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RESEARCH MEMORANDUM

STALL AND FLAME-OUT RESULTING FROM

FIRING OF ARMAMENT

By J. Howard Childs, Fred D. Kochendorfer,
Robert J. Lubick, and Robert Friedman

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NATIONAL ADVISORY COMMITTEE
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STALL AND FLAME-OUT RESULTING FROM FIRING OF ARMAMENT

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SUMMARY

An analysis is presented of the causes of compressor stall and flame-out when armament is fired during flight at high altitudes. Experimental data are also presented on this subject.

The increase in compressor-inlet temperature during armament firing is probably the most important single factor affecting engine performance. This increase in temperature is sufficient by itself to account for the observed occurrences of compressor stall and flame-out.

The changed compressor-inlet pressure, inlet-flow distortions, and combustibles in the compressor, for the most part, increase the likelihood of compressor stall beyond that for an inlet-temperature increase alone.

If the combustible materials entering the engine inlet do not burn until they reach the combustor, their effect will be very small. Also, the reduction in oxygen concentration is not sufficient to affect combustor performance appreciably.

The principal change occurring in the combustor during armament firing is the great increase in fuel-air ratio due to the reduced compressor air flow. In some engines, this increase in fuel-air ratio may be enough to cause a flame-out before compressor stall occurs. However, for the particular engine analyzed in this report, it appears that compressor stall precedes flame-out.

Measures to alleviate these engine difficulties during armament firing include all the features of variable engine geometry that increase the margin between the compressor operating point and the stall limit. A reduction in fuel flow during armament firing will also decrease the likelihood of compressor stall and should prevent combustor flame-out as long as stall does not occur. However, the best solution to the problem is to move the armament away from the engine inlets so that the hot gases never enter the engine.

INTRODUCTION

Compressor stall and flame-out in turbojet engines have occurred on numerous occasions when rocket missiles and cannon were fired at high altitudes. The U.S. Air Force has reported stall and flame-out in the F-86 and the F-94C, and the U.S. Navy in the Cutlass and the Fury. In addition, the British have encountered this problem in their Swift and Hunter airplanes. Obviously, then, the problem is rather general in nature and is not one that is peculiar to any one aircraft or to any one engine. 3412

One factor that obviously contributes to these engine difficulties is the ingestion of rocket and cannon-shell exhaust gases into the engines. The extent to which rocket exhaust can enter the engine air intakes is illustrated in figure 1, which shows a photograph of an F-94C aircraft approximately 0.7 second after firing rockets. The exhaust smoke and vapor trails from the rockets envelop a large part of the airplane and can enter the air intakes in appreciable quantities.

The following engine-inlet effects will occur during the firing of armament:

(1) Increased inlet temperature

(2) Changed inlet pressure

(3) Distorted inlet-pressure and temperature profiles

(4) Entry of combustibles into engine

(5) Reduced oxygen content in gases entering engine

The object of this paper is to examine each of these things that occur when armament is fired and to show by analysis and by experimental data which of these items are important and how each affects engine performance. Deductions are made of the causes of the compressor stall and flame-out that have been encountered in flight. Finally, some remedial measures are suggested.

EFFECTS AT ENGINE INLET

Rocket Firing

Analysis. - The magnitude of the temperature and pressure effects at the engine inlet cannot be obtained directly from existing data. Several sets of data on jet spreading and mixing must be considered together.

For the 2.75-inch air-to-air rocket (fig. 2), the combustion-chamber pressure is 1100 pounds per square inch and the temperature is 4500° R. The design exit Mach number is 2.7 and the exit static pressure is 42 pounds per square inch. Since this pressure is considerably above ambient pressure at altitude, the jet will expand greatly upon leaving the nozzle. The amount of initial expansion can be obtained directly from existing data (ref. 1), but the amount of mixing farther downstream cannot. Experimental data on jet mixing are presented in reference 2. These mixing data were obtained with low velocities for both the jet and the stream; in addition, the two streams had equal temperatures and static pressures. These conditions are in marked contrast to the high velocities and the temperature and pressure differences noted for the rocket. Nevertheless, an estimate of rocket-jet spreading can be obtained by adding the results of reference 2 to the supersonic expansion, as indicated by the diagram in figure 2.

Results of a typical calculation are shown in figure 3. The stream Mach number relative to the launching station is taken to be 0.9 and the altitude 45,000 feet. At the time shown (about 0.3 sec after firing), the rocket is 60 feet from the launcher and is moving away at a speed of 400 feet per second. Contours of temperature are shown for the exhaust from a single rocket; these temperatures are expressed as the difference between the temperature at various locations and the ambient temperature.

If the position of the inlet relative to the launcher is known, temperature increments at the engine inlet can be estimated. For example, consider an inlet 1 foot in diameter whose centerline is spaced 1.5 feet from that of the rocket. If the inlet is in the plane of the launcher, the average temperature of the entering stream would be about 200° F above ambient. The temperature increment will vary from 260° F at the inner face to 140° F at the outer, as indicated by the inset in figure 3.

If the calculations represented in figure 3 are repeated for other lengths of time after firing, the temperature profiles at any station can be obtained as a function of time. Temperature variations at the launcher station are shown in figure 4. Variations in total pressure are also shown in figure 4, and these pressures are expressed as the difference between the local pressure and the free-stream total pressure divided by the free-stream total pressure. For the example cited of an inlet 1 foot in diameter and located 1.5 feet from the rocket centerline, a maximum temperature increase of 400° F is reached at the inner face of the inlet 0.15 second after firing. At this time, there is a variation in temperature across the inlet of 400°. A maximum total-pressure increase of 25 percent above free-stream pressure is experienced at a slightly later time after firing, 0.22 second. At this time, the total-pressure variation across the inlet is 25 percent. Figure 4 also shows the duration of these increases. A temperature increase of 200° at the inner face, for example, will be reached at 0.09 second and will persist until 0.32 second after firing.

Experiment. - The only available experimental data showing the magnitude of increase in inlet temperature when firing rockets were obtained by Lockheed for the F-94C fighter firing a charge of 24 rockets during a 0.15-second time interval. Figure 5 shows the relative position of the rockets and the engine-inlet ducts in the F-94C. The rocket cluster is located in the nose section, while the fuselage scoop-type air inlets are located just rearward. Because of this positioning, it can be expected that more rocket exhaust will be diverted into the inlet duct and higher temperatures will result than for the sample inlet indicated in figures 3 and 4.

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The actual temperature rise measured by Lockheed (ref. 3) is presented in figure 6, which shows the variation in temperature measured at the engine inlet as a function of time elapsed after firing the rockets. Three curves represent the minimum, average, and maximum temperature rise where peak temperatures are approximately 300° , 600° , and 900° F, respectively. One reason for the wide variation in temperature rise lies in the large difference in burning characteristics of the individual rockets. Ignition lag may vary from 0.025 to 0.035 second and total burning time from 1.4 to 2.0 seconds. For a burst of 24 rockets, these variations can result in a large range of possible temperatures in the wake of the rockets.

The approximate time during which these hot gases pass into the engine (fig. 6) agrees quite well with the calculations presented earlier. The hot gases start to pass into the engine about 0.1 second after the beginning of rocket firing and continue to enter the engine until about 0.4 second after the beginning of firing. The values of temperature increase are higher than would be predicted from figure 4, because a large number of rockets were fired and because the nose section tends to divert the exhaust gases into the inlets.

Cannon Firing

The effects of cannon fire are different in certain respects from those for rockets. Figure 7 shows an aircraft in the process of firing cannon. As the muzzle gases move out ahead of the inlet, they mix with the incoming air and the inlet temperature is increased. In certain cases, the gun chambers are vented into the inlet; this permits a considerable blast of hot gases directly into the engine. An additional factor is muzzle flash, a sudden burning or explosion of the gases ahead of the guns. If this occurs, the hot gases can expand ahead of the aircraft and enter the engine at greater than normal rates.

In addition to the temperature effects, cannon firing can alter inlet pressures. Pressure effects for cannon should be in a direction opposite to that noted for rockets, because the momentum of the muzzle gases opposes that of the incoming air. Mixing effects are even more difficult to evaluate than those for the rocket. However, it has been estimated that, at a Mach number of 0.9 and an altitude of 45,000 feet, firing four 20-millimeter guns at the rate of 1500 rounds per minute could reduce inlet pressures by as much as 9 percent if the guns were located close beside the inlet.

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EFFECTS ON COMPRESSOR

Analysis

Increased inlet temperature. - Figure 8 is a compressor operating map for a typical single-spool axial-flow turbojet engine. The map shows the variation of compressor pressure ratio with corrected air flow for lines of constant corrected engine speed. Also shown are the steady-state operating line and the stall line for no inlet-flow distortion. Calculations of the point of compressor operation have been made for an increase in inlet temperature. The initial point of operation (point A) was taken at rated mechanical engine speed at an altitude of 45,000 feet and a flight Mach number of 0.9. For these calculations, fuel flow and engine speed were assumed to remain constant while the inlet temperature increased. This is a valid assumption, since, during the short time (approximately 0.3 sec for an actual rocket-firing case), the engine control would not have time to adjust fuel flow nor would engine speed have time to change.

To completely analyze compressor operation during an inlet-temperature increase would necessitate calculating the history of the compressor operating point as the armament gases pass through the engine. However, with only the operating map for the complete compressor, this calculation is impossible. The discussion must therefore be limited to the operating point that is reached after the armament gases reach the turbine station (point B, fig. 8). The location of point B depends on two factors: (1) The compressor equivalent speed $N/\sqrt{\theta}$ must be corrected for the increased value of θ; and (2) the reduced equivalent speed produces a reduced air-flow rate, and, since fuel flow remains constant during the short time involved, the turbine-inlet temperature rises. Point B is therefore located on a line of lower equivalent speed and is shifted upward from the steady-state operating line. (Symbols are defined in the appendix.)

Since the path followed in the transient from points A to B cannot be calculated, the dashed curve in figure 8 serves only as an indication of one possible path. The obvious question as to what happens if the path enters the stall region cannot be answered at present. Since time is required for stall to set in, it is possible that some stages of the compressor could momentarily operate above their steady-state stall limit. On the other hand, if the path crosses the stall-limit line and stall does occur, then operation at point B will not be realized.

Compressor operating points calculated by this procedure for several values of inlet-temperature increase are presented in figure 9. A 300° increase results in an operating point just at the stall line. For this particular engine, an inlet-temperature increase of slightly over 300° F would certainly cause compressor stall. 3412

Changed inlet pressure. - If a pressure change accompanies the temperature change, the stall tendency can be affected. Three paths between points A and B are shown in figure 10. Assume that the center path would be followed for a temperature change alone. Now, if the inlet pressure is reduced as a result of cannon fire, the reduction in compressor-inlet pressure effectively increases the compressor pressure ratio. The path will therefore be shifted upward toward the stall limit. The amount of shift will, of course, depend on the magnitude and duration of both the temperature and pressure change. The pressure effects of cannon fire should, in general, increase the probability of stall.

For rockets, the inlet pressure is increased and the compressor pressure ratio is momentarily decreased. This moves the path downward away from stall.

Inlet-flow distortions. - As shown in the jet-spreading analysis, both temperature gradients and pressure distortions may be expected after armament firing. Pressure distortions effectively lower the stall limit. Temperature distortions may also have a similar effect. The reduction in stall margin due to these changes is indicated in figure 11. In many instances, armament firing takes place while the aircraft is at high angle of attack. The inlet-flow distortions caused by angle of attack also affect the stall margin as indicated in figure 11.

Combustibles in compressor. - Numerous instances of afterburning from rockets and muzzle flash from guns serve to show that the armament gases can undergo further burning. If burning occurs while these gases are in the compressor, then the pressure will rise where the burning takes place. This means that the pressure ratio across the compressor stages upstream of this location will be greatly increased. The resultant effect cannot be shown on the compressor map, but the probability of encountering stall is increased.

Experiment

Lockheed (ref. 3) reports that a 300° increase in inlet temperature is the critical value in the F-94C, which uses a J48 engine. They state that a temperature increase of less than 300° can be tolerated, while a temperature increase in excess of this value causes engine difficulties. These temperature effects were probably accompanied by inlet-flow distortion and pressure changes; consequently, the 300° value is probably unique for the F-94C configuration.

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Experimental data showing the effect of an increase in inlet temperature on a modern turbojet engine are presented in figure 12 which shows that surge was produced by inlet-temperature increases of 70° F or more. These data were obtained in the Lewis altitude wind tunnel by deflecting hot air into the engine inlet. A time interval of 1 to 2 seconds was required for the complete temperature rise to be felt at the compressor inlet during these tests; consequently, these data are not strictly analogous to the more rapid temperature increase accompanying armament firing.

EFFECTS ON COMBUSTOR

Before considering the effects of armament on the combustor, the factors that affect combustor performance will be reviewed briefly. A combustor performance map is shown in figure 13. Combustion efficiency is plotted as a function of the combustion parameter $\alpha \frac{P_3 T_3}{V_r} \varphi(\alpha)$, where α is the oxygen concentration in the inlet gases, P_3 and T_3 are the total pressure and temperature at the combustor inlet, V_r is the combustor reference velocity and is equal to the inlet volume flow rate divided by the maximum cross-sectional area of the combustor, and $\varphi(\alpha)$ is an exponential function that depends primarily on the oxygen concentration. The combustion parameter was derived from theory by assuming that chemical reaction kinetics control the rate of burning in the combustor (refs. 4 and 5). For most combustors, a reasonable correlation of data can be obtained with the parameter.

A typical family of experimental curves is shown in figure 13 for several fuel-air ratios. For lower values of the combustion parameter, efficiency decreases sharply. The solid points at the ends of the curves denote flame-out. The value of the combustion parameter at which flame-out occurs is a function of fuel-air ratio, as shown in figure 14. Each of the solid points indicates an experimental flame-out for a typical turbojet combustor. Since flame-outs are not exactly reproducible, a shaded band is used to indicate the flame-out region. Above this band, combustion is stable; below, no burning is possible.

To facilitate estimation of flame-out margin for this combustor, operating points are indicated at several altitudes. These points correspond to rated engine speed and a flight Mach number of 0.9. This combustor obviously has ample steady-state flame-out limits.

The combustor operating map can now be used to analyze the effect of armament firing on combustor performance in a hypothetical turbojet engine that uses the combustor of figure 14 and the compressor of figure 9. Consider an initial operating point at an altitude of 45,000 feet, rated mechanical speed, and a flight Mach number of 0.9; this is point A on the combustor map (fig. 15). If the armament firing causes a 300° F increase in compressor-inlet temperature, then the combustor operating conditions become those of point B (fig. 15). The location of point B depends on both the change in inlet temperature and gas composition and the change in compressor operating point. On the compressor map (fig. 9), it was shown that a 300° F increase in compressor-inlet temperature causes a marked decrease in both compressor pressure ratio and mass-flow rate. The decreases in oxygen content and in combustor-inlet pressure adversely affect the combustor; however, these effects are largely offset by the increase in inlet temperature and the decrease in air-flow rate. Thus, the change in the combustion parameter between points A and B is slight. However, the fuel-air ratio increases considerably from point A to point B, since fuel flow remains constant and air flow decreases. For the combustor map illustrated in figure 15, point B lies in the stable burning region, and flame-out does not occur.

As indicated by figure 9, the operating point corresponding to a 300° F increase in inlet temperature lies just underneath the compressor stall line. If the compressor stalls at this condition, then the combustor operating point changes to the values indicated by the two points labeled C in figure 15. The upper point corresponds to the highest and the lower point to the lowest of the oscillating pressures that have been measured in a similar turbojet engine operating in a stalled condition at this particular value of equivalent speed (ref. 6). Since these operating points for the combustor during stall lie outside the stable burning region, combustor flame-out will follow the occurrence of compressor stall.

The foregoing discussion does not mean that in all engines compressor stall must precede combustor flame-out. Obviously, if this same sequence of events occurred in some engine for which the combustor did not have as wide a fuel-air-ratio range for stable operation, combustor flame-out could occur at point B, because the fuel-air ratio would be too high even though the compressor did not stall.

The possibility of the entry of some combustible material into the engine due to incomplete combustion in the armament exhaust gases has been mentioned. The effect of these materials passing through the compressor without burning and then burning in the combustor, is shown in figure 16.

For a combustion efficiency in the armament gases of 85 percent, and the remaining 15 percent of their chemical heat release assumed to occur in the combustor, point B would be moved over to the location indicated by point D. The effect is quite small.

EFFECT ON SUPERSONIC AIRCRAFT

At supersonic speeds, an additional component, the air inlet, must be considered. If reduced air flow accompanies the temperature increase, a supersonic inlet can be forced into its subcritical operating range, and the inlet then becomes an additional source for pressure and flow pulsations. The effects of firing rocket exhaust into an inlet-engine combination operating at a Mach number of 1.9 have been briefly investigated in the Lewis 8- by 6-foot supersonic wind tunnel. Although quantitative data are not available as yet, it is known that large pressure fluctuations existed at both the compressor inlet and outlet stations. Because of compressor surge or inlet instability, a quasi-steady operating condition was never reached.

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REMEDIAL MEASURES

Possible remedial measures are as follows:

Transient adjustments:
- Close inlet guide vanes
- Open compressor bleeds
- Reduce fuel flow
- Inject water at compressor inlet
- Open engine exhaust nozzle

Design changes to avoid intake of exhaust gases:
- Move armament away from inlet
- Vent gun chambers away from inlet
- Deflect muzzle gas away from inlet

The first course of action is the use of transient adjustments, such as closing the inlet guide vanes, opening the compressor bleeds, reducing fuel flow, and injecting water at the compressor inlet. These are all measures that can be put into effect an instant before armament is fired and maintained during the critical period. The length of time the hot gases are passing into the engine is approximately 0.3 second, so that the use of these adjustments may be limited by the speed of actuation. Unfortunately, all these items (except water injection) also appreciably decrease the engine thrust level, and this effect on performance must be considered. F. A. Holm stated that reducing fuel flow during rocket firing on the F-94C was effective (I. A. S. meeting, Cleveland, Ohio, March 11, 1955).

The compressor operating margin between the steady-state line and the stall limit decreases as altitude is increased. The gains resulting from design changes to increase this stall margin will be largely taken up by future increases in flight altitude. In addition, angle-of-attack operation decreases the stall margin, and the size and firing rate of armament are being steadily increased. From these considerations it would appear that the use of transient adjustments to increase the stall margin may prove to be only a temporary solution to the problem of armament firing.

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The best course of action, and the most obvious, is to avoid completely the intake of exhaust gases. Moving the armament, venting the gun chambers, and deflecting muzzle gas away from the inlet are all possible solutions. The problem for the F-86F has been greatly alleviated by installing blast deflectors on the nose-mounted cannon. It is noteworthy that the F-89, which has wing-tip-mounted rocket pods, has encountered no engine problems due to rocket firing, according to Holm.

CONCLUSIONS

The increase in compressor-inlet temperature during armament firing is probably the most important single factor affecting engine performance. This increase in temperature is sufficient by itself to account for the observed occurrences of compressor stall and flame-out.

The changed compressor-inlet pressure, the inlet-flow distortions, and the combustibles in the compressor, for the most part, increase the likelihood of compressor stall beyond that for an inlet-temperature increase alone.

If the combustible materials entering the engine inlet do not burn until they reach the combustor, their effect will be very small. Also, the reduction in oxygen concentration is not sufficient to affect combustor performance appreciably.

The principal change occurring in the combustor during armament firing is the greatly increased fuel-air ratio due to the reduced compressor air flow. In some engines, this increase in fuel-air ratio may be enough to cause a flame-out before compressor stall occurs. However, for the particular engine analyzed here, it appears that compressor stall precedes flame-out.

Measures to alleviate these engine difficulties during armament firing include all the features of variable engine geometry that increase the margin between the compressor operating point and the stall limit. A reduction in fuel flow during armament firing will also decrease the likelihood of compressor stall and should prevent combustor flame-out as long as stall does not occur. However, the best solution to the problem is to move the armament away from the engine inlets so that the hot gases never enter the engine.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, June 1, 1955

APPENDIX - SYMBOLS

The following symbols are used in this report:

M Mach number

N engine rotational speed

P total pressure

T total temperature

V_r combustor reference velocity

W_a air flow

α oxygen concentration in inlet gases

δ ratio of total pressure to absolute pressure of NACA standard sea-level conditions

θ ratio of total temperature to absolute temperature of NACA standard sea-level conditions

$\varphi(\alpha)$ exponential function depending primarily on oxygen concentration

Subscripts:

c rocket combustion chamber

e rocket-nozzle outlet

0 free stream

2 compressor inlet

3 compressor outlet

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REFERENCES

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5. Graves, Charles C.: Effect of Oxygen Concentration of the Inlet Oxygen-Nitrogen Mixture on the Combustion Efficiency of a Single J33 Turbojet Combustor. NACA RM E52F13, 1952.

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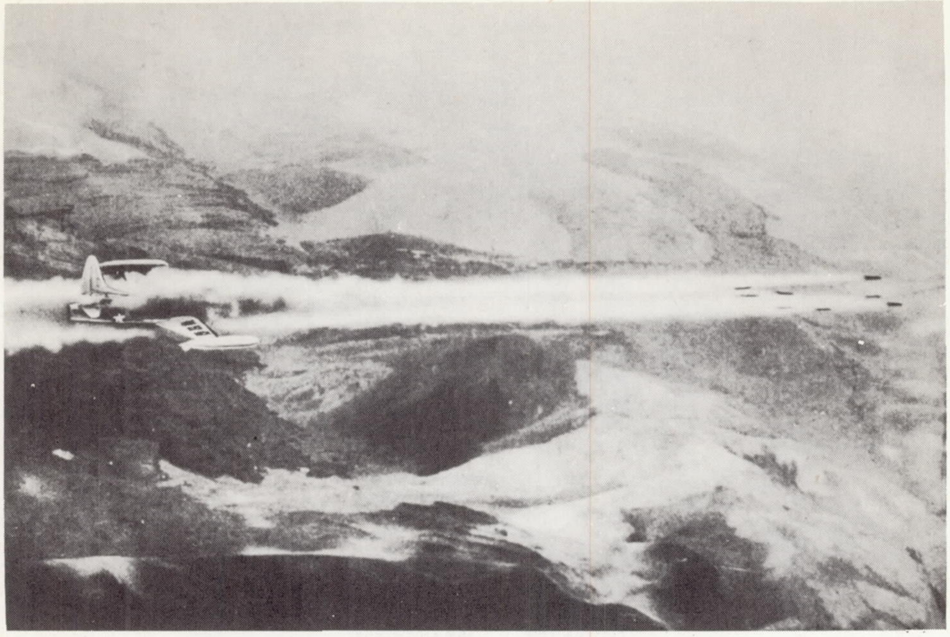

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Figure 1. - F-94C firing rockets. (Obtained from Lockheed Aircraft Corp.)

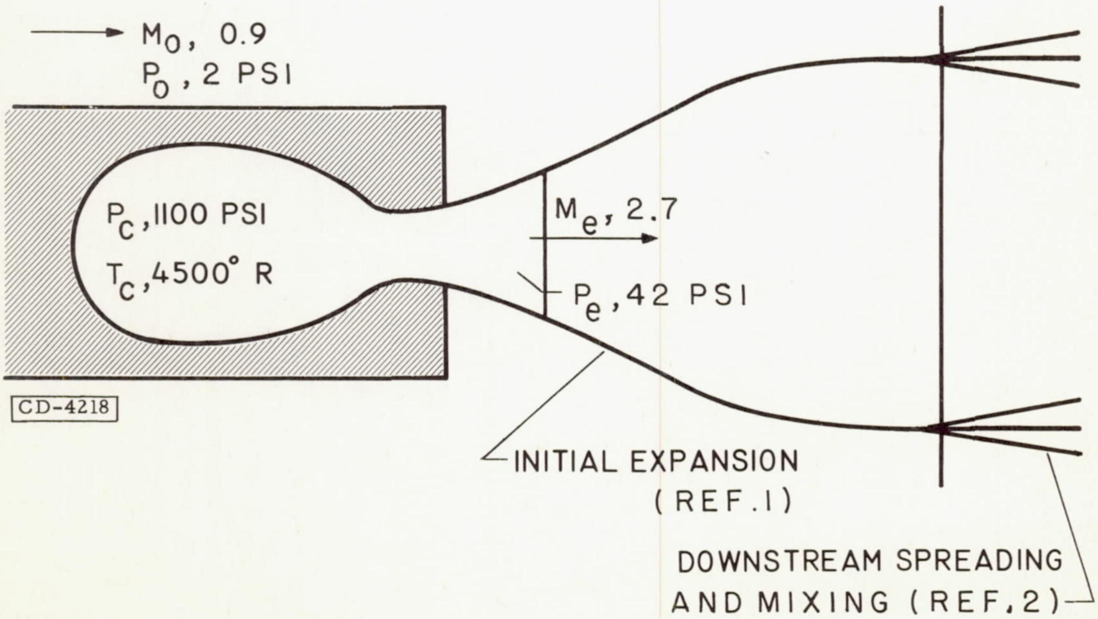

Figure 2. - Jet spreading from 2.75-inch rocket. Altitude, 45,000 feet.

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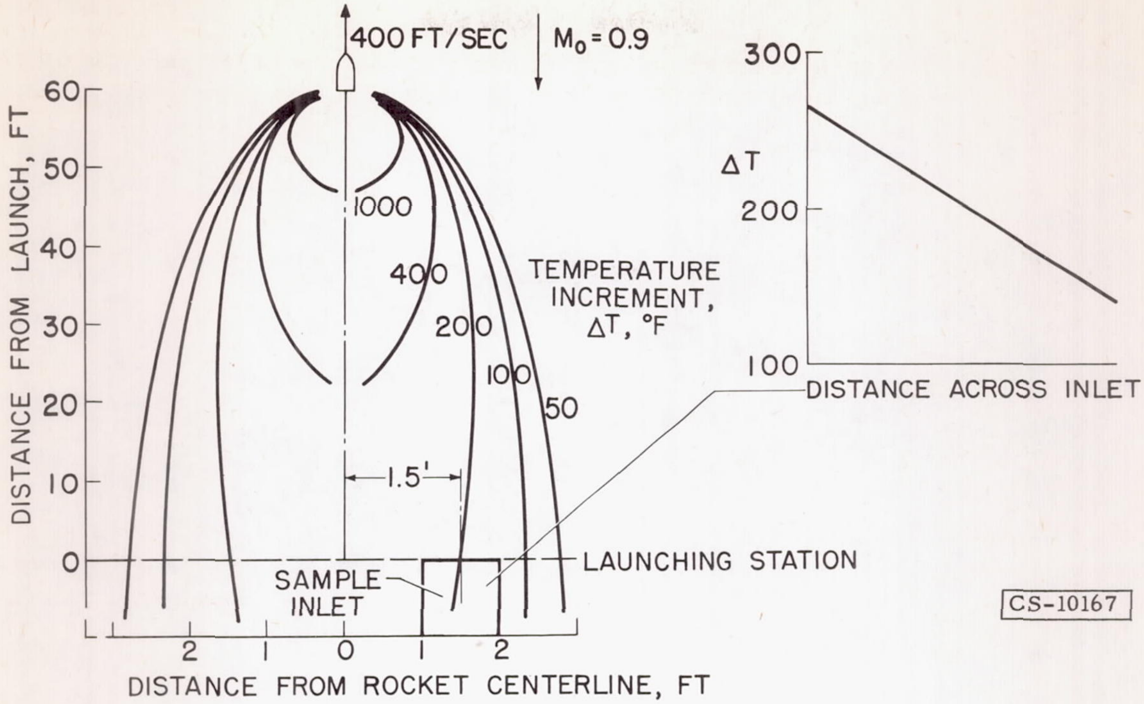

Figure 3. - Temperature contours for 2.75-inch rocket. Altitude 45,000 feet; time after firing, 0.3 second.

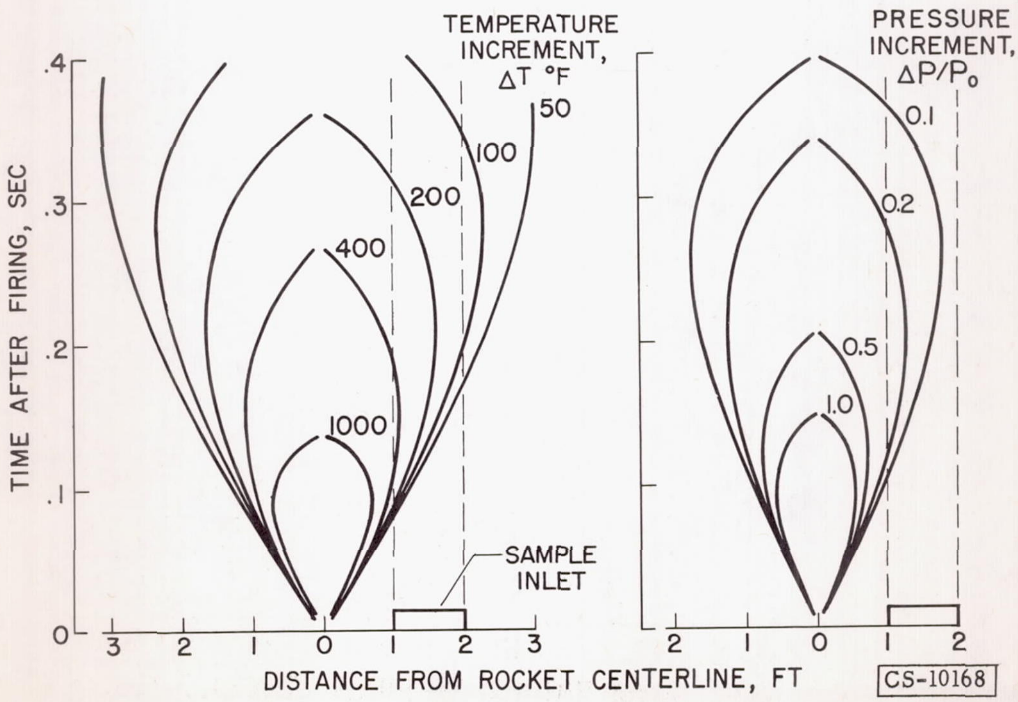

Figure 4. - Pressure and temperature increments for 2.75-inch rocket. Free-stream Mach number, 0.9; altitude, 45,000 feet.

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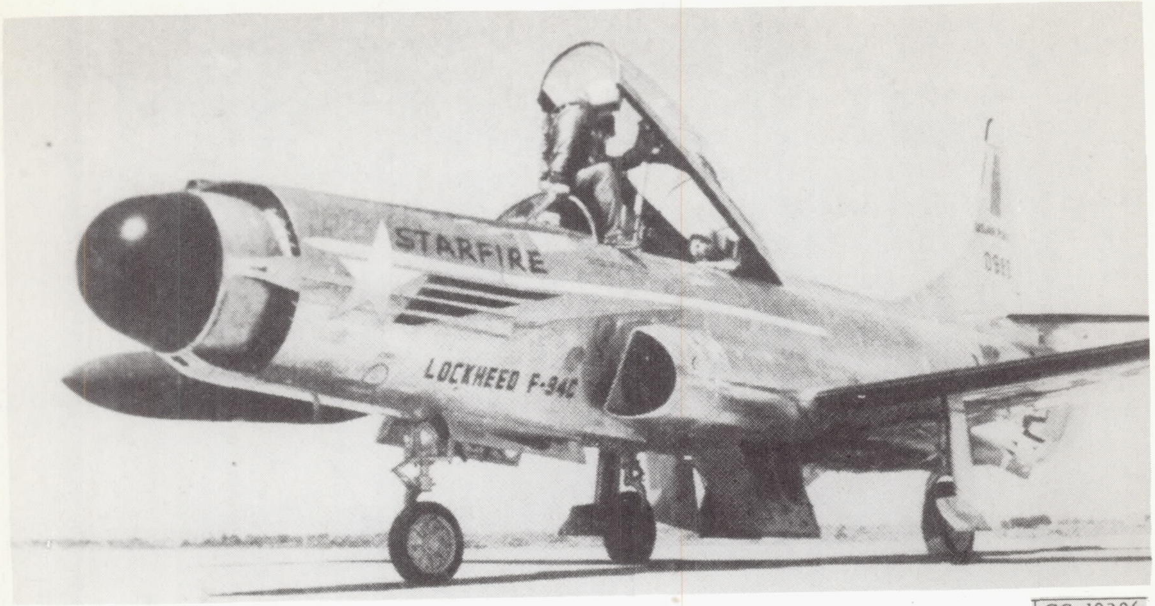

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Figure 5. - Rocket installation for F-94C.

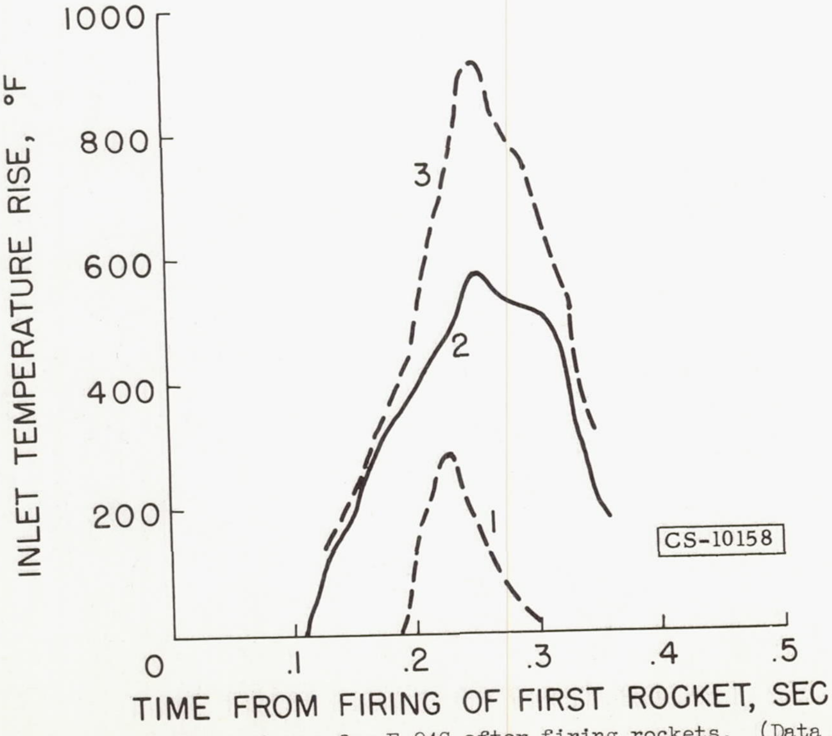

Figure 6. - Inlet temperatures for F-94C after firing rockets. (Data from ref. 3).

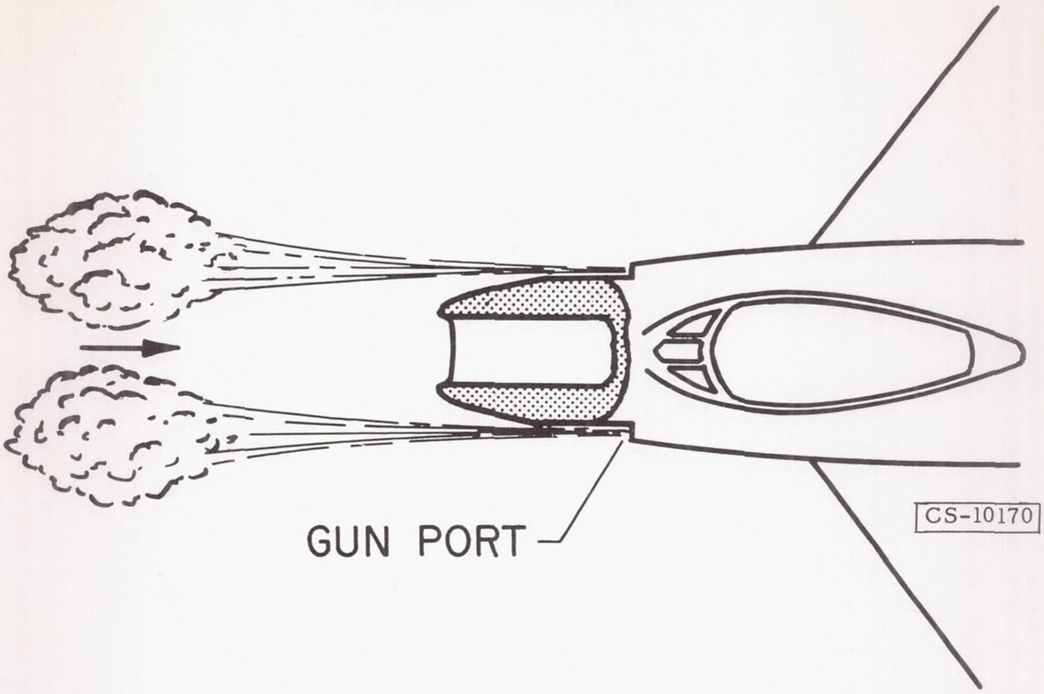

Figure 7. - Cannon blast effect.

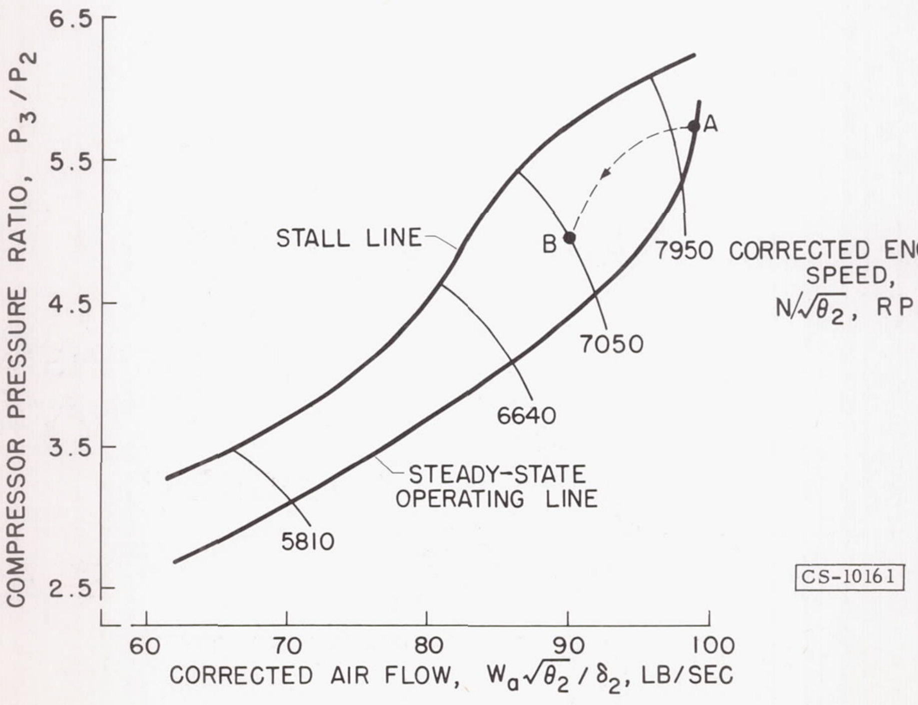

Figure 8. - Theoretical compressor operation for instantaneous inlet-temperature increase.

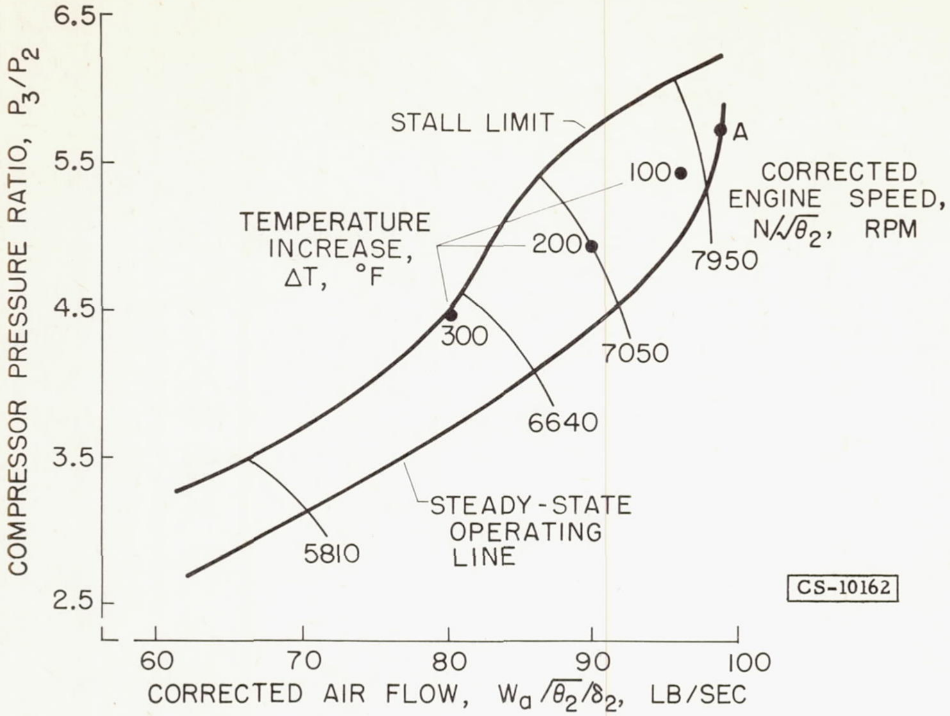

Figure 9. - Calculated effect of various inlet-temperature increases.

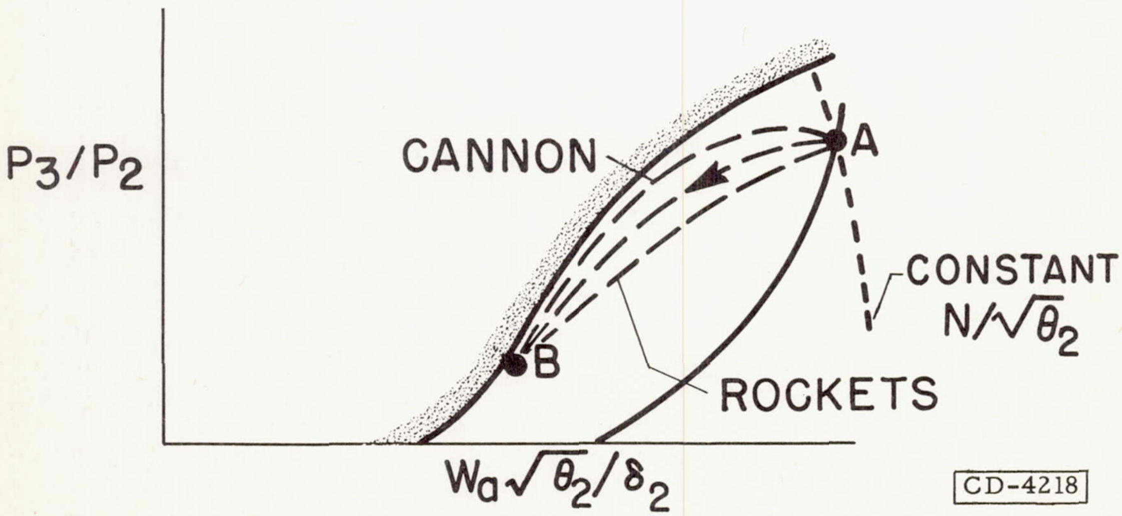

Figure 10. - Effect of changed inlet pressure.

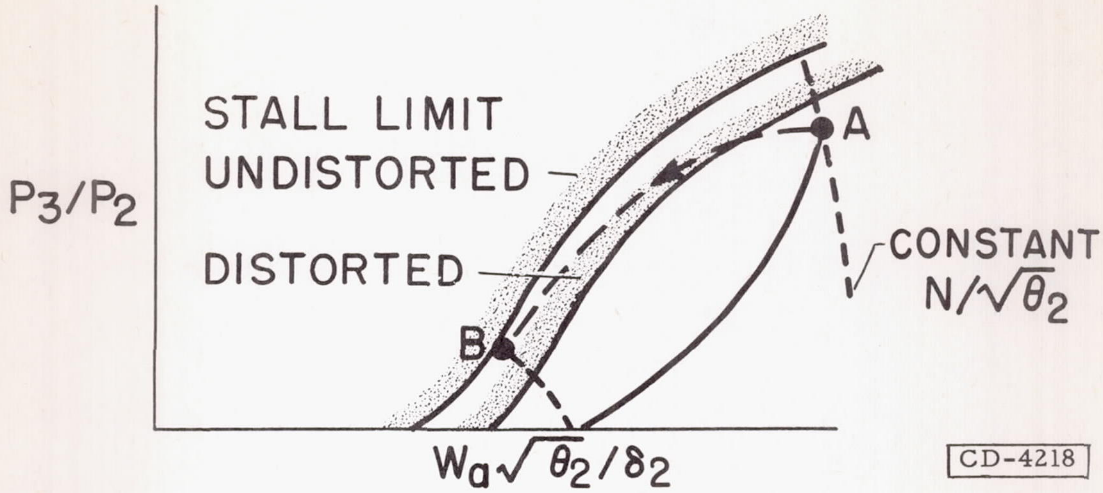

Figure 11. - Effect of inlet distortions.

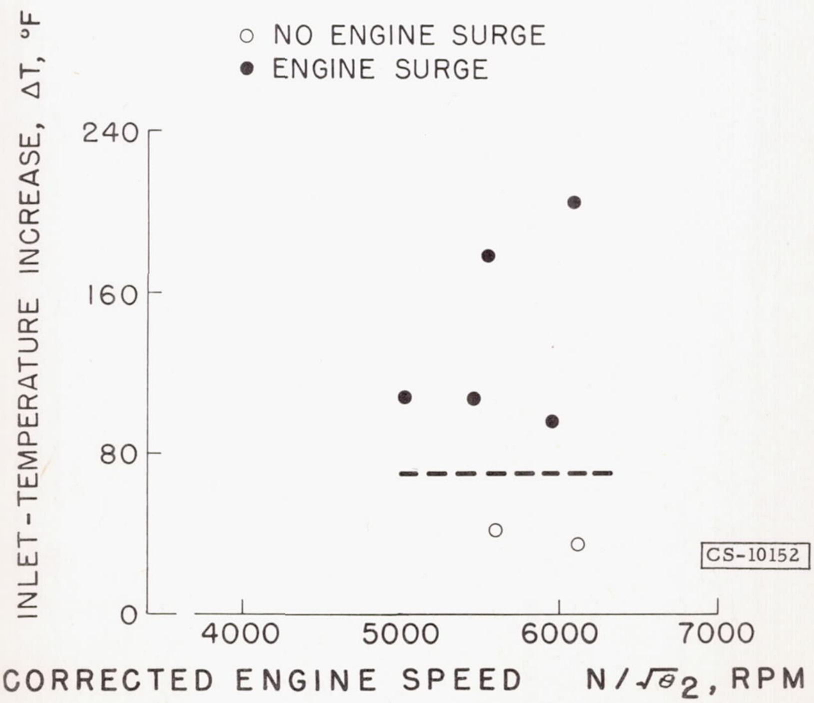

Figure 12. - Experimental effect of various inlet-temperature increases.

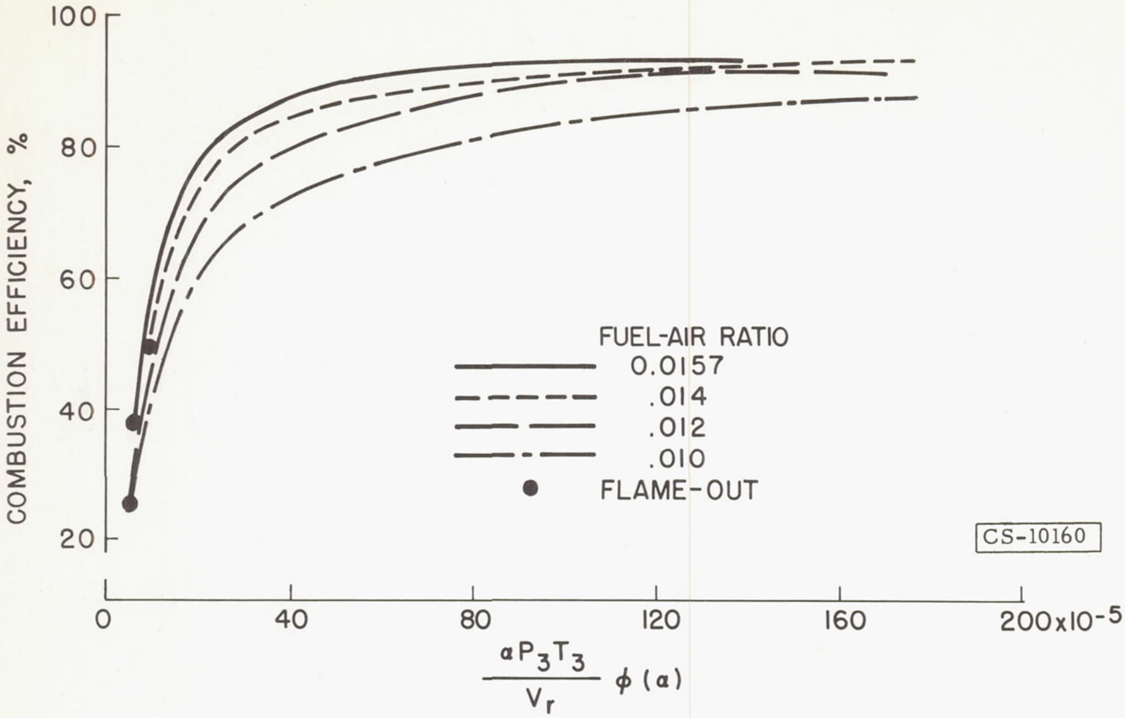

Figure 13. - Combustor operating lines for several fuel-air ratios.

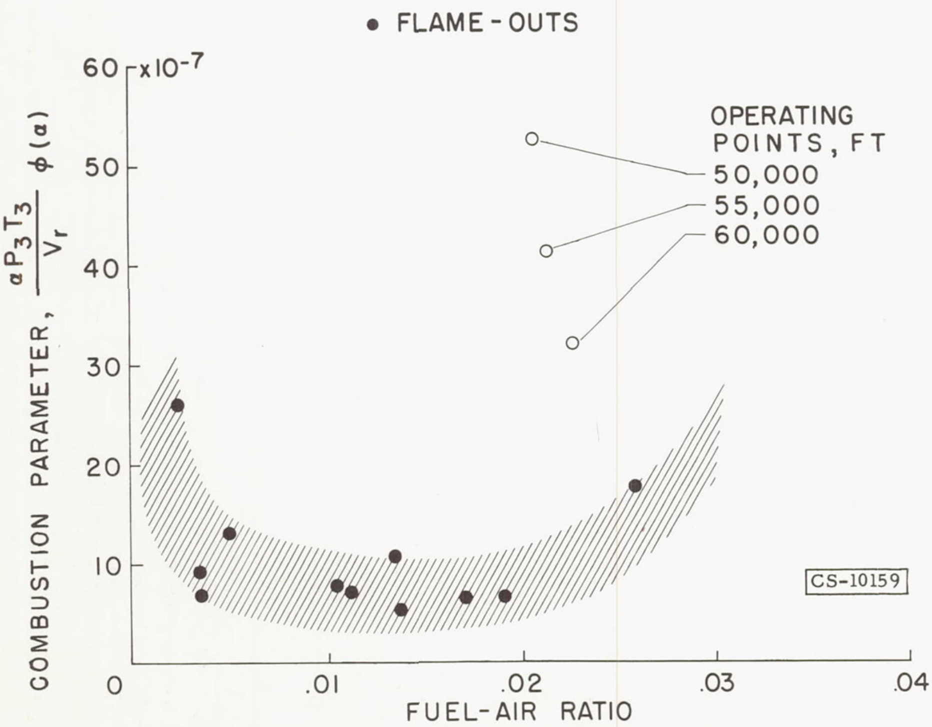

Figure 14. - Combustor flame-outs.

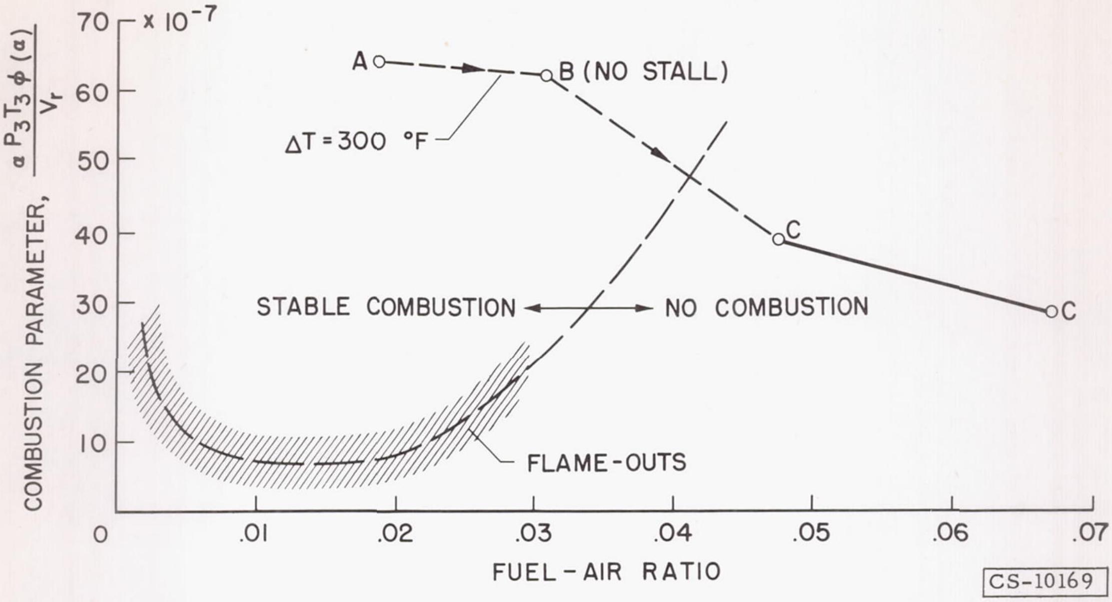

Figure 15. - Combustor conditions during rocket firing.

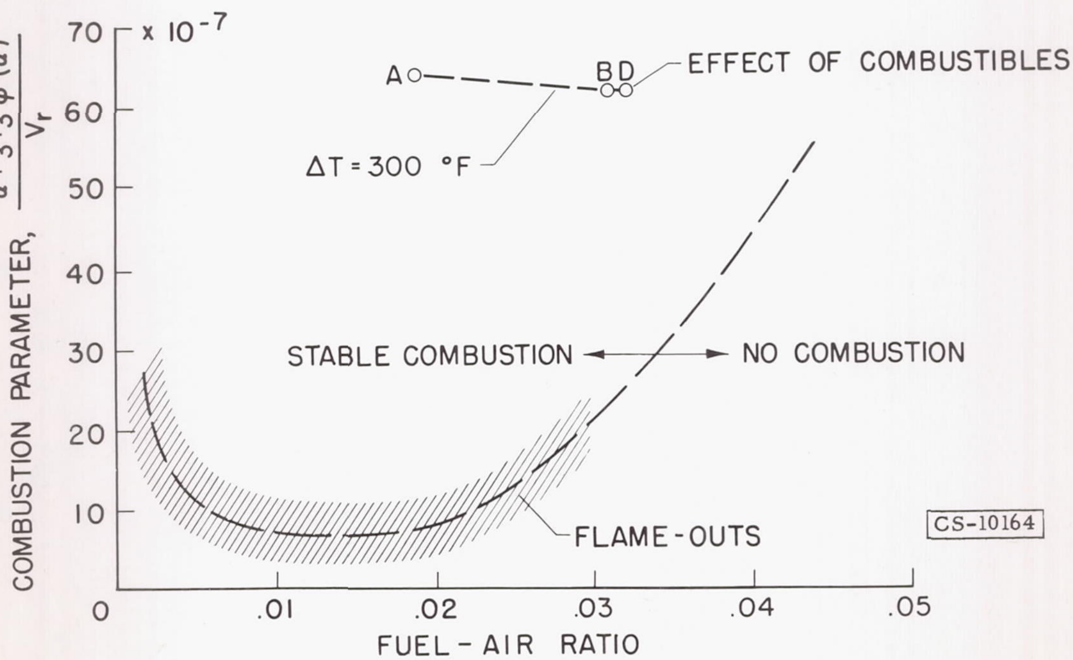

Figure 16. - Effect of armament gases burning within combustor.