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SOLID PROPELLANT GRAIN CONFIGURATION WITH PROGRESSIVE BURNING OBTAINED THROUGH ADDITIVE MANUFACTURING

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Abstract: *In this paper, a solid propellant grain concept was defined, obtained through additive manufacturing, which combines characteristics of multilayer elements with those of tubular geometric shape. The internal ballistics model adapted for such a solid propellant grain configuration allowed the calculations and highlighted the possibility of obtaining a pressure-time curve with a slightly increasing variation (progressive burning). The case studies carried out had as their starting point the existing physical configuration of the solid rocket motor for S-5K unguided ammunition. For the latter, PRTF-100 double base propellant was replaced by a composite propellant consisting of 87.3 wt% ammonium dinitramide, 10.7 wt% polycaprolactone and 2 wt% copper oxide.*

Keywords: solid rocket motor, propellant grain, unsteady lumped internal ballistic model, additive manufacturing, FDM

1. Introduction

The solid rocket motors have recently become widespread. From published data, it appears that more than 90% of existing missiles/artillery rockets are equipped with solid rocket motors.

The solid rocket motors are used in all current classes of combat mission missiles/ artillery rockets. Also, vectors equipped with a solid rocket motor are used in the national economy for the following purposes:

- combating hail (antihail rockets);
- well drilling;
- testing the upper structures of the atmosphere.

According to the functions performed, the solid rocket motors can be divided into three large groups:

- solid rocket motors that develop the thrust force necessary to move the missiles/artillery rockets;
- solid rocket motors that create moments and command forces;
- solid rocket motors with experimental and technological purposes.

The main advantages of the solid rocket motors include its simplicity, and the ability to deliver very high thrust levels without significant design compromises. The solid rocket motors offer the maximum volumetric impulse of any chemical propulsion system.

For those, the solid propellant grain is cast directly into combustion chamber or loaded as a free-standing cartridge inside the afferent combustion chamber. Traditional manufacturing techniques for cast grain can produce pressure gradients and therefore

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density variations, leading to ballistic performance decrease.
The new technology with applicability in the field of energetic materials is Additive Manufacturing, where 3D printing methods are used to obtain a solid propellant grain both costs effectively and controllable ballistic properties.
Based on the information available at this time, we assume that a solid propellant grain introduced freely into the combustion chamber of the rocket motor could be obtained through additive manufacturing.
Direct printing of the solid propellant grains with Fused Deposition Modeling (FDM) technology has also been reported [1], [2], [3]. However, in this moment FDM technology is rarely applied in obtaining of the solid propellant grains because this 3D printing process involves heat extrusion of filling a thermoplastic polymer filament with solid high energetic particles.
The main advantage of FDM technology is the simplicity and economy of the manufacturing process in comparation with other 3D printing technologies. The main drawback of FDM technology is the limited accuracy and the safety hazard caused by the rapid filament heating during 3D printing activity.

2. Solid Propellant Grain Configuration

Depending on the form of the combustion surface, the burning evolution of the solid propellant grain can be considered as having a neutral, progressive or regressive character.
The law of modification for the burning surface is established from the geometry of the solid propellant grain and the bonding of certain surfaces.
The most commonly used solid propellant grain configuration with neutral burning that could be introduced into the combustion chamber of the solid rocket motor have the tubular geometric shape (Figure 1).

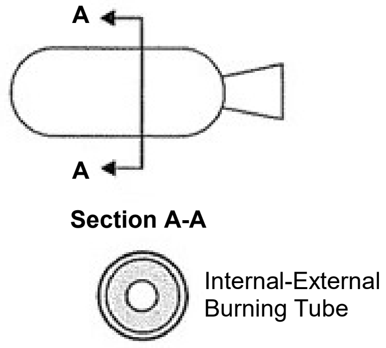

Figure 1: Solid propellant tubular grain cross-sectional geometry

In the case of multilayer solid propellant grains, they are made of at least two distinct components which may differ in ballistic properties, particularly in terms of burning rate.
In such a solid propellant grain the outer layer, exposed to ignition, is made of component that burn more slowly and the component that burn faster is placed inside.
When the transition from on layer to other takes place, even if it is not mandatory to increase the burning surface, by increasing the burning rate, the gas flow rate also increases, so the effect is quasi-similar to the increase in the burning surface.

3. Unsteady Lumped Internal Ballistic Model

Calculation of gas pressure p_0 inside the combustion chamber of the solid rocket motor in transient regime is based on the continuity equation – the mass flow rate emitted by the propellant burning surface is equal to the sum of the mass flow rate of the burned gases accumulated in the solid rocket motor and the mass flow rate which the nozzle throat can eject [4]:

$$\frac{dp_0}{dt} = \frac{RT_0}{V_l}\left[A_a u_a\left(\rho_{prop} - \frac{p_0}{RT_0}\right) - C_D A_{cr} p_0\right] \quad (1)$$

where, R is the gas constant for the combustion products mixture, T_0 is the combustion gas absolute temperature, V_l is the combustion chamber cavity volume, A_a is the propellant burning surface area, u_a is the propellant burning rate, ρ_{prop} is the propellant density, $C_D = \sqrt{\frac{\gamma}{RT_0}}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma-1)}}$ is the mass flow coefficient, γ is the ratio of specific heats for the combustion gas mixture, and A_{cr} is the nozzle throat area.

The burnt thickness e, define the distance to which the propellant burning surface has moved relative to the original unburned propellant surface location. The dependence formula for the burnt thickness with the burning rate law is presented below:

$$\frac{de}{dt} = u_a . \qquad (2)$$

The augmentation of the burning rate u_a due to the erosive cause has been observed experimentally for port passage cross-sectional area (A_p) to throat area (A_{cr}) ratio of 4 or less [4].

In the modified Lenoir and Robillard model [5], total burning rate consisting of components without erosion influence, $u_{a\,0}$, and $u_{a\,eroz}$, the contribution of the erosion effects:

$$u_a = u_{a\,0} + u_{a\,eroz} \qquad (3)$$

where,

$$u_{a\,eroz} = \alpha\left(\rho_{prop}\, u_a \,\frac{A_a}{A_p}\right)^{0,8} L^{-0,2}\, e^{-\beta\frac{A_p}{A_a}}$$

define the erosive burning contribution, L is the propellant grain characteristic length, and β is empirically determined constant whose value is about 270 for a nitropolymer propellant [6].

The expression of the heat transfer coefficient α is determined based on analogy Chilton-Colburn:

$$\alpha = \frac{0,0288\, c_p\, \mu^{0.2}\, \mathrm{Pr}^{-2/3}}{\rho_{prop}\, c_{prop}} \frac{T_0 - T_s}{T_s - T_0} \qquad (4)$$

where, c_p is the specific heat at constant pressure, μ is the dynamic viscosity, Pr is the Prandtl number, all for the combustion gas mixture, c_{prop} is the propellant specific heat capacity, and T_s is the propellant burning surface temperature.

4. Numerical Results

The stoichiometric composition of composite propellant which could be used for additive manufacturing consisted of 87.3 wt% ammonium dinitramide (ADN), 10.7 wt% polycaprolactone (PCL) and 2 wt% copper oxide (CuO) [7], [8].

Two types of PCL with different molar masses: 1250 g/mole and 10000 g/mole were used [7], [8].

For this composite propellant, the relation between base burning rate, expressed in mm/s, and combustion chamber pressure, expressed in MPa, is shown below [7], [8]:

- $u_{a\,0} = 15.3\, p_0^{0.44}$, if the binder is PCL with a molar mass of 1250 g/mole;
- $u_{a\,0} = 9.1\, p_0^{0.6}$, if the binder is PCL with a molar mass of 10000 g/mole.

The CEA thermochemical results, with frozen reactions, obtained for combustion chamber pressure of 10 MPa, needful for the numerical integration of internal ballistic model, are the following:

- T_0 = 2982.29 K;
- R = 332.16 J/(kg-K);
- γ = 1.2066;
- c_p = 1940.1 J/(kg-K);
- μ = 0.98788×10^{-4} N-s/m^2;
- Pr = 0.6969.

In this first scenario, the nozzle of the S-5K solid rocket motor does not require any constructive modification; instead, due to the fact that the burn rate value of the hypothetical generic composite propellant obtained through additive manufacturing is about four time that of existing PRTF-100 propellant from the S-5K solid rocket motor, an 80% reduction for tubular propellant grain was achieved (Figure 2).

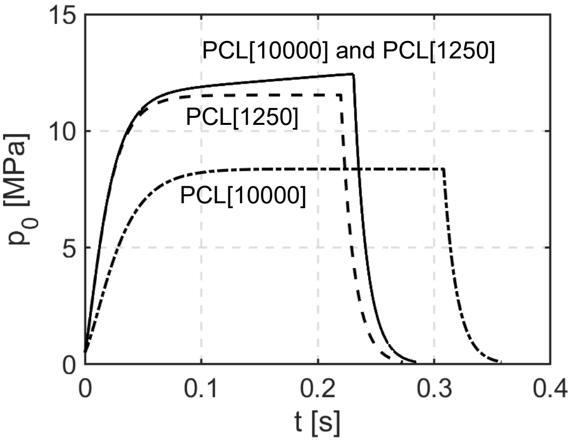

Figure 2: Combustion chamber pressure versus time graph, in the first scenario

The outer layer of the tubular grain is made of the composite propellant that burns more slowly (the binder used, PCL, has a molar mass of 10000 g/mole), and the inside is the composite propellant that burns faster (the binder used, PCL, has a molar mass of 1250 g/mole).

In the second scenario, the dimensional shape of the existing tubular propellant grain is maintained; instead, due to the fact that the burn rate value of the hypothetical generic composite propellant obtained through additive manufacturing is about four time that of existing PRTF-100 propellant from the S-5K solid rocket motor, a 2.5-fold increase in the diameter for nozzle throat was achieved (Figure 3).

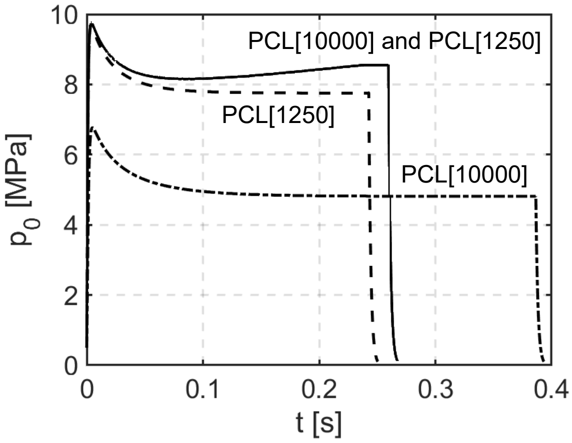

Figure 3: Combustion chamber pressure versus time graph, in the second scenario

6. Conclusions

Even though a single solid propellant rocket motor was used in this case study, it seems reasonable to extend the conclusions to all propulsion systems, regardless of size or other design data, set within technically reasonable limits.

Thus, considering the possibilities of the additive manufacturing technology, the combination of the characteristics of the multilayer elements with those of the tubular propellant charges was taken into account in order to obtain a slightly increasing variation for the pressure-time curve.

Acknowledgements

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