant 70/30 oxidizer fuel mixture is selected for best mass flow rate upon change in density of wide range of composition

- Composition of propellant is selected as (AP+AL+HTPB)
- Nozzle is designed based on maximum expansion ratio of A_e/A_c=27
- Conical nozzle design data is consider for simple gomentry
- Casing material is selected based on internal pressure of chamber
- Insulator is selected based on heat transfer
- Theoretical design is suggested and geometric model is created on design data

CONCLUSIONS

Theoretical analysis of a small solid propellant rocket motor is conducted to know the propellant performance over a wide range of oxidizer / fuel ratio density Significant improvement of the motor performance does not appear likely to be achieved by variation of nozzle design. Result demonstrate the importance of considering effect of propellant density on mass flow rate of solid propellant rocket motor . The best oxidizer / fuel ratio was found to be 70/30. The conical nozzle emerged as a highly satisfactory design when the simplicity of fabrication is considered. Marginal increase in nozzle efficiency might be obtained by reducing the convergence or divergence angle.

REFERENCES

[1] N Eisenreich, HP Kugler and F Sinn, An Optical System for Measuring Burning Rates of Solid Propellants, *Propellants, Explosives, Pyrotechnics*, **1987**, 78-80.
[2] N Kubota, Survey of Rocket Propellants and their Combustion Characteristics, Chapter 1, *Rocket Propulsion Elements*, John Wiley & Sons, **2010**.

[3] MK Razdan and KK Kuo, Erosive Burning of Solid Propellants, Chapter 10, Progress in Astronautics and Aeronautics, AIAA, New York, **1984**, (90),515-598.
[4] M Summerfield (Eds.), Fundamentals of Solid Propellant Combustion, *Series on Progress in Astronautics and Aeronautics*, American Institute of Aeronautics and Astronautics, New York, **1984**, 90, 891.
[5] SD Heister and RJ Davis, Predicting Burning Time Variations in Solid Rocket Motors, *Journal of Propulsion and Power*, **1992**, 8 (3), 298.
[6] *Solid Rocket Motor Performance Analysis and Prediction*, NASA SP-8039, **1971** (N72-18785).
[7] KK Kuo and M Summerfield (Eds.), Fundamentals of Solid-Propellant Combustion. Progress in Astronautics and Aeronautics, American Institute of Aeronautics and Astronautics, New York, **1984**, 90, 8.
[8] Truchot, Design of Solid Rocket Motor Internal Insulation, Chapter 10, *Design Methods in Solid Rocket Motors, Lecture Series* LS 150, AGARD/NATO, **1988**.
[9] V Gayathri, Chemistry of Ignition and Combustion of Ammonium-Perchlorate-Based Propellants, Chapter 2, Rocket Propulsion Elements, **2010**.
[10] TL Boggs, The Thermal Behaviour of Cyclotrimethylene Trinitrate (RDX) and Cyclotetramethylene Tetranitrate (HMX), Chapter 3, *Rocket Propulsion Elements*, **2010**.
[11] CE Hermance, Solid Propellant Ignition Theories and Experiments, Chapter 5; M Kumar and KK Kuo, Flame Spreading and Overall Ignition Transient, Chapter 6, *Progress in Astronautics and Aeronautics*, AIAA, New York, **1984,** 90.
[12] EW Price, Experimental Observations of Combustion Instability, Chapter 13, *Fundamentals of Solid Propellant Combustion*, **1984**, 733-790.
[13] James S Tien, Theoretical Analysis of Combustion Instability, Chapter 14, *Fundamentals of Solid Propellant Combustion*, American Institute of Aeronautics and Astronautics,**1984**.