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(ACCESSION NUMBER)  
15  
(PAGES)  
(NASA CR OR TMX OR AD NUMBER)

(THRU)  
(CODE) ~~27~~  
(CATEGORY)

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# TECHNICAL MEMORANDUM

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CLASSIFICATION CHANGE NOTICES NO. \_\_\_\_\_  
DATED \_\_\_\_\_ ITEM NO. \_\_\_\_\_

PERFORMANCE OF A COMPOSITE SOLID PROPELLANT  
AT SIMULATED HIGH ALTITUDES

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DECLASSIFIED: EFFECTIVE 4-29-65  
AUTHORITY F.G. DROBKA (ATSS\*A)  
no dated 5-13-65:AFSDO 5439

GPO PRICE \$ \_\_\_\_\_

OTS PRICE(S) \$ \_\_\_\_\_

Hard copy (HC) 1.00

Microfiche (MF) 50



NATIONAL AERONAUTICS AND SPACE ADMINISTRATION  
WASHINGTON  
December 1959

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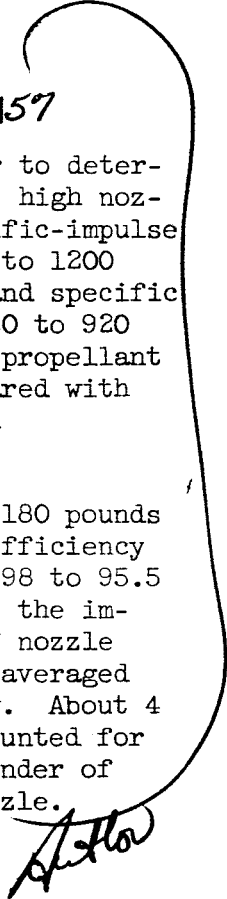
By Carl C. Ciepluch

SUMMARY

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An investigation was conducted in an altitude test chamber to determine the performance of a typical composite solid propellant at high nozzle pressure ratios and for a range of chamber pressure. Specific-impulse measurements were made over a range of pressure ratio from 115 to 1200 for fully expanded flow. The characteristic exhaust velocity and specific impulse were determined for a range of chamber pressure from 180 to 920 pounds per square inch absolute. Experimental measurements of propellant specific impulse and characteristic exhaust velocity were compared with theoretically calculated values for both frozen and equilibrium expansions.

For a reduction in operating chamber pressure from 920 to 180 pounds per square inch absolute, the characteristic exhaust-velocity efficiency based on equilibrium flow decreased 2.5 percentage points from 98 to 95.5 percent. For the same reduction in operating chamber pressure, the impulse efficiency decreased only about 1 point. Over a range of nozzle pressure ratio from 115 to 1200, the measured specific impulse averaged 7 percent lower than the theoretical value for equilibrium flow. About 4 percent of the 7 percent loss in specific impulse could be accounted for by considering combustion and nozzle inefficiencies. The remainder of the loss was apparently a result of frozen expansion in the nozzle.



INTRODUCTION

The propulsive characteristics of solid propellants that are necessary for calculation of missile or vehicle requirements can readily be obtained for moderate pressure ratios in sea-level studies. However, for high-performance stages, which operate at low chamber pressures (for reduction of case weight) and high altitudes, extreme extrapolation of sea-level performance to higher pressure ratios is required. Theoretical performance calculations can be used as a guide for this extrapolation; however, these calculations depend on the accuracy of the thermodynamic data

\*Title, Unclassified.



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involved. Furthermore, the impulse loss due to nonequilibrium flow is difficult to predict. Therefore, having experimental methods of determining the performance of solid-propellant rocket engines at high expansion ratios and low chamber pressures would be advantageous. An experimental investigation was therefore conducted in the Lewis altitude wind tunnel to determine the performance of a typical composite solid propellant at high nozzle pressure ratios and low chamber pressures. The effect of chamber pressure on characteristic exhaust velocity or combustion temperature for the propellant investigated was also determined.

The performance data include the specific impulse for nozzle pressure ratios up to 1200 (equivalent area ratio, 65) and the effect of chamber pressure on characteristic exhaust velocity and specific impulse. Theoretically calculated performance was determined for the range of variables investigated experimentally.

### APPARATUS AND PROCEDURE

#### Propellant

The solid propellant used in this investigation is designated "Arcite 368" and has the following composition:

	Percent by weight
Ammonium perchlorate (oxidizer) <sup>a</sup>	81.03
Polyvinyl chloride (binder)	8.44
Dibutylsebacate (plasticizer)	9.90
Carbon black	.05
Stabilizer and viscosity depressant	.58

<sup>a</sup>Particle-size distribution:  
 2 Parts, 30 microns  
 1 Part, 170 microns

The grain was an end-burning type approximately 12 inches in diameter and 20 inches long. Nominal grain weight was 120 pounds. The grain contained 55 axially oriented, evenly spaced, 0.005-inch-diameter silver wires used for increasing the propellant burning rate. At a chamber pressure of 1000 pounds per square inch absolute, the burning rate of the propellant containing silver wires was 2.6 inches per second. Calculated equilibrium

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combustion temperature of the propellant is 5122° R at 1000 pounds per square inch, and the calculated equilibrium specific impulse for expansion from 1000 pounds per square inch absolute to sea-level pressure is approximately 240.

### Installation

The rocket engine (fig. 1) was installed in the altitude wind tunnel, which provided control of the ambient pressure. The engine was mounted on a thrust stand that employed the flexure-plate principle for thrust measurement. The propellant was ignited by means of a spark-ignited hydrogen-oxygen torch. A conical nozzle with a 45° convergent half-angle and a 15° divergent half-angle was used. The nozzle-throat contour was an arc, the radius of which was equal to the diameter of the nozzle throat. The nozzle was uncooled and was constructed with heavy walls that were insulated internally with a "flame-sprayed" coating of zirconium oxide. There was essentially no throat enlargement due to erosion during the rocket firings.

All variables were measured with transient instrumentation. These included thrust, chamber pressure, nozzle-wall static pressure, and ambient pressure. Transient measurements were recorded on a direct-recording oscillograph. The estimated accuracy with which the chamber-pressure and thrust measurements could be made was:  $\pm 1/2$  percent. The weight of grain and inhibitor burned during each firing could be measured to the nearest 1/4 of 1 percent. Chamber-pressure static taps were located in a region of low enough Mach number so that they measured total chamber pressure. Tubing lengths to chamber-pressure pickups were kept to 6 inches or less in order to ensure adequate frequency response.

The data were taken at the following conditions:

Average chamber pressure, $P_{c,a}$ , lb/sq in. abs	Nozzle ambient pressure, $P_0$ , lb/sq in. abs	Nozzle area ratio, $A_e/A_t$	Nozzle throat area, $A_t$ , sq in.
920	0.8	65.0	2.39
920	1.2	47.7	2.39
920	2.0	32.4	2.39
920	3.2	22.8	2.39
920	8.0	11.4	2.39
333	3.2	11.4	4.37
180	2.0	11.4	5.74

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The tunnel ambient pressure was adjusted for each firing to expand approximately fully the exhaust gases for each combination of area ratio and chamber pressure investigated. However, the nozzle-exit pressure was usually slightly different from the ambient pressure. The total impulse was adjusted to represent fully expanded flow by subtracting the  $A_e(p_e - p_0)$  term from the measured thrust. (All symbols are defined in appendix A.) The weight of the propellant burned during the firing was determined by weighing the propellant and inhibitor before and after each firing. Calculations of average values of specific impulse and characteristic exhaust velocity  $c^*$  were made for each run. The details of the calculation procedure are discussed in appendix B.

## RESULTS AND DISCUSSION

### Effect of Chamber Pressure on Characteristic Velocity and Specific Impulse

The effect of chamber pressure on the characteristic exhaust velocity  $c^*$  of the propellant is shown in figure 2(a). A slight reduction in  $c^*$  efficiency ( $c^*/c_{th}^*$ ), a result of combustion inefficiency, is noted as the chamber pressure is reduced. For a reduction in chamber pressure from 920 to 180 pounds per square inch absolute,  $c^*$  efficiency based on equilibrium flow decreased approximately 2.5 points from 98 to 95.5 percent. Since the combustion temperature is approximately proportional to the square of  $c^*$ , combustion-chamber temperatures corresponding to chamber pressures of 920 and 180 pounds per square inch absolute would be 96 and 91 percent of the theoretical equilibrium value, respectively.

Since the specific impulse of a propellant varies approximately directly with  $c^*$ , the observed reduction in  $c^*$  efficiency as chamber pressure was reduced should produce approximately the same reduction in specific impulse efficiency ( $I/I_{th}$ ) at equivalent nozzle pressure ratios. However, as indicated in figure 2(b), a reduction of only about 1 percentage point was obtained. There are two possible effects that could compensate the expected reduction in specific impulse due to decreased measured values of  $c^*$  at low chamber pressures. One explanation is that combustion inefficiencies in the chamber reduced experimental  $c^*$  values, but some thrust was recovered because of combustion in the nozzle. A second possibility is that at low chamber pressures, the lower combustion temperatures produced less dissociation in the gases that expanded in the nozzle. Thus, thrust losses from dissociation would be less for the runs at low chamber pressures.



Comparison of Measured and Calculated Specific Impulse  
at High Nozzle Pressure Ratio

A comparison between measured and theoretically calculated values of specific impulse is shown in figure 3 for a range of nozzle pressure ratio. These data are for fully expanded flow and a nominal average chamber pressure of 920 pounds per square inch absolute. The measured specific impulse averaged about 93 percent of the theoretical equilibrium specific impulse for the range of pressure ratio from approximately 100 to 1200. Of the 7 percent loss in specific impulse, approximately 4 percent can be accounted for as a result of the following: (1) a 2 percent loss in specific impulse because the measured  $c^*$  was 2 percent lower than theoretical; (2) a 1.7 percent loss in specific impulse due to non-axial flow discharge from the  $15^\circ$  half-angle nozzle (ref. 1); and (3) a fraction of 1 percent due to heat absorbed by nozzle and friction losses. Correction of the measured specific impulse by 4 percent (fig. 3) results in measured values approximately equal to the frozen expansion curve, indicating that the flow is probably closer to frozen than to equilibrium flow.

A comparison of the theoretical and experimental variation in pressure ratio with area ratio for complete expansion is shown in figure 4. The experimental data were compiled from several firings at a nominal chamber pressure of 920 pounds per square inch absolute. The chamber pressure and wall static pressures used to calculate the pressure ratio are instantaneous values. Although considerable scatter occurs in the data at low area ratios, it appears that the required area ratio for complete expansion at a given pressure ratio is lower than that calculated for both equilibrium and frozen flow.

SUMMARY OF RESULTS

The following results were obtained during an investigation of the performance of a composite solid propellant at high nozzle pressure ratios and a range of chamber pressure:

1. The characteristic exhaust-velocity efficiency decreased slightly as operating chamber pressure was reduced, indicating a reduction in combustion efficiency. For a reduction in chamber pressure from 920 to 180 pounds per square inch absolute, the characteristic exhaust velocity efficiency decreased from 98 to 95.5 percent. The specific-impulse efficiency decreased only about 1 percentage point for the same variation in operating chamber pressure.



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2. Measured values of specific impulse averaged about 7 percent lower than calculated equilibrium flow values for a range of pressure ratio from 100 to 1200. Of this 7 percent loss in specific impulse, approximately 4 percent could be accounted for by considering combustion inefficiency and nozzle thrust losses. When measured values of specific impulse were corrected for nozzle and combustion inefficiencies, the measured specific impulse was approximately equal to the calculated values for frozen flow.

Lewis Research Center  
National Aeronautics and Space Administration  
Cleveland, Ohio, July 10, 1959

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## APPENDIX A

## SYMBOLS

A	cross-sectional area, sq in.
$A_e$	nozzle-exit cross-sectional area, sq in.
$A_t$	nozzle-throat area, sq in.
$c^*$	characteristic velocity, ft/sec
$c_{th}^*$	theoretical characteristic velocity, ft/sec
F	thrust, lb
g	acceleration due to gravity, 32.2 ft/sec <sup>2</sup>
I	specific impulse, (lb)(sec)/lb
$P_c$	instantaneous chamber pressure, lb/sq in.
$P_{c,a}$	time average chamber pressure, lb/sq in.
$P_e$	instantaneous static pressure in plane of nozzle exit, lb/sq in.
$P_{e,a}$	time average pressure in plane of nozzle exit, lb/sq in.
$P_0$	nozzle ambient pressure, lb/sq in.
t	time, sec
$t_a$	action time, defined as time between points at which chamber pressure has risen to 10 percent of its maximum value and fallen to 10 percent of its maximum value, sec
w	total weight of propellant and inhibitor burned, lb



APPENDIX B

CALCULATION PROCEDURE

No attempt was made to measure the instantaneous gas-flow rate because of the difficulty encountered with solid propellants, and therefore an average characteristic velocity was calculated in the conventional manner:

$$c^* = \frac{A_t g}{w} \int P_c dt$$

where  $\int P_c dt$  represents the area under the pressure-time curve. The value of  $w$  equals the total weight of burned propellant and inhibitor and was obtained by weight measurements of the propellant and inhibitor before and after firing.

Specific impulse was calculated from experimental data as follows:

$$I = \frac{\int F dt}{w}$$

where  $\int F dt$  represents the area under the thrust-time curve. The value of  $I$  thus calculated is an average value similar to  $c^*$ , since the chamber pressure (and, therefore, thrust) varies with time as indicated in figure 5. Because the chamber pressure is not constant, the nozzle pressure ratio varies. In order to correct the measured thrust to represent fully expanded flow, the following procedure was followed. An average chamber pressure was determined:

$$P_{c,a} = \frac{\int P_c dt}{t_a}$$

where  $t_a$  represents the action time (see fig. 5). The ratio of chamber pressure to nozzle-exit pressure was calculated from chamber-pressure and wall-pressure measurements. Since this ratio is constant for pressure ratios near design, an average value of nozzle-exit pressure was calculated:

$$p_{e,a} = P_{c,a} \frac{1}{P_c/p_e}$$

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The value of  $p_{e,a}$  then represents the average value of nozzle-exit pressure for fully expanded flow. Since by definition  $p_{e,a}$  should be equal to  $p_0$  for fully expanded flow, the measured value of  $\int F dt$  was corrected by subtracting the  $A_e(p_e - p_0)t_a$  term from the measured thrust as follows:

$$\int F dt \text{ (fully expanded)} = \int F dt - A_e(p_e - p_0)t_a$$

Theoretical calculations were made with the aid of a high-speed digital computer using the thermodynamic data, equations, and procedures described in references 2 and 3. Values of heats of formation of ammonium perchlorate (ref. 4), polyvinyl chloride (ref. 5), and dibutylsebacate (ref. 5) used were -69.42, -20.64, and -285.04 kilocalories per gram, respectively. The theoretical performance calculations assumed that the propellant consisted of only the fuels and the oxidizer. The small percentages of carbon black, stabilizer, and viscosity depressant, which are primarily organic, were added to the polyvinyl chloride in order to simplify the calculations.

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5. Farber, Milton, and Altman, David.: Compilation of Performance Calculations on Composite Solid Propellants. Rep. 20-73, Jet Prop. Lab., C.I.T., Dec. 15, 1953. (Contract DA-04-495-ORD-18.)

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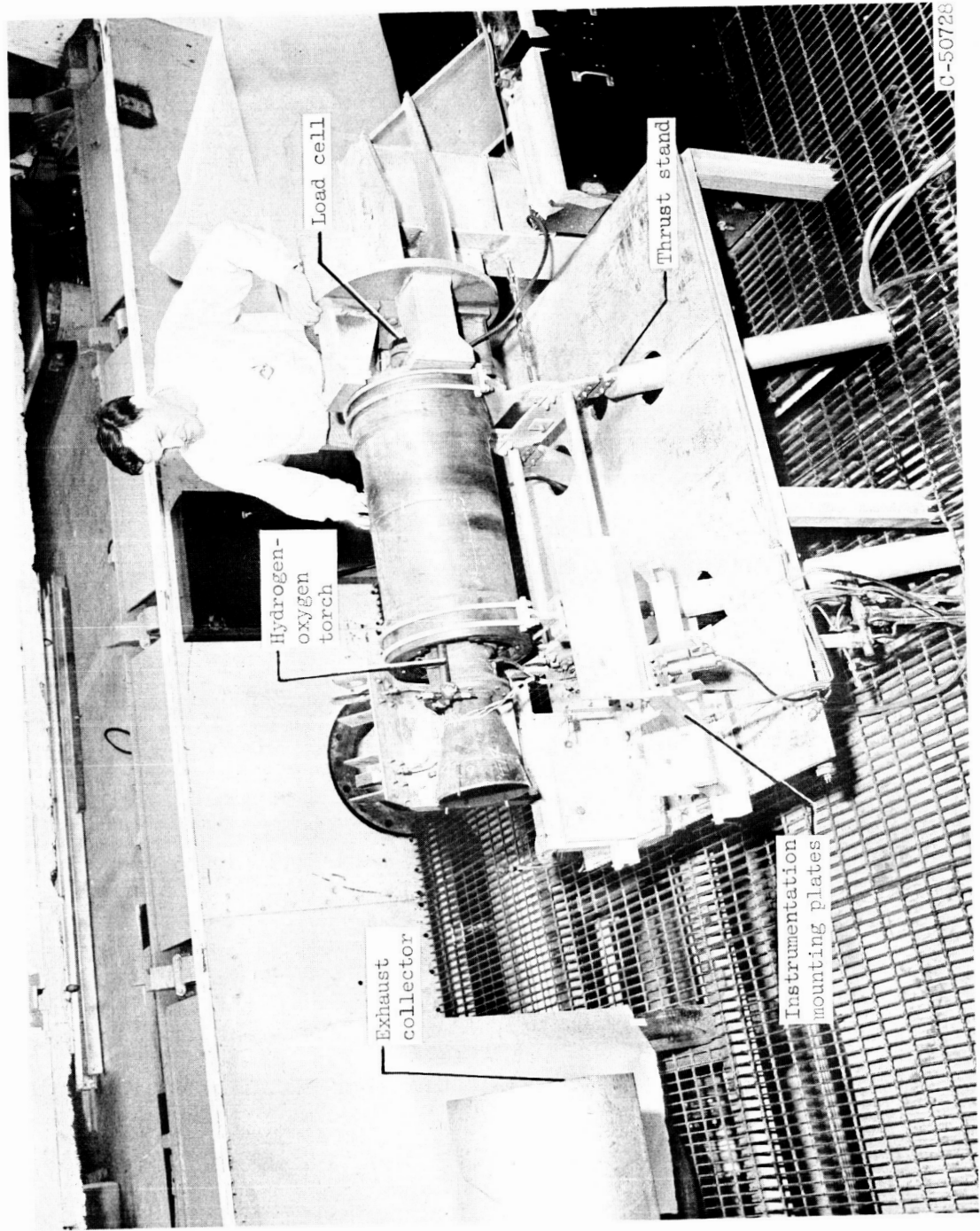


Figure 1. - Rocket-engine installation in altitude wind tunnel.

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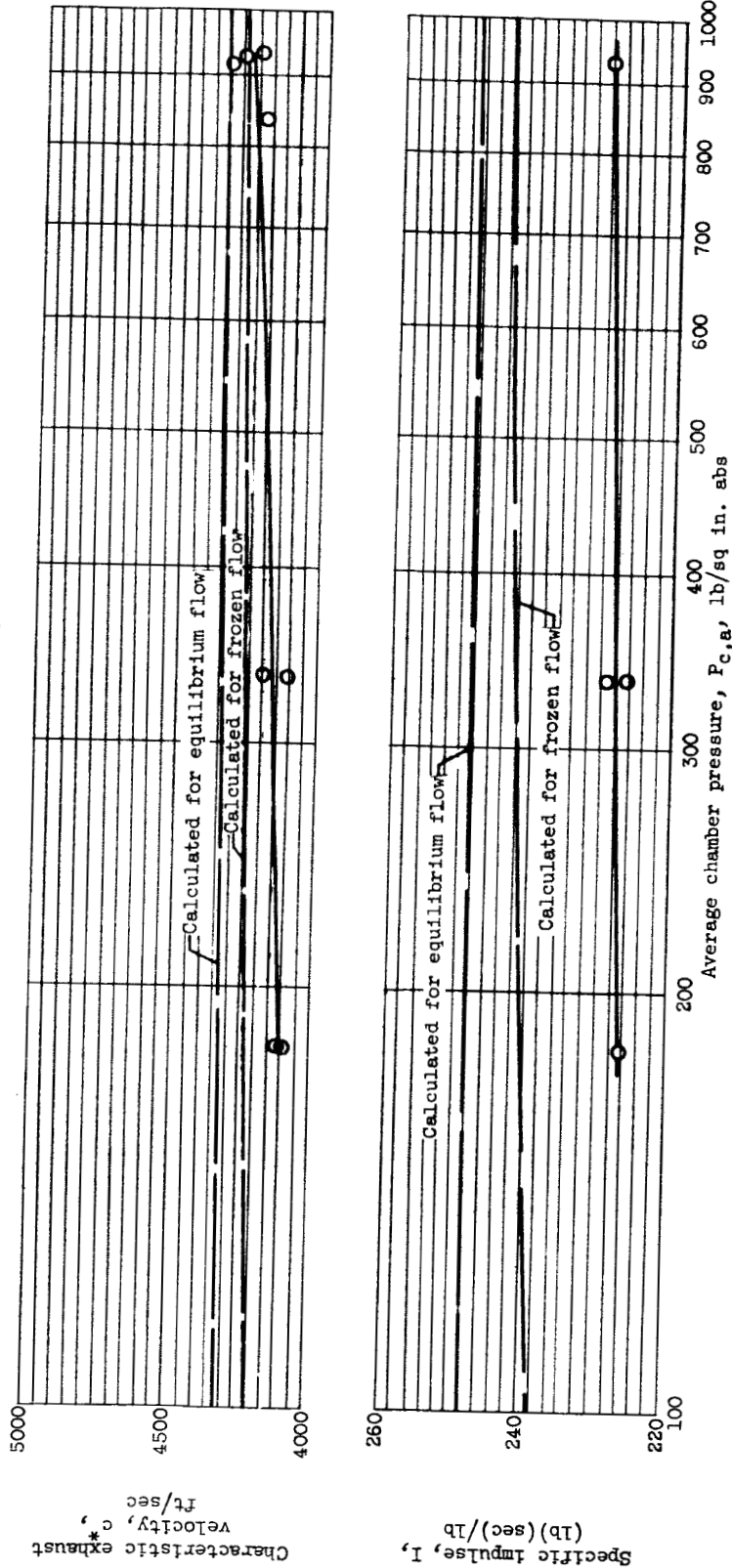


Figure 2. - Comparison of experimental and calculated values of characteristic velocity and specific impulse over a range of chamber pressure. Specific impulse for complete expansion at nozzle pressure ratio of 11.5.

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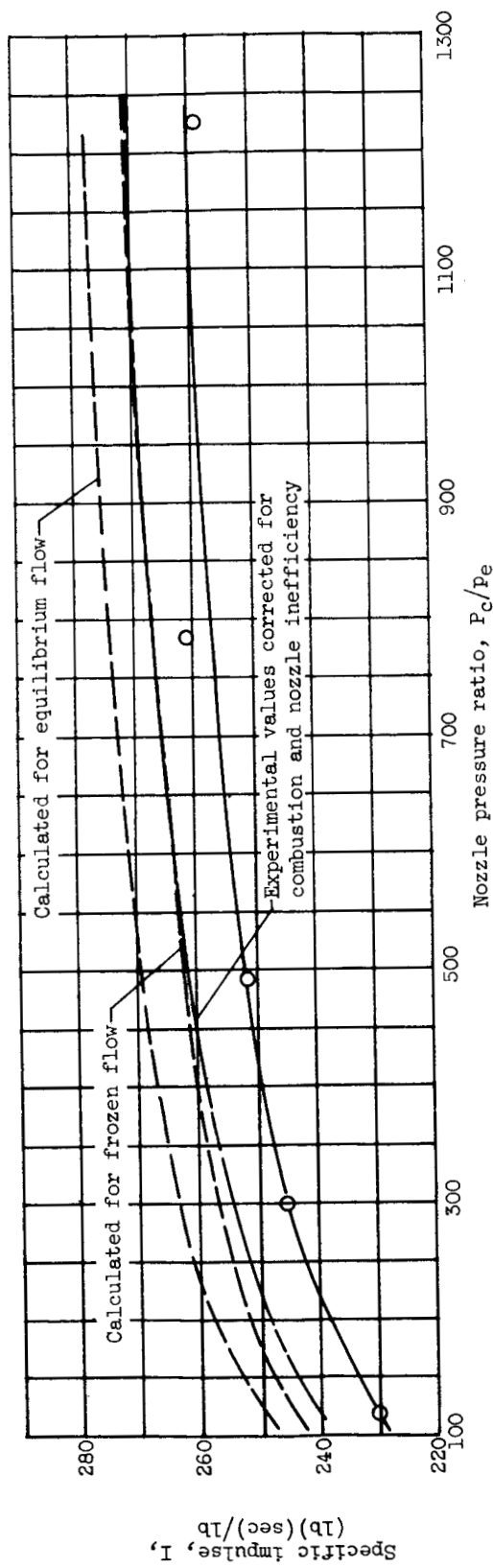


Figure 3. - Comparison of experimental and calculated specific impulse over a range of nozzle pressure ratio for complete expansion. Average chamber pressure, 920 pounds per square inch absolute.

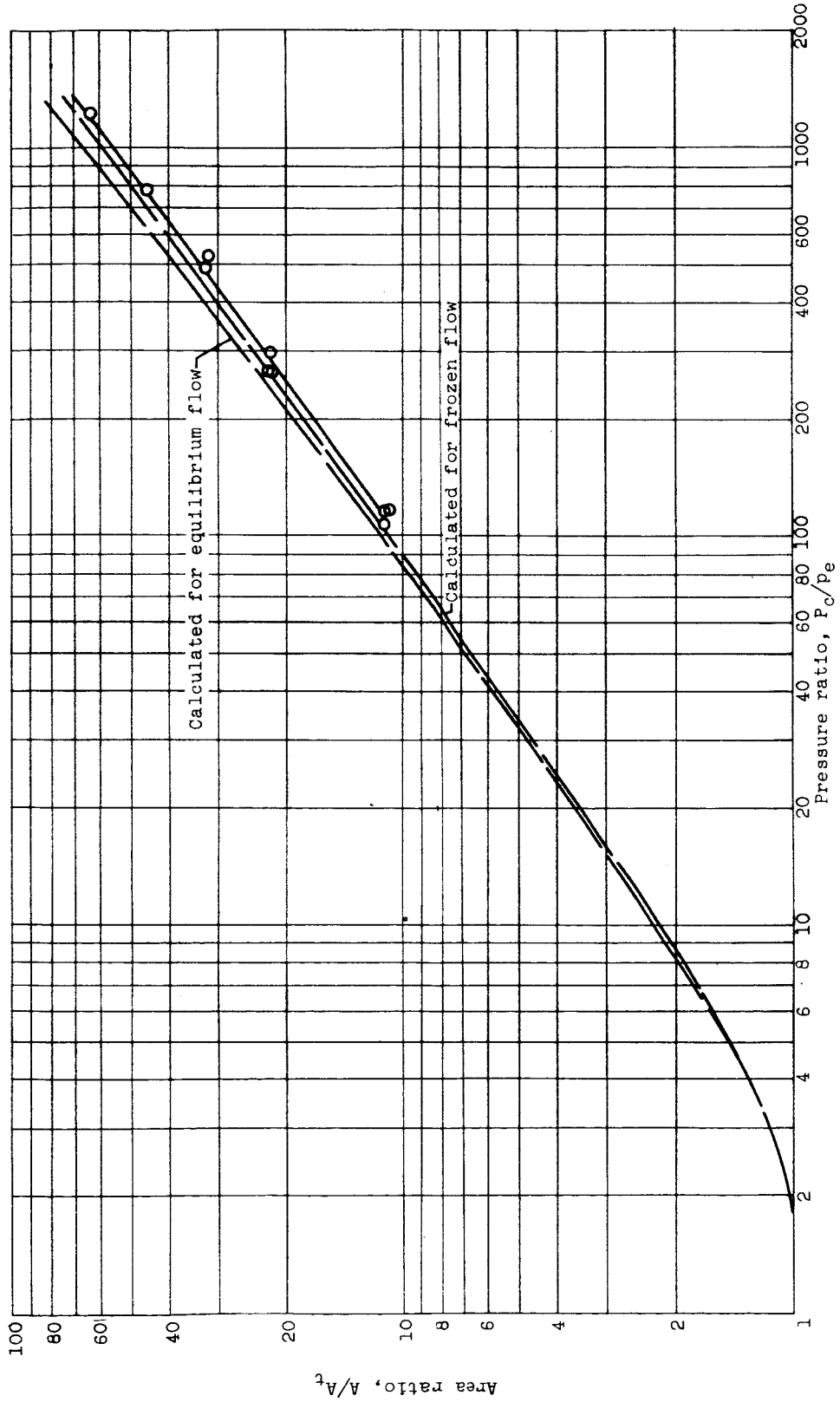


Figure 4. - Variation of pressure ratio with area ratio for complete expansion. Average chamber pressure, 920 pounds per square inch absolute.

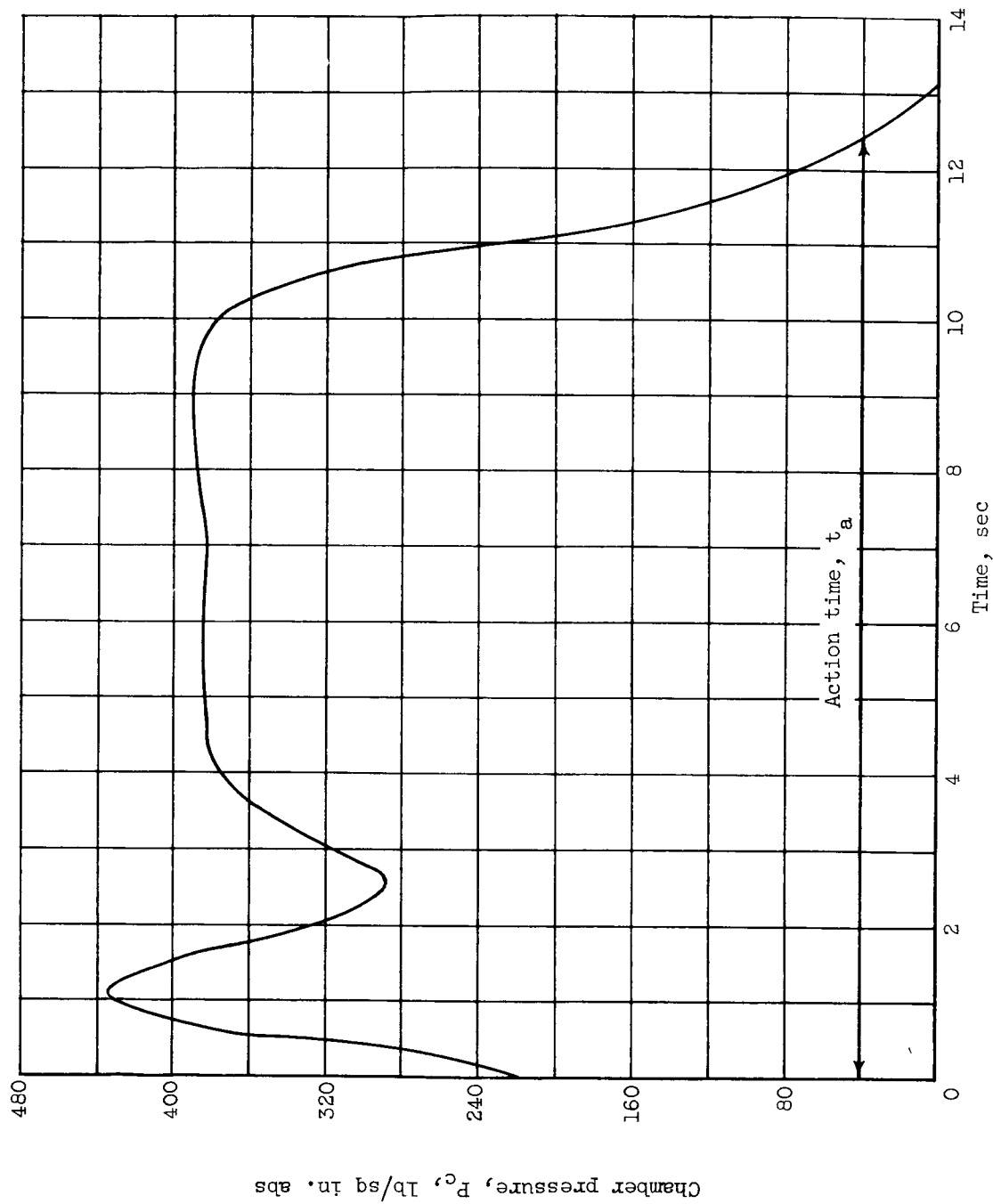


Figure 5. - Typical chamber pressure transient record for rocket firing.