AIR FILM COOLING OF A METAL SURFACE
EXPOSED TO HIGH TEMPERATURE AND HIGH
VELOCITY GASES

A Thesis
Submitted to the Graduate Faculty of the
University of Minnesota

by

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TABLE OF CONTENTS

SUMMARY

The object of this investigation was to study the variation in cooling effectiveness with variation in number and location of film cooling air orifices in a metal surface exposed to high temperature and high velocity gases.

Tests were made on one flat surface of a test blade consisting of two flat parallel surfaces connected by circular arcs at the leading and trailing edges. The blade was mounted parallel to the hot gas flow. Ten cooling orifice configurations were tested employing various cooling air and hot gas rates of flow.

The following observations were made:

Increasing the number of cooling holes increased the cooling of the test blade for a given cooling air flow.

A uniform temperature reduction over the test blade required cooling air orifices over the entire blade. This results from the rapid downstream dissipation of the effectiveness of the cooling air film.

There was no appreciable difference in the cooling air effectiveness for gas Mach Numbers of 0.775 and 1.0.

INTRODUCTION

A profitable way to realize increased efficiencies and outputs for gas turbines is to increase the working temperature of the hot inlet gases. This means that, with existing turbine blade materials, some method of blade cooling must be employed.

Commander D. O. Ness, in a Thesis "Boundary Layer Control as a Method of Gas Turbine Blade Cooling" (Ref. 1), experimentally investigated the feasibility of cooling gas turbine blades by the introduction of a controlled boundary layer of cool air over the blade surface.

The test blade of Ref. 1 was a solid Jumo 004 turbine blade with cooling air supplied to a row of six holes on the upper blade surface and a row of six holes on the lower blade surface from a spanwise drilled passage at 30 per cent of the blade chord. These 12 cooling air holes were 1/16 inch in diameter and spaced 7/16 inches apart.

A maximum leading edge temperature reduction of 280° F and a trailing edge temperature reduction of 140° F was obtained at a gas Mach No. of 0.77 and 1600° F. The cooling airflow was 2 per cent of the burner airflow, based on a full scale J-33 jet engine.

This investigation is an attempt to supply the following additional information:

The effect of varying the number and location of cooling air holes for a given air flow and gas flow.

The effectiveness of the boundary layer in cooling the area downstream of the last row of cooling air holes.

The effect of an increase in cooling air flow on the test blade chordwise temperature gradient aft of the cooling air holes.

Whether the temperature reduction is primarily caused by the conductive cooling process of cool air flowing inside the blade or whether the cool boundary layer film on the external blade surface contributes appreciably to blade temperature reduction.

The effect of varying the gas Mach No. for a given configuration of cooling air holes.

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Professors N. A. Hall, T. E. Murphy, and K. E. Neumeier of the Mechanical Engineering Department, for their assistance, advice, and suggestions.

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LCDR E. T. LaRoe, U.S.N., for his assistance in conduction of test runs.

TEST EQUIPMENT

Test Blade

It was decided to depart from the standard method of testing a three blade static cascade of actual turbine blades and test a larger single simplified blade which would facilitate the installation of internal cooling air passages and thermocouples.

The blade was made of mild steel to eliminate drilling and welding problems.

Thin sheet metal was employed to minimize the difficulty of drilling many small holes and to allow rapid temperature stabilization while testing.

The blade was made flat sided and with zero camber to insure constant gas static pressure over the blade chord. This was necessary to keep the amount of air flowing from each orifice approximately constant.

One surface of the blade was cooled because of internal space restrictions in the blade.

The above simplification restricts the test results to non-quantitative comparisons with actual turbine blades. An attempt was made to adhere as closely as possible to actual turbine blade conditions by making the cooled surface of the test blade

approximately equal in area to the curved area of both sides of an actual J-33 turbine blade.

Figures 2, 16, and 17 show the test blade which was mounted parallel to the direction of flow in the hot gas duct as illustrated in Figures 14 and 15.

Gas welding was employed in fabrication. The sheet metal was mild steel of relatively high thermal conductivity and 0.0689 in. thick. The tubes were standard 1/8 inch mild steel pipe (0.068 in. wall thickness, 0.405 in. O.D., and 0.269 in. I.D.).

Six themocouples were attached to the upper blade surface in a chordwise direction $1\frac{1}{2}$ ins. from the blade tip by machine screws through six access holes in the lower surface. The access holes were then closed with flush screw plugs. (See Figures 2 and 17)

For configuration (A) 5 rows of cooling air holes 9/16 in. apart were used. There were 17 holes in each row, $\frac{1}{4}$ in. apart.

For configurations (B), (C), and (D), 165 holes were employed; 5 rows, 9/16" apart as in configuration (A), but with 33 holes per row. The spanwise hole spacing was 1/8 in. See Figures 2, 16, and 17.

All cooling air holes were .040 inches in diameter (No. 60

Drill) and were drilled at right angles to the blade surface.

The 1/8 in. standpipe on the upper surface at the blade root and the 1/8 in. pipe extension at the blade tip were not used in the test runs. They were originally attached to determine the spanwise variation in cooling air pressure and temperature in the central cooling air passage.

Hot Gas System (See Figures 3 and 14)

Burner air was metered through a seven inch rounded orifice. The air was then ducted to the inlet side of an Allison V-1710 aircraft engine supercharger which was employed as a source of compressed air for the J-33 combustion chamber. The supercharger was the only load on the unsupercharged Allison engine.

After leaving the supercharger, the air was led to a J-33 combustion chamber where the air and fuel oil were burned to supply hot gas to the duct in which the test blade was mounted.

A bypass valve, in conjunction with an electrically driven fuel oil pump, was used to control gas temperatures.

The burner airflow was regulated with the throttle on the Allison engine.

Cooling Air System

Air was supplied by the laboratory central compressed air supply. The flow was measured by an air rotameter and regulated by a needle valve in the line (see Figures 3 and 13). The pressure and temperature were measured at the rotameter to correct the observed air flow to standard conditions. The capacity of the Fischer and Porter air rotameter was 8.5 C.F.M. at zero pounds per square inch gage and 100° F.

Total test section temperature was approximated with a radiation shielded stagnation thermocouple suitably ventillated to provide a low velocity gas flow over the thermocouple bead. Figure 15 shows attachment of test section temperature, statio pressure, and total pressure lines.

TEST PROCEDURE

The fuel oil pressure and the Allison engine R.P.M. were varied to control test section temperature and gas flow.

Cooling air flows were regulated by adjusting air rotameter readings with the needle valve to conform with values dictated by temperature and pressure conditions as indicated in Figures 4 and 5.

Test runs were made with the configurations shown in Figure 1.

TEST RESULTS

The results of this investigation are presented in Tables I, II, and III and the curves of Figures 6 through 12.

Figures 6 and 7 show gas and blade temperatures plotted against blade chord position at the 2/3 span from root position.

Figures 8 and 9 are plots of the above temperatures subtracted from the uncooled blade temperature at corresponding locations. These curves are labeled "Blade Temperature Reductions". Each group of curves represents a given cooling and gas flow configuration.

Figures 10, 11, and 12 are replots of the data of Figures 8 and 9 with each group of curves representing a given number of rows of cooling air holes rather than a given cooling air and gas flow configuration as in Figures 8 and 9.

CONCLUSIONS

Within the test limits of this investigation, it was found that film cooling was effective only near cooling air exit orifices. This is indicated by the sharp increase in blade temperature after the last row of cooling air holes.

Doubling the number of cooling air exit holes increased the internal cooling passage area by approximately 11 per cent. This increase in internal area did not seem to be large enough to account for the lowered blade temperatures of configuration (B). Therefore, the major portion in the increase in temperature reduction resulting from doubling the number of cooling air holes must be due to the flow of cooling air over the external blade surface in the vicinity of the cooling air holes.

Increasing the coolant flow or the number of cooling air holes per row did not improve the test blade temperature gradient as indicated by the slopes of the cooling curves downstream of the last row of operating cooling air orifices.

To uniformly cool the surface it was necessary to distribute cooling air holes over the entire surface.

Increasing the gas Mach No. from 0.775 to 1.0 did not seem to have any measurable effect on test blade cooling within the limits of accuracy of the investigation.

Increasing the cooling air flow increased the blade temperature reduction. The rate of increase in blade temperature reduction with increasing cooling air flows for a typical turbine blade is indicated in Figure 16 of Reference 1.

The results of this investigation suggest that this method of turbine blade cooling is not as beneficial as expected; since turbine blades, to be efficient, must be very thin from mid-chord aft. The abrupt rise in temperature following the last row of cooling holes indicated that the trailing edge would receive small benefit from cooling air ejected from a point foreward of mid-chord.

BIBLIOGRAPHY AND REFERENCES

(1) "Boundary Layer Control as a Method of Gas Turbine Blade Cooling"; Ness, D.O., CRD. U.S.N.; A Thesis Submitted to the Graduate Faculty of the University of Minnesota; August, 1949.

(2) Smith, G.G.; "Gas Turbines and Jet Propulsion for Aircraft", 1946, Fourth Edition, Aircraft Books, Inc., New York.

(3) Godsey, F.W.Jr., and Young, L.A.; "Gas Turbines for Aircraft", 1949, First Edition, McGraw-Hill Book Co., Inc., New York.

(4) "Temperatures and Stresses on Hollow Blades for Gas Turbines"; Pollmann, E., N.A.C.A. T.M. No. 1183, September 1947, Washington, D.C.

(5) "N.A.C.A. Investigations of Gas Turbine Blade Cooling"; Ellerbrock, H.H.; Institute of Aeronautical Sciences, Reprint No. 117.

(6) Vincent, E.T.; "Theory and Design of Gas Turbines and Jet Engines", 1950, First Edition, McGraw-Hill Book Co., New York.

(7) Keenan, J.H. and Kaye, Jr.; "Thermodynamic Properties of Air, Products of Combustion, and Component Gases", 1948, John Wiley and Sons, Inc., New York.

APPENDIX

NOMENCLATURE

$Q_{C.A.}$	Cooling airflow	C.F.M.
$P_{C.A.}$	Cooling air press. at rotameter	INS. HG. ABS.
P_1-P_5	Cooling air press. at test blade	INS. HG. GAGE
P_S	Test section static press.	INS. HG. ABS.
P_T	Test section total press.	INS. HG. ABS.
$P_{B.A.}$	Press. drop across burner air metering orifice	INS. H_2O
$T_{T.S.}$	Test section temperature	^oF
$T_{B.A.}$	Temp. of burner air at orifice	^oF
$T_{C.A.}$	Temp. of cooling air at rotameter	^oF
T_1-T_6	Test blade temperatures (see Fig. 2)	^oF
W_F	Burner fuel flow	LBS./HR.
$M_{T.S.}$	Mach No. at test section (no blocking)	
M_B	Mach No. at test blade (including blocking effect of blade)	
$W_{C.A.}$	Cooling air flow	LBS./MIN.
$W_{B.A.}$	Burner air flow	LB./HR.

SAMPLE COMPUTATIONS

Burner Air Flow

$$W_{B.A.} = W_{\substack{29.92"Hg \\ 100^{\circ} F}} \left[\frac{560}{T_{C.A.}} \times \frac{P_{C.A.}}{29.92} \right]^{\frac{1}{2}}$$

The W for 29.92 ins. Hg and 100° F published in Aerofin charts for 7 in. orifice.

$$W_{B.A.} = 10,570 \left[\frac{560}{564} \times \frac{29.16}{29.92} \right]^{\frac{1}{2}}$$

$$= 10,220 \text{ LB/HR.}$$

Mach Numbers

Obtained from Gas Tables knowing

Pressure ratios and Area ratios

Area of duct at working section = 15.75 in.^{2}

Area of duct at working section less area of blade

= 14.06 in.^{2}

Cooling Air Flows

Read directly from Figures 4 and 5.

ACCURACY OF RESULTS

Gas Temperature	1500° F $\pm 10^\circ$
(Approximates Total Temp.)	
Blade Temperatures	$\pm 10^\circ$
Cooling Air Flow	$\pm$ 2%
Assuming Errors .05 C.F.M. Rotameter reading 1.0 in. HG. in $P_{C.A.}$ 5° in $T_{C.A.}$	
Burner Air Flow	$\pm$ 1.4%
Assuming Errors 0.1 in. H_2O in $P_{B.A.}$ 5° in $T_{B.A.}$ Exact orifice pressure	
Fuel Flow	$\pm$ 0.5%
Assuming error of 1 LB/HR in rotameter reading	
Mach Number at M = 1.0	None
Choked	
Mach Number at M = 0.775	$\pm$3%
Assuming Errors 0.5 in. HG. in P_T 1.5 in. H_2O in P_S γ assumed to be 1.3	

The test runs were not duplicated except in instances where there appeared to be inconsistencies in the results. For example the chordwise drop in temperature in configuration (A) with no cooling air and the apparently improved cooling at a higher Mach No. in comparing the results of configurations (C) and (D) with all cooling air holes in operation. Consistent results elsewhere and the expense of operating the Allison engine and J-33 combustion chamber were the primary reasons for the limited duplication of test runs.

It was originally planned to measure the cooling airflow by means of a 3/8 inch sharp edged orifice, but it was later decided that the low cooling air flow settings could be made more rapidly and accurately with an air rotameter. The rate of flow of the rotameter was checked against the rate of flow of the orifice with the following results:

Orifice	0.231 LB/MIN.
Rotameter	0.224 LB/MIN.

This represents approximately a three per cent discrepancy.

TABLES, GRAPHS, AND ILLUSTRATIONS

TABLE I

AVERAGE TEST DATA AND RESULTS

	CONFIG.(A)	CONFIG.(B)	CONFIG.(C)	CONFIG.(D)
BAROMETER INS. HG.	29.4	29.16	29.16	29.10
P_S INS. HG. ABS.	28.77	28.57	28.53	28.51
P_T INS. HG. ABS.	40.50	39.15	39.46	36.18
$M_{T.S.}$	.740	.710	.718	.612
M_B	1.0	1.0	1.0	.775
$\Delta P_{B.A.}$ INS H_2O	5.31	5.17	5.20	3.59
$T_{B.A}$ °F	90	106	104	77
$W_{B.A.}$ LB/HR.	10,719	10,320	10,220	8,860
$T_{C.A.}$ °F	85	95	95	80
$W_{C.A.}$ LB./MIN.	0.320	0.320	0.533	0.533
W_F LB/HR.	215	212	212	187
A/F RATIO	49.8	48.7	48.2	47.4
* % BURNER AIR FOR COOLING	0.688	1.19	1.205	1.39
BLADE TEMPS.	FIG. 6	FIG. 6	FIG. 7	FIG. 7
BLADE PRESSURES	TABLE II	TABLE II	TABLE III	TABLE III

* BASED ON J-33 JET ENGINE. COOLED SIDE OF TEST BLADE EQUAL IN AREA TO BOTH SIDES OF ACTUAL J-33 TURBINE BLADE.

DATA SHEET
LOW AIR FLOW

TABLE II

NOTE. $P_{C.A.}$, P_S, P_T, AND P_1-P_5 ARE GAGE PRESSURES

REMARKS	COOLING AIR FLOW C.F.M. ($Q_{C.A.}$)	COOLING AIR PRESS IN. HG. ($P_{C.A.}$)		COOLING AIR PRESSURE AT BLADE P_1 (IN. HG.)	P_2	P_3	P_4	P_5		TEST SECTION STATIC PRESS IN. H_2O (P_S)	TEST SECTION TOTAL PRESS IN. HG. (P_T)		BURNER AIR PRESS DIFF IN. H_2O ($\Delta P_{B.A.}$)
7/25/50 CONFIG A) BAROM. 29.40" @ 80°F – 85 COOLING AIR HOLES TOTAL 0.320 LB/MIN. COOLING AIR – M = 1.0													
C.A. PASSAGE 1-5	3.77	11.6		-1.35	-1.40	-1.35	-1.4	-1.8		-.[illegible]	10.6		5.5
" 1-4	3.77	11.6		-.45	-.65	-.70	-1.5			-1.[illegible]7	11.0		5.3
" 1-3	3.65	14.1		+1.1	+1.1	+1.1				[illegible].5	11.0		5.4
" 1-2	3.52	17.7		+5.4	+6.7					-8.5	11.0		5.3
" 1	3.0	35		+11.0						-[illegible]1.0	11.5		5.2
NO COOLING AIR										-9.0	11.5		5.2
7/28/50 CONFIG (B) BAROMETER 29.16" @ 90°F – 165 COOLING AIR HOLES TOTAL – 0.320 LB/MIN COOLING AIR – M = 1.0													
C.A. PASSAGES 1-5	3.8	13.7		0	0	+0.1	0	-0.15		-9.0	10.0		5.1
" 1-4	3.75	14.1		+0.4	+0.6	+0.8	+0.5			-8.5	10.0		5.2
" 1-3	3.7	14.7		+1.2	+1.2	+1.4				-7.0	9.6		5.1
" 1-2	3.6	17.4		+2.8	+2.8					-10.5	10.1		5.2
" 1	3.22	28.5		+12.0						-7.0	10.0		5.2
NO COOLING AIR										-6.6	10.0		5.2

REMARKS	TEST SECTION TEMP. °F ($T_{T.S.}$)	BURNER AIR TEMP. °F ($T_{B.A.}$)	COOLING AIR TEMP °F ($T_{C.A.}$)	BLADE TEMPERATURES T_1	T_2	T_3	T_4	T_5	T_6	BURNER FUEL FLOW LB/HR (W_F)
7/25/50	CONFIG. (A)									
C.A. PASSAGES 1-5	1500 ±10	85	85	1370	1300	1250	[illegible]	1180	1230	218
" 1-4	"	85	85	[illegible]350	1260	1210	[illegible]80	1180	1255	217
" 1-3	"	90	85	1330	1120	1[illegible]5	220	1235	1270	214
" 1-2	"	90	85	[illegible]310	1[illegible]90	11[illegible]	245	1230	1260	215
" 1	"	95	85	1280	1210	1280	[illegible]80	1250	1260	214
NO COOLING AIR	"	95	85	1420	1410	1390	[illegible]350	1310	1305	214
7/28/50	CONFIG. (B)									
C.A. PASSAGES 1-5	1500 ±10	105	95	1315	1[illegible]35	[illegible]200	[illegible]80	[illegible]160	1220	212
" 1-4	"	105	95	1295	1200	1150	[illegible]	1190	1285	215
" 1-3	"	110	95	1285	11[illegible]	1130	[illegible]90	1270	1310	206
" 1-2	"	105	95	1270	1[illegible]45	1145	1260	1300	1330	214
" 1	"	100	95	1235	1165	1275	[illegible]320	1335	1345	213
NO COOLING AIR		100	95	1410	1420	1410	1400	[illegible]400	1400	214

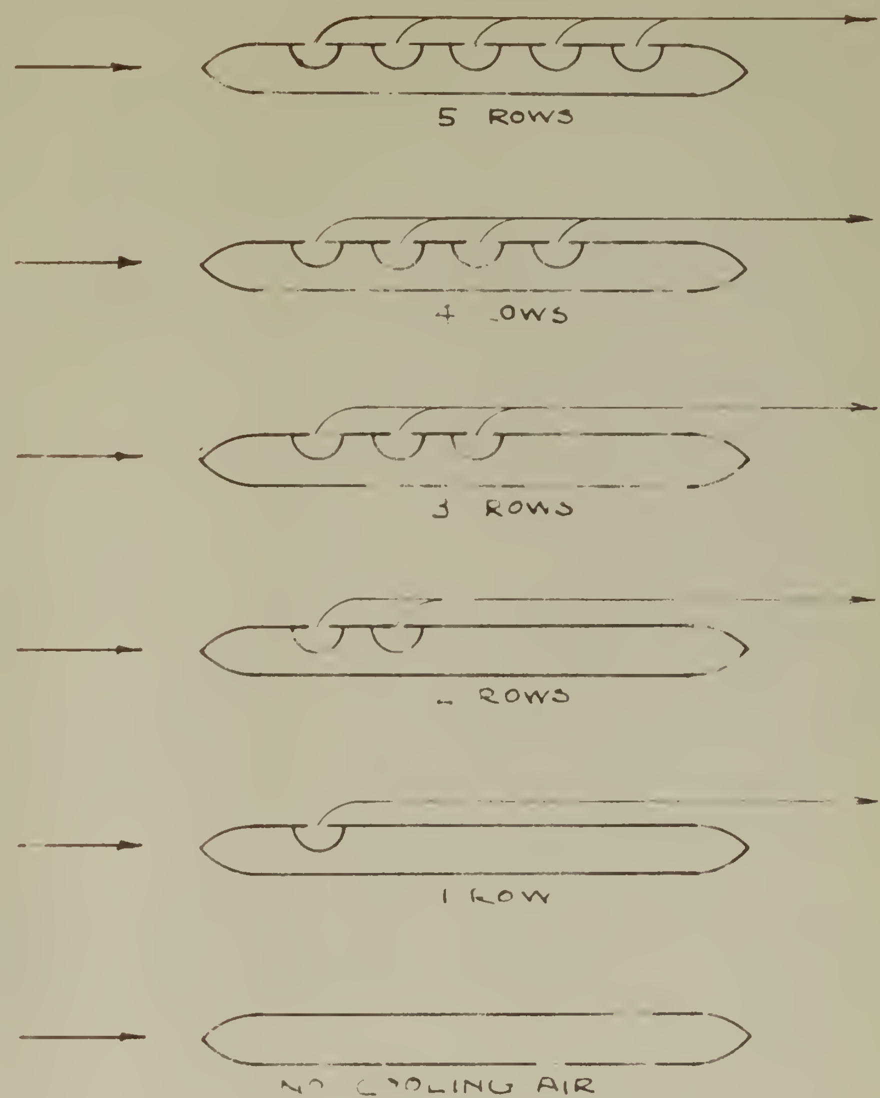
DATA SHEET
HIGH AIR FLOW

TABLE III

NOTE: $P_{C.A.}$, P_S, P_T AND P_1-P_5 ARE GAGE PRESSURES

REMARKS	COOLING AIR FLOW C.F.M. ($Q_{C.A.}$)	COOLING AIR PRESS IN. HG. ($P_{C.A.}$)		COOLING AIR PRESSURE AT BLADE P_1 IN. HG.	P_2	P_3	P_4	P_5		TEST SECTION STATIC PRESS IN. H_2O (P_S)	TEST SECTION TOTAL PRESS IN. HG. (P_T)		BURNER AIR PRESS DIFF IN. H_2O ($\Delta P_{B.A.}$)
7/28/50	CONFIG. (C)	BAROM. 29.16" Hg @ 90°F — 165 COOLING AIR HOLES TOTAL — 0.533 LB/MIN COOLING AIR — M = 1.0											
C.A. PASSAGES 1-5	5.2	32.0		-1.0	-1.0	-0.8	-1.0	-0.9		-9.0	10.0		5.3
" 1-4	5.2	33.0		+2.2	+2.2	+2.5	+2.2			-9.0	10.0		5.2
" 1-3	5.08	36.0		+4.1	+3.9	+4.2				-6.8	10.0		5.1
REPEAT 1-3	5.10	35.0		3.9	3.9	4.0				-9.7	11.0		5.2
C.A. PASSAGES 1-2	4.9	39.4		8.1	7.8					-9.6	10.2		5.2
" 1	4.22	61.0		24.9						-8.5	11.0		5.15
NO COOLING AIR										-6.6	10.0		5.20
7/31/50	CONFIG. (D)	BAROMETER 29.10" Hg @ 86°F — 165 COOLING AIR HOLES TOTAL — 0.533 LB/MIN COOLING AIR — M = .775											
C.A. PASSAGES 1-5	5.20	32.0		-1.4	+1.7	1.65	1.45	1.25		-8.5	7.2		3.55
REPEAT 1-5	5.20	32.0		+1.5	+1.6	1.70	1.45	1.30		-8.5	7.0		3.65
C.A. PASSAGES 1-4	5.15	33.0		2.5	2.4	2.55	2.20			-8.5	7.2		3.55
1-3	5.10	34.0		3.9	3.9	4.0				-7.0	7.2		3.55
1-2	4.95	39.5		7.9	7.9					-6.7	7.0		3.55
1	4.21	61.0		25.1						-7.5	7.0		3.60
NO COOLING AIR										-8.3	7.0		3.65

	TEST SECTION TEMP. °F ($T_{T.S.}$)		BURNER AIR TEMP. °F ($T_{B.A.}$)		COOLING AIR TEMP. °F ($T_{C.A.}$)		BLADE TEMPERATURES °F T_1	T_2	T_3	T_4	T_5	T_6	BURNER FUEL FLOW LB/HR (W_F)
7/28/50	CONFIG. (C)												
C.A. PASSAGES 1-5	1500 ±10		100		95		1275	1150	1080	1030	1025	1110	213
" 1-4	"		110		95		1270	1130	1060	1070	1110	1240	212
" 1-3	"		110		95		1245	1080	1030	1125	1225	1285	210
REPEAT 1-3	"		105		95		1235	1075	1010	1110	1210	1270	204
C.A. PASSAGES 1-2	"		105		95		1220	1050	1070	1220	1270	1300	216
" 1	"		100		95		1190	1110	1250	1290	1315	1330	215
NO COOLING AIR	"		100				1410	1420	1410	1400	1400	1400	214
7/31/50	CONFIG. (D)												
C.A. PASSAGES 1-5	1500 ±10		80		80		1260	1130	1070	1060	1030	1110	187
REPEAT 1-5	"		80		80		1270	1135	1080	1060	1035	1120	187
C.A. PASSAGES 1-4	"		75		80		1240	1095	1030	1035	1085	1210	187
" 1-3	"		75		80		1220	1060	1010	1095	1195	1250	187
" 1-2	"		75		80		1190	1025	1050	1200	1245	1280	185
" 1	"		75		80		1150	1070	1200	1250	1270	1290	187
NO COOLING AIR	"		77				1390	1390	1375	1375	1375	1375	186

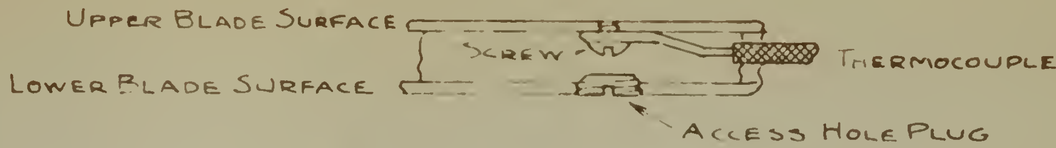


EACH OF THE ABOVE CONFIGURATIONS
WERE TESTED UNDER THE FOLLOWING CONDITIONS:

	COOLING AIR FLOW	NO. COOLING AIR HOLES	GAS MACH NO.	
(A)	.520 LB/MIN.	85	1.0	(A)
(B)	.520 LB/MIN.	165	1.0	(B)
(C)	.533 LB/MIN.	165	1.0	(C)
(D)	.533 LB/MIN.	165	.175	(D)

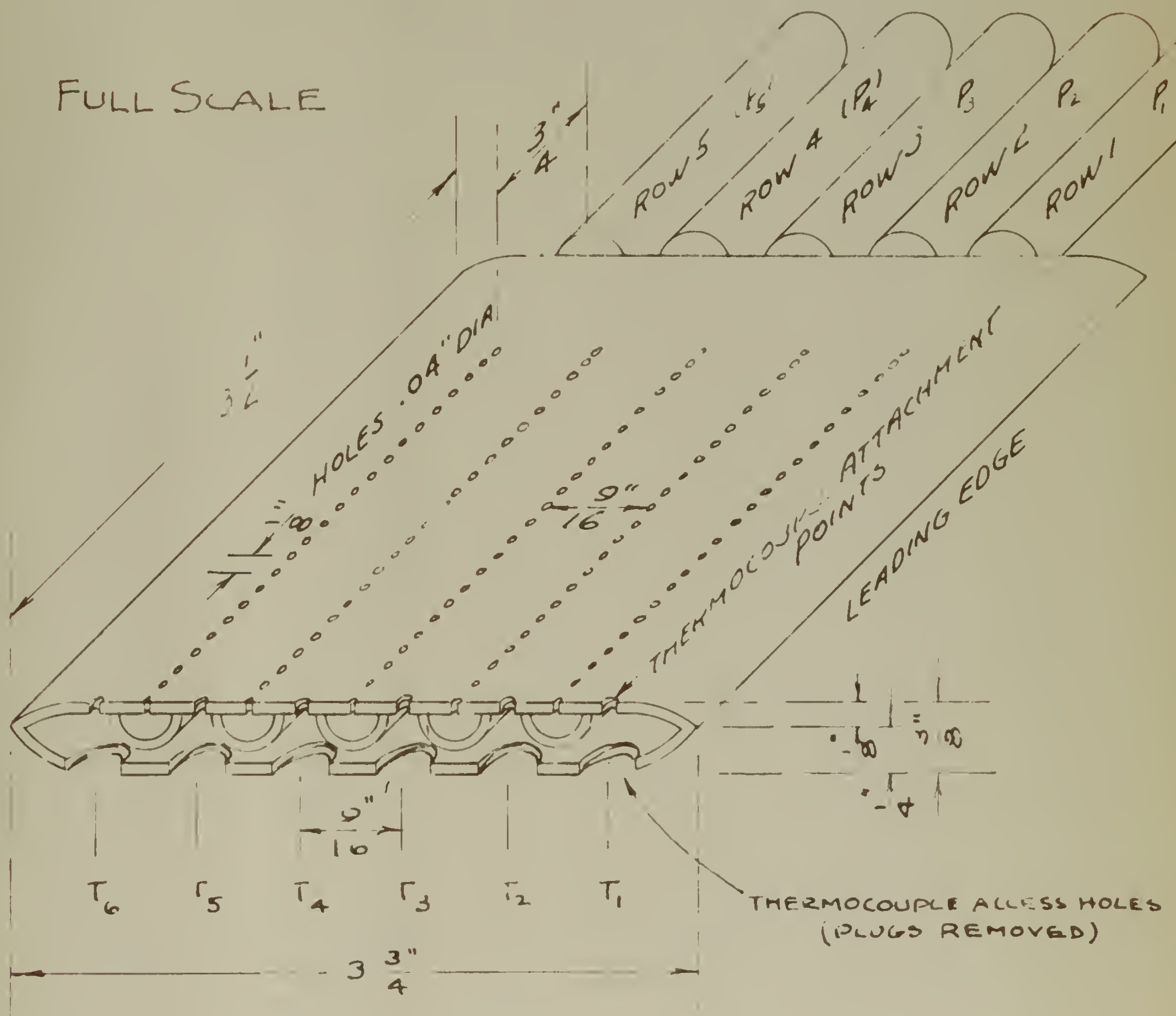
TEST CONFIGURATIONS

FIG. 1



METHOD OF ATTACHING THERMOCOUPLE TO BLADE

FULL SCALE



BLADE CROSS SECTION AT THERMOCOUPLE ATTACHMENT POINTS
165 HOLE CONFIGURATION
THERMOCOUPLE AND 1½ INCH OF BLADE TIP REMOVED
HOLES EXTEND TO ¼ INCH FROM BLADE TIP

FIG. 2

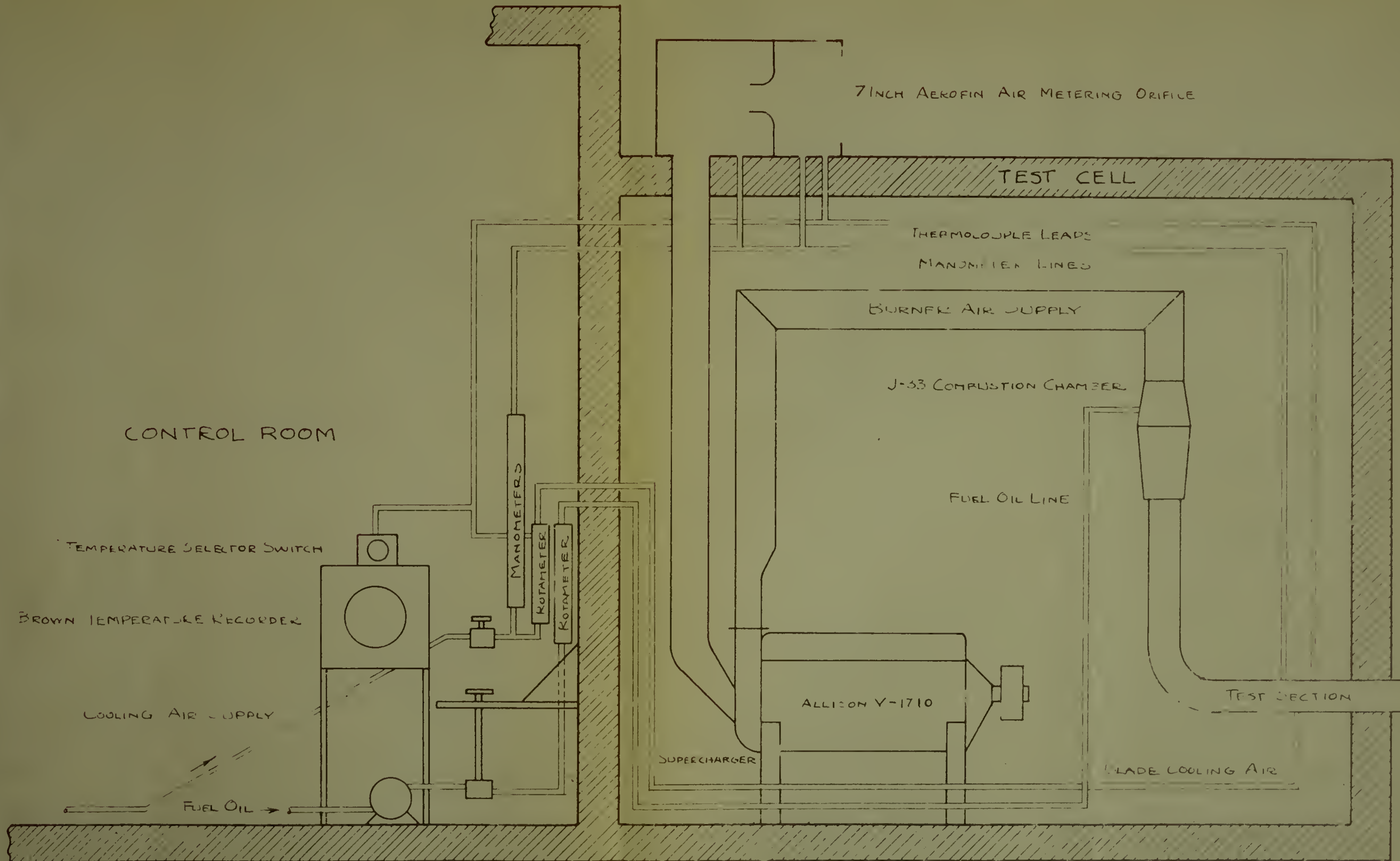
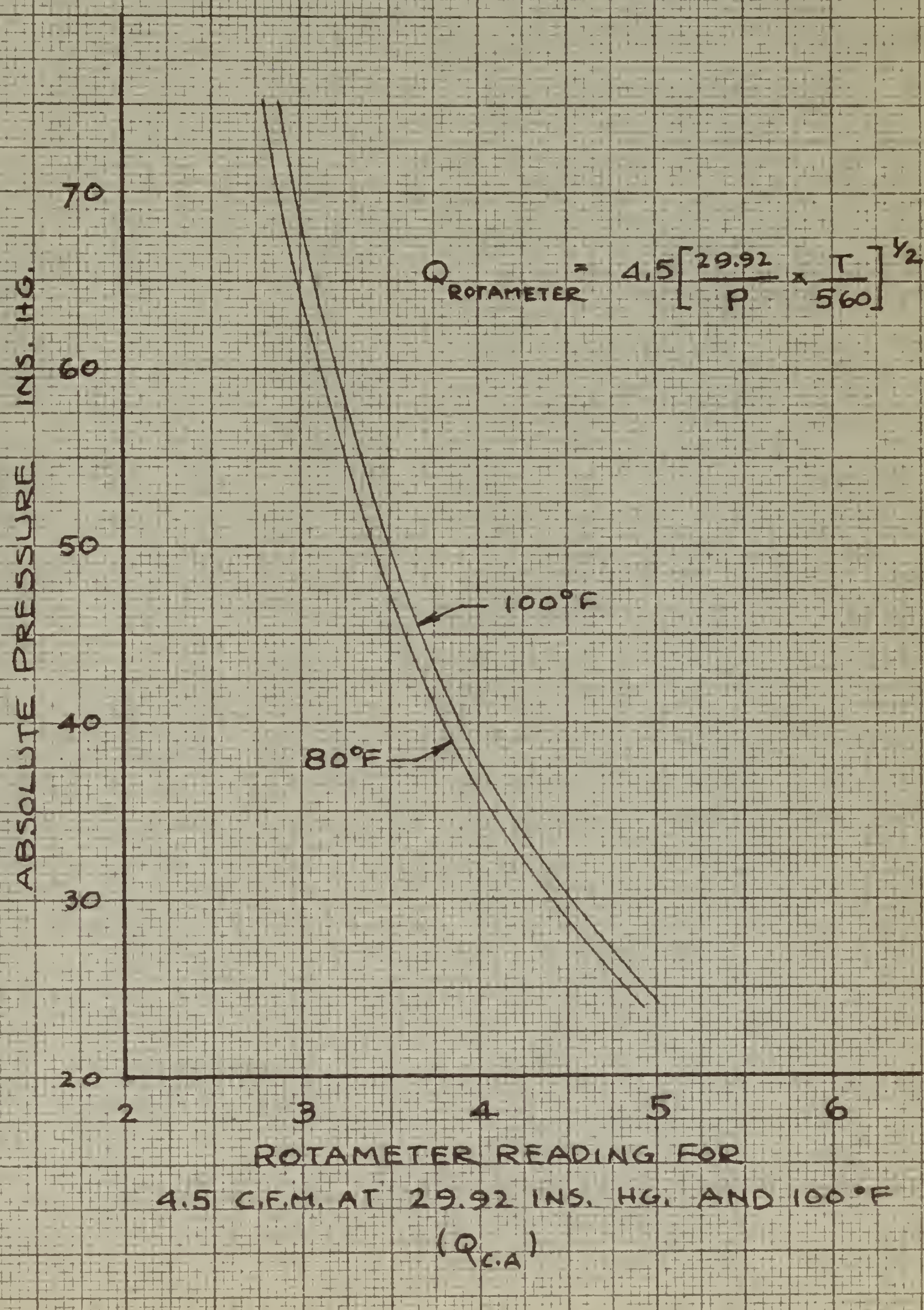


FIG. 3
SCHEMATIC ELEVATION OF TEST CELL AND CONTROL ROOM

FIG. 4

VARIATION OF ROTAMETER READING WITH TEMPERATURE AND PRESSURE FOR A CONSTANT COOLING AIR FLOW OF 4.5 C.F.M. AT 29.92 INS. HG. AND 100°F

(0.320 LB/MIN)



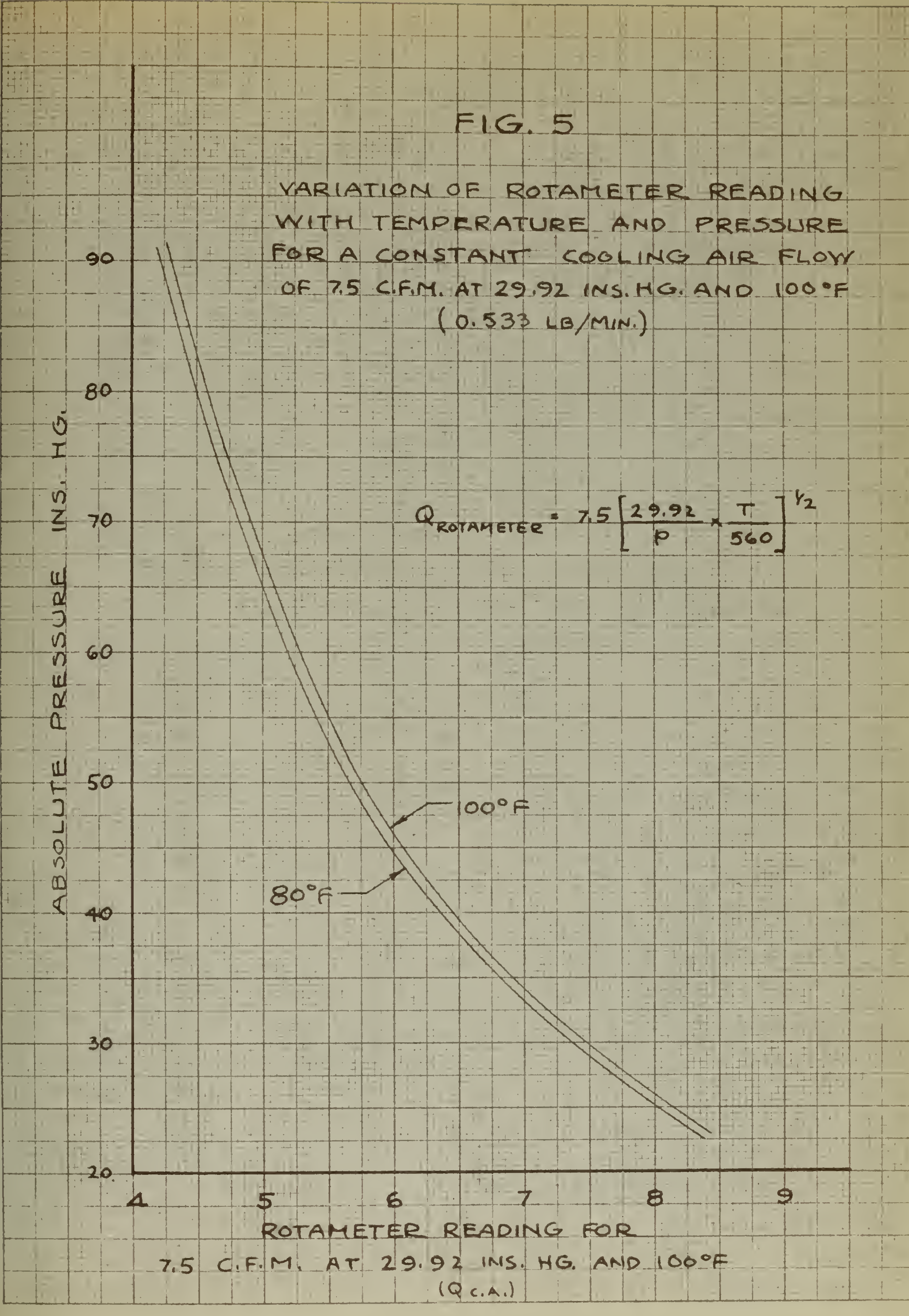


FIG. 5
VARIATION OF ROTAMETER READING
WITH TEMPERATURE AND PRESSURE
FOR A CONSTANT COOLING AIR FLOW
OF 7.5 C.F.M. AT 29.92 INS. HG. AND 100°F
(0.533 LB/MIN.)
$Q_{ROTAMETER} = 7.5\left[\frac{29.92}{P} \times \frac{T}{560}\right]^{1/2}$
100°F
80°F
ABSOLUTE PRESSURE INS. HG.
90
80
70
60
50
40
30
20
4
5
6
7
8
9
ROTAMETER READING FOR
7.5 C.F.M. AT 29.92 INS. HG. AND 100°F
(Q c.A.)

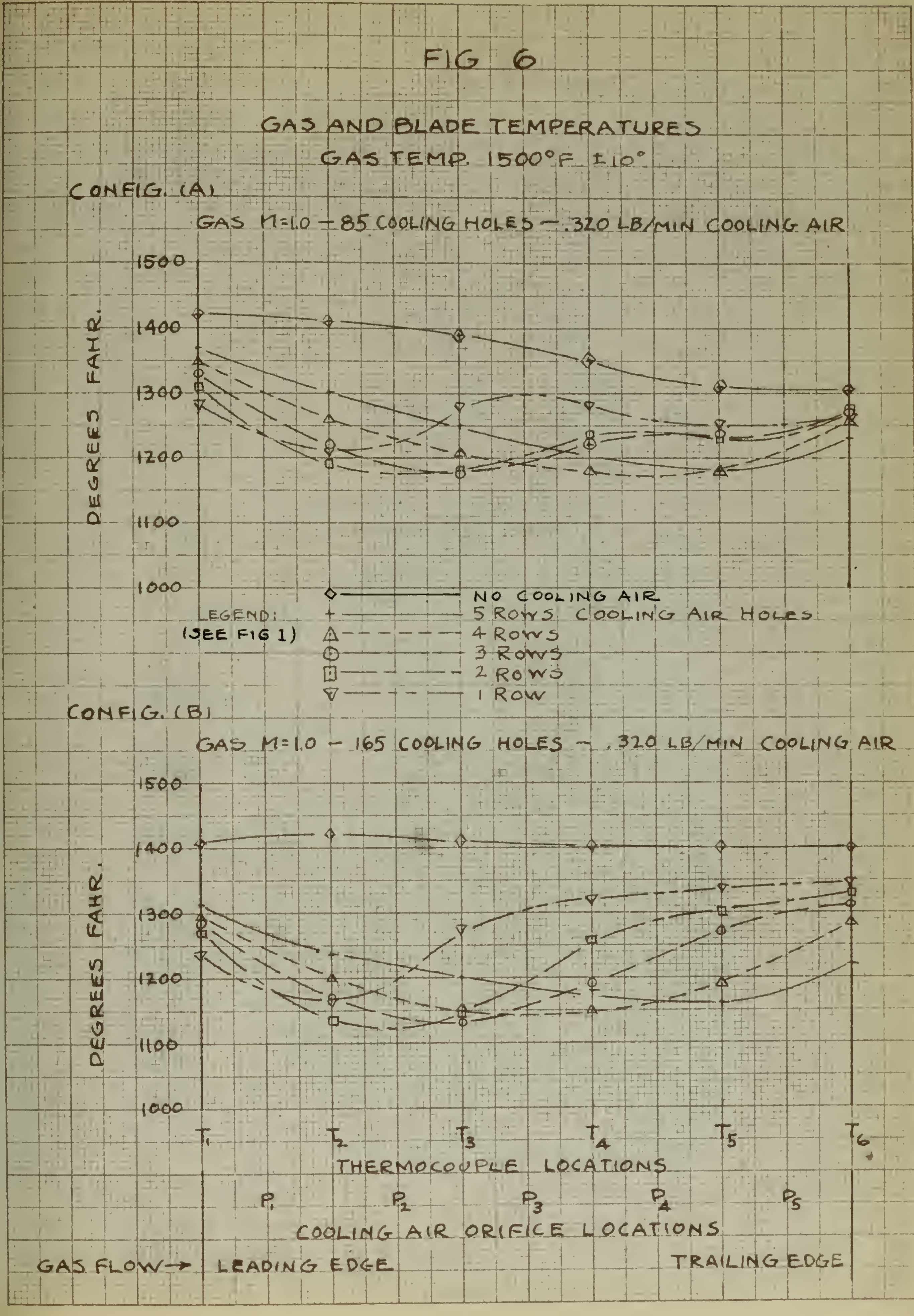


FIG 6
GAS AND BLADE TEMPERATURES
GAS TEMP. 1500°F ±10°
CONFIG. (A)
GAS M=1.0 - 85 COOLING HOLES - .320 LB/MIN COOLING AIR
1500
1400
1300
1200
1100
1000
DEGREES FAHR.
LEGEND:
(SEE FIG 1)
NO COOLING AIR
5 ROWS COOLING AIR HOLES
4 ROWS
3 ROWS
2 ROWS
1 ROW
CONFIG. (B)
GAS M=1.0 - 165 COOLING HOLES - .320 LB/MIN COOLING AIR
1500
1400
1300
1200
1100
1000
DEGREES FAHR.
T1 T2 T3 T4 T5 T6
THERMOCOUPLE LOCATIONS
P1 P2 P3 P4 P5
COOLING AIR ORIFICE LOCATIONS
GAS FLOW→ LEADING EDGE
TRAILING EDGE

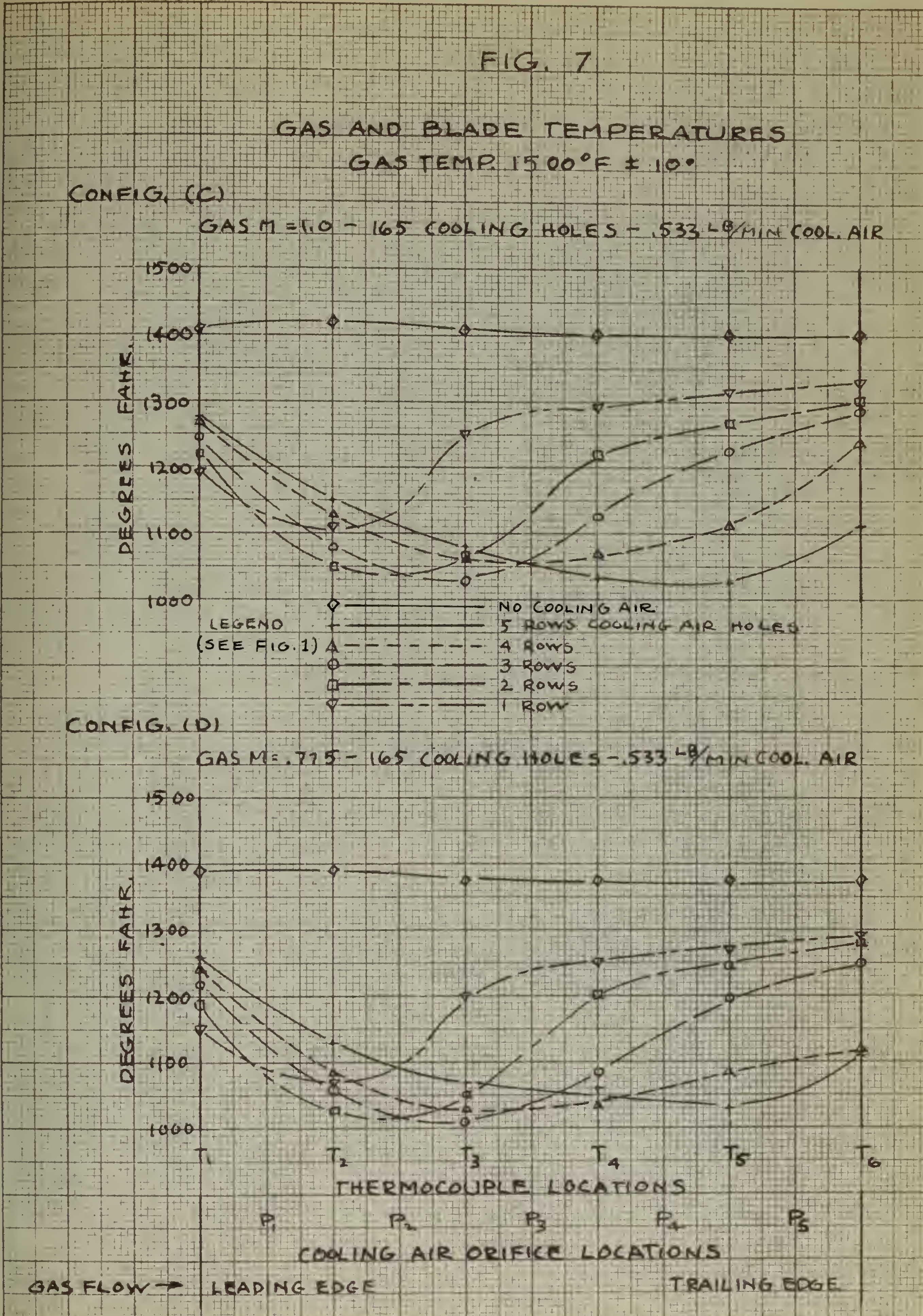


FIG. 7
GAS AND BLADE TEMPERATURES
GAS TEMP. 1500°F ± 10°
CONFIG. (C)
GAS M = 1.0 - 165 COOLING HOLES - .533 LB/MIN COOL. AIR
1500
1400
1300
1200
1100
1000
DEGREES FAHR.
NO COOLING AIR
LEGEND
(SEE FIG. 1)
5 ROWS COOLING AIR HOLES
4 ROWS
3 ROWS
2 ROWS
1 ROW
CONFIG. (D)
GAS M = .775 - 165 COOLING HOLES - .533 LB/MIN COOL. AIR
15.00
1400
13.00
1200
1100
1000
DEGREES FAHR.
T1
T2
T3
T4
T5
T6
THERMOCOUPLE LOCATIONS
P1
P2
P3
P4
P5
COOLING AIR ORIFICE LOCATIONS
GAS FLOW →
LEADING EDGE
TRAILING EDGE

FIG 8

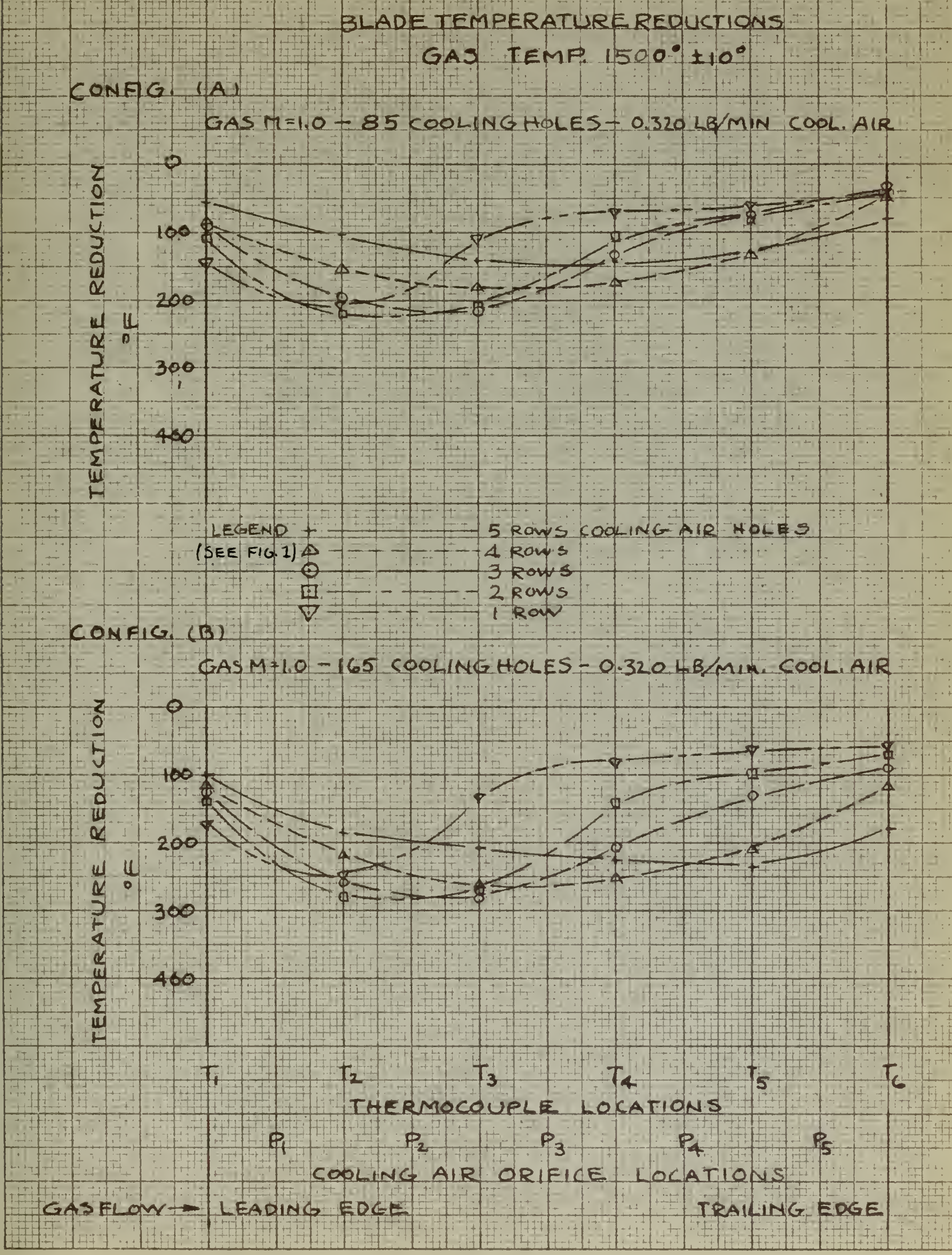

BLADE TEMPERATURE REDUCTIONS
GAS TEMP. 1500° ±10°
CONFIG. (A)
GAS M=1.0 - 85 COOLING HOLES - 0.320 LB/MIN COOL. AIR
TEMPERATURE REDUCTION °F
0
100
200
300
400
LEGEND + 5 ROWS COOLING AIR HOLES
(SEE FIG 1) △ 4 ROWS
○ 3 ROWS
□ 2 ROWS
▽ 1 ROW
CONFIG. (B)
GAS M=1.0 - 165 COOLING HOLES - 0.320 LB/MIN. COOL. AIR
TEMPERATURE REDUCTION °F
0
100
200
300
400
T_1 T_2 T_3 T_4 T_5 T_6
THERMOCOUPLE LOCATIONS
P_1 P_2 P_3 P_4 P_5
COOLING AIR ORIFICE LOCATIONS
GASFLOW → LEADING EDGE
TRAILING EDGE

FIG 9

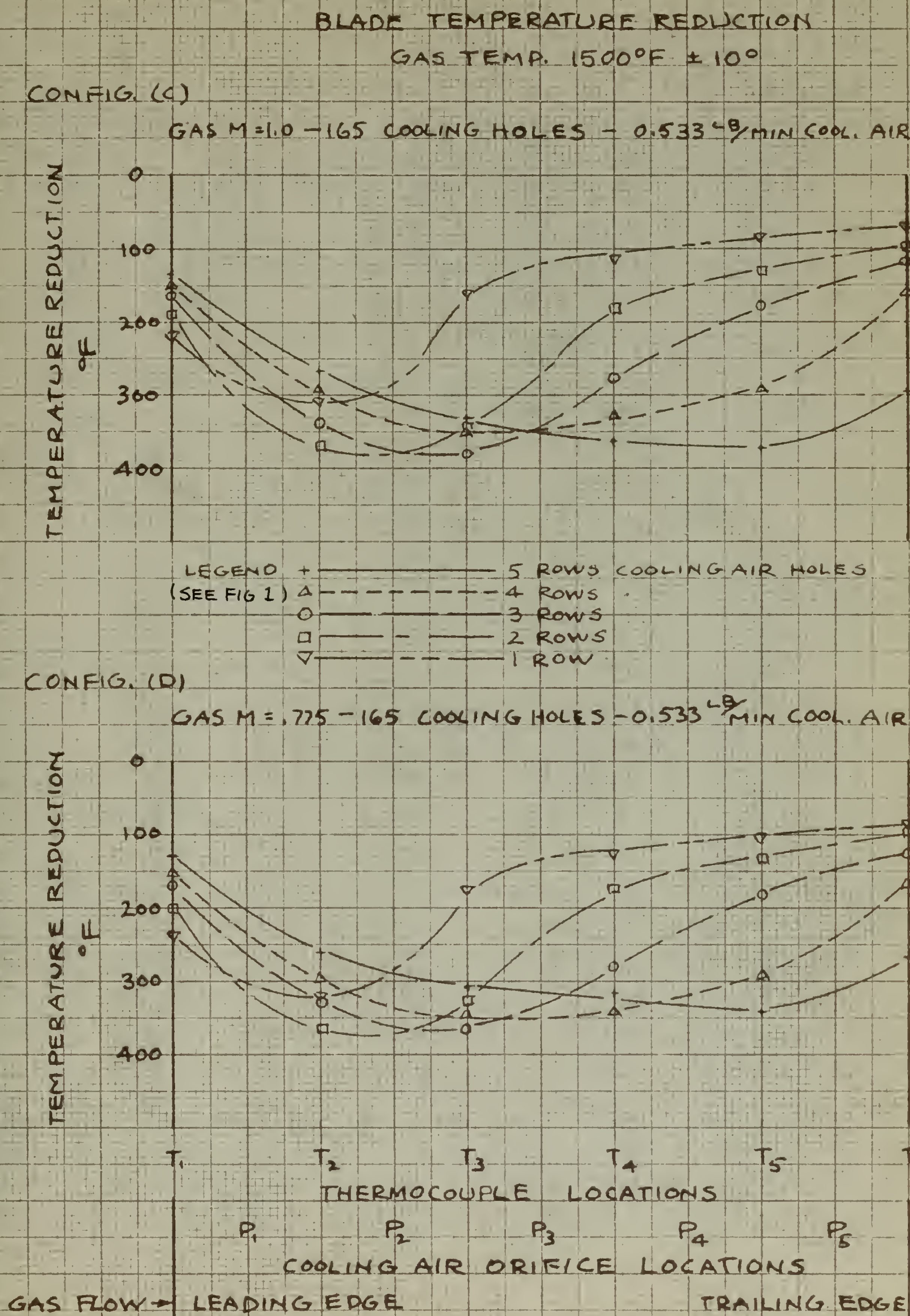

BLADE TEMPERATURE REDUCTION
GAS TEMP. 1500°F ± 10°
CONFIG. (C)
GAS M=1.0 – 165 COOLING HOLES – 0.533 LB/MIN COOL. AIR
TEMPERATURE REDUCTION °F
0
100
200
300
400
LEGEND + 5 ROWS COOLING AIR HOLES
(SEE FIG 1) △ 4 ROWS
○ 3 ROWS
□ 2 ROWS
▽ 1 ROW
CONFIG. (D)
GAS M=.775 – 165 COOLING HOLES – 0.533 LB/MIN COOL. AIR
TEMPERATURE REDUCTION °F
0
100
200
300
400
T1
T2
T3
T4
T5
T6
THERMOCOUPLE LOCATIONS
P1
P2
P3
P4
P5
COOLING AIR ORIFICE LOCATIONS
GAS FLOW → LEADING EDGE
TRAILING EDGE

FIG 10

BLADE TEMPERATURE REDUCTION

GAS TEMP. 1500°F ± 10°

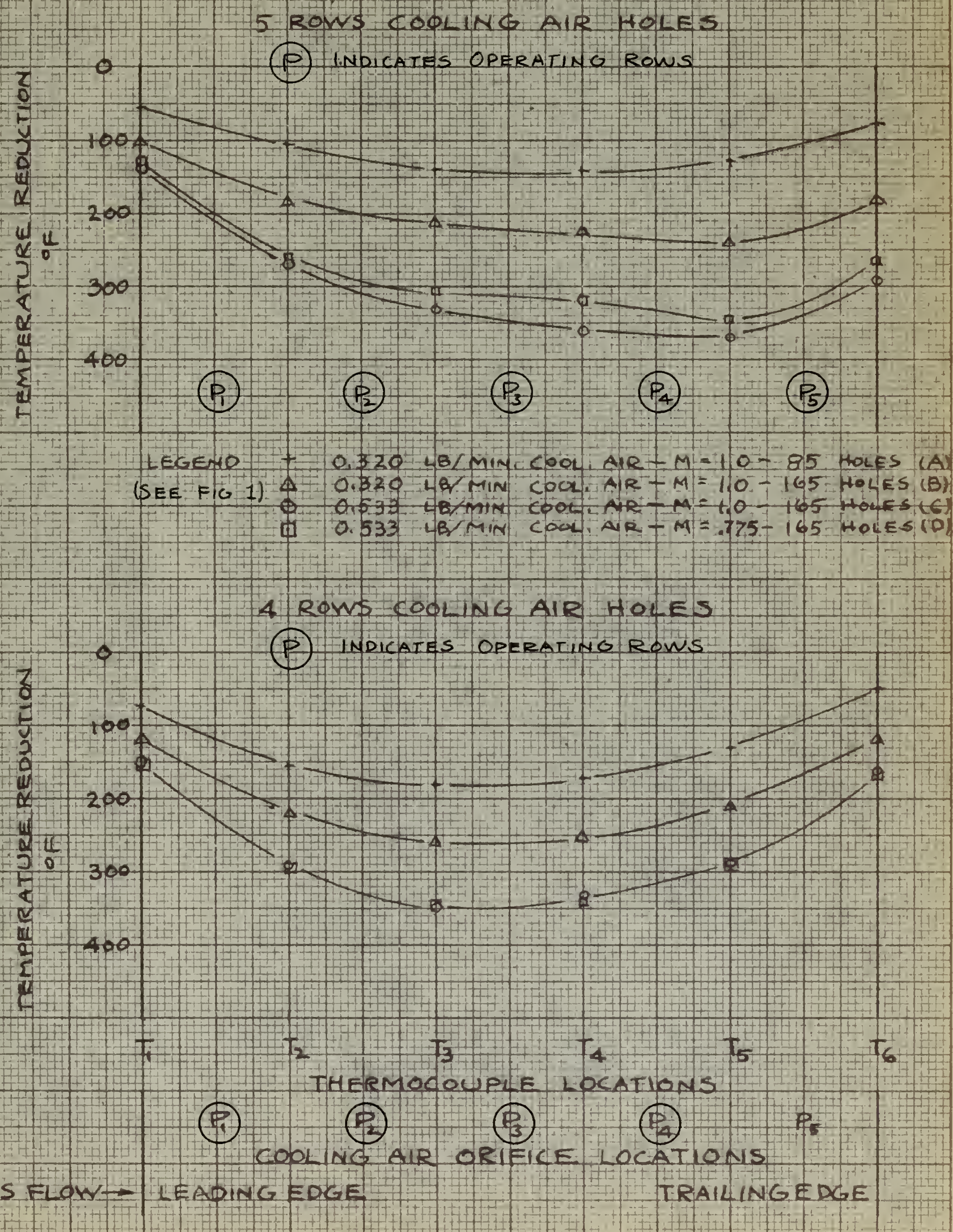


FIG. 11

BLADE TEMPERATURE REDUCTION
GAS TEMP. 1500°F ± 10°

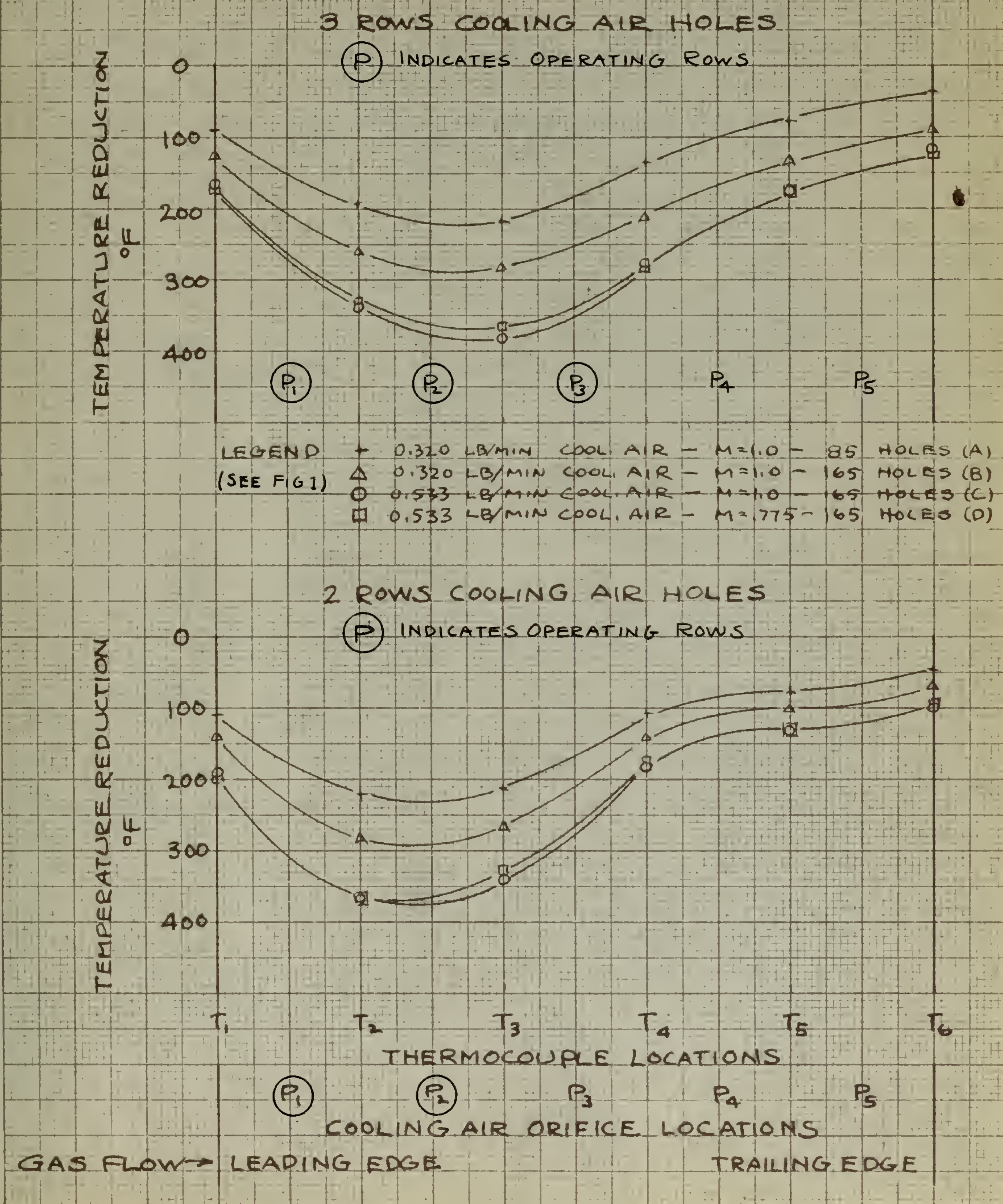
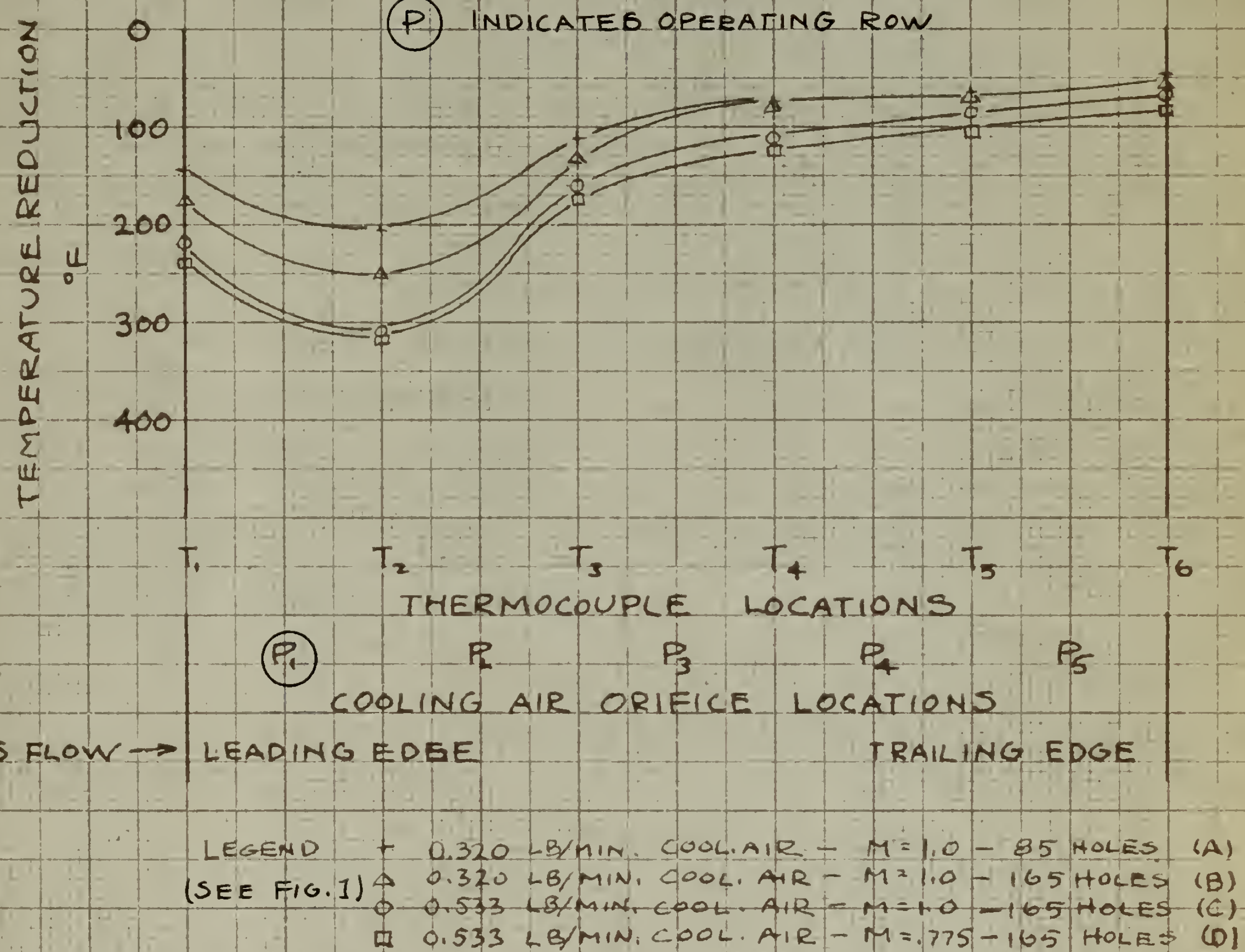


FIG. 12

BLADE TEMPERATURE REDUCTION
GAS TEMP. 1500 °F ± 10°

1 ROW COOLING AIR HOLES



LEGEND FOR EQUIPMENT LABELED IN PHOTOGRAPHS

A - Test Blade

B - Duct $3\frac{1}{2}$ ins. High and $4\frac{1}{2}$ ins. Wide at Test Section

C - Cooling Air Manifold

D - Test Blade Cooling Air Selector Switches

E - Tees for Measuring Blade Cooling Air Pressure

F - Total Pressure and Total Temperature Fittings

G - Static Pressure Fitting

H - J-33 Combustion Chamber

I - Cooling Air Rotameter and Control Needle Valve

J - Brown Temperature Recorder

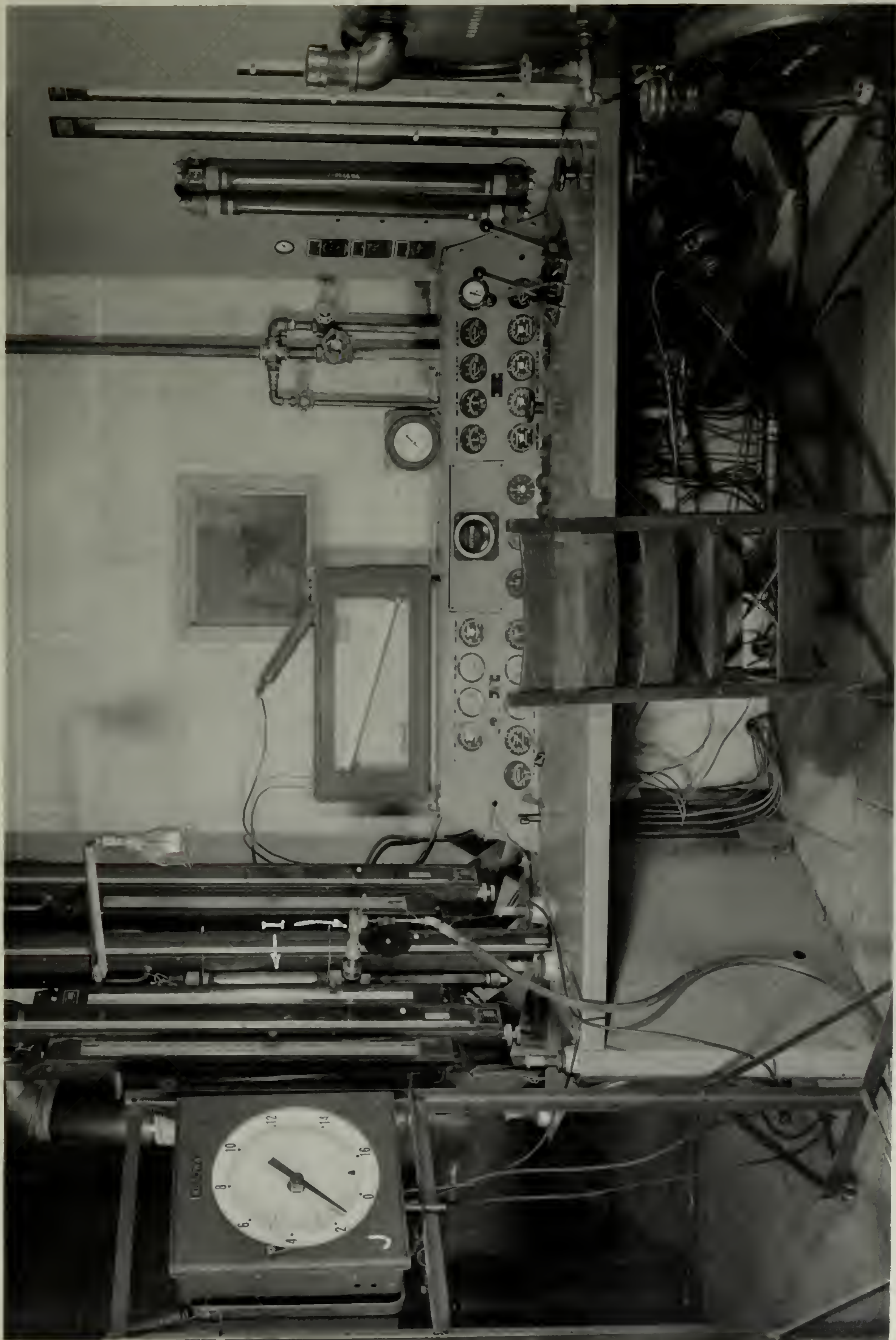


Figure 13 Control Room

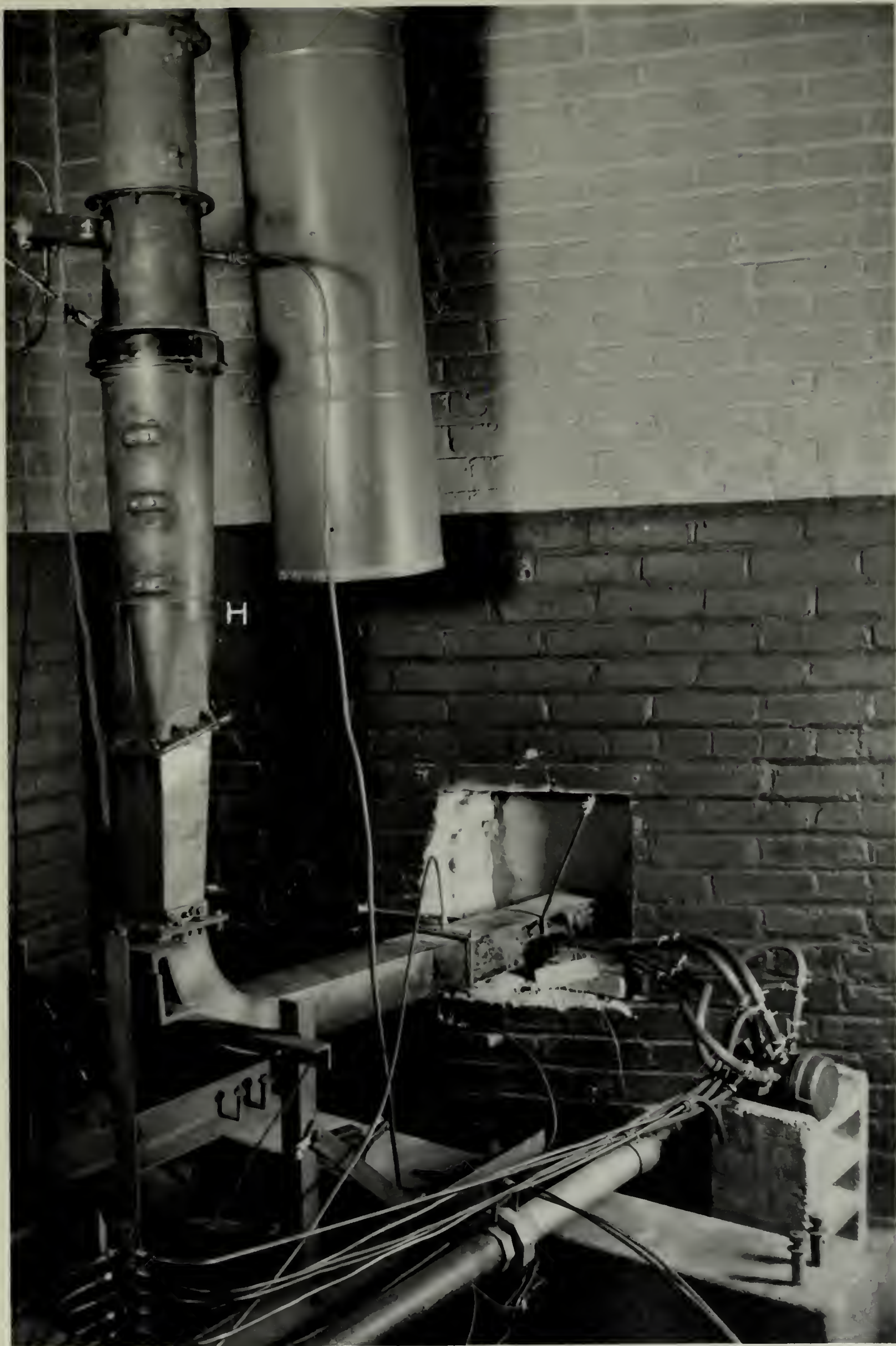


Figure 14 Burner and Test Section

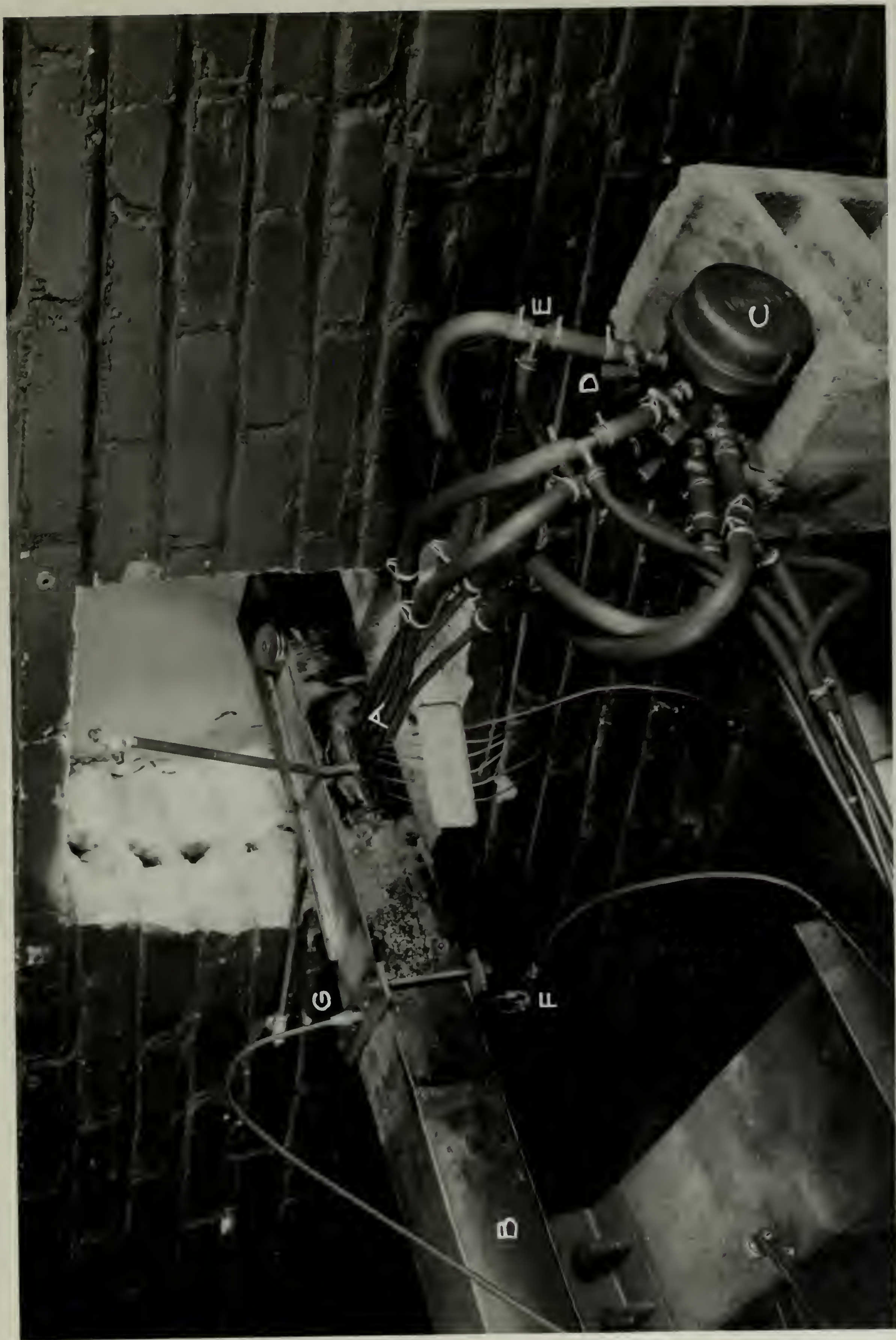


Figure 15 Test Section

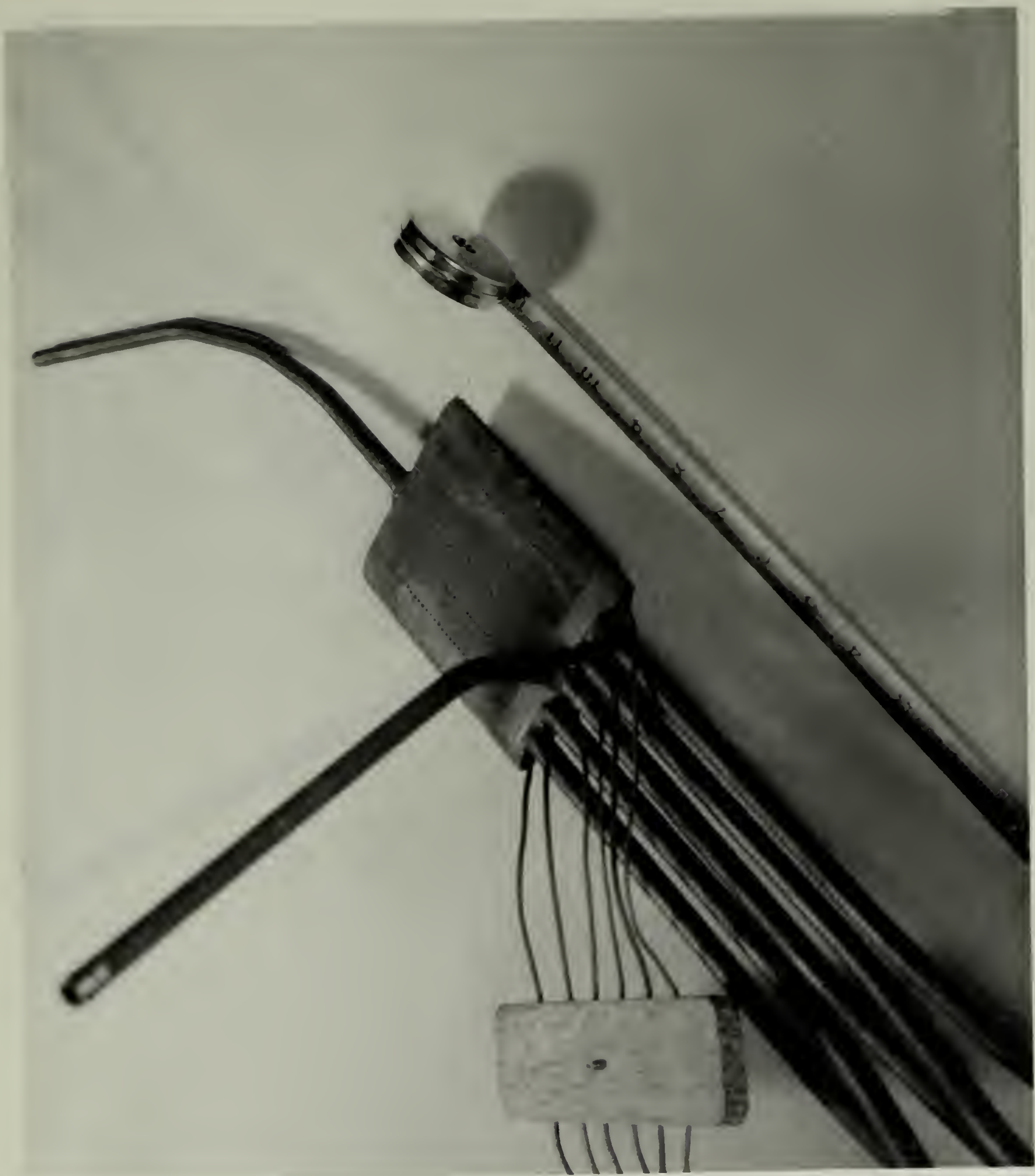


Figure 16 Top View of Test Blade

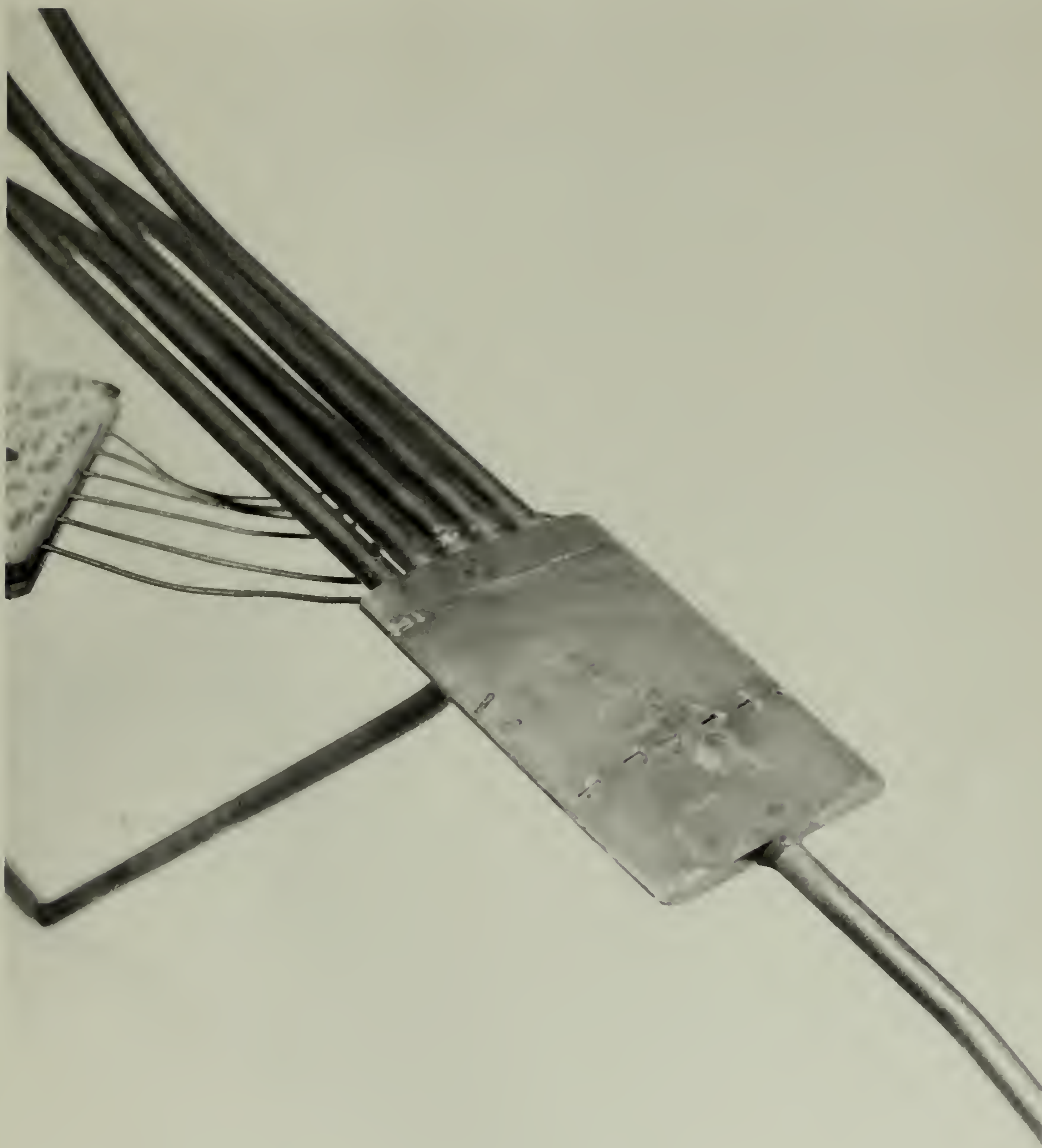


Figure 17   Bottom View of Test Blade

Thesis
M584 Mildahn 13190
Air film cooling of a metal surface exposed to high temperature and high velocity gases.