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IGNITION TRANSIENT ANALYSIS OF SOLID ROCKET MOTOR

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ABSTRACT

To predict pressure-time and thrust-time behavior of solid rocket motors, a one-dimensional numerical model is developed.The ignition phase of solid rocket motors (time less than 0.4 sec) depends critically on complex interactions among many elements, such as rocket geometry, heat and mass transfer, flow development, and chemical reactions. The present model solves the mass, momentum, and energy equations governing the transfer processes in the rocket chamber as well as the attached converging-diverging nozzle. A qualitative agreement with the SRM test data in terms of head-end pressure gradient and the total thrust build-up is obtained. Numerical results show that the burning rate in the star-segmented head-end section and the erosive burning are two important parameters in the ignition transient of solid rocket motor (SRM).

INTRODUCTION

Fig. 1 shows a schematic diagram of space shuttle solid rocket motor. As the pyrogen gas from the igniter heats the surface of a solid propellent in the star-segmented section, temperature of the propellent increases until its surface temperature reaches an ignition temperature.Energy released through a chemical reaction initiates a strong convection in the chamber and the flame spreading toward the aft-end of the rocket chamber.Following the completion of flame spreading, the gas pressure in the chamber increases rapidly due to the energy and mass released from the burned solid propellent. Ignition transient (or starting transient) refers to a short period of motor operation from the ignition of igniter gas to the rapid chamber pressure build-up.

An ability to predict and control the complex transport processes in the rocket chamber is desired for an efficient design of rocket motors. Engineering design parameters include: overall transient time of a motor; reduction of motor- to-motor variation; over-pressurization; avoiding misfires and hangfires; effects of design modification such as propellent formulation, and the geometry of a rocket; variation of total thrust and the dynamic thrust.

Peretz, et. al. [1] performed experimental and analytical modeling of high velocity transient of solid rocket motors. Caveny and Kuo [2] extended their analysis to space shuttle SRM. Caveny further extended their method and compared numerical predictions with space shuttle SRM test data [3]. Total thrust build-up predicted by Caveny-Kuo's model agrees qualitatively with the test data. However, predicted transient pressure in the head-end of the motor shows an extreme departure.

The purpose of the present investigation is to analyze ignition transient of space shuttle SRM. Anew 1-dimensional numerical model is developed and numerical results are compared with the test data.

ANALYSIS

Assumptions

It is assumed that the combustion products are ideal gas of single component without the radiation effects. Chemical reaction occurs in an imaginary this surface instantaneously when the surface temperature of the propellent reaches auto- ignition temperature. All transport coefficients remain constants. Flow and temperature fields are assumed to be one- dimensional and interactions between the gas and the solid surface are adequately represented by empirical momentum and energy transfer correlations.

Equations for Gas

Mass conservation is given by

$$\frac{\partial \rho a}{\partial t} + \frac{\partial}{\partial x}(\rho u A) = \rho_{pr} r b + \dot{m}_{ig}, \tag{1}$$

where ρ is the density, A is the cross-sectional area, u axial velocity, ρ_{pr} the solid propellent density, r is the burning rate, b is the burning perimeter and m_{ig} is the rate of mass generation. All variables are functions of time (t) and axial distance (x). The Lenoir-Robillard burning rate law is employed to evaluate the regression rate of solid propellent. The conventional form of this expression is [3]

$$r = r_{ref}\ (P/P_{ref})^n + \alpha_e G^{0.8} Ph^{-0.2}\ \exp[-\beta_e r \zeta_{pr}/G]. \tag{2}$$

The first term accounts for a standard burning and the second for an erosive burning. Erosive burning increases as the mass flow rate (G) of the gas increases. D_h is the hydraulic diameter, α_e and β_e are adjustable coefficients.

The axial momentum equation is

$$\frac{\partial}{\partial t}(\rho u A) + \frac{\partial}{\partial x}[A(\rho u u - (2u + \lambda)\frac{\partial u}{\partial x})] = -\frac{\partial AP}{\partial x} + p\frac{\partial A}{\partial x} - \gamma_w P_w + \psi, \tag{3}$$

where γ_w is the wall shear stress and is determined by using a friction factor correlation.

The internal energy equation is

$$\frac{\partial}{\partial t}(\rho e A) + \frac{\partial}{\partial x}[A(\rho u e - \frac{k}{c_v}\frac{\partial e}{\partial x})] = -p\frac{\partial uA}{\partial x} + u\gamma_w P_w + \Phi + \frac{1}{2}\rho_{pr} u^2 rb$$
$$- q_{AS} P_w + \rho_{pr} brh_f + \dot{m}_{ig} h_{ig}, \tag{4}$$

where Φ is the dissipation, q_{AS} is the amount of heat transfer and h_f is the enthalpy of gaseous solid propellent and h_{ig} is the enthalpy of igniter gas. The convective heat transfer to the solid is determined by using an empirical convection coefficient.

Initially the propellent and the gas in the chamber are at a thermal equilibrium with the surrounding atmosphere. The left side (x=0) of the chamber is a solid mass and the right side (x=L) is the nozzle exit. Atmospheric conditions are imposed at the nozzle exit when the flow is subsonic at the exit and a linear extrapolation is used when the flow is supersonic.

Equation for Solid

Fig. 2 shows a schematic temperature profile in the solid propellent and a solid wall. One dimensional conduction through the solid propellent and the solid wall is governed by

$$q_{PR} c_{PR} \frac{\partial T_{pr}}{\partial t} = \frac{1}{r} \frac{\partial}{\partial r} \left(r k_{pr} \frac{\partial T_{pr}}{\partial r} \right). \tag{5}$$

Convective boundary conditions are used at the interface between the gas and the solid and adiabatic conditions are assumed at the other surface.

Numerical Method

A modified version of SIMPLE method [4] is used to integrate governing equations iteratively. A staggered non-uniform grid system is used. This numerical method is highly versatile and modular and is proved to be a good numerical method for combustion problems.

RESULTS

Geometry of SRM

Physical geometry of a RSRM, including grain configuration and the attached nozzle, is selected. Total length of the propellent section is approximately 1320 inches and the nozzle is 178 inches. Radius of the propellent cross-section varies from 28 inches at the head-end to 42 inches at the aft-end of combustion chamber. Radius at the throat of the nozzle is 27 inches and 75 inches at the nozzle exit.

Grain geometry at the head-end section is star-segmented as shown in Fig.3. To account properly for the burning surface, a burning perimeter ratio, α , is defined as

$$\alpha = \frac{\text{actual burning surface perimeter}}{\text{circumference of the cylinder}}$$

where $0 \leq \alpha \leq 6.70$.

Igniter pyrogen gas mass flow rate is shown in Fig.4. The Autoignition temperature of the solid propellent is assumed to be 850 K. Adiabatic flame temperature of the pyrogen gas is 2450 K and solid propellent is 3361 K. All other values of physical properties are those used in ref. 2. Total number of control volumes are 60; 20 for the chamber, and 40 for the converging-diverging nozzle.

Comparison with Test Data

Fig. 5 shows a comparison of two numerical results with test data in terms of total thrust and the head-end pressure variations. In numerical results, total thrust is divided into dynamic thrust (Fdy) and nozzle thrust (Fnoz). Dynamic thrust is caused by the change in momentum within the rocket and the nozzle thrust is due to the momentum efflux at the exit. Pressure variation in the chamber during an ignition transient is closely related to the dynamic thrust.

Numerical results show that total thrust can not reach the measured value without the erosive burning effects and a careful adjustment of burning perimeter ratio, α , is needed to match the head-end pressure variations.

CONCLUSIONS AND RECOMMENDATIONS

One-dimensional numerical model is developed to analyze ignition transients of space shuttle SRM. Ideal compressible fluid equations are solved by a variant of SIMPLE method. Comparison with test data shows a qualitative agreement. Two most dominant factors influencing ignition transients are the erosive burning rate and the burning pattern in the star-segment head-end propellent. With an accurate data for the erosive burning and head-end burning pattern, one-dimensional model may be used for parametric study of ignition transients of SRM.

Multi-dimensional analysis is needed to account for more accurate burning pattern in the star section, igniter gas expansion, complex flow in the chamber and through a converging-diverging nozzle. Erosive burning rate under highly transient dynamic conditions requires more elaborate treatments on the chemical reaction as well as flow field at the surface of solid propellent.

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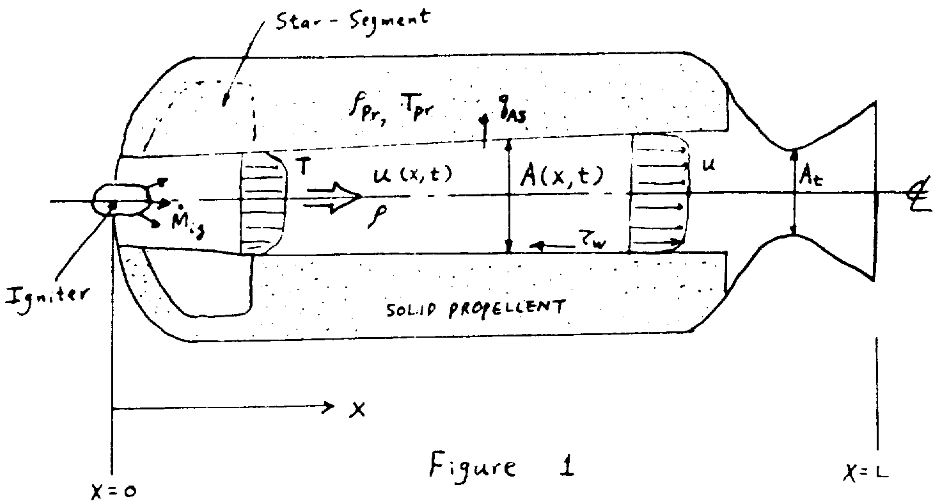

Figure 1

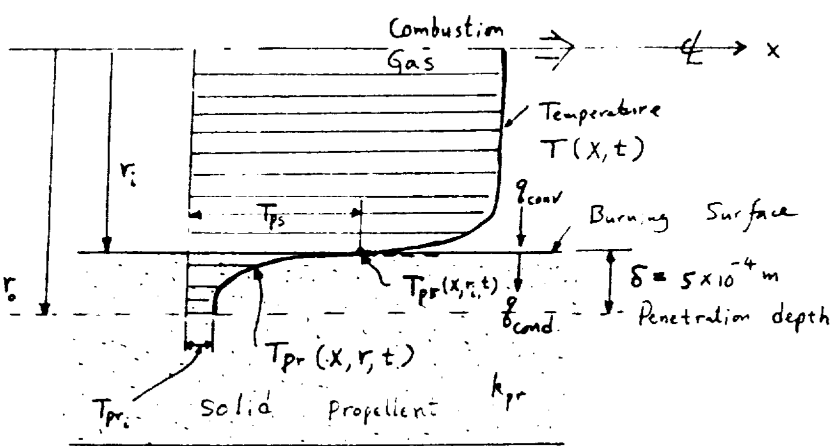

Figure 2

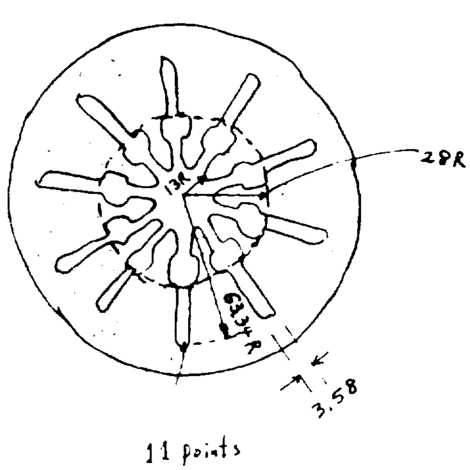

Figure 3

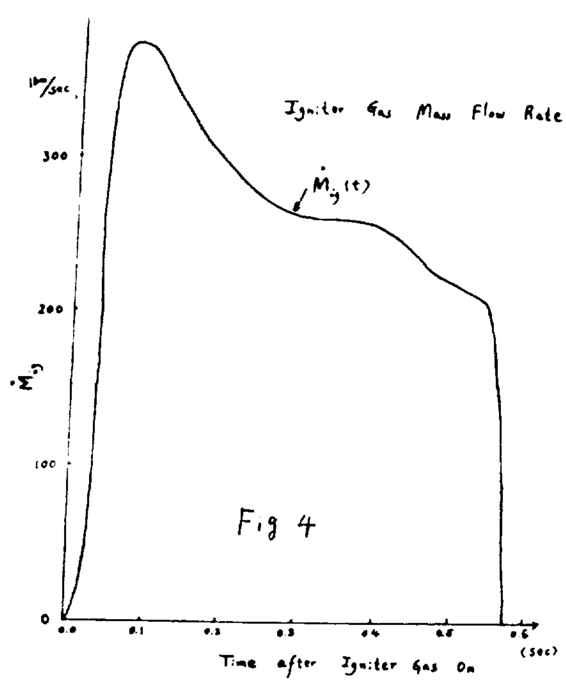

Fig 4

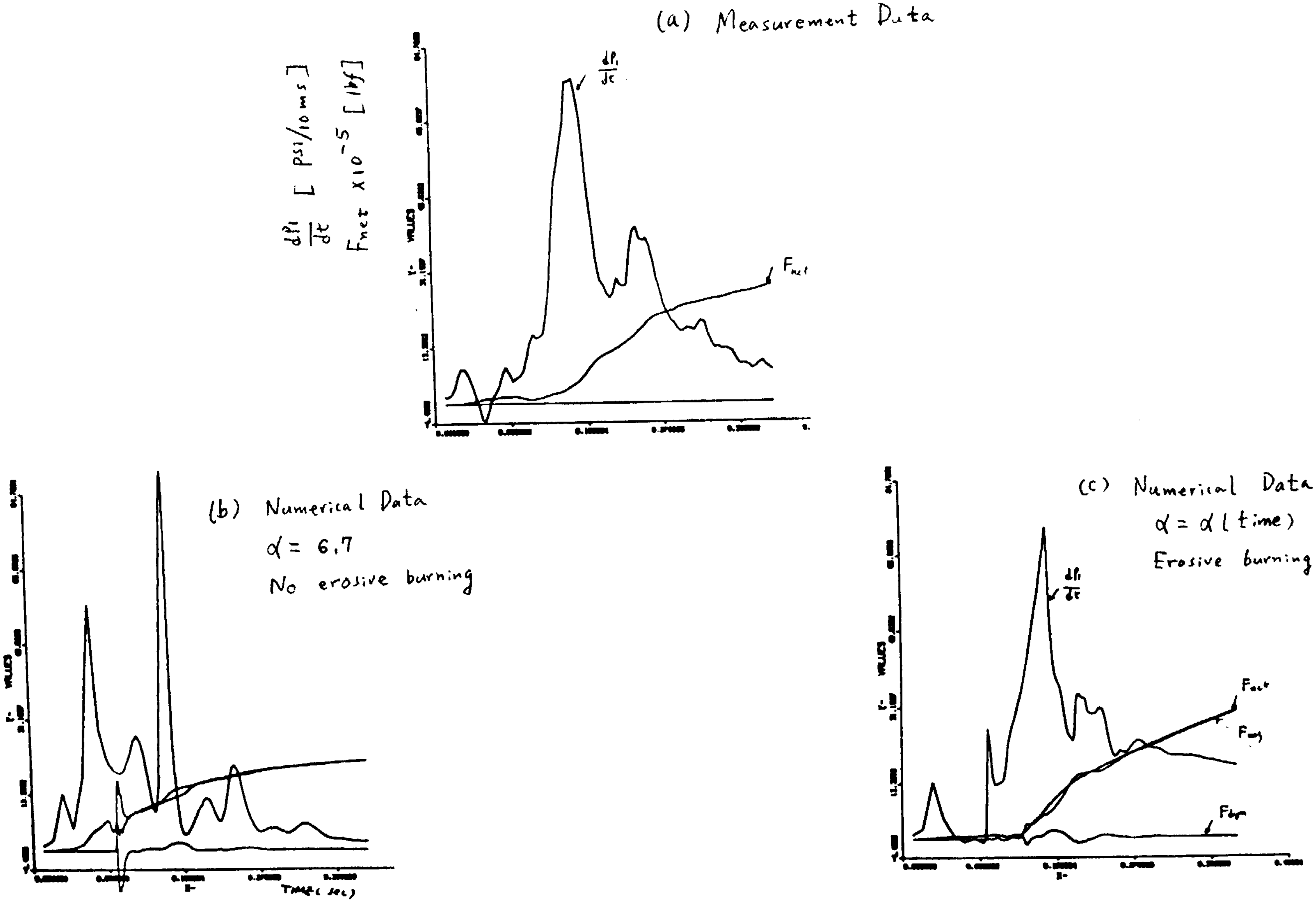
(a) Measurement Data
$\frac{dP_1}{dt}$ [psi/10 ms]
F_{net} x10^{-5} [lbf]
$\frac{dP_1}{dt}$
F_{net}
(b) Numerical Data
α = 6.7
No erosive burning
TIME (sec)
(c) Numerical Data
$\alpha = \alpha$ (time)
Erosive burning
$\frac{dP_1}{dt}$

Figure 5