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BRISTOL UNIVERSITY

SCHOOL OF AERONAUTICAL ENGINEERING

EXPERIMENTAL METHODS OF DETERMINATION OF
RAM ROCKET COMBUSTION EFFICIENCIES

by

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NOMENCLATURE

A	=	Area
C_{db}	=	Coefficient of burner drag
C_p	=	Specific heat at constant pressure
D	=	Diameter (in feet unless otherwise specified)
F	=	Impulse function $FA(1 + \gamma M^2)$
F^*	=	Impulse function at $M=1$
$^{\circ}F$	=	Fahrenheit degrees
F/A	=	Fuel to air ratio
g	=	Acceleration of gravity
h	=	Film heat transfer coefficient BTU/(hr)(ft ²)($^{\circ}F$)
I_{sp}	=	Specific impulse
k	=	Thermal conductivity BTU/(hr)(ft ²)($^{\circ}F/ft$)
L	=	Length
M	=	Mach number
m	=	Mass flow (lb/sec)
P	=	Absolute pressure
q	=	Rate of heat transfer
R	=	Gas constant (ft lb/lb $^{\circ}R$)
$^{\circ}R$	=	Rankine degrees
S_a	=	Air specific impulse at $M=1$
T	=	Absolute temperature
T_e	=	Rocket thrust

- U = Overall heat transfer coefficient
BTU/(hr)(ft²)(°F)
- V = Velocity
- γ = Ratio of specific heats C_p/C_v
- μ = Viscosity (micropoises or appropriate units)
- η = Efficiency
- ρ = Density (lb/ft³)
- $\gamma(M)$ = F/F^*

Subscripts

- o = Free stream condition
- 1 = Entrance station
- 2 = Exit or station under flow consideration
- a = Air
- e = Exit
- f = Fuel
- frict = Friction
- i = Inlet
- n = Nozzle
- r = Rocket

ABSTRACT

Two combustion efficiencies are defined: the ratio of the measured to the ideal air specific impulse, and the ratio of the actual rise in stagnation temperature to the theoretically calculated value. These efficiencies may be measured by the determination of appropriate stagnation and static pressures and by direct temperature measurements. It is determined that the former efficiency is relatively insensitive to the method of measurement. Determination of the combustion temperature on the other hand is not feasible from measurements of the appropriate upstream pressures. A fair accuracy may be obtained utilizing the stagnation pressure probe, and an accurate determination of temperature profiles is possible by utilization of the pneumatic temperature probe. The principles and design criteria of the pneumatic probe are discussed.

I. INTRODUCTION

The ram rocket configuration has many aspects of interest (Figure 1). Possible applications of this configuration would be for use as an ejector, an afterburner or a combined rocket-ram jet type power plant. A typical ram rocket configuration is indicated in Figure 1. In this diagram Station 0 indicates the stream tube at ambient conditions, Station 1 the inlet to the diffuser, Station 1 the burner entrance, Station 2 the burner exit, and Station n the nozzle.

The device may use a bipropellant fuel system with varying degrees of oxidizer to fuel ratios, or use a monopropellant fuel with the products of decomposition in the rocket forming the fuel for the ram air in the combustion chamber.

Theoretical calculations of the performance of the ram rocket indicate it to be superior both as to thrust per cross sectional area and as to specific fuel consumption to the ram jet in the lower Mach range and to tend to coincide with ram jet performance at supersonic speeds when particular fuels are used (1).

Its apparent superiority, in general, may be attributed to its increased air flow per unit of frontal area due to the rocket induction effect, the static thrust of the rocket portion, and the insensitivity of the rocket chamber to ambient conditions. The last named factor suggests that it is possible to provide fuel to the ram air in such a state that combustion can be effected over much wider air fuel ratios and pressures than are now possible, utilizing the conventional ram jet configuration.

It is the purpose of this paper to examine methods of evaluating the combustion efficiency of the ram rocket burner, experimentally, using a burner duct.

II EQUIPMENT

Air supply is provided to the burner by two 125 horsepower Spencer Centrifugal Compressors which may be run either in parallel or in series. The temperature is controlled by variable water cooled radiators. The air supply is conveyed by a twelve inch duct vertically down into a 4-1/2 foot diameter settling chamber. The vertical section contains an electrically actuated butterfly valve to control the mass flow of air (See Figure 2).

The burner is a twelve foot long, eight inch diameter, horizontal iron pipe; exhausting to atmosphere. Static pressure taps, which lead to water manometers, are located every two inches along the duct. The duct is water cooled in seven separated sections.

The rocket body is four inches in external diameter, with a windshield cap on the front, and is streamlined to a one inch exhaust diameter. It is concentrically located and aligned with the centerline of the burner by means of a steel strut. Propellant and instrumentation lines and leads to the

rocket are run along this strut inside a streamlining fairing.

Air flow measurements are made by means of a Taylor Pitot-Venturi Meter and are read on an inclined water manometer to an accuracy of one percent. Fuel flow measurements are effected by means of a Potter Flow Meter and cross checked by means of recording the variation of fuel tank weight. The tank weight is measured by means of four reinforcing electronic strain gages mounted on a steel strain ring with a dead weight calibration before and after each run.

Stagnation pressure is measured from a static tap in the settling chamber and read on a mercury manometer referenced to atmosphere.

Incoming air stagnation temperature is measured by means of a calibrated iron-constantan thermocouple located on the nose of the rocket. Cooling losses from the combustor are evaluated by means of thermocouples located at the inlet and the seven exhaust pipes. Water flow is measured by means of calibrated orifices. It should be noted that, although a pressure regulator was in-

stalled in the upstream line, it has not been possible to avoid fluctuation in water mass flow rates.

Rocket thrust is evaluated by two means. Four electrically reinforcing strain gages on the rocket strut measure the strain of the strut. This is evaluated by means of a dead weight calibration.

The rocket strut effectively is designed as a bell crank so that the forward thrust of the rocket is transmitted downward to the top of an Emery Hydraulic Cylinder. The thrust is then measured by the pressure read on a calibrated Bourdon Tube Gage.

Direct measurement of combustor temperatures presents some difficult problems. To obtain a true thermocouple stagnation temperature, the temperature recovery must be evaluated and radiation losses must be minimized by shielding. In view of the fact that, at stoichiometric mixture ratios, flame temperatures are much above the melting point of available metals, some other device than thermocouples must be employed.

High temperature measurement has customarily been accomplished by sodium D line reversal. The D line method has the obvious disadvantage that it depends upon the opinion of the operator as to when the D line reverses, and on the opinion of the operator in evaluating the temperature brightness of the D line light source with an optical pyrometer or thermopile. A second practical consideration is that the static temperature of an experimental burner will be measured with the operator a safe distance away, obviously necessitating a cumbersome optical system.

An ingenious solution to this problem was presented by David Moore of Fairchild Aircraft in the Aeronautical Engineering Review in 1946 (2). The device described was a water cooled pneumatic probe, which can be utilized at temperatures much higher than the melting point of the metal from which it is made (Figure 6).

The pneumatic probe is based upon the principle that the mass flow through a choked orifice can be measured as a function of the upstream total pressure, the orifice area, and the local speed of sound.

Noting that $(T/T_s)_{\text{sonic}} = \frac{2}{\gamma+1}$

The mass flow of gas may be written:

$$\begin{aligned} \dot{m} &= \frac{A_{\text{orifice}} P_s}{\sqrt{T_s}} \sqrt{\frac{\gamma}{R} \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{\gamma-1}}} \\ &= \frac{A_o P_s}{\sqrt{T_s}} f(\gamma, R) \text{ lb/sec} \end{aligned}$$

where P_s may be measured and A_{orifice} is known. If the mass flow through the water cooled probe is chilled, frozen equilibrium may be assumed, with γ and R constant. By now flowing this cooled gas through a second sonic orifice, the mass flow may be computed from measured values of orifice area, stagnation pressure and stagnation temperature. At the reduced gas temperatures it is possible to use an unshielded thermocouple with no correction for radiation losses. By equating the mass flows the entrance stagnation temperature may be computed:

$$\frac{A_1 P_{s1}}{\sqrt{T_{s1}}} f(\gamma, R) = \frac{A_2 P_{s2}}{\sqrt{T_{s2}}} f(\gamma, R)$$

or

$$T_{s1} = \left(\frac{A_1 P_{s1}}{A_2 P_{s2}} \right)^2 T_{s2}$$

The pressures, P_{s1} and P_{s2} , are measured on mercury manometers, A_1/A_2 is known, and T_{s2} is measured by a calibrated iron-constantan thermocouple referenced to ice bath.

All measurements are either photographically or electrically recorded on Leeds and Northrup Speedomax Recorders.

III. MEASUREMENT OF COMBUSTOR EFFICIENCIES

Combustor efficiencies are affected by the completeness of combustion, the air/fuel range of operation, the fuel injection system, the loss in total pressure across the combustor, the combustor dimensions, the exit temperature profile, and the flight conditions of the vehicle it propels. It is obvious that these conditions are interrelated. Friedman, Bennet and Zwick proposed that the combustion efficiencies, η be represented by (3):

$$\frac{1-\eta}{\eta} = f\left(\frac{V}{P_s L T_s F/A}\right)$$

where the latter is named an efficiency characteristic number.

Since the experimental apparatus has a limited range of inlet conditions, flight conditions may be extrapolated to various altitudes and Mach numbers by the proper combination of inlet pressures and temperatures and the assumption of a reasonable diffuser efficiency. The rocket design makes it desirable to operate the rocket at very nearly fixed operating condition as regards fuel flow and to control the air/fuel ratio by varying the

incoming air. Hence the flight conditions and the blowout velocities are indicated by the air fuel ratio and the geometry of the combustor. Since one of the primary interests in this configuration is the extension of the narrow range of permissible operating air fuel ratios, this will be selected as the independent variable of the experimental combustion efficiency.

If combustor sizes and pressure losses are within reasonable limits, the most indicative combustor efficiency can be directly defined as:

$$\eta_c = \frac{(\Delta T_g)_{\text{measured}}}{(\Delta T_g)_{\text{ideal}}}$$

A frequently used criterion of combustor efficiency is the ratios of the Air Specific Impulse,

$$S_a \gamma(M) = F_o/m_a$$

where

$$\eta_c = \frac{(S_a) \gamma(M)_{\text{measured}}}{(S_a) \gamma(M)_{\text{ideal}}}$$

and S_a is defined as:

$$S_a = (1 + m_f/m_a) \sqrt{\frac{RT_g(2)(1+\gamma)}{g \gamma}}$$

See Appendix I

This is somewhat more indicative of the true burner efficiency. $S_g \gamma(M)$ ideal is computed upon Rayleigh line heating from entrance temperature and Mach number to exit adiabatic flame temperature and Mach number. This efficiency would indicate both the actual temperature rise across the burner and the loss in total pressure due to friction.

Although the burner effective length, the thrust per cross sectional area, and hence the combustion heat release per unit volume are important criteria, they can be compared only with other combustors and not with an ideal case. However, the effective combustor length may be used as a parameter in plotting the various efficiencies.

Evaluation of the measured quantities is dependent upon γ . It has been found that γ may reasonably be assumed to be very close to the computed value at any given air fuel ratio (Figure 3). More precisely, γ can be determined by an Orsat analysis by quenching a gas sample to freeze the equilibrium. It should be noted that the water content must be considered in the determination of γ from the gas sample.

Measurement of the actual stream thrust at any section of the burner may be made by measuring the stagnation pressure profile and the static pressure. By employing the relation of the static to stagnation pressure, the average Mach number at the station may be computed and the stream thrust measured:

$$F = PA(1 + M^2)$$

The heat loss per second to the cooled burner duct may be evaluated by the summation of the heat transfer rates to the upstream burner cooling sections. Since the water flow fluctuates, since some time is required for the coolant to approach equilibrium, and since each water temperature is recorded only once every thirty two seconds, caution must be exercised in approximating the instantaneous heat transfer rate.

From the measured air flow and air temperature entering the combustion chamber, the entrance Mach number and temperature is known:

$$M_1 = \frac{P_1 A_1 M_1 \sqrt{\gamma T_1}}{\sqrt{\gamma R T_1}}$$

Neglecting friction, the exhaust temperature may be

computed from the measured Mach number utilizing Rayleigh line considerations. From the temperature, the specific heat of the gas may be computed. Equating heat transfer rates:

$$\dot{m}_{\text{gas}} (\bar{C}_p) \Delta T = \dot{m}_{\text{water}} \Delta T \text{ BTU/Sec}$$

A first approximation of the temperature loss of the gas may be obtained and an average specific heat, \bar{C}_p , computed. By iterating, the rise in temperature and hence the increase in Mach number may be computed to correct for the cooling loss. The assumption of Rayleigh line (frictionless) heating is well within the accuracy of the measured heat transfer rate. With a more refined measurement of the cooling losses it would be possible to replace the temperature loss of the gas, section by section, with a more accurate average specific heat. It should be noted that this correction to the temperature is only of the order of 5 to 8 percent of the total temperature rise.

Having measured the Mach number, the section impulse, F_2 , and the air and fuel flows, one can apply the principle of continuity to solve for the air specific impulse. Appendix I.

$$m_2 = (m_a + m_f)$$

$$F_2 = \frac{m_a (1 + m_f/m_a) \sqrt{R_2 T_{s2}} (1 + \gamma_2 M_2^2)}{M_2 \sqrt{\gamma_2 (1 + \frac{\gamma_2 - 1}{2} M_2^2)}}$$

at $M_2 = 1$ $F_2 = F_2^*$

$$F_2^* = m_a (1 + m_f/m_a) \sqrt{\frac{R_2 T_{s2} 2(1 + \gamma_2)}{\gamma_2}}$$

defining

$$S_a = \frac{F_2^*}{m_a}$$

$$F_2 = m_a S_a \frac{F_2}{F_2^*} = m_a S_a \gamma(M_2) = m_a S_a \frac{(1 + \gamma_2 M_2^2)}{M_2 \sqrt{2(\gamma_2 + 1)(1 + \frac{\gamma_2 - 1}{2} M_2^2)}}$$

defining the Air Specific Impulse

$$F_2/m_a = S_a \gamma(M_2)$$

From these values the stagnation temperature may be computed and the actual temperature rise of the incoming air evaluated.

Since the impulse in a constant area, frictionless duct is constant, one may compute the exit impulse from the measured values at the entrance to the duct by applying a reasonable value of burner drag. See Appendix I:

$$F_{\text{exit}} = F_{\text{entrance}} - F_{\text{frict}}$$

$$F_2 = P_{s1} A_2 + F_f = P_2 A_2 (1 + \gamma_2 M_2^2)$$

From these values, M_2 , the Air Specific Impulse, and the rise in temperature, may be measured as before.

By utilizing the pneumatic probe the stagnation temperature of the gas is measured directly at any section. By replacing the upstream temperature losses the true stagnation temperature may be accurately estimated and the measured rise in temperature computed. From these temperatures the measured Air Specific Impulse may be computed. Since the pneumatic probe depends upon the measured total pressure, the Mach number and hence the stream thrust may be computed from the static pressure tap at that station.

By virtue of the static pressure taps located along the burner one may quite accurately estimate the burning length and measure the flame temperature at this station (Figure 4). The obvious advantage is the minimization of the coolant corrections.

The burning length is determined by noting the point along the burner duct where the pressure drop through the flame becomes tangent to the frictional pressure drop. These measurements provide a means of correlating the required burner length as a function of air/fuel ratios (for a given combustor geometry).

Utilizing the momentum pressure drop or probe measurements the completeness of combustion may be measured throughout the burning length of the combustor. We may, therefore, plot a given percentage of completion of combustion versus distance along the duct for various values of the air-fuel ratio (Figure 5).

IV CONCLUSIONS

Two combustion efficiencies have been defined and three methods of evaluating these efficiencies are outlined. In measuring the Air Specific Impulse, both the pneumatic probe and the stagnation pressure probe present approximately the same degree of accuracy. Should the design of the rocket be such that separation occurs as the incoming air passes over the rocket, the error introduced, combined with the error inherent in reading small manometer deflections at the combustor entrance, would make the measurement of this efficiency at the entrance to the combustor perhaps one percent less accurate than by the two previous methods.

Insofar as the rise in temperature is concerned, evaluation by measurement of entrance conditions is very difficult. Since M_2^2 as evaluated from the impulse function is, at a maximum, of the order of .25:

$$\Gamma M_2^2 = \frac{P_{s1} A_2 + \Gamma V - P_2 A_2}{P_2 A_2}$$

and T_{s2} evaluated from continuity is:

$$T_{s2} = \frac{(P_2 A_2^2) \gamma T_2 M_2^2 (1 + \frac{\gamma-1}{2} M_2^2)}{(M_2 + M_1)^2 R_2}$$

The slightest deviation in any measurement is magnified approximately one hundred times in the computation of the exit Mach number. In these tests, since the exit Mach number ranges from about 0.2 to about 0.5, extreme accuracy is required in the manometer readings using the stagnation probe method. The method, however, is feasible and will provide a much better estimate of the true stagnation temperature. Accuracy of the order of 95 percent is estimated for this method.

The pneumatic probe is accurate to a deviation of less than one percent. Although the pneumatic probe is dependent upon manometer pressure readings also, the spread of manometer readings is such as to minimize the percentage inaccuracy. The pneumatic probe has the additional advantage that temperature profiles may be measured. This is especially advantageous in ascertaining

the combustor length at which a uniform temperature profile is obtained.

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APPENDIX I

DERIVATION OF STREAM THRUST METHOD

From simple momentum considerations it can be shown that the force accelerating a fluid in a duct is:

$$m dv = -d(PA) + P dA - dF_{frict}$$

integrating between Stations 1 and 2

$$m_2 V_2 - m_1 V_1 = -P_2 A_2 + P_1 A_1 + \int_1^2 P dA - F_{fr}$$

or defining $(PA + mV) = PA(1 + \gamma M^2) = F$

$$\int_1^2 P da = F_2 - F_1 + F_{frict}$$

impulse function

in a constant area burner then $da = 0$

$$F_2 = F_1 - F_{frict}$$

temperature. At the entrance to the combustion chamber in Section 1 there are two concentric gas streams; the rocket jet and the ram combustion air (Figure 1). Therefore at Section 1 the stream thrust will consist of the air stream thrust F_1 and the rocket stream thrust F_r . Equating:

$$F_2 = F_1 + F_r - F_{\text{frict}}$$

$$F_1 = P_1 A_1 (1 + \gamma_1 M_1^2)$$

$$F_r = P_r A_r (1 + \gamma_r M_r^2) = m_r I_{sp} = P_1 A_r (1 + M_r V_r) \text{ if}$$

perfectly expanded (1).

The assumption that the burner drag coefficient equals one is made. This value corresponds to most assumptions in the literature and has been proved experimentally to be reasonably justified (4).

$$F_{\text{frict}} = (C_{db}) \frac{\rho_1 V_1^2 A_1}{2} = (1) \frac{P_1 A_1 \gamma_1 M_1^2}{2}$$

Therefore:

$$\begin{aligned} F_2 &= (P_1 A_1 + P_1 A_1 \gamma_1 M_1^2) - (P_1 A_1 \gamma_1 M_1^2) + (P_1 A_r + m_r V_r) \\ &= P_1 A_1 (1 + \frac{\gamma_1 M_1^2}{2}) + P_1 A_r + T_r \text{ measured} \end{aligned}$$

$$\text{Isentropically } P_1 = P_{sl} \frac{P_1}{P_{sl}} = P_{sl} (1 + \frac{\gamma-1}{2} M_1^2)^{\frac{\gamma}{1-\gamma}}$$

Expanding by the binomial theorem:

$$P_1 = P_{sl} \left[\frac{1 + \gamma(\gamma-1)M^2}{(1-\gamma)^2} + \frac{(2\gamma^2 - \gamma)M^4}{6} + \dots \right]$$

and neglecting terms of order M^4 , since M_1 is small:

$$\begin{aligned} F_1 &= P_{sl} (1 + \frac{\gamma}{2} M^2) (1 - \frac{\gamma}{2} M^2 + \dots) \\ &= P_{sl} A_1 (1 + \frac{\gamma}{2} M^2 - \frac{\gamma}{2} M^2 - \frac{\gamma^2}{4} M^4 + \dots) \\ &= P_{sl} A_1 \end{aligned}$$

(Note: The subscript 1 on the Mach number has been dropped for simplicity.)

and since $A_2 = A_1 + A_r$

$$F_2 = P_{s1} A_2 + T_r - P_{s1} \frac{A_r \gamma M_1^2}{2}$$

where the last term is negligible.

The approximation has proved by measured values on the burner to differ from the exact measurement by .4 percent at an entrance Mach number of .15.

$$F_2 = P_{s1} A_2 + T_r$$

Hence from measured values of rocket thrust and total pressure the impulse function at Section 2 may be evaluated.

Exit pressure of the burner, if unchoked, is barometric. To evaluate M_2 , values of γ calculated for the measured air-fuel ratio are used. This assumption introduces a slight degree of error, in view of the fact that T_2 will be below the calculated adiabatic flame temperature for that mixture ratio.

Having computed F_2 and M_2 obtain T_{s2}

$$\text{From continuity } m_2 = m_1 = (m_a + m_f)$$

$$(m_a + m_f) = (\rho VA)_2 = \frac{P_2 A_2 M_2 \sqrt{\gamma_2}}{\sqrt{R_2 T_2}} \left[\frac{1 + \frac{\gamma_2 - 1}{2} M_2^2}{T_{s2}/T_2} \right]^{1/2}$$

Substituting for $P_2 A_2$ in $F_2 = P_2 A_2 (1 + \gamma_2 M_2^2)$

$$F_2 = \frac{m_a (1 + m_f/m_a) \sqrt{R_2 T_{s2}} (1 + \gamma_2 M_2^2)}{M_2 \sqrt{\frac{5\gamma_2 (1 + \gamma_2 - 1) M_2^2}{2}}}$$

at $M_2 = 1$ $F_2 = F_2^*$

$$F_2^* = m_a (1 + m_f/m_a) \sqrt{\frac{R_2 T_{s2} (1 + \gamma_2) (2)}{5\gamma_2}}$$

$$S_a \frac{F_2^*}{m_a}$$

and defining

$$Y(M_2) \frac{F_2}{F_2^*} = \frac{(1 + \gamma_2 M_2^2)}{\sqrt{2(\gamma_2 + 1) (1 + \frac{\gamma_2 - 1}{2} M_2^2)}}$$

The Air Specific Impulse may then be defined:

$$F_2/m_a = S_a Y(M_2)$$

It is to be noted that F_2 is a function only of m_a , m_f , M_2 , and R_2 , γ_2 , T_{s2} where T_{s2} is assumed to be the adiabatic flame temperature. Thermochemical computations of equilibrium compositions of combustion products and adiabatic flame temperature then may be plotted as theoretical $\frac{m_a S_a}{m_f}$ values versus air fuel ratios m_a/m_f .

APPENDIX II

DEVELOPMENT OF THE PNEUMATIC PROBE

The pneumatic probe is based upon the principle that the mass flow through a choked orifice can be measured as a function of the upstream total pressure, the orifice area, and the local speed of sound.

$$\text{Noting that } (T/T_s)_{\text{sonic}} = \frac{2}{\gamma+1}$$

The mass flow of gas may be written:

$$m = A_{\text{orifice}} P_s \sqrt{\frac{\gamma \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}{R}}$$

$$= \frac{A_o P_s}{\sqrt{T_s}} f(\gamma, R) \text{ lb/sec}$$

where P_s may be measured and A_{orifice} is known. If the mass flow through the water cooled probe is chilled, frozen equilibrium may be assumed, with γ and R constant. By now flowing this cooled gas through a second sonic orifice, the mass flow may be computed from measured values of orifice area, stagnation pressure and stagnation temperature. At the reduced gas temperatures it is

possible to use an unshielded thermocouple with no correction for radiation losses. By equating the mass flows the entrance stagnation temperature may be computed:

$$m = \frac{A_1 P_{s1}}{\sqrt{T_{s1}}} f(\gamma, R) = \frac{A_2 P_{s2}}{\sqrt{T_{s2}}} f(\gamma, R)$$

or

$$T_{s1} = \left[\frac{A_1 P_{s1}}{A_2 P_{s2}} \right]^2 T_{s2}$$

The general configuration of the probe is illustrated in Figure 6. The hot gas is sucked in through the front orifice at sonic velocity, cooled while passing through the water cooled probe, carried through an electrically heated tube (to prevent condensation of the water products of combustion) to the second orifice. Static pressure taps are located before and after the second orifice leading to mercury manometers. By closing the line to the vacuum pump both manometers will indicate the total pressure at the probe tip. Since the tube to the orifice area is so large, the upstream tap reads essentially stagnation pressure while the tap downstream provides a check that the pressure

ratio is such that sonic velocity will be maintained across this second orifice.

The size of the orifices is in general determined by the size and hence the mass flow of the vacuum pump which is available. For ease in construction and to minimize the percentage of deviations in areas from corrosion, etc., the maximum area available was selected consistent with the capacity of the available vacuum pump and a reasonable margin of safety.

Utilizing the NACA criterion that a pressure ratio of 3.2 will insure sonic flow through a flat plate orifice, A_2/A_1 was selected at 3.3 (5).

(D = .040/D = .075) matching drill sizes

Since

$$M_{\text{critical}} = \frac{A_1 P_{s1} \sqrt{\gamma \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}}{\sqrt{RT_{s1}}} \quad \text{lb/sec}$$

maximum flow would occur at the lowest temperatures and highest total pressures anticipated.

At:

$$T = 500^\circ \text{ R}, P_{s1} = 17 \text{ lbs/in}^2, \gamma = 1.4$$

The mass flow:

$$M = .0303 \text{ lbs/min} = 4.00 \text{ ft}^3/\text{min}$$

at a vacuum pressure of 1.661 lbs/in² absolute, which is well within the limits of the pump.

While sonic velocities may be achieved at lesser pressure ratios by utilizing a nozzle instead of an orifice, it was found by the NACA that a sizable cooling error was introduced due to the passage of the gas through the long water cooled nozzle. Hence a flat plate orifice with an L/D ratio of .25 was used (Figure 8).

The general construction of the probe is also illustrated in Figure 7. The innermost gas tube is 1/4 inch standard ~~steel~~ (.035 in. wall) stainless steel tubing. Surrounding this is a 3/8 inch by .020 inch wall stainless steel tube to form the inner cooling annulus. Stainless steel was chosen merely for its resistance to rust and corrosion in general. The .020 inch wall was desired in order to equalize the inner cooling annulus area with the outer. The outer wall forms the return cooling annulus. Standard size 1/2 inch by .035 inch wall inconel tubing was chosen because of its high melting point. The probe tip, containing the front orifice, was constructed of

beryllium copper because of the requirement of high heat conductivity to remove heat from the lip of the flat plate front orifice combined with resistance to deformation (Figure 8).

The preliminary heat transfer calculations were based upon the measured cooling water which could be flowed through an orifice 85 percent of the size of the minimum annulus. The heat transfer requirements were then computed based upon the maximum area of the probe which could be inserted into the burner duct.

Based upon McAdams (6):

$$h_{\text{water}} = 150 (1 + .011T) \frac{(V_{\text{ft/sec}})^{.8}}{(D_{\text{in}})^{.2}} = 6,850 \text{ BTU/hr ft}^2 \text{ } ^\circ\text{F}$$

$$h'_{\text{tube}} = \frac{K}{\text{wall thickness}} = 5,650 \text{ BTU/hr ft}^2 \text{ } ^\circ\text{F}$$

$$h_{\text{gas}} = \frac{K}{D} \left(\frac{C_{\text{cp}}}{K} \right)^{.5} \left[.47 \left(\frac{DV\rho}{\mu} \right)^{.52} + .35 \right] = 99.9 \text{ BTU/hr ft}^2 \text{ } ^\circ\text{F}$$

for cross flow of gas around the tube at $V=1863 \text{ ft/sec}$

$$T_{\text{gas}} = 4000 \text{ } ^\circ\text{R}$$

The overall heat transfer coefficient was computed to be

$$U = 96.7 \text{ BTU/hr ft}^2 \text{ } ^\circ\text{F}$$

or

$$q = .649 \text{ BTU/in}^2 \text{ sec.}$$

and the bulk temperature rise of the water

$$\frac{q A}{h_{\text{water}}} = 50.1 \text{ } ^\circ\text{F.}$$

These calculations were based upon the estimate that the thermal conductivity of inconel could be represented by the conductivity of stainless steel.

Under these conditions the gas flow becomes $m = .0001 \text{ 845 lb/see}$ and the heat transferred to the inner cooling annulus becomes negligible.

It is desired to quench the temperature of the incoming gas quickly to insure frozen equilibrium. However, the temperature must be high enough that the gaseous water combustion products will not condense at the pressures inside the system.

Under the pressures and mass flows experienced in the probe itself, it is apparent that the Reynolds number is such as to place the flow in the laminar regime. Hence the film coefficient h was computed from the equation (7):

$$\frac{hD}{\mu} = 1.86 \left(\frac{DW_e}{\mu} \right)^{1/3} \left(\frac{\mu C_p}{k} \right)^{1/3} \left(\frac{P}{P_0} \right)^{.14} \left(\frac{D}{L} \right)^{1/3}$$

Viscosity values at reduced pressures were taken from the dissertation of Dr. Irvin Glassman (8).

$$h_{4000 \text{ Gas}} = \frac{9.45}{L^{1/3}} \quad \text{and} \quad h_{750} = \frac{5.24}{L^{1/3}}$$

$$U_{4000} = \frac{9.45}{L^{1/3} + .00303} \quad U_{750} = \frac{5.24}{L^{1/3} + .001691}$$

utilizing McAdams counter flow heat transfer equation:

$$\text{BTU/hr} = A \left[\frac{U_1 \Delta T_2 - U_2 \Delta T_1}{\ln \frac{U_1 \Delta T_2}{U_2 \Delta T_1}} \right] \quad \text{where } A = \pi DL \text{ ft}^2$$

L was determined to be 21 inches. This does not represent an exact solution but an estimate of the exit temperature. The actual solution would have placed the length at about 23 inches, but since the gas temperature can be controlled electrically along the incoming gas line it was decided that 21 inches represented the most convenient size from a mechanical viewpoint.

Mechanical considerations dictated a bend radius of 1.45 to 1.5 inches. At the minimum bend radius of 1.45 the diameter to wall thickness as compared to the bend radius to diameter placed the outside tube in the near region of requiring internal support, while the two internal tubes were not critical.

The probe was assembled in correct alignment and a commercial bending alloy (with a very low melting point) was poured into it to assure retention of the alignment.

The probe was calibrated up to 1500° F in an electrically heated furnace. Compressed air was run through a coil of heated copper tubing immersed in the furnace, the temperature measured eight inches from the exit by a calibrated chromel-alumel thermocouple, and past the inserted probe tip. Every precaution was taken to avoid radiation errors from the water cooled probe tip. Both the calibrating and the probe thermocouples were calibrated against standard National Bureau of Standards Platinum - Platinum Rhodium Thermocouples. The values obtained showed absolutely negligible deviations from the Leeds and Northrup thermocouple tables.

The approach velocity was maintained at from 20 to 60 feet per second. This was to prevent the possibility of the stream tube scrubbing against the water cooled tip and hence introducing an erroneously cooler measured temperature. The general criterion is that the approach Mach number must be such that the free stream tube area, A_0 , must be

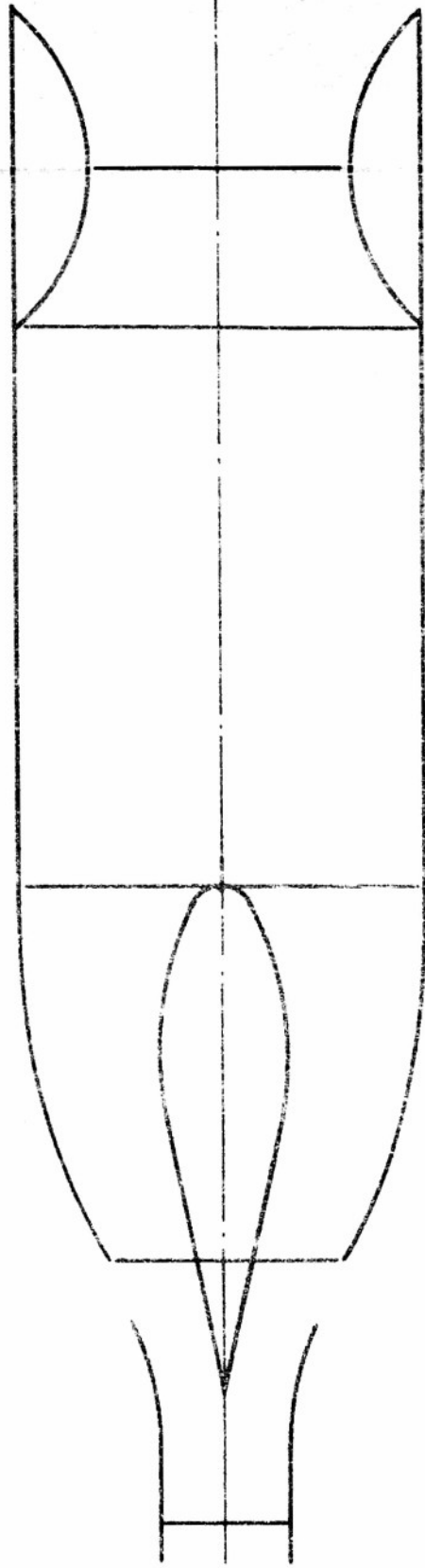
less than the frontal area of the probe (5):

$$A_0 = \frac{A_1}{K_0} \left[\frac{1 + \frac{\gamma - 1}{2} M_0^2}{\frac{\gamma + 1}{2}} \right]^{\frac{\gamma + 1}{\gamma - 1}}$$

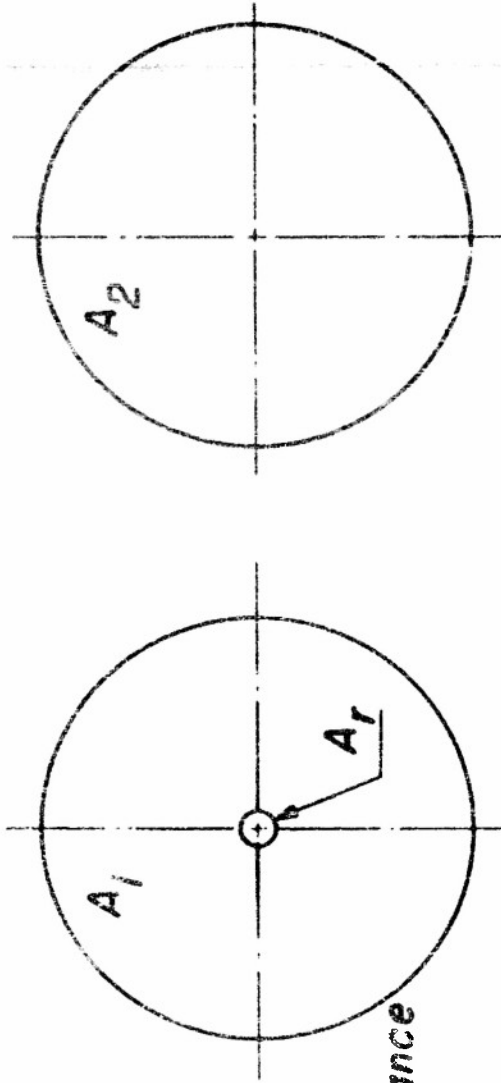
Beyond 1500° F the temperature was calibrated utilizing the Sodium D line method. Results of this calibration indicate a deviation of less than one percent.

One precaution to be noted is that since the probe is extremely sensitive to orifice area, particles of carbon or oxides which might block this area can introduce serious errors. Another consideration is that the probe will indicate an average temperature. To reduce the response time the "lead in" tube should be the minimum diameter consistent with the mass flow.

Ram Rocket Configuration



0 1 2 n



0 = Free Stream

1 = Inlet

r = Rocket

1 = Combustor Entrance

2 = Combustor Exit

n = Nozzle

FIGURE 1

Test Equipment Layout

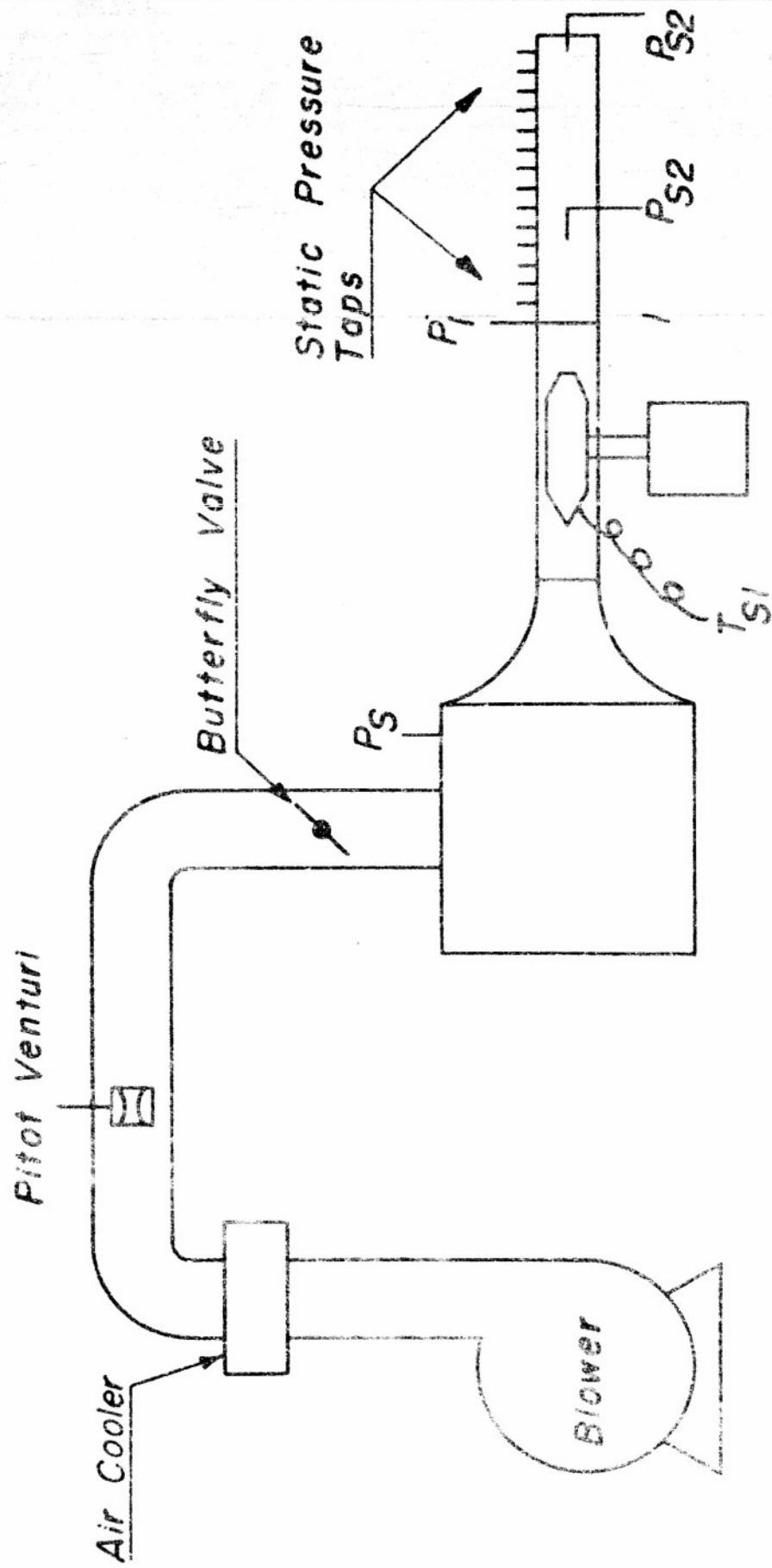


FIGURE 2

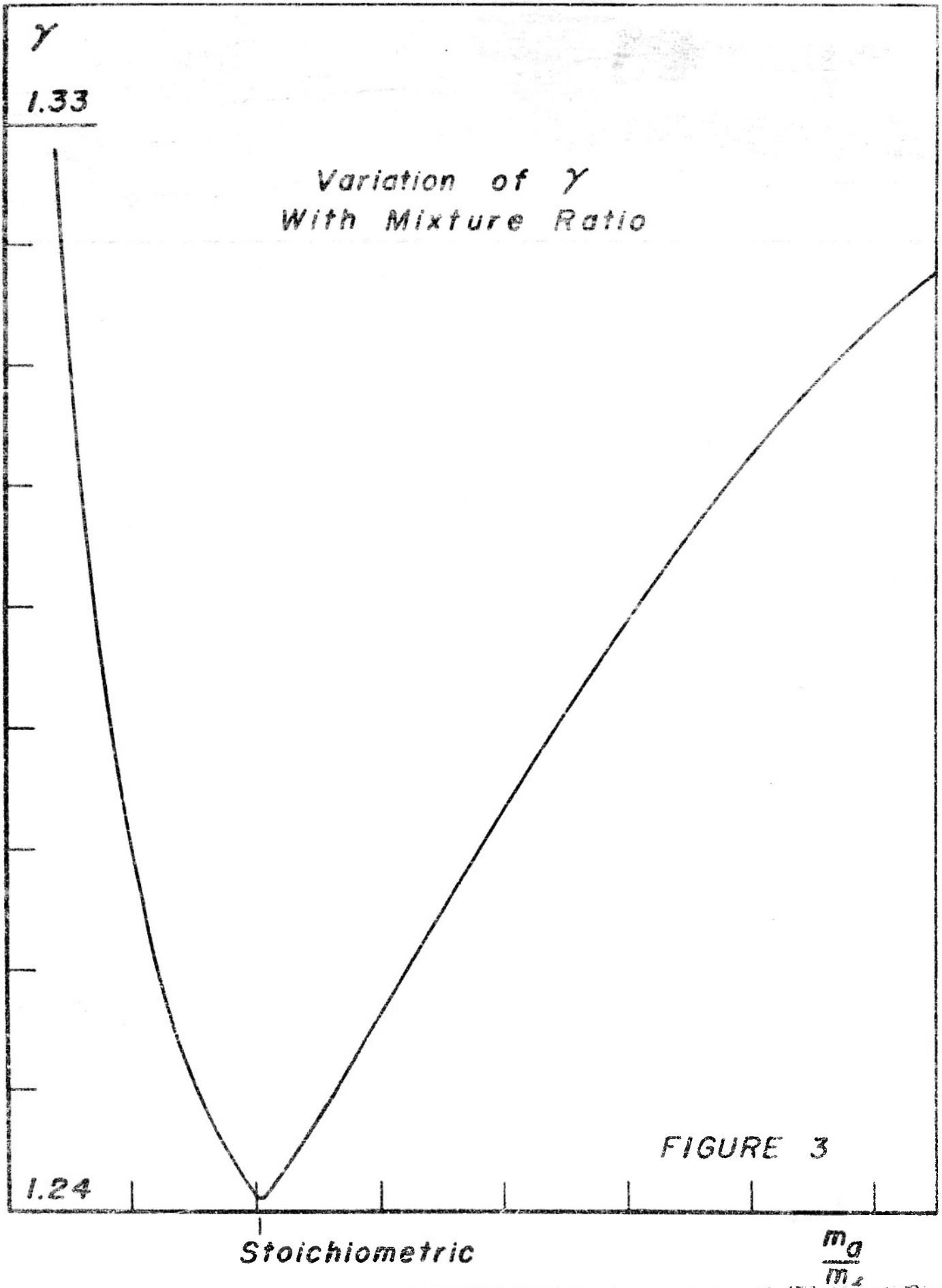
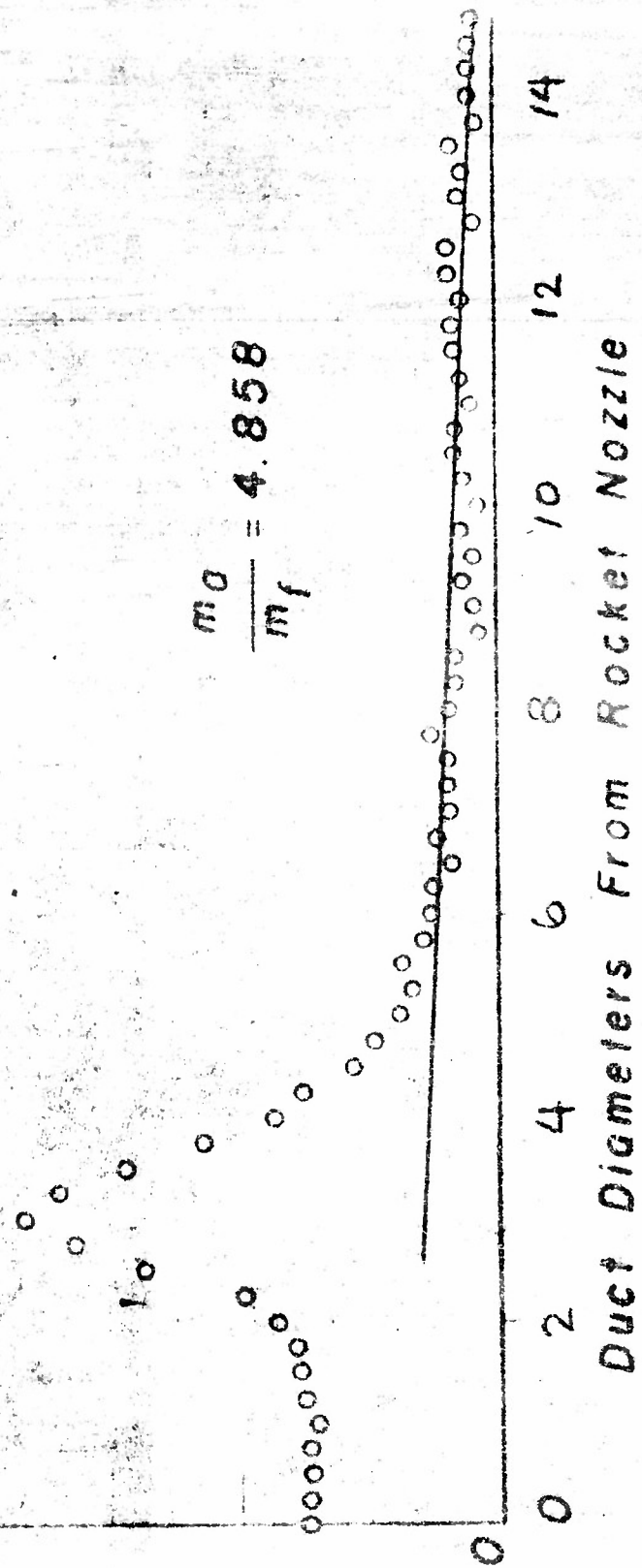


FIGURE 4

STATIC PRESSURE DISTRIBUTION
MIXING AND COMBUSTION

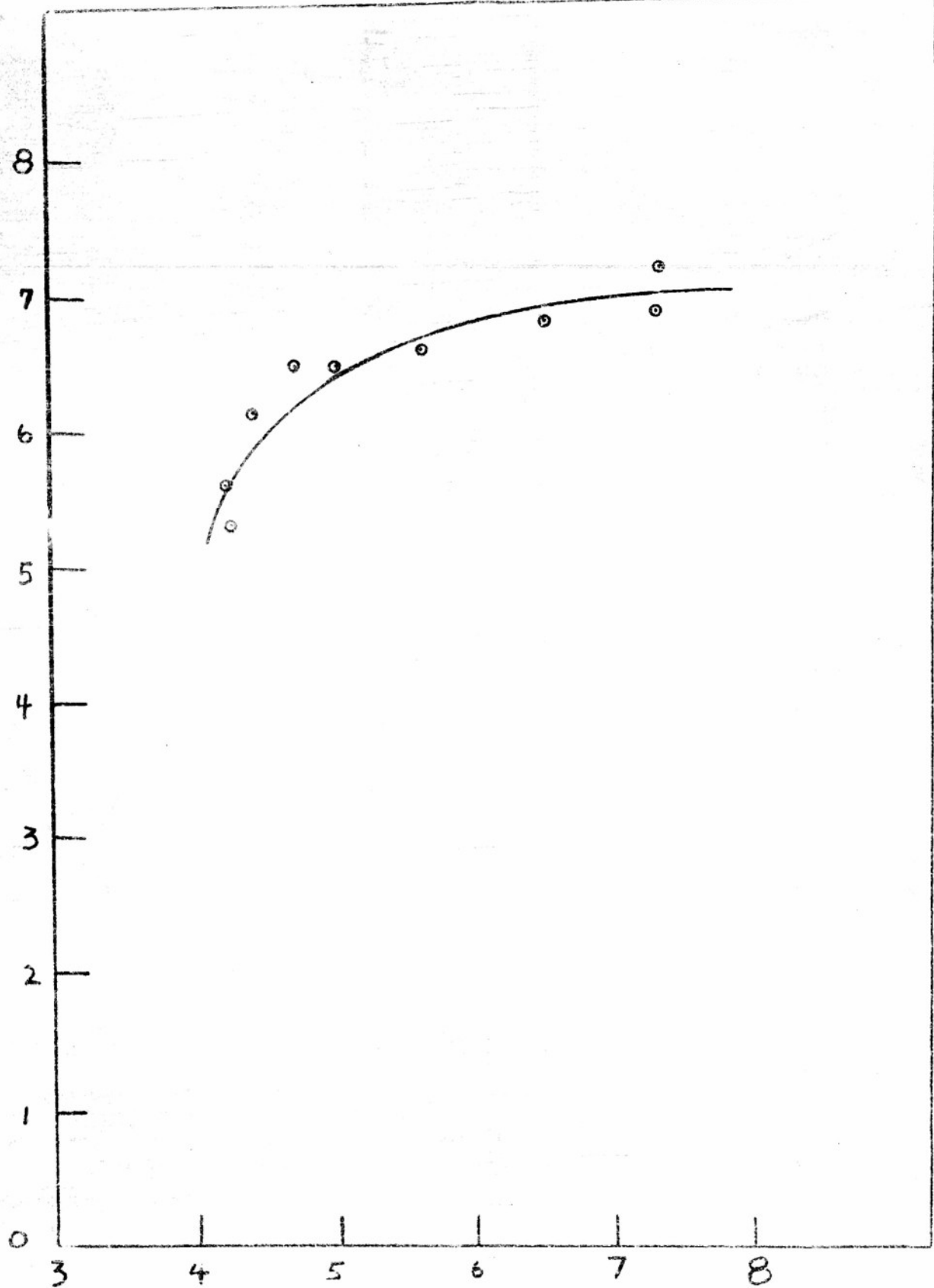
Static Pressure - Inches H₂O Gage

$$\frac{m_o}{m_f} = 4.858$$



Duct Diameter

DUCT DIAMETERS (90% COMPLETION OF COMBUSTION)



m_a/m_f

FIG 5

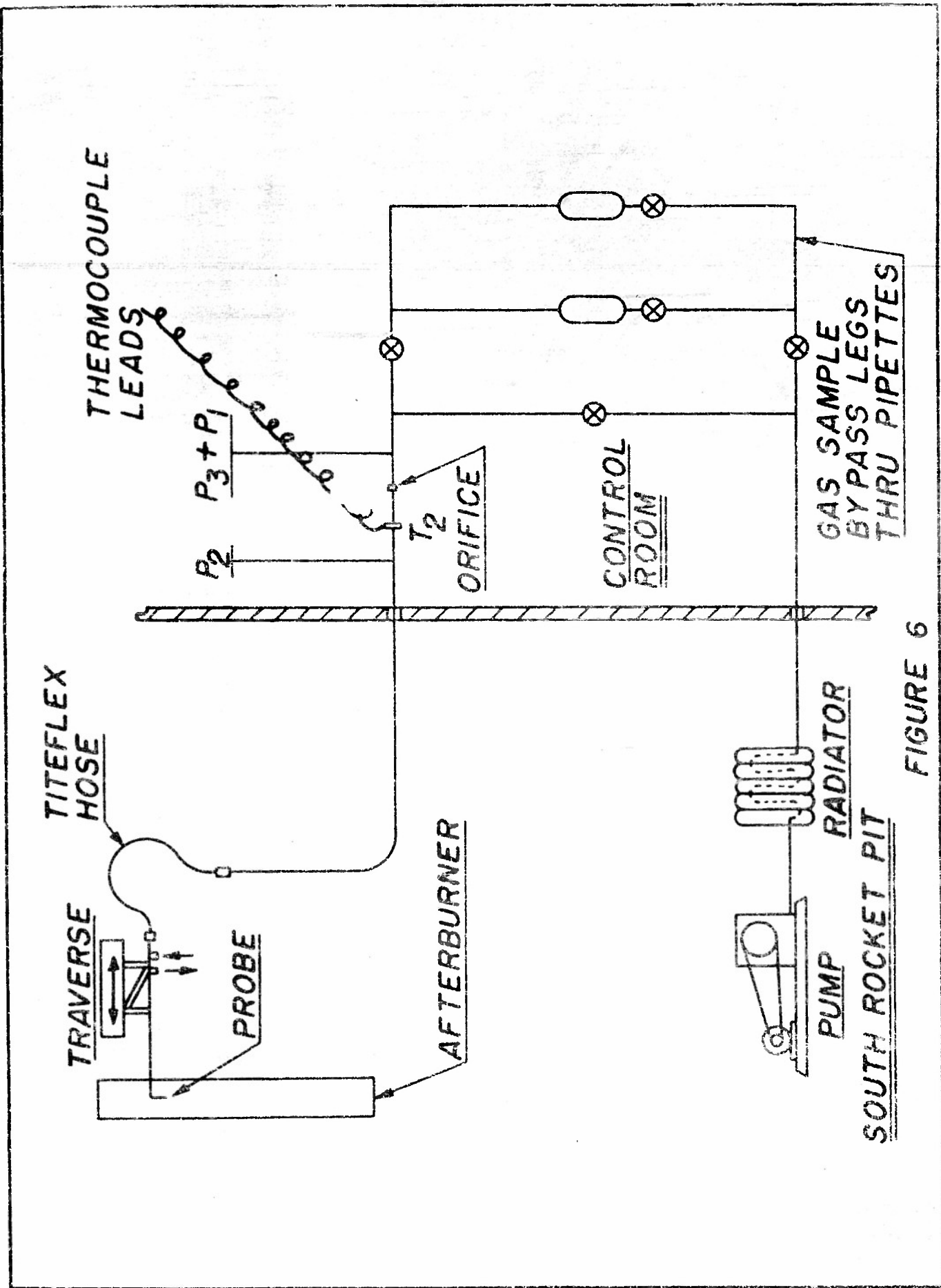
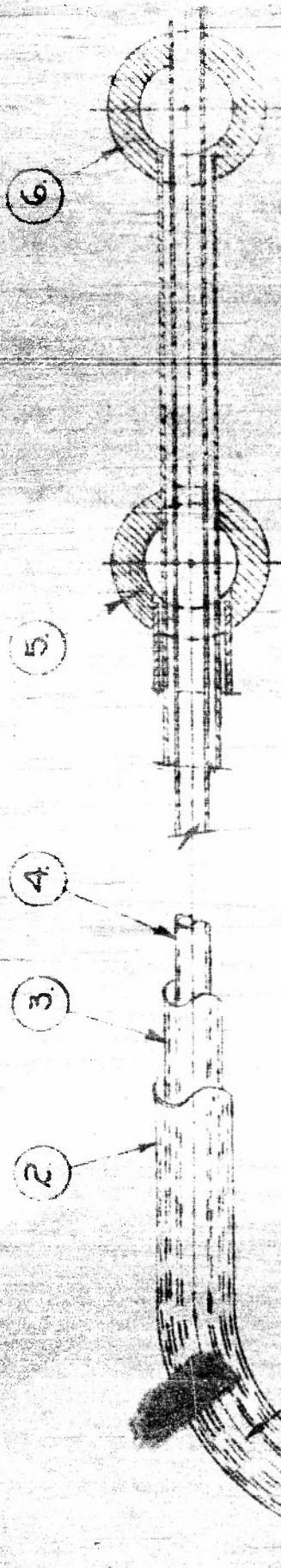
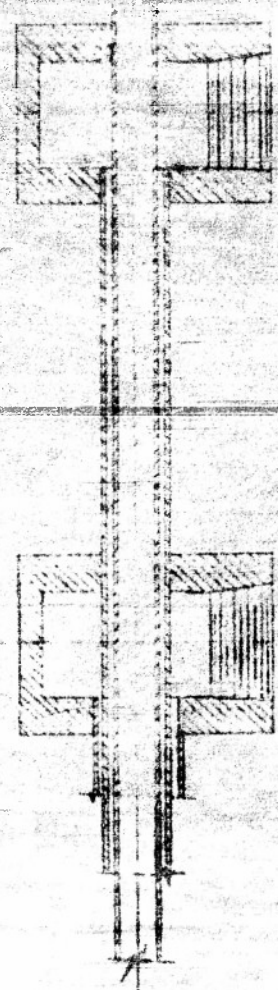
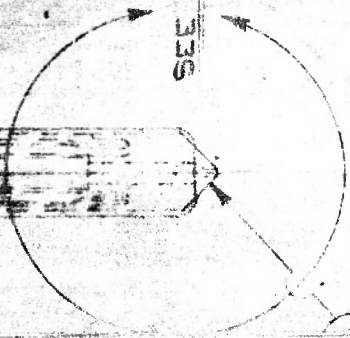


FIGURE 6



1.45" R

SEE DETAIL A - DRG. J.P. 1-MO 33 B

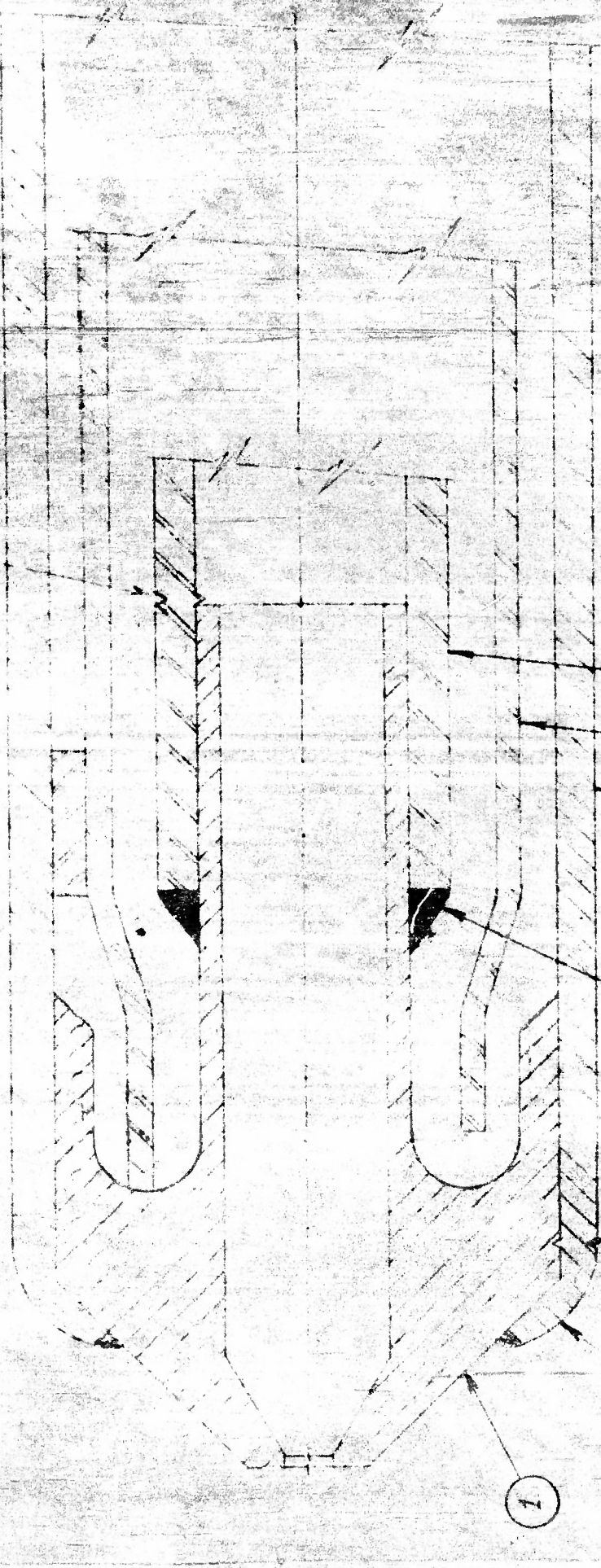


14.75"

3.00"

FIG 7

PRICK PUNCH MARK PERMISSIBLE
AT THIS POINT TO PREVENT PART 1
FROM SLIDING IN WHILE SOLDERING



1

2 & LOCK

ENTER PUNCH
3 @ 120°

2 3 4

SILVER SOLDER.

SECTION "AA"

FIG 8

Armed Services Technical Information Agency

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