Copy 1
RM SE55B07

NACA RM SE55B07

NACA

RESEARCH MEMORANDUM

for the

Bureau of Aeronautics, Department of the Navy

ALTITUDE-TEST-CHAMBER INVESTIGATION OF THE ENDURANCE AND PERFORMANCE CHARACTERISTICS OF THE J65-W-7 ENGINE AT A MACH NUMBER OF 2.0

By A. E. Biermann and Willis M. Braithwaite

Lewis Flight Propulsion Laboratory
Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON

FEB 1955

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

for the

Bureau of Aeronautics, Department of the Navy

ALTITUDE-TEST-CHAMBER INVESTIGATION OF THE ENDURANCE AND PERFORMANCE CHARACTERISTICS OF THE J65-W-7 ENGINE AT A MACH NUMBER OF 2.0

By A. E. Biermann and Willis M. Braithwaite

SUMMARY

An investigation of the endurance characteristics, at high Mach number, of the J65-W-7 engine was made in an altitude chamber at the Lewis laboratory. The investigation was made to determine whether this engine can be operated at flight conditions of Mach 2 at 35,000-feet altitude (inlet temperature, 250° F) as a limited-service-life engine.

Failure of the seventh-stage aluminum compressor blades occurred in both engines tested and was attributed to insufficient strength of the blade fastenings at the elevated temperatures.

For the conditions of these tests, the results showed that it is reasonable to expect 10 to 15 minutes of satisfactory engine operation before failure. The high temperatures and pressures imposed upon the compressor housing caused no permanent deformation.

In general, the performance of the engines tested was only slightly affected by the high ram conditions of this investigation. There was no discernible depreciation of performance with time prior to failure.

INTRODUCTION

At the request of the Bureau of Aeronautics, Department of the Navy, an exploratory investigation of the endurance characteristics, at high Mach number, of the J65-W-7 engine was made in an altitude chamber at the Lewis laboratory.

The investigation was undertaken to determine: (1) whether the J65 engine can be operated as a limited-service-life engine at a simulated flight condition of Mach 2 at 35,000-feet altitude for a period of approximately 30 minutes (This is a flight condition of interest for the Regulus II missile.); (2) the engine performance at these conditions; and (3) the estimated succession of engine component failures with increasing temperature and pressure.

Operation at the specified flight conditions of Mach 2 and 35,000-feet altitude subjects the engine to temperatures and pressures in excess of the design specifications as an extended-service-life engine. Differences in the thermal expansion of the various parts, the strength and fatigue properties of the aluminum-alloy compressor housing and the compressor blades at elevated temperatures combined with the high compressor discharge pressures involved are factors that may lead to unsatisfactory engine life at the specified flight conditions.

Study of the design limits of the J65 engine by the manufacturer prior to this investigation indicated that although bearings and compressor blading would be operated beyond normal limits there was probably a sufficient margin of strength to insure satisfactory completion of these tests. Of particular concern was the difference in thermal expansion of the nodular-iron center main bearing housing and the adjacent aluminum compressor case at elevated temperatures.

The tests were conducted in two phases. For the first phase, the severity of engine inlet temperature and pressure conditions was gradually increased toward a maximum value corresponding to a flight condition of Mach 2 and 35,000-feet altitude. Measurements of pertinent distortions, growth of parts, and critical temperatures were made in order to predict imminent failure. During the second phase of the tests, a second engine was operated at conditions simulating flight at Mach 2 and 35,000-feet altitude until failure occurred.

APPARATUS AND PROCEDURE

Engine

The J65-W-7 turbojet is a single-shaft engine having a 13-stage axial-flow compressor driven by a 2-stage reaction turbine. The compressor has a pressure ratio of 7.25 at rated conditions. The combustion chamber is of the annular type with vaporizing-type fuel injectors.

During this program both engines were provided with the same fixed exhaust nozzle having a diameter of 19.34 inches. The diameter of this nozzle was selected to give the engine of phase I a sea-level, limiting exhaust-gas temperature of 1185° F at 8300 rpm. The resulting thrust, corrected to static sea-level conditions, was approximately 8300 pounds.

The manufacturer's maximum specified inlet-air temperature is 200° F. The maximum specified temperature of the center and rear main bearings is 500° F.

Instrumentation

Instrumentation was provided for measuring temperatures and pressures at various stations throughout the engine as shown in figure 1. The instrumentation used in the engine of phase II was similar to that of phase I except that the instrumentation for the compressor discharge (station 3) was omitted.

Air flow to the engine was measured by means of a 29.9-inch-diameter ram pipe placed at the engine inlet. Measurements were also made of the temperature and weight flow of the cooling air that passes over the engine bearings and which is discharged overboard (midframe air bleed).

The variation of compressor housing diameter at station 2.8 was measured at four diametrical stations by means of eight linear potentiometers. These potentiometers were mounted on a tubular ring surrounding the engine. This ring was fixed to the engine at one point and was guided but was independent of radial engine movement at other points. During tests this tubular ring was maintained at constant temperature by the internal flow of compressed air around the ring. Linear potentiometers, one on each side of the engine at each diametrical station, as shown by the numbered lines in the diagram of figure 2, were connected differentially in such a manner as to give a direct indication of change in diameter independently of engine movement.

A linear potentiometer was also employed during the tests of phase II to measure the angular deflection or bending of the flanged joint between the aluminum compressor case and the iron center bearing housing. Bending of this flanged joint was anticipated because of the large difference in expansion of the aluminum and the iron housings. This measurement was made at radial station 2 of the diagram of figure 2.

Installation

The engine was mounted on a thrust-measuring platform in an altitude chamber as shown in figure 1. Inlet air was supplied to the engine from a plenum chamber formed by a bulkhead across the altitude chamber. Ram pressures were measured in this plenum chamber. The exhaust jet from the engine was directed into a diffuser that was so placed as to serve as an ejector for ventilating the engine compartment. This arrangement placed the engine compartment at tail-pipe exhaust-pressure conditions. A photograph of a J65 engine installed in the test chamber is shown in figure 3.

Procedure

The operating conditions for phase I and phase II, at progressively increasing severity and at maximum severity levels, respectively, are given in tables I and II. The engine was operated at the maximum rated speed of 8300 rpm and with a maximum exhaust-gas temperature of 1185° F. Because of choking in the exhaust-system diffuser of this setup, it was not always possible to obtain specified altitude pressures in the engine compartment. This deviation from specified pressures had no effect on engine performance because of the choked engine exhaust nozzle.

Because of instrument limitations at ram temperatures, the temperature of the engine compartment was maintained at approximately 100° F.

The engine control system was modified for these tests to make the compressor outlet-pressure fuel-limit mechanism of the power control inoperative.

Heated ram air was supplied from a direct-fired combustion-type heater.

The fuel used throughout the investigation conformed to the specifications for MIL-F-5624E, grade JP-4, and had a lower heating value of 18,700 Btu per pound and a hydrogen-carbon ratio of 0.171.

RESULTS AND DISCUSSION

Engine Endurance Characteristics

Phase I tests. - Table I presents a time-history of the phase I tests in which the severity of the inlet temperature and pressure conditions were gradually increased. These tests were terminated by failure of the seventh-stage rotor blading (last stage of aluminum blading). Figure 4 indicates the character of this failure. Failure was accompanied by a slight change in the pitch of the engine noise. The engine continued to rotate smoothly after failure. The compressor housing was dented by the failure, but was still intact.

The rotor blades of compressor stages 4, 5, 6, and 7 are constructed of aluminum alloy; the blades of the other stages are of steel. The blades of each stage are fitted between two steel rotor disks. The rotor-blade mountings consist of rectangular tangs which are clamped between steel mounting disks and are secured by means of two rivits.

Examination of the engine after failure indicated that the tang portion of some of the aluminum-alloy compressor blades of the seventh stage failed in tension. The greatly reduced margin of strength of the blade materials at the elevated temperatures involved lends support to this supposition. At the operating air temperature of 480° F and this stage of the compressor, the blade material has a basic yield strength of only 20,000 pounds per square inch, while a simplified stress calculation of the blade tang indicates an operating load of approximately 15,000 pounds per square inch.

The turbines were found to be in satisfactory condition at the termination of the tests.

Phase II tests. - Phase II tests, in which the engine was operated at the most severe inlet temperature and pressure conditions of the program, were terminated by failure of the sixth- and seventh-stage compressor rotor blading (aluminum alloy) and failure of the rotor blading in both turbine stages. Table II presents the history of the operation prior to failure. Figures 5(a) and 5(b) show the character of the compressor failure, and figure 5(c) shows the final condition of the blading of the two turbines. Failure of this engine was accompanied by a thump and a change in the pitch of the engine. The compressor housing was dented, but was not ruptured.

Examination of this engine indicated that rupture of a compressor-blade tang was possibly the initial cause of failure. The turbine blading failed from excessive temperatures. Sudden reduction of air flow through the engine following the compressor failure could have caused over-rich fuel-air mixtures and attendant excessive temperatures. The fact that the turbines of the phase I engine remained intact, whereas the turbines of the engine of phase II were severely damaged was possibly caused by the more complete failure of the compressor in the latter tests. Only the seventh-stage blades failed in phase I tests, whereas both sixth and seventh stages failed in phase II tests.

These tests indicate that it is reasonable to expect 10 to 15 minutes of satisfactory operation at an inlet-air temperature of 250° F before failure. Operation at this condition for periods of 20 minutes or longer is considered extremely marginal.

Effect of Ram Conditions upon Deformation of Compressor Case

Initial studies by the manufacturer of the J65 engine to determine whether the engine can be used as a limited-service-life engine for the

Regulus missile indicated that the discharge end of the compressor housing and the center main bearing support housing are regions of marginal strength because of the high shell temperatures, the high pressures, and the thermal distortions expected.

The compressor housing is of aluminum alloy and is bolted to a nodular-iron center main bearing support housing, which forms a diffuser section. Ten radial struts pass through the diffuser section. These struts are lettered as shown in figure 2. Struts B, C, E, G, I, and J are used as radial ducts for carrying cooling air from the fifth compressor stage to the main bearings.

The differential thermal expansion between the discharge end of the aluminum-alloy compressor case and the iron center main bearing housing is such as to possibly overstress these parts in the region of attachment. Accordingly, measurements were made of changes in compression case dimensions just ahead of the rear compressor flange (station 2.8). This station is possibly critical because the much greater expansion of the aluminum compressor flange over that of the iron center main bearing housing flange would tend to deflect the outer edge of these flanges toward the rear of the engine and in so doing would tend to overstress the compressor housing in the region of station 2.8. Changes in case diameter were plotted against time continuously during tests. In such plots a change in the slope of the curve may indicate stresses beyond the elastic limits and failure can be predicted before it occurs. Check measurements were also made before and after tests in order to detect permanent deformations. These measurements were made with calipers especially constructed for the purpose.

The deflection of the bolting flange, which fastens the aluminum compressor housing to the iron center main bearing housing, was negligible. There were no indications of permanent deformation of any of the engine parts measured up to the point of failure.

The effect of ram conditions upon the increase in compressor case diameter at station 2.8 is shown in figure 2. From this figure it will be noted that for the engine of phase I the increases in case diameter was more than three times as much for positions 1 and 3 as for positions 2 and 4. For the engine of phase II this difference was much less.

The temperatures of the compressor discharge air and the compressor case at station 2.8 are shown in figure 6. The temperature of the compressor case is approximately 80° F lower than the temperature of the compressor discharge air. The temperature of both the compressor case and the discharge air rise somewhat more rapidly than the inlet total temperature.

During these tests the ambient cooling air surrounding the engine was maintained at approximately 100° F to protect instrumentation and engine accessories. If the ambient cooling air had corresponded to a ram temperature of 250° F, the compressor case would only have been increased by approximately 20° F.

Effect of Ram Conditions upon Oil and Bearing Temperatures

The center main bearing of the J65 engine is cooled by leakage air from the rear compressor seal. The rear main bearing is cooled predominately by air from the fifth compressor stage. The lubricating oil to these bearings is metered. The oil and cooling air from the bearings is then dumped overboard. The front main bearing (thrust bearing) is both cooled and lubricated by a circulating oil system.

Inasmuch as the center and rear main bearings are predominately cooled by air, the bearing temperatures would be expected to follow engine-inlet-air temperature changes. Figure 7 shows a linear response of the temperature of the bearings and of the midframe bleed air with change of inlet-air temperature. It will be noted that although the temperatures of these bearings are somewhat in excess of the specified limit of 500° F, the life of such bearings is probably ample for short periods even though temperatures are above normal limits.

Inasmuch as the thrust bearing is cooled by oil, its temperature would be expected to follow oil temperature changes. Examination of the data shows that, in general, the temperature of the thrust bearing was approximately 60° F above the oil temperature. Because of the thermal capacity of the oil system and associated parts, the temperature of the thrust bearing was much slower to respond to air temperature changes than were the air-cooled bearings. Because of this thermal lag and the fact that these tests were started with various initial oil temperatures, attempts at showing the effect of inlet-air temperature upon thrust bearing and oil temperatures are inconclusive.

Time Required to Reach Constant Temperatures

From the standpoint of engine endurance, some significance may be attached to whether engine temperatures were still rising at the time of engine failure. A marginal condition is indicated if the temperature of temperature-critical areas leveled out before failure, whereas, a more critical problem exists if temperatures were still rising at the time of failure.

Data from these tests show that the temperature of the discharge portion of the compressor case and that of the two air-cooled main bearings approached a constant level very shortly after a change in inlet-air temperature. As shown in figure 8(a), the midframe air bleed temperatures tended to level off after a period of from 5 to 15 minutes of operation. The midframe air bleed temperature has been plotted here because it is somewhat indicative of the temperature of the interior portions of the engine.

Figure 8(b) shows the relation of midframe air bleed flow with time. The change of flow with time probably reflects the change of labyrinth clearances with temperature.

Engine Performance

Only sufficient data are presented to indicate the general engine performance. The engine air flow, temperature, and pressure ratios are shown in figures 9, 10, and 11, respectively. The specific fuel consumption and net thrust are shown in figure 12. The engine air flow is corrected to compressor inlet conditions. The nominal altitude is, however, based on the total pressure for a flight Mach number of 2.0 with an assumed total-pressure recovery of 80 percent. The net thrust and specific fuel consumption data have been corrected to free-stream, total-pressure flight conditions.

Comparative engine performance data at an inlet temperature of 70° F are available for the phase I engine only. In general, the corrected performance of this engine was only slightly affected by the high ram conditions of the investigation. As shown by figure 12, average values of the net thrust and specific fuel consumption for a given exhaust-gas temperature at inlet-air temperatures from 200° to 250° F are within 2 and 5 percent, respectively, of that obtained at standard inlet conditions. The reasons for the 10 percent higher net thrust of the engine of phase II is not known; however, the thrust data are consistent with the high air flow (fig. 9) and the somewhat higher pressure ratio (fig. 10) of this engine.

CONCLUDING REMARKS

This investigation of the endurance and performance of two J65-W-7 engines at a simulated flight Mach number of 2.0 (inlet-air temperature, 250° F) in an altitude test chamber was terminated by failure of the seventh-stage aluminum compressor blades of both engines. This failure is attributed to insufficient strength of the blade mounting tang.

Deflection of the aluminum compressor flange, which is bolted to an iron center main bearing section, was negligible and there was no permanent deformation of the compressor housing. The temperature of the turbine bearing, which is predominately air cooled, reached 612° F shortly after starting operation with an engine inlet-air temperature of 250° F. Temperatures of the thrust bearing and lubricating oil were too greatly affected by the engine installation to be conclusive.

Corrected engine performance was only slightly affected by the high ram conditions.

Based on the conditions and results of this investigation, satisfactory engine operation, at an engine inlet temperature of 250° F, can be reasonably expected for a period of 10 to 15 minutes. Continued operation for periods of 20 minutes or longer is considered extremely marginal.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, February 9, 1955

TABLE I. - OPERATING CONDITIONS FOR PHASE I TESTS AT

PROGRESSIVELY INCREASING SEVERITY

Nominal altitude, ft (a)	Inlet total temperature, °F	Inlet total pressure, lb/sq ft abs	Engine compartment static pressure, lb/sq ft abs (a)	Duration, min
Time at elevated inlet temperature required to set conditions				10
55,000	200	1200	280	13
Time required to set condition				6
45,000	200	1950	414	10
Time required to set condition				9
35,000	200	3115	950	(b)
Instrument failure prevented completion of this point				
Time required to set condition				12
55,000	225	1200	330	10
Time required to change inlet temperature				5
55,000	250	1200	330	12
Engine failed				
				Total time 87

[a]The nominal altitude is based on the total pressure for a flight Mach number of 2.0 with an assumed total-pressure recovery of 80 percent. Because of choking in the facility, the exhaust ambient static pressure was higher than the NACA standard atmospheric value. The engine compartment pressure was approximately the same as that at the engine exhaust.

[b]Engine was shut down at this point to repair instrumentation.

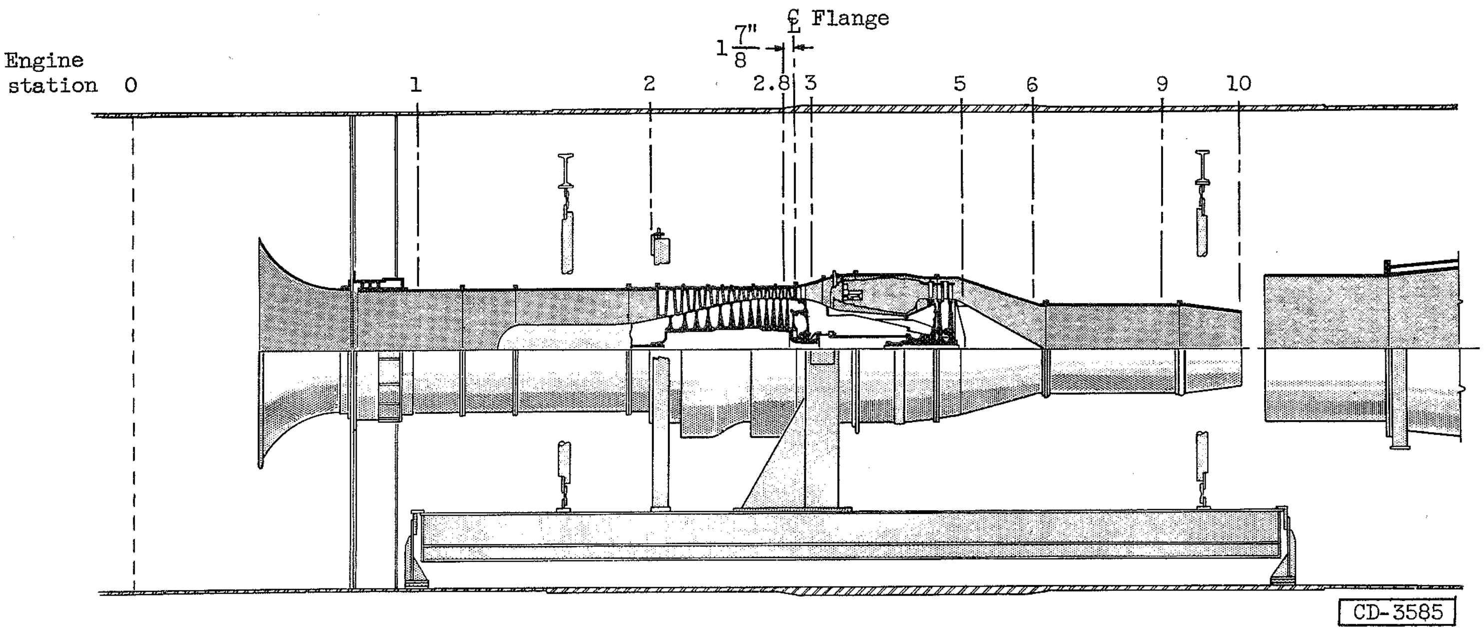
TABLE II. - OPERATING CONDITIONS FOR PHASE II TESTS AT

MAXIMUM SEVERITY LEVEL

Nominal altitude, ft (a)	Inlet total temperature, °F	Inlet total pressure, lb/sq ft abs	Engine compartment static pressure, lb/sq ft abs (a)	Duration, min
Time at elevated inlet temperature required to set flight condition				2
35,000	250	3100	950	20
Faulty operation of inlet-air preheater resulted in the following unscheduled mode of operation for the remaining time:				
35,000	100	---- (b)	---	8
35,000	175	---- (b)	---	3
35,000	100	---- (b)	---	3
35,000	175	---- (b)	---	2
35,000	200	---- (b)	---	3
35,000	250	3500 approx.	---	2
Engine failed				
				Total time 43

[a]The nominal altitude is based on the total pressure for a flight Mach number of 2.0 with an assumed total-pressure recovery of 80 percent. Because of choking in the facility, the exhaust ambient static pressure was higher than the NACA standard atmospheric value. The engine compartment pressure was approximately the same as that at the engine exhaust.

[b]During this time interval, the inlet pressures varied from that corresponding to a flight condition of Mach 2 at 35,000 feet (3100 psfa) to that for a flight condition of Mach 1.5 at 35,000 feet (1850 psfa). The engine speed was also reduced to approximately 7000 rpm.



Station	Total-pressure tubes	Wall static orifices	Static-pressure tubes	Thermo-couples (surface)	Thermo-couples (total temperatures)	Linear potentio-meters
0					4	
1		4				
2.8				7		8
3	12	3			10	
5	15				18	
9	35	5		4	30	
10			4	4		
Midframe bleed tube	2		1	1	2	

Figure 1. - Schematic diagram of J65-W-7 engine installation showing instrumentation stations.

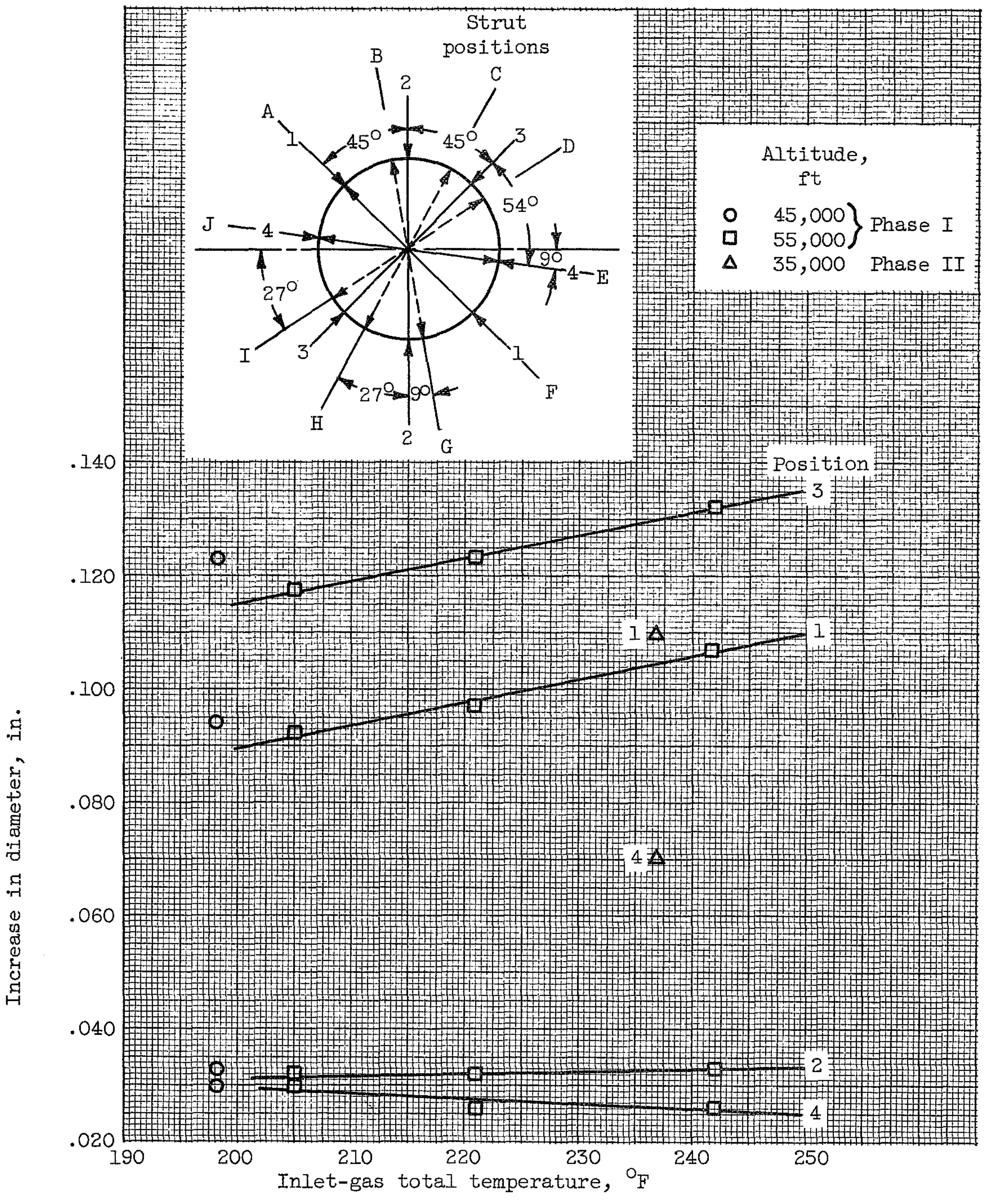


Figure 2. - Effect of ram conditions on increase in diameter of compressor case with reference to the diameter at static ambient conditions. Flight Mach number, 2.0. Data taken after 10-minutes operation at each test condition. Diagram view is looking downstream. Letters refer to positions of struts in center main bearing support housing. Struts B, C, E, G, I, and J carry cooling air to the center and rear main bearings.

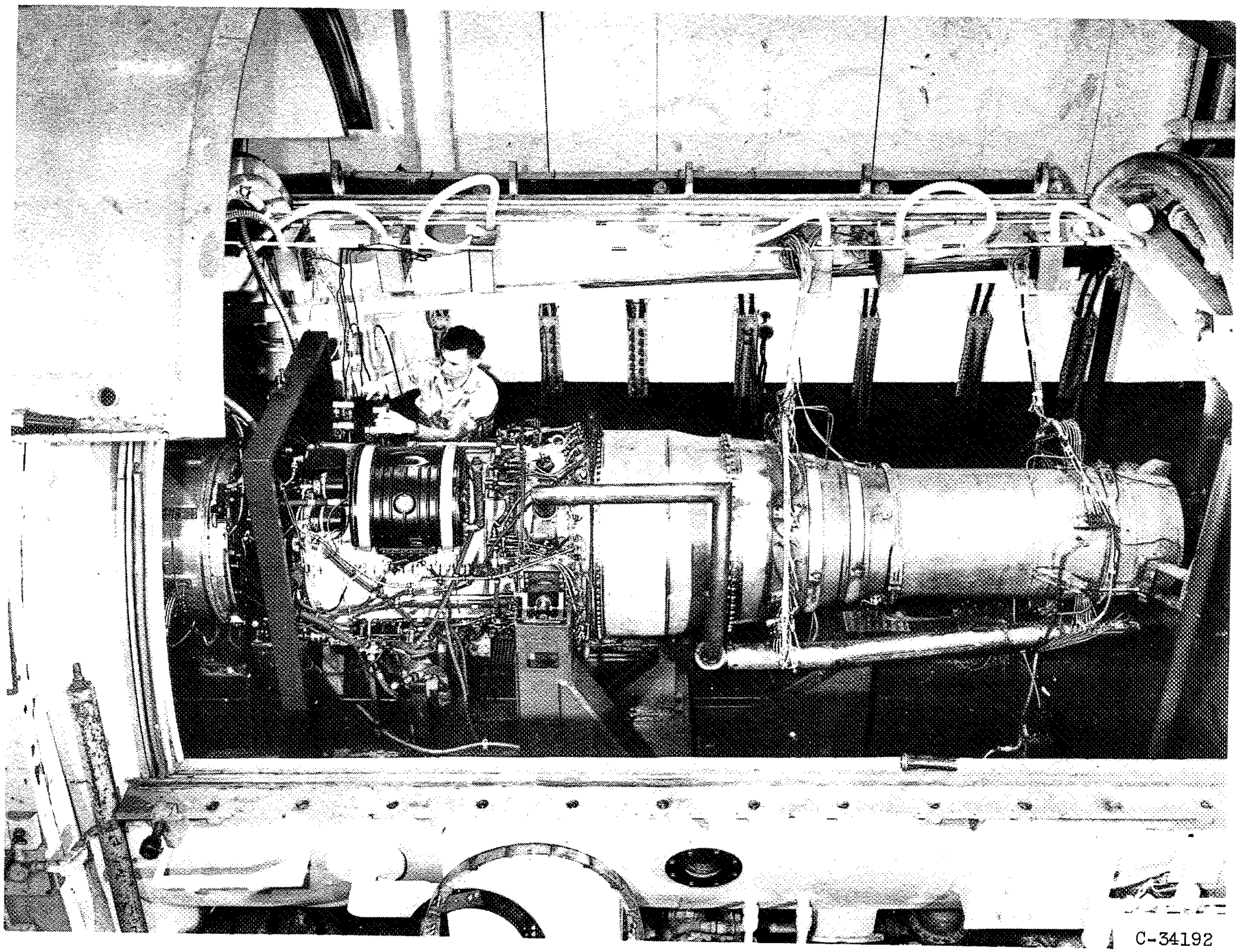
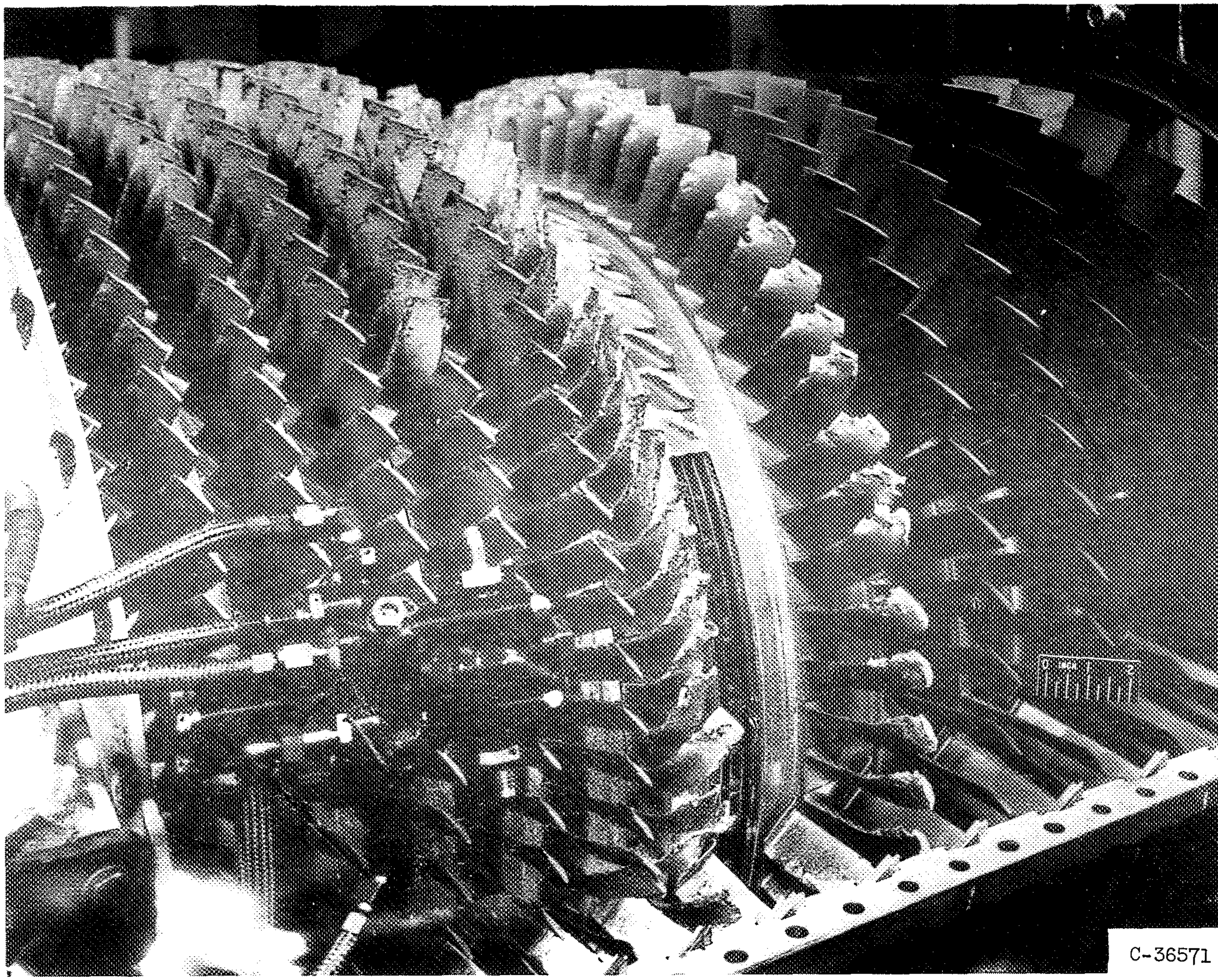
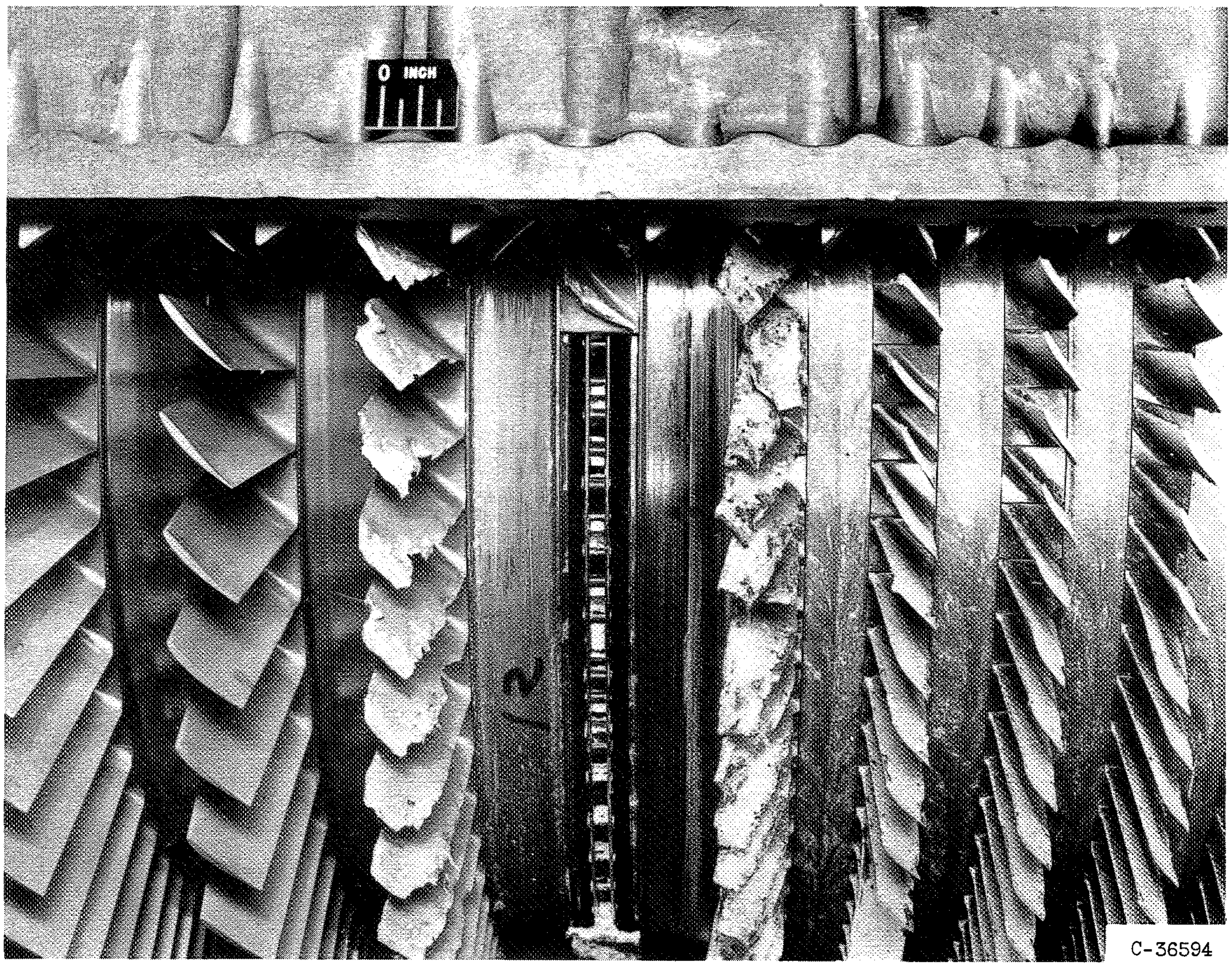


Figure 3. - J65-W-7 engine installed in altitude test chamber.



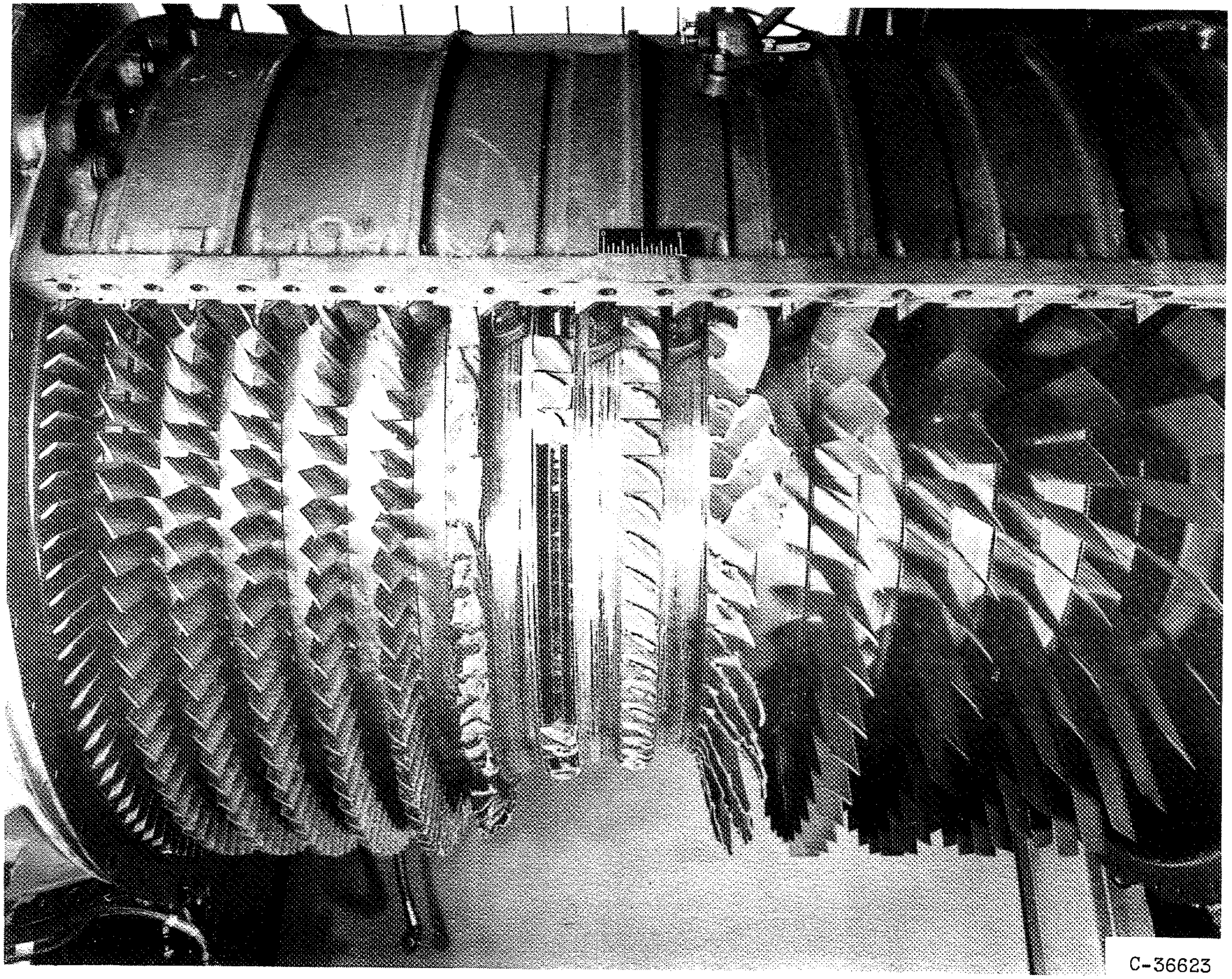
(a) Top half of compressor rotor.

Figure 4. - Compressor rotor of phase I engine showing seventh-stage blade failure.



(b) Bottom half of compressor rotor.

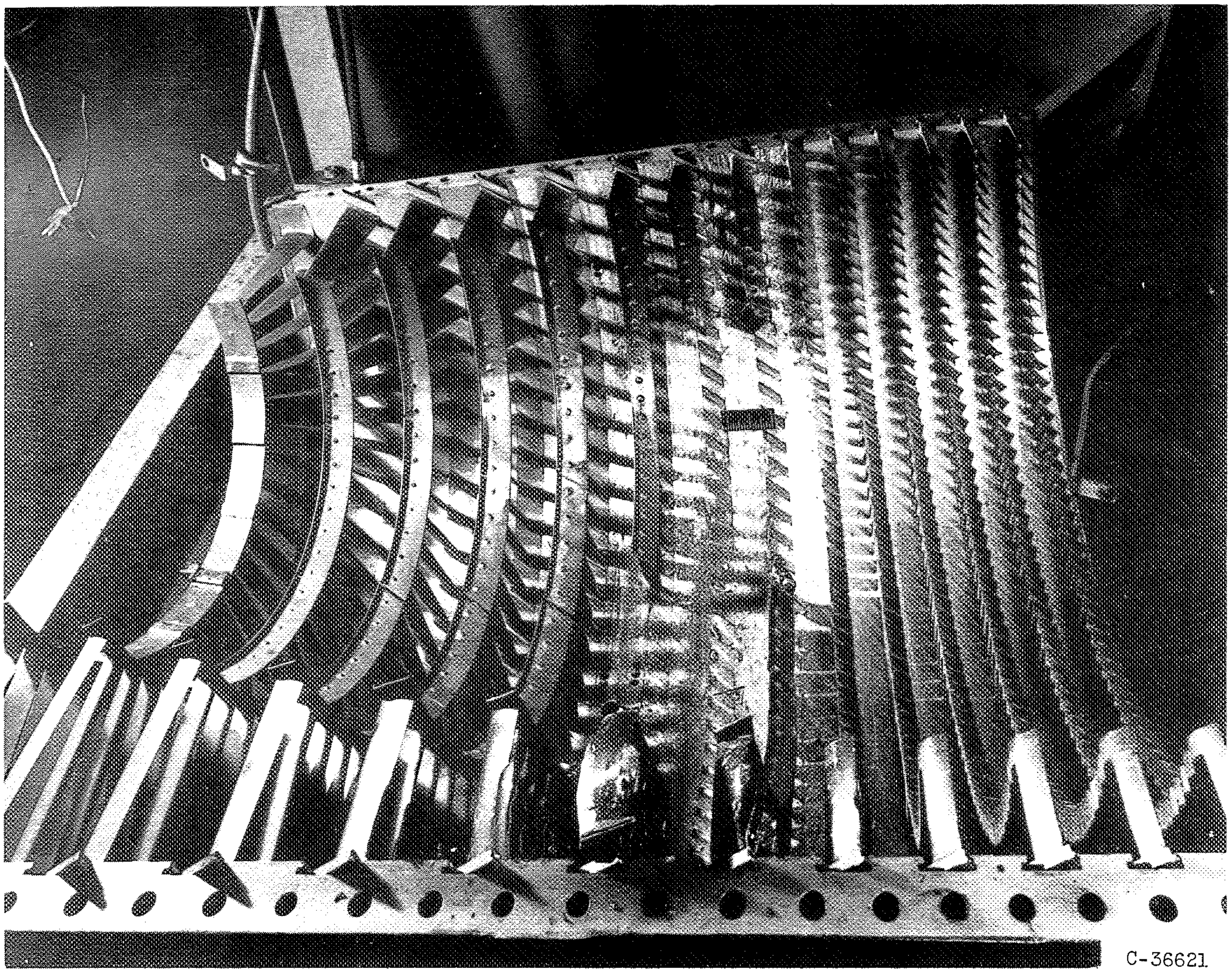
Figure 4. - Concluded. Compressor rotor of phase I engine showing seventh-stage blade failure.



C-36623

(a) Bottom half of compressor rotor showing failure of sixth- and seventh-stage blading.

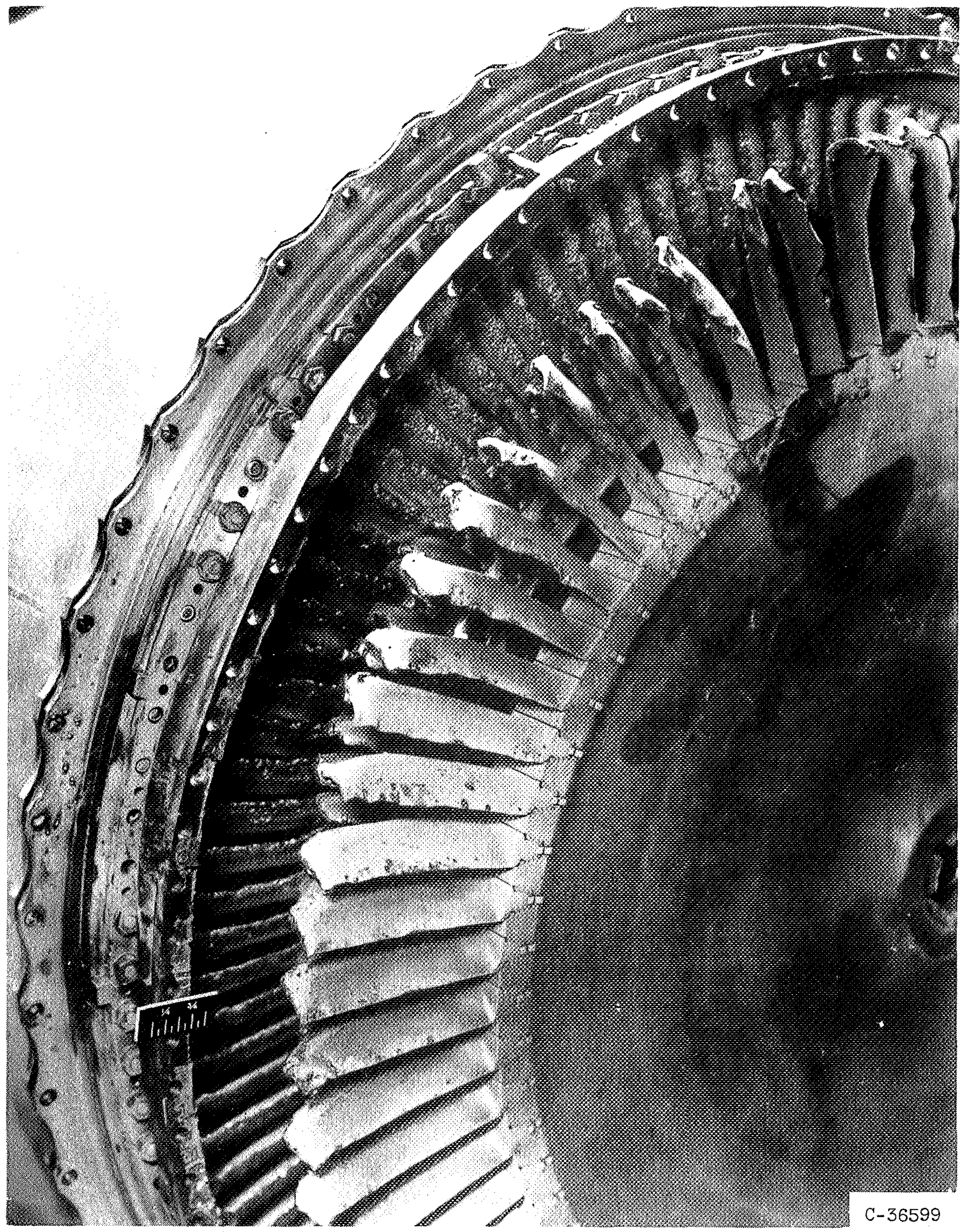
Figure 5. - Failed components of phase II engine.



C-36621

(b) Bottom half of compressor case showing condition of stator blading.

Figure 5. - Continued. Failed components of phase II engine.



(c) First-stage turbine-nozzle diaphragm and first- and second-stage turbine rotor blading.

Figure 5. - Concluded. Failed components of phase II engine.

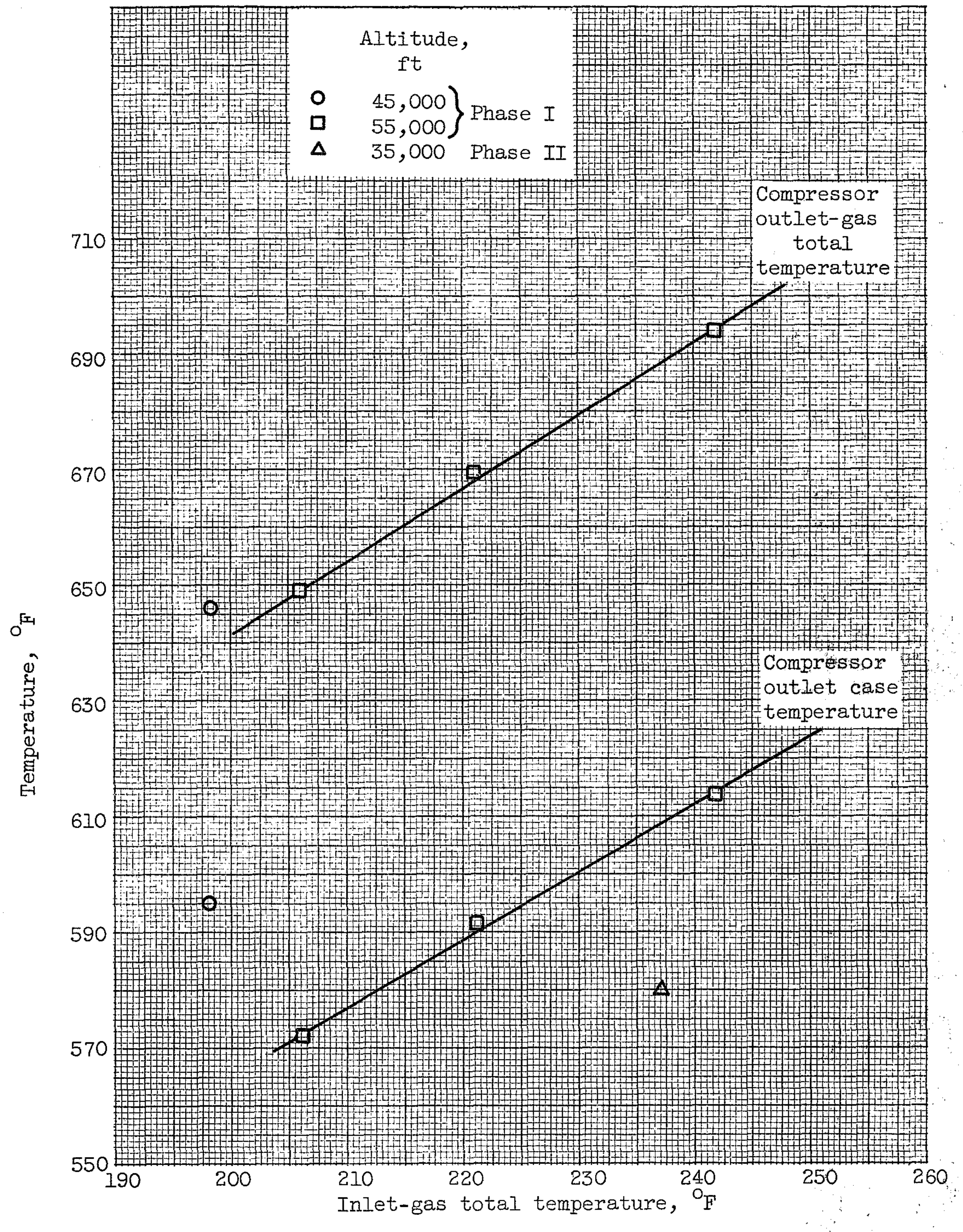


Figure 6. - Effect of ram conditions on compressor discharge and compressor case temperature. Flight Mach number, 2.0. Data taken after 10 minutes operation at each test point.

CONFIDENTIAL

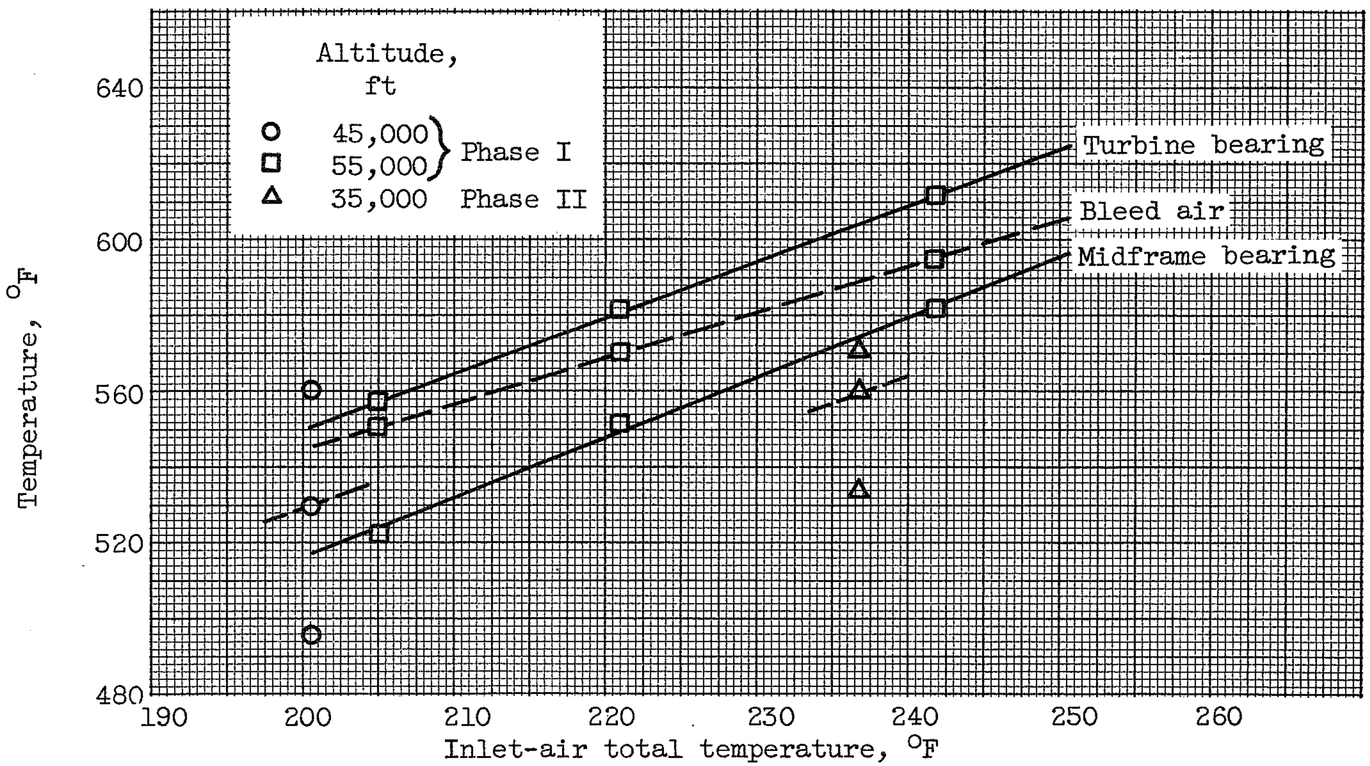


Figure 7. - Effect of ram conditions on temperatures of midframe bleed air and midframe and turbine bearings. Flight Mach number, 2.0.

CONFIDENTIAL

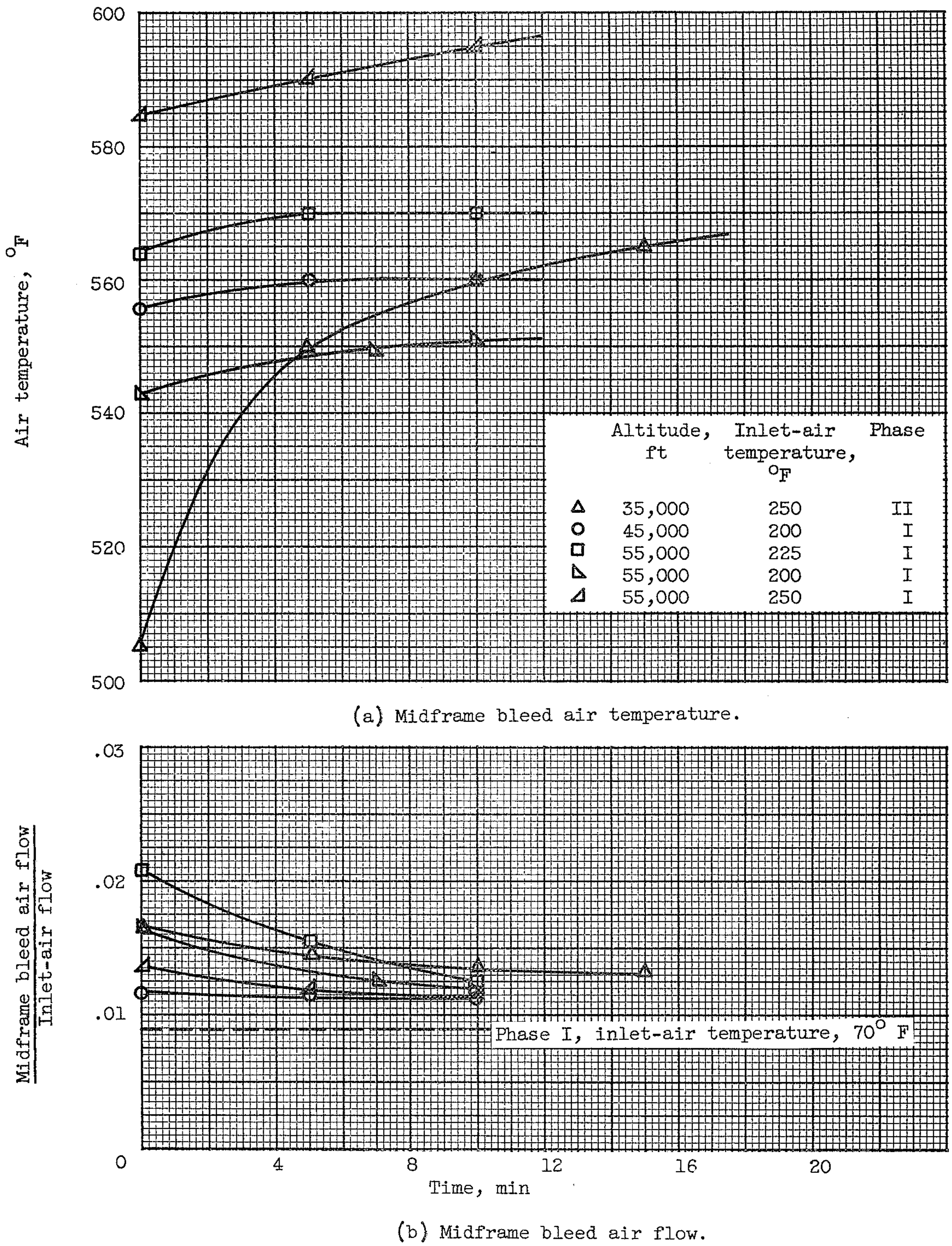


Figure 8. - Variation of midframe bleed air flow and air temperature with time. Flight Mach number, 2.0.

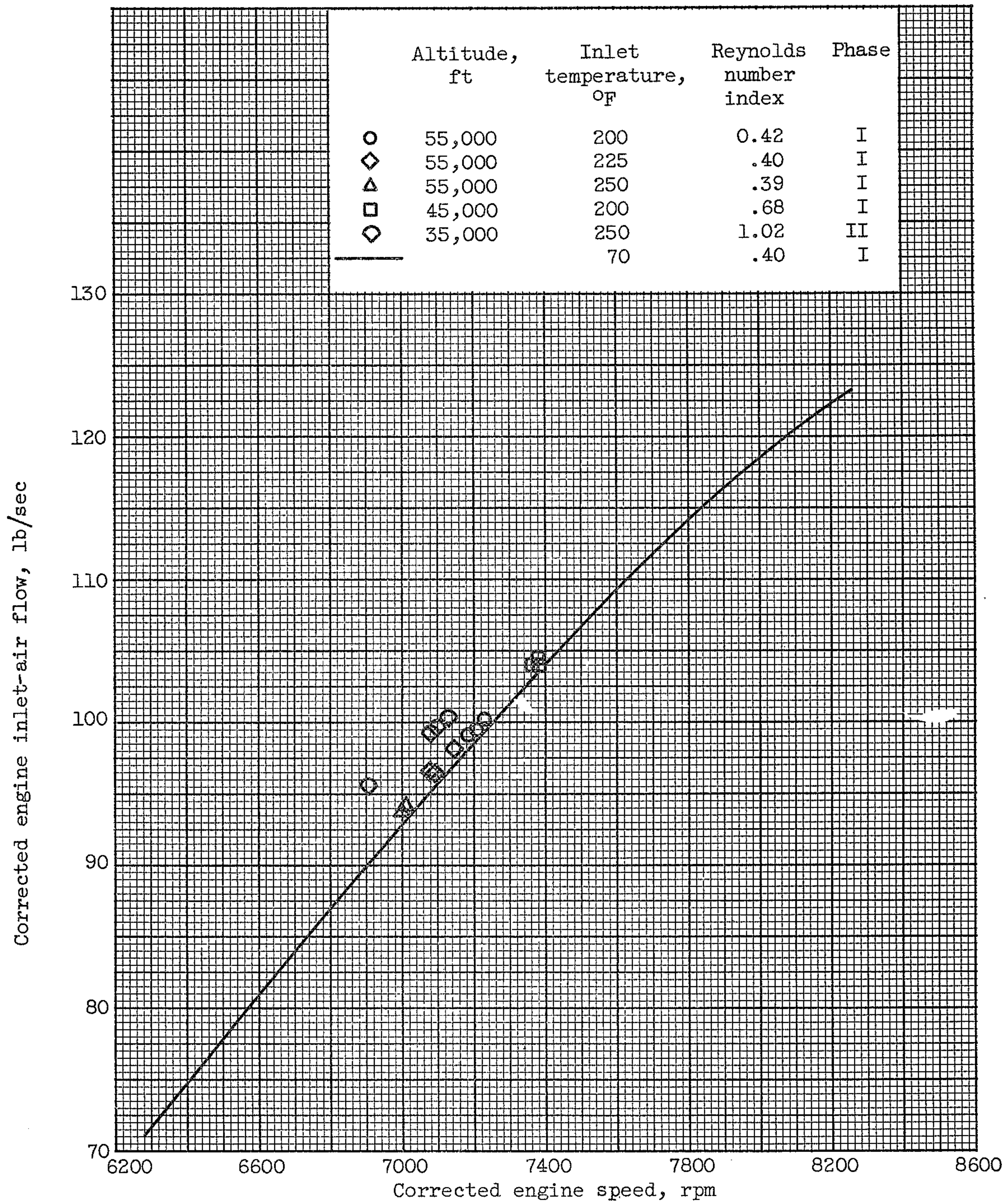


Figure 9. - Effect of operating conditions on corrected engine inlet-air flow. Flight Mach number, 2.0.

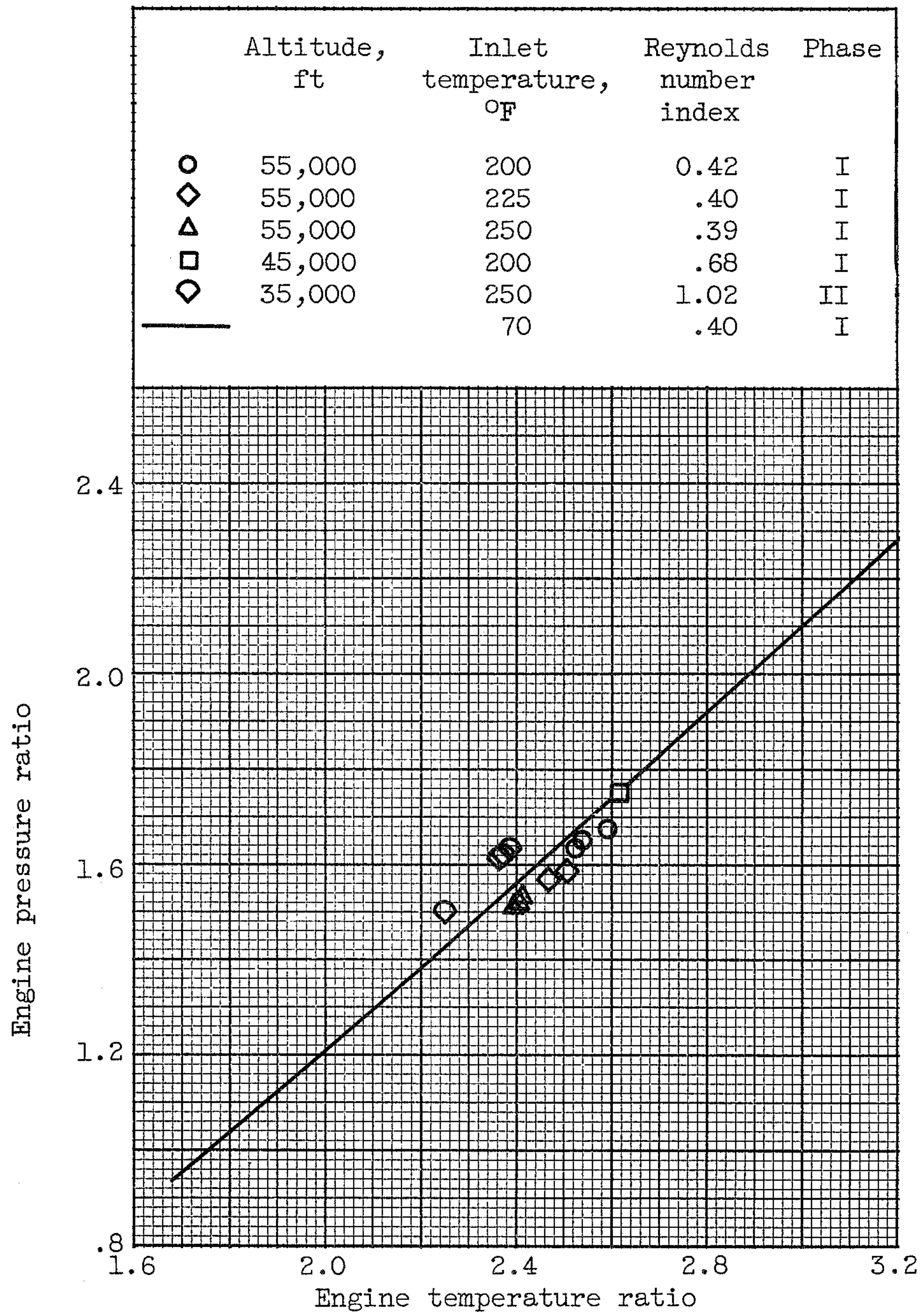


Figure 10. - Effect of operating conditions on the relation between engine temperature ratio and engine pressure ratio. Flight Mach number, 2.0.

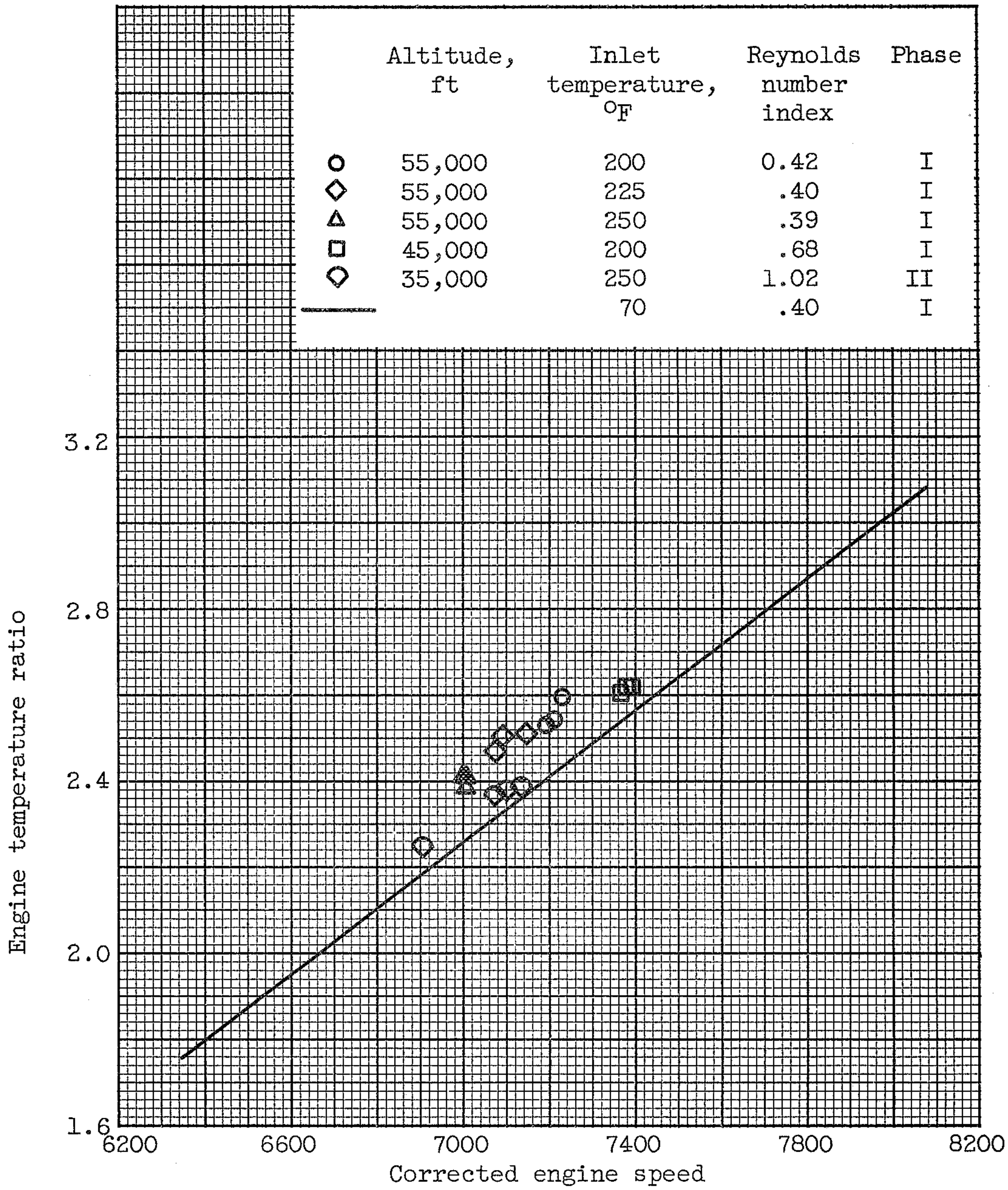


Figure 11. - Effect of operating conditions on engine temperature ratio. Flight Mach number, 2.0.

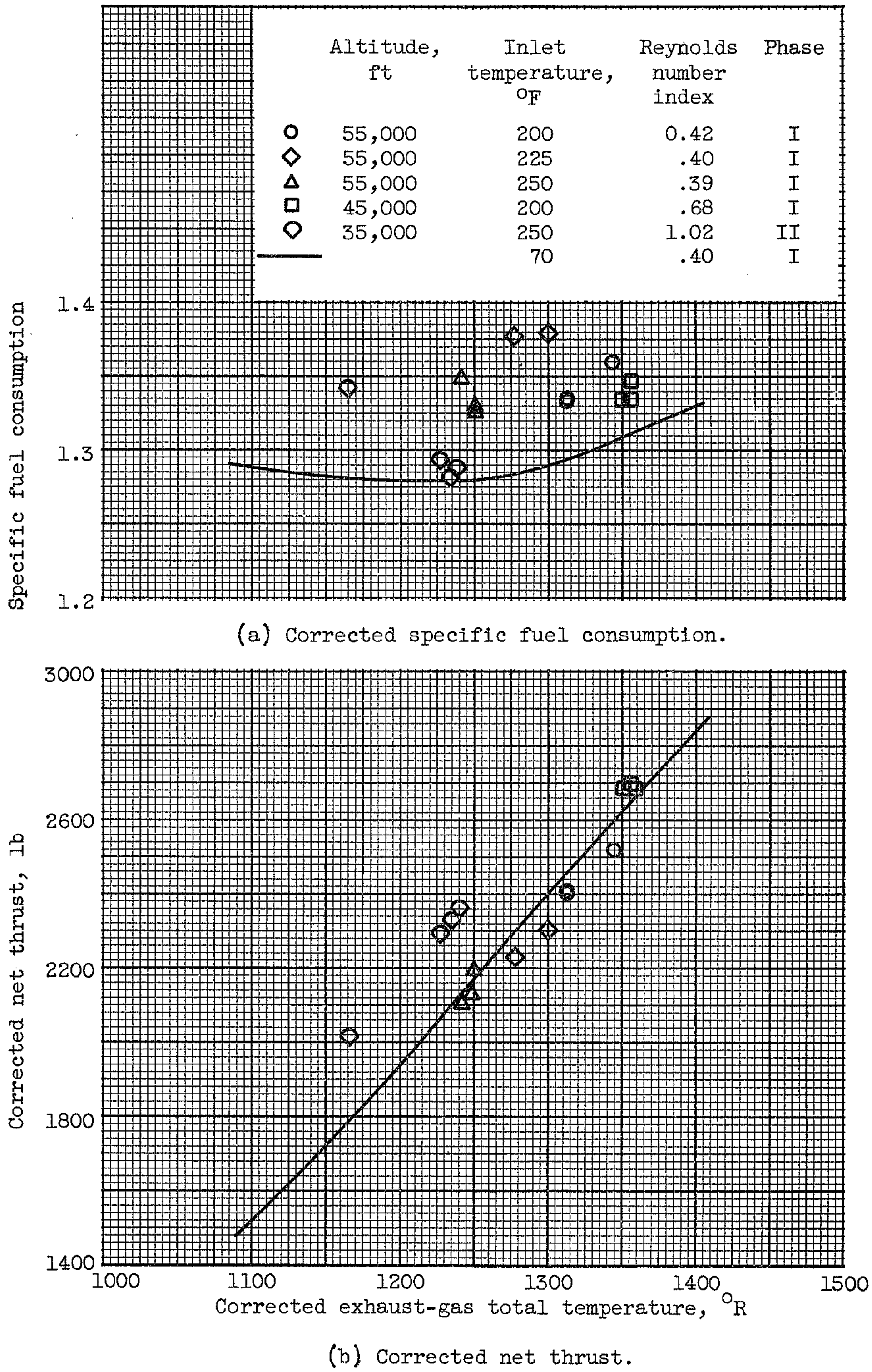


Figure 12. - Effect of operating conditions on corrected net thrust and net thrust specific fuel consumption. Flight Mach number, 2.0.

ALTITUDE-TEST-CHAMBER INVESTIGATION OF THE ENDURANCE AND
PERFORMANCE CHARACTERISTICS OF THE J65-W-7 ENGINE
AT A MACH NUMBER OF 2.0

A. E. Biermann

A. E. Biermann
Aeronautical Research Scientist
Propulsion Systems

Willis M. Braithwaite

Willis M. Braithwaite
Aeronautical Research Scientist
Propulsion Systems

Approved:

Bruce T. Lundin

Bruce T. Lundin
Chief
Engine Research Division

maa - 2/9/55