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A study of the heat transfer characteristics of a turbine blade having a ceramic sleeve with air cooling

Dressendorfer, David Eugene
University of Minnesota

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A STUDY OF THE HEAT TRANSFER CHARACTERISTICS
OF A TURBINE BLADE HAVING A CERAMIC SLEEVE
WITH AIR COOLING

A Thesis
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by

D. E. Fressendorfer
Commander, U. S. Navy

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A STUDY OF THE HEAT TRANSFER CHARACTERISTICS OF A TURBINE BLADE HAVING A CERAMIC SLEEVE WITH AIR COOLING

OBJECT AND SCOPE

The object of this investigation was to determine the high temperature protection afforded a turbine blade by using air cooling in conjunction with a blade shaped sleeve made of a sillimanite base ceramic.

The investigation was limited to a static test of an air cooled blade with a ceramic sleeve and an unprotected blade in a high velocity gas stream whose temperatures ranged from 600° F to 1600° F. Cooling air mass flow was controlled but varied. Recording of temperatures was limited to one temperature in each blade at a point approximately one fourth the blade length from the root. No attempt was made to investigate the thermal shock characteristics of the ceramic sleeve.

INTRODUCTION

In the present stages of gas turbine design, one of the factors limiting higher specific output and lower specific fuel consumption is the necessity of maintaining, because of metallurgical considerations, relatively low turbine inlet temperatures. Turbine theory and analysis of the thermodynamics of the cycles present in gas turbines reveal that the specific output of a gas turbine is proportional to the turbine inlet temperature.[1*] (See Fig. 1.) Although devices such as heat exchangers can be used to increase specific output and reduce specific fuel consumption, where weight saving is of importance such as in aircraft power plants, the employment of higher turbine inlet temperatures appears to be the most promising method of attaining the same objective.

High temperature alloys currently available for turbine blades rapidly lose strength and other desirable characteristics, such as low creep rate, at elevated temperatures, and current uncooled turbine units operate with a stress-limited temperature of approximately 1500° F. Fuels presently available will produce temperatures in the region of 3500° F. Fig. 2(a) shows

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* Superscripts refer to references contained in "References and Bibliography."

the measured temperature distribution in the blades of a test turbine, and Fig. 2(b) shows calculated centrifugal stresses along with stress-rupture curves for the blade material of the test turbine.[2] Fig. 3 shows a method of determining limiting speed and critical blade section from curves of computed temperature distribution and allowable stress limited blade temperature.[1] From these Figures it would appear that the maximum blade temperature occurs at a point near the middle of the blade, and that the critical section is at a point approximately one third the blade length from the root. Since present day alloys do not permit use of higher turbine inlet temperatures, possible solutions to the problem lie in blade cooling, the use of ceramics, and the useof metallo-ceramics or ceramals.

In the category of blade cooling, the cooling mediums most widely employed have been air and water. The Germans prior to the end of World War II were, in some of their latest aircraft turbo-jet engines using hollow blades cooled by air. With blade materials, whose high temperature qualities were inferior to American types, temperatures as high as 1653° F could be reached.[1] A water cooled stationery gas turbine was also operated in Germany, in which blade temperatures of 850° to 930° F were maintained with gas a temperature of 2200° F. However, water level control, condenser auxiliaries, and the possibility of mineral deposits in cooling

passages are among the disadvantages applicable to such a system.[3]

The National Advisory Committee for Aeronautics made an extensive study,[1] part of which was experimental, of turbine blade cooling methods. Among the methods investigated were: rim cooling by air, film cooling by air in which the cooling air flowed out of slits in the blade and along the blade surface, cooling hollow blades by air with and without an insert (Fig. 4), and cooling by circulating liquid through passages in the blades. From the above investigations it was determined that improved performance resulted with increased inlet temperatures permitted by cooling, even though cooling losses were considered. (See Fig. 1 which shows performance when losses due to liquid cooling are considered). Rim cooling was considered to be of little value other than to increase blade life. Film cooling was found to be effective, and an analytical investigation of hollow blade air cooling revealed that a gas temperature increase of 500° F could be obtained. With an insert, the permissible temperature increase was found to be 650° F. Liquid cooling indicated that temperatures up to the limit of current fuels could be obtained.

In order to show some of the desirable features of ceramics in their application to the gas turbine field, a review of some of their properties and a comparison

of a few of these properties with present metallic alloys can be presented.[4] Using one of the many formulas, for a metallic blade having a specific gravity of 8.3 (average for blade alloys), a maximum root stress of approximately 40,000 psi can be obtained. At 1200° F, a number of alloys have 100 hour stress rupture strengths in excess of that figure. However, above 1200° F the strength of metallic alloys decreases rapidly, whereas some porcelains retain most of their original, though lower, strength at elevated temperatures, and have sustained tensile stresses of 17,000 psi at 1800° F for over 100 hours, with an average elongation or creep of less than .0004 per cent per hour. At 1800° F the best alloy has a 100 hour stress-rupture strength of approximately 12,000 psi. Since the stress set up in a turbine blade is directly proportional to the density, a ceramic blade with a density of 3.0 and a tensile strength of 10,000 psi would be equivalent to an average alloy blade having a tensile strength of 47,000 psi. Ryschkewitsch, in Germany, investigated the tensile strengths of several ceramics at 1800° F, and found that their strengths ranged from 6,000 to 18,000 psi. At similar temperatures, the National Bureau of Standards conducted tests of a number of ceramic bodies and found that their bending strengths ranged from 10,000 to 33,000 psi. Although ceramics are generally weak in tension, they are surprisingly good in compression. The National Bureau of

Standards tested 330 specimens of which 55 per cent had compressive strengths in excess of 200,000 psi, and one ceramic exhibited a compressive strength of 332,000. Ceramicshave excellent resistance to oxidation at elevated temperatures, whereas only a few of the very scarce metals such as platinum and iridium have good resistance to oxidation at high temperatures.

Porcelains have been found to have excellent resistance to creep below 1900° F, and in contrast to metals do not have the characteristic of having their creep rate increase with time. Under prolonged loading they appear to become more resistant to creep. The most important aspects of ceramics favorable to their use in gas turbines are their heat resistant qualities, high melting points, low densities, moldability, resistance to oxidation, and possible low cost.

In comparison to metals ceramics have several unattractive characteristics which must be overcome to successfully make use of their more desirable qualities in high temperature work. Brittleness, thermal shock sensitivity, and manufacturing difficulties such as shrinkage, warping and fissuring during forming and sintering operations, appear to be the most important. Of these, brittleness can possibly be avoided by using compressive rather than tensile stresses, and increased knowledge and techniques in manufacturing processes may eliminate forming difficulties. Thermal shock sensitivity

is apparently a property of the particular ceramic as well as its shape. Ceramics exhibit varying degrees of thermal shock resistance, and in a listing of ten ceramic refractories, the thermal shock resistance is classified as fair to very good.[6] They are, however, generally inferior to metals in this respect.

The use of ceramics, both in the form of a coating for metal and as solid ceramic bodies in high temperature applications has shown considerable promise. The use of ceramic coatings on molybdenum pitot tubes, located in the nozzle of a ram jet, was, in simulated service tests, instrumental in prolonging the life of the instruments from 5 minutes to 45 minutes.[5] Although much of the increased life was due to the prevention of oxidation of the molybdenum, in specimen tests on a ceramic coated molybdenum strip, subjected to a gas stream at 3500° F, the temperature at the surface of the strip was effectively reduced to 2600° F.[5] In an analytical study,[1] the National Advisory Committee for Aeronautics determined that a liquid cooled turbine blade, with a low conductivity ceramic coating .01" thick permitted a 450° F increase in gas temperature over liquid cooling alone. The use of ceramic coatings on metals, however, presents additional problems of thermal shock because of the difference in rates of thermal expansion of the ceramic and the metal.

Research effort into the use of ceramics as turbine blade material had during and subsequent to World War II reached considerable proportions. In Germany various designs were considered and tested, involving the employment of the ceramic blades in tension as well as compression. Blades with a conventional dovetail securing arrangement were tested in a rotor assembly. It was found that the blades had low load bearing capacity, and according to W. Kalisch, of the Air Transport Research Institute, there seemed to be little hope of reaching high peripheral velocities with ceramic blades.[3] Another design utilized the principle of placing the ceramic blades in compression by mounting the blades radially inward on the inner surface of a rotating drum. This design was also abandoned due to increased weight of, and increased stress in, the drum because of its high peripheral speed, and because of the relatively complicated assembly.[3] The Germans finally resorted to a water cooled turbine. In an effort to evaluate the actual use of ceramic turbine blades, N.A.C.A. constructed and operated a gas turbine with rotor blades made of sillimanite ceramic.[7] This unit was operated at speeds up to 10,000 R.P.M., and at temperatures up to 1725° F. The operation of the turbine did not meet with the service requirements because of lower than normal R.P.M., little power extraction, and low pressure ratios across the turbine. Preliminary to the actual running of this

turbine, thermal shock tests were conducted on a number of blades. The blades withstood heating cycles with an 1100° F temperature change at the rate of 24,000° F per minute, but on cooling at the same rate, two out of three blades broke. However, when the cooling rate was reduced to 20,000° F per minute, the blades withstood repeated cooling cycles. Despite the fact that all of these blades failed at relatively low centrifugal stresses, it was concluded that the results warranted further research into the use of ceramics as turbine blades.

In order to improve the thermal shock characteristics and at the same time retain the temperature resistance of ceramics, some research in recent years has been directed to the use of metallo-ceramics or ceramals. Ceramals are compounded by use of powdered metals with the addition oxides, carbides, nitrides, or borides, etc.[6,8] Considerable work in this field was done in Germany,[3] and the National Advisory Committee for Aeronautics investigated a carbide type ceramal as a gas turbine blade material.[9] Although results of ceramal research has showed considerable promise, it appears that considerable more research must be done before definite conclusions as to feasibility of ceramal for rotor blades can be made.

Since ceramics are as a whole weak in tension and comparatively strong in compression, it would appear

that research into the use of a completely ceramic turbine blade has mainly benn directed to the exploitation of the more inferior strength characteristic of ceramics. Professor [illegible] Murphy of the University of Minnesota Faculty was of the opinion that, subject to considerations of thermal shock sensitivity, ceramics would be feasible in protecting turbine blades from high temperatures, providing the design was such that the ceramic could, in actual turbine operation, be placed in compression instead of tension. It was his idea that a blade shaped sleeve, free to move radially along an inner blade or core, and retained by means of a cap device or shroud ring, might satisfactorily utilize the high compressive strength of ceramics, obviate considerable of the objectionable brittleness characteristics of ceramics, and, since the sleeve walls could be relatively thin, considerable resistance to thermal shock could be maintained. Further, the clearance between the walls of the sleeve and the inner blade provided an ideal passage for the introduction of cooling air. Fig. 4 shows that air cooling of a hollow turbine blade with an insert was much more effective and was accomplished with a minimum amount of air in comparison with a blade without an insert.

SLEEVE FORMING PROCESS

The forming die shown in Figs. 5 and 6 was the final design with which sleeves were finally successfully formed. Several modifications to the original design were made to improve the quality of the sleeves. Countersinking of the sliding piston, and bevelling of the blade tip were resorted to to improve the flow characteristics in pressing. It was known prior to manufacturing the die that the design selected was basically not the best, but since simplicity, ease of manufacture, and cost considerations were paramount the design was carried out.

A sillimanite base cement was selected as sleeve material because of availability and the fact that the thermal shock characteristics have been proven to be sufficiently good to warrant the consideration of sillimanite for turbine blading investigations.[7]

The dry press process was used and after numerous trials it was found that 12 per cent fluid addition was necessary to provide sufficient plasticity to make a sleeve of rather uniform density. In addition it was found necessary to preload the die with sillimanite during assembly of the die. A pressure of 16,500 psi was used in pressing, and it was found that increased pressure did not, from all appearances, improve the

sleeve qualities, but merely resulted in bending the die blade. The greatest difficulty encountered in forming the sleeves was lubrication of the mold to effect withdrawal of the sleeve from the die blade. Various lubricants and combinations were tried, among them were: talc, lard oil, graphite grease, vaseline, silicone grease, and mica lubricant. None of the above lubricants were successful when water was used as the additive. The possibility of using a lubricant as the fluid additive was next resorted to. A solution composed of the following was made up: 50% by volume of a solution containing 50% by weight of "Carbo-wax" (a water-soluble wax) in water; 50% by volume of a solution containing 4% by weight of "Meth-o-cel" in water. Twelve per cent by weight of this solution was added to the sillimanite powder, and the die surfaces were lubricated with lard oil. This effort was again unsuccessful as the "Meth-o-cel", which was to give dry strength to the sleeve, apparently increased its adhering quality. Sleeves were successfully withdrawn from the mold using the following procedure: die surfaces rubbed with colloidal graphite and given a light coating of lard oil; 12% by weight of a solution, composed of 50% by weight of Carbo-wax in water, used as the fluid additive. Fig. 6 shows a sleeve as withdrawn from the mold and prior to shaping and sintering.

SINTERING PROCESS

It was decided, from a study of the equilibrium diagram[11] for Al_2O_3 - SiO_2, to sinter the sleeves at 3000° F. The degree of deformation of the sleeves to be expected in sintering was not definitely known. To minimize deformation, a carbon coffin (Fig. 6) with a 2" dia bore, having the same inner surface curvature as the sleeve walls, was made to hold the sleeve while sintering. Two carbon rods, (.163" dia) were inserted in the sleeve, to prevent sagging. Temperature measurements during sintering were made using a Leeds-Northrup Optical Pyrometer of the disappearing filament type. The sintering was done in a carbon-resistance type furnace, following as closely as possible a schedule shown in the Appendix.

Fig. 6 shows a sleeve shaped and ready for sintering, and also a sleeve after sintering. Little curvature deformation was observed, although some shrinkage was apparent. Withdrawal of the carbon rods subsequent to sintering was made without difficulty. Wall thickness of the sintered sleeve, designed to be .1" thick varied from ".087 to ".118. However, it is known that the wall thickness during forming was not uniform, possibly due to springing of the die blade during pressing. The weight of the sintered sleeve was .085 lbs. It is

believed that the discoloration of the sintered sleeve was due to the fact that the sleeve was sintered under reducing conditions.

TEST EQUIPMENT

Molding Die

A detailed drawing of the molding die used in forming the ceramic sleeves is shown in Fig. 5. Fig. 6 is a photograph showing the component parts of the die.

Test Blades

Details of the test blade used as the insert for the ceramic sleeve are shown in Fig. 7. The unprotected test blade was made by cutting down the central part of the stationary piston of the molding die, and is shown in Fig. 8 along with the insert test blade. Three .136" diameter holes are shown in the base flange of each blade for securing the test blades to the test block. It should be noted from the detailed drawings and Figs. 8 and 9 that there is considerable difference in thickness of the two test blades.

For reasons of simplicity the blades were designed along lines of a pure impulse type having no twist. Further, the angle through which the gas stream was to be turned was limited by design to approximately 75 degrees.

In addition to the actual test blades, two Jumo 004 buckets were cut down and used as guide blades to direct the gas flow along the test blades. Fig. 9 shows these blades as installed in the test block.

<u>Ceramic Sleeve</u>

The ceramic sleeve, shown in Figs. 6 and 9 was designed to have, prior to sintering, a wall thickness of 0"1. A securing arm, shown in Fig. 9 was used to prevent the sleeve from lifting off the blade base from the action of the cooling air.

<u>Test Block</u>

Figs. 8 and 9 show details of the test block. The test block was made from a steel block 4"5 x 4"5 x 1"5. A slot 1"7 wide and ".16 deep was milled in the block to receive the blades. Holes for cooling air, thermocouples, and hold down screws were drilled and are shown in Fig. 8. Also shown in Fig. 8 is the cooling air chamber which was chiseled out to form a distribution channel for passage of cooling air to the test blade air holes. The block and insert test blade mating surfaces were ground to provide an air tight joint and thereby prevent cooling air leakage from around the base of the blade. The raised edge around the thermocouple hole in the center of the air reservoir was also ground to form an airtight seal and prevent leakage of cooling air into the thermocouple installation.

<u>Test Section</u>

The test section was designed to give an entrance and exit area of 15.75 in^2 (3"5 x 4"5), and an approximate Mach Number of .6 for the range of operation

encountered in the investigation. The test block with blades formed one wall of the test section. Fig. 10, which shows schematically the complete test layout, illustrates the location of the test section as well as other main components of test equipment.

Allison Engine

The Allison V-1710-81 engine, naturally aspirated and shown in Fig. 11, furnished compressed air to the J-33 burner. Air flow to the Allison supercharger was measured by the pressure drop across a 5.76 dia. sharp edge orifice located in the 8" dia. intake pipe.

J-33 Burner

Hot gas for delivery to the test section was generated in a J-33 burner using diesel fuel. Temperature of the gas stream was controlled, for any set air supply, by changing fuel pressure to the burner. The J-33 burner and installation is shown in Fig. 12.

Temperature Measurements

Temperature measurements were made by use of ceramic insulated, chromel-alumel thermocouples, led through compensated lead wires to a Brown Recorder. The Recorder, reading in degrees Fahrenheit, is shown in Figs. 13 and 14.

Cooling Air Meter

Compressed air at 60 to 80 psig for blade cooling was furnished to the cooling air meter from the

laboratory main air supply. The cooling air meter shown in Fig. 13 was constructed insofar as practicable in accordance with reference (10). Cooling air pressure was controlled by a globe stop value and read on the gauge at the outlet end of the meter. Air quantity was measured by the pressure drop across a ".75 dia. sharp edge orifice in the 2" pipe of the cooling meter. The pressure drop was indicated on the manometer recording in inches of water.

Control Panel

Control of the test equipment was exercised through a standard control panel shown in Fig. 14.

TEST PROCEDURE

The blades, with thermocouples installed and mounted in the test block (Fig. 9), were inserted in the test section, and tests were conducted from the control panel. Two complete runs were made, the first at a compressor RPM of 18,250, and the second at a compressor RPM of 24,000. Temperature of the gas stream was varied, by control of burner fuel pressure, between 600° F and 1600° F in 200° F increments at a rate of approximately 1000° F per minute. At each gas temperature, cooling air flow was varied by adjusting the cooling air meter orifice pressure drop of 0 to .7 inches of water, using increments of .10 inches of water. Temperature recordings were made of blade thermocouple readings for each gas temperature and range of cooling air flow rates. Necessary pressure and temperature readings were recorded. Observed data for all tests are listed in Tables I and II.

TEST RESULTS

Figs. 15 and 16 show graphically the results of this investigation. In Fig. 15 the amount of temperature reduction obtained with the ceramic sleeve is shown plotted against weight of cooling air for burner air flows varying from 108 to 130 lbs/min. The gas stream Mach No. varied from .494 to .611. The maximum temperature reduction of 374° F occurred at a gas temperature of 1600° F and a cooling air flow of .925 lbs/min. However, the greatest rate of increase in temperature reduction occurred at lower cooling air flows. This can be seen from the decreasing slope of the curves as the cooling air weight flow increased. The temperature of the protected blade with no cooling air flow was considered to be essentially the same as that of the unprotected blade (Table I). This can possibly be attributed to the fact that there was apparently little heat transfer from the base of the test block to the atmosphere, and hence the temperature necessarily would equal that of the unprotected blade.

Fig. 16 shows temperature reduction versus cooling air weight flow for burner air flows ranging from 155 to 170 lbs/min, and for gas stream Mach Nos. from .731 to .823. It was not possible in this run to maintain combustion in the burner at a gas temperature of 600° F.

TEST RESULTS

Figs. 15 and 16 show graphically the results of these investigations. In Fig. 15 the amount of temperature reduction obtained with the porous plates is shown plotted against weight of coolant air for various gas flows varying from 100 to 150 lbs/min. The gas stream Mach No. varied from .495 to .511. The maximum temperature reduction of 370° F occurred at a gas temperature of 1600° F and a coolant air flow of .255 lbs/min. However, the greatest ratio of increase in temperature reduction occurred at lower cooling air flows. This can be seen from the increasing slope of the curves as the cooling air flow was decreased. The temperature of the porous plate with no cooling air flow was considered to be essentially the same as that of the unperforated plate (Table I). This can possibly be attributed to the fact that there was essentially little heat transfer from the face of the test plate to the atmosphere, and hence the temperature necessarily would equal that of the unperforated plate.

Fig. 16 shows temperature reduction versus cooling air weight flow for various gas flows ranging from 100 to 130 lbs/min, and for gas stream Mach Nos. from .480 to .513. It was not possible in this run to maintain conditions in the system at a gas temperature of 1600 F. [illegible]

Also, the maximum cooling air rate available from the laboratory supply line was .81 lbs/min. A maximum temperature reduction of 840° F for this run was obtained at a gas temperature of 1600° F and a cooling air flow of .81 lbs/min.

As in the first run, the slope of the curves reveals that the greatest increase in temperature reduction occurred at the lower cooling air flows.

A comparison of the curves for the two test conditions shows remarkable similarity. At the higher Mach Nos. the reduction in temperature, for a given gas temperature, was slightly but consistently greater than that for the lower gas stream Mach Nos. As could be expected, for a given cooling air flow, the greatest temperature reduction occurred at the higher gas temperatures.

Since there was apparently little heat transfer from the test block to the atmosphere, the protection afforded the blade by the ceramic sleeve, without cooling, and solely as a result of its low thermal conductivity, could not be evaluated. It is felt that this situation, with respect to heat transfer from the blade to the mounting base, is more adverse than that existing in an actual turbine. Excluding other factors and based on the temperature protection afforded by a low conductivity ceramic it is probable that a similar installation in an actual turbine would either result

in a greater temperature reduction, or similar temperature reduction with less cooling air flow.

The weight of cooling air used in providing the temperature reductions encountered in these tests was relatively small. Assuming that the maximum amount of cooling air used, (.925 lbs/min in Run 1 affecting a temperature reduction of 874° F) could effectively cool one turbine blade in an I-40 turbojet engine (54 buckets, compressor air capacity approximately 4260 lbs/min at rated sea level r.p.m.) extrapolation of data would indicate that cooling of the turbine blades in the I-40 engine could be accomplished by bleeding approximately 1.2% of the compressor air capacity.

CONCLUSIONS

Since this investigation was conducted in a static rig and under conditions unrelated to actual gas turbine operation, no definite conclusions can be drawn as to the applicability of ceramic sleeves in reducing blade temperatures in an actual turbine. However, the following conclusions were reached:

1. The configuration employing a ceramic sleeve with air cooling is, excluding service operating and other factors, an effective and economical means of effecting temperature reduction of turbine blades.

2. The amount of temperature reduction obtained is proportional to the cooling air flow, for the range of cooling air flows investigated.

3. The amount of temperature reduction per unit weight flow of cooling air is greatest at the lower cooling air flow rates.

TABLE I

RUN 1 - OBSERVED TEST DATA, COMPRESSOR RPM=18,250

GAS TEMPERATURE - °F

ΔP_{ca} "H_2O	P_{ca} psi (gage)	W_{ca} lbs/min	600 T_M	600 T_C	600 T_R	800° T_M	800° T_C	800° T_R	1000° T_M	1000° T_C	1000° T_R	1200° T_M	1200° T_C	1200° T_R	1400° T_M	1400° T_C	1400° T_R	1600° T_M	1600° T_C	1600° T_R
0	0	0	475	487	-12	670	670	0	875	870	5	1030	1030	0	1208	1230	-22	1332	1325	7
.1	12	.236	475	326	149	675	462	213	882	600	282	1043	777	266	1195	913	282	1332	1095	237
.2	18	.363	475	287	188	675	380	295	882	527	355	1043	580	463	1195	656	539	1332	760	572
.3	22	.476	475	255	220	675	340	335	882	462	420	1043	495	548	1195	620	575	1337	703	634
.4	26	.582	475	248	227	683	320	363	887	412	475	1043	475	568	1195	547	648	1337	625	712
.5	32	.688	475	228	247	690	300	390	887	373	514	1043	456	587	1195	490	705	1337	559	783
.6	38	.810	475	222	253	690	287	403	887	346	541	1043	437	606	1195	450	745	1337	507	830
.7	44	.925	215	690	260	690	279	411	887	333	554	1043	412	631	1195	-	-	1337	463	874

	600°	800°	1000°	1200°	1400°	1600°
P_f, fuel pressure, psig	49	61	67	73	79	85
W_f, fuel flow, lbs/hr	47	65	80	90	103	113
ΔP_{BA}, " Hg.	1.05	.95	.88	.83	.80	.74
W_{BA} lbs/sec	2.16	2.06	1.98	1.92	1.89	1.815
P_3, " Hg. gage	7.9	7.8	8.4	8.6	9.1	9.2
P_{T_3} " Hg. gage	8.3	8.0	8.6	8.9	9.3	9.5
P_4 " Hg gage	.8	.55	.4	.32	.30	.28
P_{T_4} " Hg. gage	5.8	5.8	6.55	6.95	7.35	7.6
q " Hg.	5.0	5.25	6.15	6.63	7.05	7.32
M_4	.494	.506	.559	.595	.60	.611

TEMP: COOLING AIR = 80°F.   PRESSURE: (ATMOS) = 29.82" Hg

TEST CELL = 120°F   TEST CELL = 29.5" Hg

TABLE II

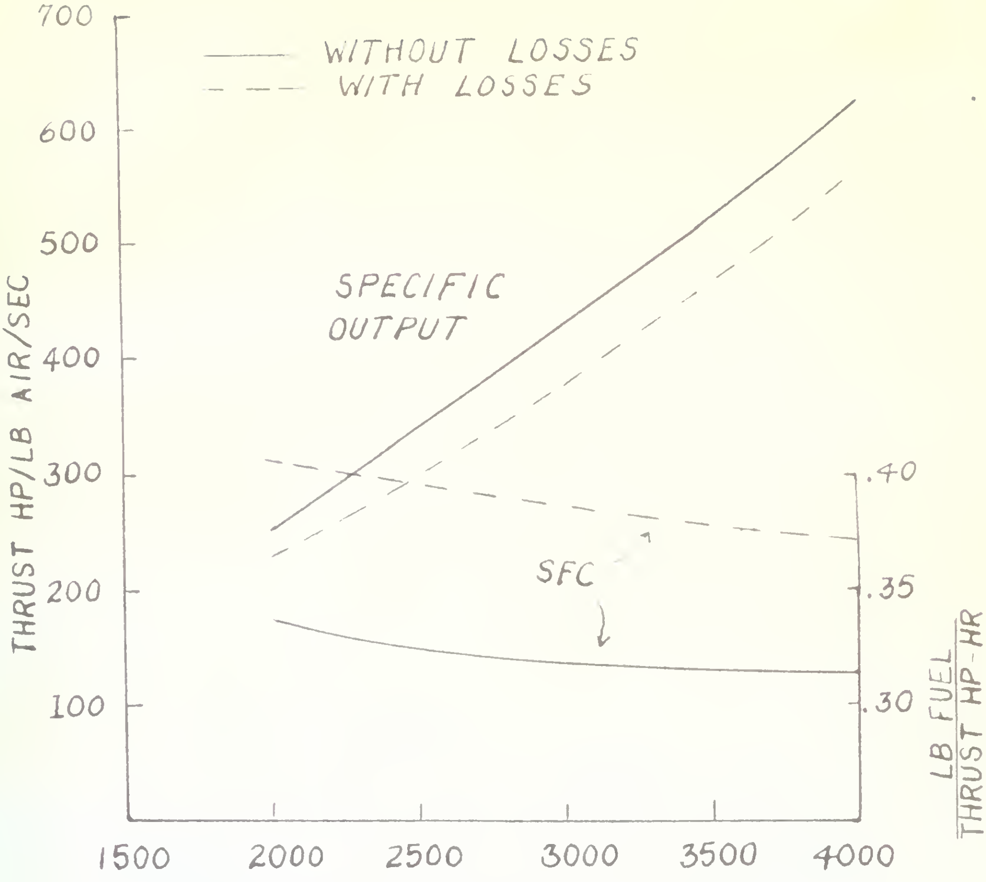
RUN 2 - OBSERVED TEST DATA, COMPRESSOR RPM= 24,000

ΔP_{ca} "H_2O	P_{ca} psi (gage)	W_{ca} lbs/min	GAS TEMPERATURE - °F: 600° T_M	600° T_C	600° T_R	800° T_M	800° T_C	800° T_R	1000° T_M	1000° T_C	1000° T_R	1200° T_M	1200° T_C	1200° T_R	1400° T_M	1400° T_C	1400° T_R	1600° T_M	1600° T_C	1600° T_R
.0	0	0	-	-	-	696	715	-19	900	925	-25	1047	1075	-28	1160	1215	-55	1270	1350	-80
.1	12	.236	-	-	-	645	738	207	844	578	266	985	670	315	1120	760	360	1270	862	408
.2	18	.363	-	-	-	650	365	285	837	439	398	990	535	455	1120	620	500	1270	735	535
.3	22	.476	-	-	-	650	327	323	837	380	457	997	475	522	1120	502	618	1270	592	678
.4	26	.582	-	-	-	657	295	362	837	347	490	997	425	572	1120	445	675	1270	522	748
.5	32	.688	-	-	-	657	275	382	837	307	530	997	380	617	1120	405	715	1270	463	807
.6	38	.810	-	-	-	657	255	402	837	280	557	997	352	645	1120	380	740	1265	425	840

	600°	800°	1000°	1200°	1400°	1600°
P_f, fuel pressure, psig	-	67	76	84	96	107
W_f, fuel flow, lbs/hr.	-	74	93	107	129	146
ΔP_{BA}, "Hg	-	1.83	1.82	1.80	1.65	1.50
W_{BA} lbs/sec	-	2.85	2.845	2.83	2.71	2.59
P_3 "Hg gage	-	14.8	15.0	16.1	17.1	16.8
P_{T_3} "Hg gage	-	15.4	15.4	16.5	17.5	17.2
P_4 "Hg gage	-	.72	.64	.60	.52	.50
P_{T_4} "Hg gage	-	11.45	11.8	13.1	13.75	13.8
δ "Hg	-	10.63	11.16	12.5	13.23	13.3
M_4	-	.731	.75	.795	.82	.823

TEMP: COOLING AIR = 80°F
TEST CELL = 120°F

PRESSURE: (ATMOS) = 29.82" Hg.
TEST CELL = 29.5" Hg



TURBOPROP ENGINE PERFORMANCE WITH + WITHOUT COOLING LOSSES. AIRPLANE SPEED, 500 MPH: MACH NO., 0.69; ALTITUDE 30,000 FT.

Figure 1

(Reproduced from Ref. 1 by Permission IAeS)

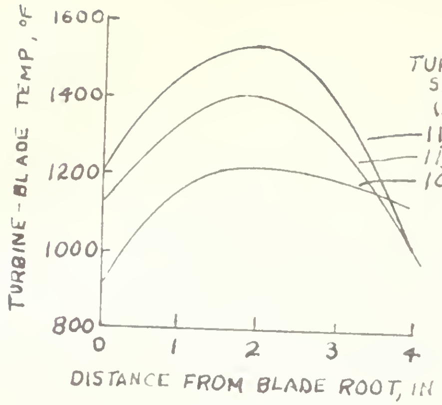


Figure 2(a)

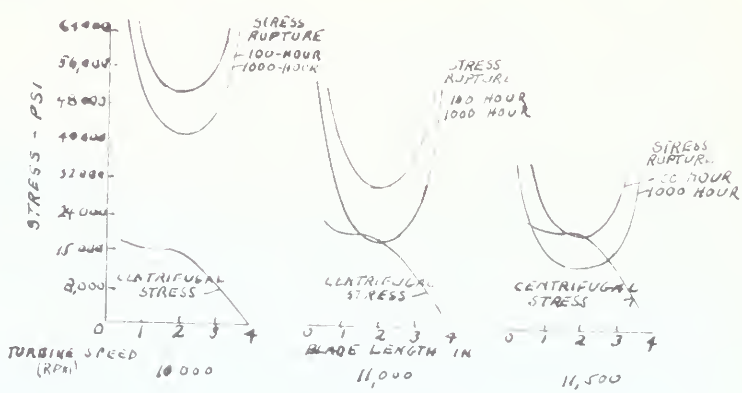
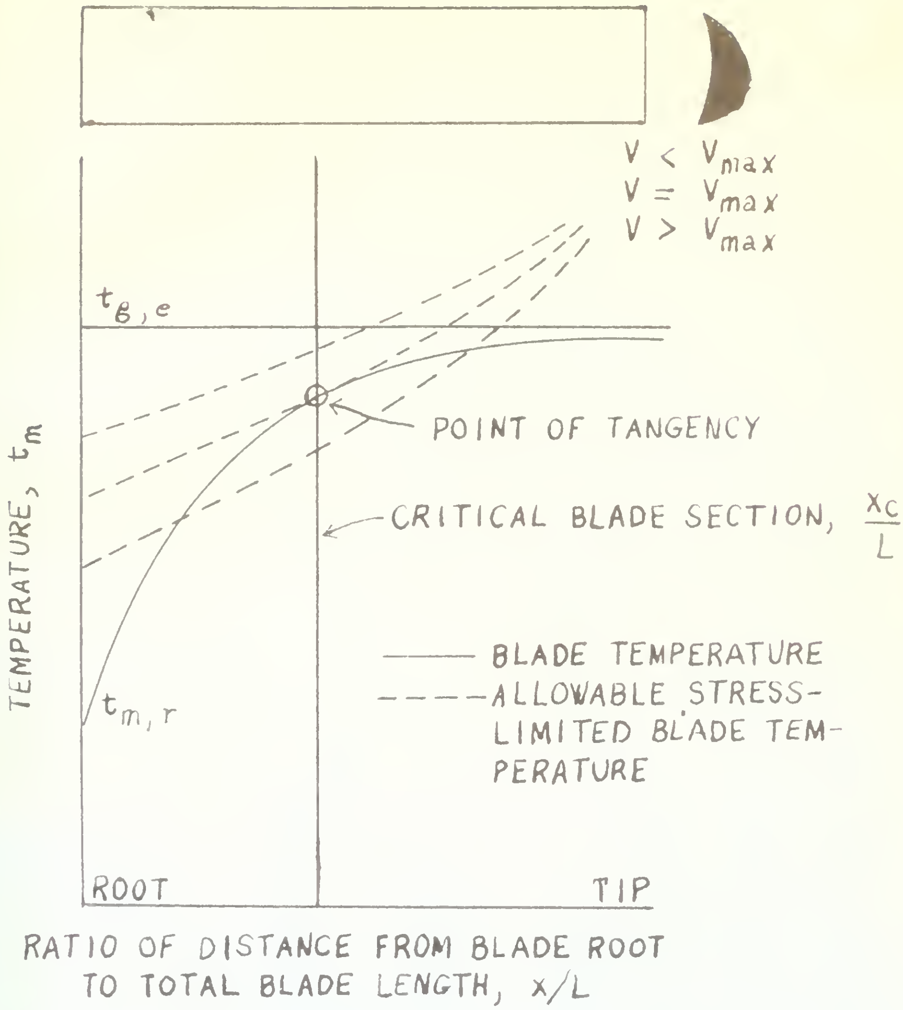


Figure 2(b)

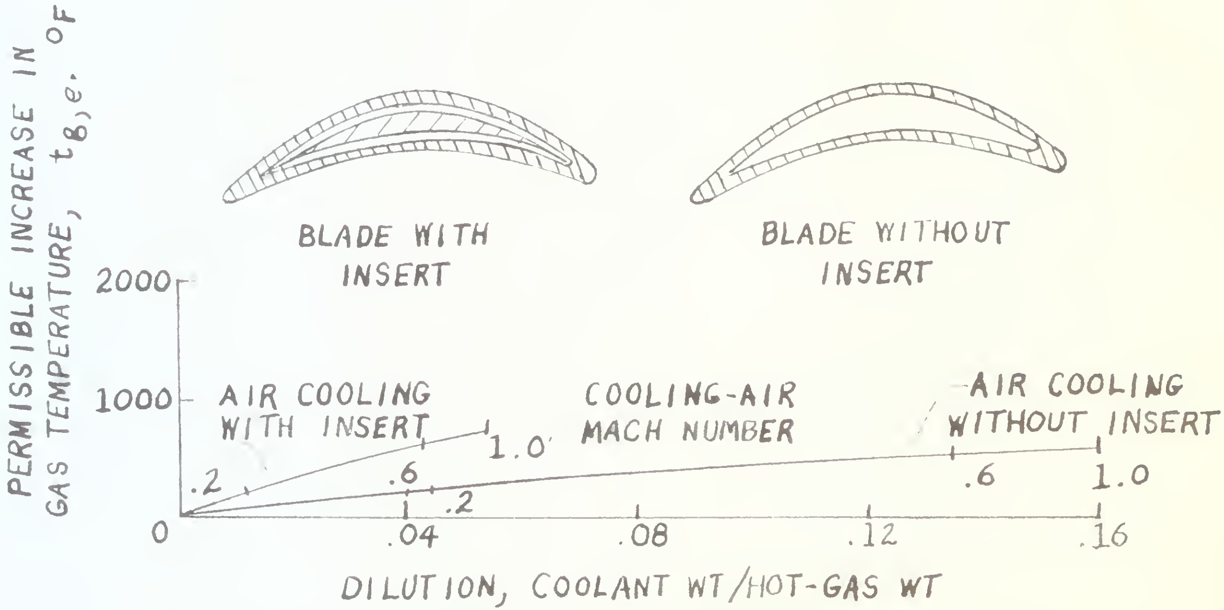
(Reproduced from Ref. 2 by Permission SAE)



METHOD OF DETERMINING LIMITING SPEED AND CRITICAL BLADE SECTION FROM CURVES OF TEMPERATURE DISTRIBUTION AND ALLOWABLE STRESS-LIMITED BLADE TEMPERATURE

Figure 3

(Reproduced from Ref. 1 by permission IAeS)



VARIATION OF PERMISSIBLE GAS TEMPERATURE INCREASE WITH DILUTION. LIFE, 1000 HRS.; BLADE MACH NO., 0.5

Figure 4

(Reproduced from Ref. 1 by permission IAeS)

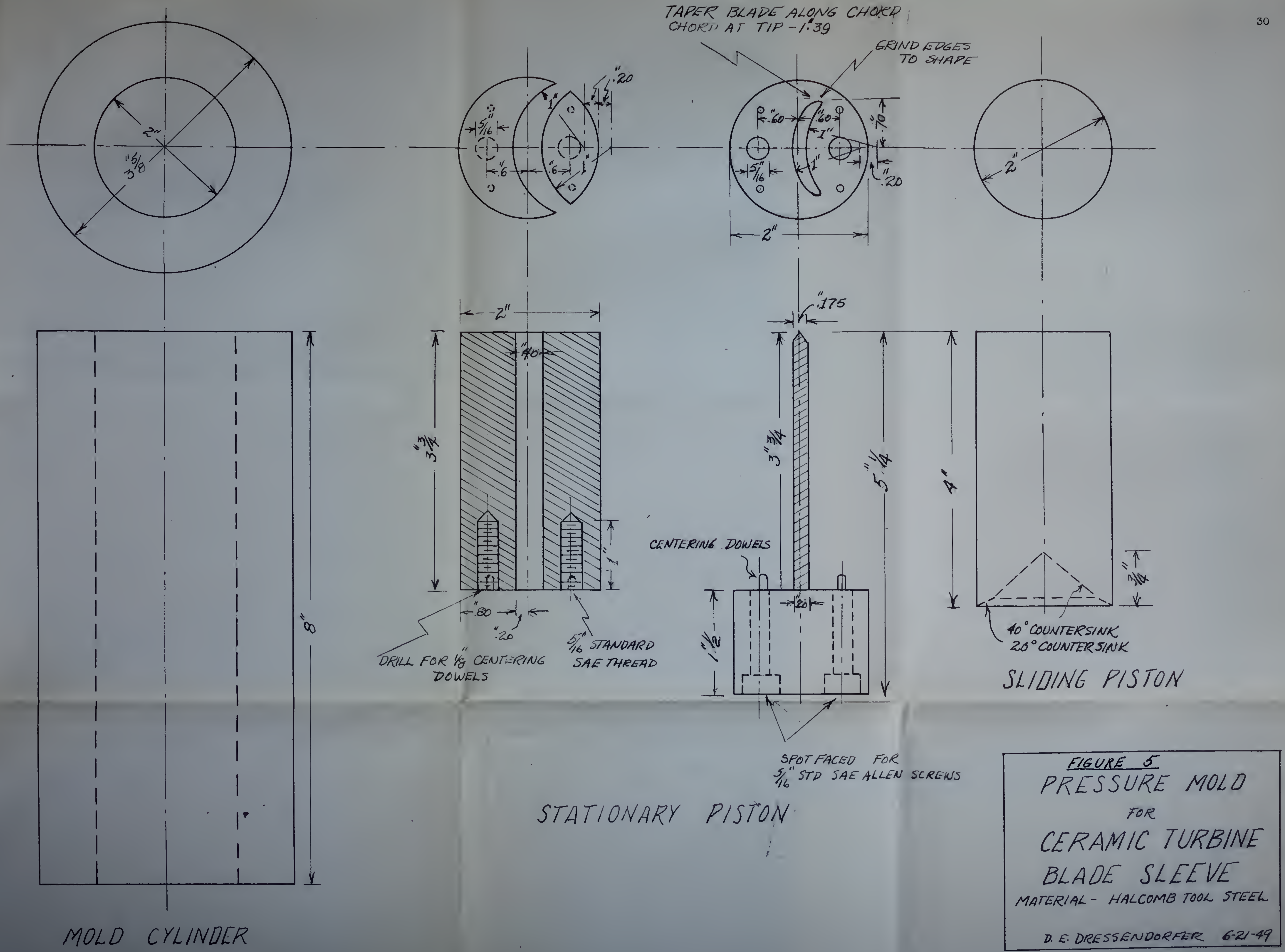
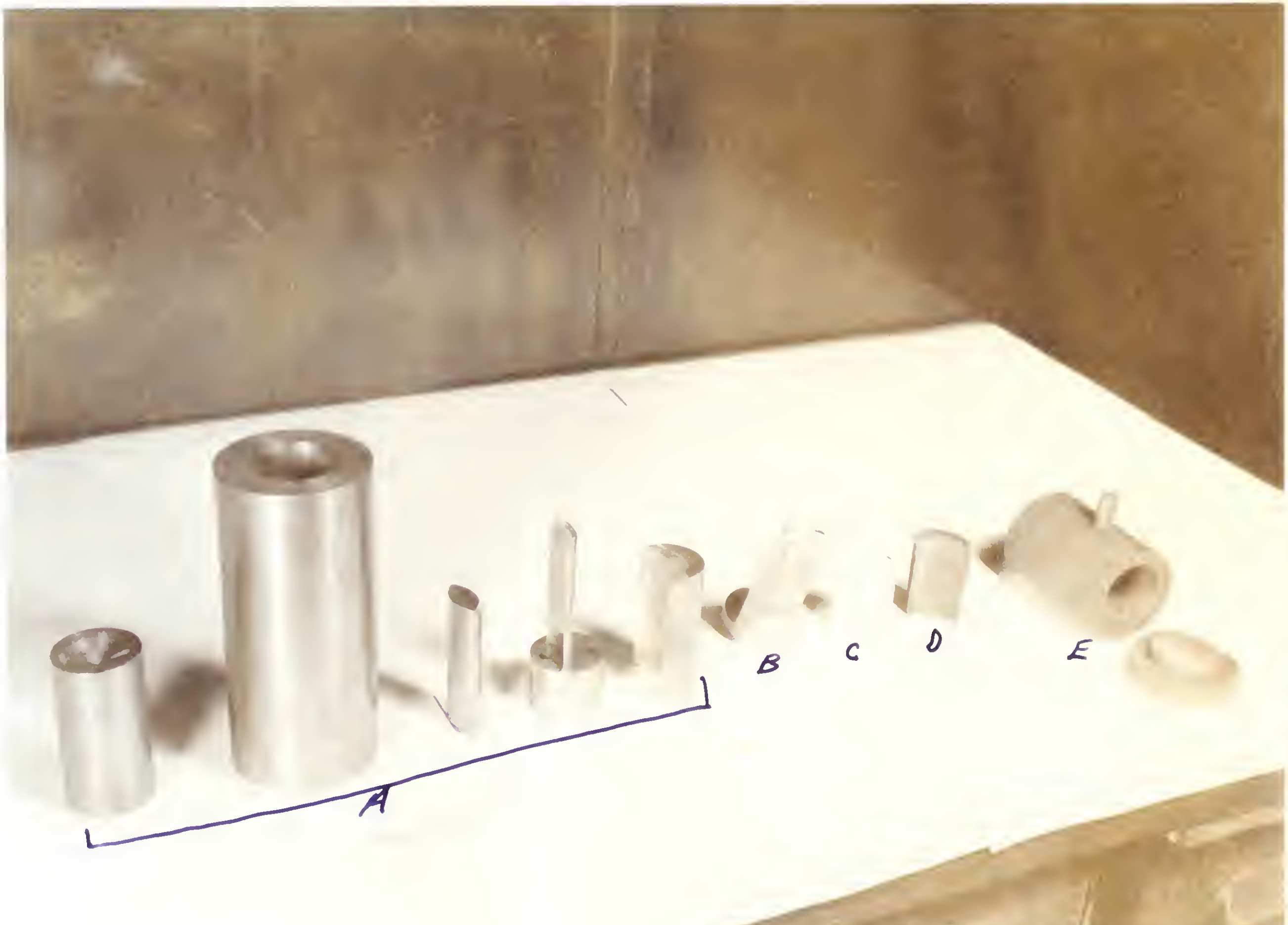


FIGURE 5
PRESSURE MOLD FOR CERAMIC TURBINE BLADE SLEEVE
MATERIAL - HALCOMB TOOL STEEL
D. E. DRESSENDORFER 6-21-49



A-Molding Die

B-Sleeve From Mold

C-Sleeve Shaped Before Sintering

D-Sintered Sleeve

E-Sintering Holder

Figure 6

Sleeve Forming and Sintering Equipment

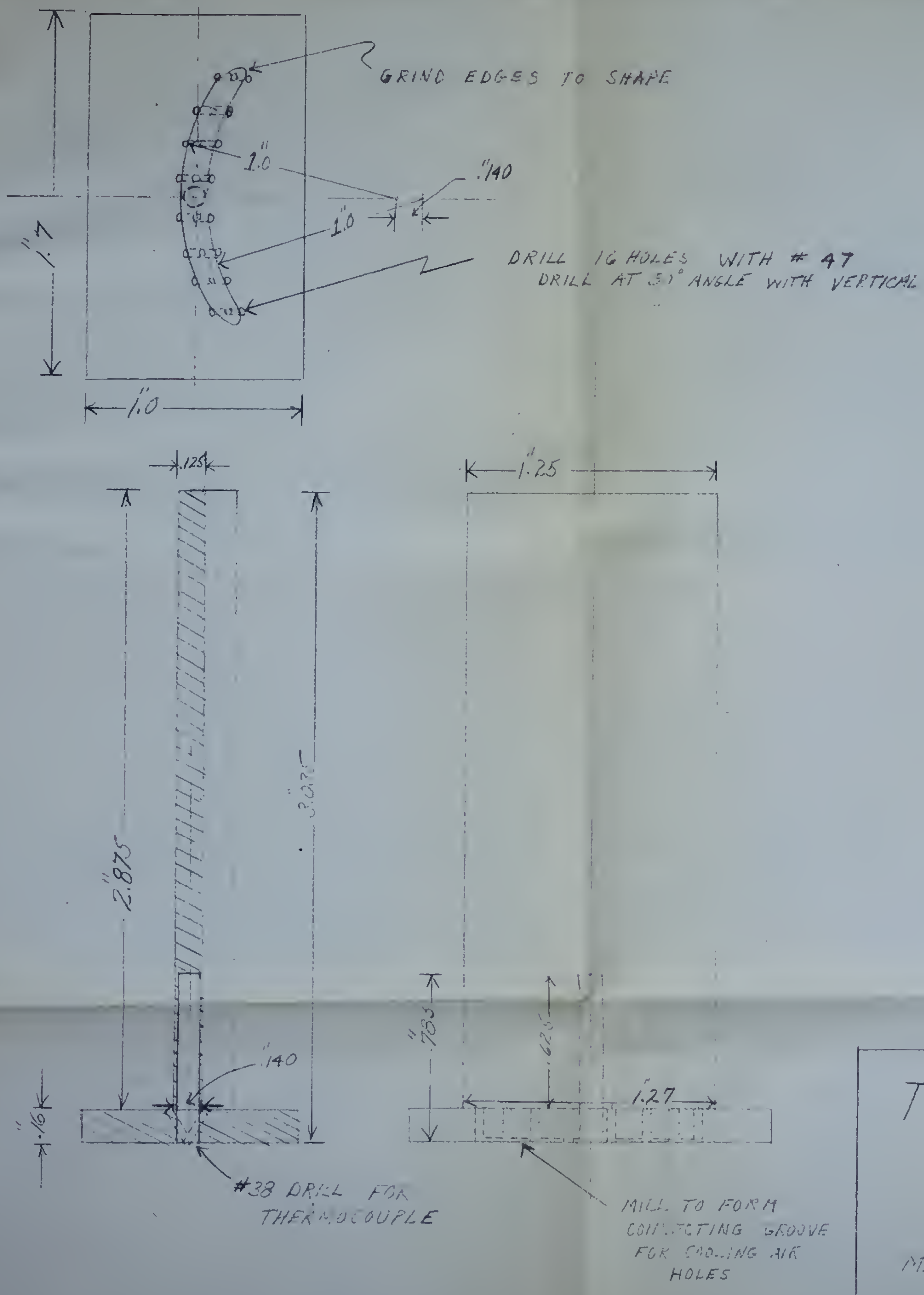


FIGURE 7

TURBINE TEST BLADE

MATERIAL - TOOL STEEL SCALE 2" = 1"

D E DRESSENDORFER 7-28-49

Figure 8

Test Block and Test Blades

Figure 9

Test Block and Blades Assembled

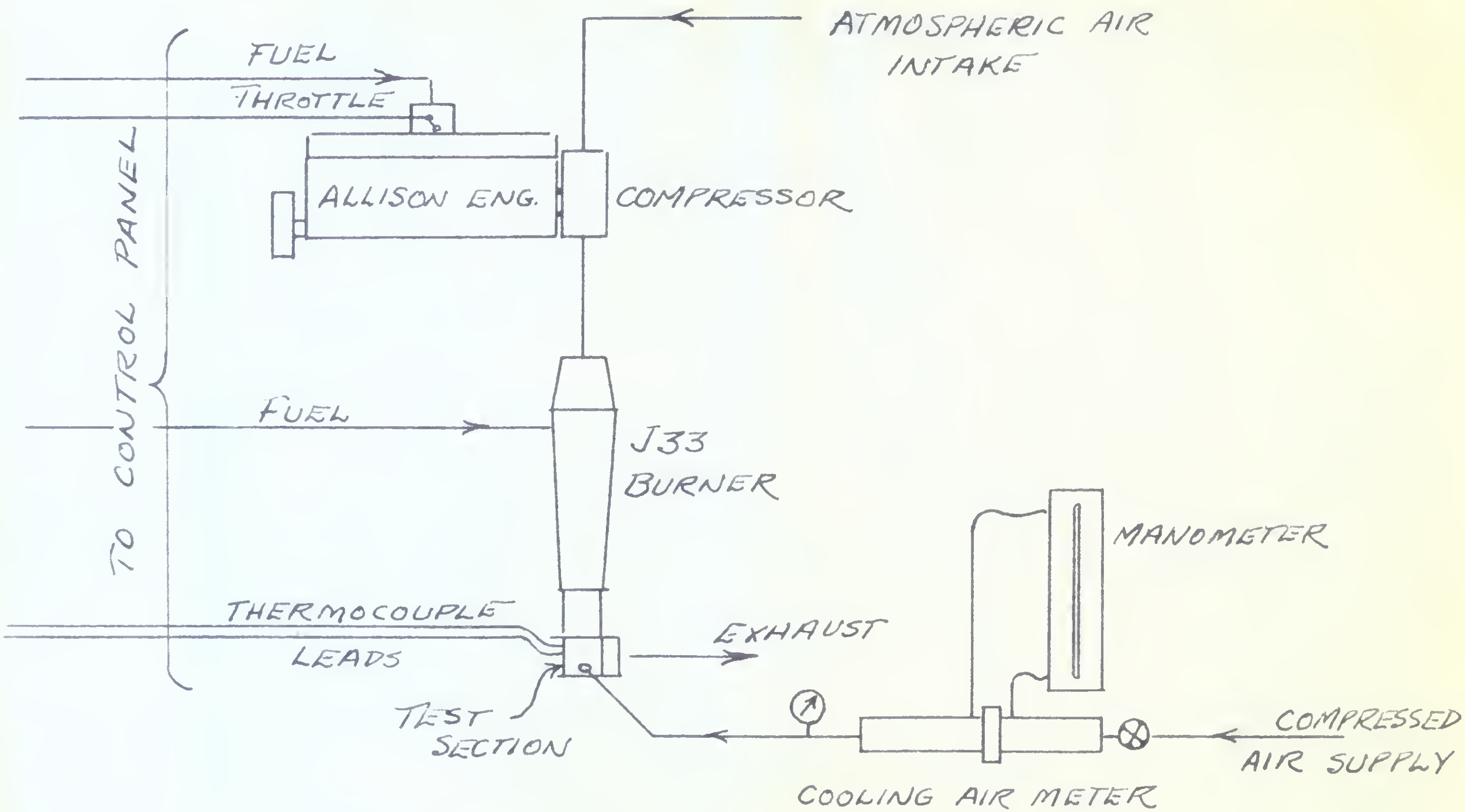


FIGURE 10

SCHEMATIC DIAGRAM OF COMPLETE TEST LAYOUT

Figure 11

Allison V-1710-81 Engine

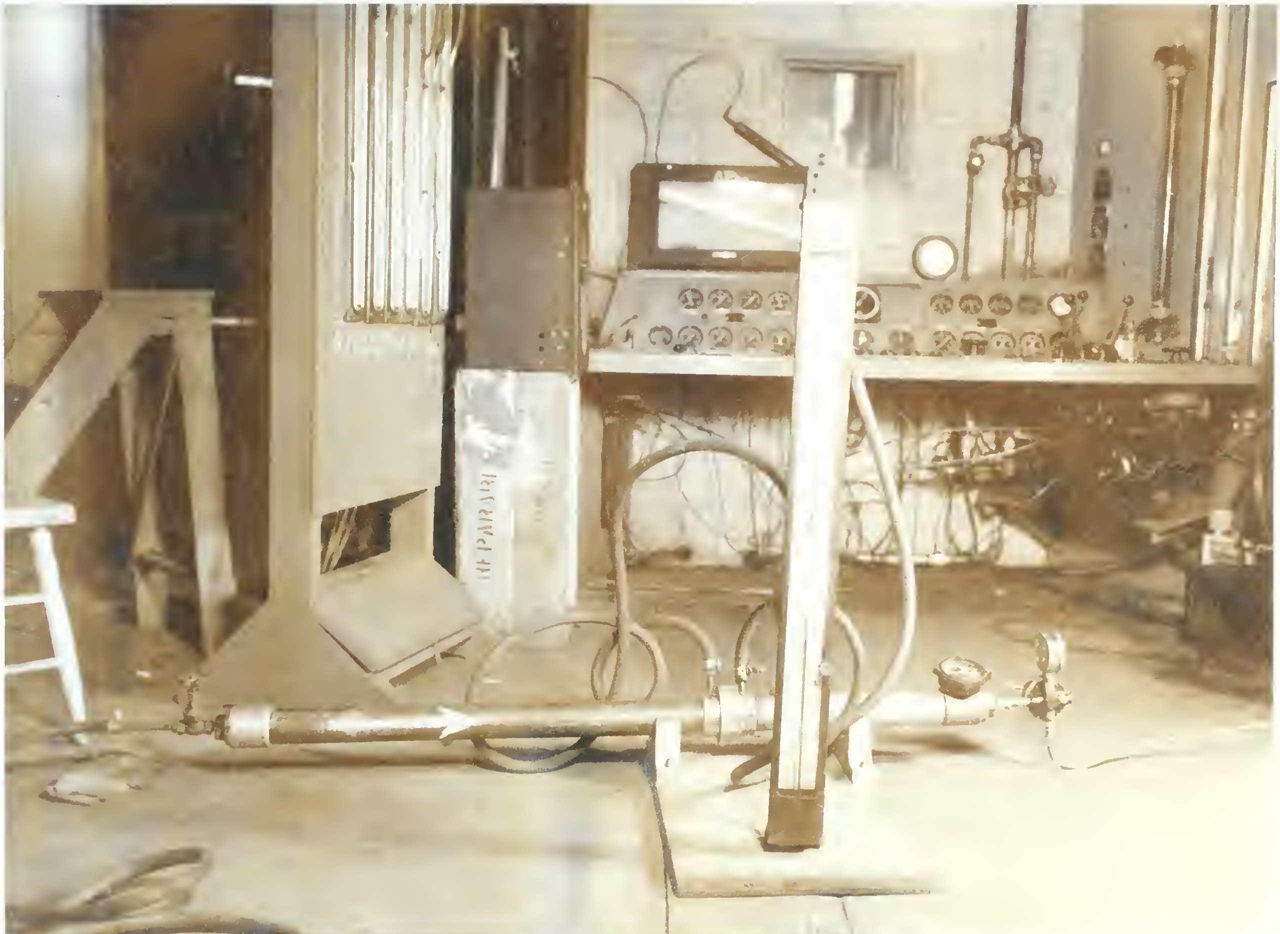


Figure 13

Cooling Air Meter

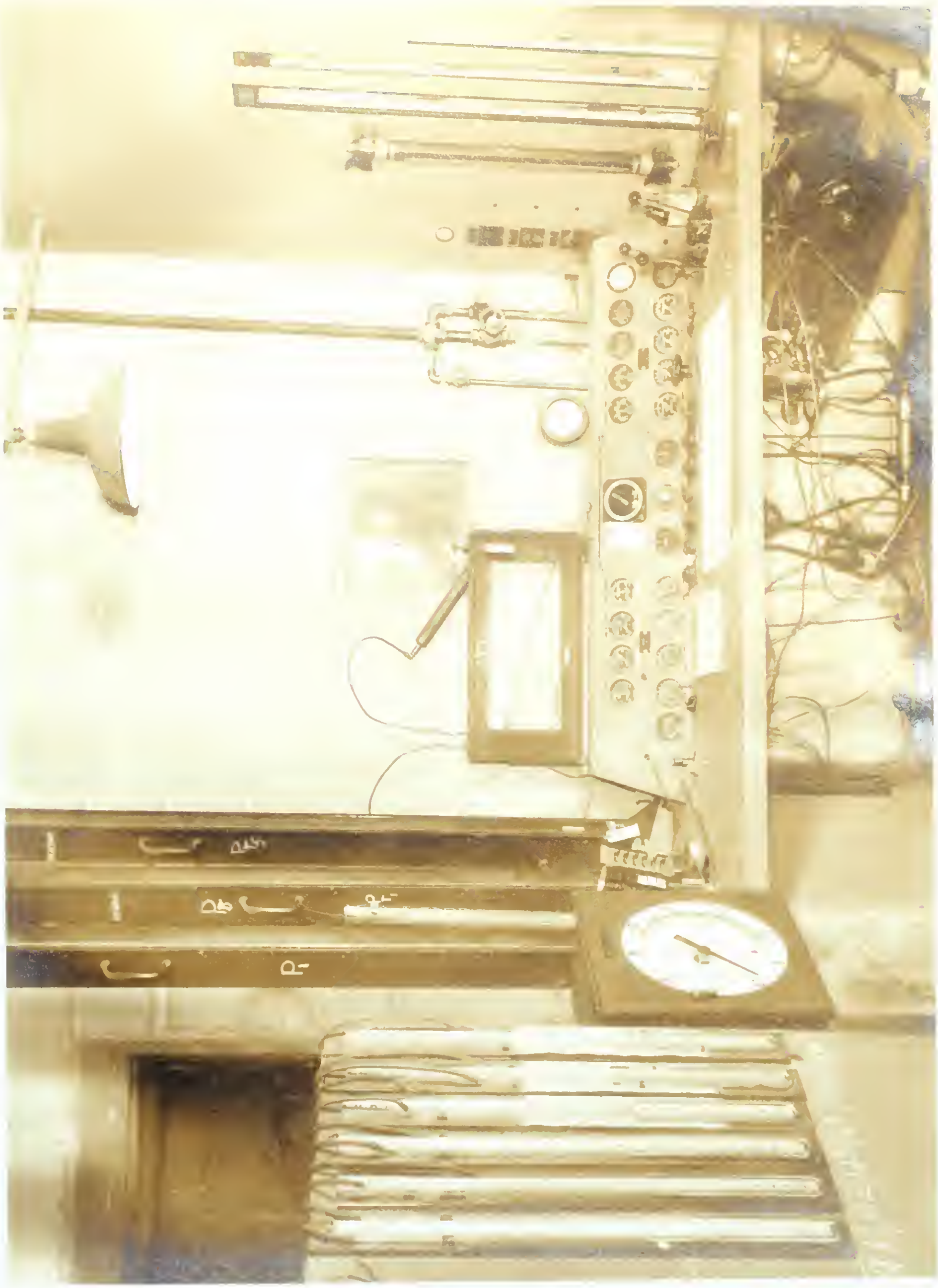
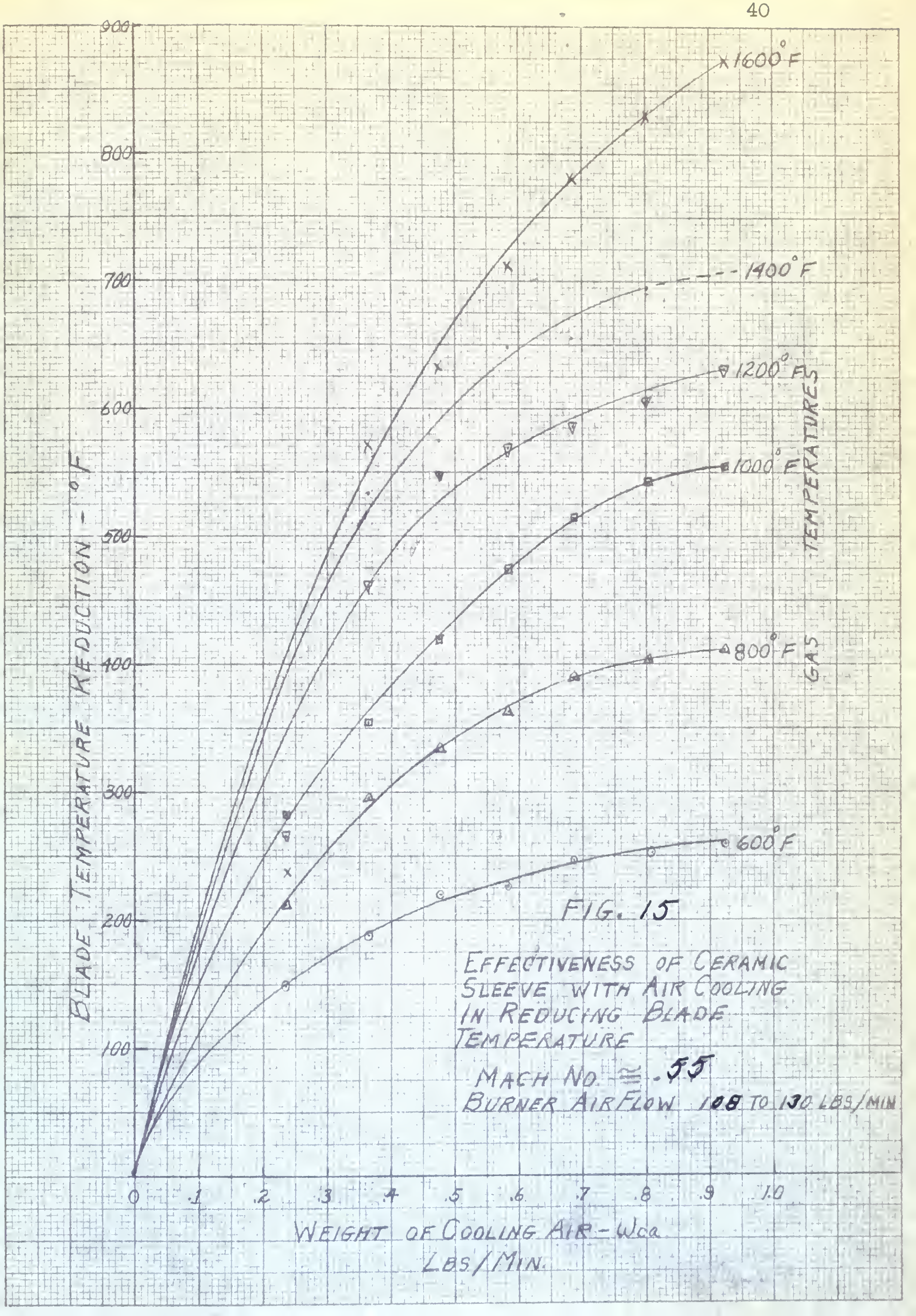


Figure 14

Control Panel



900
800
700
600
500
400
300
200
100
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
BLADE TEMPERATURE REDUCTION - °F
WEIGHT OF COOLING AIR - Wca
LBS/MIN.
1600°F
1400°F
1200°F
1000°F
800°F
600°F
GAS TEMPERATURES
FIG. 15
EFFECTIVENESS OF CERAMIC SLEEVE WITH AIR COOLING IN REDUCING BLADE TEMPERATURE
MACH NO. ≅ .55
BURNER AIRFLOW 108 TO 130 LBS/MIN

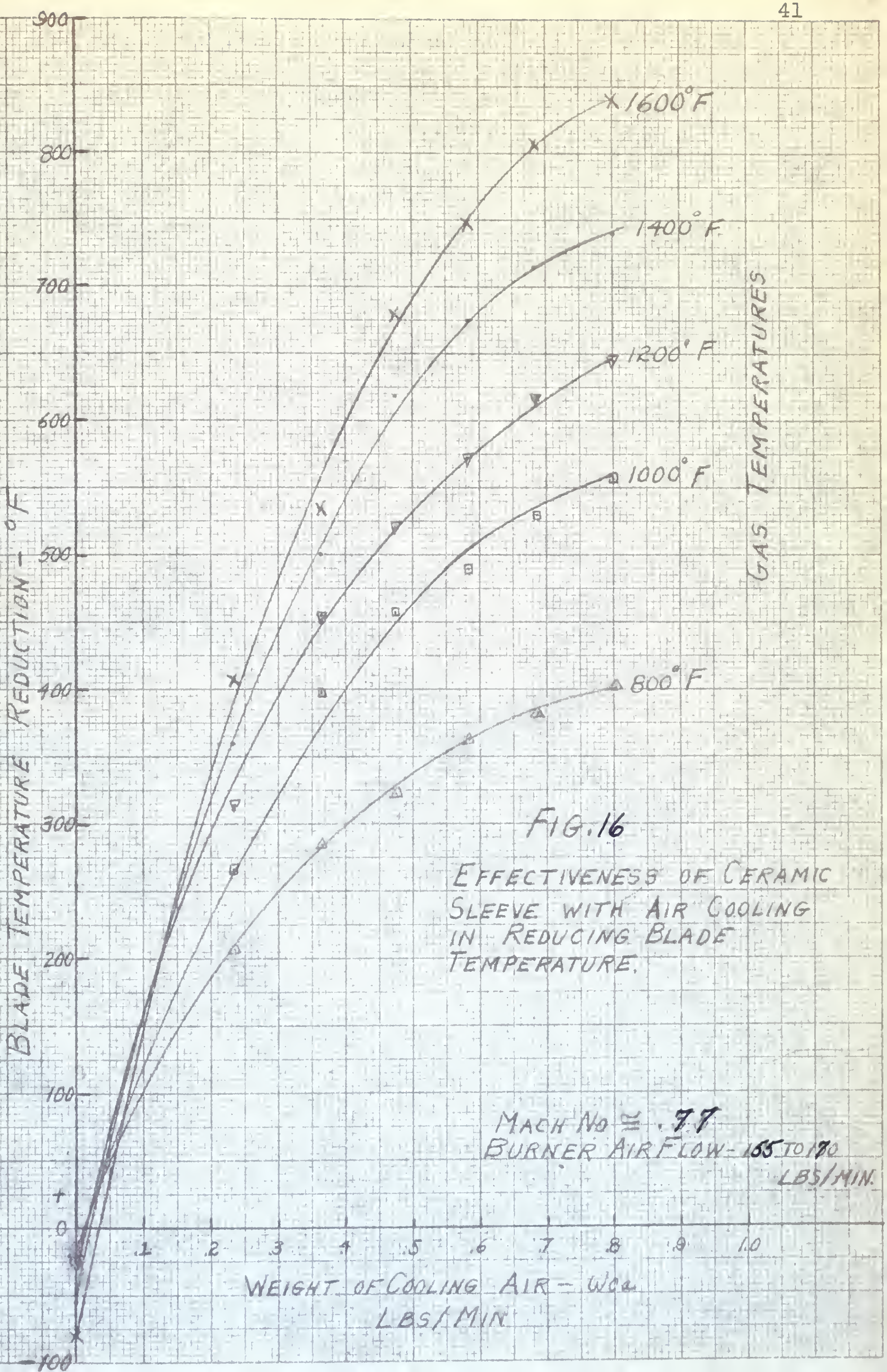


FIG. 16

EFFECTIVENESS OF CERAMIC SLEEVE WITH AIR COOLING IN REDUCING BLADE TEMPERATURE.

APPENDIX

NOMENCLATURE

P_f	FUEL PRESSURE	PSIG
P_3	BURNER INLET STATIC PRESS	"hg
P_{t3}	" " TOTAL "	"hg
P_4	TEST SECTION STATIC PRESS	"hg
P_{t4}	" " TOTAL "	"hg
ΔP_{CA}	COOLING AIR ORIFICE PRESS DROP	"H_2O
ΔP_{BA}	BURNER " " " "	"hg
q	DYNAMIC PRESS	"hg
M_4	TEST SECTION MACH NO.	
T	TEMPERATURE	°F
	SUBSCRIPTS:	
	M UNPROTECTED BLADE	
	C BLADE WITH CERAMIC SLEEVE	
	m, r " ROOT TEMP.	
	g, e GAS, AFFECTING HEAT TRANSFER	
	R TEMPERATURE REDUCTION	
W	WEIGHT FLOW	
	SUBSCRIPTS:	
	CA COOLING AIR	lb/min
	BA BURNER AIR	lb/sec
	f BURNER FUEL	lb/hr

SAMPLE CALCULATIONS

Calculation of Cooling Air Flow Rate

Formulas and Coefficients from "Flow Measurement"-ASME, 1940.

$$W = .0596KD_2^2 \sqrt{\frac{P_2 \Delta P}{T_1}}$$

where K = flow coefficient

D_2 = orifice diameter, inches

P_2 = absolute outlet static pressure, in lbs/in^2

T_1 = inlet air absolute temperature, $^\circ$ F

ΔP = orifice pressure drop in lbs/in^2

W = weight flow, lbs/sec

K = .61 for flange taps and orifice to pipe diameter ratio of .304

D_2 = .75 in

$P_2 = 12 + 14.6 = 26.6$ psia

$\Delta P = .1''/27.7 = .00360$ psi

$T_1 = 540^\circ$ F absolute

$$W = .0596 \times .61 \times (.75)^2 \sqrt{\frac{26.6 \times .00360}{540}} = .00392 \text{ lbs/sec}$$

$W_{om} = .00392 \times 60 = .236$ lbs/min

Calculation of Test Section Mach Number, M_4

$q = \frac{\gamma}{2} p M^2$ where q = dynamic pressure

p = static pressure

$\gamma = 1.3$ (assumed)

$q = P_{t_4} - P_4 = 6.55 - .4$

$q = 6.15$ in H_g

$p = .4 + 29.82 = 30.22$ in H_g

$$M = \sqrt{\frac{2 \times 6.15}{1.3 \times 30.22}}$$

Data taken from Table I
at gas temperature = 1000° F

$M_4 = .559$

SINTERING SCHEDULE

Preheat and bake at 550° F for a period of 4 hours.

Time		
Hours	Minutes	Temperature ° F
1	00	500
1	30	800
2	00	1100
2	30	1400
3	00	1800
3	30	2200
4	00	2550
4	30	2650
5	00	2750
5	30	2850
6	00	2900
6	30	2950
7	00	3000
7	30	3000

REFERENCES AND BIBLIOGRAPHY

1. Ellerbrock, H. H., "NACA Investigations of Gas Turbine Blade Cooling", Journal of the Aeronautical Sciences, Vol. 15, No. 12, December, 1948.

2. Manson, S. S., "Stress Investigations In Gas Turbine Disks and Blades", Society of Automotive Engineers, Preprint #117, January, 1948.

3. Stewart, Robert H., "Literature Survey on the Use of Ceramic Materials for Turbine Blading", Allis-Chalmers Manufacturing Company, Turbo-Power Development Department Report, March, 1948.

4. Geller, R. F., "Ceramic Bodies for Turbo-Jet Blades", Society of Automotive Engineers, Preprint #115, January, 1948.

5. Moore, D. G., Bolz, L. H., and Harrison, W. N., "A Study of Ceramic Coatings for High-Temperature Protection of Molybdenum", NACA, T.N. No. 1626, July, 1948.

6. Berger, A.L., "High-Temperature Ceramic Materials", Technical Data Digest, Vol. 13, No. 23, December, 1948.

7. Hartwig, F. J., Sheflin, B. W., and Jones, Robert J., "Preliminary Investigation of a Gas Turbine with Sillimanite Ceramic Rotor Blades", NACA, T.N. No. 1399, July, 1947.

8. Weber, B. C., "Ceramic Materials for Some Special Applications", Wright Field Technical Report No. F-TR-1163-ND, February, 1948.

9. Hoffman, C.A., Ault, G.M., Gangler, J.J., "Initial Investigation of Carbide-Type Ceramal of 80 Per Cent Titanium Carbide Plus 20 Per Cent Cobalt for Use as Gas Turbine-Blade Material", NACA, T. N. No. 1836, March, 1949.

10. "Flow Measurement" --ASME, 1940.

11. McNamara, E. P., "Ceramics", Vol II, The Pennsylvania State College, Pennsylvania, 1944.

12. McNamara, E. P., "Ceramics", Vol. I and Vol. III, The Pennsylvania State College, State College, Pennsylvania, 1944.

13. Norton, F. H., "Refractories", McGraw-Hill Book Company, New York, 2nd Ed., 1942.

14. Pollmann, Erich, "Temperatures and Stresses in Hollow Blades for Gas Turbines", NACA, T.M., No. 1183, September, 1947.

15. Harrison, W. N., Moore, D. G., Richmond, J. C., "Review of an Investigation of Ceramic Coatings for Metallic Turbine Parts and Other High Temperature Applications", NACA, T.M., No. 1186, March, 1947.

16. Olcott, E. L., "Review of the Ceramic Research Program of the Bureau of Aeronautics", Technical Data Digest, Vol. 14, No. 5, March, 1949.

17. Nobrowsky, A. R., "Ceramic Developments at the National Advisory Committee for Aeronautics", Technical Data Digest, Vol. 14, No. 5, March, 1949.

18. McAdams, W. H., "Heat Transmission", Mc-Graw-Hill Book Company, New York, 2nd Ed., 1942.

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istics of a turbine
blade having a ceramic
sleeve with air cooling.