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INVESTIGATION AT HIGH ALTITUDES OF ROCKET EXHAUST PLUME SYMMETRY AND INTERACTION WITH A PLATE WITH A SCALED MOL THRUSTER

D. W. Hill, Jr.

ARO, Inc.

April 1969

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NOTICES

INVESTIGATION AT HIGH ALTITUDES OF ROCKET EXHAUST PLUME SYMMETRY AND INTERACTION WITH A PLATE WITH A SCALED MOL THRUSTER

D. W. Hill, Jr.
ARO, Inc.

FOREWORD

The work reported herein was performed at the request of Space and Missile Systems Organization (SAMSO), Air Force Systems Command (AFSC), under Program Element 35121F/632A.

The rocket engine and simulated vehicle skin tested were designed and fabricated by Marquardt Corporation and McDonnell Douglas Corporation, Missiles and Space Systems Division (MSSD), respectively.

The results of the tests presented were obtained by ARO, Inc. (a subsidiary of Sverdrup & Parcel and Associates, Inc.), contract operator of the Arnold Engineering Development Center (AEDC), AFSC, Arnold Air Force Station, Tennessee, under Contract F40600-69-C-0001. The tests were conducted from May through December 21, 1968, under ARO Project No. SB0721, and the manuscript was submitted for publication on February 26, 1969.

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This technical report has been reviewed and is approved.

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ABSTRACT

A 1-lb-thrust rocket engine was tested to determine plume symmetry and Mach number contours at 400,000-ft altitude. Total pressure surveys were made with and without a flat plate in the plume. The tests without the plate indicated the plume was symmetrical, and measured results compared well with computed values. In tests with the plate, the shock interaction and boundary-layer buildup on the plate were measured.

CONTENTS

APPENDIXES

I. ILLUSTRATIONS

NOMENCLATURE

M	Mach number
$\dot{m}$	Thruster mass flow rate
O/F	Oxidizer-to-fuel ratio
P	Pressure
t	Time
x	Axial distance along thruster centerline
y	Distance from centerline of thruster
z	Distance from the surface of plate to centerline of pitot tube
γ	Ratio of specific heats

SUBSCRIPTS

2	Stagnation pressure behind shock
c	Combustion chamber
F	Fuel
O	Oxidizer
t	Total pressure

SECTION I
INTRODUCTION

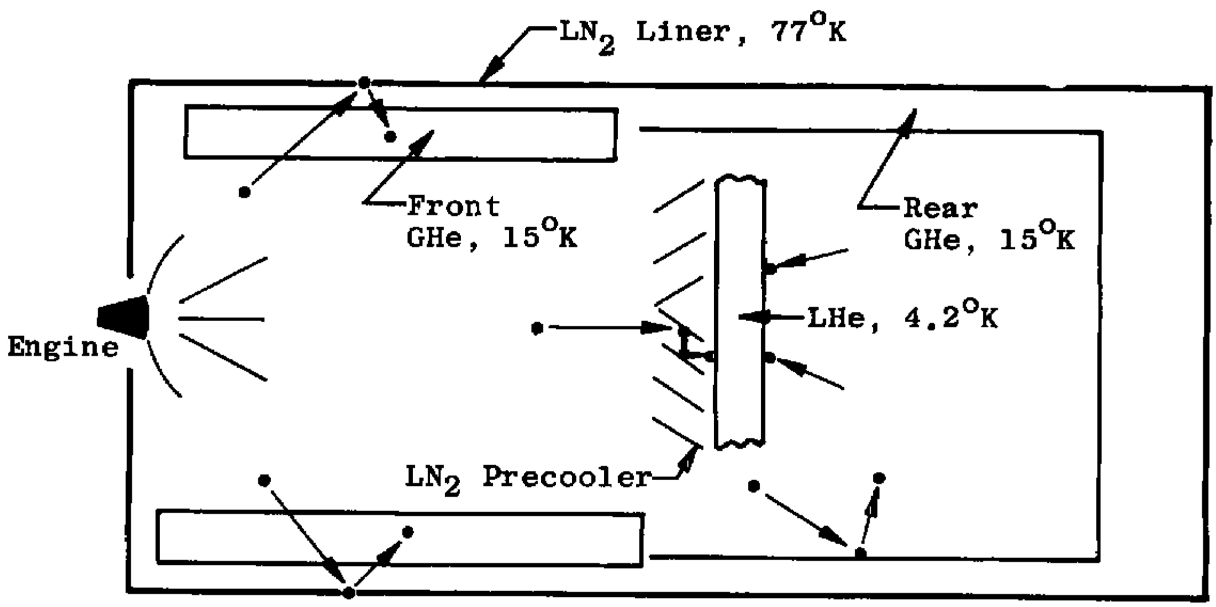
The exhaust plumes from attitude control rocket engines fired at orbital altitudes expand so that the exhaust products strike spacecraft surfaces and externally mounted components located within a large envelope extending downstream from the nozzle exit plane. The impingement of the gas results in local surface heating, surface pressures, and possible contamination of spacecraft surfaces. These effects may cause malfunctions of apparatus mounted on or inside the spacecraft. It is, therefore, important to identify and understand the conditions that cause these effects so that they may be properly considered in the design and operation of the spacecraft. There have been several analytical predictions of exhaust plume behavior (Refs. 1 and 2) and some limited experimental tests at high altitude (Refs. 1 through 4). These tests have been limited to short durations with transient altitudes from 400, 000 to 200, 000 ft.

The purpose of this test was to determine if there was plume symmetry and to determine the interaction of a 1-lb-thrust scaled thruster plume with a flat plate which simulated the vehicle skin. The test required firing the 1-lb-thrust scaled attitude control thruster continuously for up to 205 sec while maintaining a 400, 000-ft altitude.

SECTION II
TEST FACILITY

The test was conducted in the Aerospace Research Chamber (ARC) (8V) of the Aerospace Environmental Facility. The stainless steel chamber (Fig. 1, Appendix I) is 20 ft long and 10 ft in diameter. The cryopumping surfaces and arrangements were designed for removing gas products from rocket engines and low density nozzles of high enthalpy. The 620 ft^2 of liquid-nitrogen (LN_2)-cooled surfaces, 800 ft^2 of gaseous-helium (GHe)-cooled surfaces, and 50 ft^2 of liquid-helium (LHe)-cooled surfaces were arranged to remove 16 kw from the exhaust gas products in an optimum manner.

The sketch below and Fig. 1 show the arrangement of the cryosurfaces to pump the high enthalpy exhaust gas products. The gas leaving the engine passes through the radially arranged forward GHe surfaces and impinges on the annular LN_2 cryosurface where 8 kw is removed. The cooled gas is then either cryopumped by the LN_2 surface or reflected back onto the GHe cryosurface where it is condensed. There is a total GHe refrigeration capacity of 8 kw - 7 kw for the front GHe cryopump and the remaining 1 kw for the rear. Since hydrogen (H_2) has a high vapor pressure (10^{-4} torr) on 15°K GHe surfaces, LHe (4. 2°K) was used to remove the H_2. The H_2 and nitrogen (N_2) exhaust gases moving axially down the chamber impinge on the LN_2 precooler, where energy is removed, and then are cryopumped on the LHe cryosurfaces.



The front GHe cryosurfaces consist of fifty-two 8- by 1-ft panels positioned in a radial array about the axis of the chamber. The rear GHe cryosurface is 8 ft long and 6 ft in diameter. The supply of the gas to the front or rear GHe cryosurfaces could be distributed by externally operated valves.

The LHe was made in the Aerospace Environmental Facility. A 30-liter/hr He liquefier in conjunction with a 4-kw GHe refrigerator was used as a precooler for the gas. A 1000-liter dewar located on top of the chamber housed a Joule-Thompson valve for the final stages of liquefaction and storage of LHe.

SECTION III
TEST ARTICLES

3.1 1-LB-THRUST SCALED THRUSTER

The 1-lb-thrust Manned Orbiting Laboratory (MOL) scaled thruster used in the test was supplied by McDonnell Douglas. The bipropellant, monomethylhydrazine and nitric oxide (MMH-N_2O_4), thruster (Fig. 2) was designed for both steady and pulsing operation. The performance of the engine was investigated by Marquardt Corporation personnel who found that the lower thrust level resulted in lower combustion efficiency and pulsing performance. During the firing, the propellant valves and injectors were held at 60°F with cooling water.

The thruster design parameters and performance are shown below.

Thrust	1.0 lb
Fuel	MMH (Monomethylhydrazine)
Oxidizer	N_2O_4
Chamber Pressure	90 psi
Mixture Ratio	1.65 ± 1.5
Nozzle Expansion Ratio	40:1
Nozzle Geometry	Contoured
Chamber Temperature	4000°F
Throat Diameter	0.090 in.
Nozzle Exit Diameter	0.569 in.
Combustion Efficiency	0.830

Shown in Fig. 2 is the assembled 1-lb thruster. It consists of a single-doublet water-cooled injector head, high response solenoid valves, and two 5-micron nominal filters upstream of each valve. The nozzle and combustion chamber are an integral part, machined from molybdenum.

Figure 3 shows the 1-lb thruster propellant system. The system consists mainly of three parts: the engine nitrogen purge, high-point bleeds, and propellant supply system. Each propellant tank has a capacity of 2 liters. The propellants were pressurized with dry nitrogen. The propellant line was 0.180-in.-ID stainless steel tubing.

3.2 TEST PLATE

The test plate which was supplied by McDonnell Douglas Corporation (MSSD) was 30 by 32 in. and 1 in. thick. The all-welded honeycomb sandwich plate was constructed from stainless steel. The faces of the plate are 0.017 in. thick and the core thickness is 0.0025 in.

SECTION IV
TEST INSTRUMENTATION

The pitot rake (Figs. 4, 5, and 6) consisted of nine 0.25-in.-OD pitot tubes which were 10 in. long. The middle tube was positioned on the centerline of the thruster. The remaining eight tubes were positioned at angles relative to the thruster axis. Each pitot tube was connected to a Baratron® pressure transducer which was mounted on the rake. The rake was designed so it could be mounted in both vertical and horizontal positions. The rake was attached in either of the positions to a slide mechanism so it could be moved axially and laterally with respect to the thruster. The slide mechanism was manually operated from outside the chamber. The nine Baratrons had a maximum range of 10 torr. The response time of the pressure transducers in this installation was less than a second. There was a copper-constantan thermocouple mounted on each of the nine pitot probes. They were positioned 1 in. from the lip of the probe.

There were four Veeco® ionization gages located at various positions in the chamber. One gage was located behind the thruster for measuring the pressure altitude during firing. The remaining gages were located near the LHe cryopump in order to evaluate its performance. One Alphatron® gage was installed behind the thruster to measure pressures above 10^{-3} torr.

The thruster was instrumented with three Taber® 500-psia pressure transducers and two Potter® flowmeters with a range up to 3 gm/sec. The response time of the flowmeters is one second.

Figure 7 shows the data system which was used during the test. The pitot pressures, temperatures, and engine flow rates went into signal conditioning equipment, a commutator, analog-to-digital converter, digital tape, and then to the computer and/or data printout. In addition, for rapid analysis, the engine pressures and flow rates were recorded on an oscillograph.

SECTION V
PROCEDURE

5.1 PROPELLANT SYSTEM PREPARATION

The propellants were specially filtered until they met the cleanliness levels specified. Particle counts were checked before the propellants were loaded into the system to ensure that the specified propellant cleanliness was maintained.

5.2 TEST PROCEDURE WITHOUT PLATE

There were four firings with the pitot rake mounted in a vertical position and four in the horizontal position. For both of the positions, the pitot rake was located at 19.75 and 30.5 in. from the nozzle exit along the thruster centerline. The eight tests were conducted at simulated altitudes between 350,000 and 400,000 ft. The test conditions for each run are listed in Table I, Appendix II.

In preparation for firing, the chamber was pumped down with the mechanical and diffusion pumps, and all cryosurfaces except the LHe-cooled surfaces were cooled to operating temperatures. Immediately before firing, the LHe cryopump was cooled to 4.2°K. When this surface was cold, the chamber pressure was 1×10^{-7} torr. The engine was then fired for periods ranging from 46 to 304 sec. Between firings the pitot rake was moved to another axial position. A check firing was conducted at each position in order to check the repeatability of data.

The chamber was brought to atmospheric pressure in order to position the rake in an orientation 90 deg relative to the previous rake location, and the test sequence was repeated with the rake rotated.

5.3 TEST PROCEDURE WITH PLATE

There was a total of 16 firings with the pitot rake and plate mounted in a vertical plane. The thruster axis was mounted parallel to and four nozzle radii (1.14 in.) from the surface of the plate. The pitot rake was positioned manually at axial distances of 15 and 23.25 in. downstream from the nozzle exit plane. At each of these axial stations, the pitot rake was moved from 0.5 to 5.0 in. from the surface of the plate in various increments between the firings. Each of the thruster firing durations was approximately 11 sec. Four pulse firings were made during this test to evaluate chamber performance under pulse firing conditions. The pulse width, pulse delay, and pulse number were varied as shown in Table I.

SECTION VI
RESULTS AND DISCUSSION

Figure 8 shows typical engine combustion chamber pressure, oxidizer-to-fuel ratio, and oxidizer flow rate during engine firing. These quantities are practically constant after initial transients.

Figure 9 shows the predicted run time and altitude capability of the ARC 8V cryosystem as functions of engine thrust level. The data point for the 1-lb engine is plotted. Figure 10 shows measured chamber pressure versus time for a typical 1-lb engine firing. The continuous rise in chamber pressure is caused by the thermal loading of the front GHe cryopump (20°K).

6.1 WITHOUT PLATE

Plotted in Figs. 11 and 12 is impact probe pressure versus distance from the nozzle centerline at axial positions of 19.75 and 30.5 in., respectively.

Figure 13 shows the calculated (Ref. 5) equilibrium expansion solution for Mach number contours versus axial distance from the engine exit plane. The experimental Mach numbers obtained from the ratio of impact pressure to combustion chamber pressure are plotted for both vertical and horizontal positions. The average specific heat ratio, γ (1.25), used for calculating experimental local Mach numbers from the impact probe pressures, was obtained from the computer solution. The experimental Mach numbers near and on the centerline of the thruster agree very well. The deviation of the Mach numbers away from the centerline is larger, however, because of the difficulty of making accurate pitot readings in this pressure range, and this is within the expected accuracy of the measurements.

6.2 WITH PLATE

Plotted in Figs. 14 and 15 is the total pressure above the surface of the plate at axial positions along the surface of $x = 15.0$ in. and $x = 23.0$ in., respectively. Each impact probe pressure of the rake is plotted versus distance above the surface of the plate. In both Figs. 14 and 15 the pitot pressure increases with increasing distance from the plate. At $x = 15$ in. (Fig. 14) the pressure reaches a maximum at a probe height of about $z = 3$ in. and then drops off, approaching the reading obtained with no plate in the flow. At $x = 23.25$ in. no maximum pitot pressure was reached with the probe 5 in. from the surface. Since the area of particular interest was near the surface of the plate, readings were not recorded when the probe was farther than 5 in. from the plate. It is possible however, particularly from Fig. 15, to see that the boundary layer of the flow along the plate is quite thick; its estimated outer limit is indicated on the figures.

The flow field, which is influenced by the plate, is a rather complicated mixture of shock patterns, boundary layer, and continued expansion of the gas. Some investigations reported in Refs. 4 and 6 which are consistent with the current investigations indicate that there is an impingement pattern on the plate as illustrated in Fig. 16. The top view shows the interaction region on the plate, and the bottom view shows the zone of influence which the plate produces on the undisturbed plume along the nozzle centerline.

SECTION VII
CONCLUSIONS

The total pressure survey by the pitot rake in the plume without the plate indicated that the plume was reasonably axisymmetric, and the Mach numbers obtained from these pressures agreed with the analytical values which had been predicted by the equilibrium method of characteristics calculations.

The plume impingement on the plate resulted in a shock and boundary-layer region. The boundary-layer region was about 4 in. thick 23.25 in. from the nozzle exit.

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APPENDIXES
I. ILLUSTRATIONS
II. TABLE

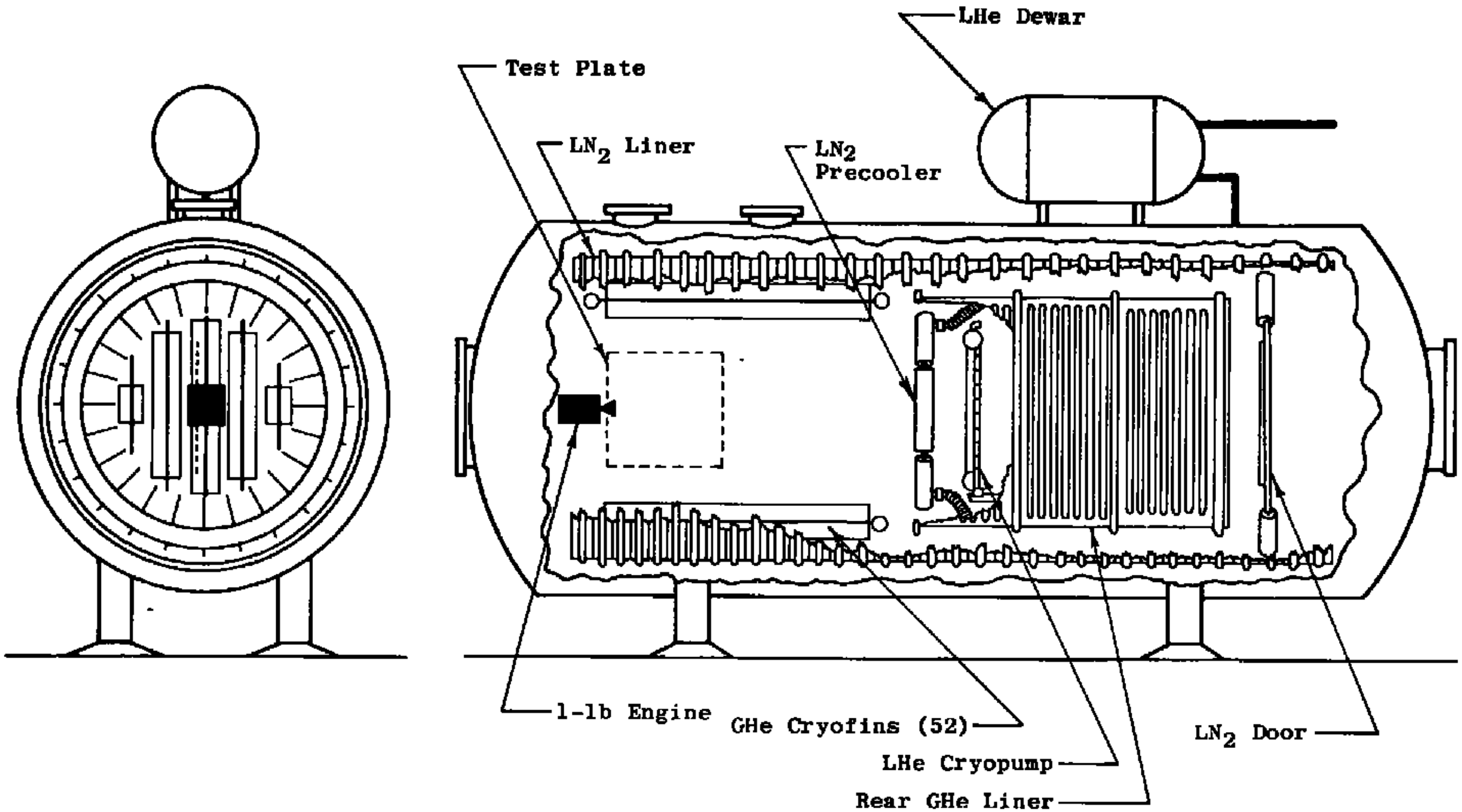
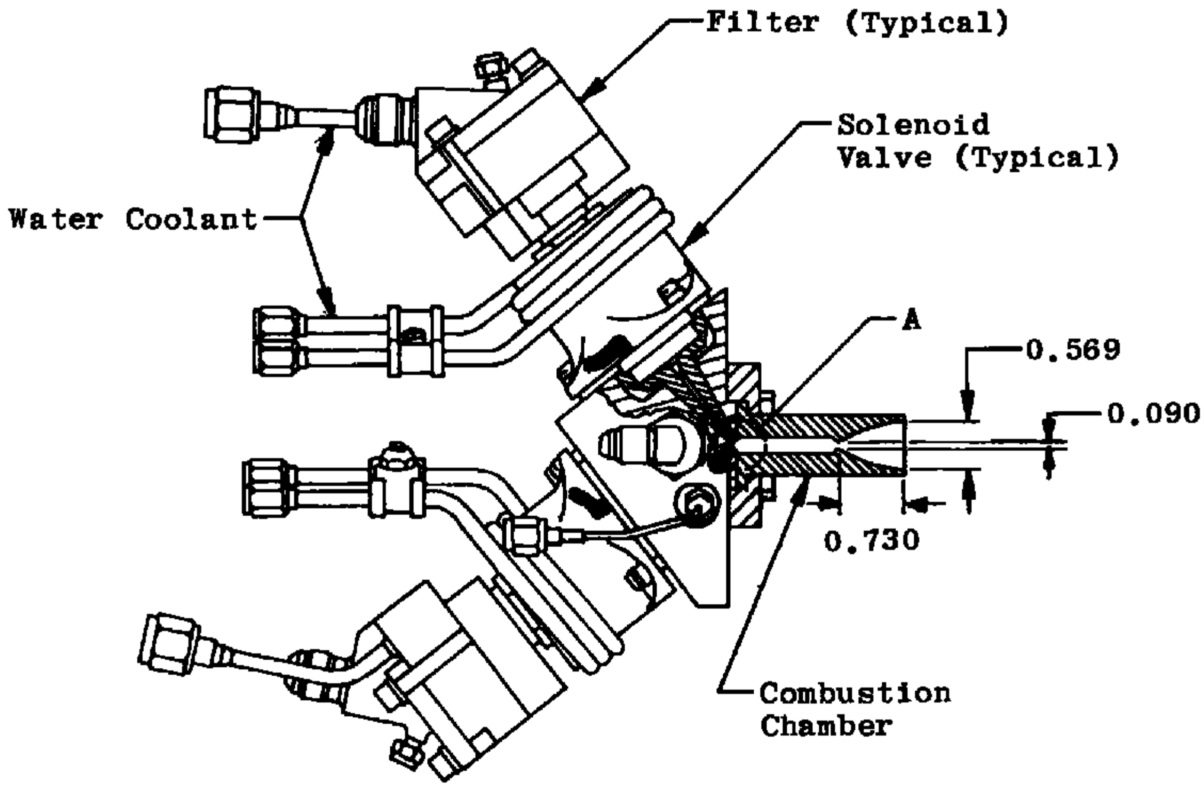
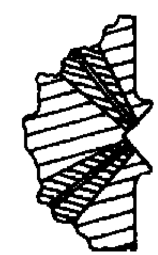


Fig. 1 Aerospace Research Chamber (8V)





Detail A
Injectors

Fig. 2 One-Pound-Thrust Engine and Valves

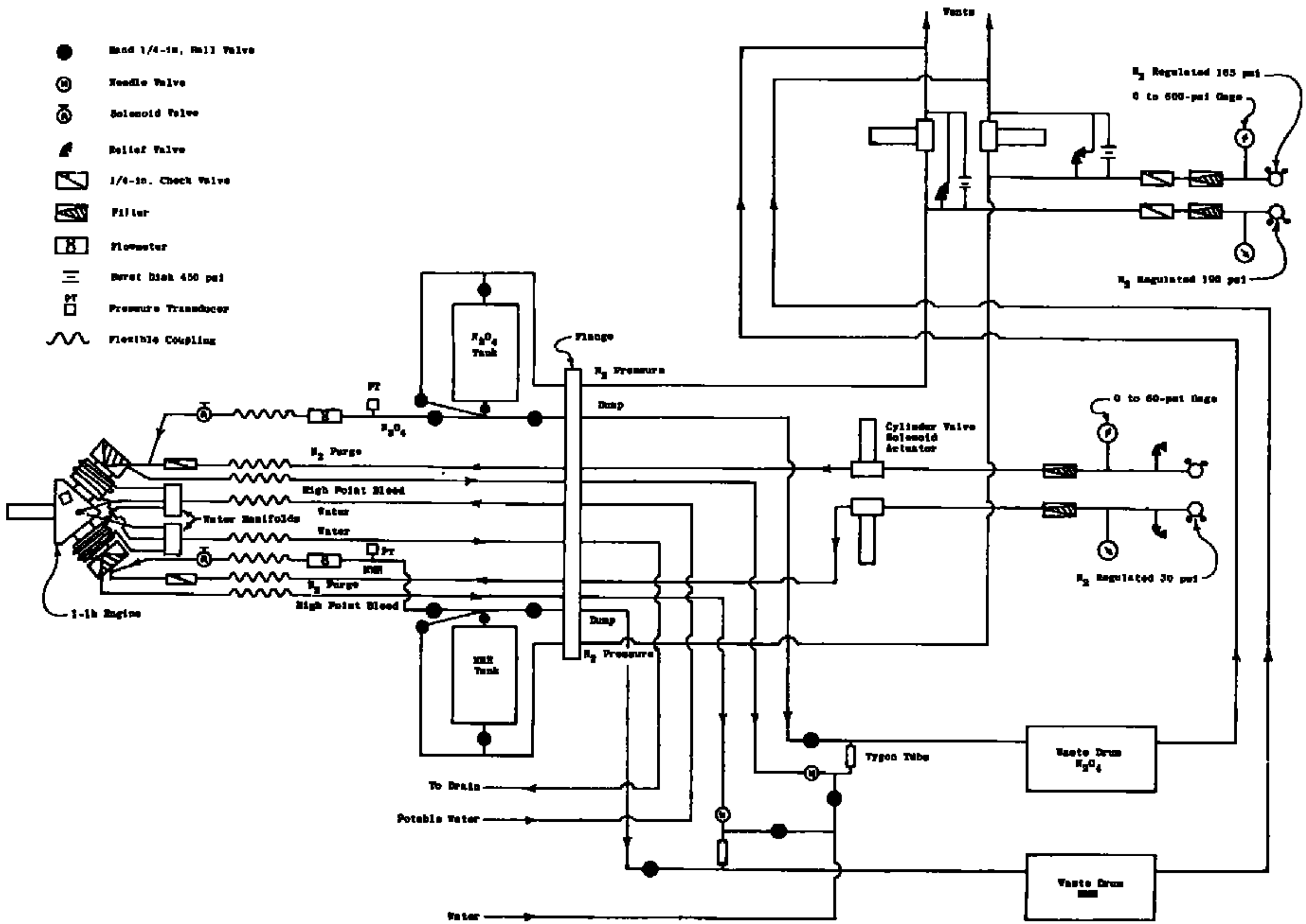


Fig. 3 Propellant System

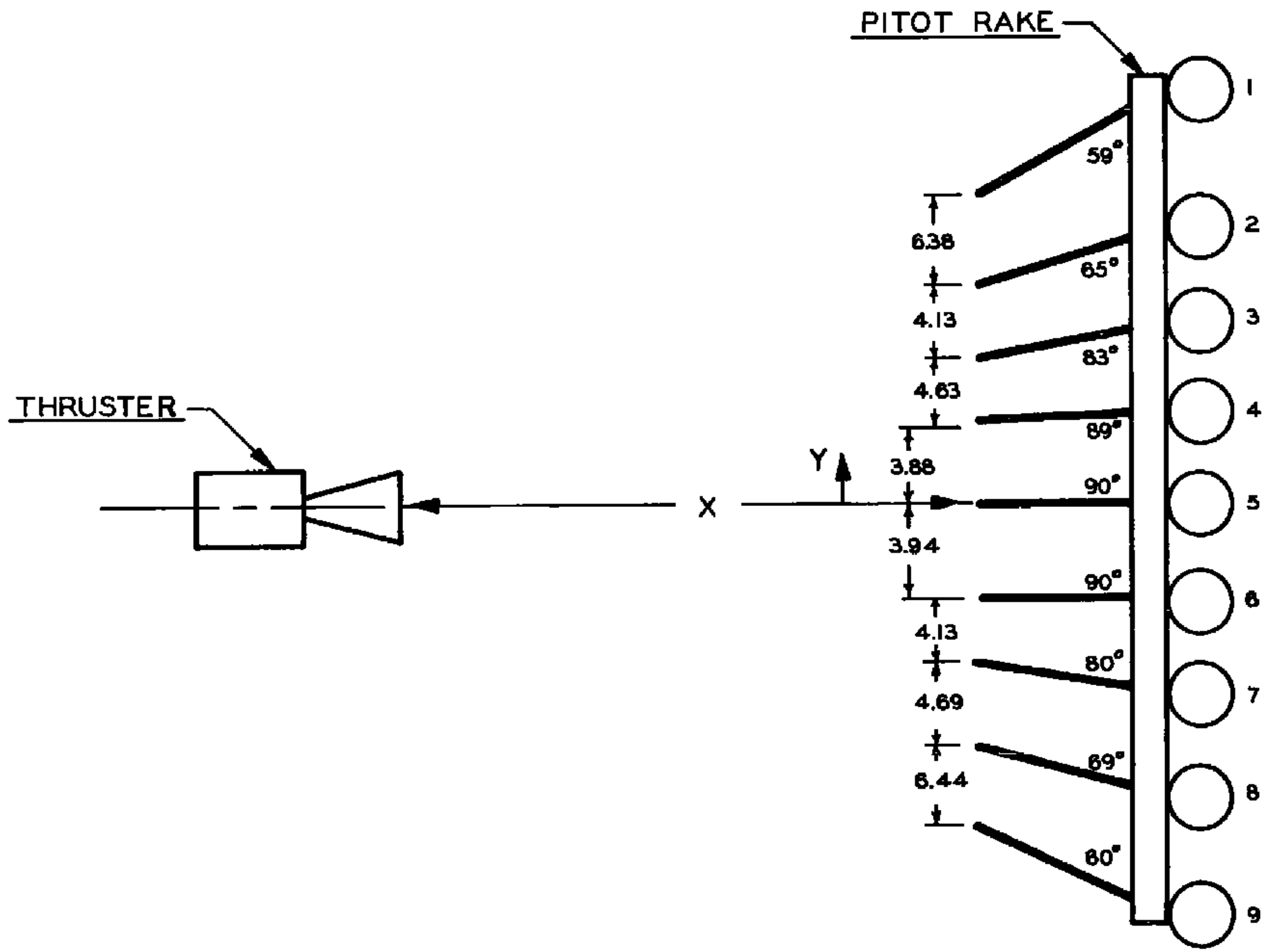


Fig. 4 Pitot Probe Rake without Plate

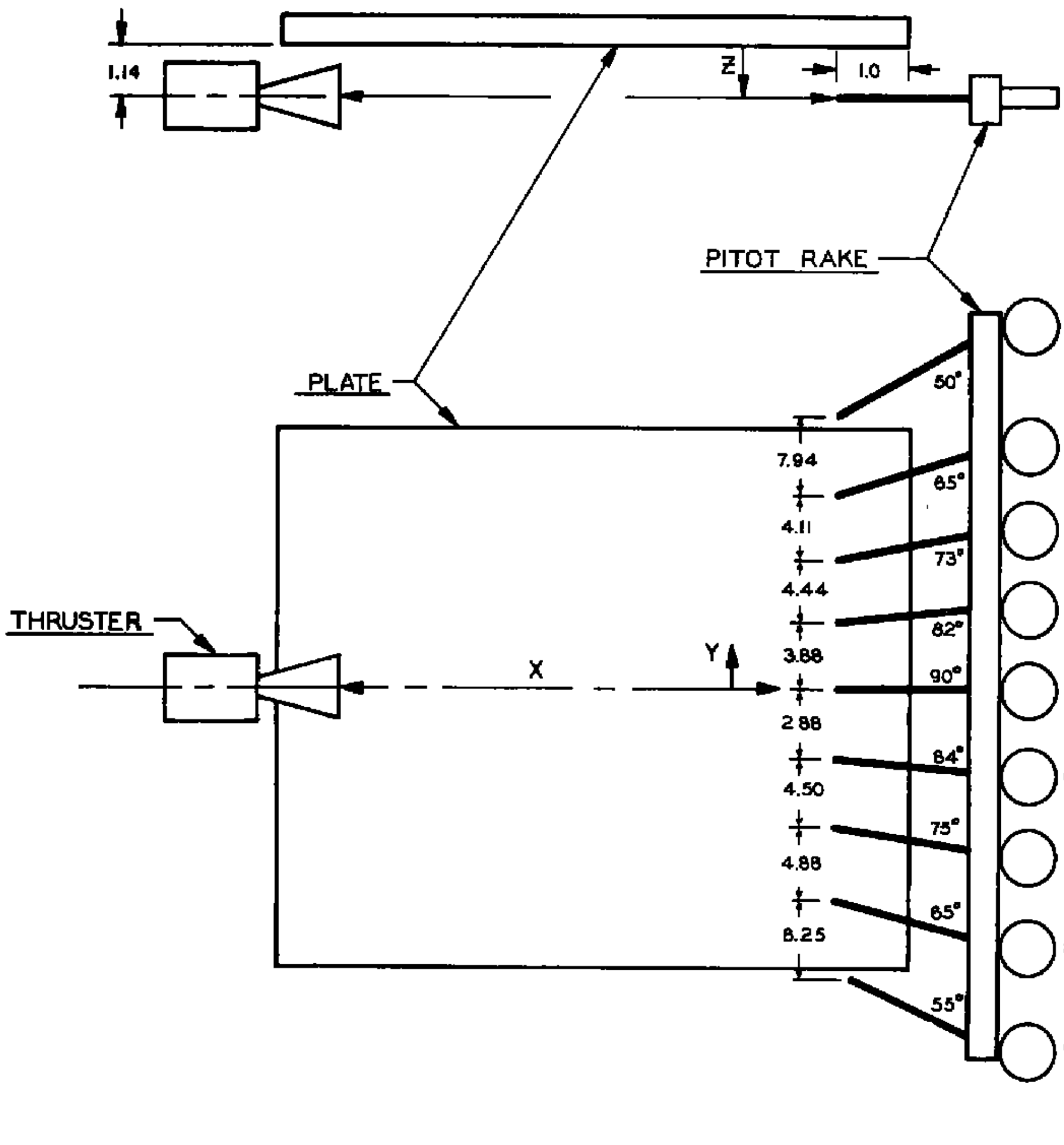


Fig. 5 Pitot Probe Rake with Plate

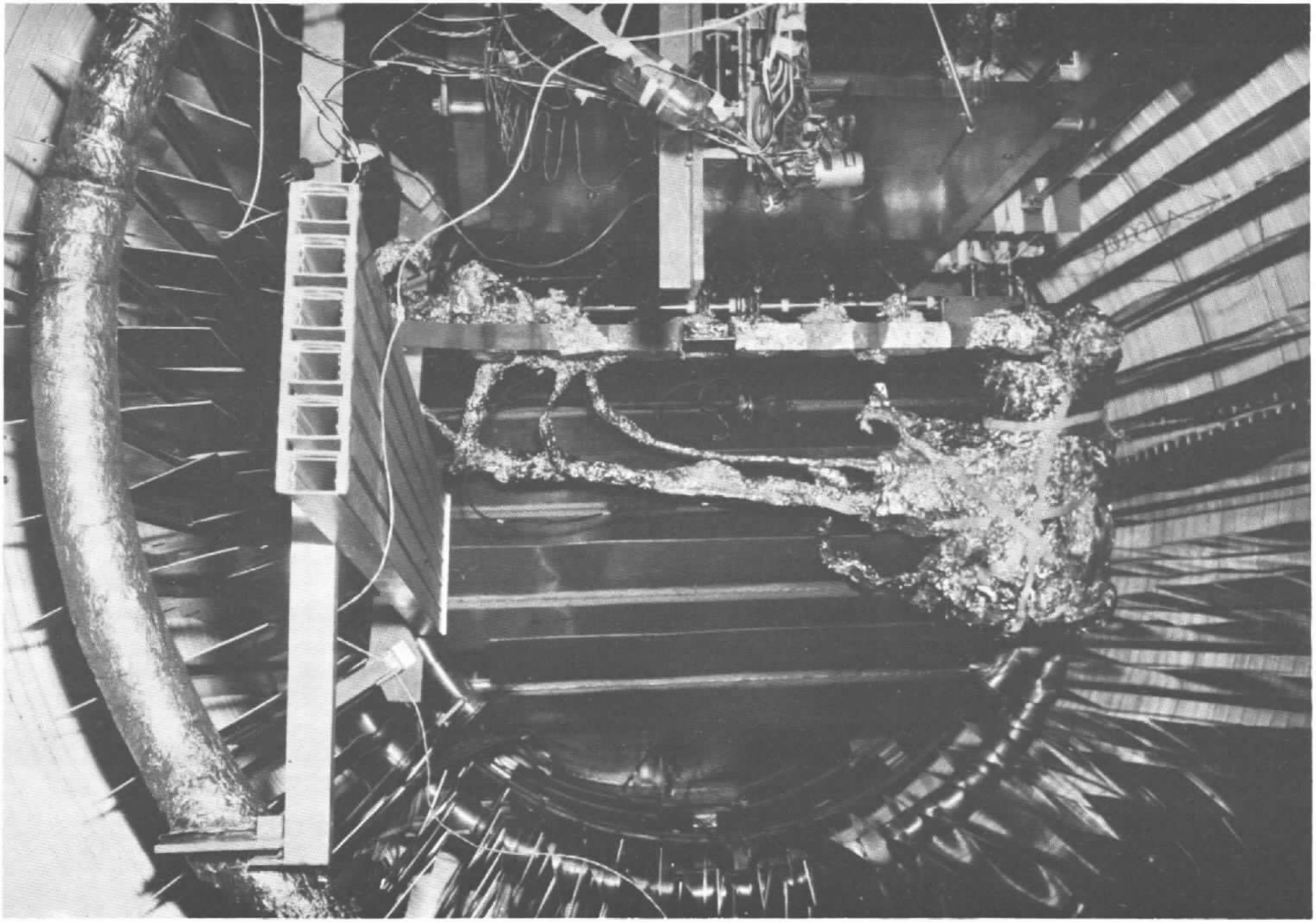


Fig. 6 Test Installation - Thruster, Plate, and Pitot Rake

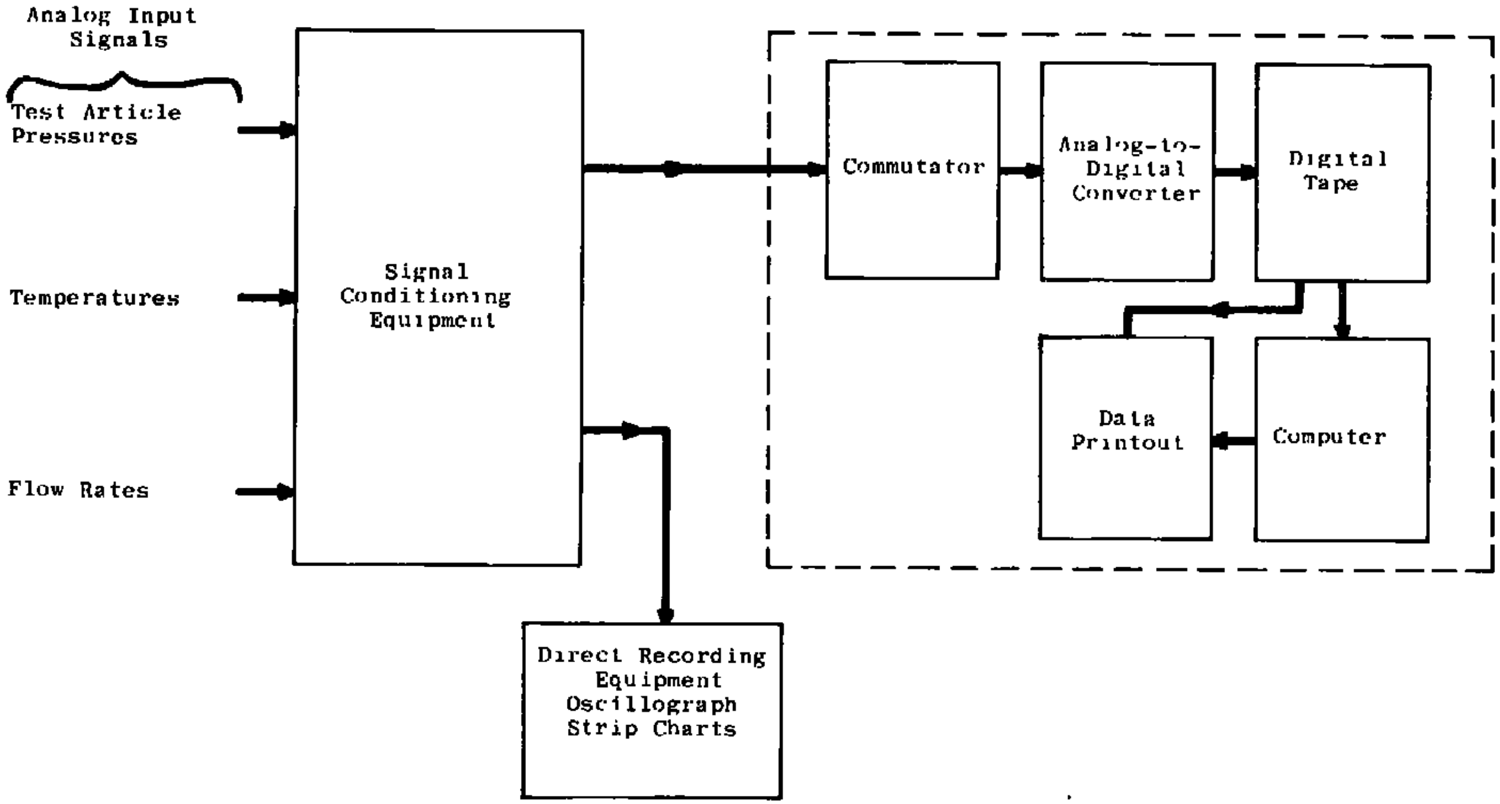


Fig. 7 Test Data System

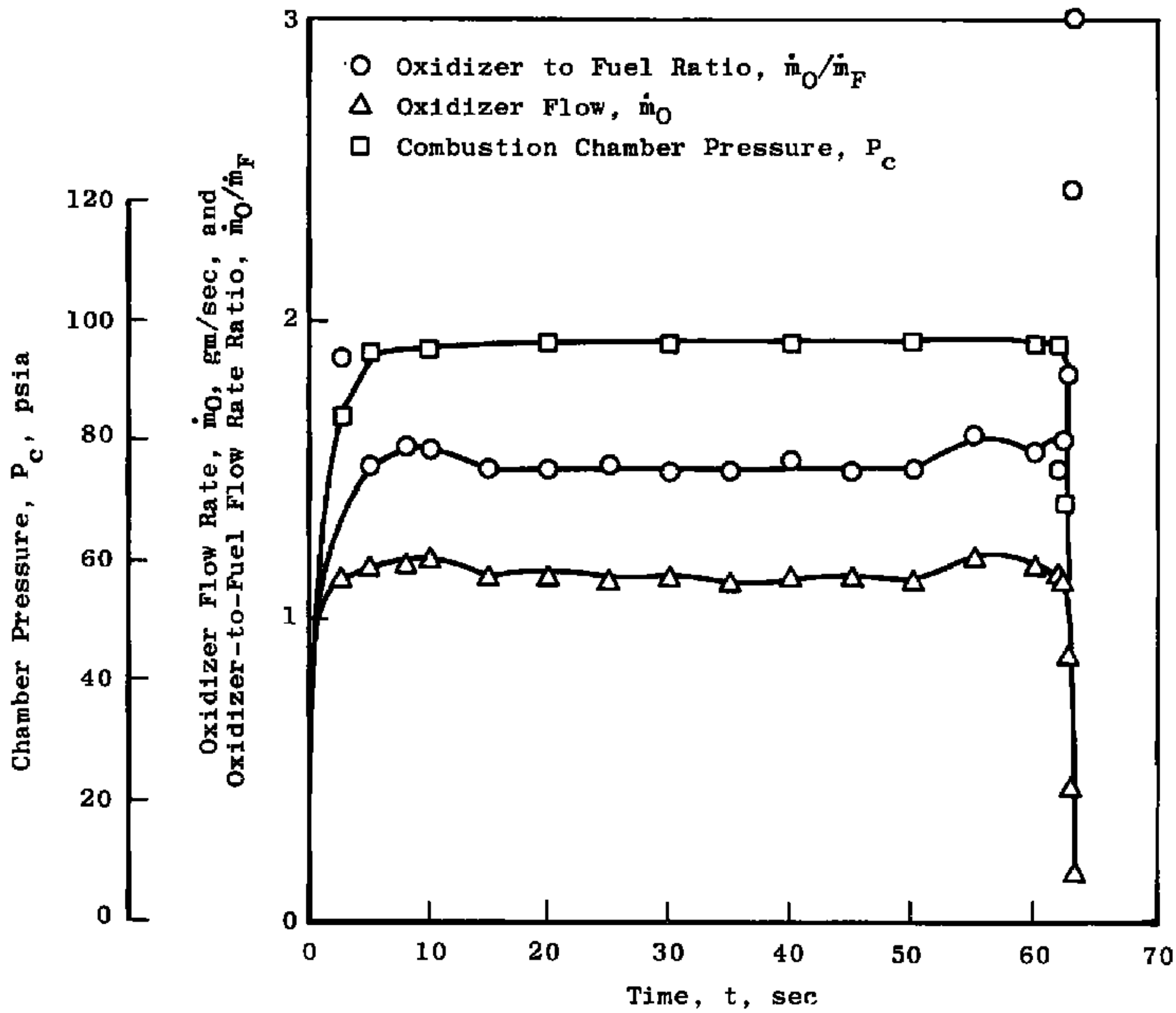


Fig. 8 Engine Pressure and Flow Rates

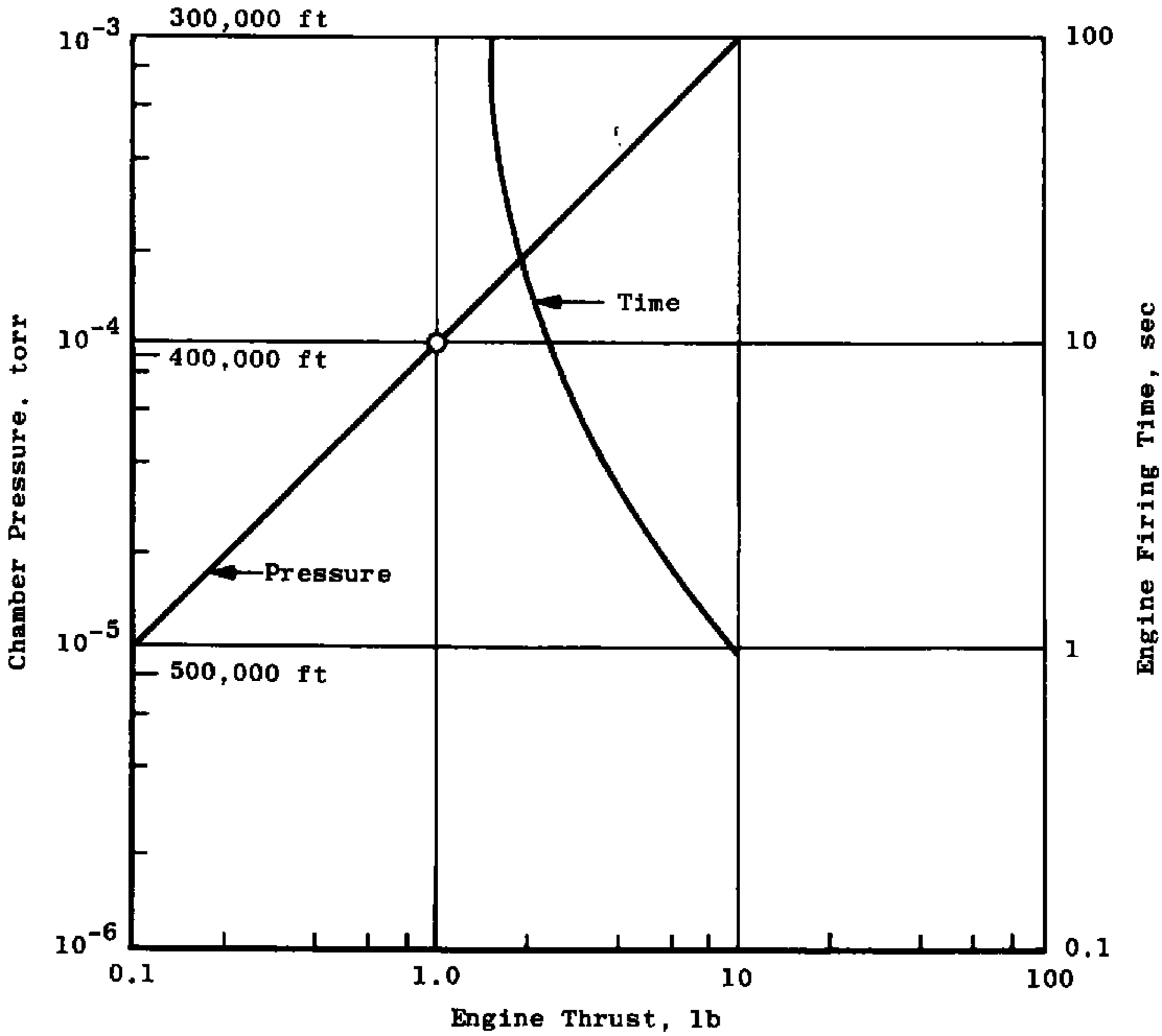


Fig. 9

Fig. 9 Predicted ARC 8V Chamber Performance

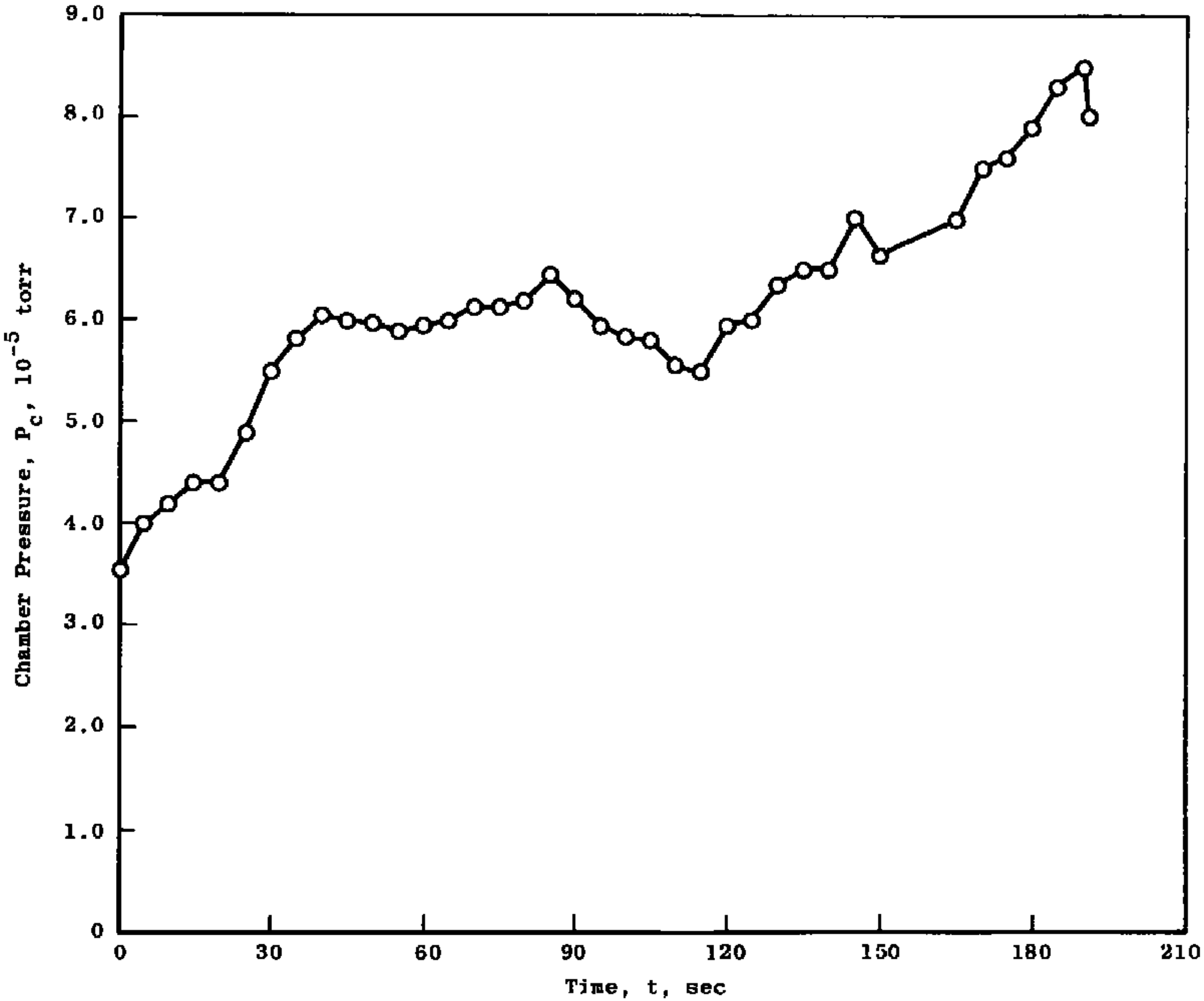


Fig. 10 ARC 8V Chamber Pressure versus Engine Firing Time

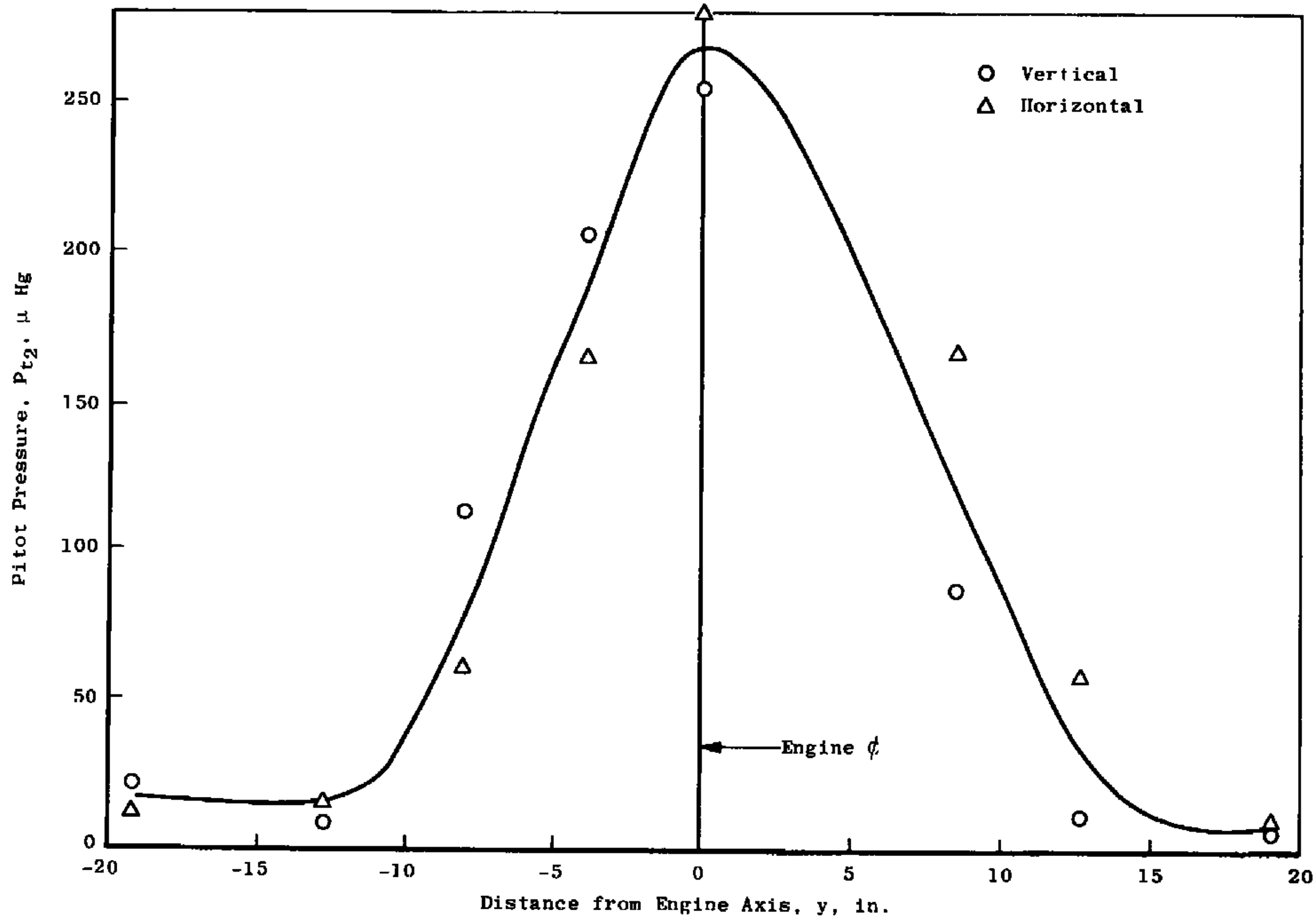


Fig. 11 Total Pressure Profile in Plume at x = 19.75 in.

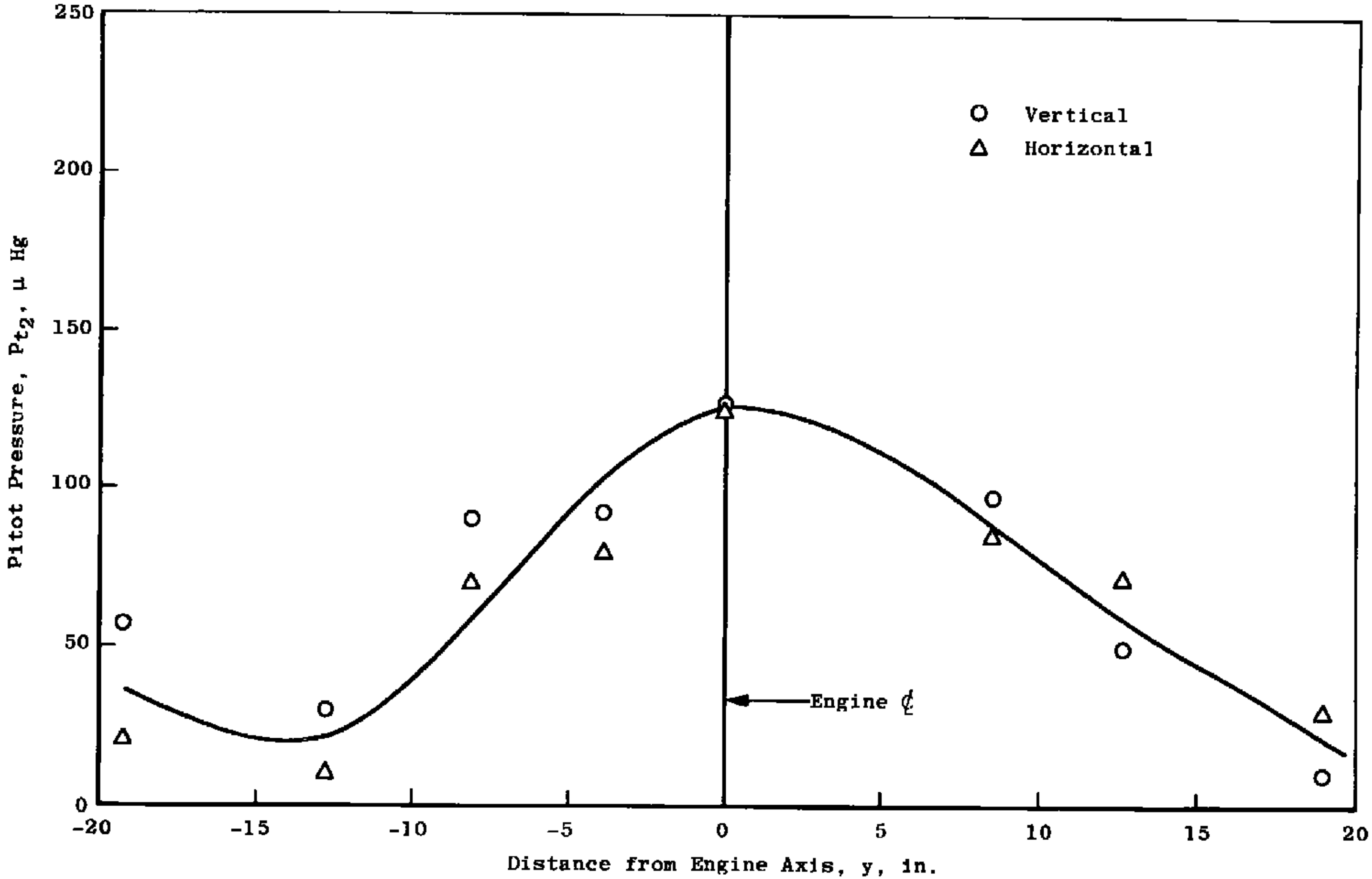


Fig. 12 Total Pressure Profile in Plume at x = 30.5 in.

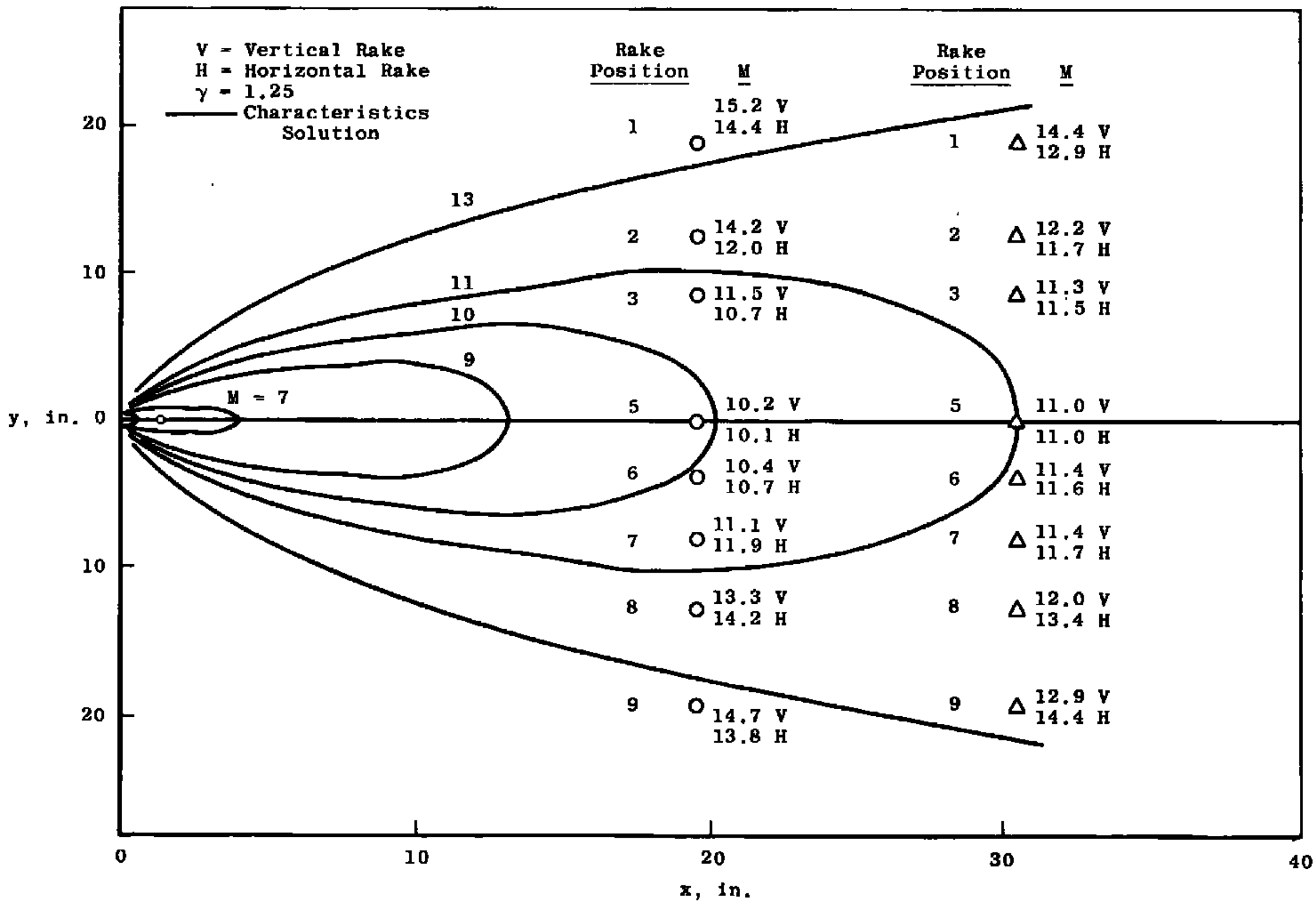


Fig. 13 Mach Number Contours for 1-lb-Thrust Engine

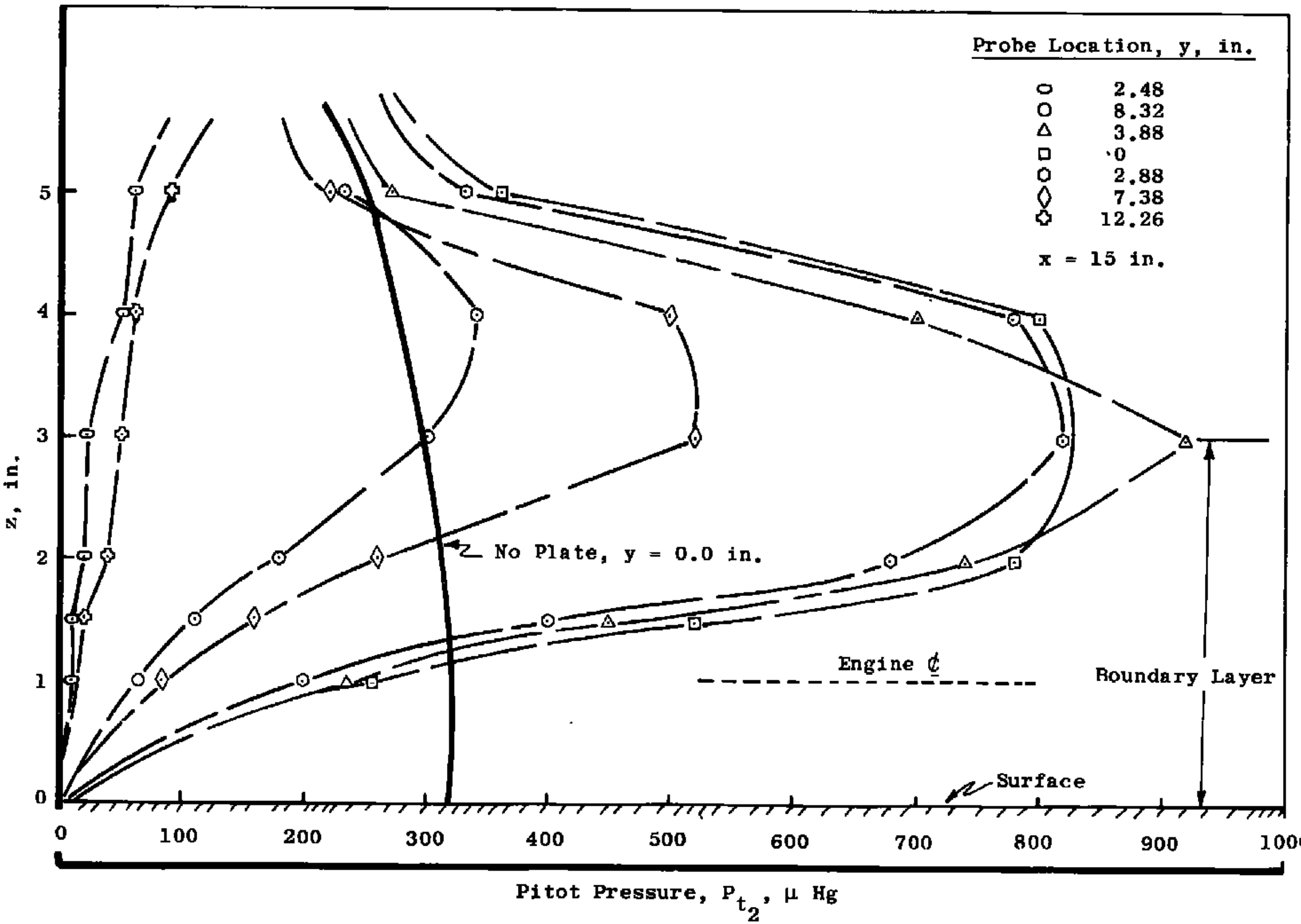


Fig. 14 Total Pressure Survey above Panel Surfaces at x = 15.0 in.

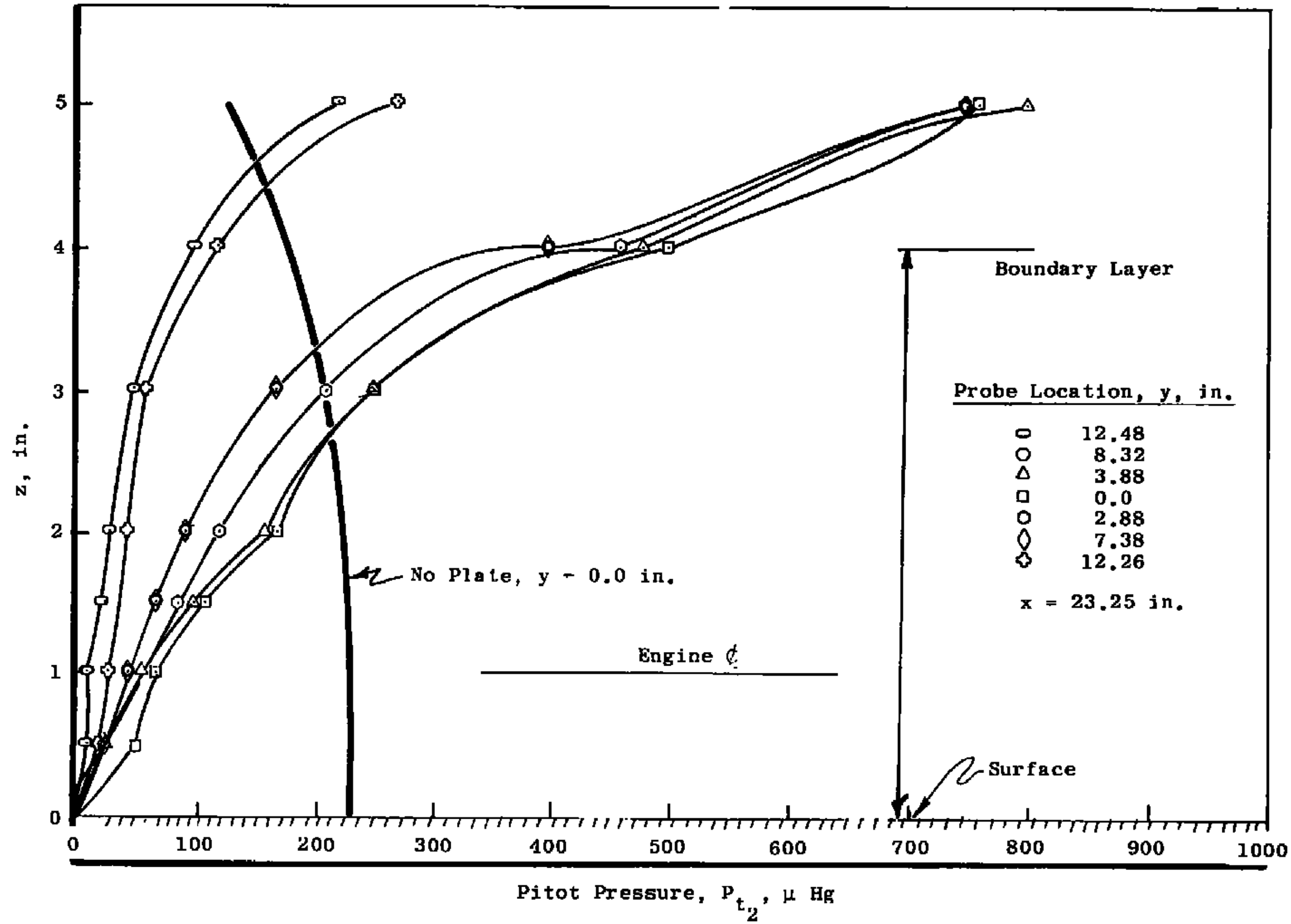


Fig. 15 Total Pressure Survey above Panel Surfaces at x = 23.25

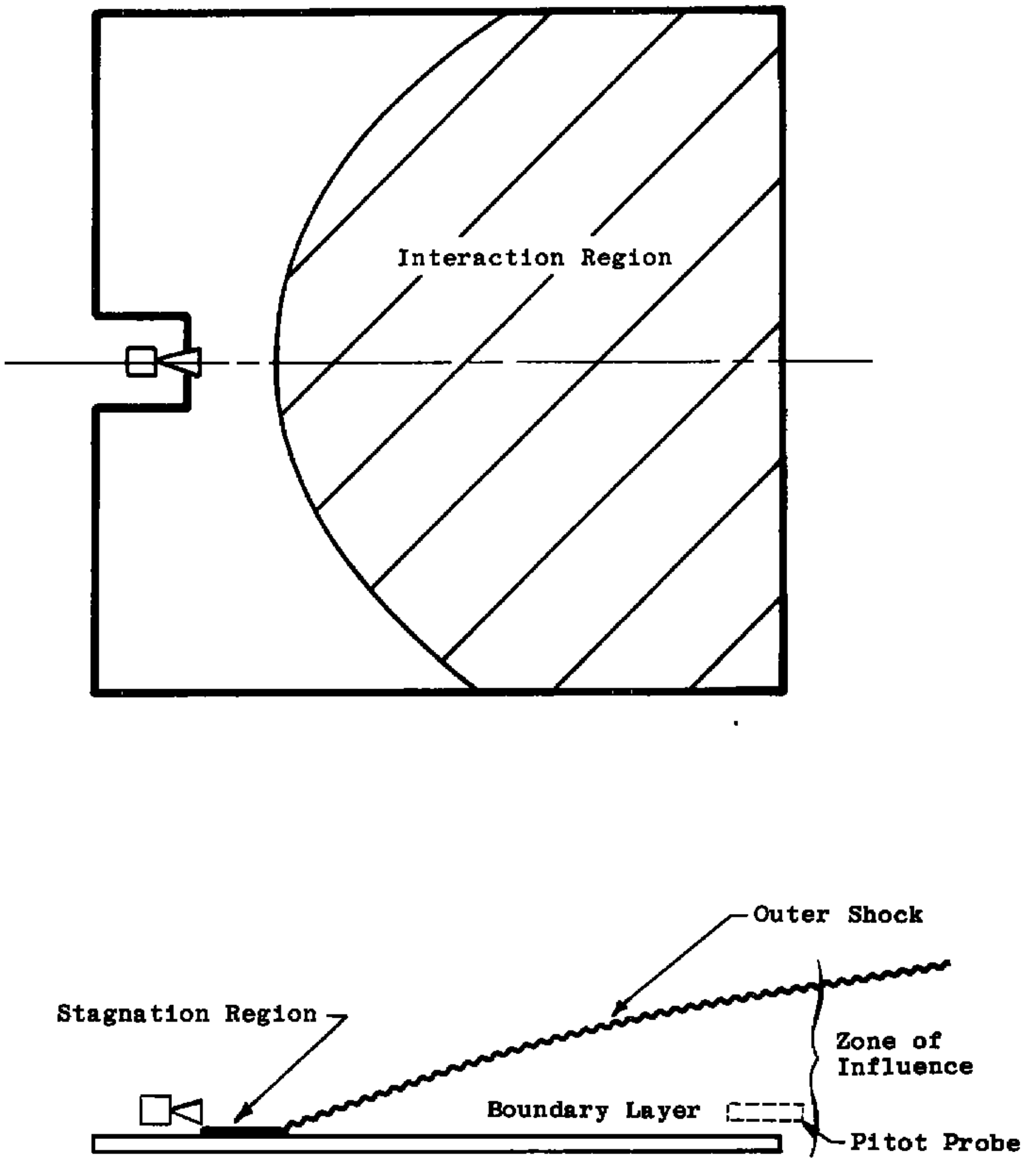


Fig. 16 Impingement Pattern on Plate

TABLE I
TEST LOG

Run No.	Run Time, sec	P_F, psia	P_O, psia	$\dot{m}_O$, gm/sec	$\dot{m}_F$, gm/sec	P_c, psia	O/F	x, in.	z, in.	Pitot Probe Rake Position, in.
1	61	125	153	1.16	0.68	87	1.71	19.75	1.14 (Thruster Centerline)	Vertical
2	60	125	↓	0.80	0.68	88	1.18	30.5	↓	↓
3	59	125		0.50	0.71	89	0.705	19.75		
4	60	126		0.60	0.69	89	0.87	30.5		
5	46	124	154	1.00	0.76	96	1.31	19.75		Horizontal
6	60	124	153	1.14	0.75	96	1.52	30.5		↓
7	304	123.5	153	1.12	0.76	97	1.47	19.75		
8	60	↓	153	1.13	0.76	96	1.49	30.5		
9	10		149	1.24	0.69	86	1.80	15.0	1.0	Vertical Rake with Flat Plate
10	11		↓	1.29	0.69	↓	1.87	↓	1.5	↓
11	↓			1.30	0.68		1.88		2.0	
12		123		1.25	0.69		1.81		3.0	
13		123		1.24	0.69		1.80		4.0	
14	12	123		1.29	0.70		1.84		5.0	
15	11	123.5		1.29	↓		1.84	23.25	0.5	
16	11	123.5		1.28			1.83	↓	1.0	
17	13	123		1.29		85	1.84		1.5	
18	11	↓		1.25		85	1.79		2.0	
19	151			1.27	0.69	87	1.84		3.0	
	Pulse Width/Delay/No.									
20	50/ 50/10								4.0	
21	50/ 500/10								5.0	
22	50/ 500/10								↓	
23	50/1000/10									
24	50/2000/10									

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13. ABSTRACT

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attitude control						
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plumes						
rocket exhausts						
altitude simulation						