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SUPERSONIC COMBUSTION APPARATUS

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SUPERSONIC COMBUSTION APPARATUS

William A. Strauss

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FOREWORD

This technical report was prepared by W. A. Strauss and R. Edse of The Department of Aeronautical and Astronautical Engineering, The Ohio State University on Contract No. F33615-70-C-1483. This investigation was performed during the period 1 July 1970 through 31 March 1972 and represents the initial phase of the study, "The Investigation of Properties of Gas Flows in Variable Area Ducts With Heat Addition." This work was supported by the Air Force Aero Propulsion Laboratory (AFSC), Ramjet Engine Division, Wright-Patterson Air Force Base, Ohio with Mr. P. J. Hutchison as Project Engineer.

Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

ABSTRACT

An apparatus to study the performance of supersonic combustion chambers was designed, constructed and checked out. However, the main airflow heater (which was supposed to be available for use in this program) was not capable of operating at the original combustor design specifications. Therefore, the tests to check out the equipment had to be made with lower air mass flows. Results of preliminary tests in a constant area combustor with hydrogen fuel are reported.

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LIST OF SYMBOLS

A = area

A_w = wetted surface area

D = burner diameter

D_H = hydraulic diameter

M = Mach number

NE = nozzle exit

Nu = Nusselt number

R = burner radius

Re_D = Reynolds number (characteristic length = diameter))

T = static temperature

T° = stagnation temperature

T_w = wall surface temperature

V = velocity

f = friction factor

h = penetration depth of injected gas

h_c = convective heat transfer coefficient

k_f = heat conduction coefficient

$\dot{m}$ = mass flow

p = static pressure

p° = stagnation pressure

x = distance from point of injection

ϵ = constant value

γ = ratio of specific heats

θ = jet injection angle (from normal)

ρ = density

LIST OF SYMBOLS (Continued)

μ = viscosity

δq_w = heat transferred through wall

δq_A = heat added to air flow

Subscripts

air = air

H_2 = hydrogen

j = injected jet

NE = nozzle exit

SECTION I

INTRODUCTION

Supersonic flow combustors are considered to be necessary for use in hypersonic-speed aircraft engines.[1] However, the mixing of the fuel with the air and the mode of combustion in the supersonically flowing air are the major problems of this type combustor.[1] The mixing and combustion must occur so maximum combustion efficiency is obtained while the stagnation pressure loss is the smallest possible and the length of the combustor very short. Recently, Dobrowolski[2] and Billig[3] have analysed the problem of heat addition in variable area supersonic ducts. These authors used the Crocco Relation[4] $\left(pA^{\epsilon/\epsilon-1} = \text{const, where } \epsilon = \text{constant}\right)$ to facilitate solution of the governing flow equations. Large amounts of heat per unit mass of air can be added to initially supersonic flows (and still maintain supersonic flow) by the use of diverging area burners. These authors have shown that critical conditions in diverging ducts can also be reached (minimum stagnation pressure loss for the amount of heat added). Billig[3] conducted a series of experiments with a conical combustor (1.58° half angle, 23 in. long) using hydrogen as a fuel and a M = 3.0 air flow; he reported good agreement between experiment and theory. Considerable experimental work has been done on the problem of combustion and mixing in a supersonic air flow. Of particular interest are combustors employing sidewall fuel injection and thermal ignition of the mixture. Various kinds of heaters (arc, pebble bed, chemical) have been used to produce the high enthalpy air needed for this type experiment. Almost all heaters generate small amounts of gases which do not occur in normal air but which may influence the rate of combustion of the propellants.

The purpose of this investigation was to design, construct and test an apparatus for determining the combustion efficiency of a supersonic combustion chamber (air and hydrogen heated by electric resistance heaters). The hydrogen is injected at the sidewall of the combustion chamber and expected to ignite spontaneously.

SECTION II

APPARATUS

A diagram of the apparatus is shown in Fig. 1. This apparatus consists of the air flow system, fuel flow system, fuel injectors, combustor and the exhaust system. Details of each of these systems are described in the following paragraphs.

A. AIR FLOW SYSTEM

The air was compressed by two Type XVH Ingersoll-Rand 4000 psig air compressors and was stored in two 750 cu. ft. tanks at a pressure of 2400 psig. Oil, introduced into the air by the compressors, was removed with a combination Swirl-Type oil separator and Type HPF Lectrodryer Filter. The oil concentration in the air is reduced to less than 1 ppm with this equipment. Thereafter, the air is dried with a Type PB Lectrodryer Dehumidifier which reduces the water concentration in the air to less than 0.01 ppm. The air flow to the combustion chamber was controlled with two Hammel-Dahl diaphragm-operated throttle valves. These valves were controlled automatically with a Taylor Instrument two-mode pressure controller and maintained a constant stagnation pressure for the supersonic nozzle (see Part B of this section). The air flow was passed through a 600 kW electric heater made of #7 gage Kanthal wire and powered by a 1900A, 300V Type TLF General Electric generator. The airflow is straightened in a screened section between the heater exit and the supersonic nozzle stagnation chamber. The air stagnation temperature was measured in this flow straightening section with a platinum-platinum 10 percent rhodium thermocouple. The air stagnation temperature at this point was held constant (generally 1950°F ± 20°F) with a Minneapolis-Honeywell controller which controls the generator field.

The combustor was originally designed to operate at a mass flow of 0.866 lbm/sec and a temperature of 1950°F. However, difficulties with the heater required that it be operated at lower mass flows in order to reduce the current densities (discussed in Section VI) in the heating elements.

B. AIR FLOW NOZZLE

The air flow nozzle was designed by the method of characteristics according to the Cohen and Reshotko laminar boundary layer method and using a computer program developed by Petrie.[5] The nozzle was designed for $M = 1.67$ flow and had an exit diameter and throat diameter of 1.3874 and 1.2218 inches, respectively. The boundary layer at the nozzle exit was calculated to be 0.016 inches and the Reynolds number at the throat 3.6×10^5. The copper nozzle was cooled by a light flow of water over the exterior surface.

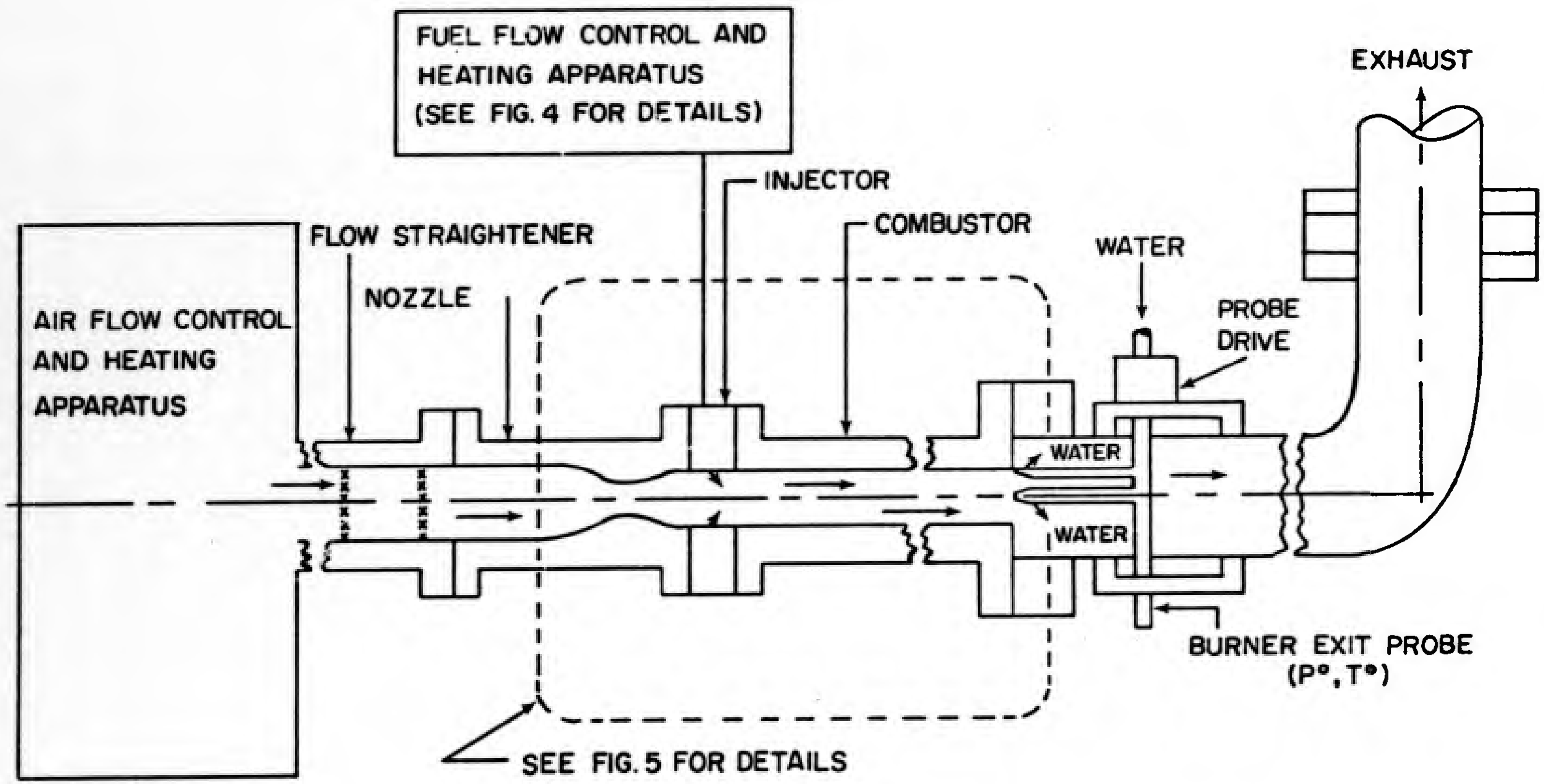


Fig. 1 - Sketch of Supersonic Combustion Apparatus

The stagnation temperature and stagnation pressure profiles at the exit of this nozzle are shown in Figs. 2 and 3, respectively. The nozzle stagnation chamber pressures for these tests were 36.9, 39.7 and 42.1 psig, while the temperatures were 1405, 1605, and 1800°F, respectively. The measured stagnation pressures (corrected for losses through normal shock) indicate that the stagnation pressure decreased only slightly in the nozzle. However, the measured stagnation temperature was approximately 160°F below the chamber value. This low value was caused by heat conduction along the thermocouple sheath (see Section III for thermocouple details). Of particular interest is the profile of the stagnation temperature near the nozzle wall since it provides information on the depth the fuel must penetrate to reach regions of sufficient temperature to produce ignition. According to Fig. 2, a fuel penetration depth of approximately 0.12 inch is required to reach the high temperature gases.

C. FUEL SYSTEM

Commercial grade gaseous hydrogen was used as the fuel for these experiments. A diagram of the fuel flow system is given in Fig. 4. The hydrogen flow was metered with a flow technology milliflow turbine flow transducer. The gas pressure at the inlet to the flowmeter was kept constant with a 4-inch Grove Dome pressure regulator. The rate of hydrogen flow to the combustion chamber was controlled remotely with a Hoke electro-mechanical throttle valve. The hydrogen was heated to approximately 1750°F with a 25 kW electric heater powered by a second 1900A, 300V TLF General Electric d.c. generator and was controlled manually. The heater exit temperature was measured with a platinum-platinum 10 percent rhodium thermocouple.

The flow system had a nitrogen (or air) by-pass (Fig. 4). The air by-pass was used to bring the hydrogen heater up to temperature and the nitrogen by-pass was used to avoid a hydrogen-air interface in the fuel heater. Also, the nitrogen purge was used during emergency shutdown of the system.

D. FUEL INJECTION

Details of the hydrogen injector assembly are given in Fig. 5. The hydrogen was injected at various angles into the airflow through holes in the sidewall of the injector. The injector housing was designed to accommodate various injector rings which allowed the injection conditions to be varied with a minimum of equipment. For the initial phase of this study, three different injector rings were constructed, the details of which are given in the table in Fig. 5. The injector holes in these rings were sized to provide the design hydrogen mass flow while operating in the under expanded mode. This arrangement was used in order to produce significant penetration of the hydrogen into the air flow, thereby promoting mixing.

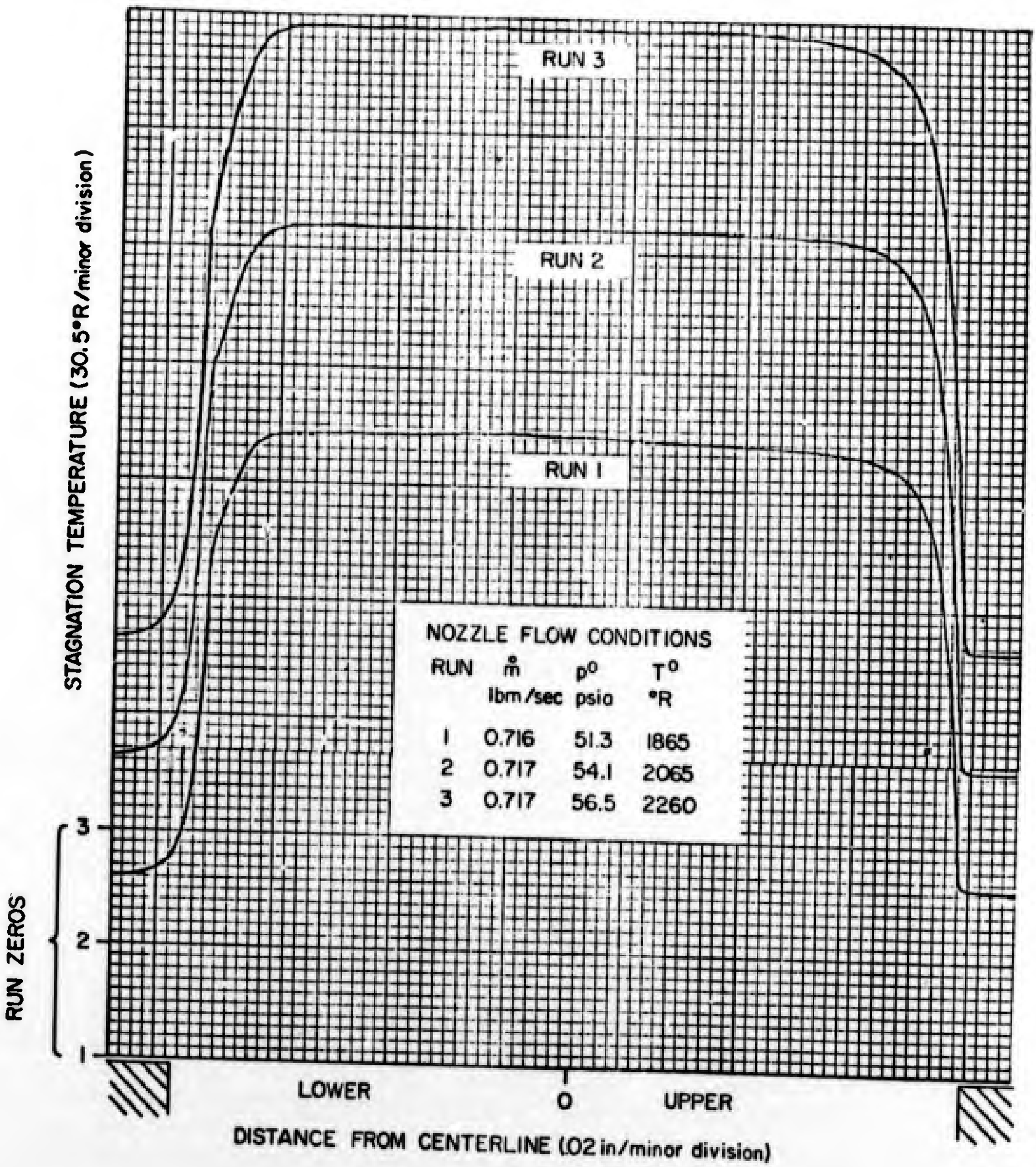


Fig. 2 - Stagnation Temperature at Exit of M = 1.7 Nozzle

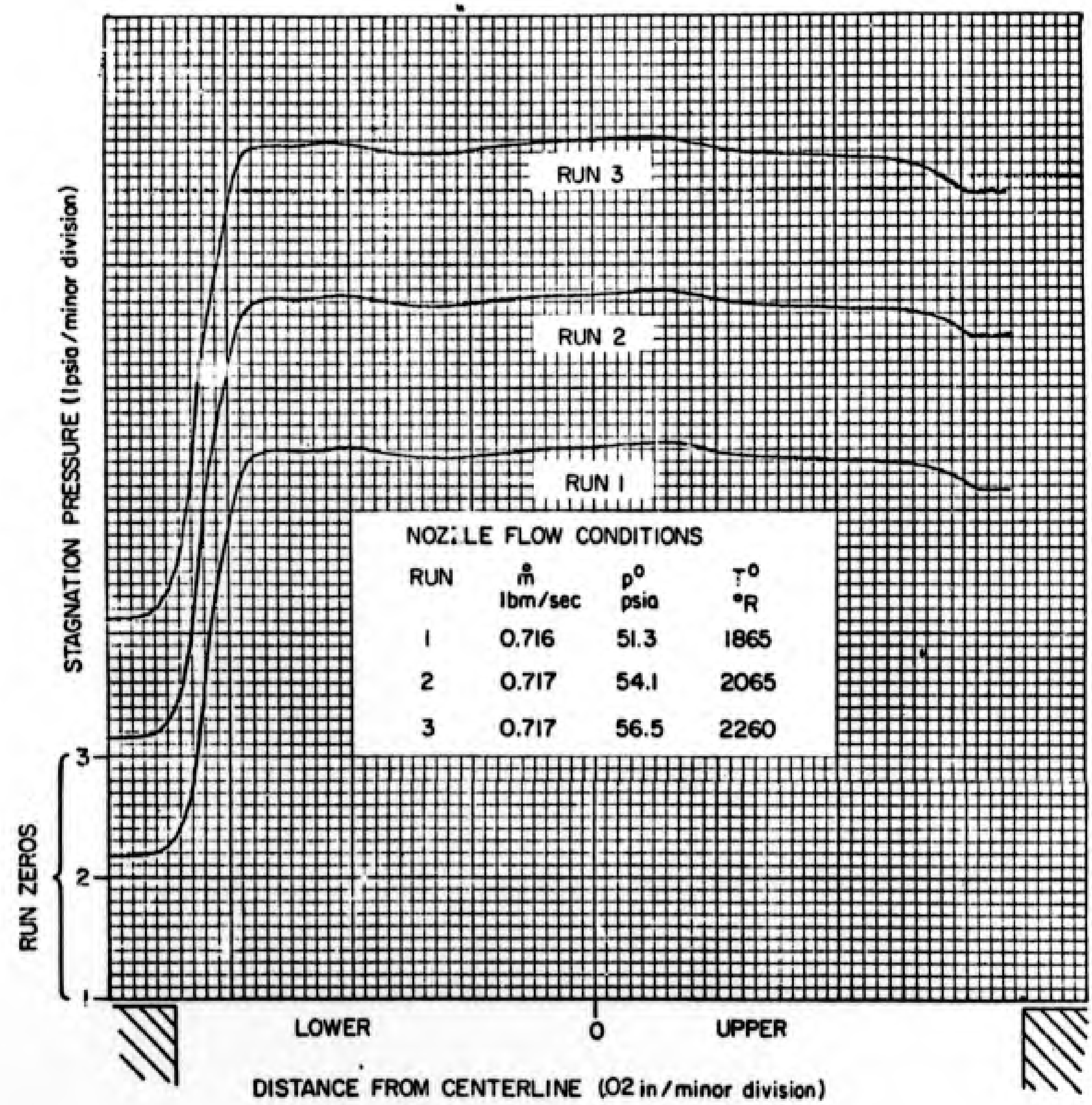


Fig. 3 - Stagnation Pressure Behind Normal Shock at Exit of M = 1.7 Nozzle

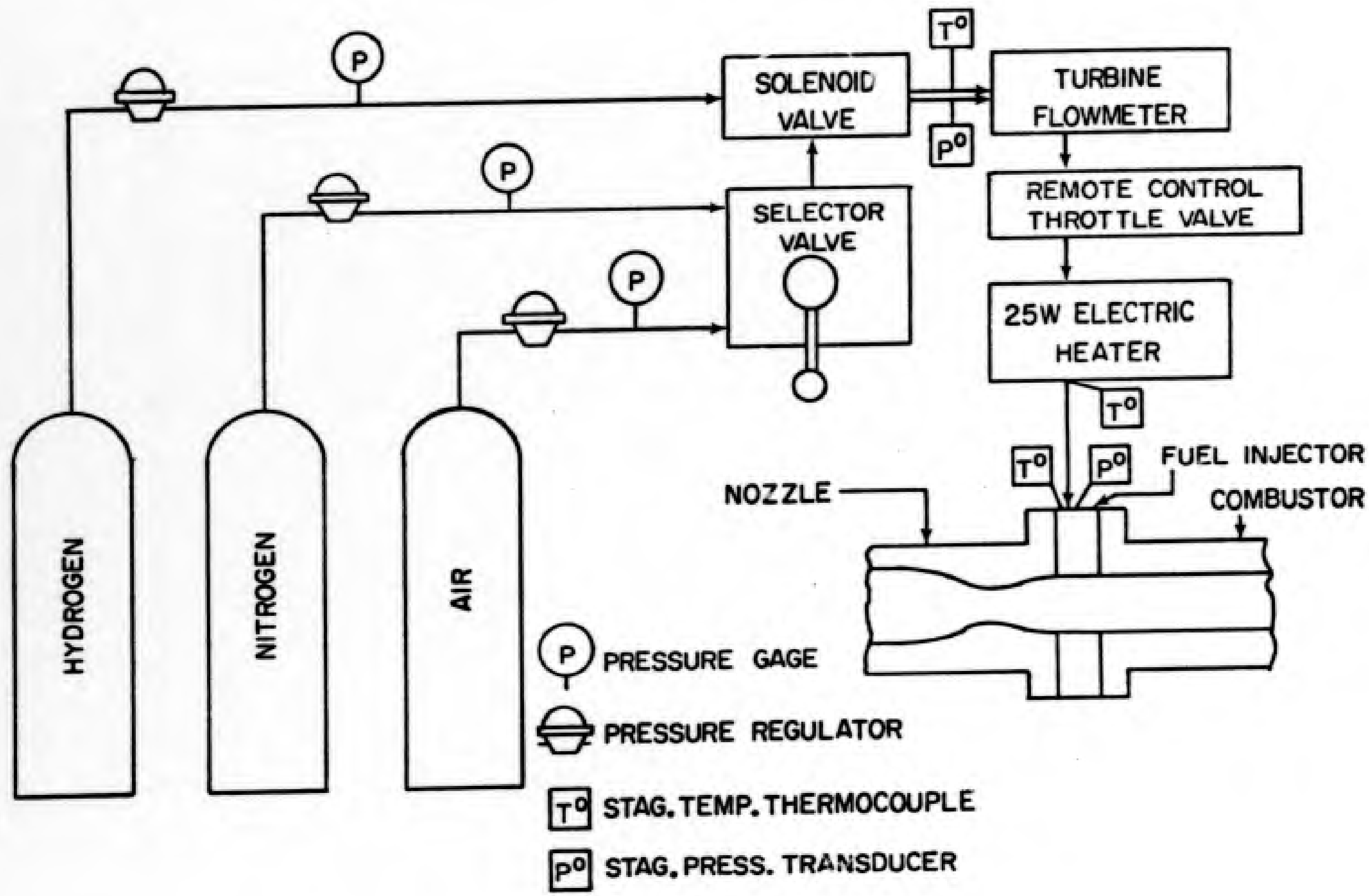
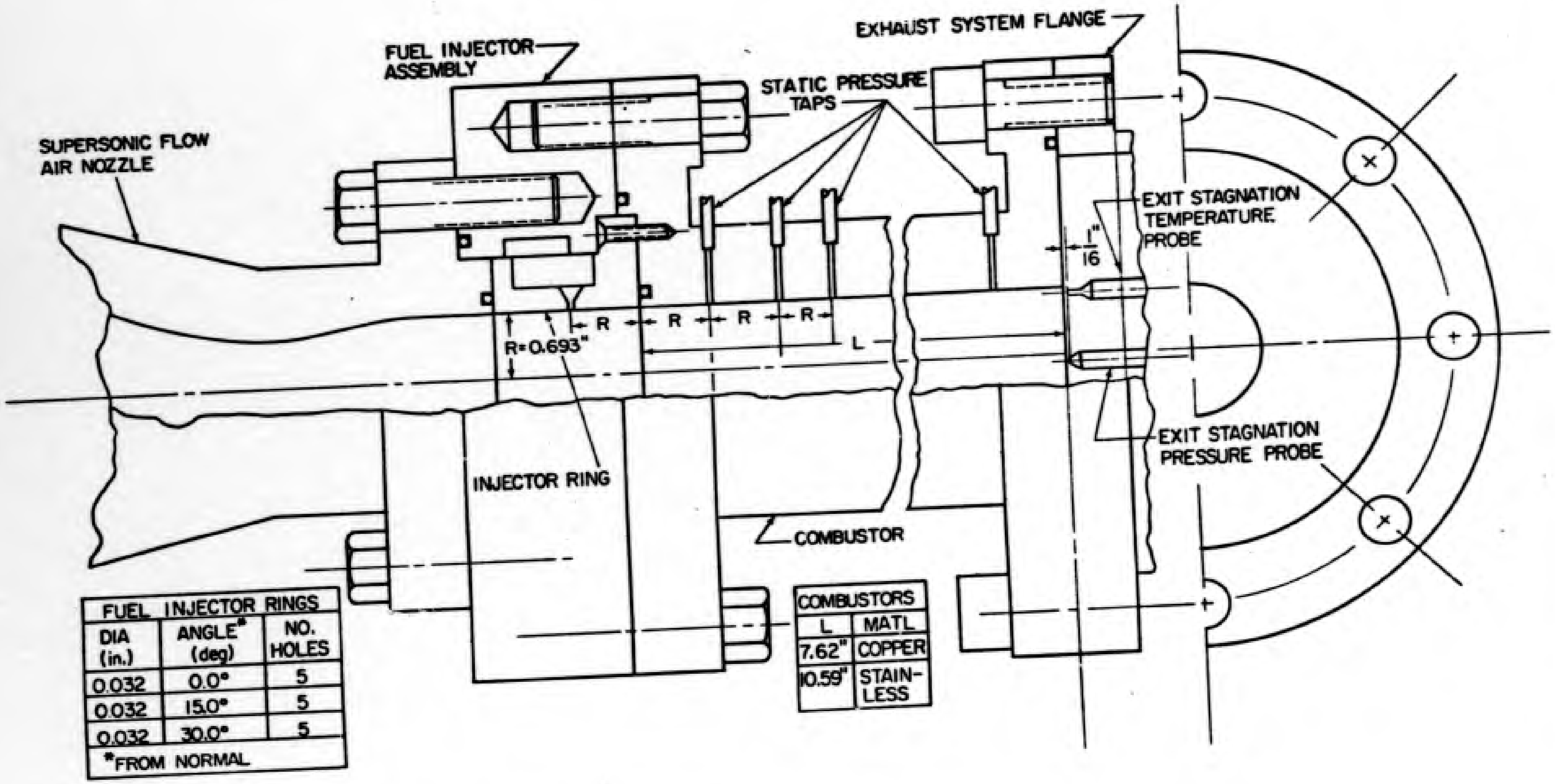


Fig. 4 - Fuel Flow Piping Diagram



FUEL INJECTOR RINGS		
DIA (in.)	ANGLE* (deg)	NO. HOLES
0.032	0.0°	5
0.032	15.0°	5
0.032	30.0°	5
*FROM NORMAL		

COMBUSTORS	
L	MATL
7.62"	COPPER
10.59"	STAINLESS

Fig. 5 - Supersonic Combustion Burner

E. SUPERSONIC COMBUSTORS

Two constant area combustors, both 1.3874 inches in diameter, one 7.62 and the other 10.59 inches long (Fig. 5) were used for the preliminary tests. The chambers were cooled by flowing small amounts of water over the external surface. The outside surface temperature of the combustor was measured approximately 2.0 inches from the point of fuel injection with a copper-constantan thermocouple. This temperature was kept at approximately 250°F to provide good metal strength but also to keep the inside surface temperature of the burner relatively high. The inside surface temperature of the combustor was measured approximately 0.7 inch from the burner exit with a copper-constantan thermocouple and was found to be approximately 1100°R when there was no combustion. Static pressure probes, used to help determine the locations of shocks and also the rate of heat addition, were spaced 0.693 inch apart along the burner sidewall (Fig. 5).

F. EXHAUST SYSTEM

The effluent gases were exhausted through a 4-inch diameter pipe. Since the nozzle exit pressures were lower than 1 atm (because of the reduced mass flows), it was necessary to use an air aspirator (Fig. 1) in the exhaust line to help the nozzle to go into flow. The exhaust gases were cooled by injecting water into the airflow near the exit of the burner. The water was injected into the hot air through the stagnation pressure-stagnation temperature probes located at the exit of the burner (Fig. 5).

SECTION III

INSTRUMENTATION AND DATA EVALUATION

A. AIR FLOW RATE

The stagnation conditions and the dimensions of the nozzle were used to determine the mass flow rate of the air. Since the thickness of the boundary layer at the nozzle throat was calculated to be 0.006 inch, the nozzle coefficient was assumed to be unity. The air stagnation temperature was measured with a platinum-platinum 10 percent rhodium thermocouple while the stagnation pressure was measured with a mercury manometer. The air flow rate was calculated by Fliegner's Formula.[6]

B. HYDROGEN FLOW RATE

The hydrogen mass flow rate was metered with a Flow Technology milliflow turbine flow transducer. The output pulses from this transducer were converted to a d.c. signal with a Flow Technology PRI-102 flow rate monitor. The turbine meter was calibrated (volume flow/pulse frequency) by the manufacturer with air. The manufacturer's calibration was compared at this Laboratory for both air and hydrogen at various pressure levels against both a flat plate orifice and a sonic nozzle flow meter. The mass flow rate of the air was then computed from the observed volume flow rate, and the measured temperature and pressure assuming the gases to be thermally perfect. The temperature of hydrogen at the turbine meter was measured with a copper-constantan thermocouple while the pressure was measured with a 0-300 psia Statham strain gage pressure transducer.

C. COMBUSTOR INSTRUMENTATION

The burner sidewall static pressures were measured at 0.963 inch intervals along the burner with mercury manometers. The stagnation temperature at the burner exit was surveyed along the vertical with a Model K302A High Temperature Instruments Corporation stagnation temperature probe. In position and at 1950°F, this probe was found to read approximately 150°F below the actual gas temperature. This discrepancy was found to be due to thermal conduction along the thermocouple sheath wall. The stagnation pressure probe moved along the same diameter as that of the stagnation temperature probe but was displaced by 3/4 inch. The transducer used for the stagnation pressure probe was a 0-300 psia Statham strain gage pressure transducer.

SECTION IV

THEORY

A general computer program was set up to calculate the performance of a supersonic combustor. The effects of variable area, heat addition, friction and heat transfer through the combustion chamber wall were taken into consideration in determining the effluent flow properties. Neglected in the calculations were the changes of mass, momentum, and molecular mass due to the fuel addition and the effects of gas mixing and shocks generated by fuel injection. The Mach number change caused by area change, friction of the walls of the duct, and heat exchange as obtained from the conservation laws (see Shapiro[7]) is

$$dM^2 = 2M^2 \frac{1+\frac{\gamma-1}{2}M^2}{M^2-1}\,\frac{dA}{A} - \gamma M^4 \frac{1+\frac{\gamma-1}{2}M^2}{M^2-1}\,4f\,\frac{dx}{D} - M^2(1+\gamma M^2)\frac{\left(1+\frac{\gamma-1}{2}M^2\right)}{M^2-1}\,\frac{dT^o}{T^o} \tag{1}$$

where:

$$dT^o = \frac{\delta q_W + \delta q_A}{c_p}$$

The heat transfer through the wall (q_W) was calculated from the convective heat transfer equation given by Kreith[8] as

$$\delta q_W = \bar{h}_c A_W (T_W - T) \tag{2}$$

where:

$$\bar{h}_c = Nu \frac{k_f}{D_H} \quad \text{for tubes} \tag{3}$$

and

$$Nu = Re\ Pr\ \frac{f}{8} \tag{4}$$

For air flows, the Prandtl number does not vary appreciably and can be assumed to be equal to 1. Also, for flows in smooth tubes and with flows having Reynolds numbers less than 1.2×10^5, the friction factor (f) can be written as

$$f = 0.184\ Re_D^{-0.2} \tag{5}$$

With these considerations, the Nusselt number can be written as

$$Nu = 0.023 \, Re_D^{0.8} \tag{6}$$

and the convective heat transfer ($\bar{h}_c$) can be written as

$$\bar{h}_c = 0.023 D^{-0.2} k_f \left(\frac{\rho V}{\mu}\right)^{0.8} \tag{7}$$

While the friction factor (f) may be calculated from Equation 5, the airflow exiting from the supersonic nozzle is not developed and the actual friction factor is considerably less than that given by the equation. The effective friction factor for the present flow was determined from the experimental sidewall static pressures along the burner. The calculated values of friction factor and flow properties with distance along the burner without fuel injection are listed in Table I.

Table I - Empirical Friction Factor Values
for Constant Area Duct

Loc.*	p_{exp}** (atm)	M	T (°R)	T° (°R)	p° (atm)	f
1.0d	0.531	1.500	1745.0	2406.4	1.901	
1.5d	0.532	1.497	1746.7	2406.1	1.897	0.0005
2.0d	0.535	1.491	1750.4	2405.7	1.889	0.0010
2.5d	0.538	1.485	1754.1	2405.2	1.883	0.0015
3.0d	0.545	1.471	1763.5	2404.7	1.868	0.0030
3.5d	0.552	1.457	1772.8	2404.6	1.855	0.0030
4.0d	0.559	1.443	1782.0	2404.4	1.841	0.0030
4.5d	0.567	1.426	1793.0	2404.1	1.825	0.0033
5.0d	0.576	1.408	1804.6	2403.5	1.806	0.0037
5.5d	0.586	1.389	1817.1	2402.8	1.788	0.0039
6.0d	0.596	1.370	1829.4	2402.5	1.771	0.0040
6.5d	0.608	1.349	1843.2	2401.9	1.753	0.0042
7.0d	0.620	1.328	1856.8	2401.6	1.737	0.0042

*Loc - distance from fuel injector holes (d = 1.3874 in.)
**p_{exp}- average values of run B1, Table III

Using these values for the friction factor, the amount of heat added by the chemical reaction can then be determined from the sidewall static pressure.

The penetration depth of the hydrogen flow into the supersonic air stream for these experiments was calculated from an empirical relation by Vranos and verified by Torrence.[9] The penetration depth (h) is given as

$$\frac{h}{D} = 0.168 \left(\frac{\rho_j V_j^{2}}{\rho V} \cos^2 \theta \frac{M}{M_j} \right)^{0.5} \left(\frac{x}{D} \right)^{0.0866} \tag{8}$$

SECTION V

TEST PROCEDURE

The various steps in the test procedure are given below:

(1) Set up the analog computer for on-line data recording.

(2) Establish a low mass flow of air through the burner assembly.

(3) Establish a low mass flow of air through the hydrogen heater and raise the temperature of the air to 1000°F.

(4) While continuing to heat the air flowing through the fuel heater (final temperature = 1750°F), the main air flow is brought to a temperature of 1950°F. The cooling water is added to the total temperature and total pressure probe cooling system when the main air flow temperature reaches 300 to 500°F. This water flow also cools the exhaust ducting and exhaust gases.

(5) The Honeywell automatic temperature controller is set to keep the main air flow temperature at 1950°F and then the Taylor Instruments automatic pressure controller is set to maintain a constant nozzle stagnation pressure (and thus a constant mass flow rate).

(6) The air flowing through the hydrogen heater is then replaced with nitrogen and after the air has been purged from the hydrogen heater, the nitrogen is replaced by hydrogen.

(7) The hydrogen mass flow and heater exit temperature are adjusted to the desired values, allowed to stabilize and then the experimental data are recorded. This step may be repeated as many times as necessary for data at various mixture ratios for this nozzle-injector-burner combination.

(8) To terminate an experiment, hydrogen in the fuel system is replaced by nitrogen and shortly thereafter by air. The power to the air and hydrogen heaters is cut off and the heaters are allowed to cool. Thereafter, the coolant water is turned off and the air flow through both heaters is shut off.

SECTION VI

PRELIMINARY RESULTS

A total of 31 supersonic combustion tests were conducted. All of the tests were made with a constant area combustor (diameter = 1.3874 in.). Hydrogen was injected radially for 28 of the experiments and at angle of 15 degrees from the radial direction (downstream) for three of the experiments. The combustion chamber was designed to simulate that of a M = 5.0 flight aircraft flying at an altitude of approximately 100,000 feet. The design required a diffusion of the air to M = 1.67, producing a static temperature of 1530°R and a static pressure of 1 atm. Since oblique shocks were formed as a result of the injection of hydrogen, it was estimated that the air temperature behind these shocks was 1800°R. Using the static ignition delay data for this mixture reported by Walker,[10] delay times of approximately 300 μs were expected. For this delay time, and the state of the air in the combustor, it was estimated that a combustor length of 8 inches would be required for combustion of the mixture. This delay time period applies only to a homogeneous mixture of hydrogen and air. The mixing of the hydrogen and air depends on the depth of penetration of the hydrogen into the airstream. The penetration depth depends on the degree of underexpansion of the hydrogen nozzle. While some difficulties to establish ignition at design conditions were anticipated, the chances to obtain combustion were considered good since Walker[10] reported shorter ignition delay times in flowing hydrogen-air mixtures and combustion was expected to aid the mixing.

During checkout of the equipment, the electric air heater burned out while operating at the design mass flow rate (0.866 lbm/sec) and a stagnation temperature of approximately 1800°F. At this time, other types of heaters (arc, chemical, pebble bed) were considered for possible use. However, since it was desired to conduct the experiments with the least possible amount of impurities in the air, the electric heater design was modified by the original designers. This modified heater initially operated well at 1950°F and a mass flow rate of 0.35 lbm/sec. The reduced mass flow was considered necessary because of high heater element temperatures. The lower air mass flow rate was a handicap to the ignition of the gases since ignition delay times are inversely proportion to the air density.

Several tests were made to establish the effect of friction and shocks on the sidewall static pressures of the two combustors used in these studies. These tests included no gas injection and inert (nitrogen) gas injection, the data of which are tabulated in Tables II and III. Graphs of the sidewall pressure increases for combustors A and B for each of the tests tabulated in Tables II and III are given in Figs. 6 and 7, respectively. For the cases of no injection,

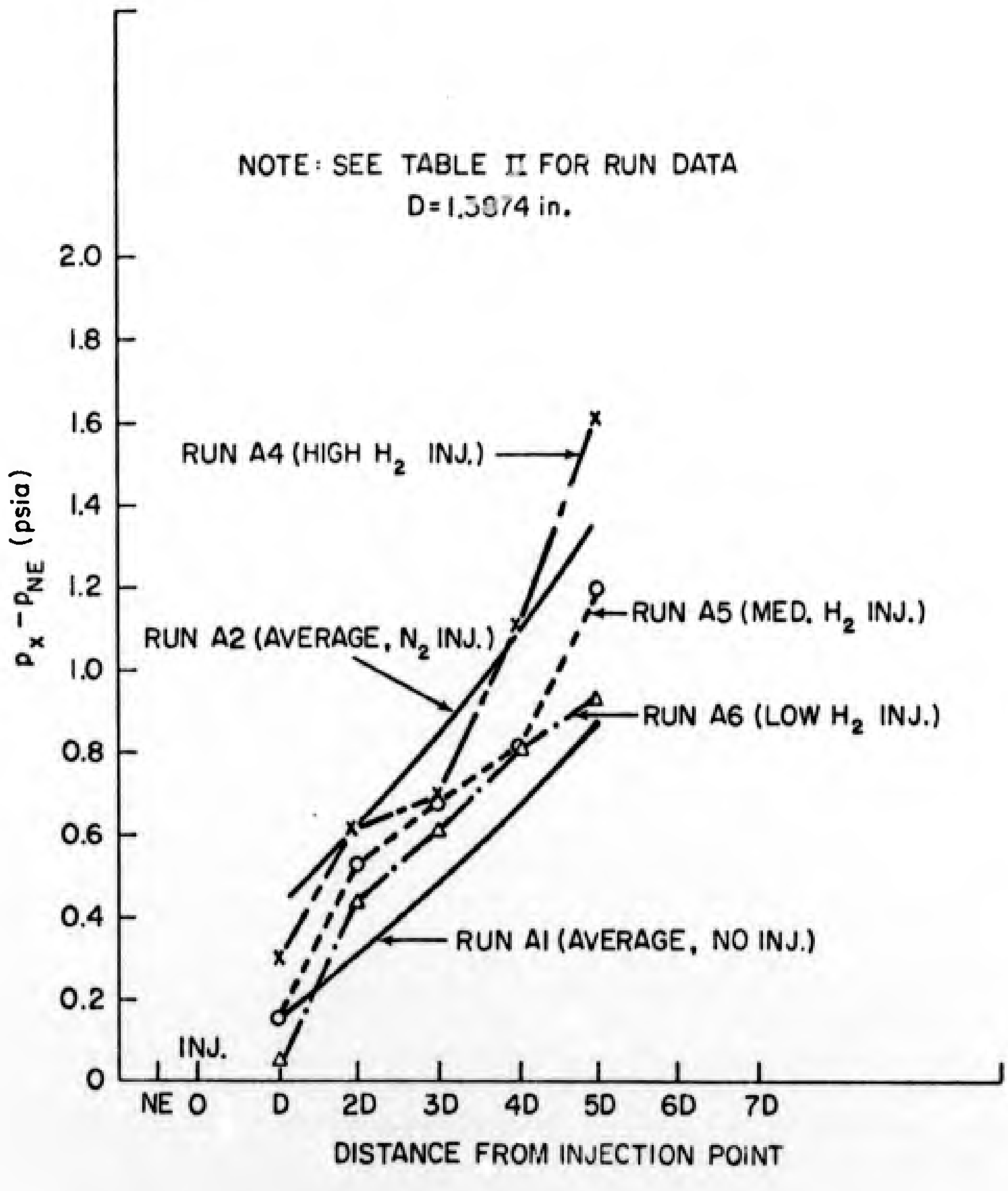


Fig. 6 - Static Pressure vs. Distance Along Burner A

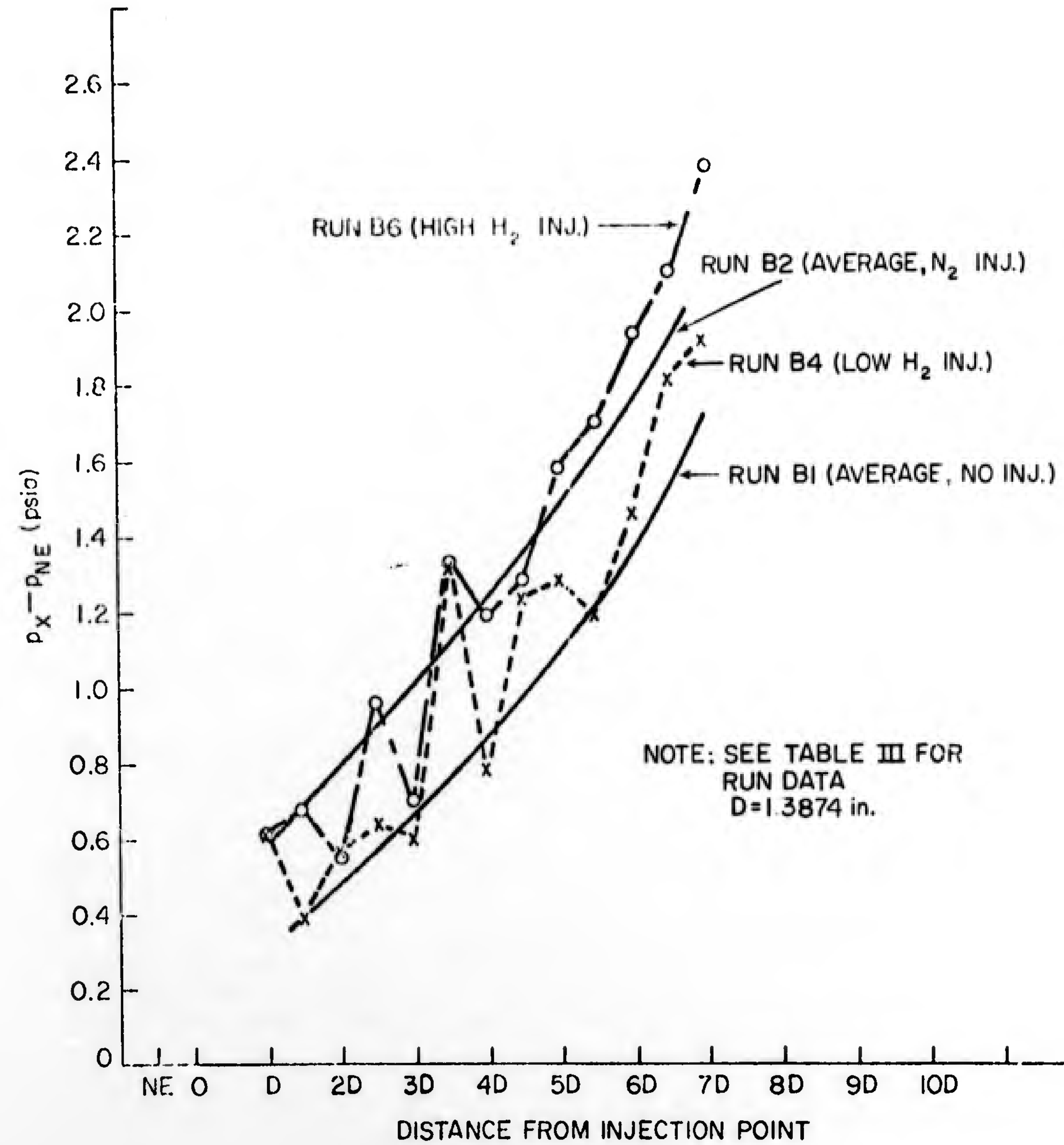


Fig. 7 - Static Pressure vs. Distance Along Burner B

the graphs of the sidewall pressure show a weak oblique shock wave generated at the nozzle exit-injector interface and traveling down the combustor, reflecting off the combustor walls. With nitrogen injection, the shocks are seen to be significantly stronger, yielding higher static pressures as the end of the burner is approached. For tests with hydrogen, the change in pressure with distance approximated that when a similar amount of nitrogen was injected. It was, therefore, concluded that there was very little heat addition to the airflow as a result of combustion of the fuel. The hydrogen mass flow rates of runs A1 and B3 are approximately equal to the amount of hydrogen required to choke the flow (if combustion were complete and neglecting shock effects). The rise in static pressure obtained when the $M = 1.67$, $p = 7.2$ psia, flow is choked (from Rayleigh line theory) is approximately 13.5 psia. This increase is considerably higher than that observed in the experiments which is additional evidence that very little heat was added to the air stream. The high hydrogen flow runs (A1 and B3) had the deepest penetration and best mixing and, therefore, the greatest chance for combustion. While the penetration depth required for certain combustion was estimated to be 0.12 inch (Section IIB), runs A4 and B6 had penetration depths of only 0.10 inch. Therefore, with the long ignition delay times and the shallow fuel penetrations for the above experiments, significant combustion could not be expected. Plans were made to promote combustion by means of a platinum catalyst and/or the redesign of the fuel injectors; however, the airflow heater again burned out and the preliminary tests were concluded. The air heater is being redesigned prior to undertaking a full series of supersonic combustion experiments with this apparatus.

A survey of the stagnation pressures and stagnation temperatures at the burner exit for experiments A4, A5 and A6 is given in Figs. 8 and 9. As mentioned in Section III, the stagnation temperature is not correct because of heat loss through the thermocouple sheath. This temperature is also expected to be in error for the case of incomplete combustion, since combustion can occur behind the shock which stands ahead of the stagnation temperature probe. The measured stagnation pressure is seen to be lower than the calculated value at the burner exit. This discrepancy is attributed to shock losses and combustion associated losses. However, these graphs show symmetry and the gradients at the burner exit.

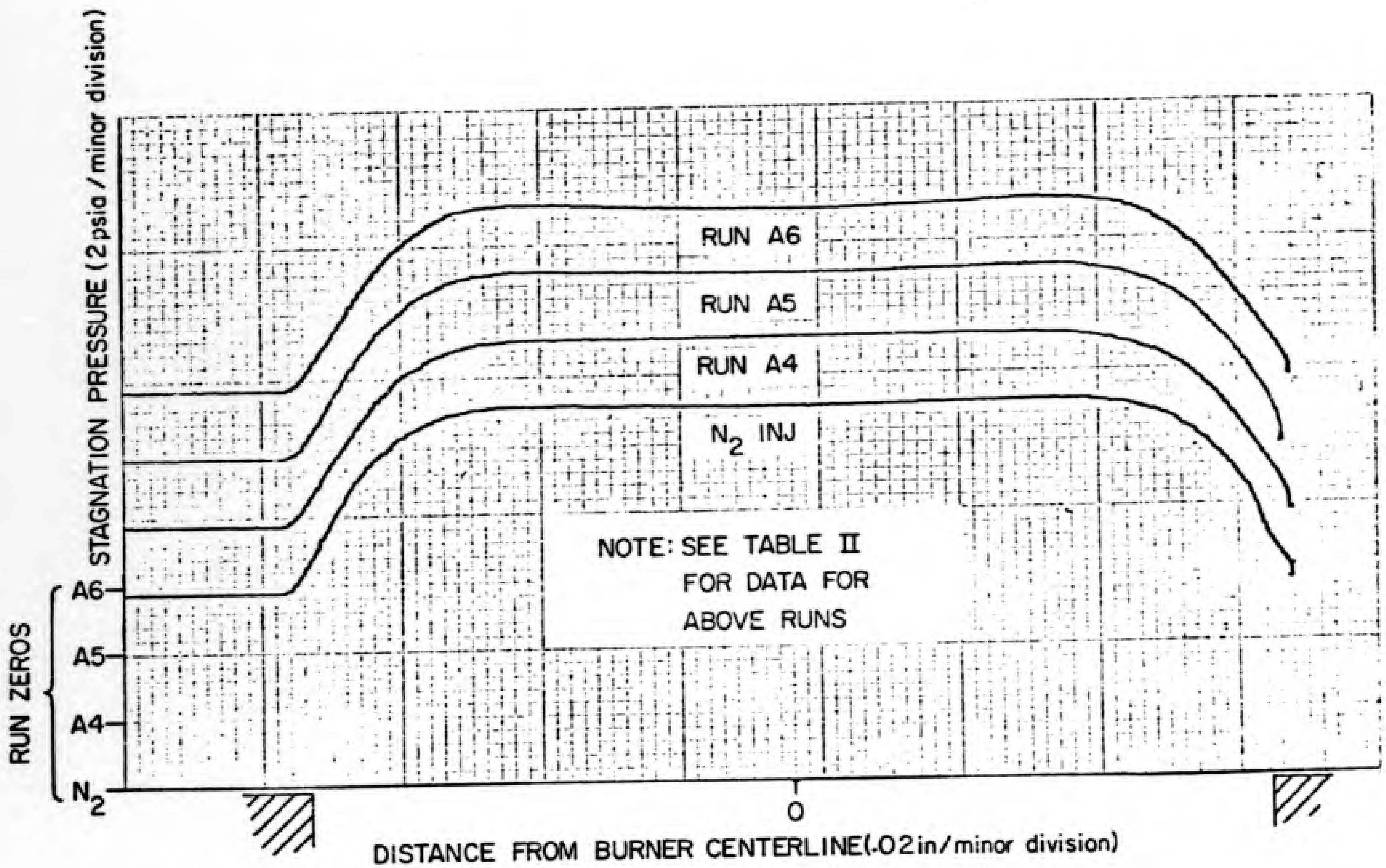


Fig. 8 - Stagnation Pressure Behind Normal Shock at Exit of Constant Area Burner

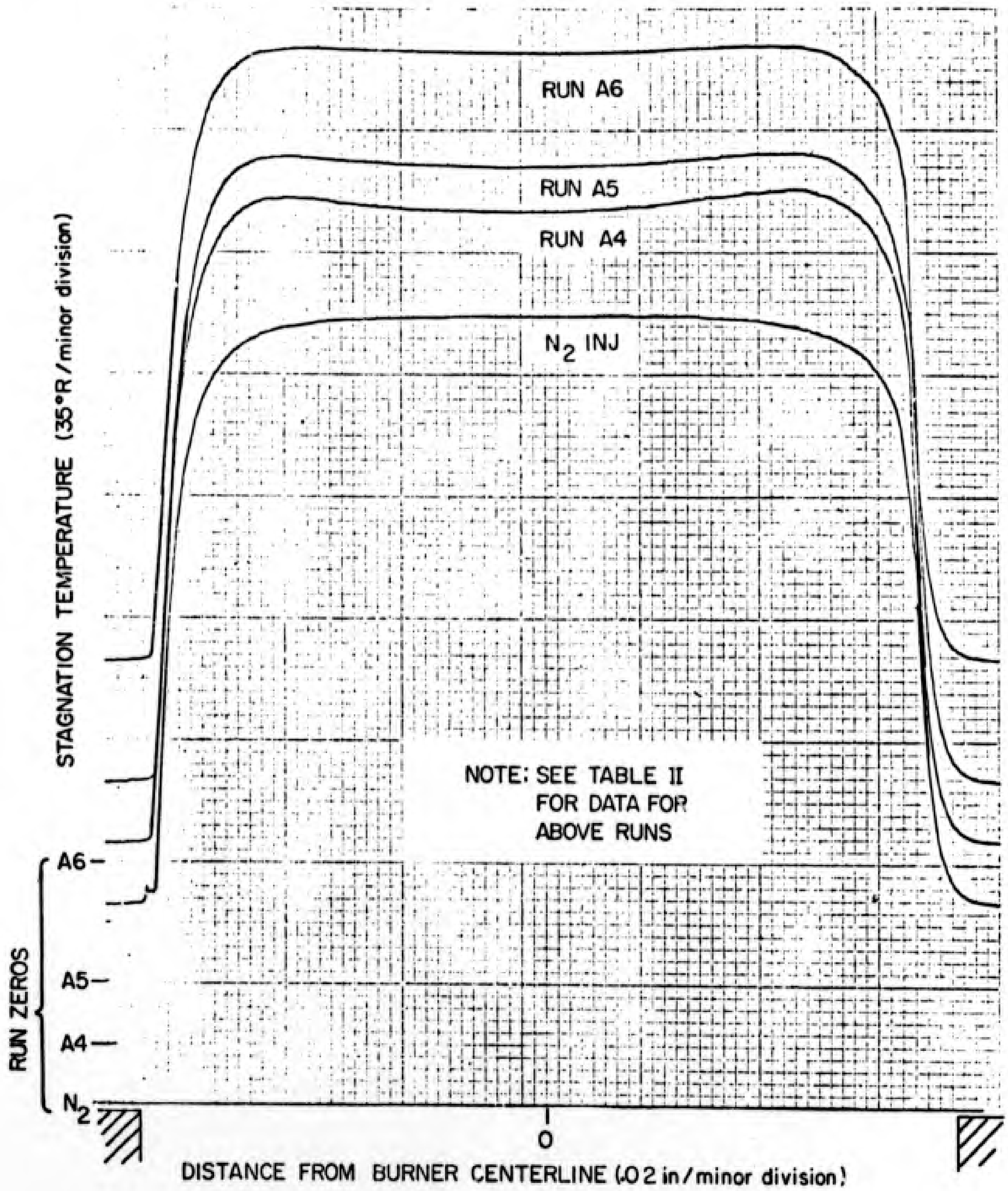


Fig. 9 - Stagnation Temperature at Exit of Constant Area Burner

Table II - Typical Preliminary Data For Burner A

		Run No.				
		A1	A2	A4	A5	A6
$\dot{m}_{air}$	(lbm/sec)	0.425	0.379	0.363	0.364	0.363
T°_{air}	(^{o}R)	2305	2346	2442	2442	2444
p°_{air}	(psia)	30.26	30.46	29.82	29.72	29.72
INJECTED GAS		-	N_2	H_2	H_2	H_2
$\dot{m}_{INJ}$	(lbm/sec)	-	0.0045	0.000677	0.000447	0.000280
T°_{INJ}	(^{o}R)	-	972	1171	989	901
p°_{INJ}	(psia)	-	51.56	45.32	31.40	23.24
h	(in)	-	0.11	0.103		0.074
p_{NE}	(psia)	7.66	7.44	7.27	7.27	7.27
$p_{1.0d}$	(psia)	7.78	8.17	7.57	7.43	7.32
$p_{2.0d}$	(psia)	7.99	7.96	7.88	7.80	7.71
$p_{3.0d}$	(psia)	8.11	8.04	7.97	7.95	7.88
$p_{4.0d}$	(psia)	8.36	9.08	8.38	8.19	8.08
$p_{5.0d}$	(psia)	8.28	9.05	8.88	8.47	8.20
$\Delta p_{1.0d}$	(psia)	0.12	0.73	0.30	0.16	0.05
$\Delta p_{2.0d}$	(psia)	0.33	0.52	0.61	0.53	0.44
$\Delta p_{3.0d}$	(psia)	0.45	0.60	0.70	0.68	0.61
$\Delta p_{4.0d}$	(psia)	0.70	1.64	1.11	0.92	0.81
$\Delta p_{5.0d}$	(psia)	0.62	1.61	1.61	0.12	0.93

Table III - Typical Preliminary Data for Burner B

		Run No.				
		B1	B2	B4	B5	B6
$\dot{m}_{air}$	(lbm/sec)	0.355	0.358	0.344	0.353	0.349
T°_{air}	(^{o}R)	2410	2402	2406	2380	23.88
p°_{air}	(psia)	28.87	29.11	27.95	2852	28.24
INJECTED GAS		-	N_2	H_2	H_2	H_2
$\dot{m}_{INJ}$	(lbm/sec)	-	0.00403	0.000249	0.000443	0.000666
T°_{INJ}	(^{o}R)	-	1048	893	1027	1201
p°_{INJ}	(psia)	-	55.3	21.60	31.18	44.96
h	(in)	-	0.12	0.073	0.086	0.104
p_{NE}	(psia)	7.30	7.36	7.08	7.22	7.12
$p_{1.0d}$	(psia)	7.74	8.14	7.89	8.08	7.92
$p_{2.0d}$	(psia)	7.78	7.87	7.75	7.88	7.77
$p_{3.0d}$	(psia)	7.93	7.98	7.88	7.91	8.02
$p_{4.0d}$	(psia)	8.12	8.61	8.05	8.12	8.51
$p_{5.0d}$	(psia)	8.32	9.01	8.56	8.91	8.90
$\Delta p_{1.0d}$	(psia)	0.44	0.78	0.81	0.86	0.80
$\Delta p_{2.0d}$	(psia)	0.48	0.51	0.67	0.66	0.65
$\Delta p_{3.0d}$	(psia)	0.45	0.60	0.70	0.68	0.61
$\Delta p_{4.0d}$	(psia)	0.70	1.64	1.11	0.92	0.81
$\Delta p_{5.0d}$	(psia)	0.62	1.61	1.61	0.12	0.93

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13. ABSTRACT

An apparatus to study the performance of supersonic combustion chambers was designed, constructed and checked out. However, the main airflow heater (which was supposed to be available for use in this program) was not capable of operating at the combustor design specifications. Therefore, the tests to check out the equipment had to be made with lower air mass flows. Results of preliminary tests in a constant area combustor with hydrogen fuel are reported.

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