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RESEARCH MEMORANDUM

AN EXPERIMENTAL INVESTIGATION OF THE AIR-FLOW STABILITY OF A SCOOP-TYPE NORMAL-SHOCK INLET

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AN EXPERIMENTAL INVESTIGATION OF THE AIR-FLOW STABILITY OF A SCOOP-TYPE NORMAL-SHOCK INLET

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SUMMARY

An experimental investigation has been made of the air-flow stability of scoop-type normal-shock inlets located on the fuselage of a model of an interceptor or fighter-type airplane. The pressure recovery and amplitude of the pressure pulsations were measured at Mach numbers of 0.80, 1.30, 1.55, and 1.90 for mass-flow ratios from 0.2 to the maximum obtainable.

Both the original and the modified inlets incorporated wedge-type boundary-layer bleed diverters in conjunction with a splitter plate to provide a gutter for removing fuselage boundary-layer air. The original air-induction system had an included diverter wedge angle of 130°, a short splitter plate, and a slight undercut in the fuselage ahead of the inlet. The included diverter wedge angle of the modified inlet was reduced to 65° while the splitter-plate length was increased to approximately three times that of the original inlet. In addition, the fuselage undercut ahead of the inlet was eliminated. It was found that these modifications reduced greatly the severe air-flow instability of the original air-induction system.

When pulsating flow occurred in either the original or modified inlet, the resulting pressure fluctuations were random and had a maximum frequency of about 400 to 600 cycles per second. Amplitudes of the pressure fluctuations as large as 22 percent of the free-stream total pressure were measured in the original inlet, while pulsations of smaller amplitude (less than 6 percent of the free-stream total pressure) were measured in the modified inlet.

The modified inlet showed a significant increase in pressure recovery, over the original inlet, at supersonic speeds. For both the original and the modified inlets, at relatively high mass-flow ratios, the magnitude of the flow pulsations increased for Mach numbers above 1.3.

At Mach numbers above 1.3 the pressure rise through a normal shock is sufficient to separate a turbulent boundary layer.

INTRODUCTION

With external shock-type inlets, rapid flow pulsations are usually encountered as the mass-flow ratio is reduced below its maximum value. Several explanations have been proposed to describe the mechanism of this air-flow instability or "buzz," and the "triggering" force necessary for its start (refs. 1 through 6). Two main "triggering" forces are now known to incite flow instability: (1) separation associated with shock-wave boundary-layer interaction, and (2) the Ferri-Nucci vortex sheet (refs. 3 and 6). The amplitude of flow instability on scoop inlets can be largely a function of the particular installation, thus, an experimental investigation was made of the stability of flow of a particular air-induction model equipped with two scoop-type normal-shock side inlets. Data were previously reported for similar normal-shock scoop inlets (refs. 7 and 8) but they apparently were not instrumented to show any of the details of the flow pulsations and they utilized boundary-layer suction scoops in contrast to the boundary-layer bleed system of the present inlets. The model investigated in this test was instrumented to record the amplitude and frequency of pulsation of the inlet pressure.

NOTATION

A_1 inlet area, 0.0228 sq ft

F.S. model fuselage station, in. (nose at F.S. 7.43)

H_c average total pressure at the simulated compressor entrance, lb/sq ft

H_o free-stream total pressure, lb/sq ft

h minimum boundary-layer-bleed height at bleed inlet, in.

M Mach number

m mass flow through inlet, ρAV, slugs/sec

m_o mass flow at free-stream conditions passing through area equal to inlet area, $\rho_o A_1 V_o$, slugs/sec

q dynamic pressure, lb/sq ft

T	temperature, °R
V	velocity, ft/sec
α	angle of attack of wing chord measured in a plane perpendicular to the wing chord plane, deg
δ	boundary-layer displacement thickness measured at F.S. 31.36, in.
ρ	mass density of air, slugs/cu ft
φ	included wedge angle of boundary-layer bleed diverter, deg (fig. 4)

Subscripts

o	free stream
1	inlet station (leading edge of upper lip)
c	compressor entrance, F.S. 39.46

APPARATUS AND PROCEDURE

The model on which the inlets were tested is shown mounted in the test section of the Ames 6- by 6-foot supersonic wind tunnel in figure 1. The external longitudinal cross-sectional area distribution of the model is presented in figure 2. Figure 3 shows the variation of the internal duct area with fuselage station. Details of the two inlet configurations are shown in figures 4(a) and 4(b). Note that these figures show the leading edge of the upper lip to be ahead of the leading edge of the lower lip of the inlet, giving the inlets a negative incidence relative to the fuselage reference plane. Pertinent differences between the original and modified inlets are given in the following table:

Original	Modified
1. Fuselage undercut in front of inlet	1. No undercut in front of inlet
2. $\varphi = 130^\circ$	2. $\varphi = 65^\circ$
3. Splitter-plate length = 0.318 in.	3. Splitter-plate length = 0.905 in.
4. Distance from apex of diverter to leading edge of splitter plate = 0.58 in.	4. Distance from apex of diverter to leading edge of splitter plate = 0.58 in.

(The internal duct of the modified inlet was 0.88 inch longer than that of the original installation. Preliminary tests showed lengthening of the duct to have a negligible effect on the duct characteristics.)

Mass flow through the air-induction system was measured by a survey rake at the simulated exit of the tail pipe. The survey rake contained 13 total-pressure tubes spaced at the center of equal areas and 4 static-pressure tubes. Mass-flow ratio was varied by inserting different size flow restrictor plates between the compressor inlet and the exit of the tail pipe. The locations of the rake for measuring the pressure recovery and the pressure cell for measuring the duct pressure pulsations are given in figure 5. The pressure cells are of the strain-gage type and have response invariant with frequency from 0 to 10,000 cycles per second. However, the carrier current amplifier and the recording oscillograph apparatus reduced this linear frequency range for the over-all instrumentation from 0 to approximately 500 cycles per second. Values of the maximum total amplitude of the pressure pulsations were obtained from pressure time records of the strain-gage pressure cell mounted in the duct system. A typical record is shown in figure 6.

Experimental data were obtained for the two inlet configurations over a range of mass-flow ratios at Mach numbers of 0.80, 1.30, 1.55, and 1.90 at a Reynolds number of 3×10^6 per foot for 0° angle of attack.

The tests were conducted in the Ames 6- by 6-foot supersonic wind tunnel. A description of this wind tunnel is given in reference 9.

RESULTS AND DISCUSSION

Duct Air-flow Stability

Throughout the mass-flow-ratio range of the test, the original inlet configuration exhibited severe pressure pulsations at all supersonic speeds. The maximum total amplitude of the pressure pulsations increased with decreasing mass-flow ratio (see fig. 7) and had a maximum magnitude of 22 percent of the free-stream total pressure at a Mach number of 1.55. Even at the highest mass-flow ratios obtainable with these models the pressure fluctuations were from 3 to 12 percent of the total pressure. Examination of the pressure time records showed the pressure pulsations to be random, and to have a maximum frequency of about 400 to 600 cycles per second.

With the original inlet, flow instability was present at all supersonic Mach numbers. The schlieren photographs of figure 8(a) show the details of the flow in the vicinity of the original inlet. At the highest mass-flow ratio, for $M = 1.55$, separation can be seen to occur immediately behind the depression in the contour of the fuselage. As the mass-flow ratio is reduced, the separation point moves upstream, forming a large wedge of separated air flow in front of the inlet. It was believed that the severe disturbances, which are transmitted upstream of the inlet through the fuselage boundary layer, are caused by the blunt wedge underneath the boundary-layer splitter plate (fig. 4(a)), the interaction of the shock waves with the boundary layer, and the additional adverse pressure gradient resulting from the flow over the undercut portion of the fuselage forward of the inlet.

Modifications to the inlet were limited, generally, by the existing airplane structure. Reference 10, which discusses the performance of wedge-type boundary-layer diverters, shows that considerable gains in inlet performance can be realized by reducing the wedge angle of the diverter. Accordingly, the wedge angle was reduced from 130° to 65°. It should be pointed out, however, that even the 65° angle is considered in reference 10 to be excessive. The splitter-plate length was increased also, so that at Mach numbers greater than 1.4 the normal shock would not move ahead of the plate until the mass-flow ratio was reduced below about 0.7. To eliminate the expansion region in front of the inlet, the undercut on the fuselage forward of the inlet was eliminated. Boundary-layer investigations (e.g., ref. 8) have shown that the height of the splitter plate (h) above the fuselage should be approximately the same as the height of the fuselage boundary layer (δ) in the region of the inlet. Measurement of the boundary layer on the modified fuselage with the inlets removed showed that h/δ was 1.0 at a Mach number of 1.3, in the plane of the inlet.

Comparison of the pressure pulsations obtained with the modified and the original inlets (fig. 7) shows a marked reduction for the modified inlet throughout the speed range. From the tests that were made it was not possible to determine what proportion of the improvement in stability was due to each configuration alteration. Schlieren photographs of the modified inlet at $M_o = 1.55$ (fig. 8(b)) indicate that the separated region in front of the inlet is reduced considerably, but not completely eliminated. At Mach numbers of 0.80 and 1.30 the amplitude of the pressure fluctuations, 1 to 2 percent, is low considering the fact that the air-induction system had two bends. For Mach numbers of 1.55 and 1.90 the pulsations were reduced to a maximum of about 6 percent of the free-stream total pressure.

From figures 7 and 9 it can be seen that the slope of the pressure recovery versus mass-flow-ratio curve is not necessarily a satisfactory measure of the amplitude of the flow pulsations. In these tests flow pulsations occurred with both positive and negative slopes, but the amplitude of the pulsations was always less when the slope was negative. It is felt, therefore, that the concept that flow stability occurs when the slope is negative must be considered in relation to the amplitude of the pressure pulsations.

A qualitative correlation has been observed between the magnitude of the flow pulsations and the pressure rise necessary to separate a turbulent boundary layer. Several investigations (refs. 11 through 15) show that the pressure rise necessary for separation of a turbulent boundary layer is about 1.8 for the supersonic Mach number range up to 2.0, and that the pressure rise is relatively insensitive to Reynolds number. Data from these references are presented in figure 10(a) and appear to be grouped near the value of 1.8 as indicated by the straight line. Included in this figure is a curve of the pressure rise occurring across a normal-shock wave as a function of Mach number. The intersection of the two curves at $M = 1.3$ leads to the expectation of separation of the flow and increased flow instability at Mach numbers greater than 1.3. In figure 10(b) the measured maximum amplitudes of the pressure pulsations at a mass-flow ratio of 0.8 are plotted for both the original and the modified inlets. There appears to be a qualitative correlation between the increase in pulsation amplitude above the values for $M_o = 1.3$ with the separation prediction shown in figure 10(a).[1]

For the modified inlet, the base of the normal shock in front of the inlet remains on the splitter plate until the mass-flow ratio is reduced below about 0.7. However, figures 7 and 10(b) show that even this inlet experienced flow instability with pulsation amplitudes as large as 6 percent of the free-stream total pressure at Mach numbers above 1.3 for mass-flow ratios greater than 0.7. For mass-flow ratios below 0.7 at $M = 1.55$ and below 0.5 at $M = 1.9$ the amplitude of the pressure pulsations for the modified inlet was reduced (see fig. 7). This reduction in pressure pulsations is believed to be due to the fact that the normal shock moved off the splitter plate and impinged on the fuselage boundary layer (see fig. 8(b)) and although separation still occurred due to shock-wave boundary-layer interaction, a large portion of the separated air was removed by the boundary-layer bleed system. Separation on the splitter plate may be minimized by reducing the Mach number of the flow through the use of a wedge-type compression surface in place of a splitter plate or perhaps by employing suction through a porous splitter plate.

Pressure Recovery

A comparison of the total pressure recovery at the simulated compressor entrance for the original and the modified inlets is given in figure 9. It should be remembered that whenever there are severe pressure fluctuations, the values of pressure recovery and mass-flow ratio are in error by an undetermined amount. It can be shown also (see ref. 16) that a total-pressure tube in an air stream with a fluctuating velocity will always indicate a pressure higher than the mean pressure. For this

[1]The average local Mach number at the inlet was not measured for the original inlet. However, measurements in the region of the modified inlet showed the Mach number to be near the free-stream value.

reason, most of the total-pressure ratios at $M = 1.55$ and $M = 1.9$ shown in figure 9 are probably too high. However, the values of H_c/H_o for the modified inlet at $M_o = 0.80$ and 1.30 are probably near the mean area-weighted average. At mass-flow ratios less than about 0.7, the pressure recoveries greater than normal shock recovery shown for the modified inlet at Mach numbers of 1.3 and 1.55 result from a bifurcation of the normal shock in front of the boundary-layer splitter plate (see figs. 8(b) and 9). The wedge-type separated region on the fuselage surface, caused by the blunt-wedge diverter under the splitter plate and the shock-wave boundary-layer interaction, produces oblique shock waves which decrease the total pressure losses of a portion of the air entering the inlet.

Presented in figure 11 are contour maps which show the total-pressure variation at the compressor entrance of the original and modified inlets.

CONCLUSIONS

An experimental investigation of the air-flow stability and pressure recover of two normal-shock, scoop-type inlets on a model of an interceptor or fighter-type airplane has led to the following conclusions:

1. The amplitude of the pressure pulsations of the original air-induction system was reduced by 1 to 20 percent in the mass-flow range of the investigation at all supersonic Mach numbers by reducing the wedge angle of the diverter under the boundary-layer splitter plate, lengthening the splitter plate, and eliminating the air-flow expansion in front of the inlet.

2. The modified inlet showed a significant increase in pressure recovery over the original inlet at supersonic speeds.

3. A qualitative correlation was observed between the magnitude of the flow pulsations and the pressure rise necessary to separate a turbulent boundary layer.

4. With pulsating flow in either the original or modified inlet, the resulting pressure fluctuations were random and had a maximum frequency of from 400 to 600 cycles per second.

National Advisory Committee for Aeronautics
Ames Aeronautical Laboratory
Moffett Field, Calif., Jan. 13, 1955

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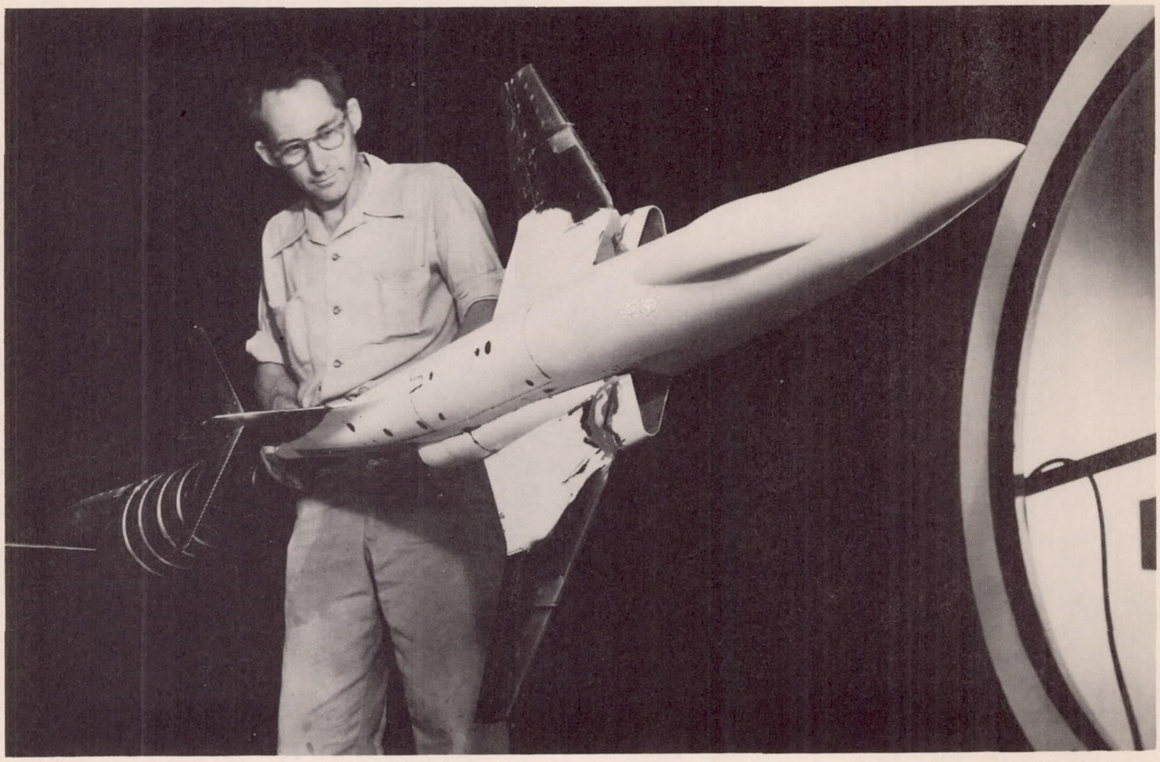

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Figure 1.- The air-induction model in the Ames 6- by 6-foot supersonic wind tunnel.

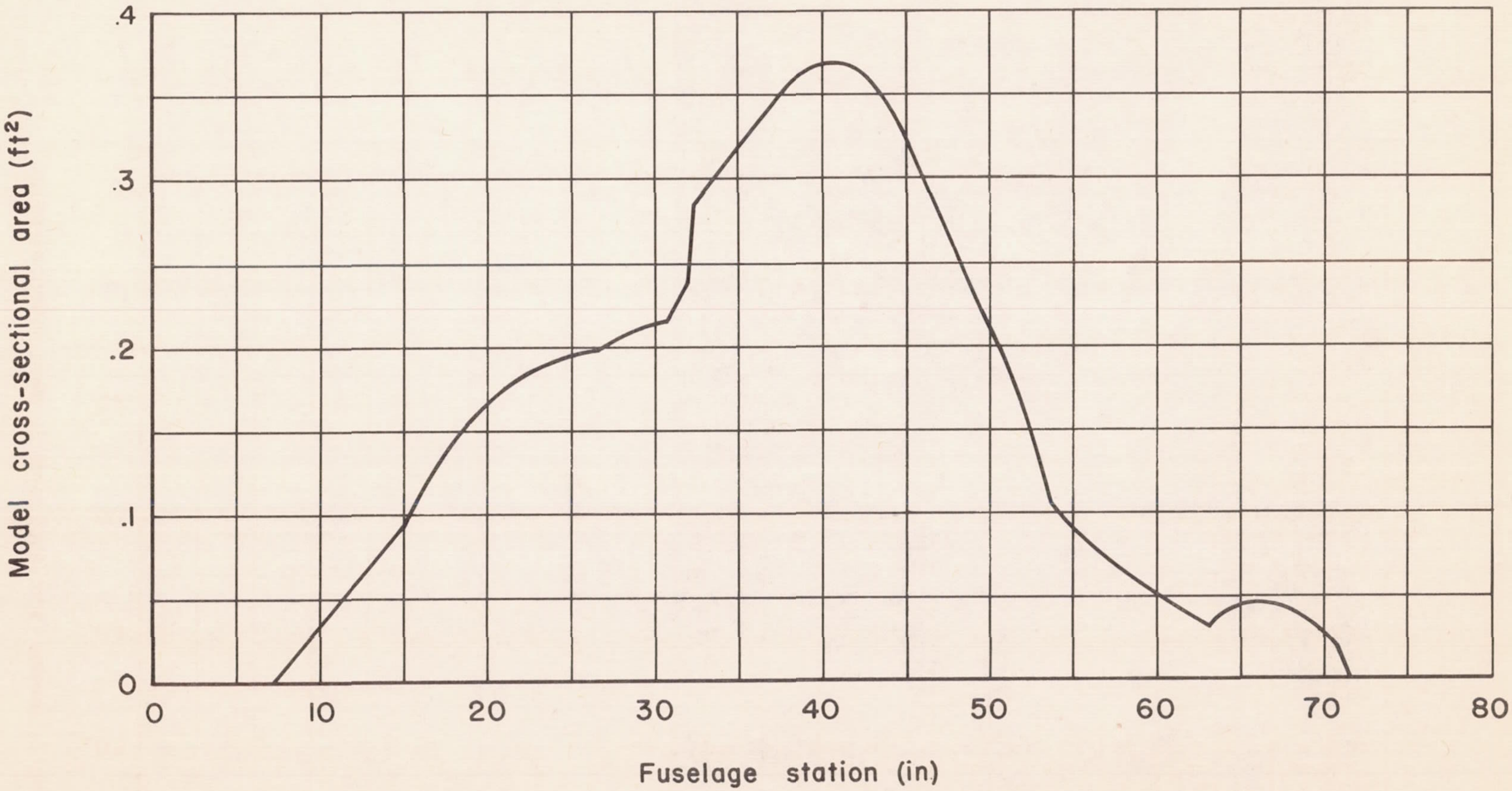


Figure 2. - Variation of model cross-sectional area with fuselage station.

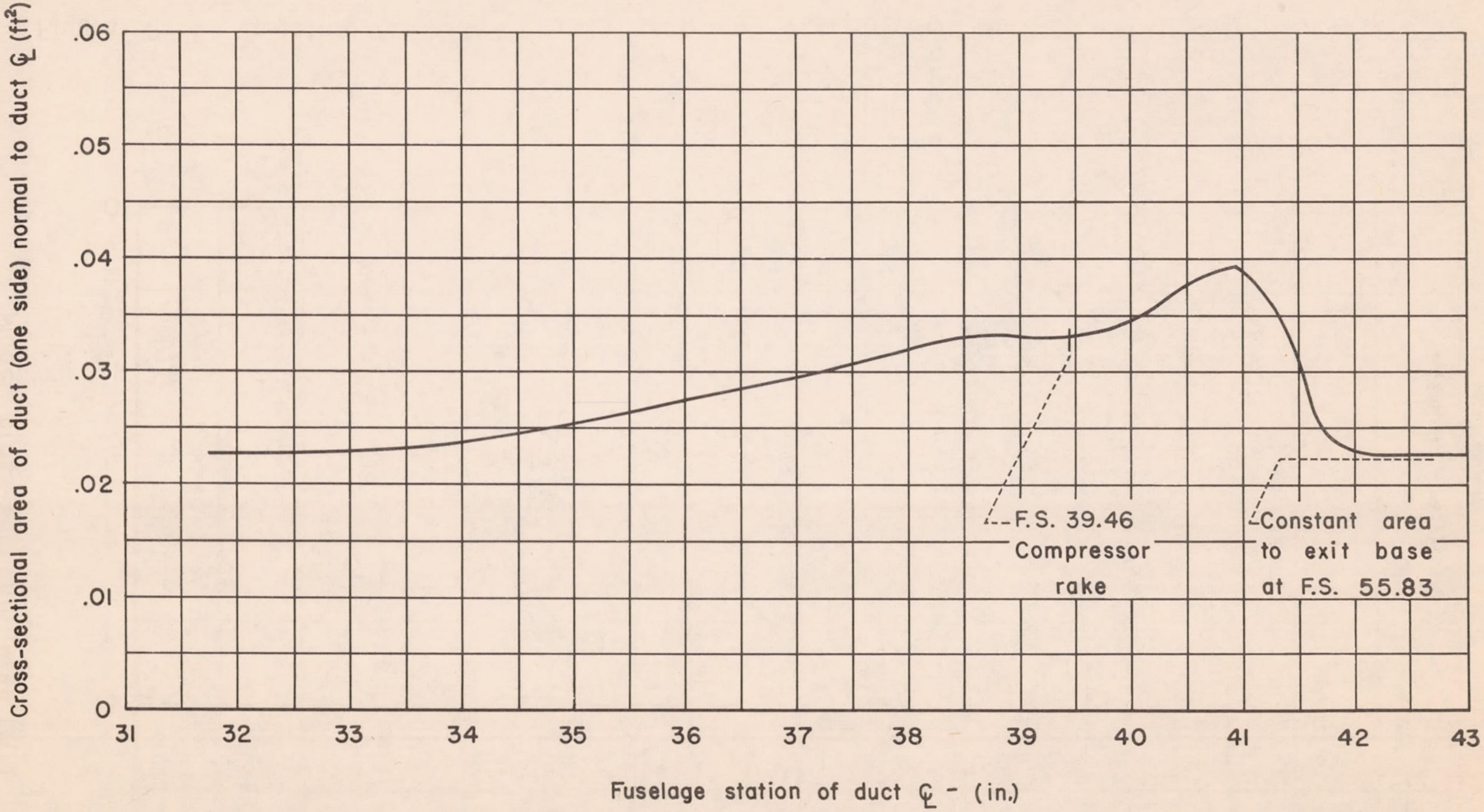


Figure 3. - Variation of duct cross-sectional area with fuselage station.

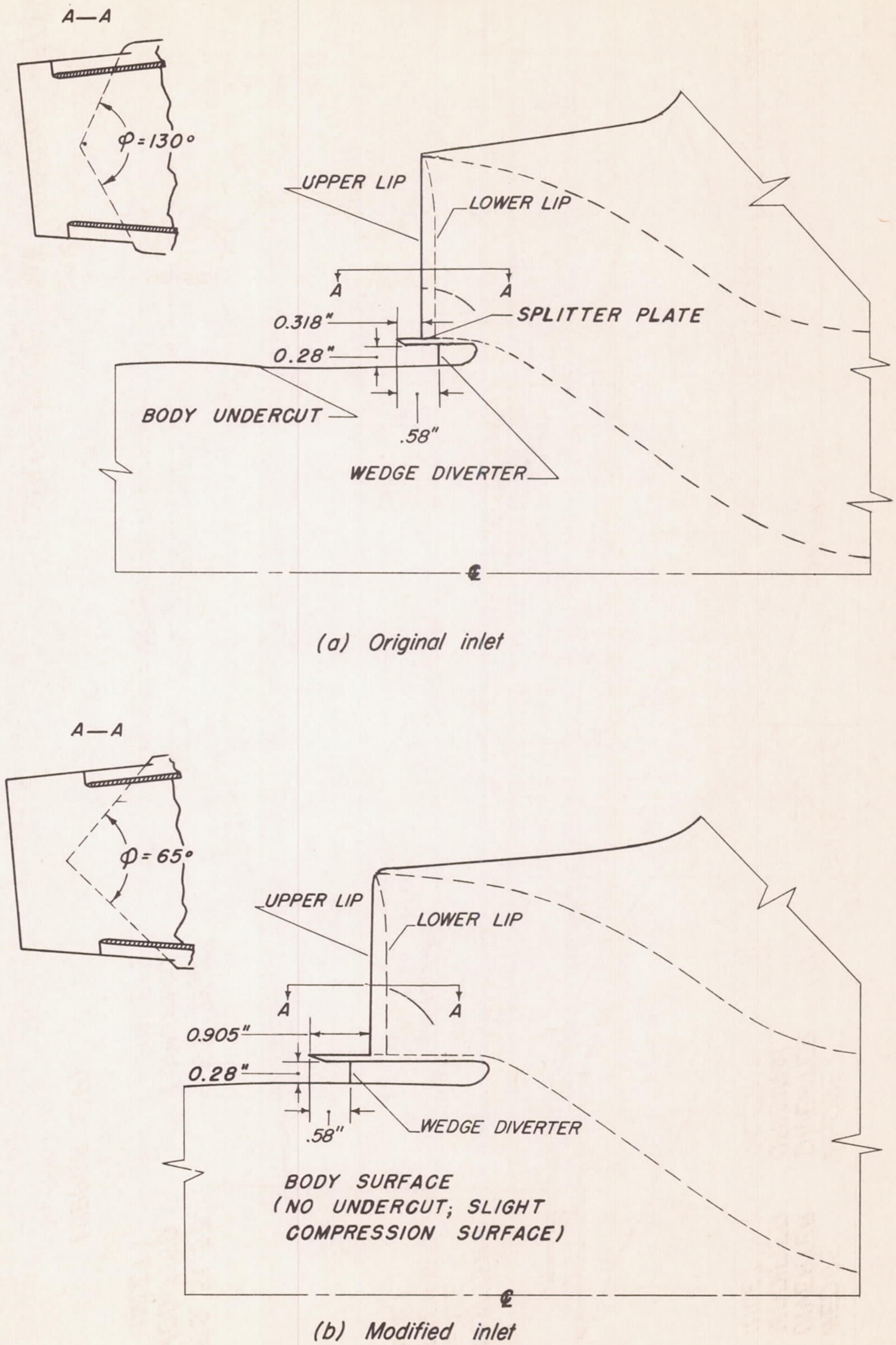


Figure 4.- Sketch showing inlet configuration details.

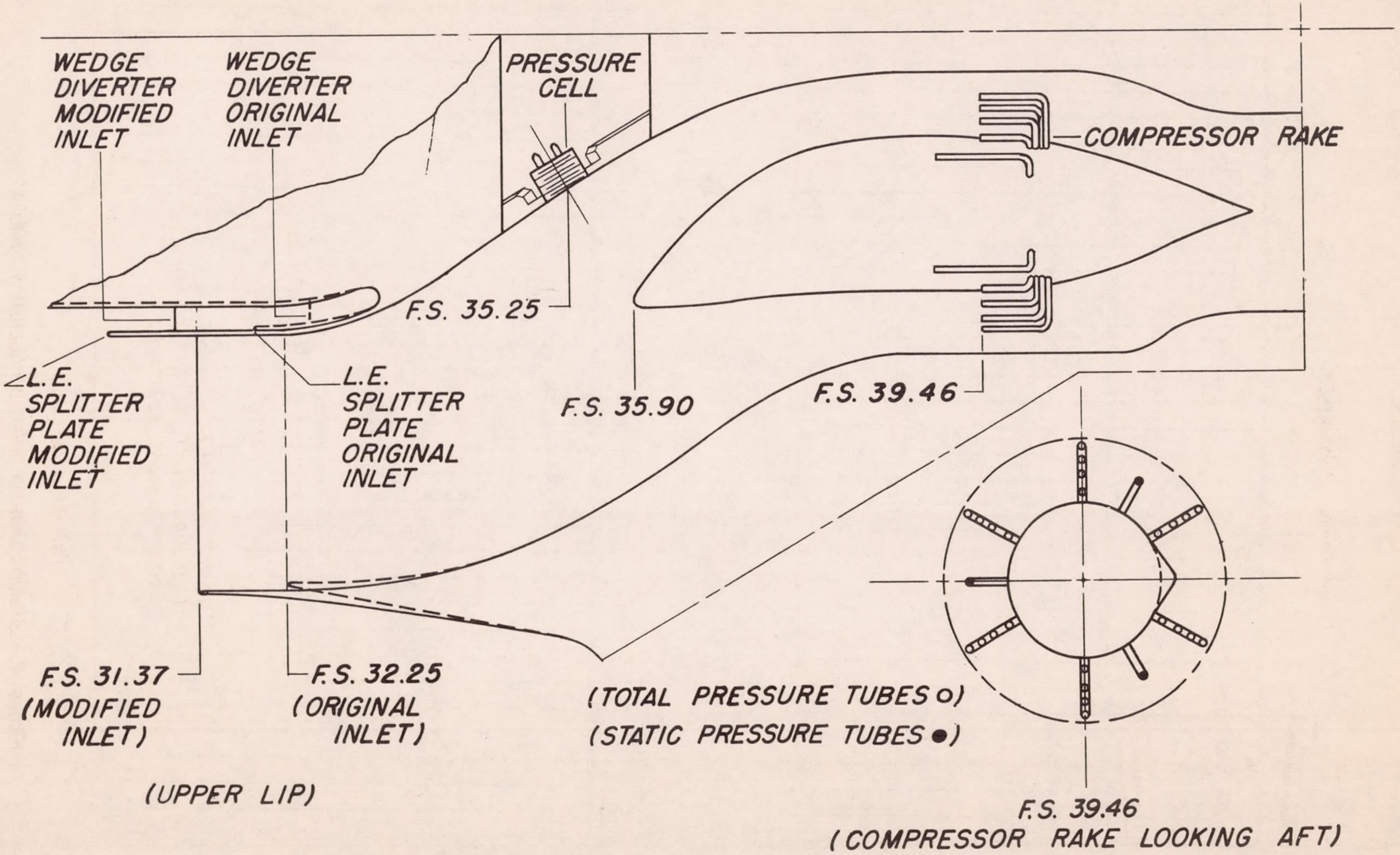


Figure 5.- Sketch showing compressor rake position and pressure cell location.

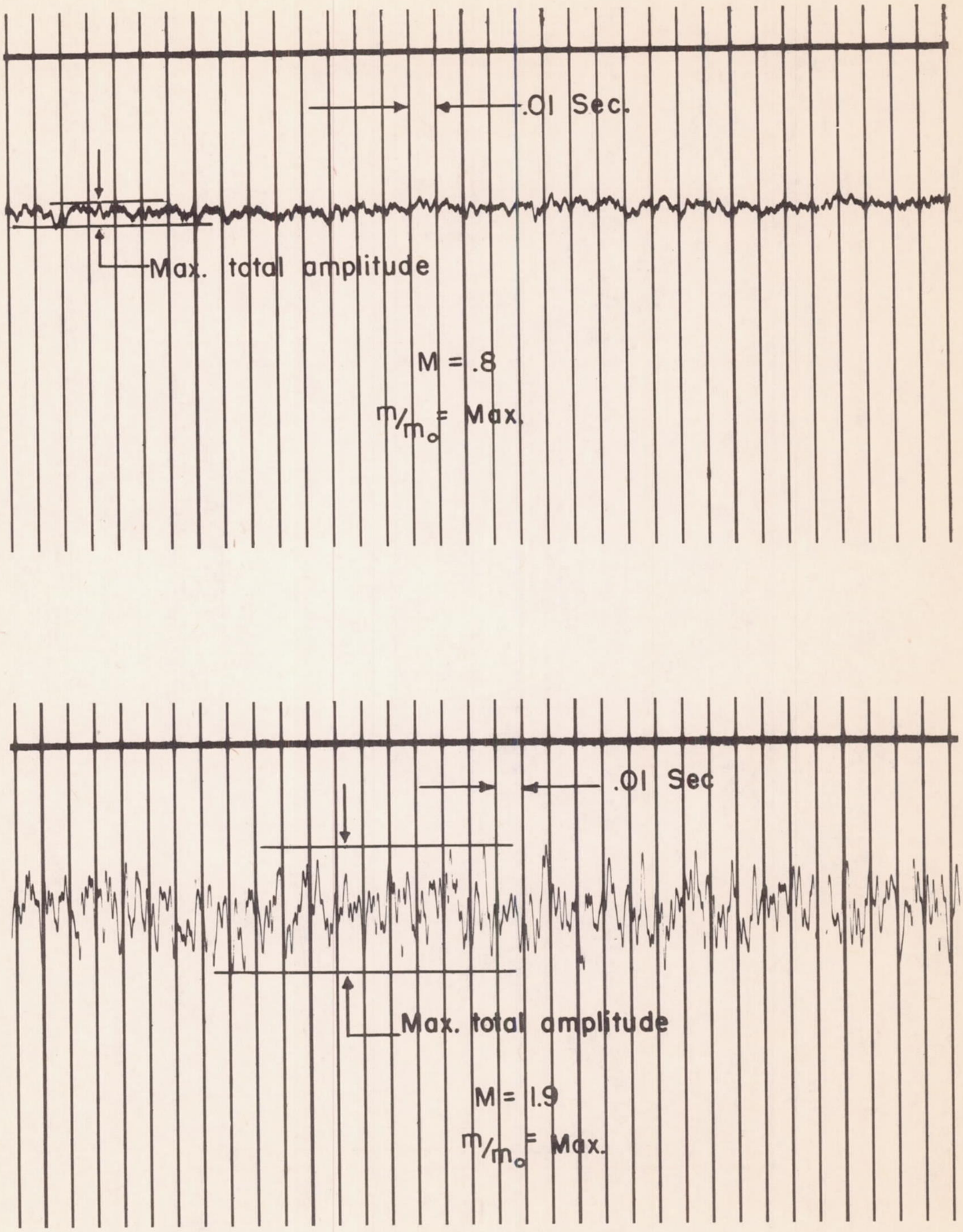


Figure 6.- Typical pressure time record of inlet pressure fluctuations.

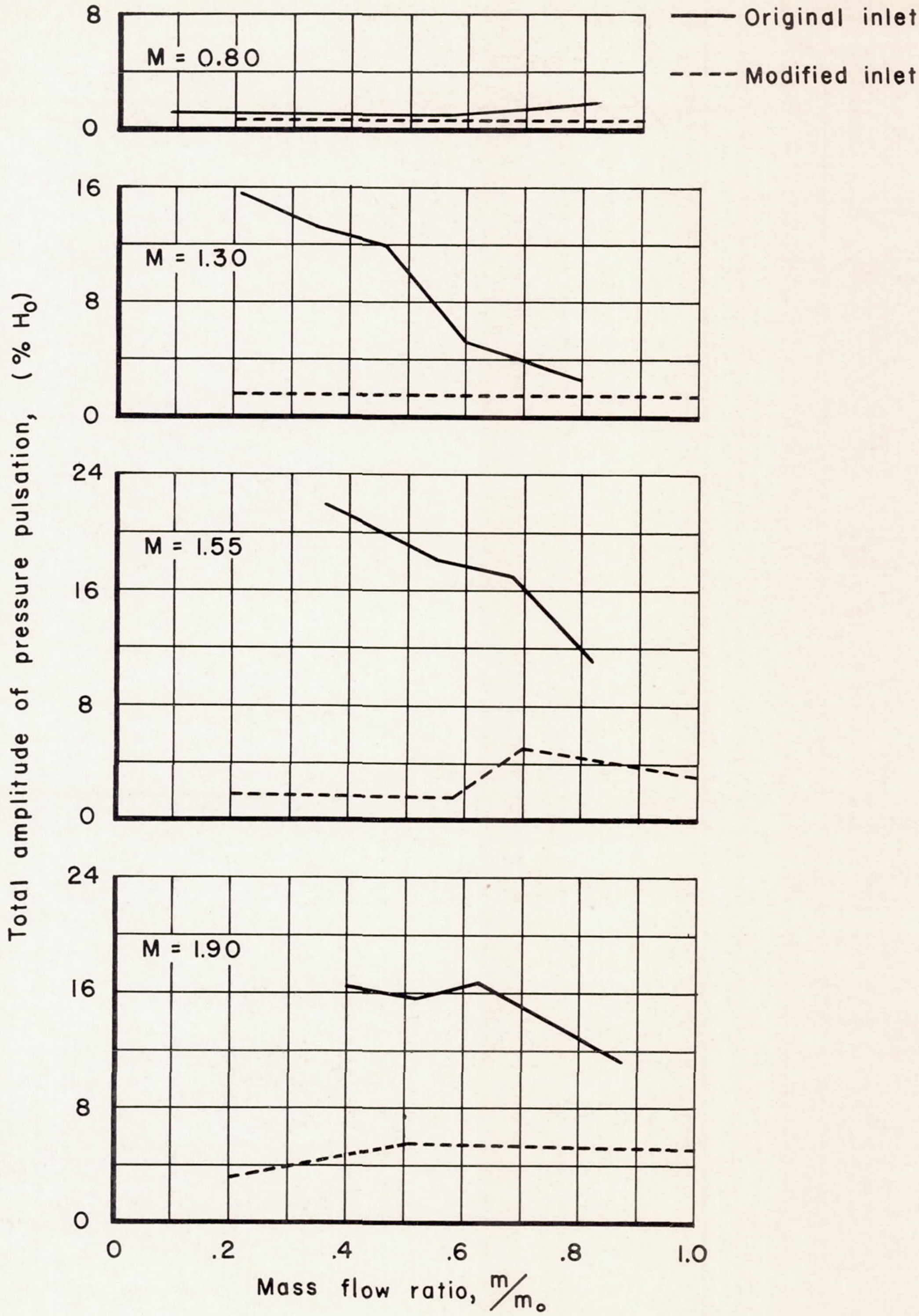
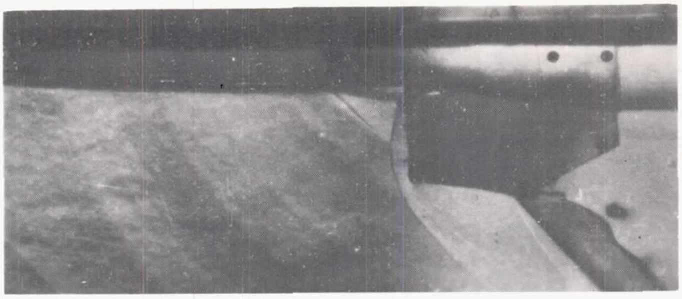
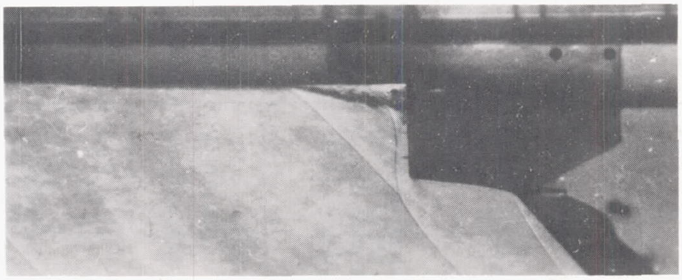


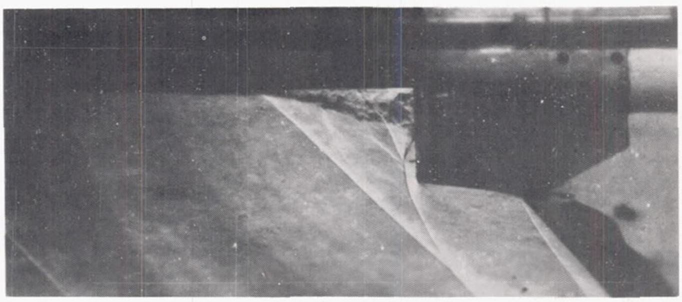
Figure 7.- Total amplitude of pressure pulsations of the original and modified inlets. $\alpha = 0°$



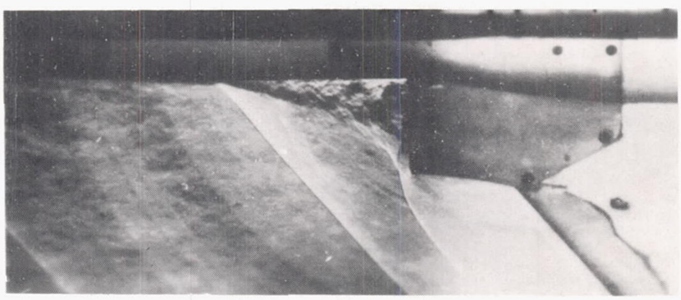
$m/m_o = 0.85$



$m/m_o = 0.69$



$m/m_o = 0.56$



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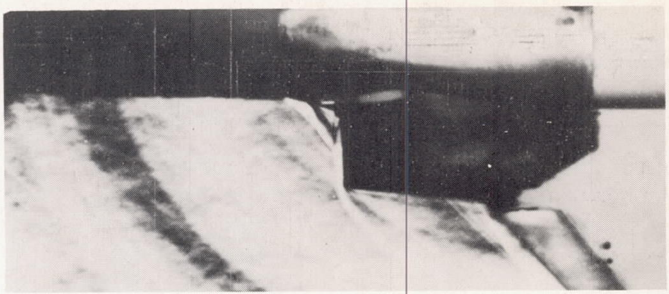
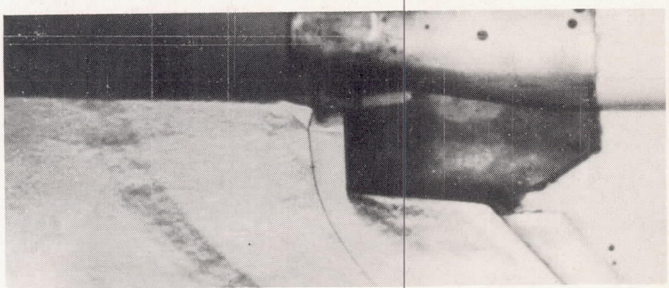
$m/m_o = 0.36$

(a) Original inlet.

Figure 8.- Schlieren photographs of the air flow in the inlet region; $M_o = 1.55$.

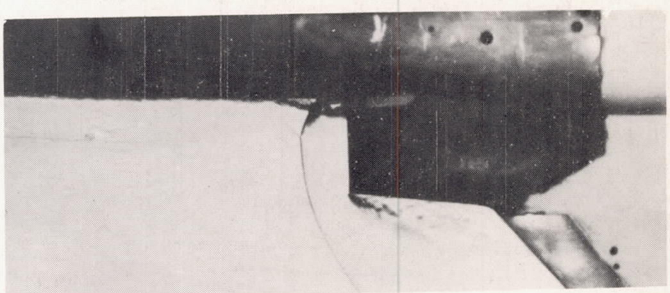


$m_1/m_o = 1.00$



$m_1/m_o = 0.64$

$m_1/m_o = 0.44$



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$m_1/m_o = 0.22$

(b) Modified inlet.

Figure 8.- Concluded.

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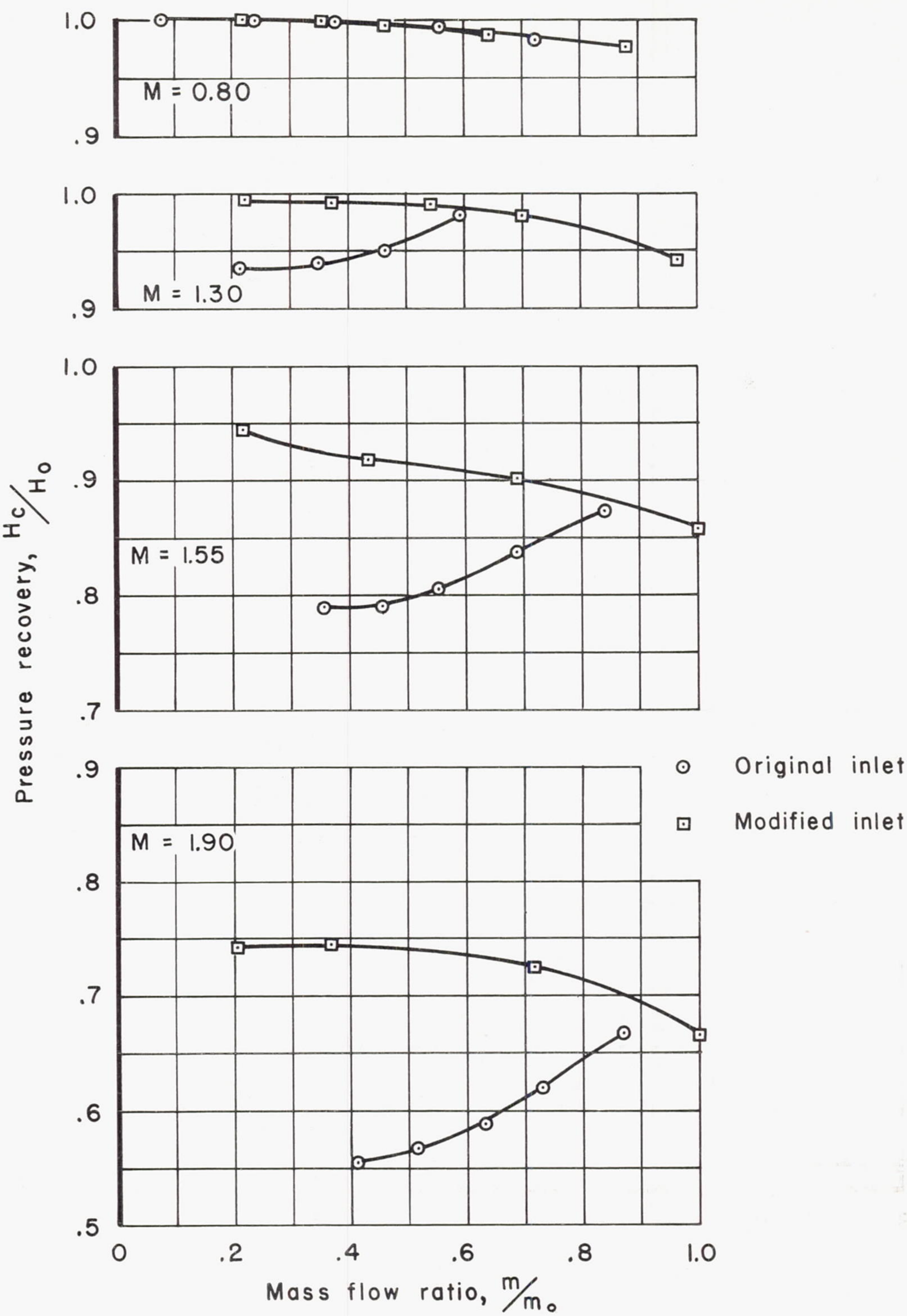
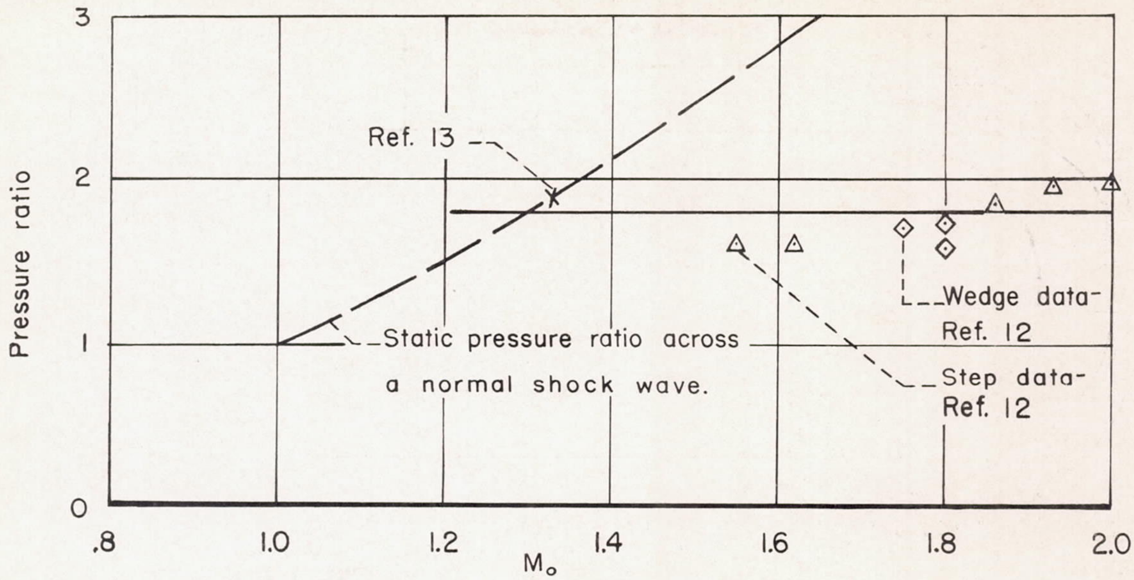
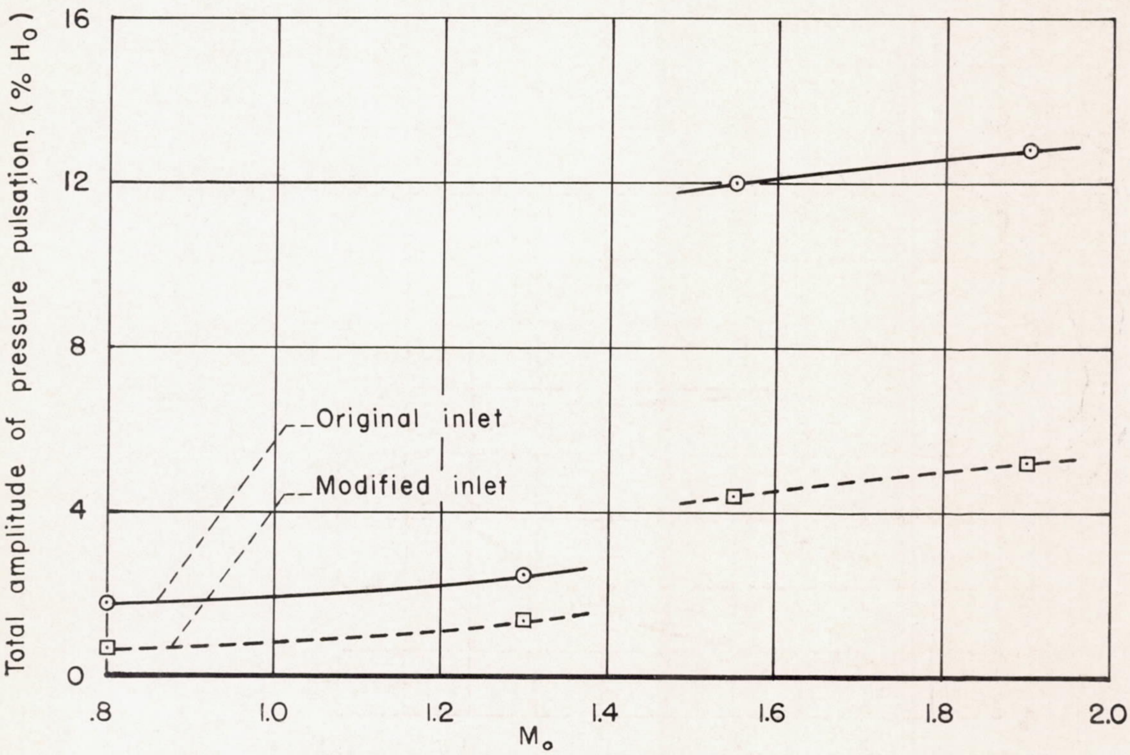


Figure 9. - Pressure recovery characteristics of the original and modified inlets; $\alpha = 0°$

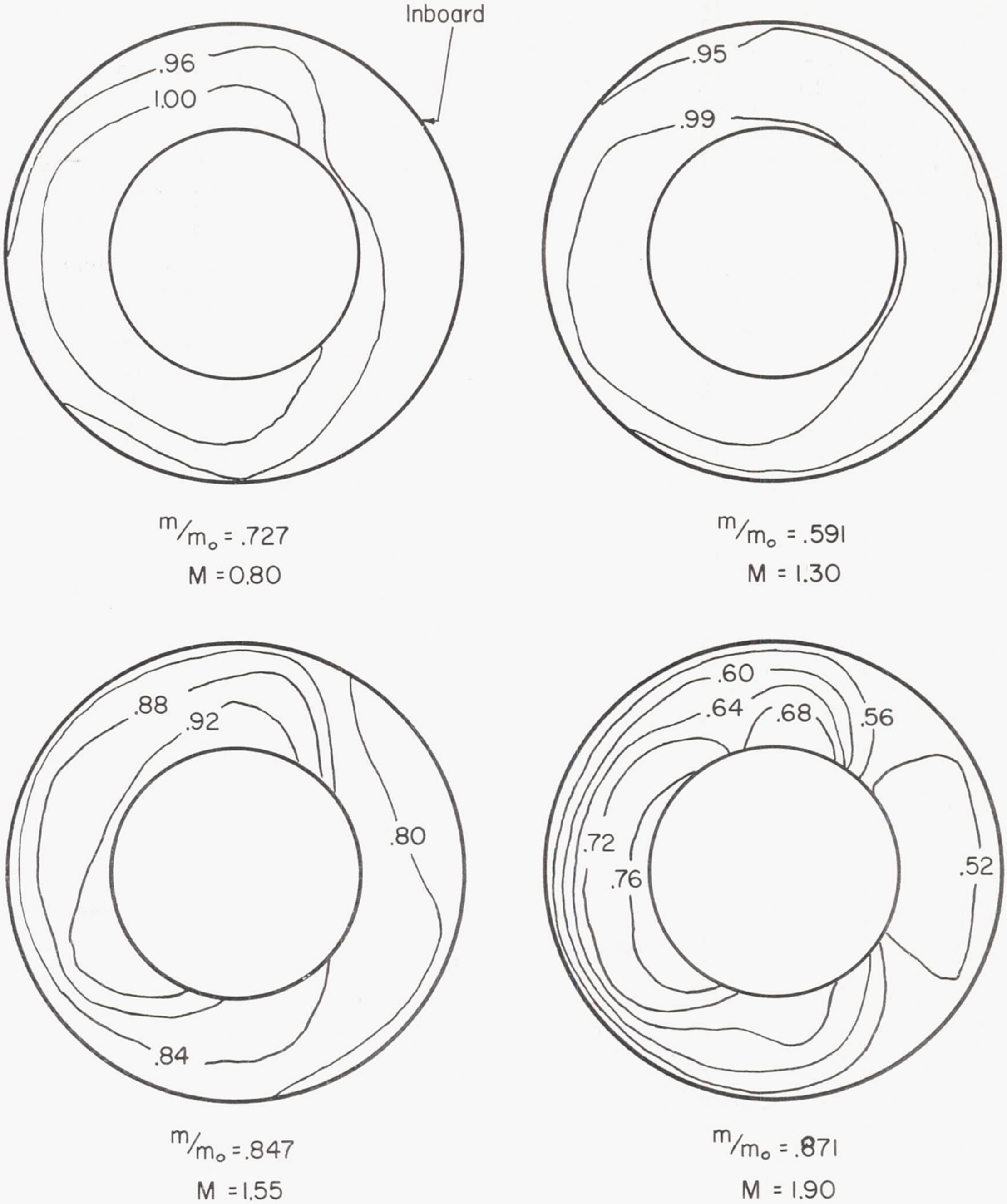


(a) Separation pressure ratio



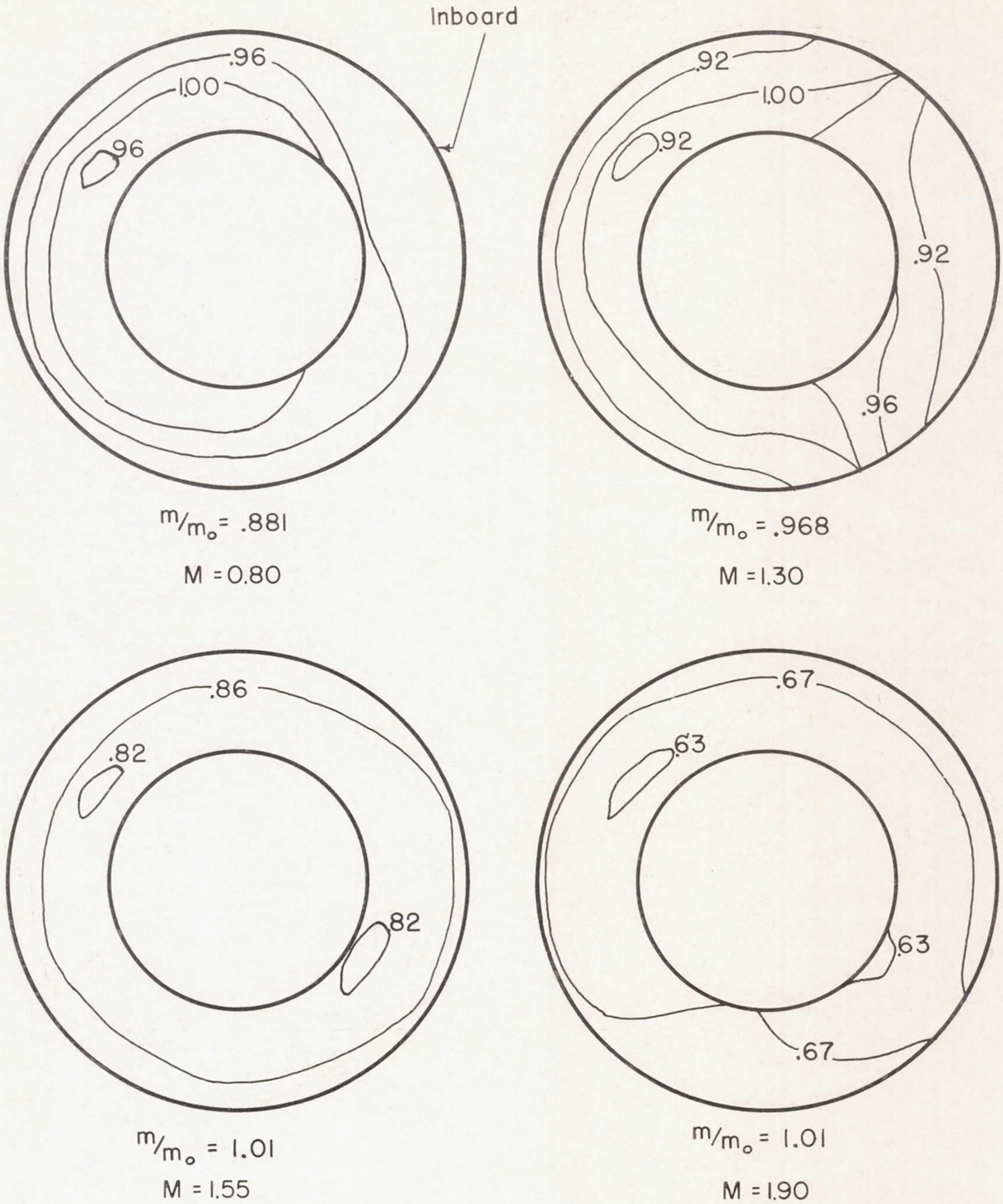
(b) Pressure pulsation, $m/m_o = 0.8$

Figure 10.- Correlation of air-flow instability with separation due to boundary-layer shock-wave interaction.



(a) Original inlet

Figure 11.- Typical total-pressure contour maps at the compressor entrance of the original and modified inlets. ($\alpha = 0°$).



(b) Modified inlet

Figure 11.- Concluded.