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Estimation of Pressure Index and Temperature Sensitivity Coefficient of Solid Rocket Propellants by Static Evaluation

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ABSTRACT

Burning rate of a solid rocket propellant depends on pressure and temperature. Conventional strand burner and Crawford bomb test on propellant strands was conducted to assess these dependent parameters. However, behaviour of propellant in rocket motor is different from its behaviour in strand form. To overcome this anomaly, data from static evaluation of rocket motor was directly used for assessment of these burning-rate controlling parameters. The conventional empirical power law ($r=a_o\exp[\pi\{T-T_o\}]P^n$) was considered and a method was evolved for determination of pressure index (n) and temperature sensitivity coefficient (π) of burning rate for solid rocket propellants from static evaluation data. Effect of pressure index and temperature sensitivity coefficient on firing curve is also depicted. Propellant grain was fired in progressive mode to cover a very wide pressure range of 50 kg/cm^2 to 250 kg/cm^2 and propellant burning rate index was calculated to be 0.32 in the given pressure range. Propellant grain was fired at +35 °C and –20 °C temperatures and temperature sensitivity coefficient of burning rate was calculated to be 0.27 % per °C. Since both the values were evaluated from realised static evaluation curves, these are more realistic and accurate compared to data generated by conventional methods.

Keywords: Solid rocket propellant, internal ballistics, pressure index, temperature sensitivity, chamber pressure, burning rate, static evaluation.

NOMENCLATURE

a	Burning rate coefficient
a_o	Standard burning rate coefficient
A_t	Throat area of rocket motor
C^*	Characteristic velocity of solid propellant
d_t	Throat diameter
m_d	Rate of mass discharge from nozzle
m_g	Rate of mass generation by burning of propellant
n	Pressure index of burning rate
P	Chamber pressure
r	Burning rate
S	Burning surface area of solid propellant
T	Firing temperature
T_o	Standard temperature for determination of a_o
w	Web burnt
ρ	Density of solid rocket propellant
π	Temperature sensitivity coefficient for propellant

1. INTRODUCTION

Solid rocket propellants are the power behind propulsion of all modern missiles and most of the launch vehicles. Rocket propellant is fired on ground in a rocket motor with known throat diameter for realisation of pre-specified pressure inside rocket motor chamber. The performance prediction requires internal ballistic calculations, and in most of the cases, mass conservation is imposed to get various ballistic parameters. The applicability of empirical power law ($r = a\,x\,P^n$) for relating burning rate with generated pressure has two important performance parameters namely pressure index (n) and burning rate coefficient (a). From this relation higher burning rate is obtained for higher pressure in rocket motor chamber. Since burning rate increases with rise in temperature, burning rate coefficient is assumed to be dependent on temperature also. The standard correlation, using temperature sensitivity coefficient is given as $a = a_o \exp[\pi(T - T_o)]$. These two parameters, n and π are conceived for ease of internal ballistic calculations, but have got status of a physical quantity in rocketry. Present paper elaborates a method for the determination of pressure index and temperature sensitivity coefficient of burning rate for solid rocket propellants using firing curve (pressure-time profile) from static evaluation, directly. This is more accurate, realistic, and gives relevant value compared to data obtained by conventional strand burner or other methods.

2. PRIOR STATE-OF-THE-ART

Propellant burning front propagation rate is expressed as an empirical function dependent on propellant composition, propellant conditioning, combustion-chamber environment, and motor dynamics. Grain temperature, local static pressure, and mass flow significantly influence local burning rate. In the empirical power law ($r = a \times P^n$), if pressure index is constant, the power law is known as de Saint Roberts

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law[1]. However, this is rarely pressure-independent and most of the time; value of n is different in different pressure ranges. A similar explanation of power law is advocated by other contemporary and subsequent literature[2-8]. The references dealing with influence of pressure on burning rate of solid rocket propellant do indicate variation of burning rate with temperature. Sutton[6] indicated that temperature sensitivity coefficient is dependent on pressure also. Temperature sensitivity coefficient is conceived in various forms[9-11].

Pressure index and temperature sensitivity coefficient are essentially empirical quantities for ease of performance prediction. Solid propellant strand was fired in Crawford bomb or strand burner at different pressures and pressure index was calculated by curve fitting. Once pressure index is available for a propellant in a pressure range, value of burning rate coefficient (a) is calculated from burning rate, pressure and calculated n value numerically using ($a = r/P^n$). If propellant samples are evaluated at different temperatures, determination of temperature sensitivity coefficient is also possible in the same set-up.

However, the values of pressure index and temperature sensitivity coefficient, realised through evaluation of strands is seldom obtained in actual rocket firing and performance prediction, based on strand burner data does not give performance matching to prediction. In his paper, an attempt has been made to predict pressure index and temperature sensitivity coefficient of burning rate directly from static firing, so that all propellant variables, including motor parameters, are taken into account and performance matching to prediction in the static firing is realised.

In addition to this, flow velocity of gases, pressure variation in port, acceleration of combustion gases, temperature non-uniformity in port, erosive burning, etc are also considered important influencing parameters for burning rate of solid rocket propellants. But for small grains, these effects are negligible and these parameters can be calculated from pressure-time curves, obtained from static evaluation of propellants. Deliberately, a smaller-length propellant was considered, where negligible erosive burning, almost zero longitudinal variation of flow-velocity, and pressure were observed. The constant of performance prediction obtained directly from static firing is more realistic than data obtained from other types of tests.

3. INTERNAL BALLISTICS CALCULATIONS AND APPROACH

Internal ballistic calculation for most of the solid propellant evaluation starts with prediction of pressure-time curve from propellant and rocket motor (including nozzle) parameters. Propellant density ρ and characteristic velocity C^* are thermodynamic parameters. Burning surface area S of propellant is geometric parameter and as web is consumed during combustion, burning surface area at various web burnt can be calculated from geometry of propellant grains. So for a given geometry, burning surface area S with web consumed Δw is available before hand. Throat area A_t is a rocket motor parameter, which is measured before firing.

Burning rate of solid propellant is given by Eqn (1)

$$r = a \times P^n \quad (1)$$

It is an empirical relation, showing dependence of burning rate on pressure in rocket motor chamber during motor operation and is used for performance prediction of rocket motors. Burning rate (r) and pressure in rocket motor chamber (P) are measurable parameters, but a and n are empirical constants, which vary with propellant and operating conditions. Burning rate of solid propellant is also dependent on temperature and burning rate coefficient takes care of this variation using relation shown as Eqn (2).

$$a = a_o \times \exp\left[\pi \times (T - T_o)\right] \quad (2)$$

The prediction of pressure is based on mass conservation, where rate of mass generation by burning propellant $m_g = r.S.\rho$ and rate of mass discharged through nozzle $m_d = P.A_t.g/C^*$ are equated to give equilibrium pressure as Eqn (3).

$$m_d = P \times A_t \times g / C^* = m_g = r \times S \times \rho = a \times P^n \times S \times \rho$$

$$P = \left[a \times S \times \rho \times C^* / A_t \times g\right]^{1/(1-n)} \quad (3)$$

The generation of pressure-time profile takes pressure from Eqn (3) and for calculation of time web increment is adopted. At each web consumed, burning rate can be obtained using Eqn (1), as pressure at that web can be obtained from instantaneous burning area, using Eqn (3). For each increment of web consumed, time taken was calculated as web burnt divided by instantaneous burning rate. The correlation between web burnt and time step is given by Eqn (4) as

$$\Delta t = \Delta w / \{a \times [a \times S \times \rho \times C^*/A_t \times g]^{n/(1-n)}\} \quad (4)$$

For calculation of complete pressure-time profile, initial web burnt and time was taken as zero. Pressure was calculated from available initial burning surface area using Eqn (3). For next web, burning surface area was available from geometrical consideration and which gave instantaneous pressure and subsequently instantaneous burning rate. Time take to consume this web was calculated using Eqn (4). So at new web, both pressure and time were available. This process was repeated till complete propellant was consumed to get pressure-time for the entire propellant.

Burning rate at a given pressure and temperature can be evaluated directly by various methods, but there is no direct method for determination of n and π. To evaluate n for a wider pressure range, propellant grain with wide variation of burning area and subsequently pressure levels during propellant combustion is necessary. A highly progressive burning profile was considered and performance prediction was carried out for different n values using Eqn (3) and Eqn (4). The solid propellant grain was statically evaluated in rocket motor and pressure-time profile was recorded. The n value, whose prediction matches with actual firing curve is n value of propellant in the given pressure range. This firing was conducted at +35 °C. Propellant grain in the same configuration and composition was soaked at low temperature (–20 °C) and statically evaluated. Low temperature gives lower pressure and burning rate and higher burn-time. With earlier calculated *n* value, plot was drawn for different temperature sensitivity coefficients and matching profile gives temperature sensitivity coefficient of burning rate directly for the given solid propellant composition.

4. CASE STUDY

A typical multi-perforated solid propellant grain has been designed to give a very high variation of burning area during combustion. It has seven holes, one central and three equidistant holes at two different pitch circle diameters. The burning area versus web burnt profile is given in Fig. 1.

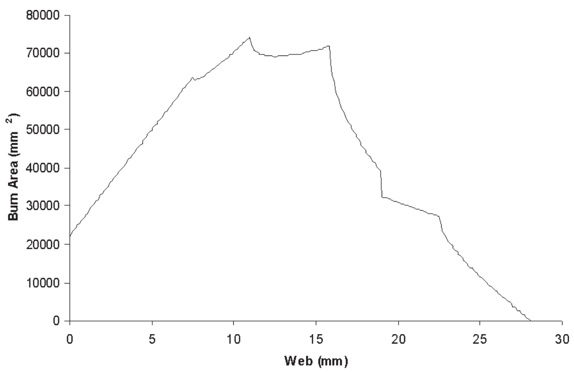

Figure 1. Predicted burning area versus web burnt profile.

The initial burning area of 222 cm^2 reaches maximum burning area of 741 cm^2, giving a progressivity ratio of burning area as 3.33 (741/222). The propellant and rocket motor parameters for further calculations are given in Table 1.

Table 1. Salient input parameters for performance prediction

Parameters	Values
Density (ρ)(g/cc)	1.65
Characteristic velocity (C^*) (m/s)	1480
Throat diameter (d_t) (mm)	17.5
Burning rate at 70 kg/cm^2 (r_{70}) (mm/s)	19.3

The configuration was fired at two temperatures, viz., +35°C and –20 °C and superimposed pressure-time profile for both the firings is given in Fig. 2. For nominal temperature of +35 °C, pressure realised varies from 50 kg/cm^2 to 250 kg/cm^2 (range around 200 kg/cm^2) and burn time was around 1.1 s. For low temperature (–20 °C) firing, pressure range reduced to around 170 kg/cm^2 (From 37 kg/cm^2 to 206 kg/cm^2) and burning time increased to 1.315 s. The curve extracted in Fig. 2 is for a burning time of 1.2 s for both the firing temperatures.

Performance prediction is done for pressure index value (*n*) ranging from 0.2 to 0.5 at an interval of 0.1. The propellant grain was statically evaluated in rocket

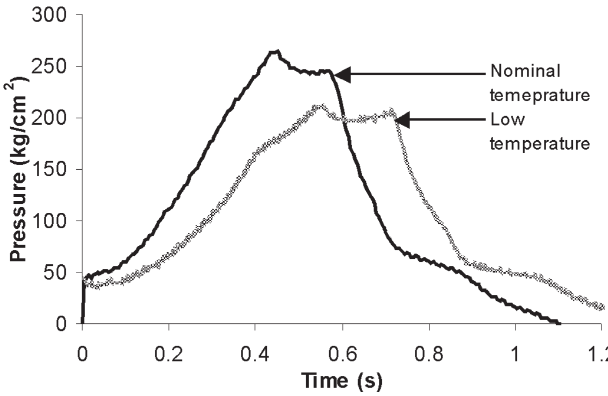

Figure 2. Static evaluation curves at different temperatures.

motor at +35 °C and pressure time profile recorded is superimposed over prediction (Fig. 3). As *n* increases, value of pressure increases and total burning time reduces. By repetitive iteration for n, predicted pressure-time profile matching to actual firing curve was obtained for the pressure index value of 0.32. This value of pressure index (*n*=0.32) for burning rate of solid propellant is valid for pressure range from 50 kg/cm^2 to 250 kg/cm^2.

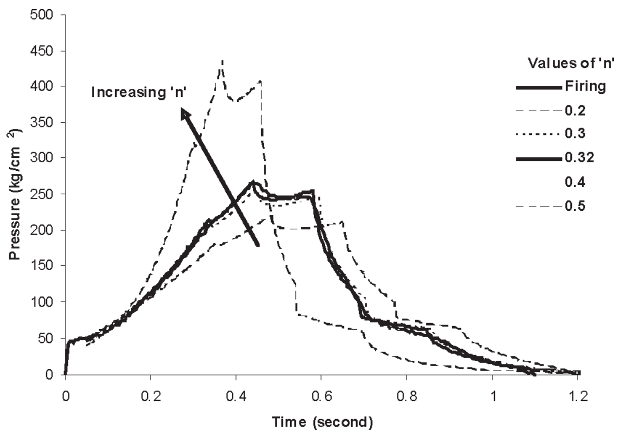

Figure 3. Determination of pressure index from firing curve.

To evaluate temperature sensitivity coefficient, –20 °C was considered adequate to make derived data suitable for a wider temperature range. Performance prediction with calculated pressure index value (*n* = 0.32) was carried out with different temperature sensitivity coefficients. Temperature sensitivity coefficient values were changed from 0.1 per cent per °C to 0.3 per cent per °C in steps of 0.1 per cent per °C. Since firing was conducted at low temperature (–20 °C) compared to reference temperature (+35°C), lower temperature sensitivity coefficient (π) gives higher pressures and lower burning time. This is depicted in Fig. 4.

If static evaluation is conducted at higher than reference temperature, higher π gives higher pressures and lower burning time. Firing data for static evaluation at low temperature (–20 °C) is also superimposed in Fig. 4. It is clear that temperature sensitivity coefficient of 0.27 per cent per °C gives matching curve and can be taken as temperature sensitivity coefficient for the propellant in the tested

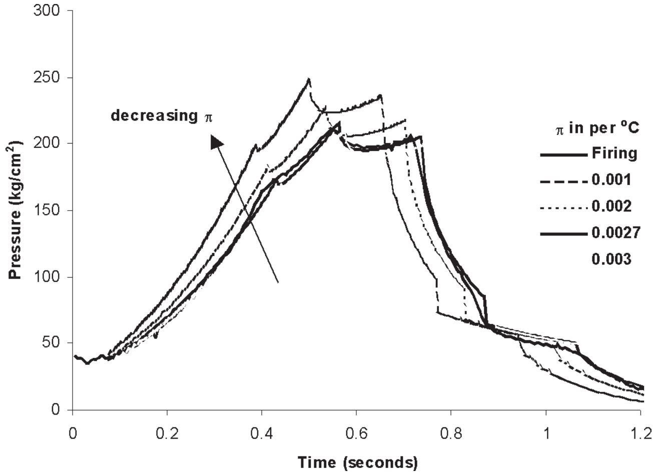

Figure 4. Determination of temperature sensitivity coefficient from firing curve.

temperature range.

5. CONCLUSIONS

Firing curve from static evaluation is used for practical determination of pressure index and temperature sensitivity coefficient for solid rocket propellants. Firing propellant grain in progressive mode gives pressure index value for a wide pressure range and for the case study depicted; pressure index value of 0.32 is valid for pressure range of 50 kg/cm^2 to 250 kg/cm^2. Temperature sensitivity coefficient can also be predicted with better accuracy by static firing. In the case study, temperature range of –20 °C to +35 °C is considered and temperature sensitivity coefficient is obtained as 0.27 per cent per °C. The method does not need any separate arrangement like strand burner, Crawford bomb, etc for such measurements. The approach is less calculation-intensive and more realistic in predicting pressure index and temperature sensitivity coefficient. The approach gives more accurate, realistic, and predictable pressure-time profiles. Values of determined constants directly from static firing ensure negligible variation from actual results in rocket operation.

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