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Conceptual Design and Structural Analysis of Solid Rocket Motor Casing

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ABSTRACT
This paper is concern with theoretical design of the 100 kg solid rocket motor with predefined values of the burning rate is 100 mm/ sec, specific impulse is 240 sec and chamber pressure is 1000 psi. confined with selection of the material and basic concept of the rocket motor and its aspect during static test. Rocket motor is a highly complex aerospace component that consists of a metal casting, ablative liner and propellant grain. Also to determine the design pressure and burst pressure of a solid rocket motor casing .Preliminary design provided key propulsion outputs that would later be refined and assessed in the final design.
Keywords: *Rocket motor, propellant grains, Ansys, Motor casing, propulsion, design, Grain shape, Materials*

I. INTRODUCTION

A solid rocket or a solid-fuel rocket is a rocket with a motor that uses solid propellants (fuel/oxidizer).The earliest rockets were solid-fuel rockets powered by gunpowder; they were used by the Chinese, Indians, Mongols and Arabs, in warfare as early as the 13th century. All rockets used some form of solid or powdered propellant up until the 20th century, when liquid rockets and hybrid rockets offered more efficient and controllable alternatives. Solid rockets are still used today in model rockets on larger applications for their simplicity and reliability. Since solid-fuel rockets can remain in storage for long periods, and then reliably launch on short notice, they have been frequently used in military applications such as missile.

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. Solids are, however, frequently used as strap-on boosters to increase payload capacity or as spin-stabilized add-on upper stages when higher-than-normal velocities are required. Solid rockets accused as light launch vehicles for low Earth orbit (LEO) payloads under 2 tons or escape payloads up to 1000 pounds.[1]

Solid propellant motor designers employ a number of important parameters (a satiable or an arbitrary constant) to define the performance of the propellants used and the motors which power rocket propelled vehicles. Since most of these terms will be used throughout the discussions which follow, brief definitions of the more common ones are presented below.

The first and most common tome used in rocketry is 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. In actual calculations, of course, gravity, the pressure of the surrounding medium, and other considerations must be taken into account.

The thrust (total force) of a football player is much like that of a rocket. The force generated is a product of player's weight (mass) time's player's rate of acceleration.

After the thrust developed by the rocket has been determined, this value is used to compute another important parameter, specific impulse (I_m), which provides a comparative index to measure the number of pounds of thrust each pound of propellant will produce. Expressed as pound force seconds pm pound mass, I_{SP} is referred to in the rocket engineer's shorthand language as seconds of impulse. [2,4]
For designing solid propellant rocket motors, there is no single, well-defined procedure or design method. Each class of operation 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,

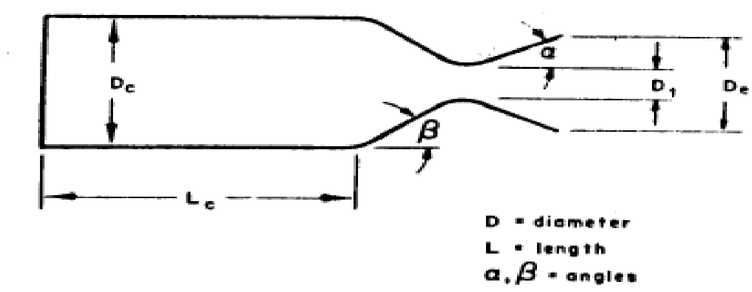

Fig.1 Motor design configuration [3]

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 configurations made early in the preliminary design.
It 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.[4]

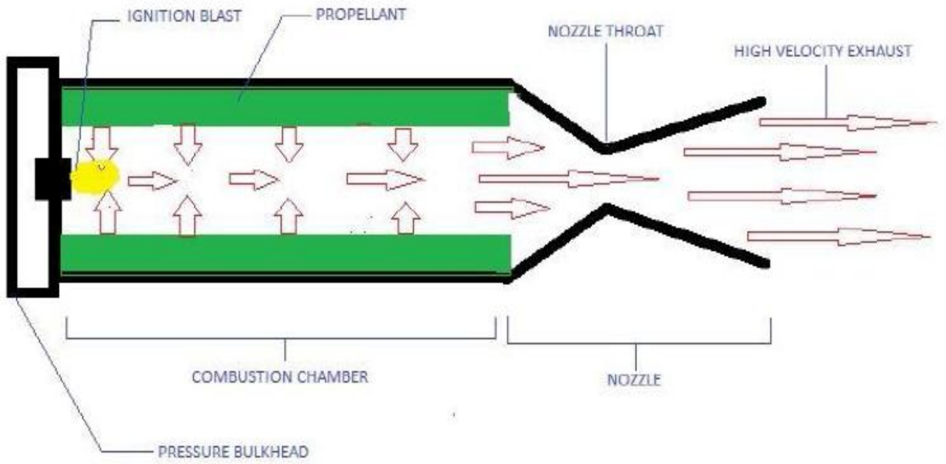

Fig.1.1 Behaviour of gases after the ignition of propellant inside combustion chamber [4]

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, γ There are 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. 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.

II. PERFORMANCE PARAMETERS

The nozzle throat cross-sectional area may be computed if the total propellant flow rate is known and the propellants and operating conditions have been chosen. Assuming perfect gas law theory.[5]

$$A_t = w_t / P_t \sqrt{\frac{RT_t}{\gamma g_c}} \quad \ldots\ldots\ldots\ldots \quad (1)$$

T_t is the temperature of the gases at the nozzle throat. The gas temperature at the nozzle throat is less than in the combustion chamber due to loss of thermal energy in accelerating the gas to the local speed of sound (Mach number = 1) at the throat. Therefore

$$T_t = T_c \left[\frac{1}{1+\frac{\gamma-1}{2}}\right] \quad \ldots\ldots\ldots\ldots \quad (2)$$

For $\gamma = 1.2$

$$T_t = (0.909)\,(T_c) \quad \ldots\ldots\ldots\ldots \quad (3)$$

T_c is the combustion chamber flame temperature in degrees Rankine (degR), given by

$$T(°R) = T(°F) + 460 \quad \ldots\ldots\ldots\ldots \quad (4)$$

Since, P_t is the gas pressure at the nozzle throat. The pressure at the nozzle throat is less than in the combustion chamber due to acceleration of the gas to the local speed of sound (Mach number = 1) at the throat. Therefore

$$P_t = P_c\left[1+\frac{\gamma-1}{2}\right]^{-\frac{\gamma}{\gamma-1}} \quad \ldots\ldots\ldots\ldots \quad (5)$$

For $\gamma = 1.2$

$$P_t = (.564)\,(P_c) \quad \ldots\ldots\ldots\ldots \quad (6)$$

The hot gases must now be expanded in the diverging section of the nozzle to obtain maximum thrust. The pressure of these gases will decrease as energy is used to accelerate the gas and we must now find that area of the nozzle where the gas pressure is equal to atmospheric pressure. This area will then be the nozzle exit area.
Mach number is the ratio of the gas velocity to the local speed of sound. The Mach number at the nozzle exit is given by a perfect gas expansion expression

$$M^2{}_e = \frac{2}{\gamma-1}\left[\left(\frac{P_c}{P_{atm}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \ldots\ldots\ldots\ldots \quad (7)$$

P_c is the pressure in the combustion chamber and P_{atm} is atmospheric pressure, or 14.7 psi.

III. DESIGN METHODOLOGY

For designing solid propellant rocket motors , there is no single ,well-defined procedures or design method. The approach also varies with the amount of available data on design issues, propellants, grains, hardware, or materials with the degree of novelty. The layout is used to estimate volumes, inert masses, or propellant mass., and thus the propellant mass fraction.

It is common to have several iterations in the preliminary design and the final design. Any major new feature can result in additional development and testing to prove its performance, reliability, operation, or cost; this means a longer program and extra resources. A simplified diagram of one particular approach to motor preliminary design and development activities or a rocket motor & other steps, such as igniter design and tests, liner/insulating selection, thrust vector control design and test, reliability analysis, evaluation of alternative designs, material specifications, inspection/ quality control steps, safety provision, special test, equipment, special test instrumentation, and so on. [6,8]

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.
Cost is always a major factor and a portion of the design effort will be spent looking for lower-cost materials, simpler manufacturing process, fewer assembly steps, or lower-cost component designs. [7]

Basic Design.

- The total impulse I_t and propellant weight at sea level.
- W_b is obtained from equation i.e. $I_t = F_t t_b = I_s W_b$: W_b=100 kg i.e. = 220 lbf .

- The propellant weight is W_b=100 kg. Allowing for a loss of 2% manufacturing tolerances
- The total propellant weight is 1.02 * 220 = 224.4 lbf .
- The volume required for this propellant v_b is given by $V_b = w_b/\rho_b$= 3349.2537 in^3.
- The web thickness b = $r*t_b$= 1 in.

Case Dimensions.

- The outside diameter is fixed at 12 inch. Heat –treated steel with an ultimate tensile strength of $\sigma = 290000$is to be used.
- The wall thickness t can be determined from simple circumferential stress as t = $\frac{P^*D}{2\sigma}$ .The value of P depends on the safety factor selected = 2. The wall thickness, t = 0.0413. [7]
- A spherical head end and a spherical segment at the nozzle end is considered.

Grain Configurations

- 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 0.10 inch inside the case .The outside diameter D for the grain is determined from the case thickness and linear to be 11.7174in. and the inside diameter is 9.7174 in. For a simple cylindrical grain ,the volume determines the effective length, which can be determined from the equation:

$$V_b = \frac{\pi}{4} L\left(D_0{}^2 - D_i{}^2\right)$$

Since, L = 49.73 in.

- The web fraction would be $\frac{2b}{D_O} = 0.1706$. The L/D_o is 4.2.
- The initial or average burning area will be found from , F =$P_b A_b r l_s$
- 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). [9,10]

$$V_b = \frac{1}{2}\left(\frac{\pi}{6}\right) D_O{}^3(1+0.6) + \left(\frac{\pi}{4}\right) D_O{}^2 L - \left(\frac{\pi}{4}\right) D_i{}^2\left(L + \frac{D\,i}{2} + \frac{0.3D_i}{2}\right)$$

- This is solved for L, with 11.7174 . and the inside diameter , Di = 9.7174. The answer is L = 144 cm.

The initial internal hollow tube burn area is about :

$$\pi D_i(L + Di/2 + 0.3Di/2) = 1710.98\ in^2$$

Nozzle Design

- The thrust coefficient C_F can be found from K=1.20 and a pressure ratio of P_1/P_2= 333. Then C_F =1.73.
- The throat area: $A_t = F/p.\ C_f$ = 6.612 in^2
- The throat diameter is D_t = 2.9016 in.
- The nozzle area ratio for optimum expansion) A_2/A_t is about 27. 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.

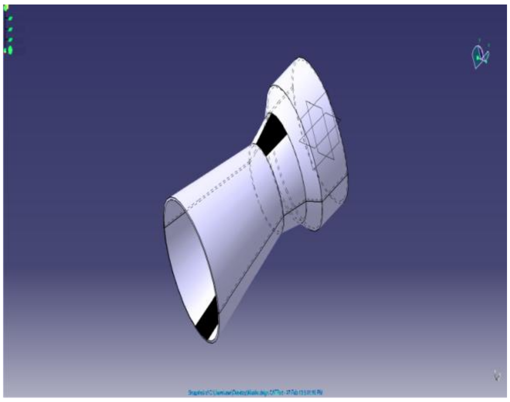

Fig.3 Nozzle

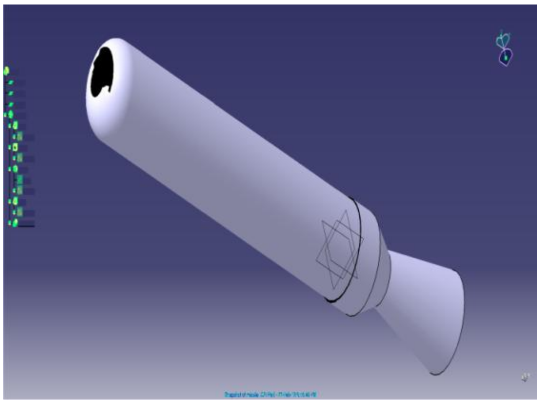

Fig.4 Screen -100 Kg Assembled of SRM

IV. STRUCTURAL ANALYSIS

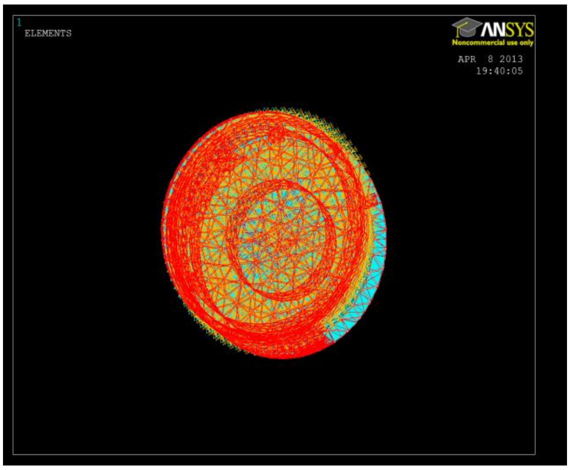

Fig.5 Applied forces on Bulkhead

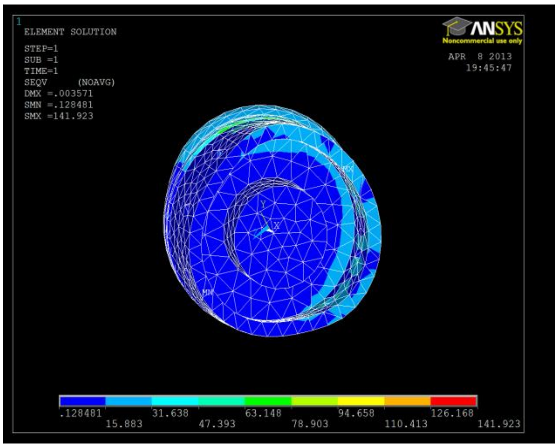

Fig.6 Stress Bulkhead

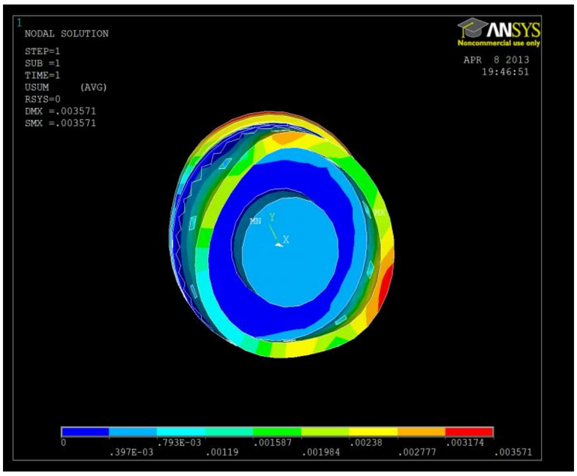

Fig.7 Displacement on Bulkhead

Fig.8. Meshed Case

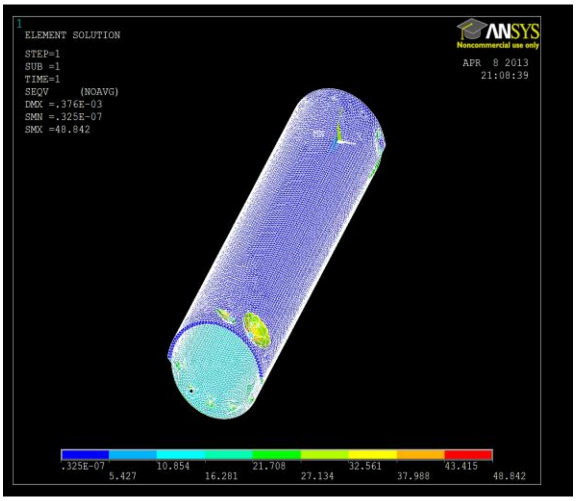

Fig.9 Deformed Case

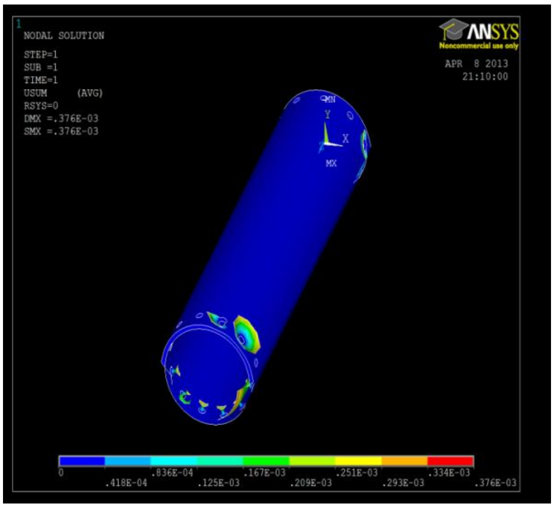

Fig.10 Degree of freedom displacement of case

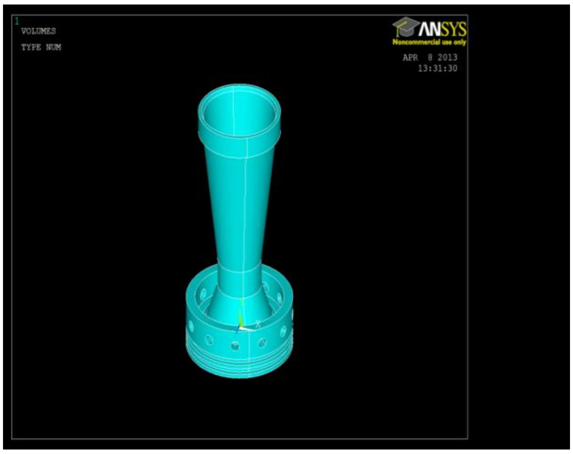

Fig.11 Nozzle

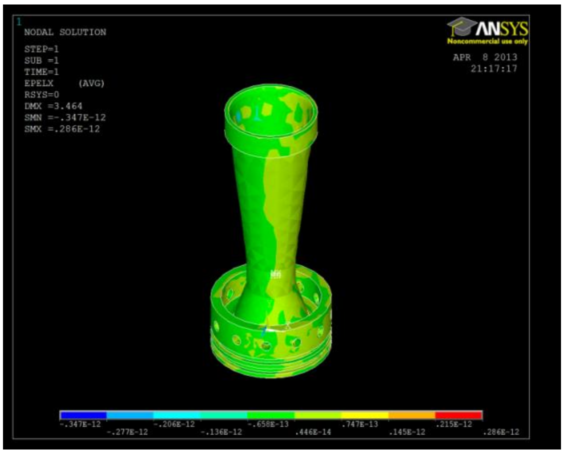

Fig.14 Elastic Strain.

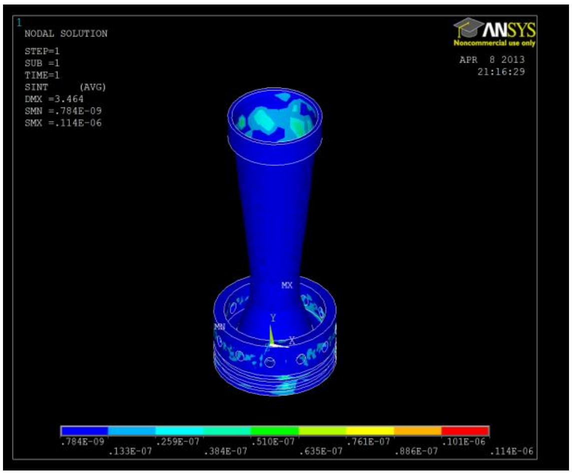

Fig.13 Stress Deformation

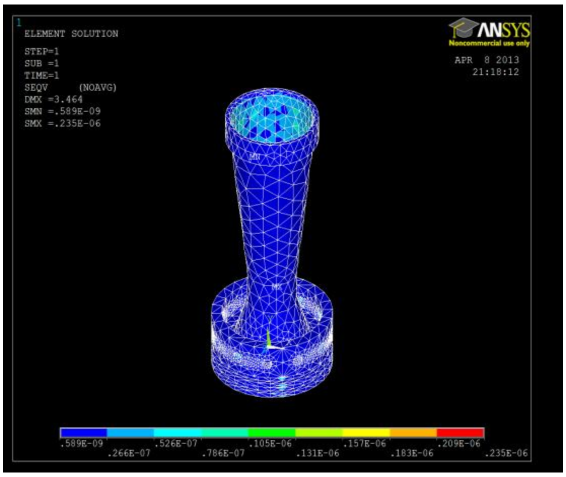

Fig.14 Elastic Deformation

IV. CONCLUSION

Theoretical analysis of a small solid propellant rocket motor is conducted using Ansys to know the propellant performance over a wide range of oxidizer/fuel ratio density. Significant improvement of the motor performance appeared likely to be achieved by variation of nozzle design.

Result demonstrates 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.

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