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RESEARCH MEMORANDUM

A STUDY OF TEMPERATURE TRANSIENTS AT THE INLET OF A TURBOJET ENGINE

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Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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A STUDY OF TEMPERATURE TRANSIENTS AT THE INLET OF A TURBOJET ENGINE

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SUMMARY

In an effort to learn why inlet temperature transients disturb the operation of turbojet engines, extensive wind-tunnel experiments were conducted on the effect of rapid temperature increases at the inlet of an engine. The study was made over a range of altitudes, temperature-rise rates, and engine speeds. The engine was instrumented with high-response instrumentation to evaluate the compressor performance during the temperature changes.

At all the altitudes investigated, compressor stall was induced by rapid compressor-inlet temperature rises between 50° and 250° R. The stall originated in the stage group having the smallest stall margin, in this case the 12th and 13th stages, and then propagated through the compressor. Stall is induced because the compressor is not able to adjust to the rapidly changing flow conditions resulting from the rapid temperature transients. Study of the flow mechanism within the compressor during these temperature transients of several thousand degrees per second indicated that, in general, high-pressure-ratio axial-flow compressors will be inherently as sensitive to inlet temperature transients as was the compressor of this engine. This was borne out by some tests run on another engine of a different manufacturer; this alternate engine was also stalled by correspondingly small inlet temperature rises.

Compressor stall was induced by consistently smaller temperature transients as altitude was increased. This altitude effect becomes particularly important, because at high altitudes combustion was extinguished by pressure pulsations from the stalled compressor. For example, when the inlet pressure of this engine was only 190 pounds per square foot, combustion blowout occurred each time the compressor stalled. In contrast, at engine-inlet pressures higher than about 500 pounds per square foot, combustion was sufficiently stable to withstand

the pressure surges; and the engine recovered from stall as soon as the inlet temperature transient was stopped. Inlet pressures below which blowout can be anticipated will be in this range, although they will vary to some degree among engines depending upon the relative stability of the combustor and the level of the compressor pressure ratio.

Sudden fuel-flow reductions immediately prior to temperature tran-
sients permitted somewhat larger temperature rises to be imposed before
stall occurred. In fact, this was the only remedial action studied that
afforded a measurable improvement. However, a fuel-flow reduction of 4441
40 percent prior to the inlet temperature transient permitted a tempera-
ture rise of only 220° R at a temperature-rise rate of 10,000° R per
second.

INTRODUCTION

With the advent of rocket and cannon armament on airplanes, gas ingestion into the inlet of turbojet engines has caused compressor surge and combustion blowout. Both of these increase in severity at high altitudes, where the stall margin is normally reduced and the combustion process becomes marginal (see refs. 1 to 4). In seeking a solution for the trouble, there are three possible changes that can be made to the engine: (1) move the armament to a position where it will not interfere with the engine; however, this is generally not practical in prototype or production aircraft; (2) modify the armament by methods such as using chemical additives in the rocket charges (ref. 5) or adding exhaust deflectors (ref. 3); (3) modify the engine or its mode of operation to make it less sensitive to the effects of armament firing. An operating technique that was found useful in one instance was the reduction in engine fuel flow prior to armament firing (ref. 1), thus lowering the compressor operating pressures and increasing the stall margin. Although each of these changes has improved particular aircraft operation, none appears to be a general solution to the problem.

The search for a way to eliminate surge and blowout has proceeded with much diligence but with little understanding of the basic phenomena involved during stall or blowout resulting from inlet temperature transients. In reference 6 it is pointed out that the firing of armament has four general effects on the air entering the engine: (1) rapidly increases the engine-inlet temperature, (2) causes pressure pulsations at the inlet, (3) contaminates the air supply, and (4) distorts the temperature and pressure profiles. It was believed that the inlet temperature transients associated with armament firing was one of the principal problems. Therefore, the investigation reported herein was designed and conducted to furnish information on the flow phenomena within the compressor following rapid inlet temperature rises. This information should, it was felt, provide for a more sound and fundamental approach in future work directed toward eliminating the problem.

The two requisites for proper definition of temperature transients are rate and magnitude of temperature change. The temperature rates covered in this study were between 2000° and 15,000° R per second (hereinafter called degrees per second) in order to obtain inlet temperature rises from 50° to 250° R. Although most of the work was carried out at an engine-inlet pressure of 242 pounds per square foot (corresponding to a Mach number of 0.8 at an altitude of 59,000 ft), the effect of pressure changes from 190 pounds per square foot to 498 pounds per square foot on temperature transients was investigated. Some effort was expended to learn the effectiveness of simple changes, such as reducing fuel flow or bleeding off compressor air, in alleviating the problems arising from inlet temperature transients.

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APPARATUS

Engine

The turbojet engine used in this investigation was in the 10,000-pound-thrust class with a compressor pressure ratio of about 7.5 in 15 stages. It had a cannular-type combustor, a two-stage turbine, and a fixed-area exhaust nozzle. The compressor was equipped with interstage air bleed valves and variable-position inlet guide vanes, both of which were actuated at about three-fourths of rated engine speed. The engine fuel control varied the fuel flow depending on engine-inlet total pressure, compressor-discharge total pressure, and mechanical speed. The engine installation in the altitude wind tunnel of the NACA Lewis laboratory is pictured in figure 1.

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Transient Temperature Device

Transient increases in engine-inlet temperature were introduced by means of a combustion gas heater and quick-acting valve combination that were placed upstream of the engine. A schematic diagram of this equipment is shown in figure 2. Large butterfly valves before and after the heater were used to control the pressure and velocity in the heater to permit ease of ignition and combustion. Eight can-type turbojet combustor chambers were used in the combustion gas heater. A quick-acting valve system, which controlled the supply of hot air to the engine, had eight rectangular butterfly valves placed around the pipe. These valves were divided into two sets with alternate valves operating 90° out of phase. Four of these valves led to stacks (shown in fig. 2) that permitted the hot gases to be diverted to the wind-tunnel airstream and thus bypass the engine during ignition of the heater and adjustment to the desired temperature level. When these valves were closed and the alternate four valves were simultaneously opened, the hot gases were directed to the cylindrical plenum chamber that evacuated to the inlet of the engine. These valves, which were light in construction and controlled a relatively large flow area, were actuated by a servomotor. The operating time from closed to open was 0.03 second.

Transient Engine Fuel Valve

During the study of engine fuel-flow reduction as a means of stall alleviation, a quick-acting, hydraulically controlled, pressure-regulating valve was used to make rapid and controlled changes in engine fuel flow. 4441 This valve was based on design considerations of reference 8, and a detailed description of the valve components is contained therein. The frequency response of the valve was sufficiently high to permit fuel changes of as much as 90 percent in as little as 0.01 second.

Instrumentation

The pressure and temperature instrumentation installed in the engine is listed in table I.

The resistance-type thermocouples used for transient temperature measurements were made of 0.0005-inch-diameter tungsten filament and were designed similar to the anemometers of reference 9. They had a flat frequency response from 0 to 100 cycles per second at sea-level pressure and density conditions. Those used in the compressor inlet had a flat response from 0 to 40 cycles per second at the altitude conditions under investigation, while those used at the compressor outlet had a flat response from 0 to 75 cycles per second. The difference is accounted for by variation of the gas density. The lag of this type of thermocouple was estimated to be between 0.001 and 0.006 second. The thermocouples had a normal life of about 40 hours of operation. The carrier current of the filament was kept at low values so that the filament was near the gas temperature being measured.

Reluctance-type pressure transducers with a flat response from 0 to 40 cycles per second at the compressor inlet and 0 to 35 cycles per second at the compressor discharge were used for transient pressure measurements. Responses are for the altitude conditions of this investigation.

The transient test data were recorded on galvanometric oscillographs with galvanometer elements having natural frequencies of about 850 cycles per second. Recording was on photosensitive paper. The recorders had a flat response within ±2 percent from 0 to 60 percent of 850 cycles per second. During operation, test conditions were monitored with a direct-inking oscillograph with a flat frequency response of 40 to 60 cycles per second.

PROCEDURE

Before introducing the transients, operation was stabilized at the engine speed and flight conditions desired. The transient temperature increase was then supplied to the engine by means of the quick-acting valve combination described in the section APPARATUS. Before the valve was opened, flow conditions and temperatures in the heater were stabilized. The range of gas temperatures in the plenum chamber was varied between 200° and 1000° F. Because of dilution, however, the temperatures at the engine inlet were less than half those in the plenum chamber. The length of time the engine was exposed to the elevated temperature was varied from 0.1 to 1.5 seconds. The range of test conditions covered in the study are summarized in table II.

Although only a small number of the transients actually appear in this paper, the large amount of data taken contributed to the basic understanding of the problem. In addition, a large number of the transients were run to establish the stall and blowout limits of the engine by a trial-and-error process.

RESULTS AND DISCUSSION

In order to study the effects of inlet temperature transients on turbojet-engine operation, a large series of tests were made with comprehensive compressor-inlet and -interstage instrumentation. A typical temperature transient which resulted in compressor stall is shown in figure 3; in figure 3(a) are shown time histories of temperatures at nine positions around the inlet together with the maximum, average, and minimum temperatures. All the thermocouples at the inlet indicate about the same trend, and generally the temperature extreme is less than 50° different from the average temperature. Throughout this report, the inlet temperature referred to is always the average gas temperature.

In figure 3(b), traces of engine fuel flow and speed are shown as well as the interstage pressures. During the entire temperature transient, there is no noticeable change in engine fuel flow or speed. This results because the temperature change occurred in about 0.1 second, which is much faster than the time needed for the engine control to take corrective measures. The noteworthy thing indicated by the pressure traces is that the wide fluctuations in pressure arising from compressor stall begin considerably before the maximum temperature change occurs (see fig. 3(a)).

During an armament firing study at the Naval Ordnance Test Station, China Lake, California, some compressor-inlet temperature measurements were made during rocket firing (fig. 4). This armament was a cluster of nineteen 2.75-inch rockets fired in sequence from a launcher mounted about a foot below the wing root inlet of the engine air-induction system. The airplane was flying at an altitude of 45,000 feet and an indicated airspeed of 165 knots. The maximum temperature-rise rates during the flight tests varied from 4500° to 9500° per second; this is within the range of the tests reported herein. Although the engine used in the flight tests was an older design (pressure ratio, 4.2) than those used in the wind-tunnel study, and consequently had a less-sensitive compressor, combustion blowout generally resulted during the rocket firing. 4441

Pressure-Ratio Histories

The effect of inlet temperature transients on compressor pressure ratio is shown in figure 5, where pressure ratio is plotted as a function of corrected engine speed for temperature-rise rates of 2500°, 3500°, and 6500° per second. Timing ticks were put on the histories to indicate the elapsed time. At the lowest temperature-rise rate, the rise in temperature lowers the corrected speed to a point where the 12-14 stage group stalls at a time between 0.02 and 0.03 second (signified by the solid point); this stage group then recovers without surging the compressor. Further increase in temperature simply reduces the corrected speed and compressor pressure ratio, closely paralleling the steady-state operating line. The displacement of the transient history to the left of the operating line arises from the fact that the temperature did not increase simultaneously throughout the compressor; that is, the inlet temperature rise used to compute corrected rotating speed at any given instant was higher than the transient temperature rise at any downstream stage. Therefore, during temperature transients the speed correction using inlet temperature is higher than should be applied to the over-all compressor.

At an inlet temperature-rise rate of 3500° per second, the curve up to the point where the 12-14 stage group stalled is similar to the curve of the lower-temperature rate. However, after 0.03 second the compressor pressure ratio starts to drop rapidly; at 0.07 second after the introduction of the transient, the pressure ratio has dropped from 8.1 to 3.4. In this instance, stalling of stages 12 to 14 is followed by successive stalling of all the stage groups, thus completing a surge cycle. At 0.07 second, the near-vertical rise in pressure ratio represents the end of the temperature rise, the recovery from surge, and the return to near-normal operation at the higher inlet temperature, and hence lower corrected engine speed.

At a temperature-rise rate of 6500° per second, the corrected engine speed decreases to about 7170 rpm before stall is obtained in the 12-14 stage group. After stall inception in this critical stage group, the drop in pressure ratio is rapid to about 3.64 at 0.06 second. Combustion blowout then occurs.

Superimposed on figure 5 is the quasi-static compressor stall line, which was obtained by ramp increases in fuel flow sufficiently large to stall the compressor. It is noteworthy that this line did not represent the singular compressor operating limit for the three temperature transients shown; that is, the 12-14 stage group stalled on both sides of the quasi-static compressor stall line. The corrected engine speed (based on inlet temperature) at which this stage group stalls, then, depends on the rate of inlet temperature change.

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Some insight into the stall-inception process is shown by the data in figure 6, which is a time history of the compressor stage-group pressure ratios together with inlet and discharge temperatures for a transient which resulted in compressor surge. Although the inlet temperature starts to rise at zero time, it is some time later before any of the compressor stages show any appreciable change in pressure ratio. At 0.025 second, the pressure ratio of the 12-14 stage group starts to drop. This drop indicates stalling of this stage group. Thus, this is the critical stage group that is the first to stall, which generally signals the inception of compressor surge in the high-speed region. After the pressure ratio in the 12-14 stage group drops, the upstream stage group (stages 8 to 11) loads up, until at 0.035 second this stage group also stalls, and so through to the inlet stage group. This stalling action progressing through the compressor constitutes a part-surge cycle and is similar to the process observed previously in other altitude-wind-tunnel engine studies (unpublished data). The over-all pressure ratio starts decreasing slowly at 0.01 second, which does not represent stall initially, but rather reflects the slight drop in pressure ratio of several of the stage groups resulting from the decrease in corrected stage speeds. At 0.045 second, however, the over-all pressure ratio decreases at an accelerated rate, resulting from the steady unloading of several stalled compressor stages. The compressor-discharge temperature does not change appreciably until about 0.015 second after the inlet temperature rise. This is a measure of the transport time for the hot air to move from the front to the rear of the compressor.

A more precise measure of transport time from front to rear of the compressor is shown by the data presented in figure 7, where compressor-inlet and -discharge temperatures are plotted as functions of time. The inlet temperature oscillation is about 55°, and the frequency is 6 cycles per second. Although the data scatter is considerable, it appears that the discharge temperature lags behind that at the inlet by an average of 0.015 second. This is the time required for the air to travel from the inlet measuring station to the discharge of the compressor. It is evident that a rapid increase in temperature at the compressor inlet would require some time before it is felt back in the compressor, depending upon which stage (hence transport time) is of interest.

The effect that transport time has on the stage speed match is shown by the curves in figure 8, illustrating the corrected stage speeds during steady-state operation and during an inlet temperature rise of 5000° per second. Whereas the difference in corrected stage speed between the inlet and the 12th stage is 23 percent for steady-state operation, it drops to 17 percent 0.02 second after the temperature transient is introduced. From these data it is quite evident that the steady-state stage match is upset considerably during a rapid inlet temperature transient. This helps to explain the variation in compressor-inlet corrected speed and compressor pressure ratio at which the 12-14 stage-group stall occurs with various temperature-rise rates, as indicated in figure 5.

Required Temperature Rise for Stall

An insight into the mechanism of temperature-induced stall can be obtained by a study of transient temperature rise at stall inception. The variation of measured inlet temperature rise with temperature-rise rate is shown in figure 9 for several temperature transients at three engine speeds. The higher the temperature-rise rate, the higher the measured inlet temperature rise to stall. At an engine speed of 7700 rpm, an inlet temperature rise of about 67° R is required for stall at a temperature-rise rate of 3500° per second; whereas at a temperature-rise rate of 6500° per second, a temperature rise of 98° R is required. As indicated in figure 9, temperature-induced stall does not occur to the left of the cross-hatched regions (below 2500° to 3500° per sec) because the compressor is able to readjust itself to the stage mismatch.

The inlet temperature rises at which stall occurs are very low in comparison with the reported engine-inlet temperature rises during rocket firing (ref. 4 and other unpublished data). It might be expected that a temperature change at the inlet would result in a larger temperature increase at the discharge because of the compression process. However, a given temperature rise at the compressor inlet produces about the same temperature rise at the discharge, that is, the heat lost to the compressor casing and blades approximates the temperature rise from the compression process.

In examining the data of figure 9, it is of interest to estimate the temperature rise required at the critical stage of the compressor to induce stall. As demonstrated by the results in figure 6, the 12-14 stage group is the first to stall during these temperature transients. By knowing the transport time through the compressor and the inlet temperature-rise rate, it is possible to make a first-order approximation of the transient temperature rise at the 12th stage at the time of stall inception. At an engine speed of 7700 rpm, the measured inlet temperature rise for stall is about 82° R at a temperature-rise rate of 5000°

per second. With an estimated transport time of 0.01 second from the inlet to the 12^{th} stage, the transient temperature rise ΔT at the 12^{th} stage for stall would be

$$(\Delta T)_{12^{th} \text{ stage}} \approx 82^{\circ} \text{ R} - 0.01 \text{ sec}(5000^{\circ} \text{ per sec}) \approx 32^{\circ} \text{ R}$$

Similarly, the transient temperature rise at the 12^{th} stage was computed for the three engine speeds and is shown by the dashed line in figure 9. It was found that the estimated transient temperature rise at the 12^{th} stage for stall inception is independent of temperature-rise rate.

The reason a given stage group generally starts the temperature-induced stall process can be seen from the data in figure 10, where stage pressure ratio is plotted as a function of corrected stage speed. In addition to the steady-state pressure ratios for the stage groups, stall lines (which were obtained from pressure-ratio histories obtained during surge) are presented for each stage group.

The data of figure 6 indicate little or no stage pressure-ratio changes from steady state to stall. By assuming a constant stage pressure ratio during a temperature transient, the temperature rise required for a corrected speed change from the steady-state line to the stall line was computed for each stage group. The temperature rise required to stall the 1-4 stage group is 380° R; whereas only 75° R is required to stall the 12-13 stage group (fig. 10). (Because additional interstage data were available for this study, it was possible to narrow the critical stage group to include only the 12^{th} and 13^{th} stages. In some of the previous figs., this group included stages 12 to 14.) Obviously, then, for these conditions, the 12-13 stage group would stall before the inlet group during a temperature transient. If, however, a sufficiently high inlet temperature-rise rate were put into the engine, so that a temperature rise of 380° R occurred at the inlet stages before any change occurred at the 12^{th} stage, the inlet stages would, of course, stall first and thus start the surge cycle.

The estimated transient temperature rise required at the 12^{th} stage for stall is about 75° R (uncorrected engine speed, 7500 rpm) in figure 10; whereas the data in figure 9 indicate the temperature rise to be about 40° R for stall at this engine speed. It should be pointed out that the temperatures presented in figure 9 were measured on hot-wire anemometers and had no lag correction applied to them. Although this correction was not determined, it was estimated to be between 0.001 and 0.006 second. Assuming a lag correction of 0.005 second and an inlet temperature-rise rate of 5000° per second, a temperature correction of 25° R should be added to the measured rise. This would bring the measured temperatures in figure 9 toward agreement with the estimate of required transient temperatures for stall shown in figure 10.

It is apparent that the engine of this investigation stalls with relatively small inlet temperature increases at rise rates above about 2000° per second. In order to check this significant finding, tests were run on an engine of different manufacture. This engine was also in the 10,000-pound-thrust class but had a 16-stage axial-flow compressor (rated pressure ratio, 7.9) and a three-stage turbine. Results of inlet temperature transients with this engine revealed that it too stalled with temperature rises of only 100° to 200° R for rise rates up to 5000° per second. Thus it appears that, with current moderate- or high-pressure-ratio axial-flow turbojet engines, compressor stall will accompany rapid inlet temperature transients for temperature rises in the range between 50° and 250° R. 4441

From the foregoing data, it is evident that the steady-state and stall characteristics of the compressor stage groups are of prime importance in determining how large an inlet temperature change an engine can withstand. If this compressor information is known, it is possible to estimate what temperature rise will result in stall. The analytical procedure for such a solution, which was checked with experimental data, is given in reference 10.

Effect of Partial Admission of Hot Gases to Compressor Inlet

For some of the inlet-temperature transients a splitter plate was placed in front of the compressor, which effectively restricted the hot gases to only half of the compressor face; the other half of the compressor inlet remained exposed to the cool tunnel air during the rapid temperature increase. The purpose was to determine the effect of temperature gradients on the allowable temperature rise for stall. The results of these tests are summarized by the data in figure 11, where temperature rise for stall is plotted as a function of temperature-rise rate at the compressor half exposed to the hot gas. For comparison, the curve for full admission at an engine speed of 7700 rpm was reproduced from figure 9. Increased inlet temperature-rise rate has the same effect on allowable temperature rise to stall for both partial and full admission of the hot gases to the compressor inlet. However, when only half of the compressor face is exposed to the increasing gas temperature, a higher temperature rise is obtained before stall is encountered.

However, the temperature rise for stall at 50-percent admission is ~~considerably~~ less than twice the inlet temperature rise for full admission. For example, at rated speed and an inlet temperature-rise rate of 7000° R per second, full admission of the hot gases at the compressor face results in stall at a temperature rise of 103° R; with 50-percent admission, stall is not obtained until the temperature in the heated portion of the inlet has risen 145° R. This would be expected because of the hot gas diffusion in passing through the compressor and the higher heat transfer to the blading resulting from the intermittent passing through the hot and cold regions. Thus, except for very localized inlet temperature transients, the sensitivity of the compressor to stall with inlet temperature gradients would not be greatly different from the uniform inlet temperature transients presented herein. Nevertheless, the temperature rise for stall with uniform inlet profiles is somewhat conservative where temperature gradients exist.

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Effect of Altitude on Stall and Blowout

The marked effect of altitude on combustion blowout induced by inlet temperature transients is shown in figure 12, where the inlet temperature-rise rate is plotted as a function of altitude. Any transient temperature-induced compressor stalls in the area above the curve plotted in this figure resulted in combustor blowout; that is, although it was possible to obtain compressor stall at all altitudes investigated, stall at the temperature-rise rates in the lower area shown on the figure resulted usually in one or two surge cycles without combustor blowout. At an altitude of 44,000 feet and a flight Mach number of 0.8, rapid temperature transients caused surge, but the combustion process was sufficiently stable so that a normal recovery always took place without combustion blowout. At an altitude of 64,000 feet and a flight Mach number of 0.8 (inlet pressure, 190 lb/sq ft), inlet temperature transients about 1800° per second caused stall and resulted ultimately in blowout. Thus, inlet temperature changes that posed little difficulty with this engine at an inlet pressure of 498 pounds per square foot gave rise to a severe problem at an inlet pressure of 190 pounds per square foot.

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Although the pressures at which blowout was obtained should be representative of present-day engine designs, some variations might be expected, depending on the relative stability of the combustor and the level of the compressor pressure ratio.

Effect of Remedial Measures

A means of alleviating compressor stall resulting from rapid inlet temperature transients is to reduce the engine fuel flow prior to the temperature change, thereby shifting the engine operating point so as to lower the compressor pressure ratio and thus increase the stall margin. This process is illustrated by the two inlet temperature transients shown in figure 13. For the inlet temperature rise where no change was made in engine fuel flow, the compressor stalled 0.02 second after the introduction of the temperature transient. A 31-percent reduction in engine fuel flow prior to the temperature change reduced the compressor pressure ratio about 10 percent; then a similar temperature input to the engine did nothing more than reduce the corrected speed and thus change the steady-state operating point.

Results of a series of tests made to determine the effectiveness of engine fuel-flow reduction as a means of preventing stall are summarized in figure 14. Fuel flow was reduced as much as 40 percent prior to imposing inlet temperature transients from about 3000° to 12,000° per second. The lead time of the fuel-flow reduction over the inlet temperature change was varied from 0.3 to 1.1 seconds but was found to have no effect on the results. Although there is considerable data scatter, lines representing fuel-flow reductions of 10, 20, and 40 percent were faired through the experimental points. As would be expected, the greater reduction of fuel flow permits higher inlet temperature rises before reaching compressor stall limit. A 20-percent fuel-flow reduction at an inlet temperature-rise rate of 8000° per second increased the allowable temperature rise before stall from 110° to 160° R. The noteworthy thing about these data is that a 40-percent reduction in engine fuel flow would still only permit an inlet temperature rise of 220° R at a temperature-rise rate of 10,000° per second. This is a relatively small allowable inlet temperature rise in comparison with that obtained during rocket firing (ref. 4). 4441

The engine with which these tests were conducted was equipped with variable-position inlet guide vanes and compressor-interstage bleed. Both these modifications are used to improve off-design speed performance or stall margin. It was found that where the stall margin was improved the engine could withstand somewhat higher temperature transients; likewise, where use of either the bleed valves or guide vanes reduced the stall margin (at high speeds), the allowable inlet temperature rise was decreased. These modifications might thus be useful at low corrected engine speeds, such as would be obtained at a flight Mach number of 2.0, for example.

CONCLUDING REMARKS

An extensive study was made to determine why rapid temperature increases at the inlet of turbojet engines cause compressor stall and combustor blowout. With the help of high-response instrumentation, it was found that present-day turbojet engines could be expected to stall with surprisingly small inlet temperature rises. For example, with one engine investigated, it was found that compressor stall would be encountered before the inlet temperature had risen 100° R when the rate of temperature increase was 5000° R per second. Compressor stall in this engine was initiated by stalling of the 12-13 stage group, where the stall margin was least at high rotative speeds.

At high altitudes where the combustion process is always less stable, compressor stalls induced by inlet temperature transients were particularly critical because they generally resulted in combustion blowout.

It was found that internal flow changes within the axial-flow compressor lagged considerably behind the rapid inlet temperature transients. Thus it was possible to stall the rear compressor stages by decreasing the corrected speed with temperature transients before the pressure ratio
4441 changed appreciably. As a result, the maximum allowable transient temperature rise of a compressor can be estimated by knowing the stage stall and steady-state operating lines.

The stage mismatch that resulted from the rapid inlet temperature transients meant that the over-all compressor would no longer stall on the usual quasi-static stall-limit line. Depending on the rate of temperature increase, a compressor can be made to stall on either side of the conventional stall line.

From some work on partial admission of the hot gases to the compressor inlet, it was found that the engine could withstand larger inlet temperature rises before stall if the high temperatures were somewhat localized. For example, at rated speed and an inlet temperature-rise rate of 7000° R per second, full admission of the hot gases at the compressor face resulted in stall at a temperature rise of 103° R; with 50-percent admission, stall was not obtained until the temperature in the heated portion of the inlet had risen 145° R.

It was found that the only remedial action that afforded any measure of relief from the temperature-induced stalls was a sudden reduction in engine fuel flow just prior to the temperature increase. A 20-percent fuel-flow reduction at an inlet temperature-rise rate of 8000° R per second increased the allowable temperature rise before stall from 110° to 160° R.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, March 28, 1957

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[redacted]

TABLE I. - INSTRUMENTATION

Station	Steady state				Transient				
	Total-pressure probes	Wall static-pressure probes	Stream static-pressure probes	Temperature probes	Temperature probes	Total-pressure probes	Total-pressure probes (a)	Temperature probes (a)	Static-pressure probes (a)
Upstream of compressor	-	-	-	-	10	-			
Compressor inlet	45	4	14	12	0	0			
1st Stator	14	4	0	2	0	2	1	1	0
2nd Stator	11	4	0	2	0	0	0	1	0
4th Stator	5	2	0	1	0	1			
7th Stator	5	1	0	1	0	[b]1			
11th Stator	5	1	0	1	0	[b]1			
13th Stator	10	1	0	1	0	[b]1			
14th Stator	10	1	0	1	0	1			
Compressor outlet	20	3	0	19	1	2	1	0	0
Turbine inlet	12	0	0	28	0	0			
Turbine outlet	18	0	0	15	0	0			
Tailpipe	16	0	0	26	2	1	1	2	3
	170	21	14	109	13	10	3	4	3

[a]Direct-inking oscillograph recordings with moderate transient response used for monitoring purposes only.

[b]Not in for all runs.

Individ - 109 temp
Scani - 205 press
52
17 temp
36 press
(157 channels)

[redacted]

TABLE II. - TEST CONDITIONS

Type of test	Inlet-pressure, lb/sq ft abs	Number of Transients
Temperature increase	498 392 308 242 190	248 24 48 448 24
Temperature increase (partial admission of hot gases at compressor inlet)	242	174
Cyclic temperature change	498 242	64 40
Engine fuel reduction	242	192
Compressor bleed actuation	242	72
Inlet-guide-vane actuation	242	96
Combined inlet-guide-vane and fuel reduction	242	136
Establishment of stall lines	1194 498 242 190	40 232 88 152

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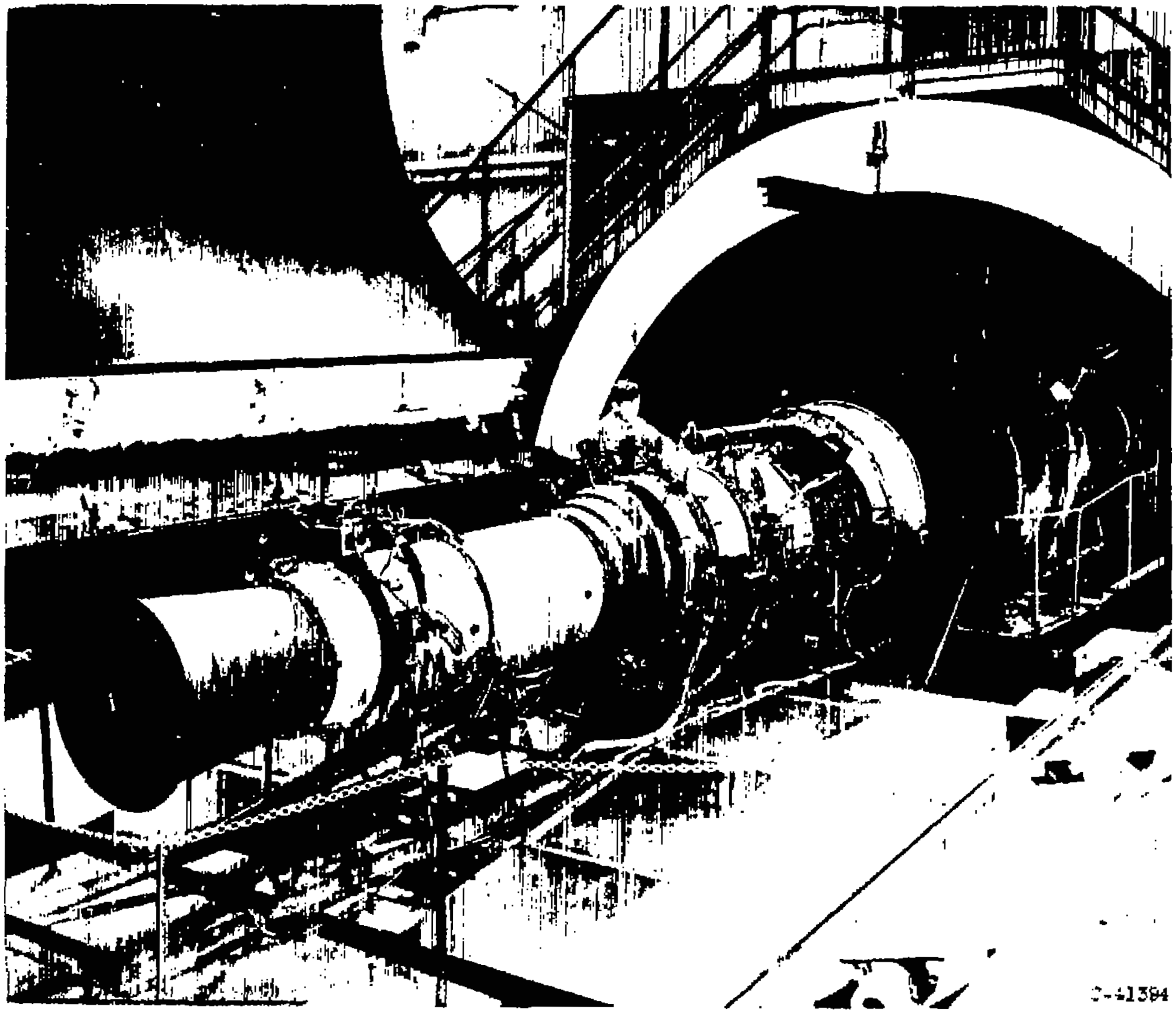

Figure 1. - Turbojet engine and inlet-air heater installed in altitude wind tunnel.

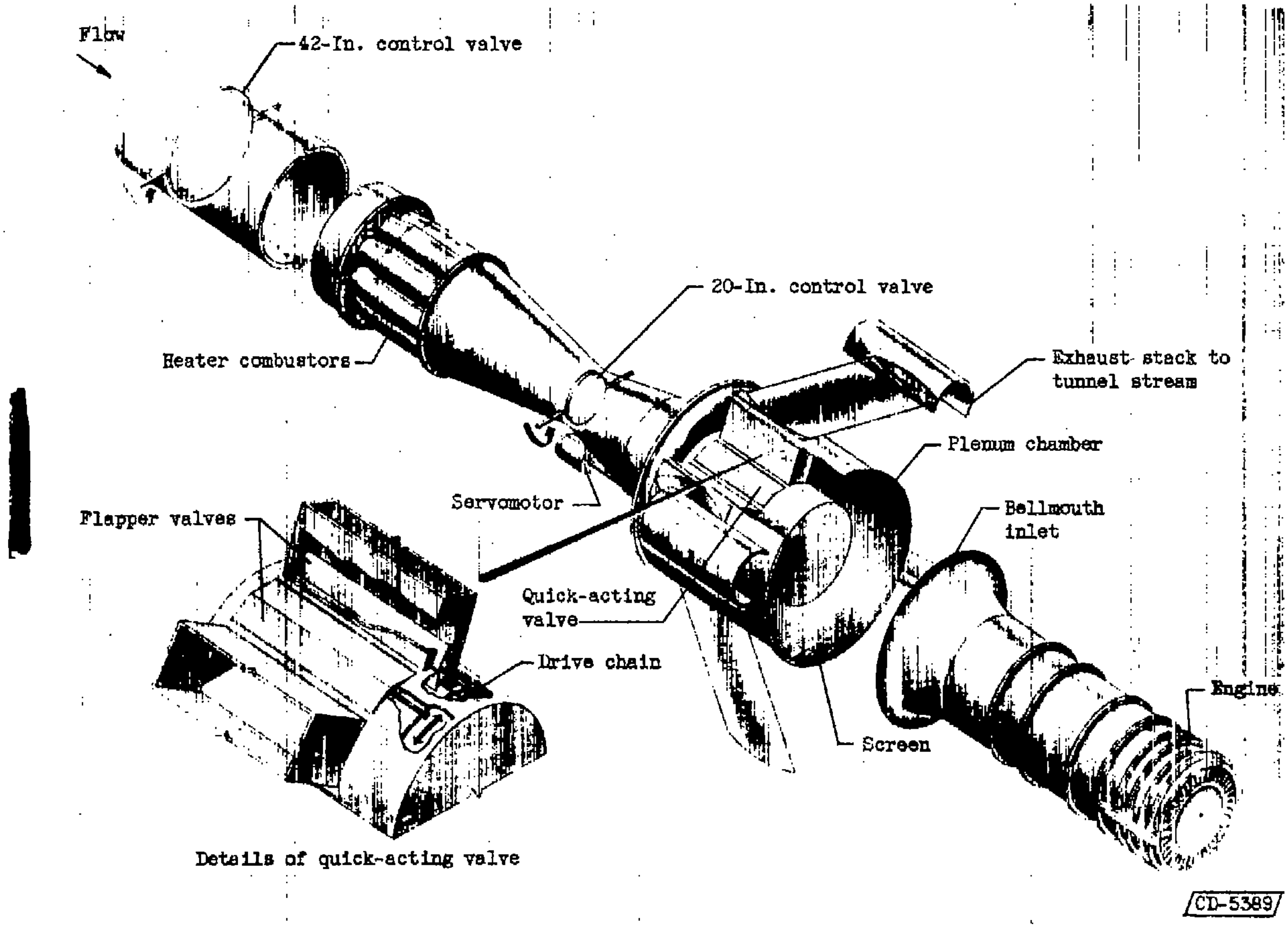

Figure 2. - Schematic diagram of heater and quick-acting valve combination used to introduce temperature transient at face of turbojet engine.

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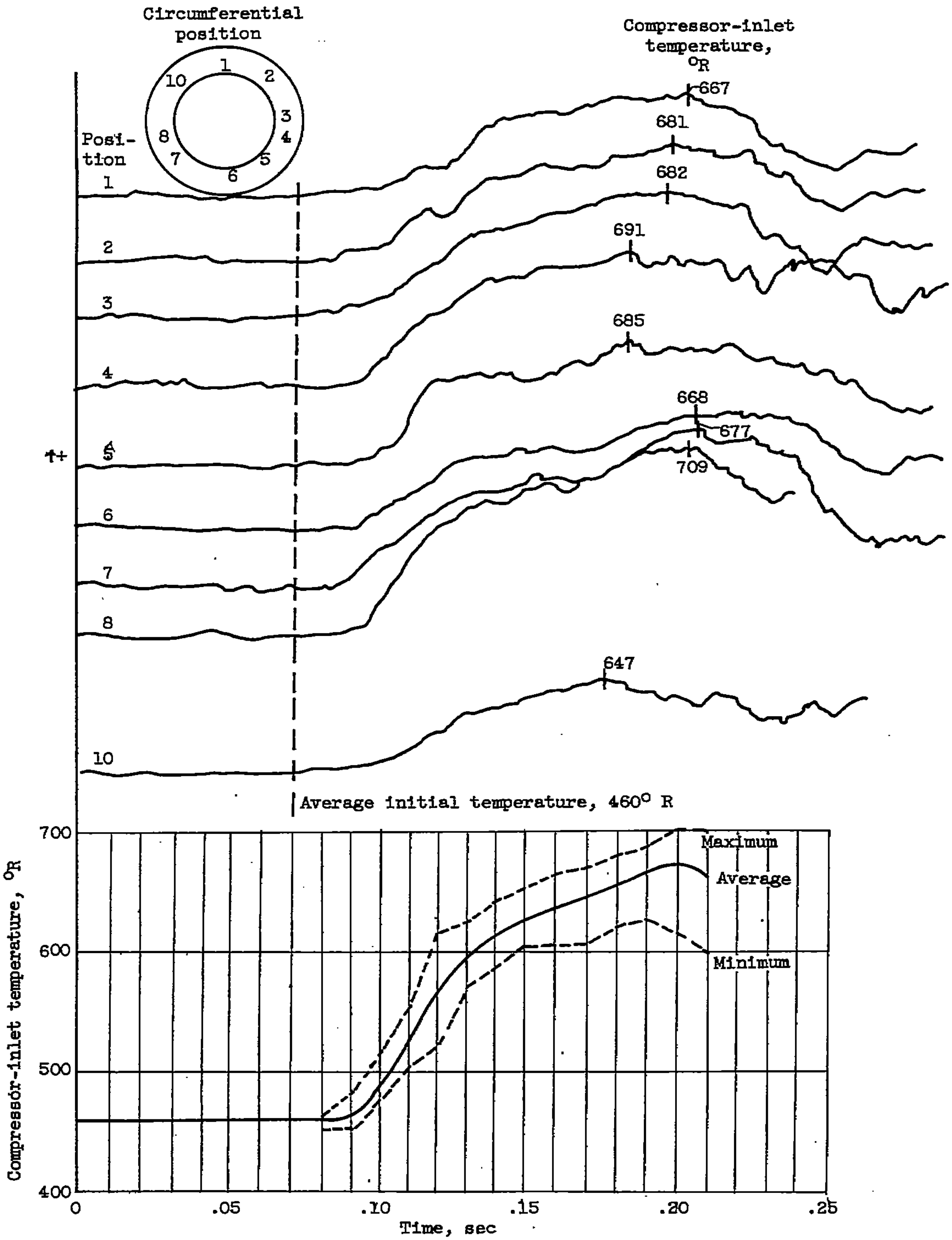
CU-3 back



(a) Individual and average compressor-inlet temperatures.

Figure 3. - Typical inlet temperature transient.

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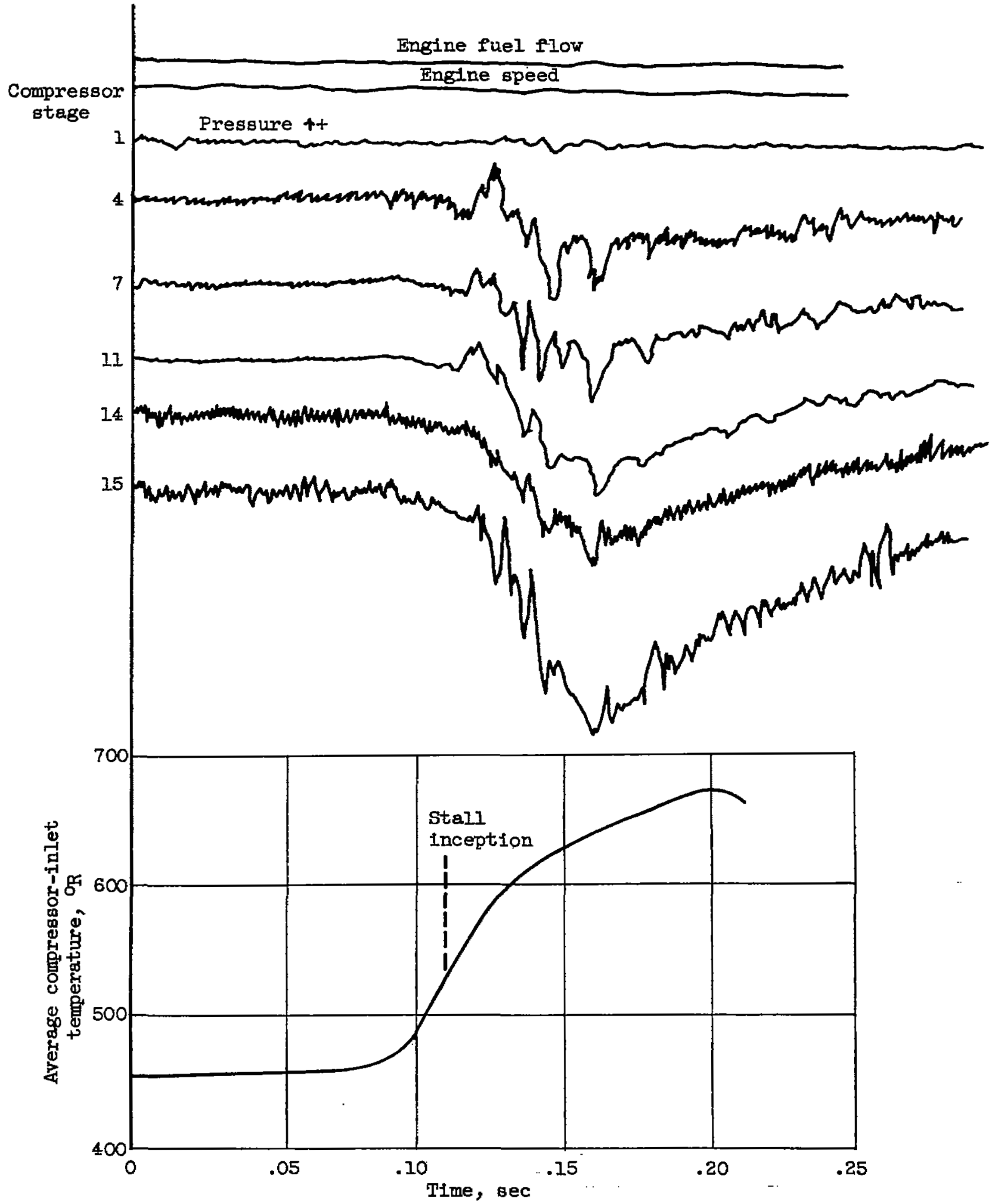

(b) Compressor stage pressures, fuel flow, and engine speed.

Figure 3. - Concluded. Typical inlet temperature transient.

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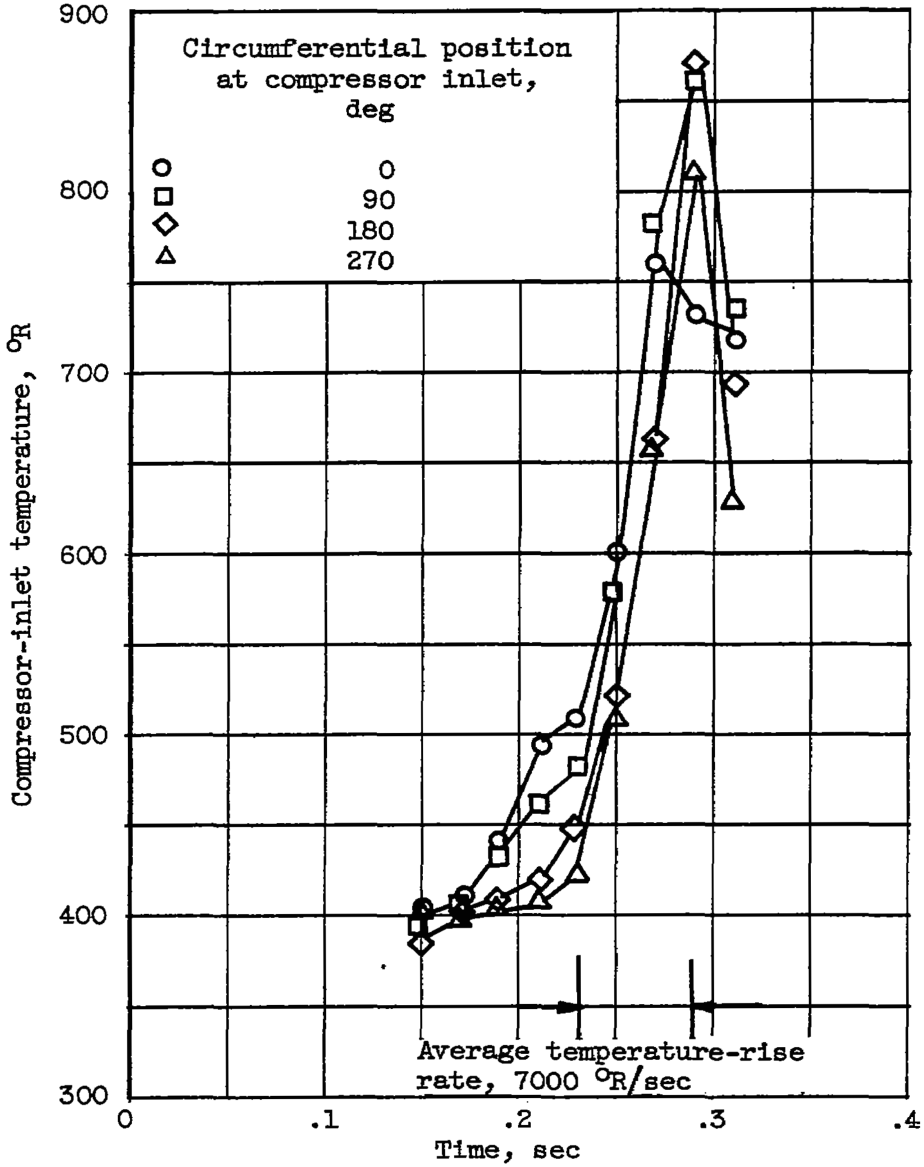

Figure 4. - Measured compressor-inlet temperatures during rocket firing from airplane flying at altitude of 45,000 feet at indicated airspeed of 165 knots. Data from Naval Ordnance Test Station, China Lake, California.

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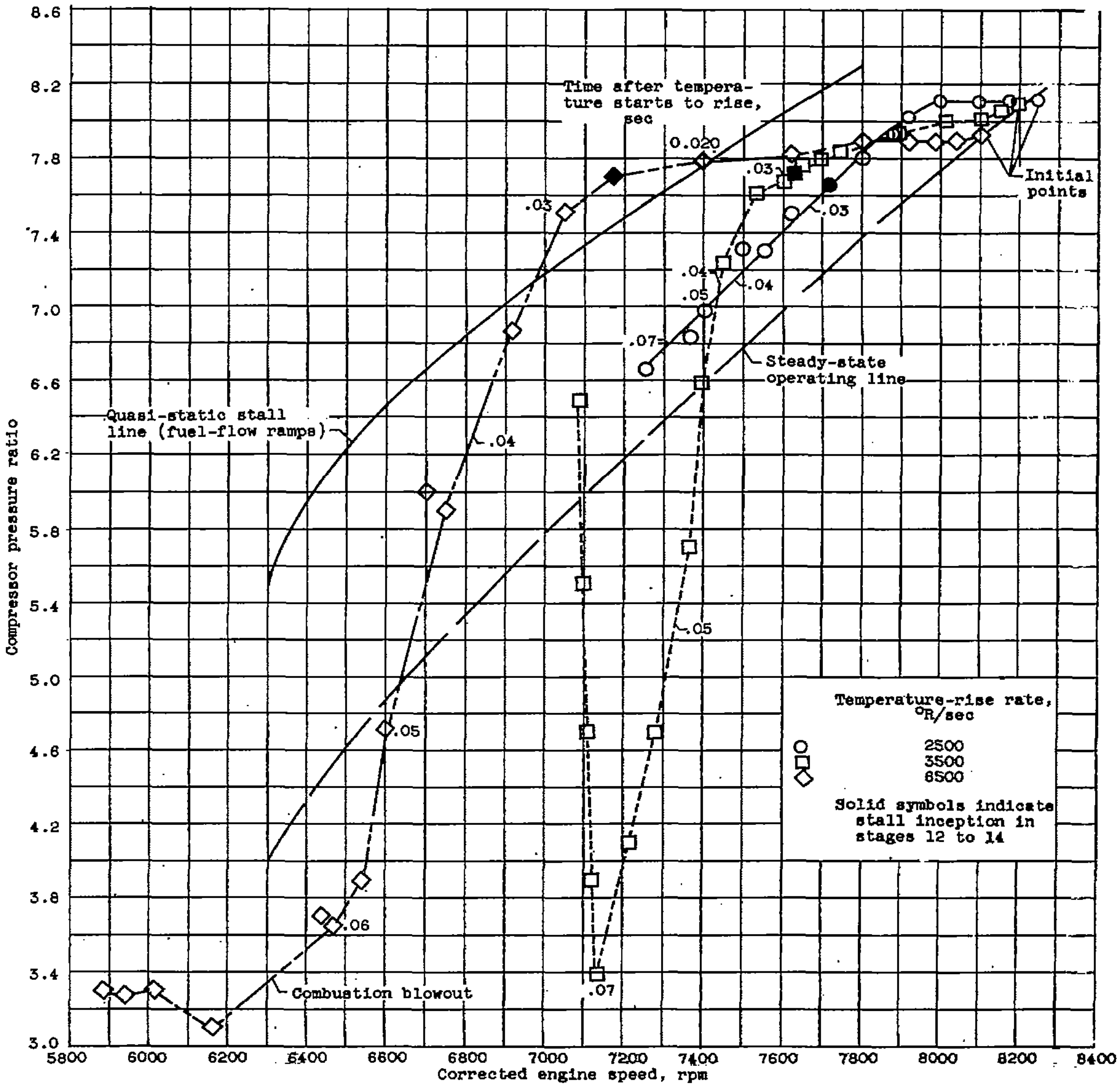

Figure 5. - Variation of compressor pressure ratio with corrected engine speed for several inlet temperature transients covering a range of temperature-rise rates. Inlet total pressure, 242 pounds per square foot.

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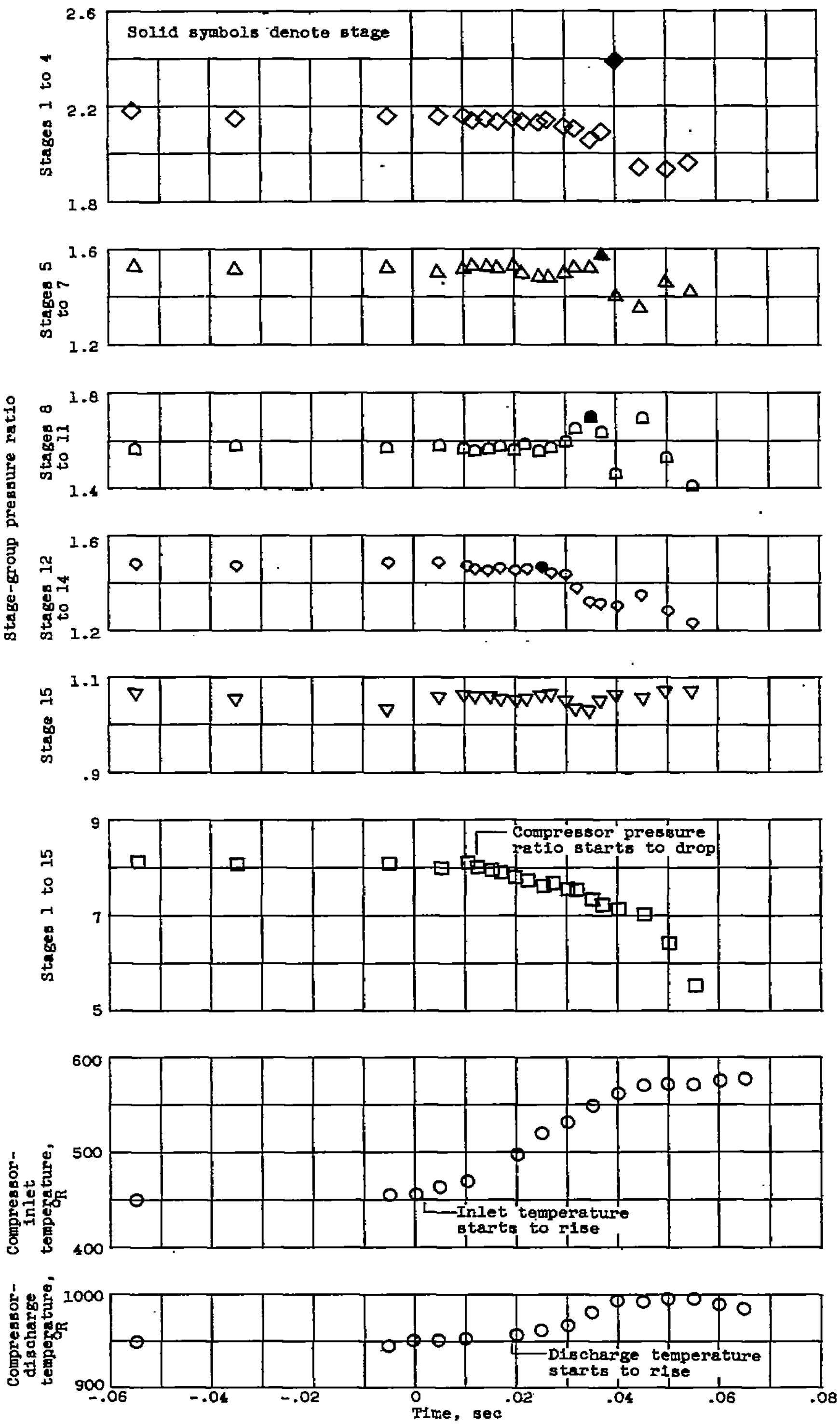

Figure 6. - Time history of compressor pressure ratios and temperatures during a typical inlet temperature transient. Inlet total pressure, 242 pounds per square foot.

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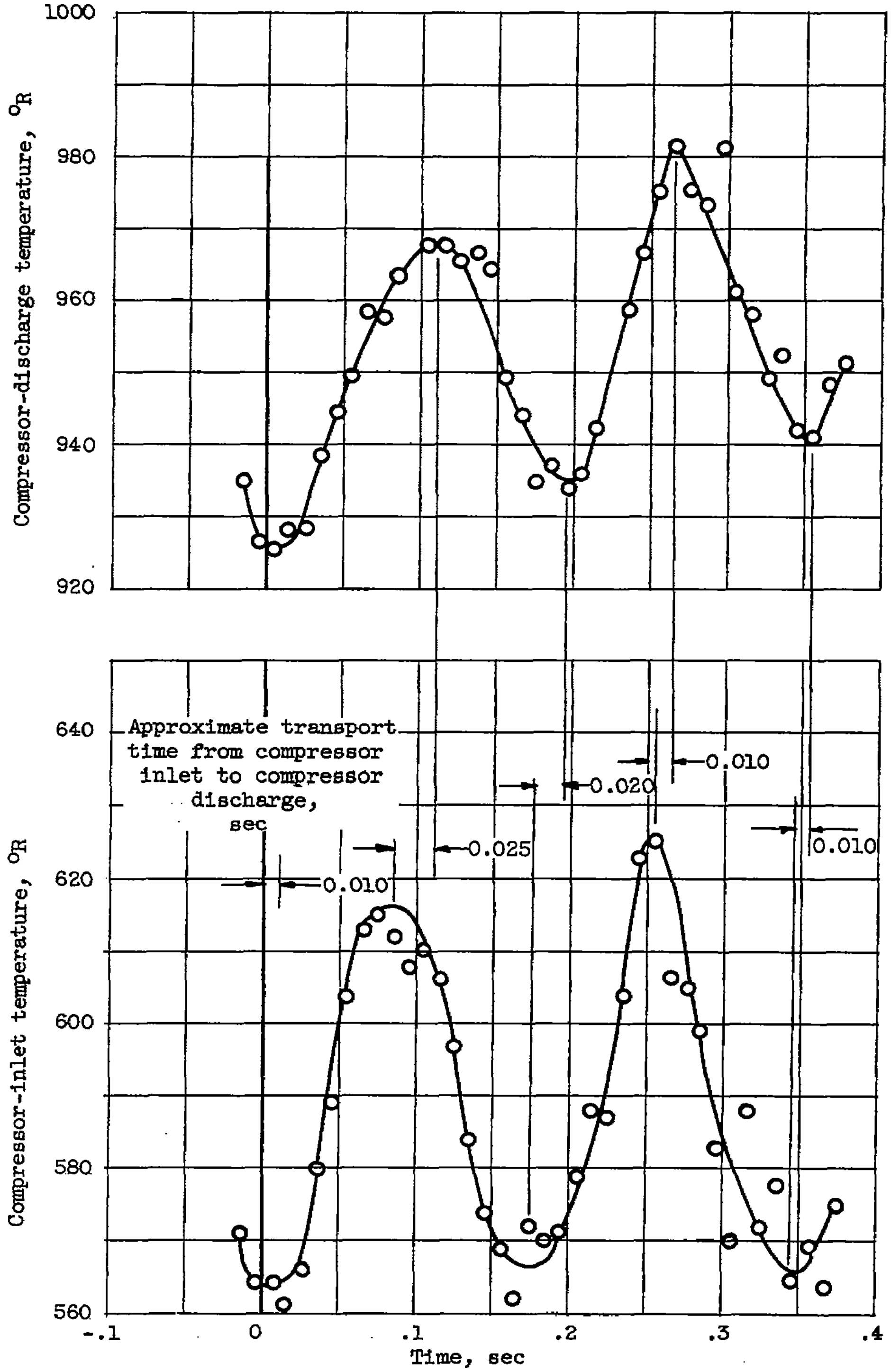

Figure 7. - Compressor temperature measurements during inlet temperature cycling.

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CU-4

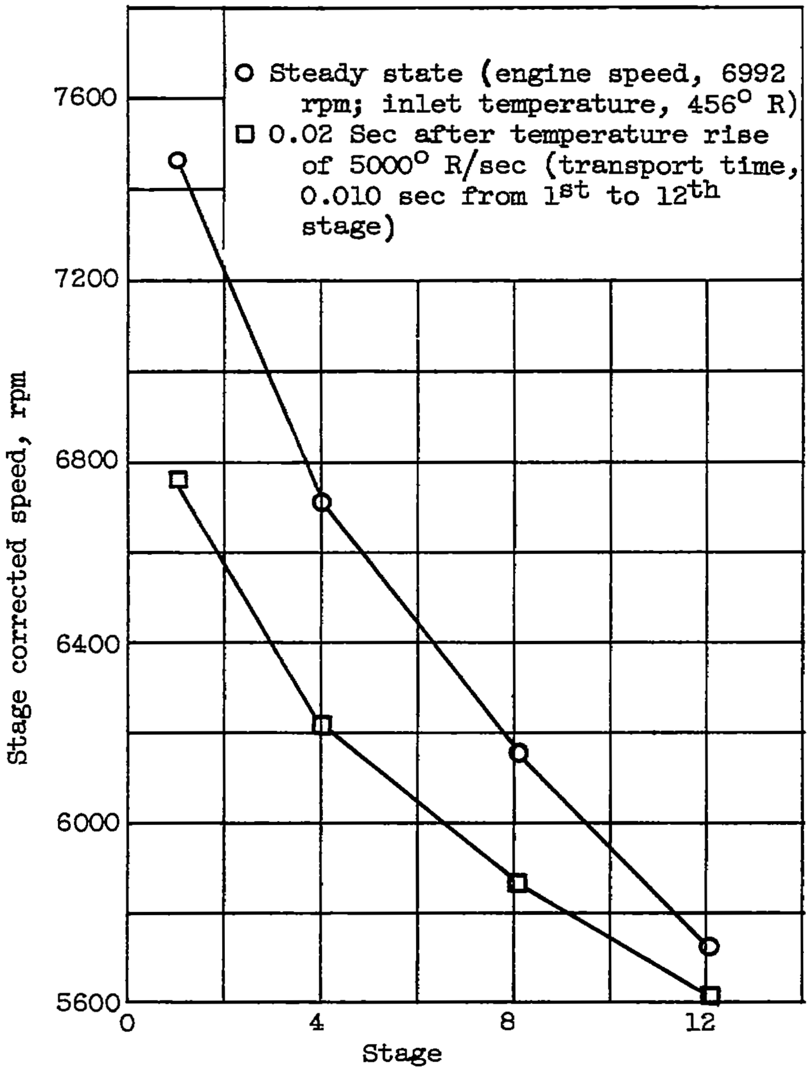
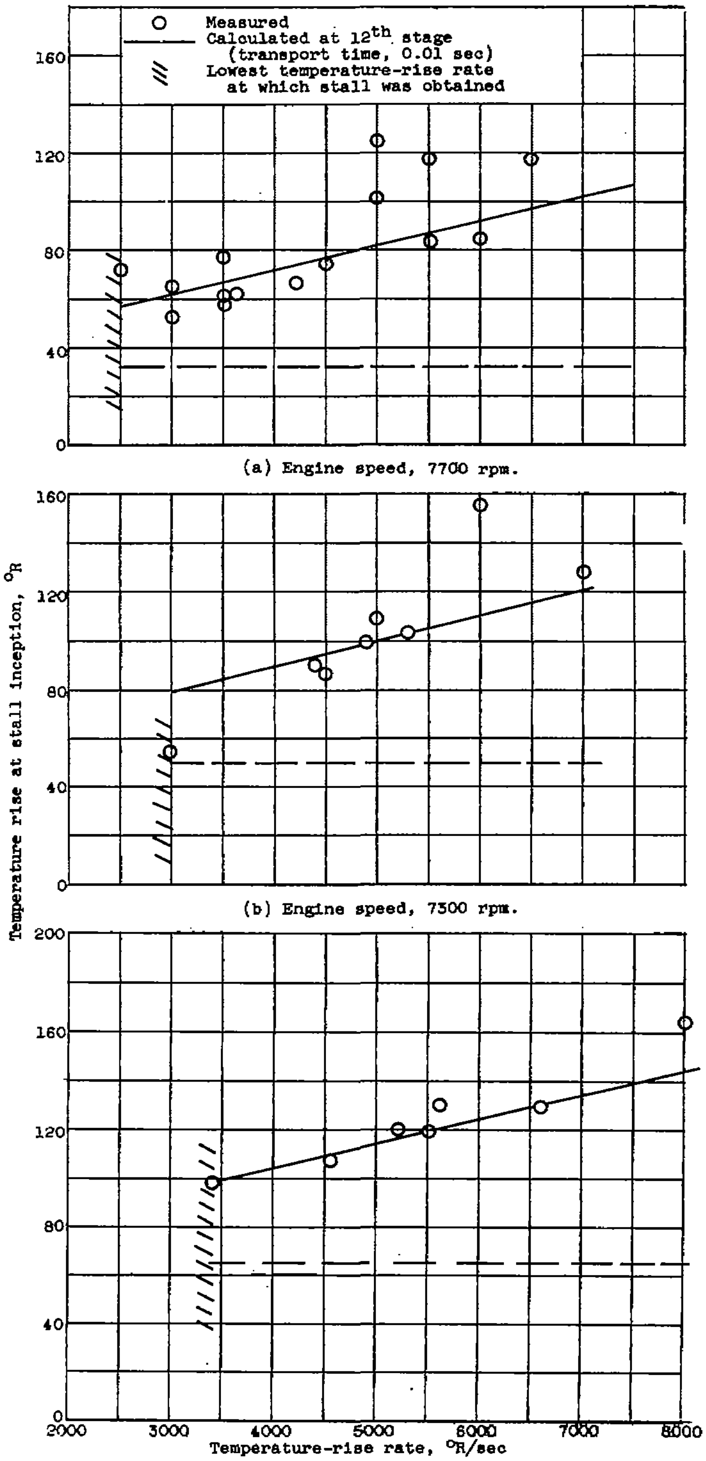


Figure 8. - Effect of inlet temperature transient on compressor stage speed match.



(c) Engine speed, 6900 rpm.

Figure 9. - Measured compressor temperature rise as function of temperature-rise rate. Inlet total pressure, 242 pounds per square foot.

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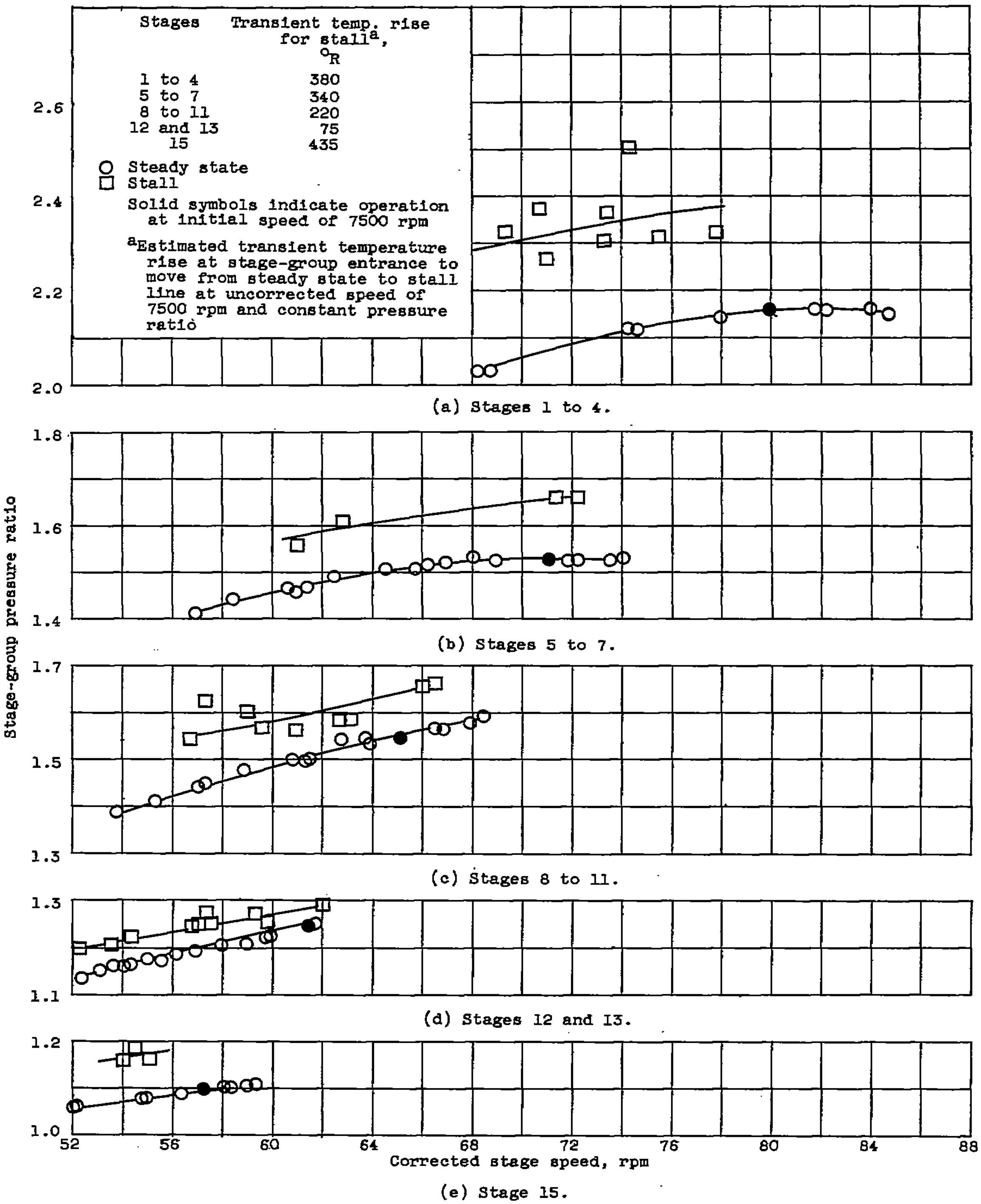

Figure 10. - Steady-state and stall lines for compressor stage groups obtained from temperature transient (fuel-flow steps). Inlet total pressure, 242 pounds per square foot.

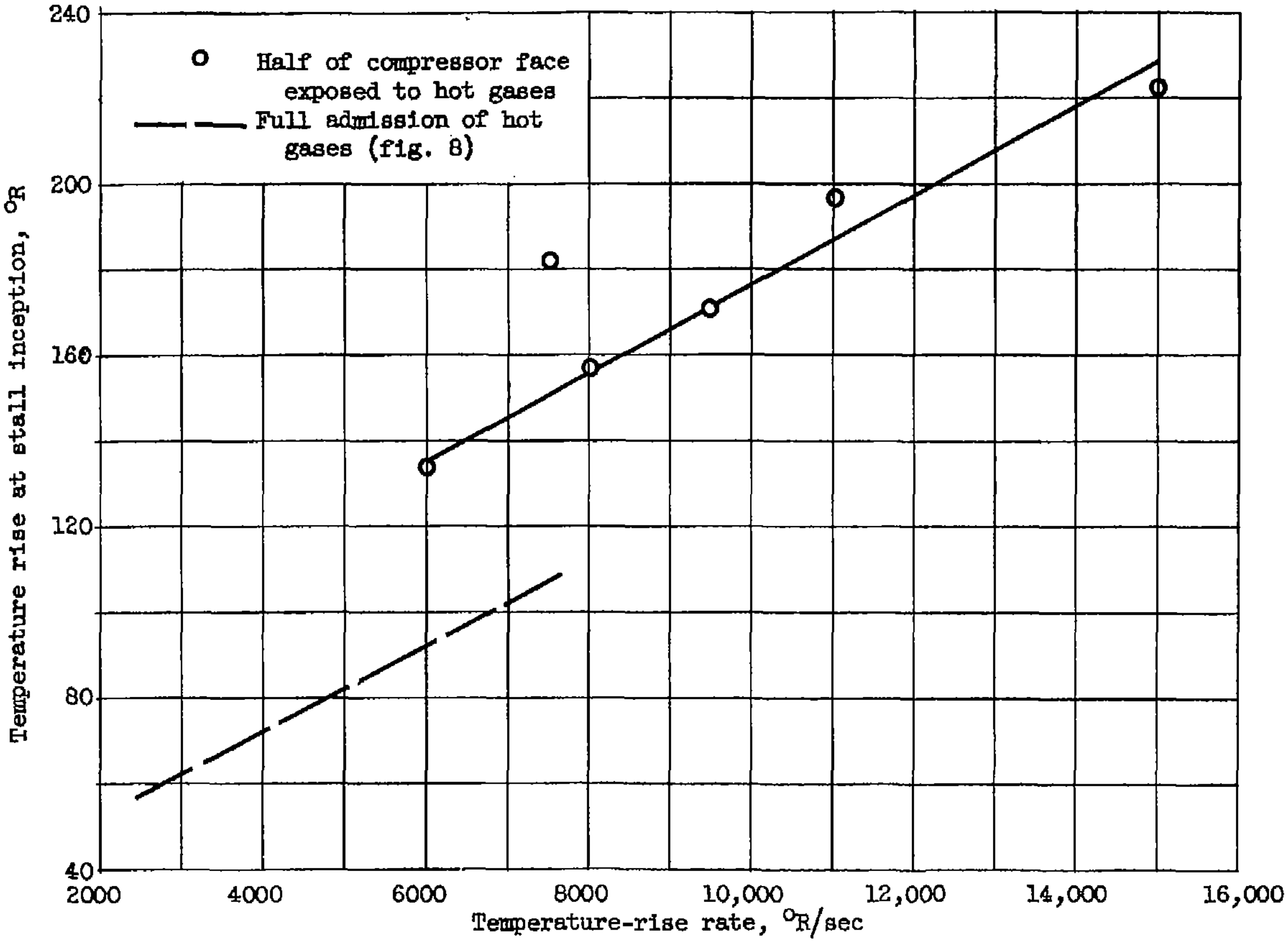

Figure 11. - Effect of partial admission of hot gas at compressor inlet during rapid inlet temperature rise. Initial engine speed, 7700 rpm.

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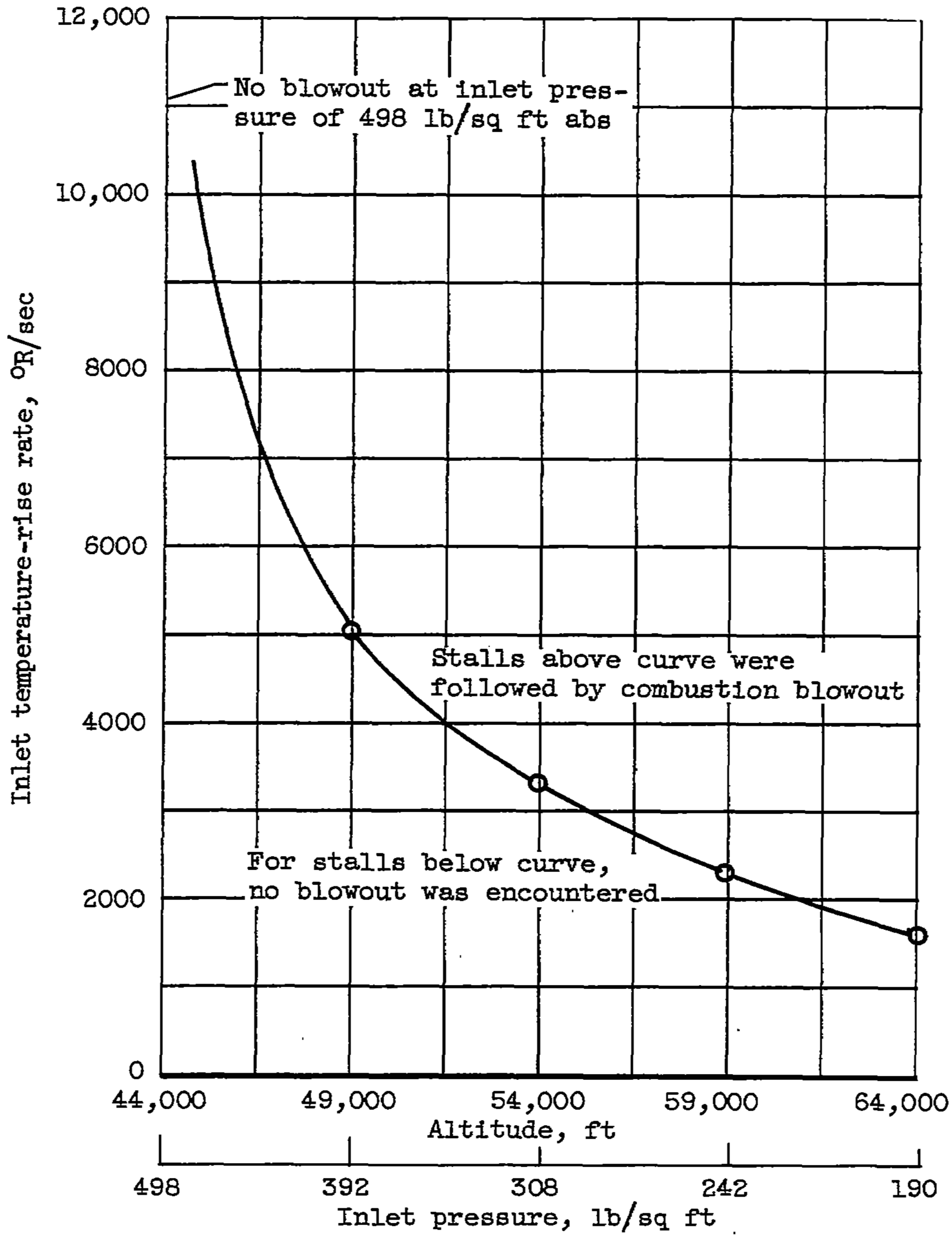

Figure 12. - Effect of altitude on minimum allowable temperature-rise rate resulting in combustor blowout. Flight Mach number, 0.8.

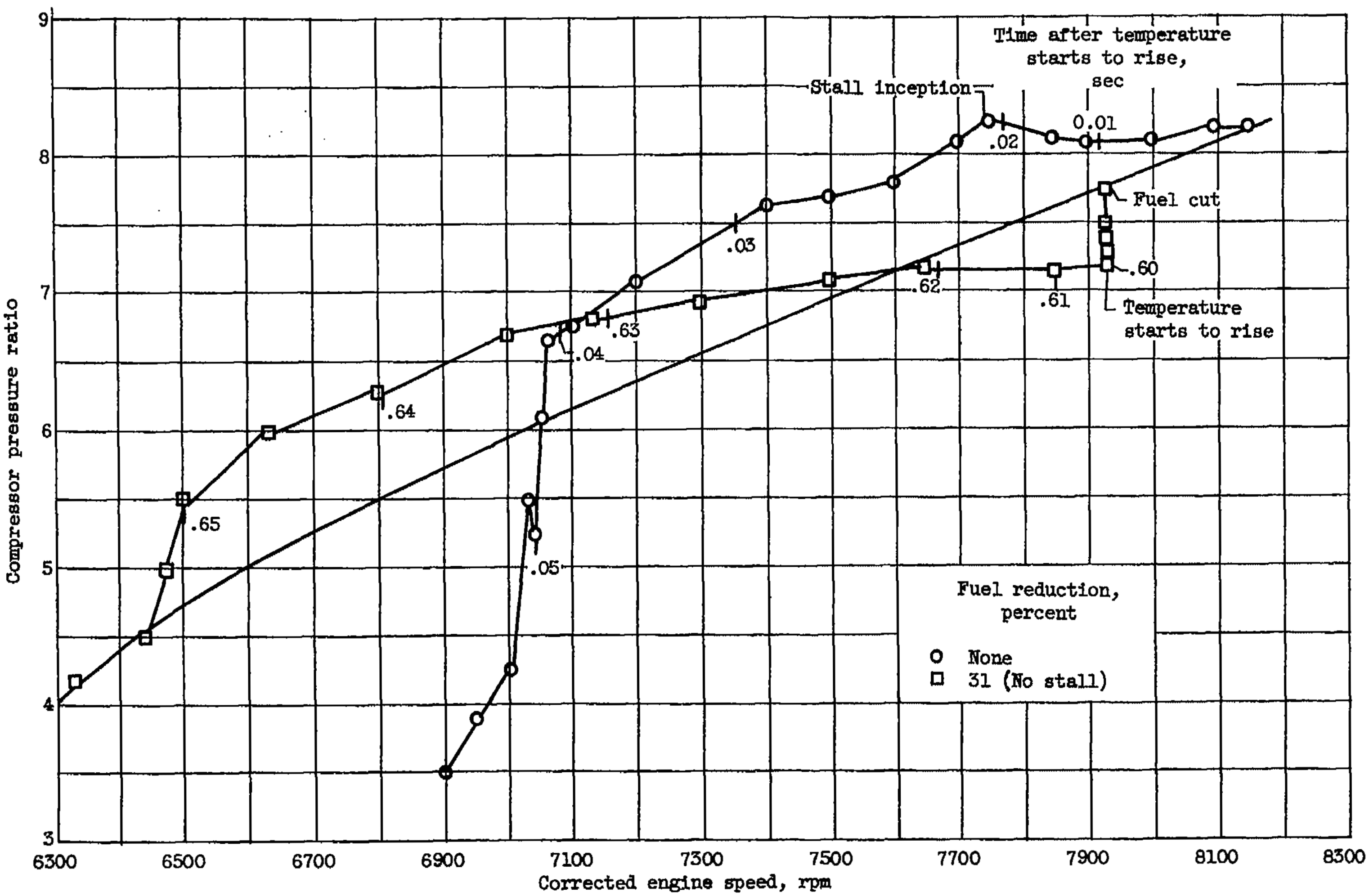

Figure 13. - Effect of engine fuel reduction prior to inlet temperature change. Inlet total pressure, 242 pounds per square foot.

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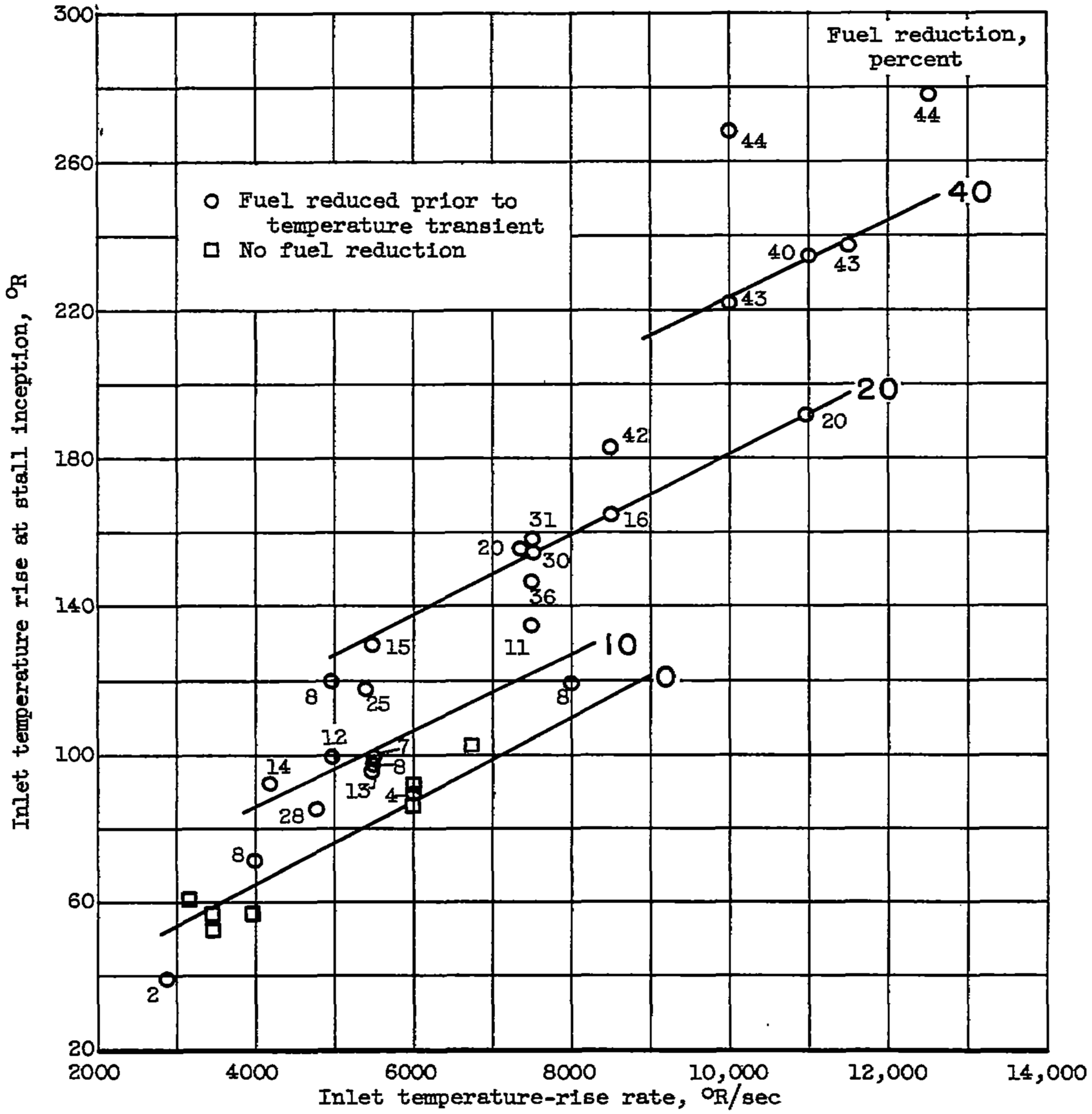

Figure 14. - Effect of engine fuel reduction on allowable inlet temperature rise. Inlet total pressure, 242 pounds per square foot.

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