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RESEARCH MEMORANDUM

VISCOUS FLOWS IN INLETS

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and Lewis A. Anderson

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NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

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INTRODUCTION

It is well known that total-pressure ratios approaching 0.9 at Mach numbers of about 4 are desirable but have not been attained because of a group of problems termed "viscous effects." The purpose of this paper is to demonstrate how viscous effects can be minimized. Results of the basic research are first presented; these results are then applied to the design of an idealized inlet and the performance of this inlet is described.

The primary effect of viscosity at increasing free-stream Mach numbers is to increase the total pressure losses due to friction. This effect, coupled with the increasing positive pressure gradients in inlets, results in difficulty in preventing separation. It may not be possible to do much about this primary viscous effect other than to remove the low energy flow; however, there are a group of secondary viscous effects that can be eliminated at their sources. Some of these occur in the inlet shown in figure 1. This inlet is rectangular and has partial internal contraction. The effects of corners and of the oblique pressure fields on the side walls are typical secondary viscous effects. These effects result in local regions of thickened boundary layer, and the first separations, in strong positive pressure gradients, would occur in these regions. When the positive pressure gradient is continued downstream, the regions of separation grow into wakelike flows having strong large-scale low-frequency mixing. The transonic region in inlets is particularly sensitive to the development of wakelike flows because of the effect of fluctuations in effective throat area on the throat Mach number and thus on the total pressure losses at the terminal shock and in the subsonic diffuser. It appears that care must be exercised in inlet design to maintain uniform steady boundary-layer and core flows in the throat. The initial reaction to such an idealized requirement is that it is unnecessarily restrictive, but that this is not the case will be demonstrated.

SYMBOLS

M	Mach number
p_t	total pressure
p_{t_∞}	ambient total pressure
$(\Delta p_t)_{max}$	peak-to-peak fluctuation in total pressure
r	distance from surface
R	local duct radius
S/A_t	ratio of wetted area of supersonic diffuser to throat cross-sectional area

BASIC RESEARCH RESULTS

The basic research has been directed toward providing sufficient understanding of unsteady transonic diffusing flows to allow the design of a Mach number 4.0 inlet with a weak terminal shock and steady transonic flow. The most important result of this research is demonstrated with motion pictures. These pictures have been taken by use of an instrumentation system which provides a continuous display and record of transient-total-pressure profiles in boundary layers. The test apparatus is shown in figure 2. A four-tube total-pressure rake with transient-pressure transducers and a similar wall-static-pressure transducer are located at the exit of an 8° conic diffuser in which supersonic flow exists to a Mach number of about 1.1 just downstream of the throat. The outputs of the transducers are displayed on an oscilloscope, as shown in figure 3, with total-pressure ratio as the abscissa and radial position in the duct as the ordinate. Fluctuations in local total pressure appear as horizontal motions of the spot at each tube position. The local static pressure is plotted at the wall to provide an indication of separation.

The first flow that is shown is called unsteady; that is, large-scale, low-frequency total pressure losses occur with separation at the wall and simultaneous large losses in pressure at the tube farthest from the wall. Selected frames from the first motion picture are reproduced in figures 4(a) and 4(b). The time between frames is 0.125 second.

The second flow is relatively steady in that the largest and lowest frequency disturbances have been eliminated. The total pressure farthest from the wall is almost constant and the flow at the wall is not quite separated. Selected frames from the second motion picture are reproduced in figure 5. The time between frames is 0.125 second. The difference between the two flows is due to an improvement in initial flow uniformity of less than 1 percent in total pressure. This is so small that the critical disturbances may not be total-pressure fluctuations, but rather small fluctuations in local stream angles or vorticity. In either case, the conclusion is that the initial flow should be uniform and steady in order to attain steady exit flows in diffusers.

The sensitivity of diffusing flows to small initial disturbances was unexpected and it was believed that some rational explanation was essential if the result was to be employed with confidence. The general question is: How does large-scale turbulence grow from some initial shear? This is one of the most basic fluid-mechanics problems and one possible explanation of the phenomenon is shown in figure 6.

It is assumed that some initial shear exists in a field of random three-dimensional disturbances. When angular momentum is conserved, some spectrum of random streamwise vortices is produced, each of which is a conventional secondary-flow vortex. (See ref. 1.) Any vortex has a low-energy core which grows in diameter with distance downstream, and because of this growth a positive static-pressure gradient must exist along the axis. Now, assume the existence of a single disturbance sufficiently strong to slow the core appreciably. The core must expand rapidly in response to this pulse and may be forced to continue to expand because of the abrupt increases in positive pressure gradient demanded by the growing diameter and because of the deflection of the ambient flow about the rapidly growing core. Thus, some of the vortices in the shear layer can be unstable and explode, as indicated. This concept is consistent with the requirement of self-preservation of turbulent flows because each explosion results in the two features required for more explosions, namely, new vortices and a pressure wave. In such a flow it would be expected that each generation of vortices and explosions would become larger in size.

When this hypothesis is applied to the flow in a choked diffuser, it is apparent that the effect of the additional positive pressure gradient, the confining effect of the duct walls on the pressure pulses, and the effect of the terminal shock are to accentuate the growth of initial shear into large-scale turbulence. The effect observed in the motion pictures appears to be consistent with this simple physical reasoning and the physical reasoning is consistent with many diverse features of turbulence. (See ref. 2.) Thus, both the unsteadiness observations and the hypothesis can be used with some confidence.

SUPERSONIC-INLET DESIGN AND TEST RESULTS

The need for maintaining uniform and steady throat flows has been emphasized. This result will now be applied to the design of an inlet for a Mach number of about 4. The inlet shown in figure 7 is one of several possible designs that can satisfy the need for uniform throat flow. All the features of this inlet - axial symmetry, internal contraction, isentropic compression, scoop location, terminal-shock compensation, and the 4° diffuser - were selected to allow attainment of steady, uniform throat flow.

Two features of this axisymmetric isentropic inlet are unusual; namely, the location of the boundary-layer scoop just ahead of the inflection point and the abrupt growth in cross-sectional area in the terminal-shock region. The scoop location provides a thin boundary layer in the region of maximum pressure gradient, which should prevent separation and thus provide steady flow in the throat. Terminal-shock compensation has been used for several reasons and the most interesting reason can be demonstrated by use of the shear-to-turbulence hypothesis presented earlier. It is well known that turbulent boundary layers have two basic components: eddies, or small finite-length streamwise vortices, and "balls" of turbulence. If it is assumed that the terminal shock will increase the local production of turbulent balls from the eddies, the local flow cross section must be expanded to allow for the local total pressure loss in the turbulence. Thus, the use of terminal-shock compensation allows for departure from the assumed uniform throat flow. It should be noted that the mixing due to production of new eddies at the terminal shock is probably of some significance in separation or reattachment just downstream of the shock.

It is somewhat incidental to the main subject of this paper, but the question of flow-starting with internal-contraction inlets always occurs. In the inlet shown in figure 7, the top half of the supersonic flow region was arranged to move upward and forward on links to allow the excess air to be bypassed around the throat during starting.

Tests have been conducted with components of the idealized inlet at Mach numbers from 3.6 to 3.94, and tests of the complete inlet were conducted at Mach numbers of 3.70 to 3.83 for several configurations. The test Reynolds numbers were from 1.3×10^6 to 2.6×10^6, based on inlet diameter. The component tests were conducted to allow boundary-layer surveys at the inflection point and at the minimum-area station. Further flow profiles were measured just downstream of the terminal shock and at the duct exit with the complete configurations. Typical profiles obtained at these stations are presented in figures 8 and 9 in terms of the mean total-pressure ratio and the peak-to-peak total-pressure-fluctuation ratio. It should be noted that there is no exit-fluctuation profile because the mean profile was obtained with a liquid manometer.

At the inflection point (fig. 8), the mean total-pressure profile shows the concavity that is due to the friction total pressure loss. The fluctuation profile (fig. 9) indicates large fluctuations to occur well off of the surface, and this result is an indication of separation ahead of the inflection point. Unfortunately, in this investigation it was not possible to move the scoop forward; therefore, the next best thing was to remove all the separated flow. As a result, the shock waves due to the separation remain in the core flow and prevent attainment of isentropic compression. At the minimum-area station, the mean input to the terminal shock is not unusual and the fluctuation profile has a low-amplitude peak near the surface. The output from the terminal shock in terms of the mean profile appears favorable; however, the fluctuation profile, as well as the examination of the oscillograph records, suggests occasional separation. The last mean profile is at the duct exit and the distortions, both radial and circumferential, are small - less than 3 percent. This, in view of the occasional separation at the diffuser entry, indicates that the choice of a 4° divergence in the subsonic diffuser was judicious.

The overall performance of the idealized inlet is shown in figure 10 as the variation of total-pressure ratio with Mach number. Test data for the present inlet are for a Reynolds number of 2×10^6 because no improvement in recovery occurred with further increase in Reynolds number. The best available results from several other inlet investigations at Mach numbers above 3 are shown for comparison. (Some of these data are presented in refs. 3, 4, and 5.) Values of the ratio of supersonic wetted area S to throat cross-sectional area A_t are listed in figure 10 to indicate roughly the relative size per unit mass flow of the various inlets. The inlet of the present investigation is relatively short, but it can be seen that it has the highest pressure recovery. A pressure recovery of 85 percent was obtained at a Mach number of 3.70 and a recovery of 82 percent was obtained at a Mach number of 3.83. Since the boundary-layer-bleed flow rates are roughly from 20 to 25 percent of the total flow in all the inlets presented, the comparison in figure 10 is sufficient to indicate relative merit. Although the level of pressure recovery that has been achieved is unusual, it is evident that the goal of 90-percent pressure recovery has not been achieved. However, by using the total-pressure profiles, the component losses in the duct can be evaluated to find out why the goal was not achieved.

A summary of the component losses in the idealized inlet is as follows:

Subsonic diffuser, percent p_{t_∞}	2
Terminal shock, percent p_{t_∞}	1
Scoop-to-throat friction, percent p_{t_∞}	6
Supersonic core flow, percent p_{t_∞}	9
Total loss, percent p_{t_∞}	18

The data shown are temporal mean values and were obtained at $M = 3.83$ from total-pressure profiles with the assumption that the flow is actually axisymmetric. It should be noted that the possible error in each of the component losses is about $\pm 0.01 p_{t_\infty}$ and the possible error in the total loss is $\pm 0.005 p_{t_\infty}$. The total pressure loss in the subsonic diffuser is small, which was also indicated qualitatively by the fact that the exit distortions were small. The Mach number ahead of the terminal shock is about 1.1 and the total pressure loss at the terminal shock is less than 1 percent. The fact that a Mach number as low as 1.1 was measured is evidence that the flow must have been steady and relatively uniform in the throat. The friction loss from the boundary-layer scoop to the throat is about 6 percent and the remainder of the total pressure loss is in the supersonic core flow and is roughly 9 percent. In theory, there should be no supersonic core-flow loss, but the separation ahead of the boundary-layer scoop has prevented attainment of the theoretical flow. It would be expected that elimination of the separation, by moving the scoop forward or decreasing the local pressure gradient, would reduce the core-flow loss substantially. This should result in a pressure recovery near 90 percent at a Mach number of 3.83, and thus the initial goal is not unrealistic.

CONCLUDING REMARKS

It has been shown that the favorable end result of the present investigation has been obtained because of a basic research program directed toward understanding unsteady flows. A new experimental technique, the continuous display of transient-total-pressure profiles, has been used to demonstrate one origin of unsteady internal flows, and a qualitative understanding of such flows has been achieved. This has resulted in a useful hypothesis regarding one possible origin of turbulence and has indicated that uniform and steady throat flows are required for efficient inlets. Rigid application of this design requirement has produced an efficient inlet and thus it has been shown that, for the Mach numbers of this investigation, adverse viscous effects can be minimized.

Ames Aeronautical Laboratory
National Advisory Committee for Aeronautics
Moffett Field, Calif., Mar. 20, 1958

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4. Anon.: Combined Bimonthly Summary No. 60. Jet Propulsion Lab., C.I.T., Aug. 15, 1957, pp. 28-32.

5. Flaherty, Richard J., and Luidens, Roger W.: Use of Shock-Trap Bleed To Improve Pressure Recovery of Fixed and Variable-Capture-Area Internal-Contraction Inlets, Mach Number 2.0 to 3.0. NACA RM E58D24, 1958.

RECTANGULAR INLET

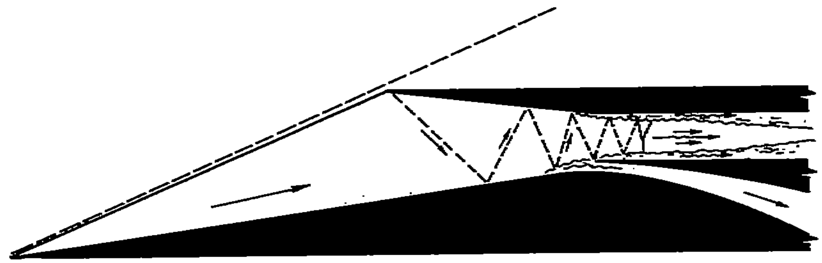

Figure 1

DIFFUSER TEST APPARATUS

MEAN M_{MAX}=1.1

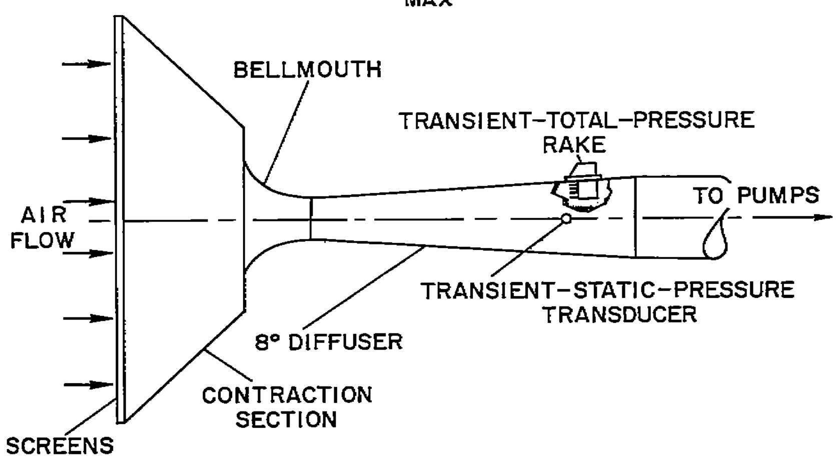

Figure 2

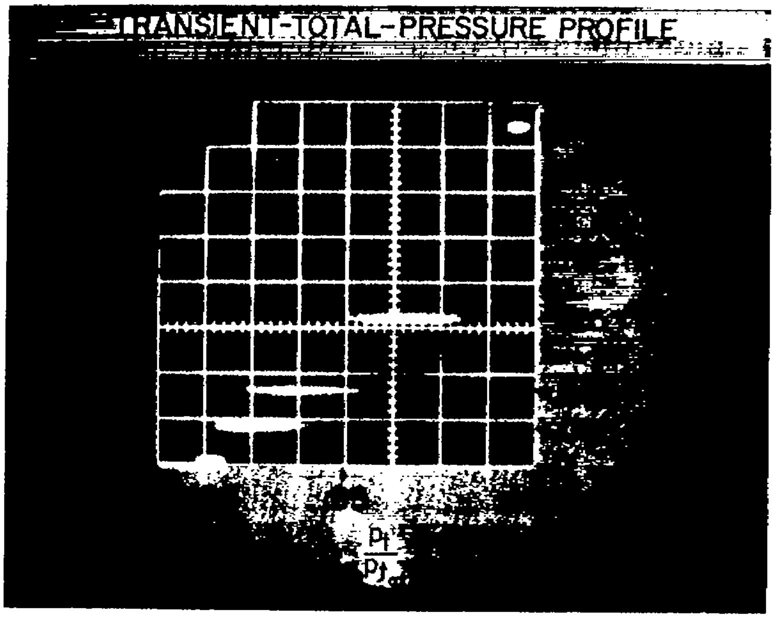

Figure 3

SELECTED FRAMES FROM MOTION PICTURE OF UNSTEADY FLOW
FRAMES 1 TO 35

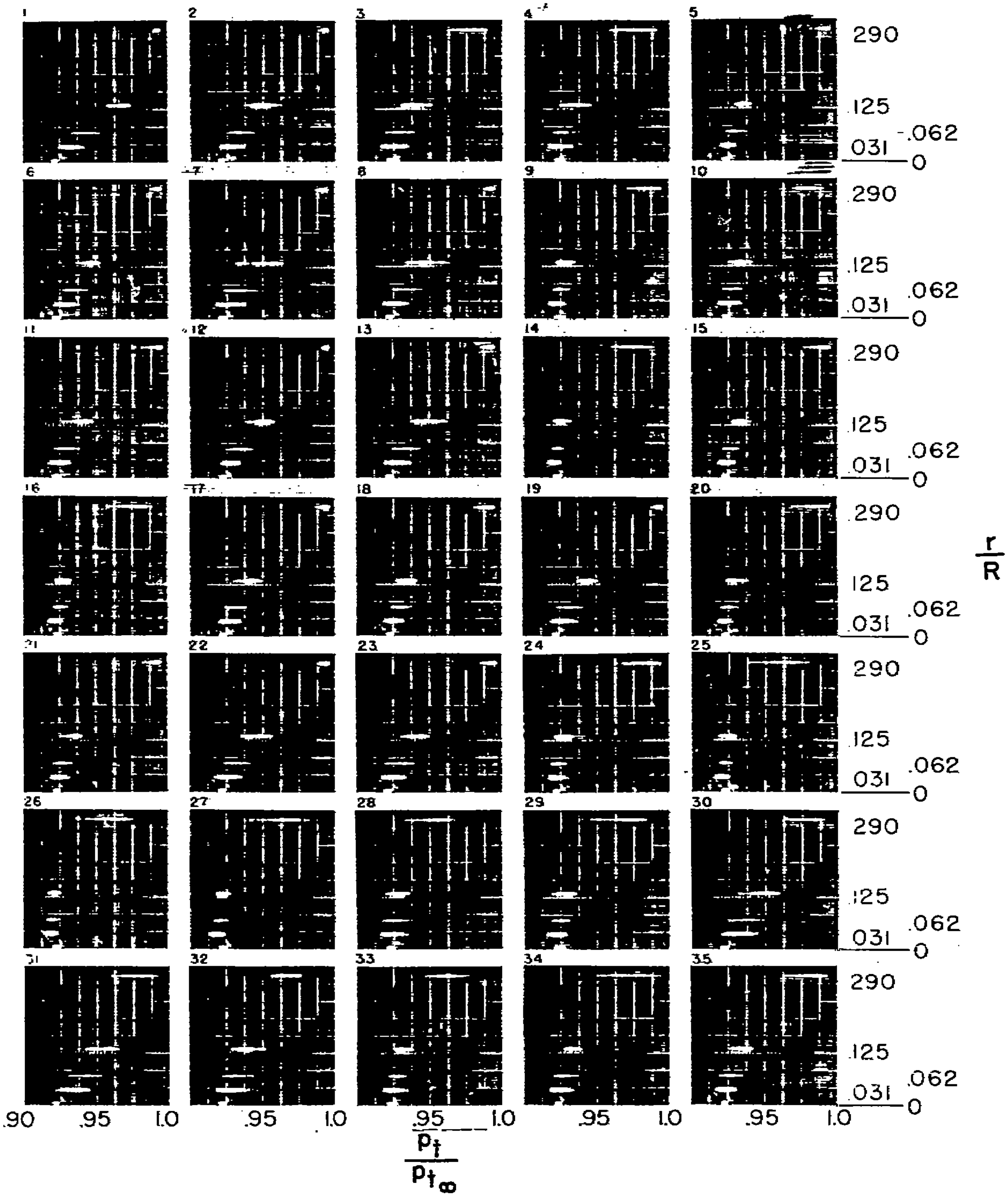

Figure 4(a)

SELECTED FRAMES FROM MOTION PICTURE OF UNSTEADY FLOW
FRAMES 36 TO 70

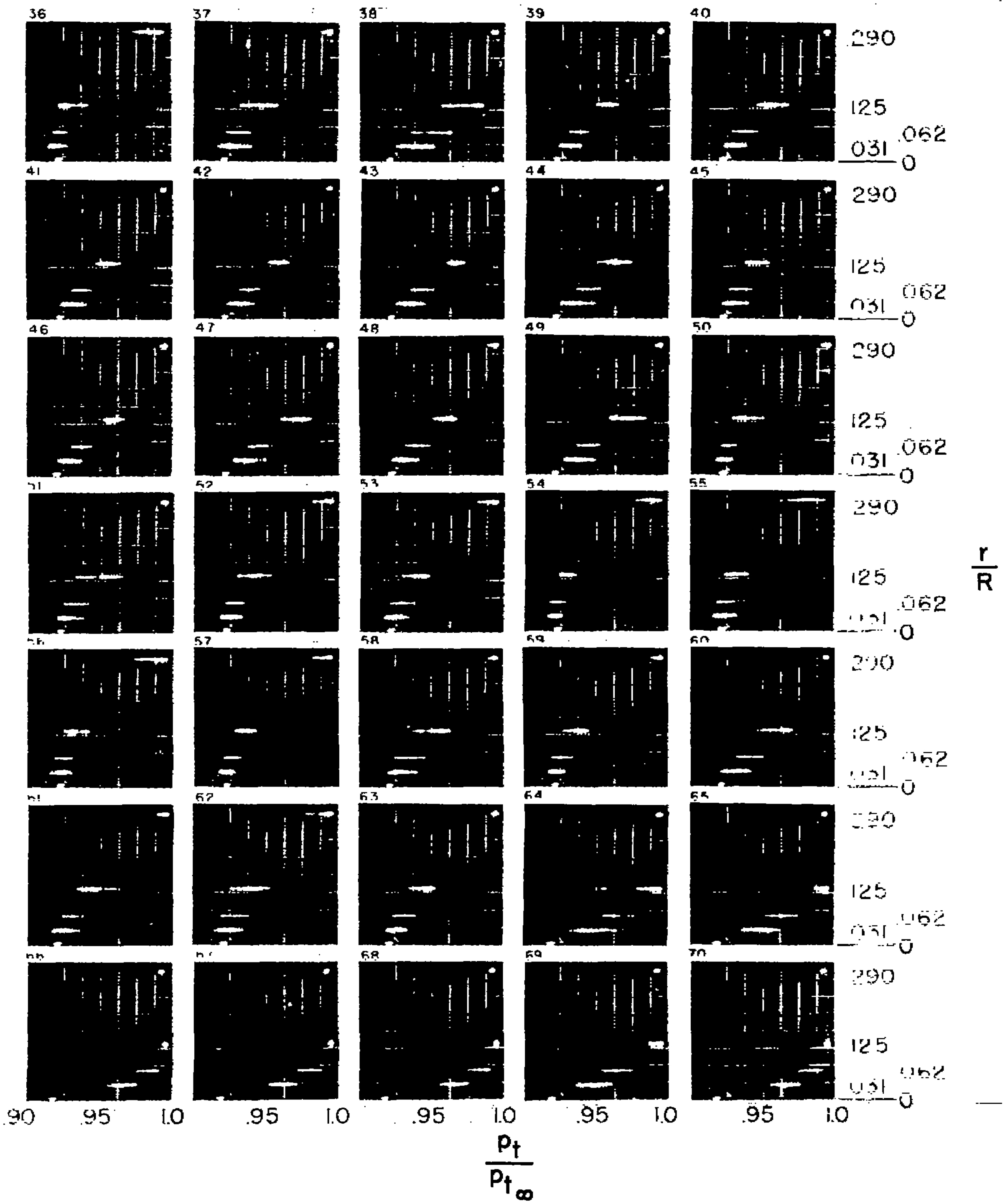

Figure 4(b)

[redacted]

SELECTED FRAMES FROM MOTION PICTURE OF STEADY FLOW

FRAMES 1 TO 35

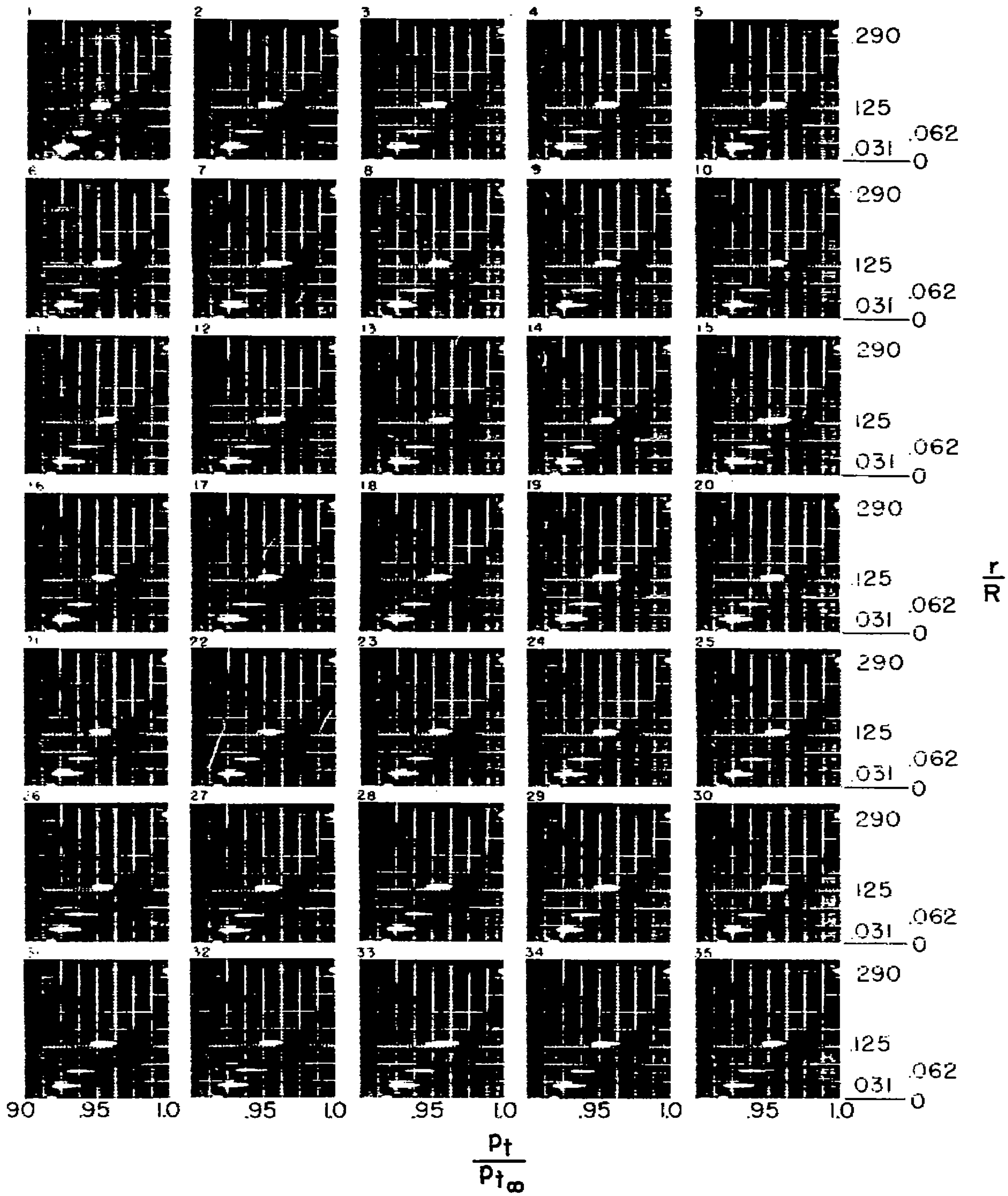

Figure 5

SHEAR-TO-TURBULENCE HYPOTHESIS

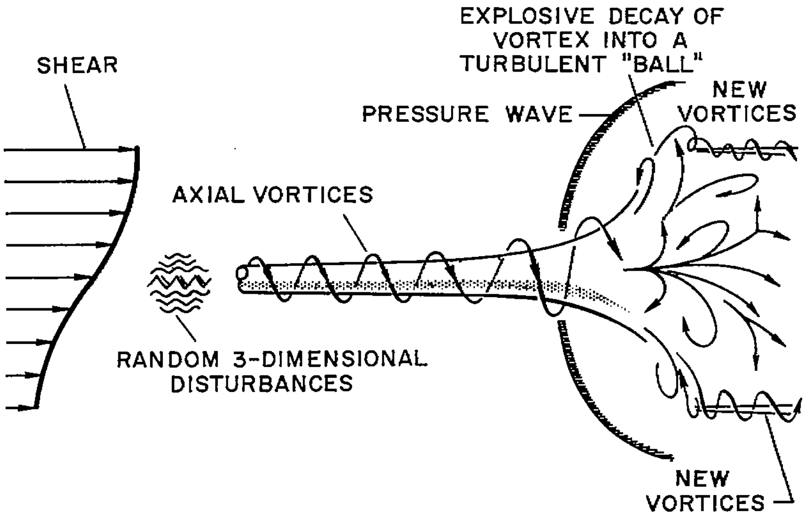

Figure 6

AXIALLY SYMMETRIC, ISENTROPIC,
INTERNAL-CONTRACTION INLET

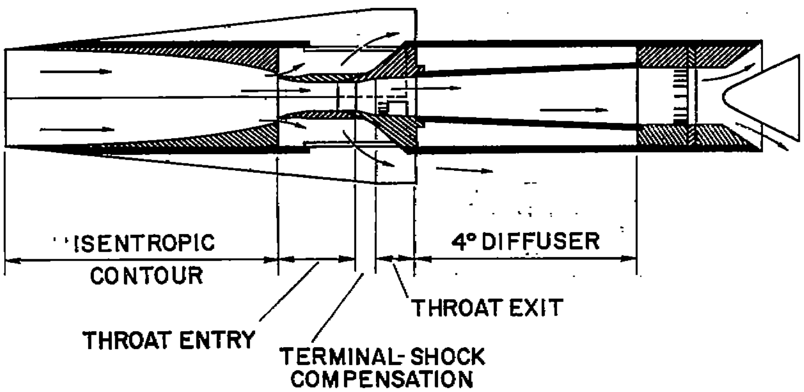

Figure 7

TYPICAL PRESSURE PROFILES

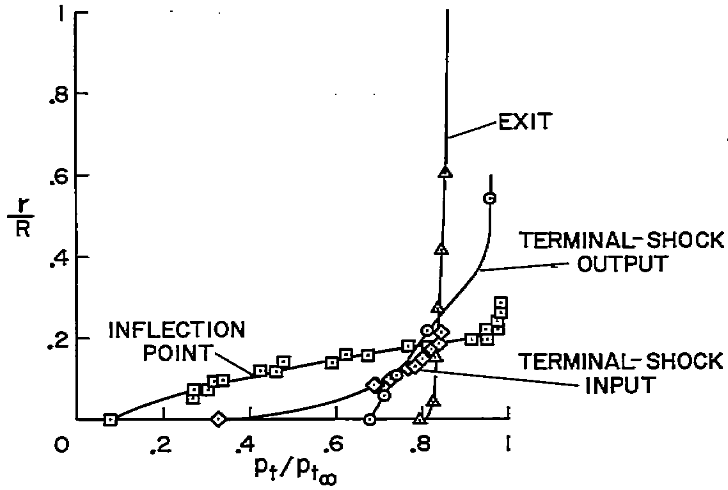

Figure 8

TYPICAL PRESSURE-FLUCTUATION PROFILES

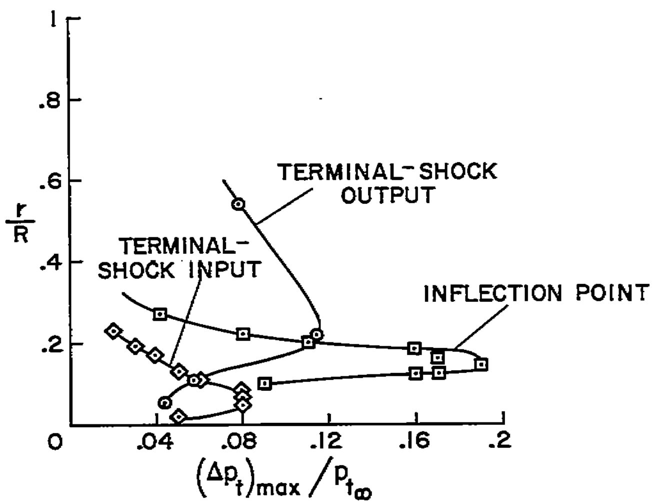

Figure 9

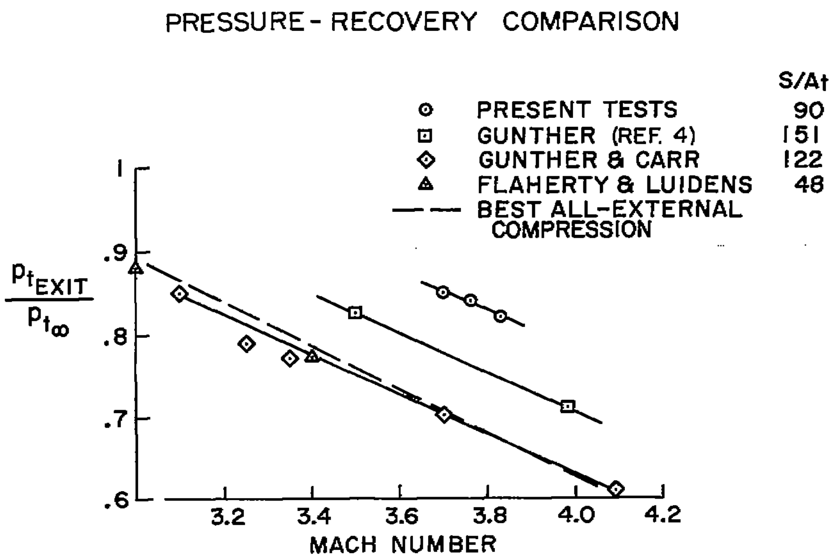

Figure 10

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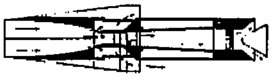
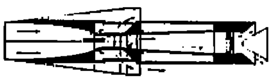
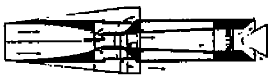
[redacted]

NOTES: (1) Reynolds number is based on the diameter of a circle with the same area as that of the capture area of the inlet.

(2) The symbol * denotes the occurrence of buzz.

Report and facility	Description			Test parameters				Test data				Performance		Remarks
	Configuration	Number of oblique shocks	Type of boundary-layer control	Free-stream Mach number	Reynolds number $\times 10^{-6}$	Angle of attack, deg	Angle of yaw, deg	Drag	Inlet-flow profile	Discharge-flow profile	Flow picture	Maximum total-pressure recovery	Mass-flow ratio	
RM A58D17a Ames 1- by 3-foot supersonic wind tunnel		Isen-tropic	Scoop	3.6 to 3.94	2×10^{6}	0	0	No	Yes	Yes	No	0.85 at M = 3.70	1.0	Report includes a discussion of the unsteady internal flow problem and presents transient total-pressure profiles at several stations.
RM A58D17a Ames 1- by 3-foot supersonic wind tunnel		Isen-tropic	Scoop	3.6 to 3.94	2×10^{6}	0	0	No	Yes	Yes	No	0.85 at M = 3.70	1.0	Report includes a discussion of the unsteady internal flow problem and presents transient total-pressure profiles at several stations.
RM A58D17a Ames 1- by 3-foot supersonic wind tunnel		Isen-tropic	Scoop	3.6 to 3.94	2×10^{6}	0	0	No	Yes	Yes	No	0.85 at M = 3.70	1.0	Report includes a discussion of the unsteady internal flow problem and presents transient total-pressure profiles at several stations.