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RESEARCH MEMORANDUM

OPERATION OF AN EXPERIMENTAL AIR-COOLED TURBOJET ENGINE AT TURBINE-INLET TEMPERATURES FROM 2200° TO 2935° R

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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OPERATION OF AN EXPERIMENTAL AIR-COOLED TURBOJET ENGINE AT TURBINE-INLET TEMPERATURES FROM 2200° TO 2935° R

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SUMMARY

An experimental investigation was conducted on a production-model turbojet engine modified by the substitution of special combustors and air-cooled turbine stator and rotor assemblies to determine some of the problems pertinent to the operation of air-cooled engines at turbine-inlet temperatures in excess of 2200° R. The test engine was assembled from a number of available parts which, although not ideal for such an investigation, served as a research expedient to obtain information that will be valuable in future air-cooled engine operation and design. A supply source of air external to the engine was used for cooling the stator and rotor blades. In addition, provisions were made for refrigerating the cooling air supplied to the turbine rotor blades to keep the temperature of the noncritical rotor blades in a safe range at the high values of turbine-inlet temperature.

The investigation was conducted in an altitude chamber at simulated altitudes of 50,000 and 60,000 feet and a simulated flight Mach number of 0.8. A range of rotor cooling-air flow ratios was covered at each turbine-inlet temperature investigated up to 2580° R. At temperatures above 2580° R, only one turbine rotor flow ratio was set at each gas temperature. The investigation was terminated at a turbine-inlet temperature of 2935° R by failure of an air-cooled turbine stator blade, which in turn damaged the rotor blades. During operation a large leakage developed in the turbine stator cooling-air system upstream from the stator blades. Loss of cooling air through this leakage undoubtedly contributed heavily to the failure of the stator blade. Other engine parts such as the combustion system, turbine disks, tailcone, and tailpipe that were directly affected by the high gas temperatures were in good condition at the completion of the investigation. From a very limited amount of engine operation, it appears that gas-turbine engines can operate satisfactorily at turbine-inlet temperatures of 2500° R or higher by providing cooling to the turbine rotor and stator blades. It also seems that temperatures up to approximately 3000° R are feasible if, in addition to cooling the rotor and stator blades, the combustors have adequate secondary air to cool the transition sections, and there is some, but not a large amount of, tailcone and exhaust-nozzle cooling.

INTRODUCTION

In order to obtain a better understanding of problems that may be involved in the operation of high-temperature, air-cooled turbojet engines, a series of runs was made with an experimental air-cooled engine at successively higher turbine-inlet temperatures until failure of an engine component resulted. A discussion of the mechanical problems encountered during operation of this air-cooled engine at turbine-inlet temperatures from 2200° to 2935° R is presented herein. Heat-transfer results from this investigation over a range of cooling-air to combustion-gas flow ratios are presented in reference 1 for the turbine-inlet temperature range from 2200° to 2580° R.

Turbine cooling studies have been conducted for a number of years on (1) various means of providing blade cooling, (2) fundamental studies of heat transfer, and (3) analytical and experimental studies of the effects of cooling on engine performance. Most of the NACA research up to 1955 is summarized in reference 2. Full-scale air-cooled engine studies that have since been conducted are reported in references 3 to 5. In all previous NACA experimental air-cooled turbine investigations the turbine rotor and turbine rotor blades were the only components that were cooled. Temperature limitations imposed by engine components other than the turbine rotor and turbine rotor blades (i.e., combustors, outer casings, stator blades, etc.) restricted these investigations to turbine-inlet temperatures below about 2200° R.

Analyses of various air-cooled turbine blade configurations by methods such as those outlined in references 6 to 8 indicate that several types of turbine blades can be adequately cooled at turbine-inlet temperatures in excess of 2500° R. Tests on an experimental combustor (ref. 9) also indicated that combustors could be made to perform adequately at combustor-outlet temperatures up to at least 2760° R. No information was available, however, on how all components of an engine would operate simultaneously and which components would provide the first turbine-inlet temperature limitation.

In order to gain some experience, without undue delay, operating at turbine-inlet temperatures higher than those in current use, available engine components (not originally intended for this investigation) were assembled or modified. Production-type tubular combustors were modified in accordance with suggestions in reference 9 so that operation at high turbine-inlet temperatures would be permissible. The turbine rotor and rotor blades were taken from the engine used in reference 2, which presents performance results obtained from a production-model engine that was modified to provide air cooling for noncritical steel turbine rotor blades. An air-cooled adjustable turbine stator assembly originally intended for use in an investigation of some special uncooled turbine rotor blades capable of operation at turbine-inlet temperatures in excess of 2100° R was also available. The stator blades, made from a high-temperature alloy, were of a free-vortex design, while the rotor blades were of an untwisted design. It was realized that with this combination turbine performance would be poor, however, the cooled stator and rotor along with the special modified combustors were assembled into an engine to obtain preliminary information on problems that might be involved in operation of turbojet engines at turbine-inlet temperatures well above 2200° R.

This investigation was divided into two phases. During the first phase heat-transfer data were obtained over a range of turbine-inlet temperatures from about 2200° to 2580° R. The second phase was conducted at turbine-inlet temperatures above 2580° R to obtain operating experience at these higher temperatures and to determine the maximum temperature that could be obtained with this engine. As stated previously, the heat-transfer data obtained during the first phase are reported in reference 1, and this report presents and discusses the mechanical problems encountered during this investigation over the entire range of high turbine-inlet temperatures covered (2200° to 2935° R). The engine operation was conducted at simulated altitudes of 50,000 and 60,000 feet and a simulated flight Mach number of 0.8 in an NACA 10-foot-diameter altitude test chamber. A range of turbine rotor blade cooling-air to combustion-gas flow ratios (hereinafter referred to as cooling-air flow ratios) from 0.02 to 0.138 was covered. Rotor cooling-air temperatures covered a range from about 400° to 700° R at the blade base, and stator cooling-air temperatures varied from 520° to 540° R. Some tentative evaluations of the cooling requirements of various engine parts under the conditions of high turbine-inlet temperature operation are discussed.

APPARATUS AND INSTRUMENTATION

Engine

The engine used was a production-model 12-stage axial-flow turbojet engine modified for this investigation. The modifications consisted of special combustion sections, air-cooled turbine stator and rotor assemblies, and an altered tailcone in place of the standard engine parts (figs. 1 and 2). The rotor and stator cooling air was supplied from a source external to the engine. The rotor cooling air was refrigerated for most of the investigation. The modified engine was installed in a 10-foot-diameter altitude chamber which is described in detail in reference 10.

Combustor modifications. - Previous experience (ref. 3) with attempts to run this type of engine at higher-than-design temperature levels have shown that cooling of some of the structural parts of the engine in the vicinity of the combustor outlet is required. To provide this additional cooling, production-model tubular combustors for this engine were modified as described in reference 9 to permit a larger flow of secondary (cooling) air around the inner and outer peripheries of the combustion chambers. Figure 2 gives a comparison of the production-type and the modified combustors. The support struts of the unmodified transition liner (fig. 2(b)) were removed in the modified version as a precautionary measure, that is, to prevent the possibility of these struts failing and damaging the turbine stator and rotor blades during operation at the proposed high turbine-inlet temperatures. The diameters of the aft end of the combustor liner, the combustor-outlet ring, and the forward end of the transition liner were reduced to increase the annular area between the liners and the combustor casings. A sheet metal shroud was built around the transition liner to direct the cooling-air over the outside surface of the liner. The scalloped surface on the outer radius of the aft end of the transition liner was more deeply drawn for added cooling-flow area, and scallops were also formed on the surface of the inner radius of the liner.

Air-cooled turbine stator assembly. - It was deemed necessary to cool the stator blades in order to ensure their withstanding the high turbine-inlet temperatures proposed for this investigation. An experimental air-cooled stator assembly with adjustable blades which was originally obtained for another research program was installed in the engine to fulfill this need. The adjustability of the stator blades was an incidental feature which proved both useful and detrimental as will be pointed out later.

The stator blades (fig. 3) were fabricated of a high-temperature alloy, N-155. The airfoil shell was formed in two halves of 0.020-inch-thick sheet stock. Corrugated heat-transfer surfaces (figs. 3 and 4) 0.010-inch thick with an amplitude of 0.070 inch and a pitch of 0.070 inch were fitted to the inside surfaces of both halves of the shell. A 0.010-inch-thick insert that extended from near the hub to near the tip of the blade (see fig. 4) was attached to the inner surface of the corrugations. This insert was capped at both ends; thus the cooling air was forced to flow over the corrugated heat-transfer surfaces in passing through the blade. The space between the end cap on the airfoil shell and the end cap of the insert at both the hub and tip of the stator blade (fig. 4) was intended to provide for the distribution of the cooling air over the entire chord length. The blade was assembled with a bearing lug at the root and a support shaft at the tip. All joints in this blade assembly were made with Nicrobraz except the leading- and trailing-edge joints which were heliarc welded.

A cross-sectional sketch of the air-cooled stator assembly is shown in figure 4, and a view of the stator assembly installed in the test engine can be seen in figure 5. The cooling air for the stator blades was supplied from the laboratory service air system (pressures near 55 lb/sq in. abs) through eight $\frac{15}{16}$-inch-inside-diameter tubes to eight segments of a cooling-air manifold on the inner radius of the stator diaphragm as shown in figures 1 and 4. Each segment of this manifold supplied eight stator blades (64 blades in complete stator assembly). From this manifold, the cooling air entered the blades at the hub (inner radius) and flowed radially outward to be discharged outside the engine.

The individual stator blades were supported on the outer casing and contacted the inner radius only at a sliding joint (fig. 4). There was the possibility that the relatively short (about 1/8 in.) inner bearing lug would not be of sufficient length to accommodate the differential expansion between the inner and outer radii of the stator assembly. Because of the required assembly procedure, this condition could not be remedied or improved without considerable redesign.

An attempt was made to reduce the amount of this differential expansion by cooling the turbine outer casing in the vicinity of the attachment points for the stator blades. This was accomplished by directing numerous jets of laboratory service air on the casing from external manifolds encircling the engine at this position (fig. 1). Reversing the stator cooling air flow by introducing cooling air through the exit ports (figs. 4 and 5) would have reduced the effect of this leakage, but such an arrangement was not considered mechanically feasible for this investigation.

Air-cooled turbine rotor assembly. - An air-cooled turbine rotor assembly had previously been built and tested at sea-level static conditions as a part of the NACA research program on turbine cooling (ref. 3). This turbine assembly was used in this investigation. A schematic view of this part of the engine is shown in figure 1. Figure 5 shows the rotor installed in the test engine.

The turbine rotor blades were of a corrugated-insert type of an untwisted design (figs. 5 and 6). The blade shell was formed from a tapered-wall tube (0.020 in. at the tip and 0.040 in. at the root) of noncritical Timken alloy 17-22A(S). The cooling-air-passage configuration consisted of corrugations made from 0.010-inch SAE 4130 sheet stock 0.17 inch in pitch and 0.10 inch in amplitude (fig. 6) placed around the inside perimeter of the blade shell. The inner portion of the blade was blocked off by an insert capped at the root of the blade. The insert and cap were made from 0.010-inch and 0.030-inch SAE 4130 sheet stock, respectively. The blade base was of a bulb-root type and was cast of SAE 4130. Timken 17-22A(S) and SAE 4130 are about 96 and 97 percent iron, respectively. Reference 3 gives a more detailed description of the blade and the fabrication methods followed. Reference 3 also points out that excessive quantities of Nicrobraz present in the shell-to-base joint during the fabrication of the blades used in this investigation caused clogging of a considerable number of coolant passages within the blade base. As a result, the effectiveness of the blade cooling was greatly reduced.

The turbine rotor was a split disk with a downstream cooling-air supply (figs. 1 and 5). Each disk was fabricated from a composite production rotor for this engine consisting of an SAE 4340 hub section and a Timken 16-25-6 rim section. A more detailed description of the turbine rotor is given in reference 3. A 4.0-inch-diameter hole was provided in the center of the downstream (rear) half of the rotor for introduction of the cooling air. The tube ducting air to the rotor was held concentric with the axis of the turbine rotor by pilot bearings (fig. 5) mounted on the small shaft protruding through the center of the air inlet. To ensure proper lubrication of these bearings even when extremely low temperatures (around 295° R) were reached in the refrigerated cooling air, it was necessary to insulate the bearing housing from the direct effects of this cold air. This was accomplished by the addition of a manifold encompassing the bearing housing (fig. 1). Alcohol was circulated in this manifold to provide a thermal shield between the bearing housing and the refrigerated cooling air.

Because the turbine stator and rotor were designed for different types of flow (free-vortex flow for the stator and nontwisted blades for the rotor), this combination of stator and rotor was far from ideal aerodynamically. As stated previously, the mechanical and heat-transfer characteristics of the stator and rotor assemblies were considered problematical. However, the two components were used because they were the only air-cooled turbine components available for this investigation.

Tailcone and exhaust system. - The altered tailcone of reference 3 was used on the test engine during this investigation. The alterations included provisions for ducting cooling air to the rotor and for a turbine rupture shroud over the turbine rotor. A schematic view of this tailcone appears in figure 1. Cooling air for the turbine rotor was ducted to a hole in the rear face of the rotor by means of a piping system that entered the tailcone assembly through one of the support struts. Either refrigerated air or laboratory service air could be supplied to the rotor through this system. No direct cooling was provided for the tailcone. However, elimination of the insulating blanket on the outer cone permitted some convection and radiation cooling by the relatively cool surroundings and test-chamber air passing over the engine.

A 21-inch-diameter tailpipe section about 8 feet in length and a clam-shell adjustable exhaust nozzle completed the exhaust system. An insulating blanket was used on the tailpipe and exhaust-nozzle sections to minimize radiation corrections to the readings of thermocouples in this vicinity. To ensure proper functioning of the exhaust nozzle, a supply of laboratory service air was directed against the movable clamshell. A reflective high-temperature paint was used on the inner surfaces of the tailcone, tailpipe, and exhaust nozzle as a precautionary insulating measure.

Instrumentation

Temperatures. - The temperatures of the turbine rotor blades at the one-third-span position were read by means of thermocouples installed on two diametrically opposite blades (ref. 1). On one of these blades, thermocouples were located on the leading edge, midchord suction surface, midchord pressure surface, and the trailing edge. On the other blade, thermocouples were located at corresponding leading- and trailing-edge positions only. Turbine-disk temperatures were read from thermocouples located at 11.4- and 13-inch radii on both the front and rear disks, the 13-inch radius being on the rim of the disk. Five thermocouples were used to read temperatures on three of the turbine stator blades. Thermocouples were installed at the midspan position on the leading edge, midchord suction surface, and trailing edge of one blade, and on the trailing edge only of the other two. All the thermocouples on the turbine rotor blades, turbine disks, and the turbine stator were chromel-alumel. The readings from the thermocouples on the rotating parts were transferred to the recording instruments by means of a slipring thermocouple pickup located at the front of the engine. The thermocouples were connected to the pickup by leads that passed through drilled holes on the centerline of the compressor and turbine shafts.

Temperatures of the engine air were measured with iron-constantan thermocouples at the compressor inlet and discharge. Exhaust-gas temperatures were measured in the tailpipe with a number of chromel-alumel thermocouples. The temperature of the turbine rotor cooling air was measured with stationary iron-constantan thermocouples at a point just upstream of the entrance to the rotor. The temperature of this cooling air was also measured by two chromel-alumel thermocouples located in the base of the first instrumented turbine rotor blade described above. These latter two thermocouples were part of the rotating thermocouple system.

Flow measurements. - The engine air flow rate was measured by means of a calibrated Venturi tube upstream of the compressor inlet. The quantities of cooling air flowing to the stator and rotor were measured by static-pressure taps, integrating total-pressure tubes, and thermocouples located in the supply lines to each of these components.

EXPERIMENTAL PROCEDURE

This investigation was divided into two phases. During the first phase, rotor blade cooling-air flow ratio was varied at various settings of turbine-inlet temperature to obtain turbine rotor blade heat-transfer data (ref. 1). A maximum engine speed was set at predetermined values of altitude, flight Mach number, turbine-inlet temperature, and cooling-air flow ratio. After data were recorded at this operating condition, a range of cooling-air flow ratios was covered by successively reducing the cooling-air flow rate while other operating conditions were maintained constant. Engine and temperature data were recorded at each setting of cooling-air flow ratio. A range of turbine-inlet temperatures (calculated by the method described in ref. 11) from 2200° to 2580° R at simulated altitudes of 50,000 and 60,000 feet and a simulated flight Mach number of 0.8 was investigated. The engine speeds (corrected for compressor-inlet conditions) during this phase varied from about 85 to 90 percent of rated. At the low end of the turbine-inlet temperature range just mentioned, the rotor blades were cooled with laboratory service air. The cooling-air flow ratios were varied from 0.09 to 0.035. The temperature of the cooling air at the inlet to the turbine rotor blades (fig. 1) was in the range of 600° to 700° R. The remainder of the testing in the first phase was conducted with refrigerated air used to cool the turbine rotor blades. With this refrigerated air supply, the cooling-air flow ratios were varied from 0.116 to 0.02 while the cooling-air temperature at the inlet to the turbine rotor blades varied from about 400° to 600° R. The lowest coolant flow was determined by the average rotor blade temperature, which generally was not permitted to exceed 1500° R. The duration of each test setting was determined during this phase by the length of time required to obtain the turbine rotor blade heat-transfer data reported in reference 1.

During the second phase of the investigation, only one setting of turbine rotor blade cooling-air flow ratio was made at each setting of turbine-inlet temperature. The turbine-inlet temperature was increased in successive steps above the 2580° R level reached in the first phase to determine the maximum temperature that could be obtained with this engine. Refrigerated turbine rotor blade cooling air was used during this phase. As the turbine-inlet temperature increased, the cooling-air flow ratio was increased (to keep the rotor blades from overheating) from an initial value of 0.118 until a value of 0.138 was reached at a turbine-inlet temperature of 2935° R. The temperature of the cooling air at the inlet to the rotor blades varied from about 400° to 600° R for the range of turbine-inlet temperatures covered in this phase of the investigation. During this second phase the corrected engine speed varied from about 92 to 94 percent of rated. The length of time at each turbine-inlet temperature during this phase was determined by the time required to record data similar to that obtained in the first phase.

During the entire investigation, the cooling-air flow to the stator blades was maintained at the maximum quantity permissible with available laboratory service air pressurized to about 55 pounds per square inch absolute. This resulted in cooling-air temperatures at the inlet to the stator blades ranging from 520° to 540° R.

RESULTS AND DISCUSSION

A summary of the engine operating conditions is given in table I. A total operating time of about 6 hours was run at turbine-inlet temperatures from about 2200° to 2935° R.

Because of the poor turbine design, this engine was incapable of operation at rated speed (7950 rpm). The maximum engine speed attained was 94 percent of corrected rated speed. Decreasing the area of the adjustable turbine stators in an effort to increase the maximum engine speed resulted in compressor surge. The area setting of the stators was reset twice during the investigation in order to determine an area that would be the best compromise between high turbine work and compressor surge margin. Because of thermal expansion and deformation, the exact stator areas during engine operation were unknown.

The primary effect of the reduced engine speed was a lowering of turbine blade stresses. But since neither the blade configuration nor the blade material was representative of what would be encountered in an engine specifically designed for high-turbine-inlet-temperature operation, the stress level (16,500 psi at blade root) was not of major significance. The tests did show experimental trends of cooling requirements for corrugated-insert turbine rotor blades with changes in turbine-inlet temperature and altitude (ref. 1), and the effects of turbine-inlet temperature on other parts of the engine.

The following sections discuss the operating temperatures and physical aspects of the major components in the hot section of the engine used in this investigation. Based on this study and the study reported in reference 1, some tentative conclusions can be drawn regarding possibilities of air-cooled engines operating at high turbine-inlet temperatures and some problems involved. It should be remembered, however, that these conclusions are based on a very limited amount of operation. Continued engine operation at very high turbine-inlet temperatures may bring to light problems that were not apparent in this investigation.

Combustion System

There were no thermocouples located on the turbine casing or on the modified combustor and transition liners (figs. 2 and 4), but the heat patterns observed on these parts during and at the conclusion of the investigation gave a good indication of the range of the wall temperatures. The forward portion of the turbine casing which encloses the transition liners (fig. 4) was protected from the hot gases by the large flow of cooling air around the modified combustion system. No heat pattern was observed here during operation. At the rear portion of the turbine casing where the adjustable stator blades were mounted, this cooling air was not in direct contact with the casing (fig. 4). Excessive heating of this region was observed, and large radial expansion resulted, as will be discussed in connection with the turbine stator. The heat patterns observed on the modified combustor and transition liners at the conclusion of the investigation indicated that at no time were the wall temperatures excessively high. There were no signs of erosion or deformation on these parts due to the high temperatures involved. The modified combustor liners satisfied the need for cooling to adjacent engine parts at least for this short-term test.

A single combustor of the same design as used in this investigation had previously been tested and results are reported in reference 9 as previously mentioned. The tests were conducted with average combustor-outlet temperatures up to 2760° R, and 32 hours of endurance operation were obtained at average combustor-outlet temperatures in the range from 2460° to 2760° R. Operation for 23 of the hours was conducted at the higher temperature. No warping or burning of parts was observed. In the investigation reported herein, the combustors were operated at higher temperatures, but for shorter periods of time. Since the combustors were in excellent condition at the completion of the investigation, it appears that there will be no serious combustor problems associated with operation up to a combustor-outlet temperature of about 3000° R, if the proper type of design is utilized.

Turbine Stator

Because of the leakage of stator cooling air, the exact quantity of air delivered to the stator blades was not known. This leakage resulted from the differential expansion between the inner and outer radius of the stator assembly which caused the stator blades to separate from the cooling-air manifold at the inner radius (fig. 4). The external cooling of the turbine casing in the vicinity of the stator was not sufficient to compensate for this expansion. Because of the leakage, correlations between cooling-air flow quantity and stator blade temperature and comparisons of temperatures at the various chordwise positions were not generally meaningful. However, to give some indication of the chordwise temperature gradients that did exist, some measured values are quoted herein. At turbine-inlet temperatures of 2200° to 2250° R, the leading-edge temperatures were about 1650° R. For the same conditions, the midchord temperatures were about 1235° R, and the trailing-edge temperatures varied from 1800° to 2000° R on the three instrumented blades. During the second phase of the investigation, maximum temperatures of 2500° R on the leading edge and 2435° R on the trailing edge were recorded. (The midchord thermocouple was not functioning by that time.) The accuracy of the thermocouple readings during the second phase is questionable because of damage to the thermocouple leads.

The loss of stator cooling air through leakage penalized the cooling effectiveness of the stator to a large but indeterminable degree. The lowered cooling effectiveness of the stator plus some rather severe temperature distributions at the exit to the combustors made the stator the component that determined the maximum permissible turbine-inlet temperature in this investigation. At the end of the first phase of the investigation, a few small spanwise and chordwise cracks had developed in the leading edges of some of the stator blades located directly behind the combustors. Erosion and buckling of the shell surfaces in these hot regions of the stator blades were also evident. During the second phase of the investigation, one of these stator blades failed at a chordwise break about 3/4 inch from the blade tip. Inspection of the combustion chamber in front of this blade showed that the fuel nozzle was not seated properly with respect to the combustor liner. The severe heat damage to blades adjacent to the failed blade indicated that this misalinement of parts caused an adverse temperature profile which resulted in the blade failure.

Figure 7 shows views of the leading edges of the stator blades behind two of the combustion chambers. Figure 7(a) shows the section with the most severe damage (exclusive of the sector in which the failure described above occurred), and figure 7(b) shows the sector with the least damage. The damage patterns shown here indicate that the peak temperature of the gas profile at the stator inlet was between one-half and three-quarters of the span distance from the hub to the tip of the blade. Severe burning of the leading edges was apparent on more than half the blades in the stator assembly.

Differential thermal expansion such as that which was encountered in this stator assembly will be a concern in any engine operating at high gas temperatures with air-cooled components. Proper allowances must be made in the design for these expansions. From the results of this investigation, the turbine-inlet temperature limitation imposed by corrugated-insert stator blades is not known. However, with a type of design where cooling-air leakage could be avoided and where more cooling air could be directed to the leading and trailing edges of the stator blades, average turbine-inlet temperatures at least as high as those encountered in this investigation (2935° R) would be feasible. This conclusion is drawn with the reservation that the combustor-outlet temperatures be reasonably uniform so that some of the stator blades would not be subjected to gas temperatures excessively higher than the average gas temperature.

Turbine Rotor Blades

The average turbine rotor blade temperatures registered during the first phase of the investigation are reported in detail in reference 1. During the second phase, the average rotor blade temperature fluctuated between 1300° and 1400° R because of the simultaneous changes in turbine-inlet temperature, cooling-air flow ratio, and temperature of the refrigerated cooling air. The highest local temperatures recorded during these tests were 2039° R at the leading edge, 1375° R at the midchord pressure surface, and 1384° R at the trailing edge. These temperatures were read at a turbine-inlet gas temperature of 2935° R, a cooling-air flow ratio of 0.138, and with a cooling-air temperature at the rotor hub of 295° R. At these conditions, the temperature of the cooling air at the base of the rotor blade was 384° R. After a series of runs had been made with turbine-inlet temperatures of 2400° to 2600° R, it was noted that the leading edges of a number of the turbine rotor blades stretched excessively (as much as 1/16 in. in some cases). This elongation was confined to a region within about 1/2 inch from the leading edge chordwise and about $1\frac{1}{2}$ inches from the blade tip spanwise. The highest temperature recorded on the two instrumented blades prior to the discovery of blade stretch was 1900° R on the leading edge at a turbine-inlet temperature of 2595° R and a cooling-air flow ratio of 0.07. From the temperature pattern shown on the stator blades, it appears that the peak temperature of the turbine-inlet temperature profile occurs between the half- and three-quarter-span position. This would indicate that the metal temperature in the region of the stretch was probably higher than the 1900° R value recorded at the one-third-span location. Data from various sources show that the tensile strength of 17-22A(S) is very low in this range of temperatures. To complete the investigation, the blade tips were ground off to the original tip radius.

The turbine rotor blades were damaged by the failure of a stator blade to the extent that it was impossible to determine whether there was any further elongation of the blades during the high-temperature operation of the second phase. However, the rotor blades did not provide a turbine-inlet temperature limitation. From the results of this investigation, it is difficult to make definite conclusions on limiting turbine-inlet temperature levels that would be directly applicable for other turbine rotor blades for the following reasons: (1) these rotor blades were made of noncritical materials incapable of withstanding very high temperatures, (2) the cooling configuration used was relatively ineffective, (3) a number of cooling-air passages were blocked by the excess braze material, (4) the stress level was low because of the low engine speed, and (5) the cooling air for the rotor blades was refrigerated to very low temperatures. It is believed, however, that blades made of better materials and having a more effective coolant passage configuration should operate safely at temperature levels at least as high as those encountered in this investigation.

Turbine Disks

The heat-transfer part of this investigation reported in reference 1 was primarily an investigation of the turbine rotor blades. Therefore, a given cooling-air flow condition was maintained only long enough to allow the rotor blade temperatures to stabilize and to be recorded. For the thin-shell rotor blades, the time required for temperature stabilization was relatively short. As a result, the temperatures in the heavy sections of the turbine disk did not have time to stabilize. However, it is believed that the measured disk temperatures give an indication of the temperature levels and the gradients between disks that may be expected in an air-cooled turbine rotor of the split-disk type operating under high-gas-temperature conditions.

At no time during the investigation did the disk temperatures reach a point where metal strength would begin to decrease appreciably. The temperatures registered at the rims of the two halves of the rotor were in the range of 900° to 1100° R. Temperature differences of the order of 0° to 50° R between the two halves at this radius existed during most of the investigation. At the 11.4-inch radius the disk temperatures varied from 500° to 900° R with changes in turbine-inlet temperature, cooling-air flow ratio, and cooling-air temperature. Temperature differences between the two disks were generally higher at this radius, ranging from 50° to 200° R. The analysis of reference 12 gives some indication of the stresses that can be introduced if the two halves of the rotor operate at different temperature levels. The temperature differences encountered in this investigation did not cause excessive stresses.

Tailcone and Other Engine Parts

Other parts of the engine which formed the passage for the hot combustion gases were also subjected to severe operating conditions. The tailcone, tailpipe, and exhaust nozzle were the major components in this category. There were no thermocouples or other temperature measuring devices installed on these parts, but visual inspections during operation and after operation afforded an evaluation of the effects of high temperatures for the short time involved in this investigation.

Some warpage of the tailcone due to alterations and previous operation was evident prior to the start of this investigation. Visual inspection at the completion of the investigation showed slight additional warpage, but none of major proportions. The modification of the combustion system to provide more secondary air to cool the transition liners may have caused a relatively cool layer of air to flow along the root and tip regions of the turbine. If such cool layers extended into the tailcone, they may have provided some protection for the surfaces of the inner and outer cones. The cooling air being discharged at the tips of the turbine rotor blades may also have contributed some cooling of the surface of the outer cone. The support struts of the tailcone (fig. 1) which had no cooling held up surprisingly well for the operating conditions of this test. The tailpipe and exhaust-nozzle section showed no signs of excessive warpage at the end of the test.

Because this test was of short duration, the condition of the tailcone, tailpipe, and exhaust nozzle at the completion of the test cannot be considered to be representative of that expected for operation of longer duration. Continuous operation at turbine-inlet temperatures of 2935° F or higher would undoubtedly require some cooling of these parts. However, this investigation indicated that the quantity of cooling required would probably be small and that considerable increase in turbine-inlet temperature above that possible with uncooled turbines should be possible without using any cooling on these parts (except for the adjustable lips of the exhaust nozzle).

From this investigation it appears that gas-turbine engines can be operated at turbine-inlet temperatures of 2500° R or higher by providing cooling to the turbine rotor and stator blades. Temperatures of about 3000° R seem feasible if, in addition to cooling the rotor and stator blades, the combustors have adequate secondary air to cool the transition sections and there is some, but not a large amount of tailcone and exhaust-nozzle cooling.

SUMMARY OF RESULTS

The following observations were made during a short-term experimental investigation conducted on a modified production-model turbojet engine equipped with air-cooled stator and rotor assemblies and operated at turbine-inlet temperatures as high as 2935° R:

1. A modification of the standard combustion system to provide increased flow of secondary (cooling) air provided adequate protection for the combustion-chamber liners and adjacent engine parts.

2. The stator assembly proved to be the component which determined the limiting operation temperature. This was partly due to differential thermal expansion which caused the stator blades to partially separate from the support and air supply on the inner radius of the stator assembly. Complete failure of one stator blade and severe damage to most of the others resulted. Mechanical design changes could probably eliminate most of the trouble due to differential expansion and improve distribution of cooling air to all parts of the blades.

3. No general evaluation of air-cooled corrugated-insert turbine rotor blades at high gas temperatures could be made from this investigation because of the use of refrigerated cooling air to protect the noncritical blade material, the low cooling effectiveness of this particular corrugated blade, and the lower-than-rated-speed operation. However, in spite of

excessive stretching in a localized leading-edge region on the tips of some of these blades, the rotor blades were not the component that established the limiting gas temperature for this engine.

4. The good condition of the tailcone and tailpipe at the conclusion of the investigation without the benefit of cooling indicated that these components could probably withstand sustained operation at high gas temperatures with the use of only a relatively small quantity of cooling.

5. It appears that gas-turbine engines can be made to operate at turbine-inlet temperatures of 2500° R or higher by providing cooling to the turbine rotor and stator blades. It also appears that temperatures up to approximately 3000° R are feasible if, in addition to cooling the rotor and stator blades, the combustors have adequate secondary air to cool the transition sections and there is some, but not a large amount of, tailcone and exhaust-nozzle cooling.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, April 26, 1956

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TABLE I. - SUMMARY OF ENGINE OPERATING CONDITIONS

Calculated average turbine-inlet temperature, °R	Altitude, ft	Approximate corrected engine speed, percent of rated	Range of rotor cooling-air flow ratios	Range of rotor cooling-air temperatures at blade base, °R	Operating time, hr
2200 to 2260	50,000	88 to 89	0.020 to 0.080	430 to 710	$2\frac{3}{4}$
2520	50,000	90	.077 to .104	385 to 420	$\frac{1}{2}$
2300	60,000	88	.036 to .077	450 to 605	$1\frac{1}{4}$
2380	60,000	85	.031 to .102	424 to 660	$\frac{1}{2}$
2580	60,000	86	.070 to .114	400 to 470	$\frac{1}{2}$
2600 to 2935	60,000	92 to 94	.118 to .138	380 to 610	$\frac{1}{2}$

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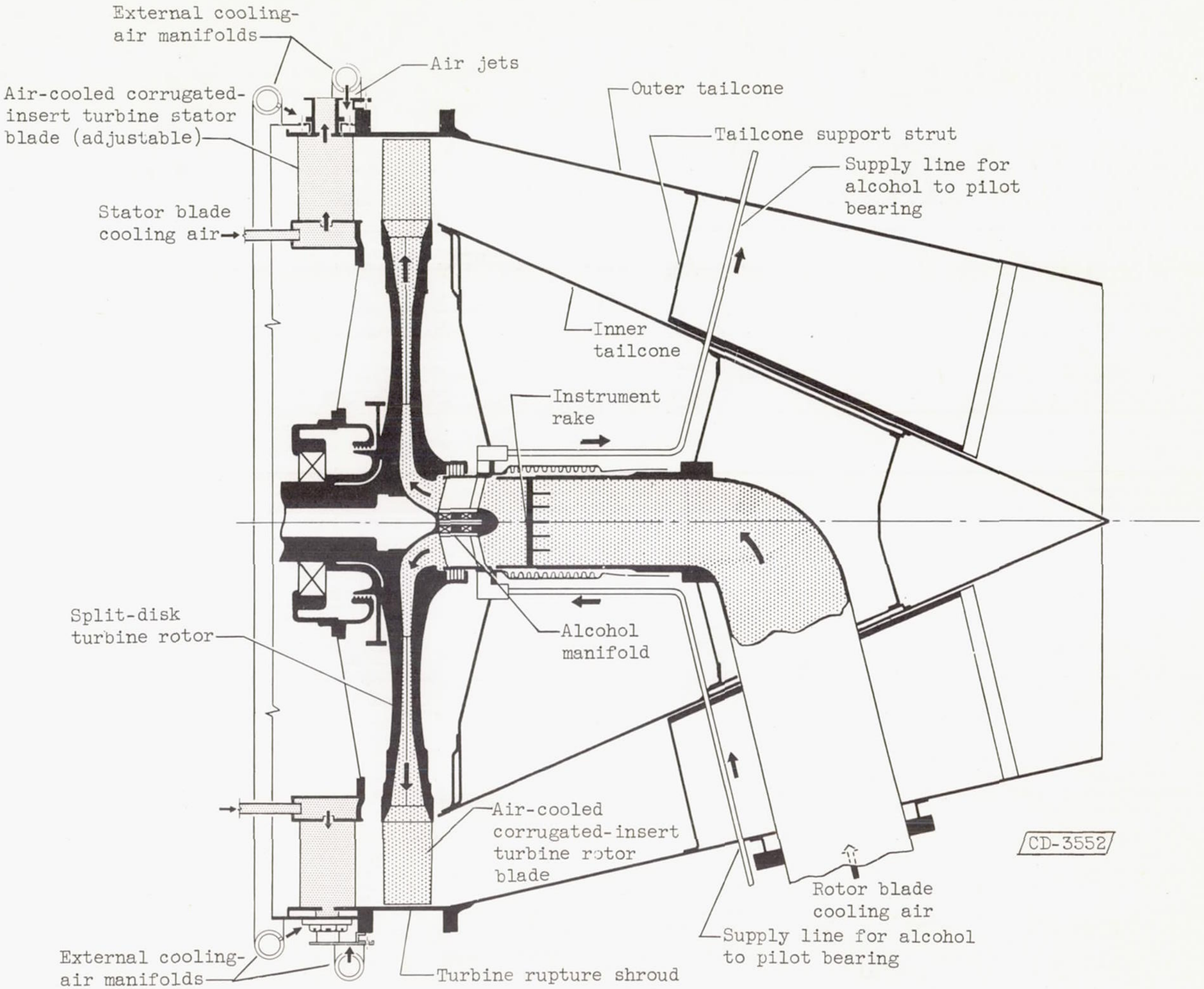

Figure 1. - Schematic view of air-cooled turbine component and tailcone.

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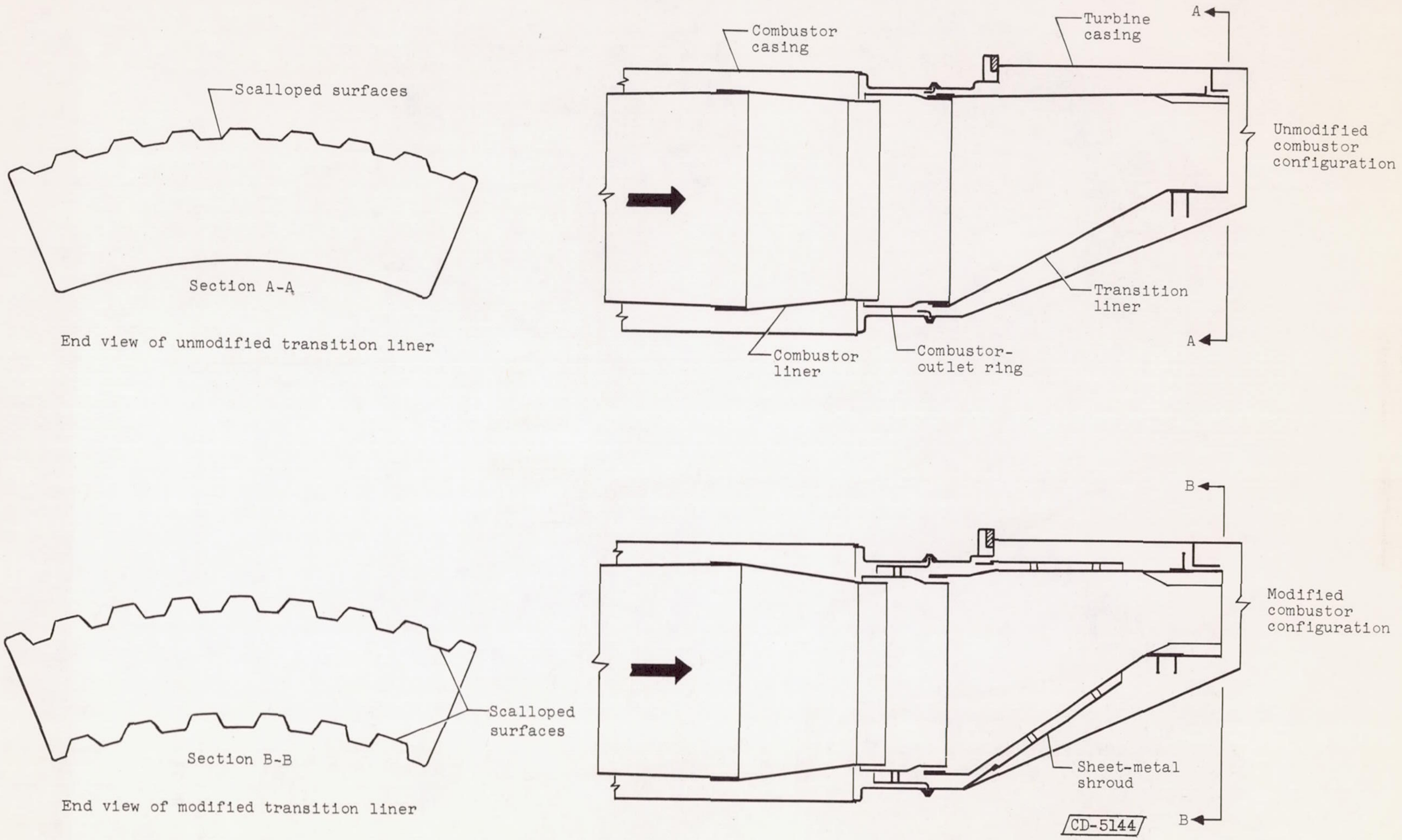
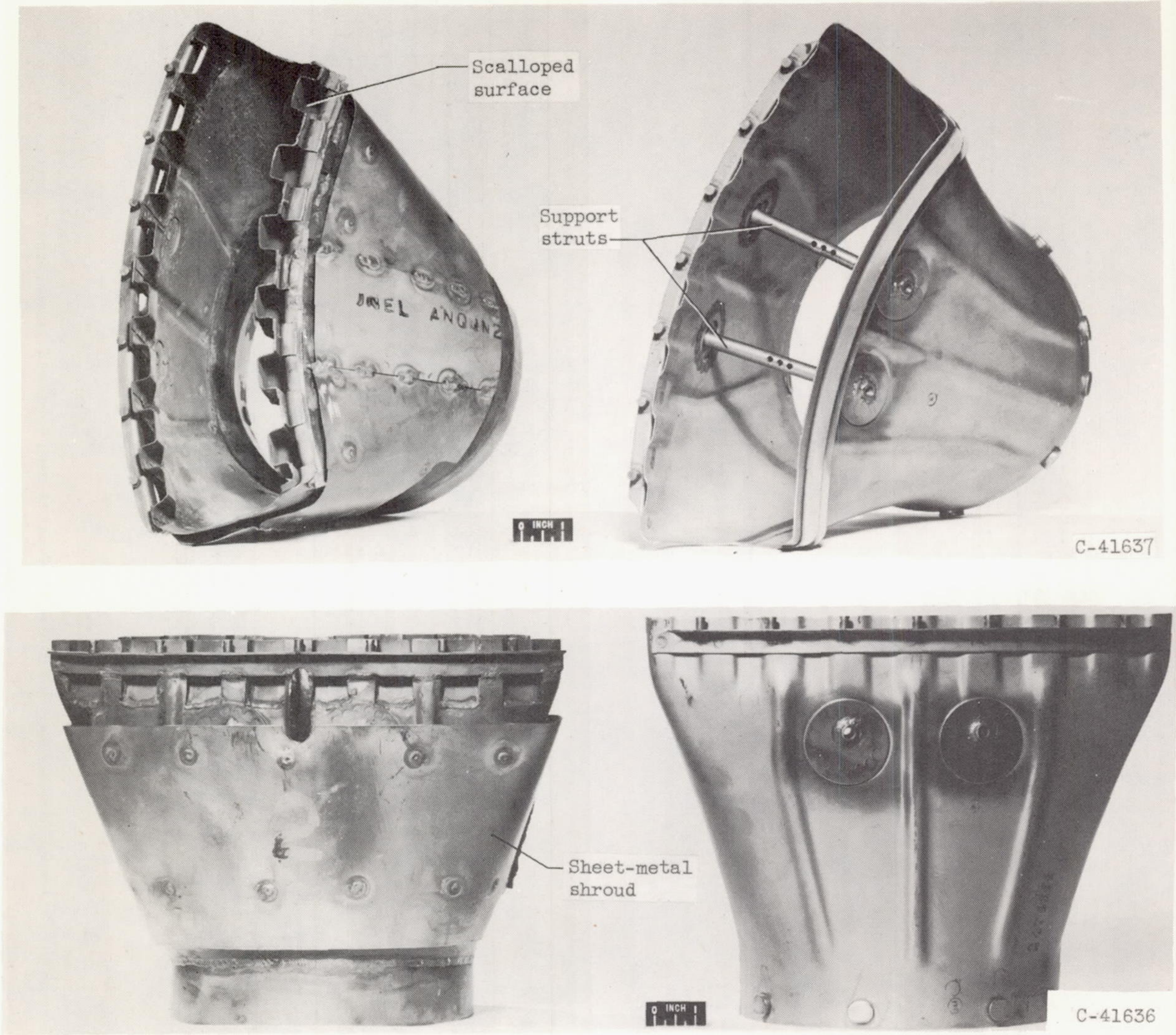


(a) Cross-sectional drawings of combustors and liners.

Figure 2. - Modified and unmodified production-model tubular combustors and transition liners.

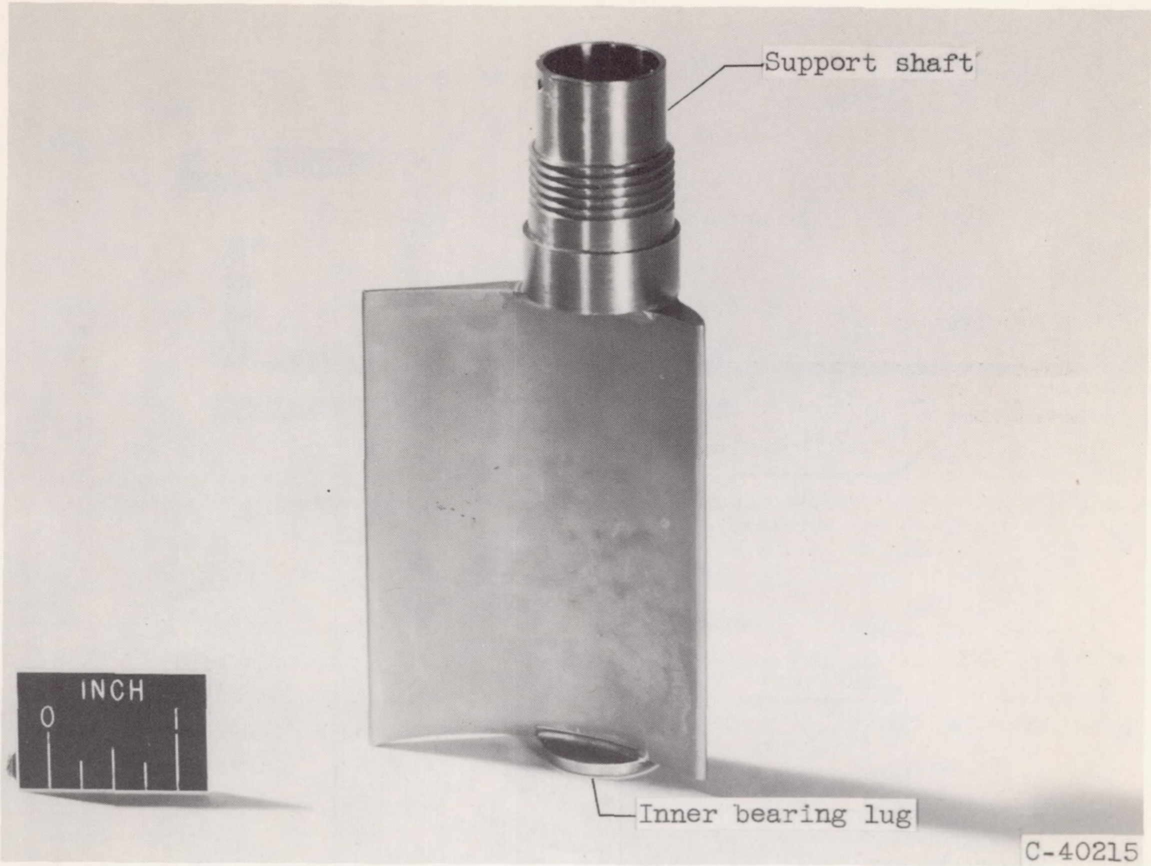


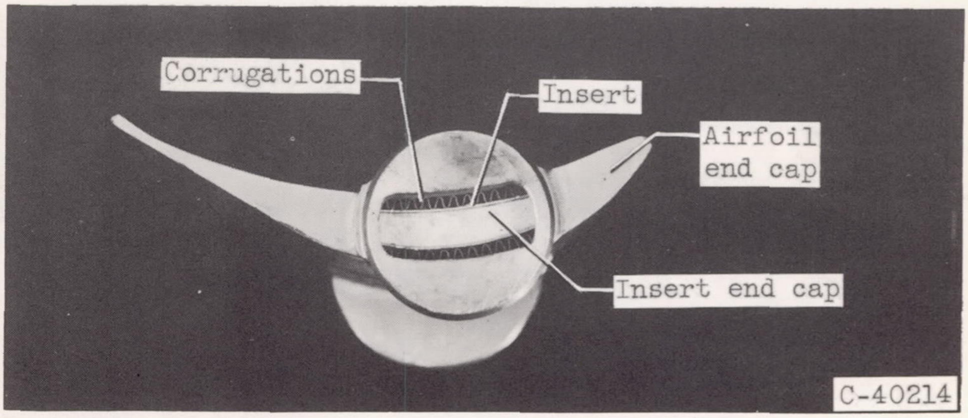
Modified liner

Unmodified liner

(b) Photographs of sheet metal transition liners.

Figure 2. - Concluded. Modified and unmodified production-model tubular combustors and transition liners.





End view of stator blade showing cooling-air entrance

Figure 3. - Adjustable air-cooled corrugated-insert turbine stator blade.

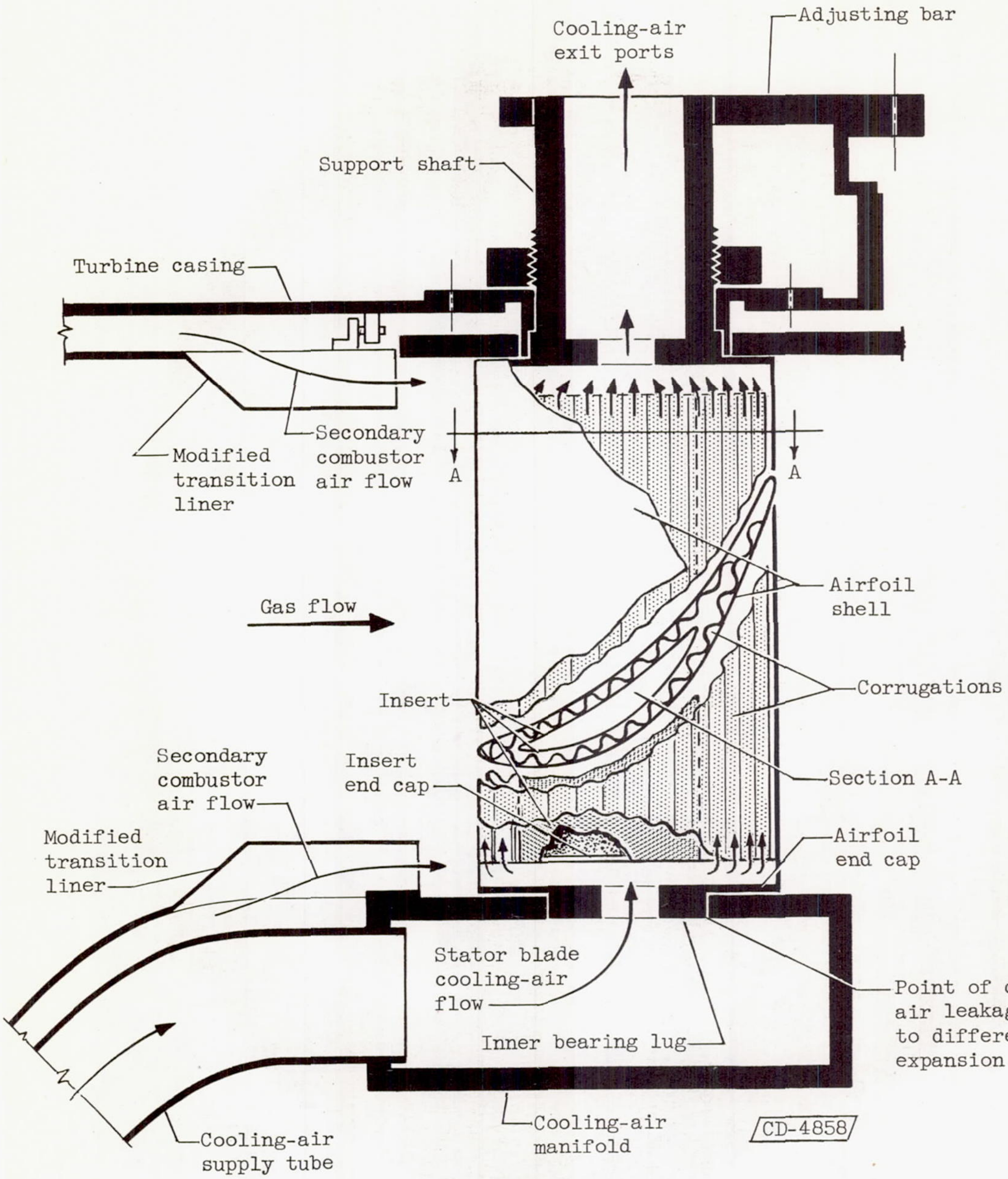


Figure 4. - Schematic view of adjustable air-cooled stator assembly.

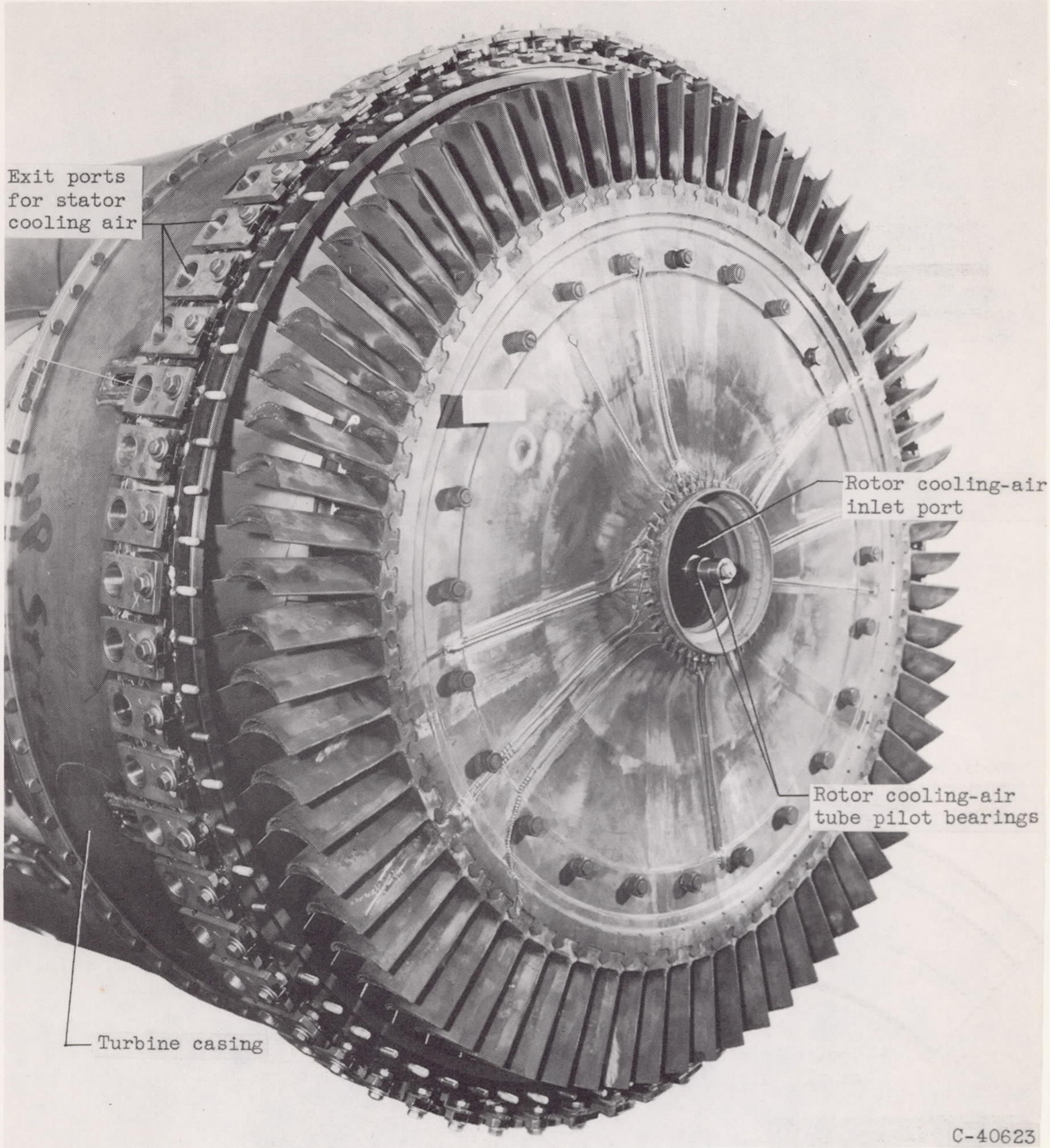


Figure 5. - View of air-cooled stator and rotor assemblies in test engine.

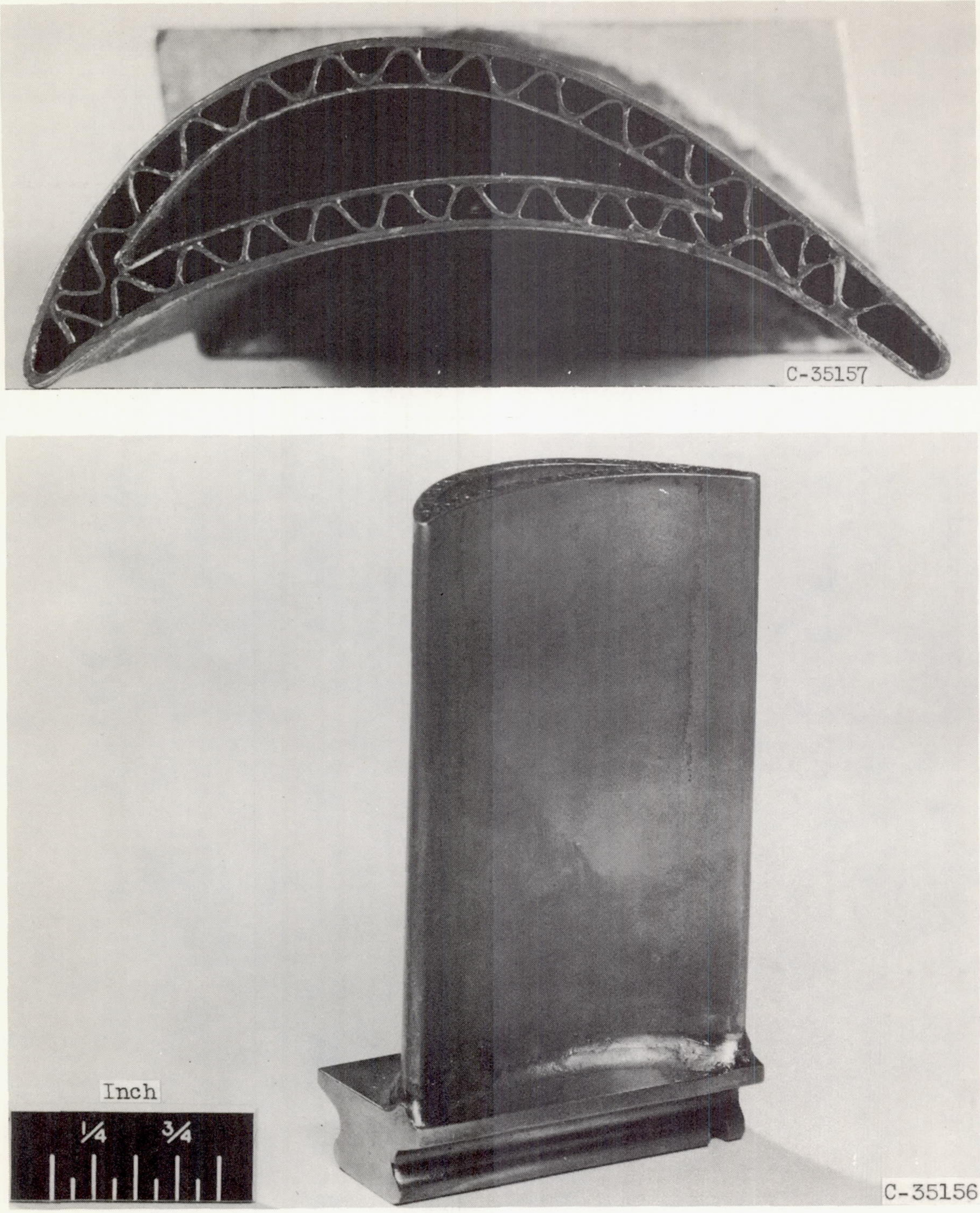
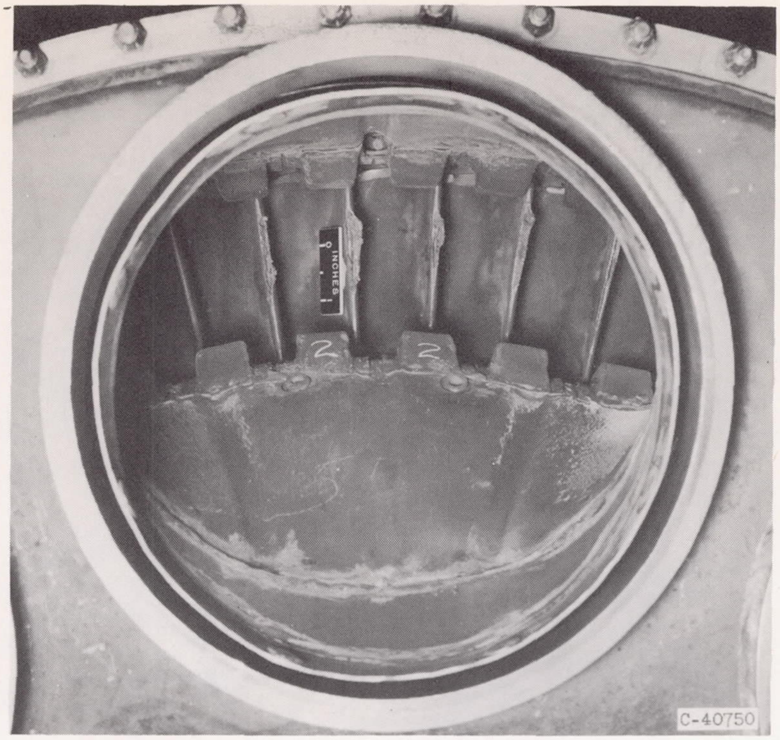


Figure 6. - Air-cooled corrugated-insert rotor blade.



(a) Most severely damaged sector.

(b) Least damaged sector.

Figure 7. - Views of leading edges of stator blades at conclusion of investigation.