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EXTERNAL INTERFERENCE EFFECTS OF FLOW THROUGH STATIC-PRESSURE ORIFICES OF AN AIRSPEED HEAD AT SEVERAL SUPERSONIC MACH NUMBERS AND ANGLES OF ATTACK

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON
March 1959

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SUMMARY

Wind-tunnel tests have been made to determine the static-pressure error resulting from external interference effects of flow through the static-pressure orifices of an NACA airspeed head at Mach numbers of 2.4, 3.0, and 4.0 for angles of attack of 0°, 5°, 10°, and 15°.

Within the accuracy of the measurements and for the range of mass flow covered, the static-pressure error increased linearly with increasing mass-flow rate for both the forward and rear sets of orifices at all Mach numbers and angles of attack of the investigation. For a given value of flow coefficient, the static-pressure error varied appreciably with Mach number but only slightly with angle of attack. For example, for a flow coefficient out of the orifices of 0.01 (the approximate value for a vertically climbing airplane for which the airspeed system incorporates an airspeed meter, a Mach meter, and an altimeter), the error increased from about 5 percent to about 12 percent of the static pressure as the Mach number increased from 2.4 to 4.0 with the airspeed head at an angle of attack of 0°.

INTRODUCTION

Airspeed installations on climbing or diving airplanes are subject to air flowing into or out of the static-pressure orifices because of the volume of the instruments and the connecting tubing. This air flow causes a pressure loss, and hence the instruments are subjected to a pressure that is different from the pressure at the static- or total-pressure sources. This is the well-known pressure-lag error. (See refs. 1 and 2.) Another error associated with flow into or out of the airspeed measuring system is the interference effect of the flow through the airspeed static-pressure orifices on the external flow when the external flow is supersonic. If the interference is appreciable, the static pressure in the region of the orifices may be expected to be different from that with no flow through the system. For example, at supersonic speeds flow out of the static-pressure orifices may cause a shock wave and hence an increase in static pressure, whereas flow into the orifices may cause an expansion wave or a decrease in static pressure.

The results of tests to determine this error at a Mach number of 3 and an angle of attack of 0° are reported in reference 3. Because the magnitude of this error was appreciable, it was considered of interest to extend the tests to include other Mach numbers and angles of attack. Consequently, additional tests were made at Mach numbers of 2.4, 3.0, and 4.0 for angles of attack of 0°, 5°, 10°, and 15° and the results are presented herein.

Measurements of static pressure in the orifice chamber were made with flow out of the orifices at mass-flow rates up to about 2.2×10^{-6}, 2.9×10^{-6}, and 3.7×10^{-6} slugs per second for the Mach numbers of 2.4, 3.0, and 4.0, respectively. The maximum Reynolds numbers (based on the diameter of the airspeed head) were 0.82×10^{6}, 0.77×10^{6}, and 1.15×10^{6} for Mach numbers of 2.4, 3.0, and 4.0 and tunnel stagnation pressures of 50, 75, and 190 lb/sq in., respectively.

SYMBOLS

A	total area of static-pressure orifices leading to each chamber of airspeed head, 1.25×10^{-4} sq ft
C_w	mass-flow coefficient, $\frac{w}{\rho_{\infty} V_{\infty} A}$
l	length of mass-flow measuring tube, 3.72 ft
M_{∞}	free-stream Mach number
p	static pressure, lb/sq ft
p_c	pressure in chamber of airspeed head, lb/sq ft
$p_{c,0}$	pressure in chamber of airspeed head with zero flow through orifices, lb/sq ft
Δp	pressure drop in mass-flow measuring tube, lb/sq ft

R gas constant for air, 53.3 ft/deg

r internal radius of mass-flow measuring tube, 3.2×10^{-3} ft

T temperature, $^{\circ}R$

v total volume of airplane airspeed installation, cu ft

V_{∞} free-stream velocity, ft/sec

w mass flow, $\frac{\pi r^4}{8\mu l} \Delta p \rho_2$, slugs/sec (positive values for flow out of airspeed-head orifices)

α angle of attack of airspeed head, deg

γ flight-path angle, deg

ϵ nondimensional pressure error, $\frac{p_c - p_{c,0}}{p_{\infty}}$

μ absolute viscosity of air in mass-flow measuring tube, slugs/ft-sec

ρ mass density of air, slugs/cu ft

Subscripts:

∞ tunnel or free stream

1 airplane airspeed installation

2 mass-flow measuring tube

APPARATUS AND TESTS

The NACA airspeed head and the test method used in the present investigation were the same as those used in the tests of reference 3. (See fig. 1.) The tests were again conducted in the Langley 9- by 9-inch high Mach number jet.

In the tests reported in reference 3, two different stagnation pressures were used to determine whether there was an effect due to Reynolds number. Because no effect was found over the range of Reynolds numbers covered, only one stagnation pressure was used at each Mach number in the present tests, namely 50, 75, and 190 lb/sq in. for the Mach numbers of 2.4, 3.0, and 4.0, respectively. The corresponding Reynolds numbers, based on airspeed-head diameter, were 0.82×10^6, 0.77×10^6, and 1.15×10^6.

The tests of reference 3 were made for conditions of flow both into and out of the orifices to simulate the conditions of an airplane diving or climbing. Because the results of those tests indicated essentially the same rate of change of static-pressure error with mass flow for both flow directions, tests were made only for the condition of flow out of the orifices in the present investigation. The Reynolds number of the flow through the orifices, based on the diameter of the larger orifice (0.052 inch) was 350 for the mass-flow rate of 3.7×10^{-6} slugs per second.

Results of the tests of reference 3 indicated that, although varying flow rates for the rear orifices had no effect on the pressure at the forward orifices (a result which would be expected for supersonic flow), there was a small effect (about 1 percent) on the pressure at the rear orifices for the full range of mass-flow rate used at the forward orifices. Because this effect had already been found to be small, no attempt was made to isolate this effect in the present tests.

Measurements were recorded by means of standard differential-pressure cells. Differential pressures were measured between: (1) tunnel static and chamber static pressures, (2) chamber static pressure and pressure at the end of the mass-flow tube, and (3) the pressure drop in the mass-flow tube. (See fig. 2.) Also recorded during the tests were tunnel stagnation temperatures and pressures and the temperature of air flowing through the mass-flow measuring tube. Photographs were taken of the airspeed head in the tunnel for the purpose of verifying the angle of attack, which was set in by a coarse indicator; in all cases the angle of attack was verified to within $\pm 1/2^{\circ}$ of the desired setting. The mass flow of the tests ranged from 0 to a maximum of about 2.2×10^{-6}, 2.9×10^{-6}, and 3.7×10^{-6} slugs per second for the Mach numbers of 2.4, 3.0, and 4.0, respectively.

RESULTS AND DISCUSSION

The static-pressure error was determined as the difference between the static pressure in the airspeed-head chambers and the static pressure at a tunnel-wall orifice about 1 foot upstream of the orifices in the airspeed head. The static-pressure error due to the external interference effects of flow through the orifices was obtained by subtracting the pressure difference for zero mass flow from the corresponding pressure differences with various values of mass flow.

The static-pressure error due to flow through the orifices, made nondimensional by dividing by the tunnel static pressure and indicated by ϵ, is shown plotted against the nondimensional mass-flow coefficient C_w in figures 3(a), 3(b), and 3(c) for Mach numbers of 2.4, 3.0, and 4.0, respectively, for angles of attack of 0°, 5°, 10°, and 15°. For some of the test conditions the data exhibit substantial scatter and hysteresis, which were mainly the result of pressure lag due to the small-diameter tubing leading to the airspeed-head chambers.

Although there was some scatter, hysteresis, and, in some cases, a scarcity of data points in figure 3, it was believed that a straight-line fairing of the data was the best fit in view of the linearity of the results of reference 3, wherein there was a substantial number of data points with little scatter. The data of figure 3 show that the static-pressure error ϵ increased with increase of the flow coefficient C_w for both the forward and rear orifices at all three Mach numbers and all four angles of attack. The data also show that at a given value of flow coefficient the static-pressure error varies with both angle of attack and Mach number.

In order to show more clearly the effects of angle of attack and Mach number on the static-pressure error, the slopes of the curves $\left(\frac{d\epsilon}{dC_w}\right)$ of figure 3 at each Mach number are plotted in figure 4 against angle of attack for both the forward and rear orifices. In general there appears to be a substantial variation of static-pressure error with Mach number but only a small variation with angle of attack over the range covered. The reason for the somewhat larger variation with angle of attack for the rear orifice at $M_\infty = 4$ is not known.

The variation with Mach number of $\frac{d\epsilon}{dC_w}$ at an angle of attack of 0° is shown in figure 5. Also shown in figure 5 are two test points at Mach numbers of 0.915 and 1.125 ($\alpha = 0^\circ$) as obtained by Mabey (ref. 4) from tests of a British Mark 9A pitot-static head, a standard probe on British high-speed aircraft, which has two sets of static slots instead of multiple round static holes. The two test points, expressed in different quantities in reference 4, have been converted to correspond to the quantities of figure 5, and have been used as an aid in extending the fairing of the curve to a Mach number of zero. The curve shows a substantial increase in the static-pressure error with increase in Mach number. For example, there is an increase in $\frac{d\epsilon}{dC_w}$ from about 5 to about 12 as the Mach number increases from 2.4 to 4.0.

Because of the method used in determining the interference of the flow through the orifices with the external flow, the values of the pressure errors as determined by these tests include the loss across the orifices (from the chamber to the outside). Some bench tests made with no external flow but with flow through the orifices over the same range of conditions as in the tunnel tests indicated an average orifice discharge coefficient of about 0.66. On the basis of this discharge coefficient the loss across the orifices for an airspeed installation with a nondimensional flow coefficient C_w of, for example, 0.01 in flight, was computed to be 0.18, 0.36, and 0.95 percent of the free-stream static pressure at Mach numbers of 2.4, 3.0, and 4.0, respectively. In these estimates the temperature of the air in the airspeed system was assumed to be the stabilized surface temperature in the boundary layer with a recovery factor of 0.85. These losses in pressure across the orifices would presumably be accounted for in the usual pressure-lag corrections.

The significance of the results of figures 4 and 5 for a climbing airplane (flow out of the orifices) may be more readily indicated if the flow coefficient C_w is converted into a related parameter involving flight conditions. As has been shown in reference 2, one form of the mass-flow coefficient in terms of the flight quantities involved is

$$C_w = \frac{w}{\rho_\infty V_\infty A} = \frac{v}{ART_1} \sin \gamma$$

With this expression for flow coefficient, a quicker interpretation of the results of figures 4 and 5 may be made. For example, for an airplane in a vertical climb at high altitude for which the airspeed installation incorporates an airspeed indicator, a Mach meter, and an altimeter, the flow coefficient C_w is of the order of 0.01. For this value of flow coefficient and an angle of attack of 0°, the data of figure 5 indicate static-pressure errors of about 5 percent of free-stream static pressure at $M_\infty = 2.4$, about 8 percent at $M_\infty = 3.0$, and about 12 percent at $M_\infty = 4.0$. Because the variation of the static-pressure error with flow coefficient is linear, static-pressure errors for other values of flow coefficient C_w can be obtained directly by multiplying the values of $\frac{d\epsilon}{dC_w}$ from figures 4 and 5 by such values of C_w. The percentage error in Mach number is essentially one-half the static-pressure error for this range of Mach numbers and these relatively small pressure errors.

CONCLUSIONS

Results of tests to determine the static-pressure error resulting from external interference effects of flow through the static-pressure orifices of an NACA airspeed head at Mach numbers of 2.4, 3.0, and 4.0 and at angles of attack of 0°, 5°, 10°, and 15° have indicated that, within the accuracy of the measurements and for the range of mass flow covered, the static-pressure error increased linearly with increasing flow rate at all Mach numbers and angles of attack of the investigation. For a given value of flow coefficient, the static-pressure error varied appreciably with Mach number but only slightly with angle of attack. For example, for a flow coefficient of 0.01 (the approximate value for a vertically climbing airplane at high altitude for which the airspeed system incorporates an airspeed meter, a Mach meter, and an altimeter) the static-pressure error increased from about 5 percent to about 12 percent of free-stream static pressure as the Mach number increased from 2.4 to 4.0 with the airspeed head at an angle of attack of 0°.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., November 13, 1958.

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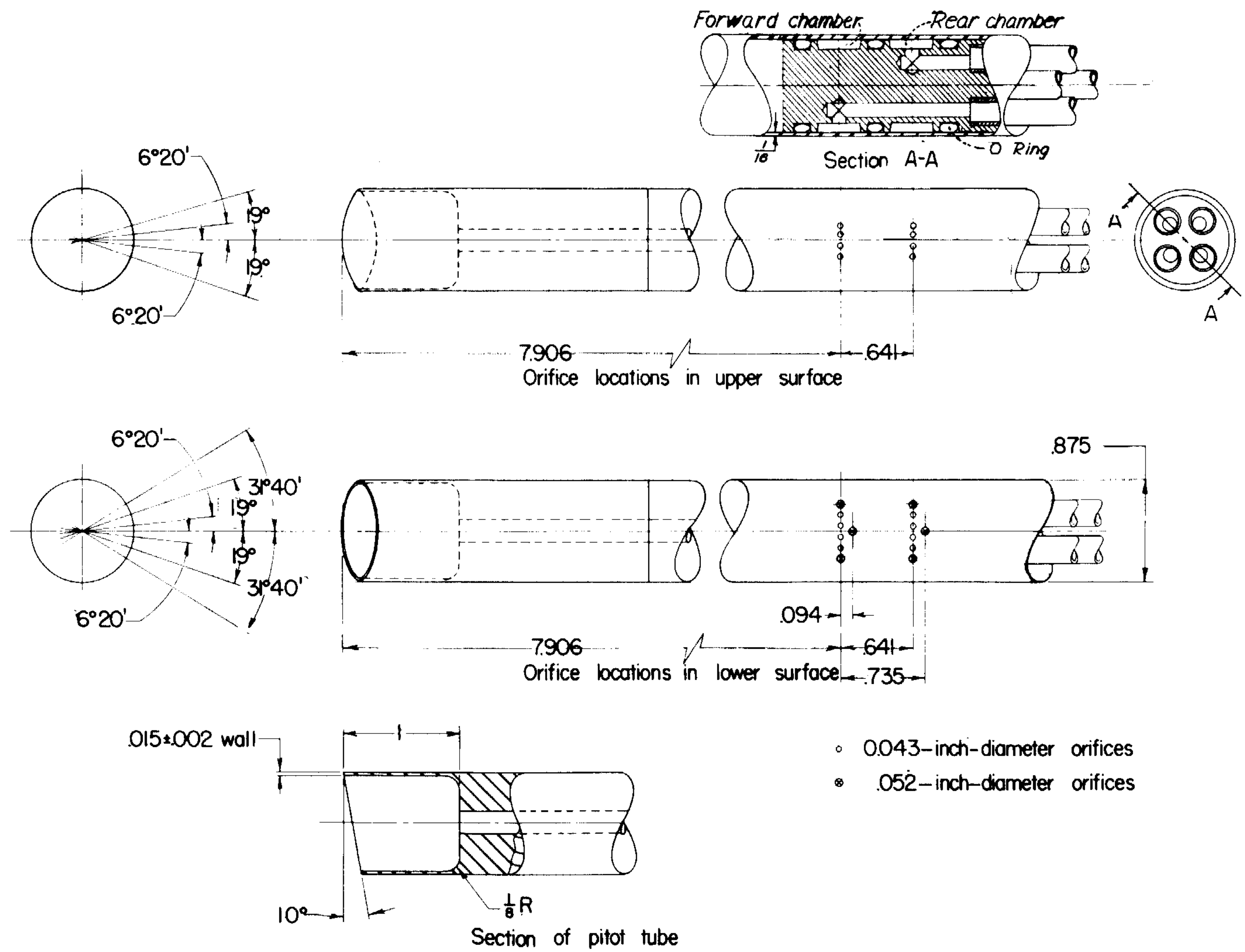


Figure 1.- Airspeed head as modified by insertion of an additional tube into each static-pressure chamber. Pitot tubing has been removed and pitot opening plugged. All linear dimensions are in inches.

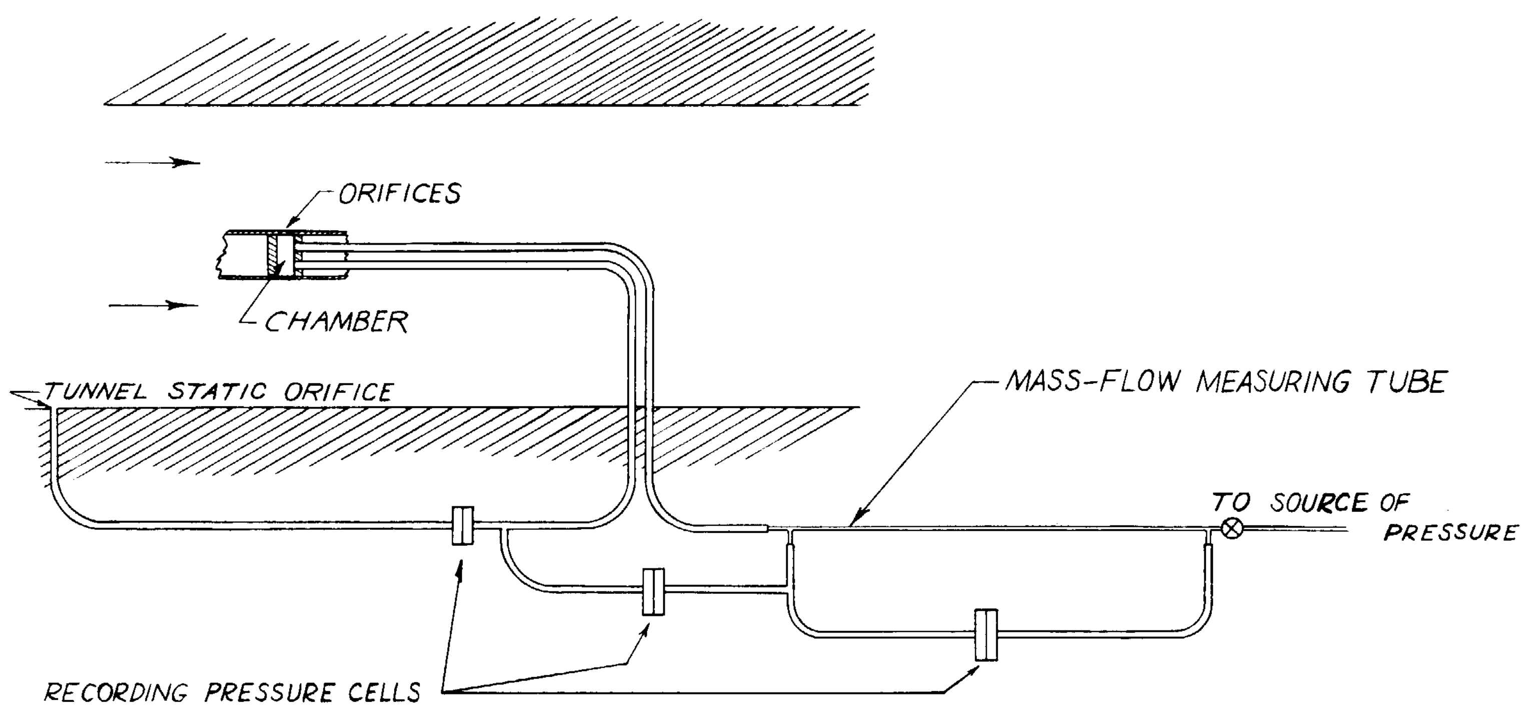
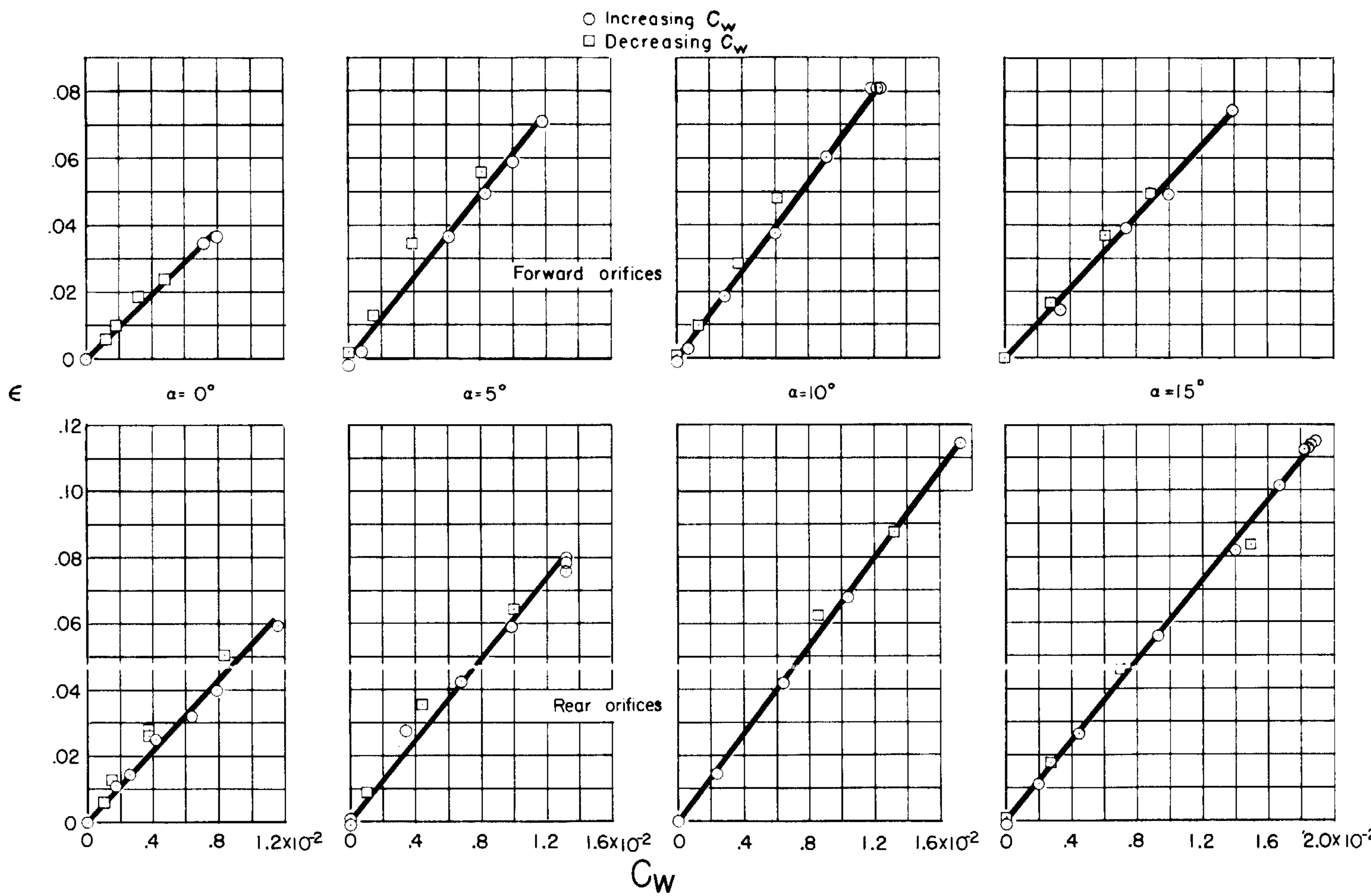
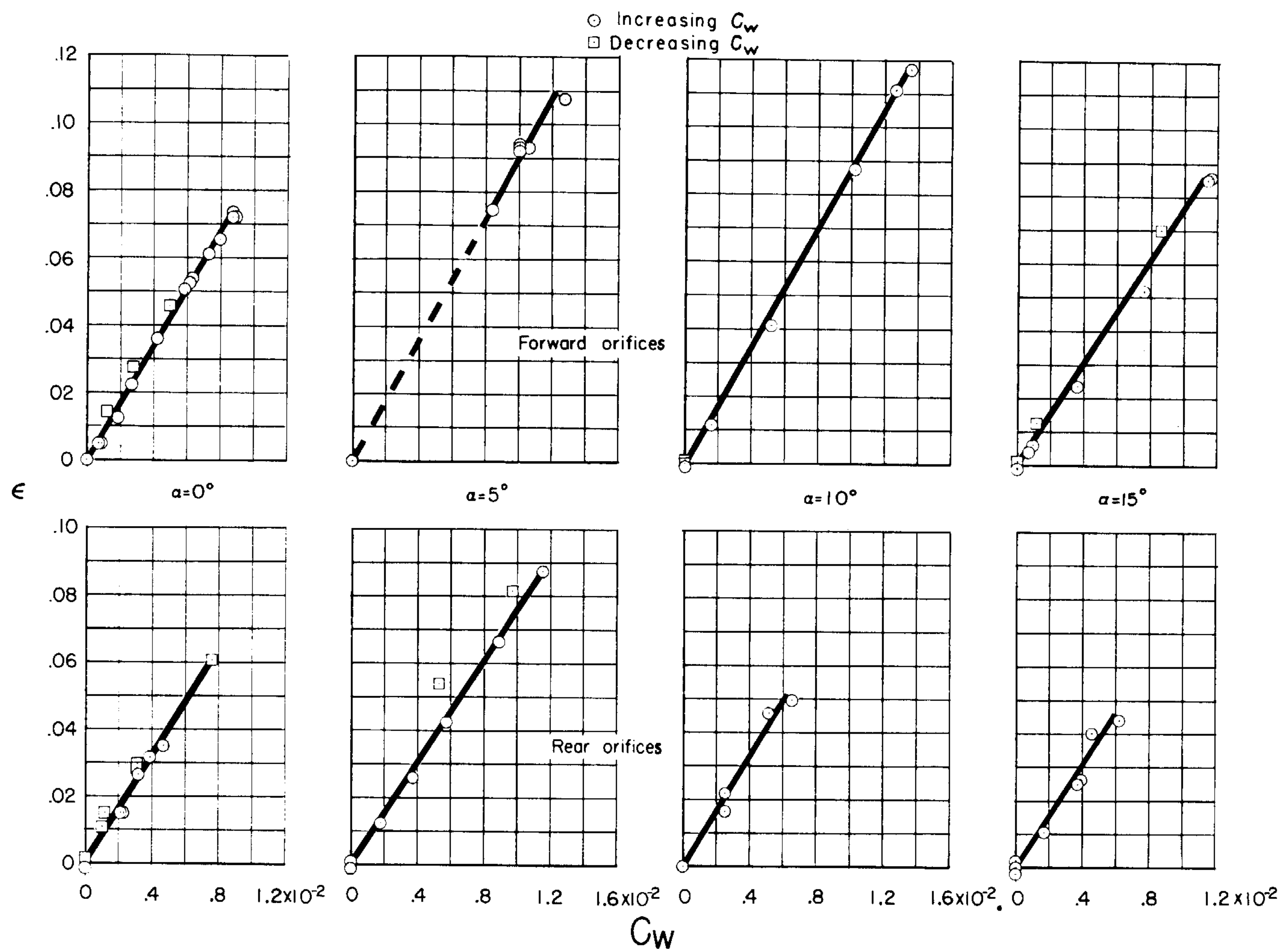


Figure 2.- Sketch showing mass-flow and pressure-measuring setup for one chamber of airspeed head.



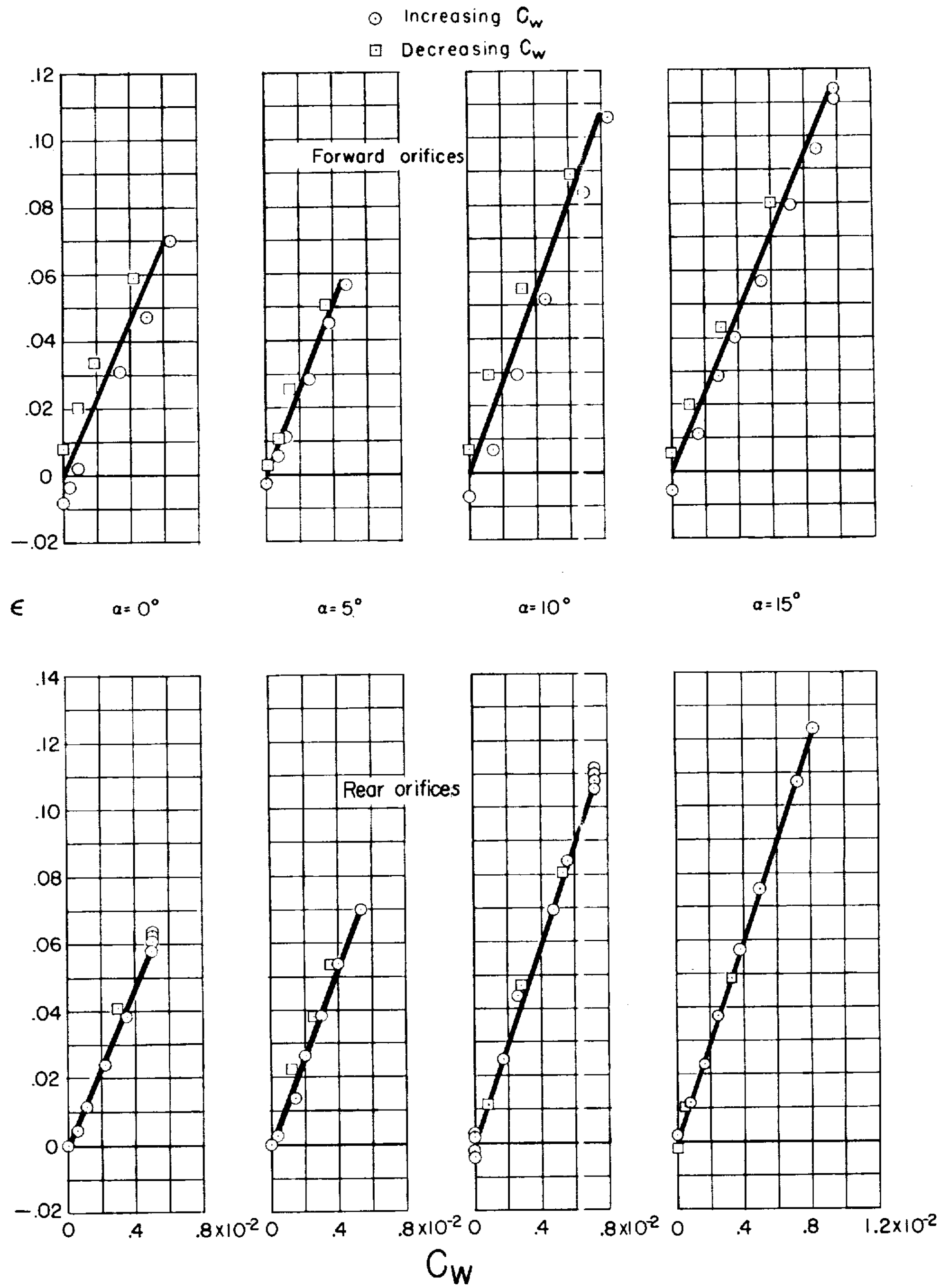
(a) $M_\infty = 2.4$.

Figure 3.- Variation of static-pressure error ϵ of airspeed head with mass-flow coefficient, C_W.



(b) $M_\infty = 3.0$.

Figure 3.- Continued.



(c) $M_\infty = 4.0.$

Figure 3.- Concluded.

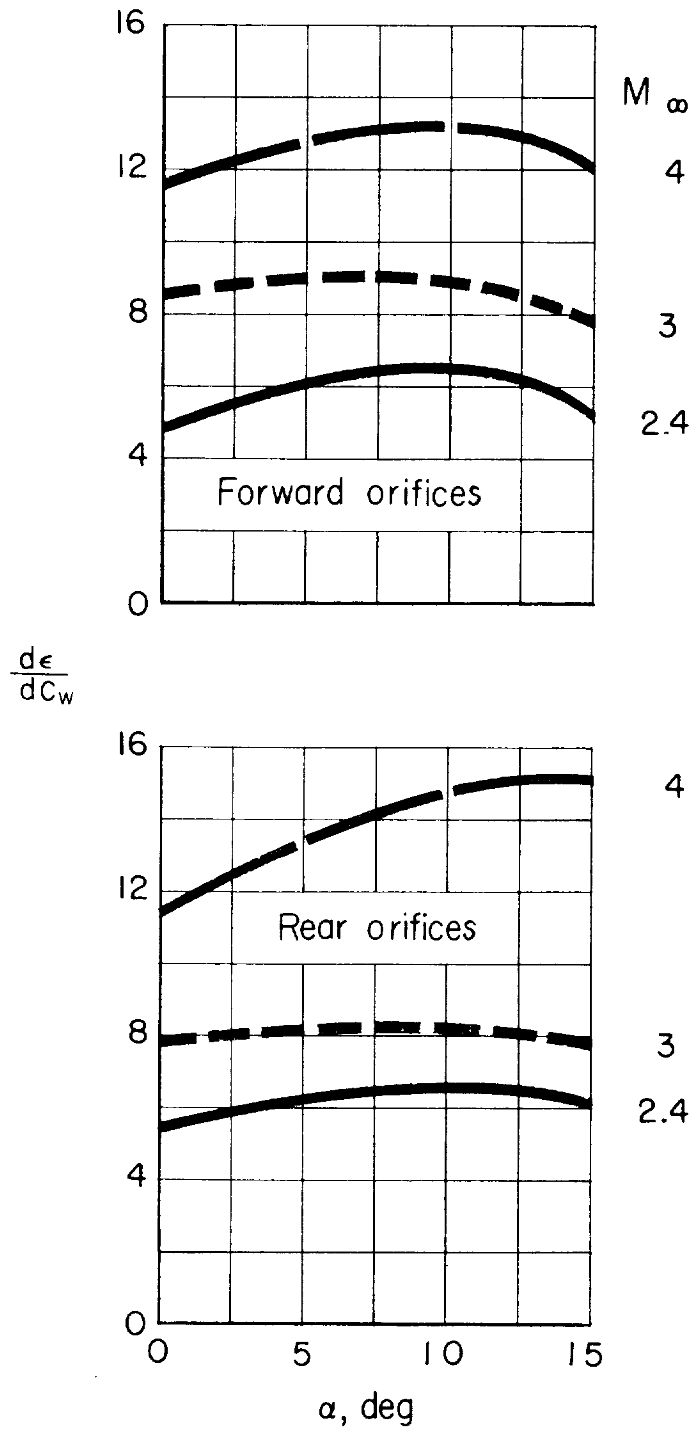


Figure 4.- Variation with angle of attack of rate of change of static-pressure error with rate of change of mass-flow parameter.

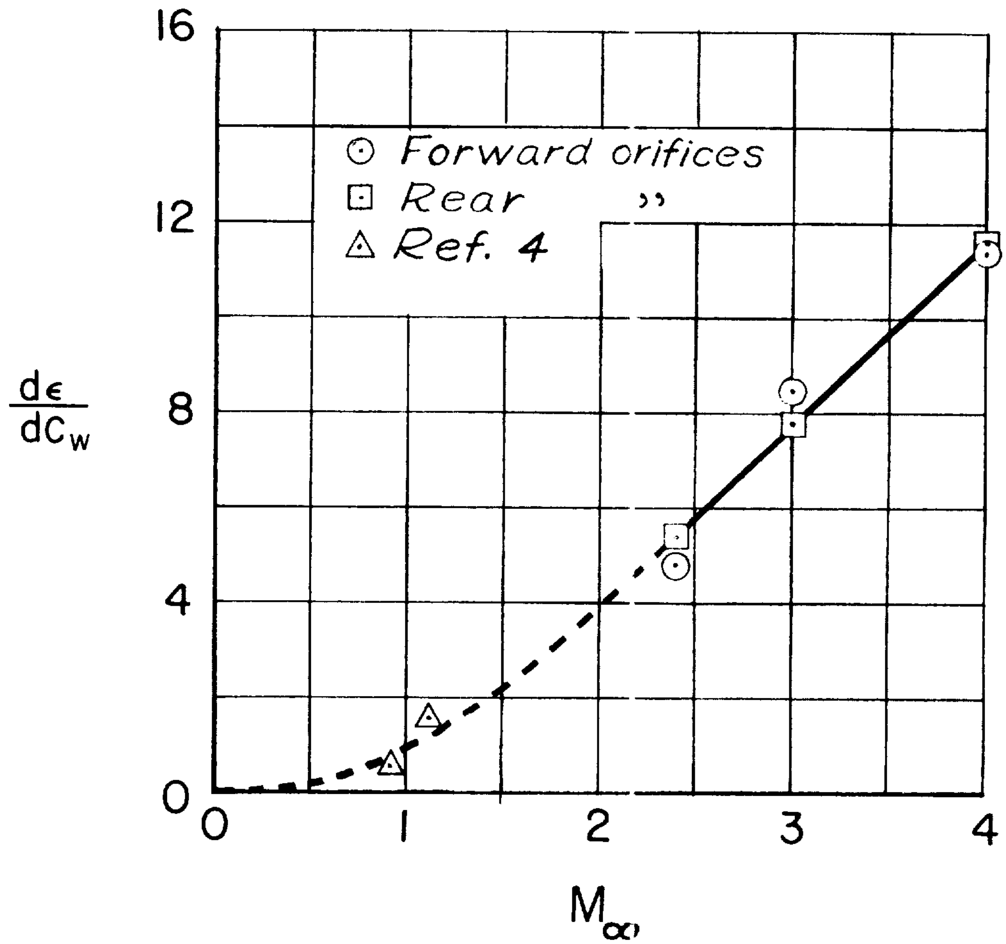


Figure 5.- Variation with Mach number of rate of change of static-pressure error with rate of change of mass-flow coefficient. $\alpha = 0^{\circ}$.