Pulse Performance Analysis for Small Hypergolic-Propellant Rocket Engines

GERALD W. SMITH*
NASA Marshall Space Flight Center, Huntsville, Ala.
AND
RICHARD H. SFORZINI†
Auburn University, Auburn, Ala.

Theme

SMALL rocket engine tests were conducted for the purpose of obtaining pulse performance data to aid in preliminary design and evaluation of attitude control systems. Performance data obtained from tests on two hypergolic rocket engines are compared with theoretical performance calculated from idealized chamber filling and evacuation characteristics. Factors which influence engine transient response during pulse mode operation are identified. The theoretical analysis is modified to obtain semiempirical performance equations for hypergolic rocket engines which are demonstrated to be reasonably accurate for two different engine configurations over a considerable range of duty cycles.

Contents

The extensive use of bipropellant systems for attitude control of launch vehicles and spacecraft has prompted numerous investigations into the design, development and testing of small (1–100 lbf thrust) bipropellant engines. In a typical attitude control system (ACS) application, the normal firing cycle (duty cycle) is that of pulse mode operation where the pulse duration may be as short as a few milliseconds. This may be combined with longer burn times of several minutes. This varying mode of operation affects engine performance by lowering the engine specific impulse when compared to a continuous steady-state operation.

Hypergolic bipropellants react in both liquid and gaseous states with ignition delay times for hydrazine-nitric acid mixtures, for example, ranging from 0.1 msec to 3 msec.[1] Corresponding flame speeds have been measured which show that for small rocket engines 3 in. in length or diameter, the flame sweeps the chamber in about 1 millisecond. Chamber pressurization occurs over a more prolonged period than ignition initiation and flame spreading. For very short pulses the thrust trace may be completely transient although normally there is a steady-state portion characterized by relatively constant pressures, temperatures and flow rates. The final portion of the trace is that of thrust decay or tailoff. Most theoretical analyses treat the shutdown sequence as an instantaneous stopping of gas generation followed by either an isentropic or isothermal decay.[2,3] Such idealizations, however, fail to give accurate representation of the transient behavior.

The experimental results are based on engine tests conducted at the NASA Marshall Space Flight Center. The basic features of the test hardware are summarized in Table 1. The tests were conducted using nitrogen tetroxide (N_2O_4) and monomethylhydrazine (MMH). Engine A was selected as the primary data source based on engine configuration, quality of the test data and the type of instrumentation used during the test. The propellant valve for engine A was torque motor operated with mechanically linked oxidizer and fuel valves, whereas separate oxidizer and fuel solenoid valves were used with Engine B.

The tests were conducted in a vaccum chamber using a steam ejector system with diffuser to simulate altitude conditions up to 115,000 ft. A drag body type flow meter was used for pulse flow measurements requiring rapid response; a turbine type meter was used for verification and steady flow measurement. The combustion chamber pressure was measured by a strain gauge pressure transducer capable of measuring a rise time of 1.0 msec from 10% to 90% of any pressure step input. Difficulties in accurately measuring thrust necessitated the calculation of thrust based on chamber pressure measurements. The duty cycle used during the Engine A tests included pulse on-time ranging from 30 msec to 5 min with pulse off-time varying from 100 msec to 100 sec.

The pressure and flow rate transients (buildup and decay) appeared to be repetitive regardless of duty cycle except for the shortest pulse off-time (excluded from the analysis) where the off-time is insufficient to allow the pressure and flow rates to decay completely between pulses. Analysis of the test data for both Engines A and B revealed that delays of approximately 10 msec occurred after the valve electrical open signal before the chamber pressure started to rise. Similar delays follow the valve close signal and are attributed to the electromechanical characteristics of the valve, the hydraulic response characteristics of the propellant and the ignition initiation

Table 1 Description of test engines[a]

Test configuration	Engine A	Engine B
Propellants	N_2O_4/MMH	N_2O_4/MMH
Ratio of specific heats	1.24	1.24
Thrust, lbf (vacuum)	100	22
Chamber pressure, P_c, psia	142	95
Mixture ratio, 0/F	1.6	1.6
Nozzle expansion ratio	60:1	40:1
Characteristic velocity, c^*, fps	5300	5000
Throat area, A_t, in.^{2}	0.397	0.1327
Injector configuration	Unlike impinging doublets	Impinging doublet
Characteristic length, L^*, in.	11.00	7.0
Cooling technique	Film and radiation	Radiation
Film coolant, % fuel flow rate, $\dot{m}_f$	40.0	0

[a] Performance values are based on steady-state test data.

Received October 8, 1971; synoptic received February 22, 1972; revision received April 13, 1972. Full paper is available from National Technical Information Service, Springfield, Va., 22151, as N72-24823 at the standard price (available upon request).

Index categories: Liquid Rocket Engines; Rocket Engine Testing.

* Aerospace Engineer, Power and Propulsion Branch, Astronautics Laboratory. Associate Member AIAA.

† Professor, Aerospace Engineering. Associate Fellow AIAA.

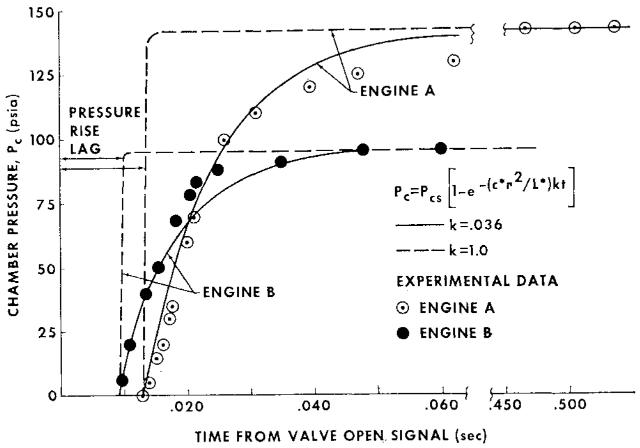

Fig. 1 Chamber pressure start transient, engine A and B.

delay. Lags also occur in propellant flow rates and have an appreciable effect on short pulse performance.

For analysis, it was assumed that the rise and decay lags are known or can be reasonably estimated. Assuming further that the propellant burns at the steady-state propellant injection rate, the perfect gas law applies, and the chamber temperature is constant during pressurization, then the continuity equation can be written as

$$dt = dP_c/c^*\Gamma^2(\dot{m}_p c^* - P_c A_t)/V_c \tag{1}$$

which when integrated gives

$$P_c = P_{cs}[1 - \exp(-c^*\Gamma^2 t/L^*)] \tag{2}$$

Here, $\dot{m}_p$ = propellant flow rate, V_c = chamber volume, $\Gamma^2 = \gamma[2/(\gamma+1)]^{(\gamma+1)/(\gamma-1)}$ and s refers to steady-state conditions (see Table 1 for basic nomenclature). Solution of the above equation using Engine A characteristics yields the result presented in Fig.1. Although substantial differences exist between theoretical and actual performance, Eq. (2) establishes a plausible basic form for the pressure transient. The solid lines in Fig. 1 are curve fits of an equation of the form of Eq. (2) to the test data. A multiplying constant "K" is used in the exponential term (see Fig. 1). A "K" value of 0.036 yields good correlation with the test data for both Engines A and B.

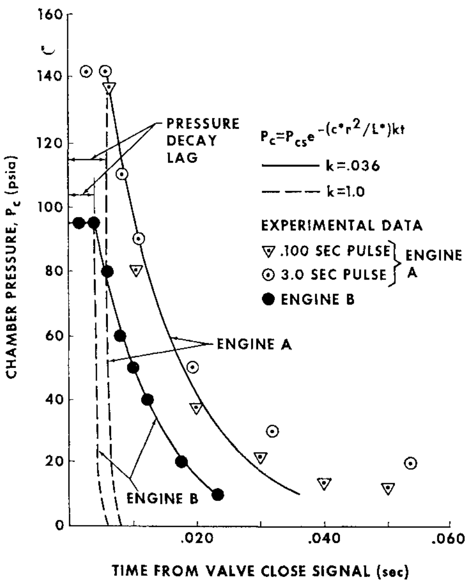

Fig. 2 Chamber pressure decay transient, engine A and B.

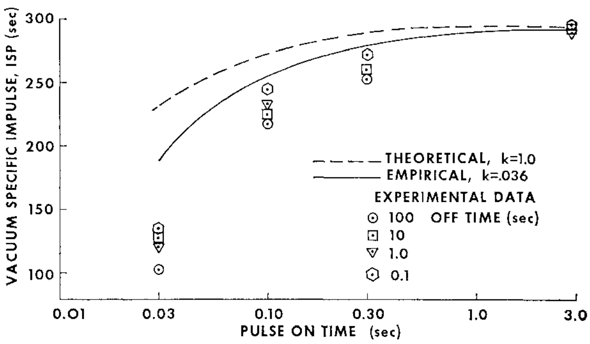

Fig. 3 Typical pulse performance, engine A.

Most theoretical models of the pressure decay transient describe the shutdown sequence as an instantaneous cessation of gas generation followed by a depressurization. For the isothermal case the result is[3]

$$P_c = P_{cs} \exp(-c^*\Gamma^2 t/L^*) \tag{3}$$

As shown in Fig. 2, the tests indicate a substantially longer decay rate than that calculated using Eq. 3. Again using the basic form of this equation with the "K" factor, a value of 0.036 best described the decay transient for both Engines A and B. For Engine A, the longer pulse on-times produced longer decay transients due possibly to increased chamber wall temperatures resulting from the longer pulse durations.

The assumption of a step flow rate at the steady-state level beginning with the first evidence of flow provided a reasonable estimate of the gross initial propellant flow rate response. Straight line approximations of the flow decay rate for Engine A provided excellent correlation with the test data using a decay slope of 7.4 lb/sec² and 33.7 lb/sec² for the fuel and oxidizer, respectively.

Specific impulse was determined by the following expression:

$$I_{sp} = A_t c_F \int P_c dt \Big/ \int \dot{m}_p dt \tag{4}$$

where c_F is the thrust coefficient. This expression with the previously determined equations for pressure and flow rate transients gives the predicted performance shown as a solid line in Fig. 3. A prediction based directly on theoretical start and decay transients without the "K" factor but including experimentally determined values for pressure and flow rate lag times is indicated by the dashed line on the figure.

Predictions based on the semiempirical analysis resulted in deviations of less than 10% from the measured values for pulse on-times of 100 msec or greater. For the very short pulse, propellant overshoot characteristics in both the fuel and oxidizer, which are not accounted for in the model, become significant causing the theoretical specific impulse to be somewhat higher than the actual test data.

The model described herein is based on limited test data; however, similarity was found in the existence of a common "K" factor for both engines. Further testing and analysis with different engines and different propellants are needed before the generality of the results can be established.

References

[1] Kilpatrick, B., *A Study of Fast Reactions in Fuel-Oxidant Systems*, Williams and Wilkens, Baltimore, Md., p. 196.

[2] Rodean, H. C., "Rocket Thrust Termination Transients," *ARS Journal*, Vol. 29. No. 6, June 1959, pp. 406–409.

[3] Barrère, M., Jaumotte, A., De Veubeke, B. F., and Vandenkerckhove, J., *Rocket Propulsion*, Elsevier, New York, 1960, pp. 242-244.