Technical Notes

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Grain Deformation of Gas Flow Passages in Solid Rocket Motors

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Introduction

HIGH-performance solid rocket motors often contain so much propellant that it becomes difficult to provide adequate passages for the combustion gases. Opening such flow passages is usually achieved by removing propellant, which degrades performance. Restricting these passages below a safe minimum, however, results in high-pressure gradients and structural failures. Therefore, a design compromise must be made between performance and a safety margin regarding gas passage sizing. This problem is complicated by the many time-dependent phenomena involved, such as ignition flame spreading, a propellant grain shape that changes as it is consumed, and the viscoelastic mechanical response of typical propellants. Additionally, the operating temperature of the rocket motor has a strong influence on the problem because the combustion process, propellant stiffness, and dimensions of the propellant grain all undergo significant changes over the operational temperature range of typical motors.

It is possible to solve these nonlinear time-dependent problems in great detail with currently available computational equipment,[1] but the process is time consuming—with the consequence that it is seldom done during the early stages of the design process. Instead, past experience with similar motors is often the only guide employed. Unfortunately, catastrophic motor failures continue to occur due to inadequate combustion gas passages in the initial design. Once such a problem is encountered, the appropriate calculations are made and the problem is resolved. In many of these problems, it is not necessary to perform elaborate calculations to achieve a near-optimum gap size, but this aspect is not always recognized at the outset. In view of this situation, a simplified approach has been constructed to facilitate the early design process. Application of the method permits a visualization of the influence that the various design parameters have on the flow passage requirements and, in many cases, leads to a rational safety factor definition for this aspect of rocket motor design.

Problem Definition

The various mechanisms influencing gas passage sizing exist to some degree in all solid-propellant rocket motors, but the criticality of this design attribute is most evident in a submerged-nozzle design. A specific hypothetical submerged-nozzle rocket motor (Fig. 1) illustrates both the problem and the associated design methodology.

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The ends of the propellant grain are insulated to prevent combustion on those surfaces not attached to the motor case. The purpose of the unbonded ends is to reduce the strains in the bore of the propellant grain. The inner bore of the propellant grain, including the surface behind the surmerged nozzle (surface A in Fig. 1), is the initial burn surface. Combustion gases are generated on this burn surface and flow out the nozzle in the manner depicted in Fig. 1. This gas flow is accompanied by a pressure field with associated gradients that drive the flow. The forward and aft domes are subjected to constant static pressure of a magnitude greater than that at the nozzle, as depicted in Fig. 1. Of particular concern is the pressure differential ΔP that exists between the aft done and the entrance of the nozzle. This pressure differential causes the propellant grain deformations to close the gap between the grain and the submerged nozzle, as shown in Fig. 1. Closure of this gas flow gap ΔG results in higher flow velocities and a greater ΔP. Combustion gas generation increases with the pressure, further increasing the gas flow. The higher-pressure gradient produces more deformation. In some instance, these mechanisms can result in complete closure of the gas flow passage or a ΔP so great that the motor case fails.

In order to avoid such problems, the gas flow passage needs to be properly sized. Since the gap will widen as propellant is consumed, it is possible to balance the rate at which the deformation-related gap closure changes against the propellant burn back rate and thus achieve a true optimum. In actual practice, this approach results in a marginal solution fraught with uncertainties regarding the many parameters that simultaneously influence the result. A different approach is to require that the flow path be large enough to insure a stable equilibrium between the deformation and flow mechanisms. Clearly, such an equilibrium exists for a grossly oversized gap. In practice, due to the parameters involved in a typical rocket

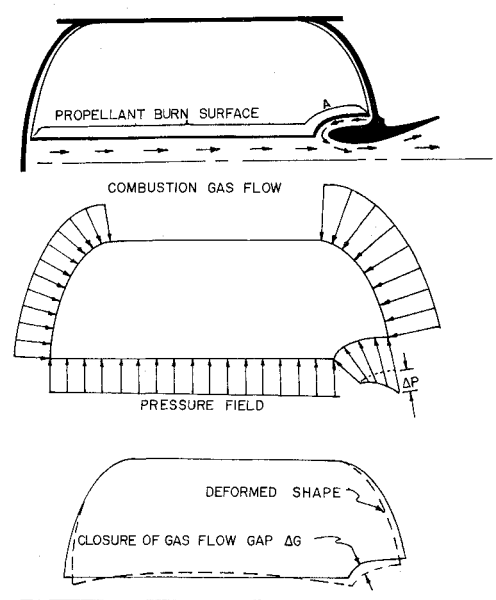

Fig. 1 Operating deformations in a submerged-nozzle solid rocket motor.

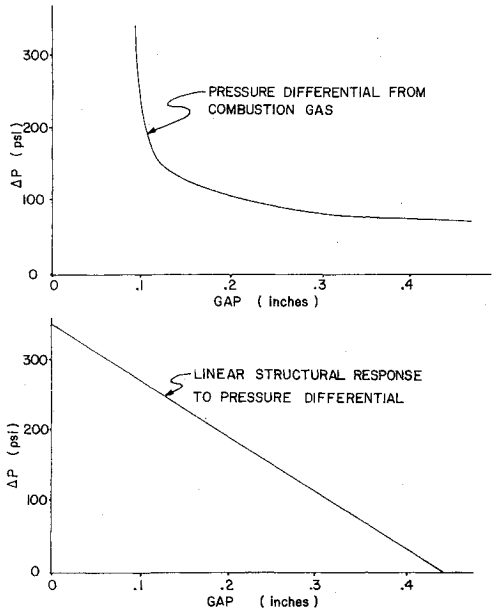

Fig. 2 Independent relationships between gap size and ΔP.

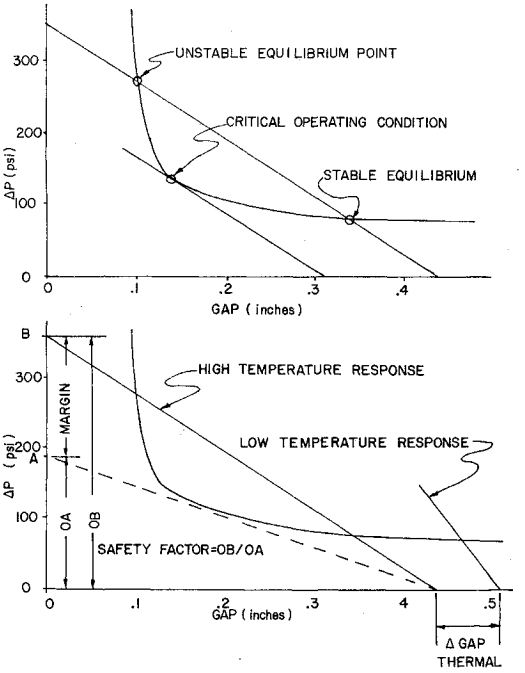

Fig. 3 Combined mechanism plots.

motor, the stable equilibrium exists for gaps that are very close to the true optimum, thereby reducing the problem to one of determining the point of stable equilibrium.

Analysis

To find the stable equilibrium point, we proceed to solve the flow and deformation problems independently, superimpose the plotted results, and look for points in common between the solutions. The fluid flow problem is solved for fixed gap sizes using numerical integration and the results are plotted as maximum pressure differential vs gap size, as shown in Fig. 2. Next, the structural deformations are computed using an elasticity computer program[2] and appropriate effective moduli. This structural analysis yields a gap dimension that varies linearly with the applied pressure differential, as shown in Fig. 2. Finally, the two results are combined (Fig. 3) and the intersections of the two lines are identified as the equilibrium points, since they simultaneously satisfy both analyses. One is also a point of stable equilibrium, as can be seen by considering small excursions about it and recognizing that the existing mechanisms will drive the system back to the intersection point. In general, there are two intersection points, the one at the higher ΔP representing an unstable equilibrium point, as shown in Fig. 3.

Composite Plot Applications

Composite plots (Fig. 3) represent an equilibrium solution at the point of intersection of the structural and gas flow curves; when no such intersection exists, equilibrium cannot be achieved. The value of such a plot lies in its ability to depict the margins existing relative to the equilibrium operations. Consider the situation that occurs as the initial gap is made smaller. For small perturbations such as this, the structural response will follow a parallel line. Eventually, by continuing to reduce the gap, we will obtain the critical operating condition line shown in Fig. 3, which no longer achieves stable equilibrium. This condition establishes the critical initial gap dimension and provides a measure of the dimensional margin inherent in a specific gap design. Other parameters will influence the gap selection process, however, and need to be considered.

When the rocket motor is cooled from its maximum operating temperature to its lowest operating temperature, the propellant grain undergoes a thermal deformation. This deformation can alter the gap significantly in either direction. It is necessary to perform a finite element structural thermal deformation analysis to determine this aspect of the problem. In this instance, the gap increases at low temperature and, furthermore, the propellant mechanical response becomes stiffer. Both of these factors (depicted in Fig. 3) reduce the significance of the problem at low temperature in this design, with the consequence that the high-temperature operation governs the gap size. Other loads such as propellant slump (long-term creep) during storage should be considered when pertinent. It is desirable to construct a safety margin for the gap design to accommodate the many variables that cannot be accurately controlled during production and subsequent motor utilization. To this end, the actual gap dimensions are relatively simple to control since trimming or other processing operations can reliably produce a specific gap size. However, the propellant stiffness is often a difficult parameter to control. This is because processing controls are more oriented toward meeting ballistic parameters such as the burn rate, with the consequence that propellant stiffness tends to be a secondary parameter.

To construct a safety factor related to propellant stiffness, a critical propellant modulus line is plotted. This critical operation line (dashed line in Fig. 3) begins at the same point as the design response line, but has a slope adjusted to make it tangent to the gas flow curve. Comparing the slopes of these lines leads to the safety factor OB/OA shown in Fig. 3.

It has been found that a reasonable safety factor to adopt is one in which OB/OA is greater than or equal to two. Other factors could be employed when the specific conditions warrant it. By relating the variable of concern to the critical operation line, it is usually possible to construct an appropriate quantifiable safety factor.

Summary

A simplified methodology has been presented for sizing combustion gas flow passages in solid rocket motors where propellant deformations must be accommodated. Combined plots of the gas flow and propellant deformation mechanisms are shown to be useful guides to the design process. Critical conditions relative to stable equilibrium operation are identified and a safety margin relative to this critical sizing has been defined. This methodology has been presented in outline form and should be tailored to specific problems and to the

computational capability available. No attempt has been made to justify the many assumptions and simplifications involved, for they may differ in other systems; rather the intent has been to convey an approach that has proved to be useful in such problems encountered over the past several years.

References

[1]Messner, A.M., Taylor, G.Q., and Price, C.F., "Interdisciplinary Computer Analyses of Three-Dimensional Solids Defined by Polyhedral Surfaces," *Journal of Spacecraft and Rockets,* Vol. 11, Jan. 1974, pp. 52-54.

[2]Messner, A.M. and Schliessman, D., "Transient Thermal Stresses in Solid Propellant Grains," *Journal of Spacecraft and Rockets,* Vol. 2, July 1965, pp. 565-570.

Performance Parameters of Some New Hybrid Hypergols

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Introduction

THE hypergolic nature of thiocarbonohydrazones, the solid derivatives of thiocarbonohydrazide with various carbonyl compounds, on coming into contact with fuming nitric acid, was discovered recently in the authors' laboratory.[1] The short ignition delays of some of these systems, measured using a drop-tester-type device, indicate the potential use of the thiocarbonohydrazones (TCH) as hypergolic hybrid rocket fuels. In addition to short ignition delays, these materials have other desirable fuel properties that include high stability toward aerial oxidation, nonhygroscopic nature, high melting/decomposition temperatures,[2] etc. All of these aspects led to the evaluation of such performance parameters as specific impulse, characteristic velocity, flame temperature, etc., of propellant systems comprising of TCH as solid fuels and anhydrous nitric acid (HNO_3) as liquid oxidizer. In doing so, however, it was realized that the thermochemical data of the various TCH needed for evaluation of the performance parameters are not reported in the literature. Therefore, the present study was undertaken to determine experimentally the heat of combustion and formation data and to evaluate the theoretical performance parameters of the various TCH-HNO_3 systems as hybrid propellants.

Experimental

Materials

Monothiocarbonohydrazones were synthesized by reacting thiocarbonohydrazide with various aldehydes and ketones, taken in 1:1 mole ratio and characterized as reported earlier.[1,3] For preparing bis-acetonethiocarbonohydrazone, thiocarbonohydrazide was refluxed with excess of acetone and the material precipitated was recrystallized using ice-cold petroleum ether.[4]

Heat of Combustion

Measurements of the heat of combustion (calorific value) were carried out using a conventional constant volume (300 ml) isothermal static bomb-calorimeter under constant pressure (30 atm) of oxygen. The water equivalent of the calorimeter was determined by combusting Analar-grade benzoic acid. To determine the heat of combustion, the sample thiocarbonohydrazone was pelletized and weighed, and wrapped with a known amount of surgical cotton to effect complete combustion. Appropriate corrections were applied for the heats of combustion of fuse wire and cotton. The average deviation in the heat of combustion values was on the order of ± 30 cal/g.

The Huffman-Ellis[5] method was adopted to evaluate the heats of combustion of various TCH. No water was taken in the bomb initially and $H_2SO_4 \cdot xH_2O$ was assumed to be the product of combustion, where $x+1$ moles of water were produced during combustion. The standard heats of formation were evaluated assuming idealized combustion and taking into account the heat of formation of $H_2SO_4 \cdot xH_2O$ actually formed.[5] A sample calculation of the standard heat of formation of formaldehydethiocarbonohydrazone ($C_2H_6N_4S$) is given in Table 1. The heat of combustion and formation of various TCH are given in Table 2.

Performance Calculation

The performance parameters were calculated using input data such as heat of formation of the fuel and oxidizer, composition of the propellant, chamber pressure and temperature, etc., on a DEC system 1090 computer, making use of the NASA-SP273 program. The chamber pressure *Pc* in hybrid rockets being of the order of 25-40 atm, was fixed at 30 atm in the present study. The parameters were evaluated by both equilibrium flow and frozen flow conditions at various oxidizer/fuel (*O/F*) ratios and exit pressures. However, for realistic considerations of hybrid systems, the parameters are reported at exit pressure $P=0.5$ atm, i.e., at expansion ratio *Pc/P* equal to 60 assuming equilibrium flow conditions.

Results and Discussion

The heat of formation data presented in Table 2 show that many of the TCH reported herein are endothermic or weakly exothermic compounds. This is significant from an energy point of view. However, it may, be pointed out that measurement of heats of combustion of compounds containing sulfur involves several difficulties. Although rotating bomb-calorimeters have been recommended for very high precision work, the Huffman-Ellis[5] method using static bomb-calorimeters has been reported[6] to yield moderately accurate values when used in carefully planned comparative measurements. Perhaps this method is appropriate for the present study, since, in evaluating the heats of formation, only heat of dilution of H_2SO_4 by the theoretically formed water is taken into ac-

Table 1 Sample calculation of the standard heat of formation

Molecular weight	118.16
Sample weight	0.3127 g
Total calories to calorimeter	1808 cal
Total correction	384.7 cal
Molar heat of combustion at contant volume (ΔE_c)	537.82 kcal
Mean value	537.04 kcal
Idealized combustion equation	
$C_2H_6N_4S_{(s)} + 5O_{2(g)} \rightarrow 2CO_{2(g)} + 2N_{2(g)} + H_2SO_4 \cdot 2H_2O_{(l)}$	
Molar heat of combustion at constant pressure (ΔH_c)	537.63 kcal
Standard heat of formation at constant volume (ΔE_f)	
$[2\Delta E_f(CO_2) + 3\Delta E_f(H_2O) + \Delta E_f(H_2SO_4 \cdot 2H_2O)] - \Delta E_c$	
	+11.55 kcal/mole
Similarly, Standard heat of formation at constant pressure (ΔH_f)	+8.57 kcal/mole

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