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A CATALOG OF DEVICES APPLICABLE TO THE MEASUREMENT OF BOUNDARY
LAYERS AND WAKES ON FLIGHT VEHICLES

CASE FILE
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A literature search was conducted to assemble a catalog of devices and techniques which have possible application to boundary layer and wake measurements on flight vehicles. The indices used in the search were NACA, NASA STAR, IAS, USGRDR and Applied Science and Technology Index. The period covered was 1950 through 1970. The devices contained in the catalog were restricted to those that provided essentially direct measurement of velocities, pressures and shear stresses. Pertinent material was included in the catalog if it contained either an adequate description of a device and associated performance data or a presentation of applicable information on a particular measurement theory and/or technique. When available, illustrations showing the configuration of the device and test condition data were also included.					
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INTRODUCTION

A need exists for instrumentation to measure boundary layers and wakes of flight vehicles. In most cases, the measurement program requires probes or other devices which must be developed specifically for the task at hand. The effort involved in such development can usually be reduced considerably if an existing device can be modified or at least serve as a source of ideas. With this in mind, a literature search was conducted to assemble a catalog of devices and techniques which have possible application to boundary layer and wake measurements on flight vehicles. This catalog is restricted to the devices that provide essentially direct measurement of velocities, pressures and shear stresses.

The following indices were used: NACA, NASA STAR, IAS, USGRDR and Applied Science and Technology Index. The period covered was 1950 through 1970.

Pertinent material was included in the catalog if it contained either an adequate description of a device and associated performance data or a presentation of applicable information on a particular measurement theory and/or technique. Graphical data were reproduced directly from the reference sources; this resulted in some variation in nomenclature. Definitions were added when necessary.

The catalog is assembled in three sections. The first section contains the listings pertaining to probes and other devices for which adequate information is provided in the referenced publication to allow an initial evaluation of the pertinent geometric and performance characteristics of the device. Each entry contains the title, author, publication identification, accession number and a summary of the publication. Where possible, the test conditions are also given as well as illustrations showing the configuration of the device and performance data. It is assumed that the final evaluation of the suitability of the device will require acquisition of the listed publication by the user. The second section contains publications providing general information pertaining to devices, calibration techniques and measurement theory. A significant portion of this section is devoted to those references providing information on the measurement of unsteady flow characteristics. Emphasis is given to the general area starting with the most recent publication dates. Special attention should be directed to the entries in the general reference categories, B.1. and B.7. The third section is labeled Bibliography and contains an author reference list to the entries included in the catalog. As an aid to those wishing to add to this catalog, lists are presented of those publications which were reviewed and rejected and of those publications which were not acquired in time for review.

A. MEASUREMENT DEVICES WITH PERFORMANCE DATA

1. Pneumatic Probes Suitable for Wake Measurements

No. 1

TITLE: Characteristics of Wake Pressure Type Velocity Head Magnifying Probes

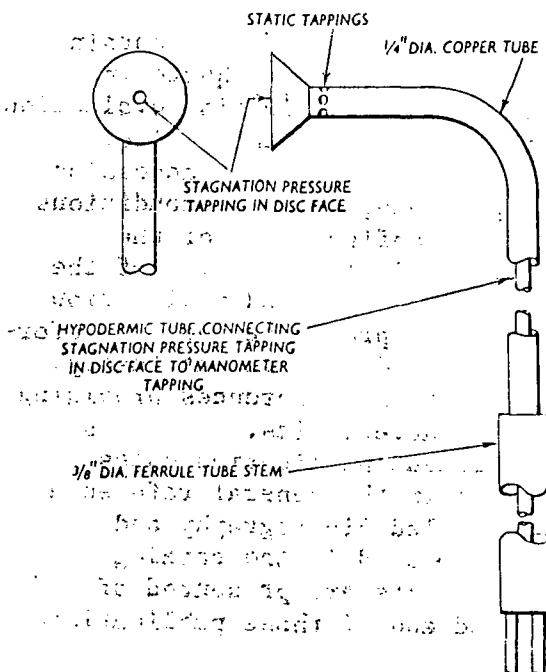
AUTHOR: A. J. S. Pratt, T. B. Fergusson and R. G. Bedlow

PUBLICATION: The Engineer, vol. 223, May 26, 1969, A67 29529

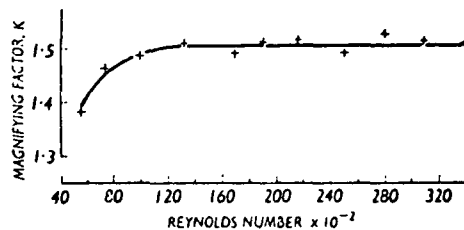
SUMMARY: A disc type magnifying probe is described, the magnifying factor of which is relatively insensitive to both stem and Reynolds number effects.

TEST CONDITIONS: Velocity, 0-90 ft/sec; Reynolds Number based on disk diameter, $5 \times 10^3 - 3.4 \times 10^4$; yaw angle, $\pm 30^\circ$; pitch angle, assumed 0° .

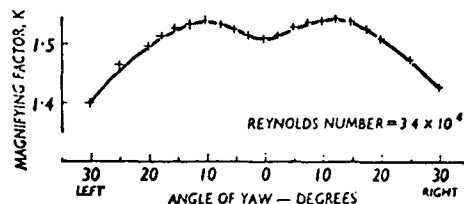
DEFINITION: $K = \Delta P / \frac{1}{2} \rho V^2$



Disc Type Magnifying Probe



Reynolds Number Characteristic of Disc Probe



Yaw Characteristic of Disc Probe

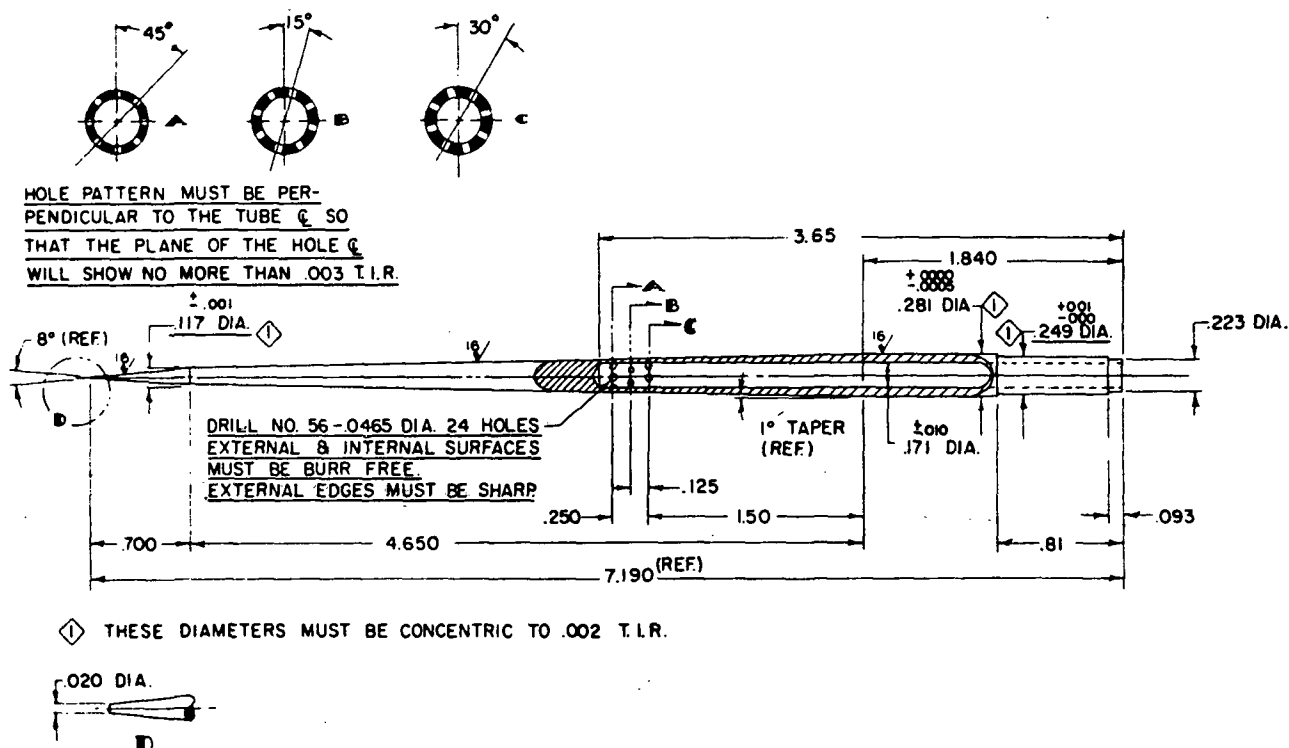
TITLE: A Conical Static Pressure Probe

AUTHOR: H. R. Vaughn

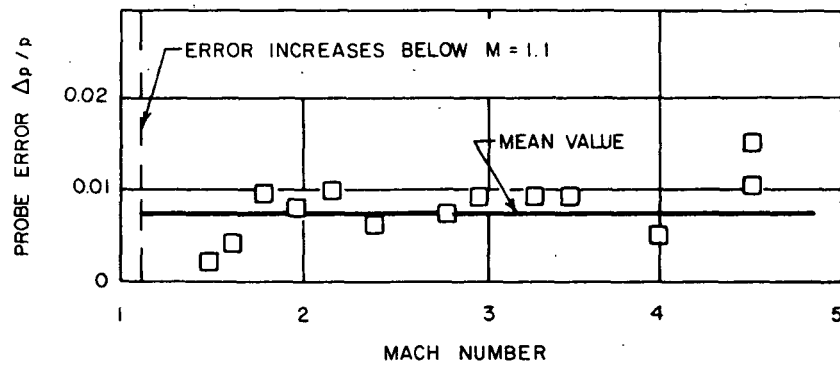
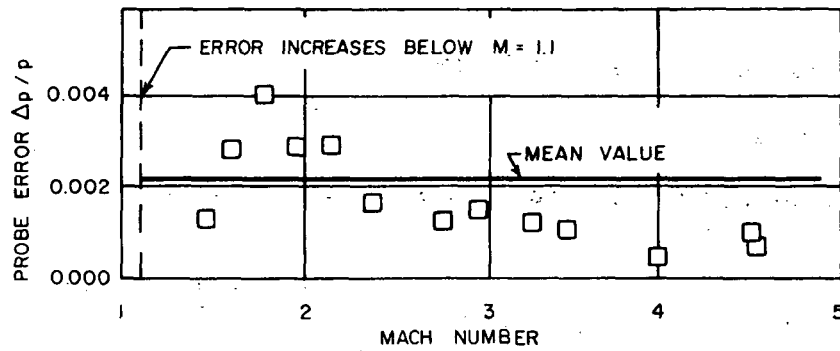
PUBLICATION: Journal of the Aerospace Sciences, vol. 28, October 1961,
A61 8618

SUMMARY: A conical probe, only 19-diameters long, which measures static pressure with good accuracy within the range $1.1 < M < 4.54$ is presented. Its strength is such that flight use is practical up to the point where aerodynamic heating becomes excessive.

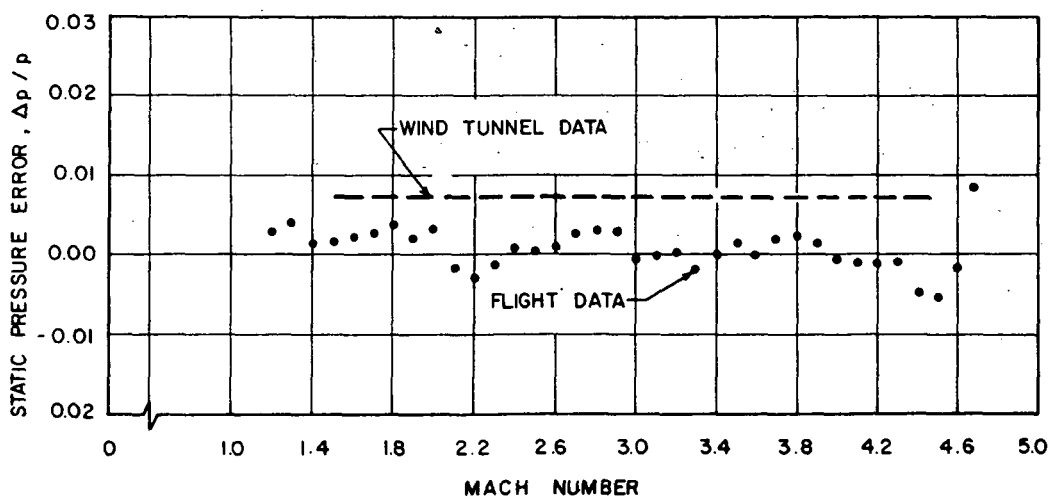
TEST CONDITIONS: Mach Number, 1.1 - 4.54; Reynolds Number, not available; yaw angle, 0°; pitch angle, 0°.



Conical Static Probe. Dimensions in Inches



Error in Measured Static Pressure



Comparison of Wind Tunnel Average Reading and Flight Tests

TITLE: Wind-Tunnel Investigation of a Number of Total-Pressure Tubes at High Angles of Attack - Subsonic, Transonic, and Supersonic Speeds

AUTHOR: William Gracey

PUBLICATION: NACA TN 3641, May 1956

SUMMARY: The effect of inclination of the airstream on the measured pressures of 54 total-pressure tubes has been determined for angles of attack up to 60° and over a Mach number range from 0.26 to 1.62. The investigation was conducted in five wind tunnels at the Langley Aeronautical Laboratory.

For simple, nonshielded tubes, the usable angular range was found to depend on the external shape of the nose section, the size of the impact opening (relative to the tube diameter), and the shape of the internal chamber behind the impact opening. The best combination of these design features (that is, a tube having a cylindrical nose shape, an impact opening equal to the tube diameter, and a 30° conical chamber) produced the highest usable range (about 28° at a Mach number of 0.26) of any of the unshielded tubes.

In cases where a usable angular range larger than $\pm 28^\circ$ is required, a much higher range may be obtained with shielded tubes. The usable range of a tube with a shield having a conical entry, for example, is about $\pm 41^\circ$ at a Mach number of 0.26. Changing the shape of the entry of the shield to a highly curved section increases the usable range to about $\pm 63^\circ$, the highest of any of the tubes tested. The throats of these tubes were vented through the walls of the shield, a design feature permitting end-mounting of the tube on a horizontal boom.

For airspeed applications where it is not essential that the tube have a large negative angle-of-attack range, the usable range may be extended to higher positive angles of attack by means of a leading-edge slant profile. For slant profiles up to 20° , the usable range at positive angles of attack can be extended by about 1° for each degree of slant.

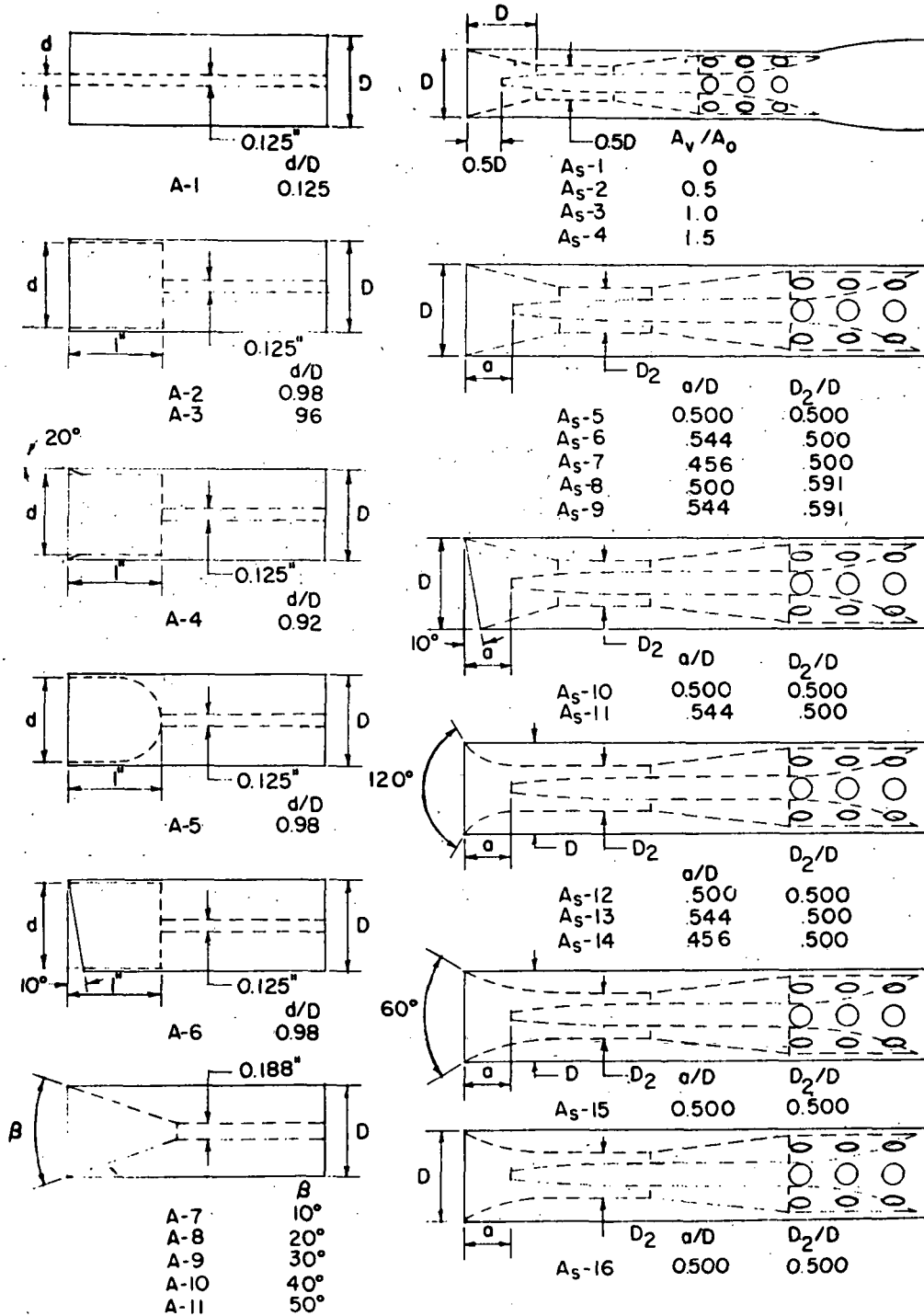
The effect of Mach number on both unshielded and shielded tubes was found to be considerable. For most of the unshielded tubes the usable angle-of-attack range increased with Mach number, whereas that of the shielded tubes decreased with Mach number.

TEST CONDITIONS: See table on following page.

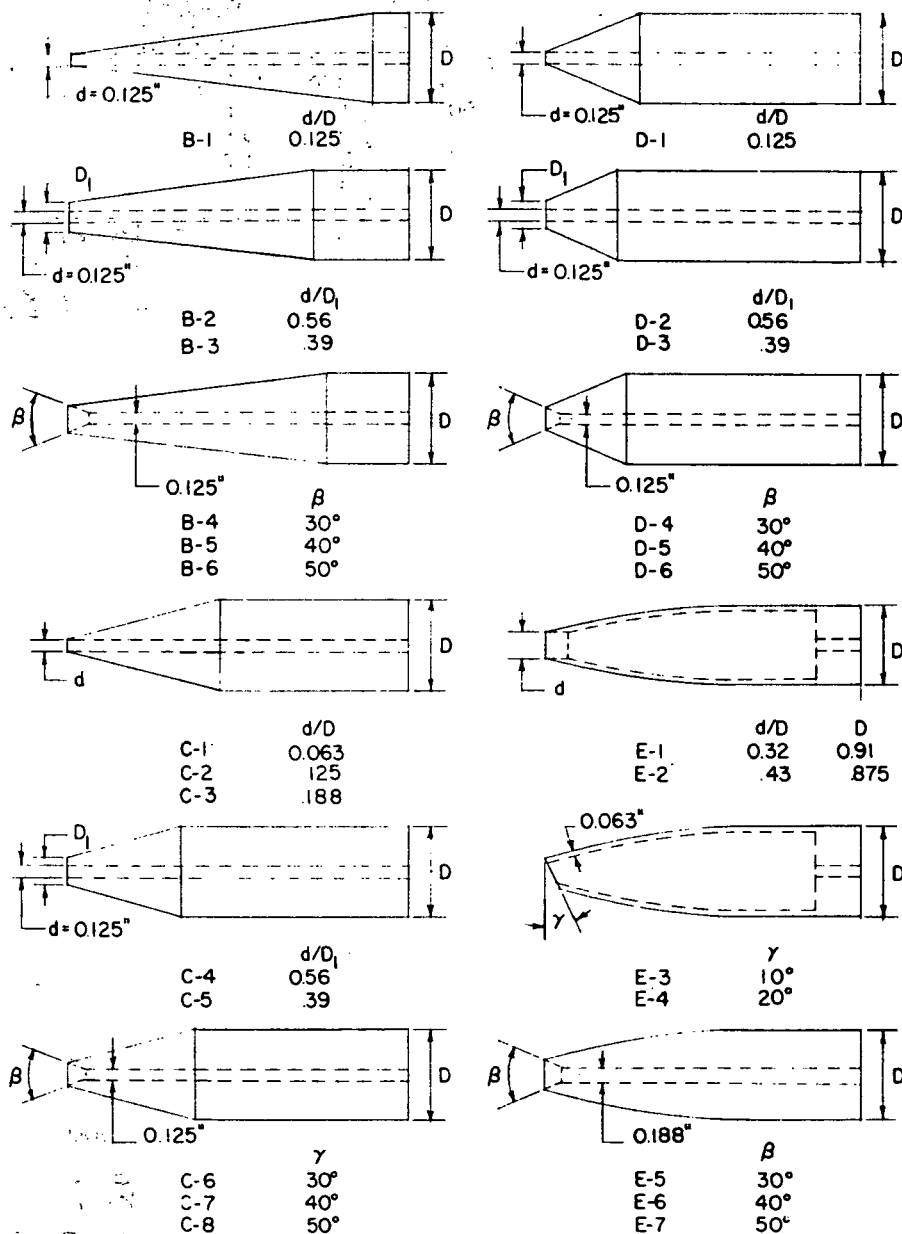
Tube	Range of angle of attack for Mach numbers of -										
	0.26	0.5	0.6	0.7	0.8	0.9	0.95	1.00	1.05	1.10	1.61
Cylindrical tubes											
A-1	±11										±11
A-2	±23										±29
A-3	±23										±29
A-4	±23										±27.5
A-5	±25.5										
A-6	-13, 32										±20, 38
A-7	±28.0										±32
A-8	±27.5										±31.5
A-9	±27.5										±30
A-10	±25.5										
A-11	±25.5										
Shielded tubes											
A _s -1	±24										
A _s -2	±38.5										
A _s -3	±41										
A _s -4	±41.5										
A _s -5	±41.3	±40.4		±39.5		±38.6					
A _s -6	±40.1										
A _s -7	±41.5										
A _s -8	±44.0										
A _s -9	±42.5										
A _s -10	±31.6, 51					±30.7, 46.5					
A _s -11	±32.5, 47.7										
A _s -12	±63.2	±63.3		±60.6		±57.1	±56.6	±57.0	±56.3	±55.7	±40.0
A _s -13	±63.8										
A _s -14	±63.8										
A _s -15	±51.4					±45.5					
A _s -16	±53.9					±50.6					
15° conical tubes											
B-1	±21										±29
B-2	±13										±17
B-3	±10.5										±14
B-4	±21										±29.5
B-5	±19										±28.5
B-6	±18										±27.5
30° conical tubes											
C-1	±15.5										
C-2	±17.5										
C-3	±19										
C-4	±10.5										
C-5	±9										
C-6	±20.5										±28.5
C-7	±18										±28
C-8	±17.5										±27.5
45° conical tubes											
D-1	±14										
D-2	±9.5										
D-3	±8										
D-4	±15.5										
D-5	±15										±24.5
D-6	±13.5										
Ogival tubes											
E-1	±16										
E-2			±21	±21.5	±22.2	±22.8	±23.1	±23.5	±23.8	±24.2	
E-3	-1.5, 11.5										
E-4	8.5, 21.5										
E-5	±19										±28
E-6	±18.5										
E-7	±18										

*Estimated.

Range of Angle of Attack Over Which Total-Pressure Tubes Remain Insensitive to Inclination to Within 1 Percent ($P_{\text{Total}} - P_{\text{Static}}$)



Diagrams of Total-Pressure Tubes
 Series A (cylindrical) and A_s (shielded)
 (Vent area A_v/A_0 of tubes A_s-4 to A_s-16 is 1.5)



Diagrams of Total-Pressure Tubes(continued)
 (b) Series B (15° conical); series C (30° conical);
 series D (45° conical); and series E (ogival)

TITLE: Improved Calibration Method for a Five-Hole Spherical Pitot Probe

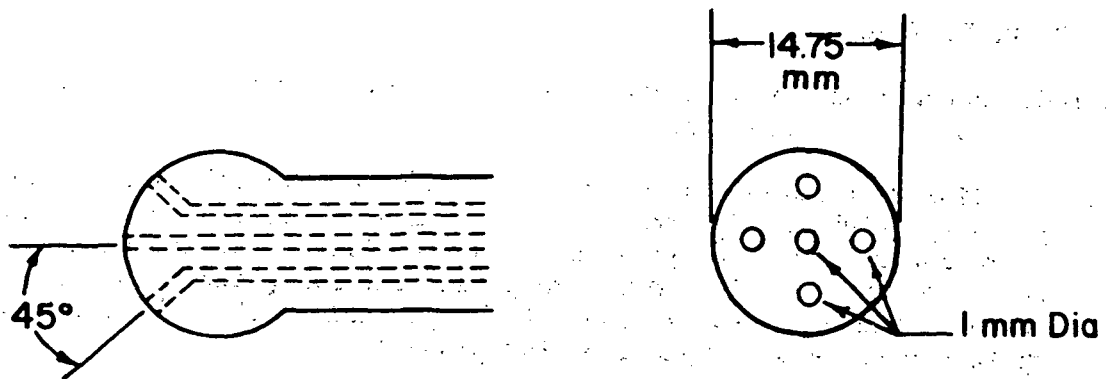
AUTHOR: C. F. R. Nowack

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments,
vol. 3, January 1970, A70 17621

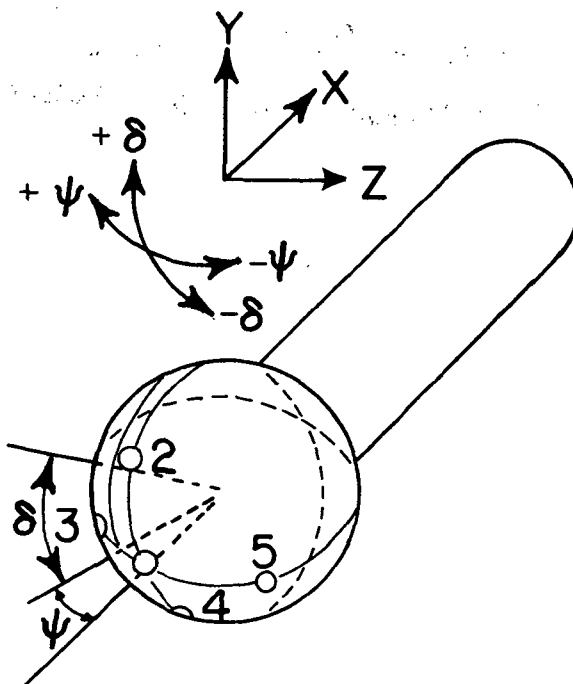
SUMMARY: A calibration method is described for a five-hole spherical pitot probe by which the direction of a velocity vector is fixed by two Cartesian angles, one lying in the horizontal equatorial plane and the other in a vertical meridian plane containing the velocity vector. Measurements are found to be possible within the range of up to 65° half-cone angle from the probe axes.

TEST CONDITIONS: Velocity, 2-40 m/sec; Reynolds Number based on sphere diameter, $2.3 \times 10^2 - 4.1 \times 10^4$; yaw angle, $\pm 70^\circ$; pitch angle, $\pm 90^\circ$.

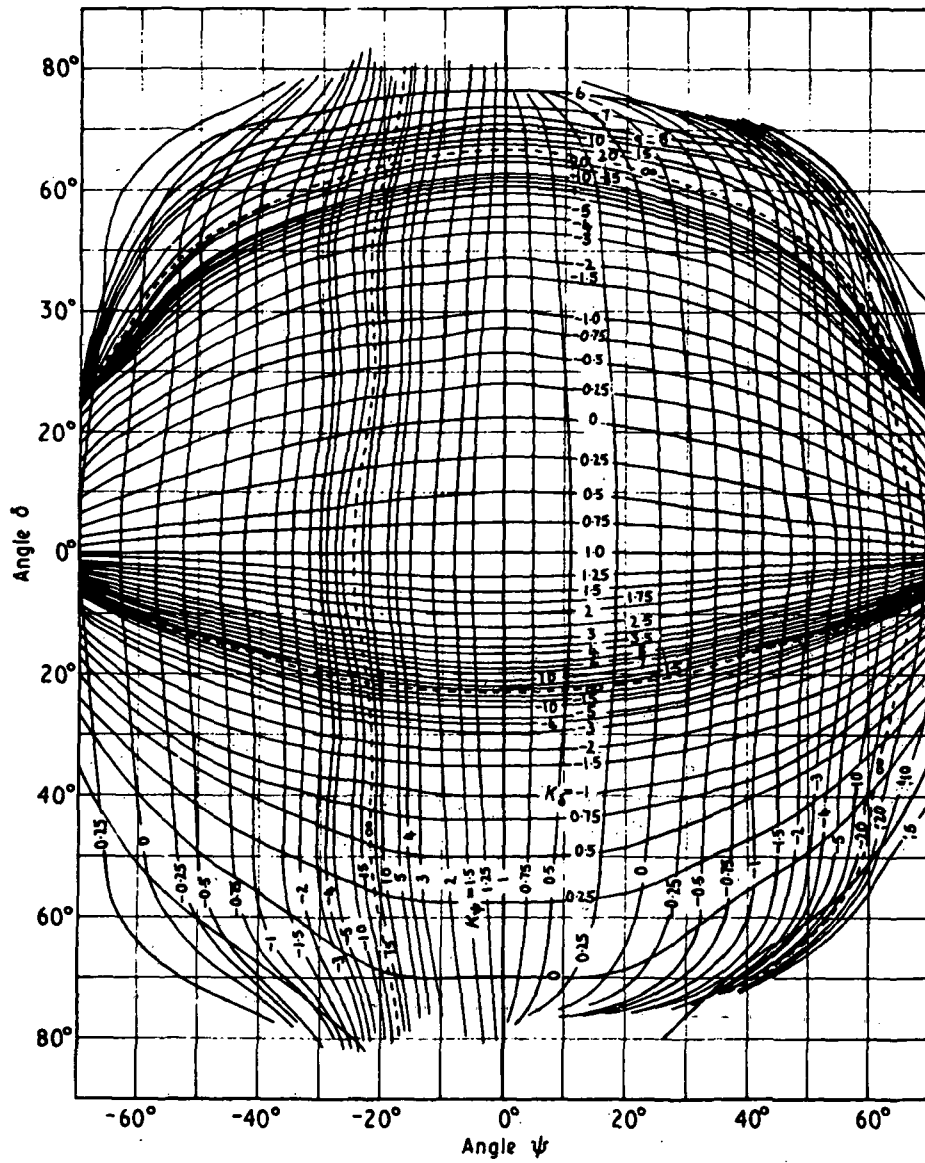
DEFINITIONS: $K_\delta \equiv (P_1 - P_2)/(P_1 - P_4)$ Inclination Factor
 $K_\psi \equiv (P_1 - P_3)/(P_1 - P_5)$ Inclination Factor
 $P_{st_1} \equiv (P_1 - P_{Static})/(P_{Total} - P_{Static})$ Static Pressure Factor
 $V_{12} \equiv (P_1 - P_2)/(P_{Total} - P_{Static})$ Velocity Factor



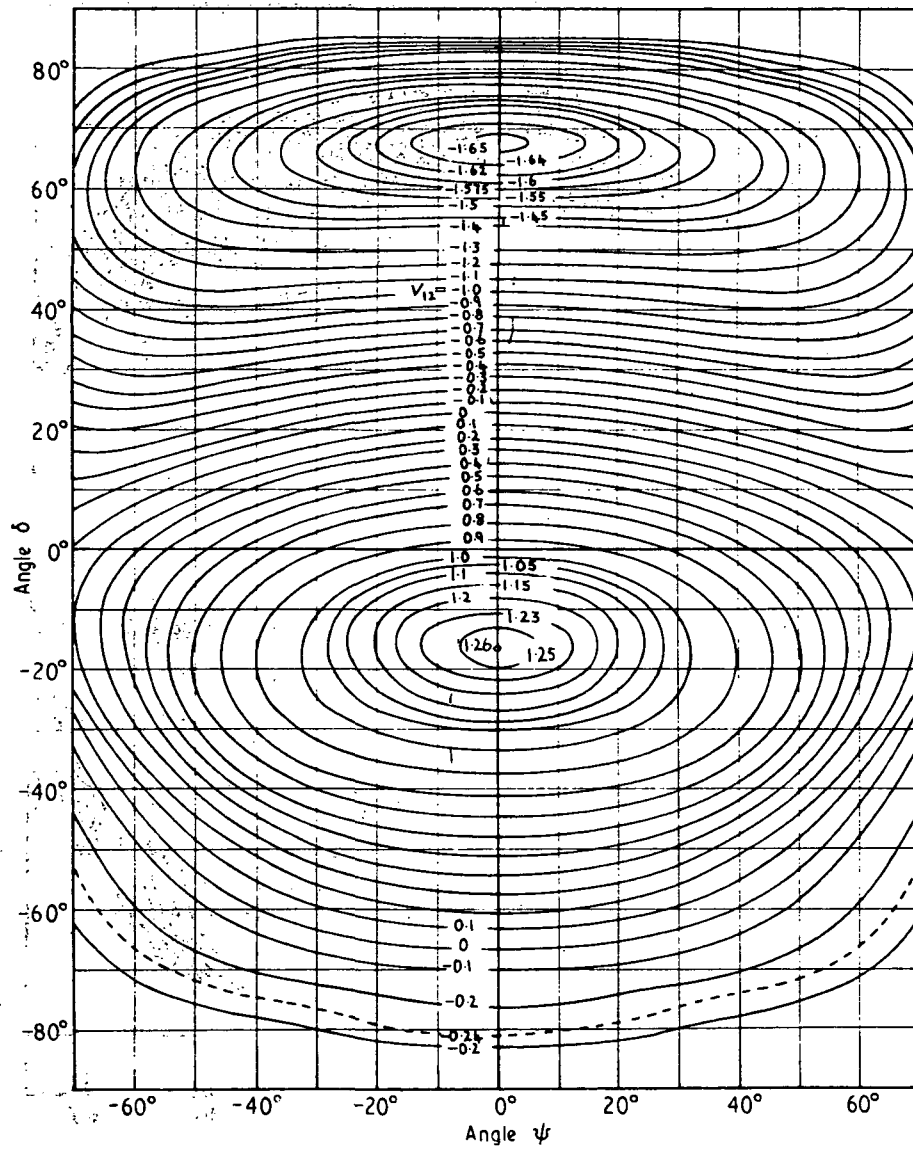
Spherical Pitot Probe Head



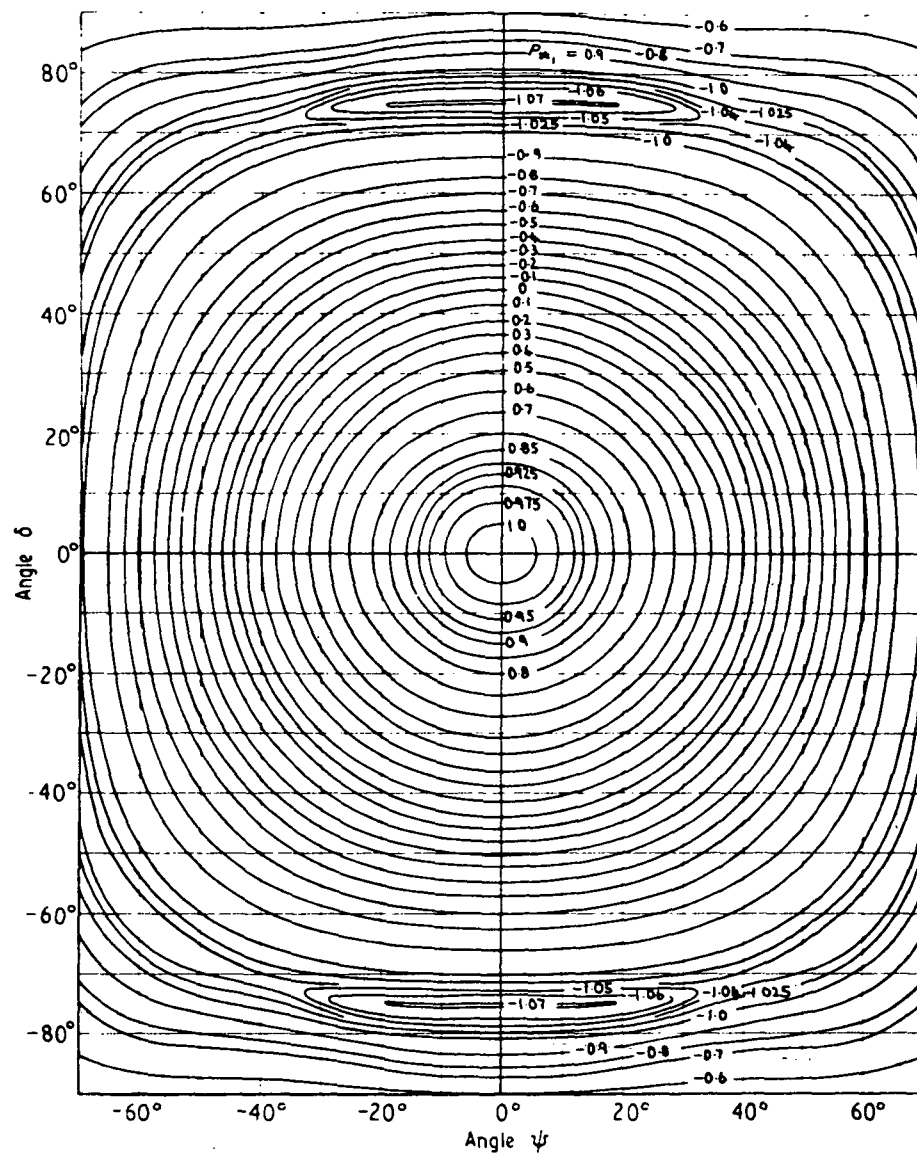
Relation Between Yaw and Pitch Angles
With Velocity Vector Configuration



Calibration Curves for the Inclination Factors K_δ and K_ψ



Calibration Curves for the Velocity Factor V_{12}



Calibration Curves for the Static Pressure Factor P_{st1}

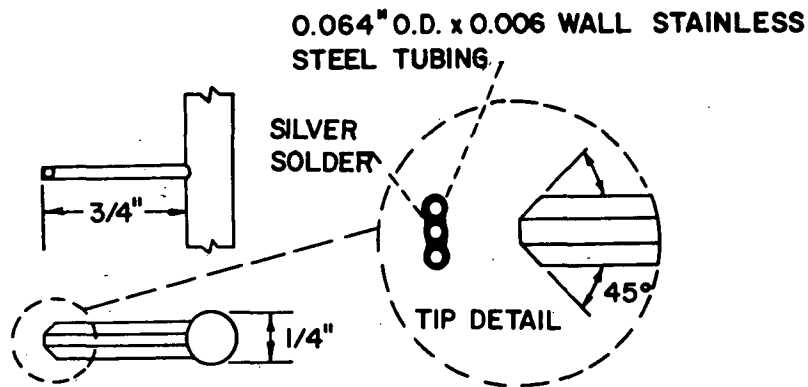
TITLE: Flow-Direction Measurement With Fixed-Position Probes in Subsonic Flow Over a Range of Reynolds Numbers

AUTHOR: Lloyd N. Krause and Thomas J. Dudzinski

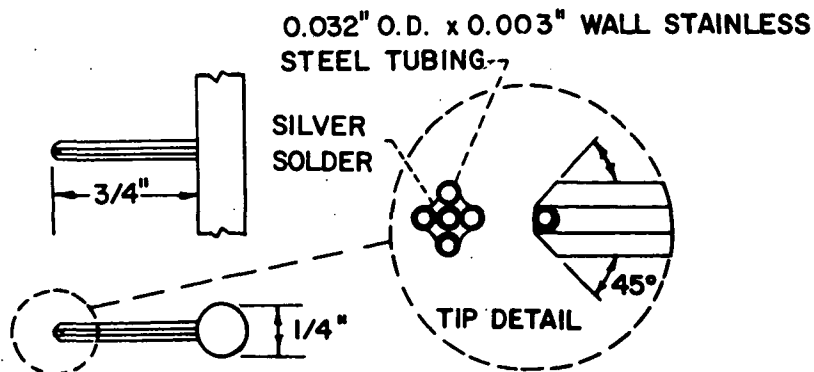
PUBLICATION: NASA TM X 52576, 1969, N69 40059

SUMMARY: Flow measurements are presented on two-dimensional and three-dimensional total pressure probes for systematic variations in pitch and yaw. These probes allow the measurement of flow angle within one degree for the flow angles tested.

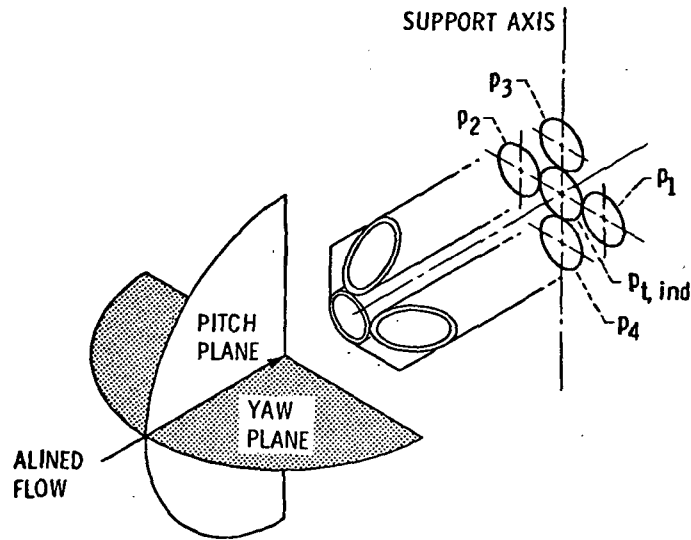
TEST CONDITIONS: Mach Number, 0.3 - 0.9; Reynolds Number based on probe outside diameter, $10^3 - 4 \times 10^4$; yaw angle, $\pm 30^\circ$; pitch angle, $\pm 30^\circ$.



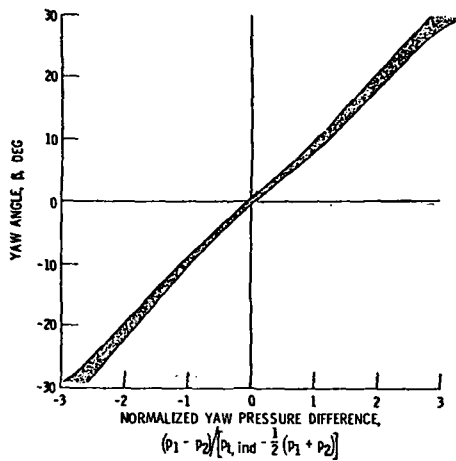
Two-Dimensional Probe



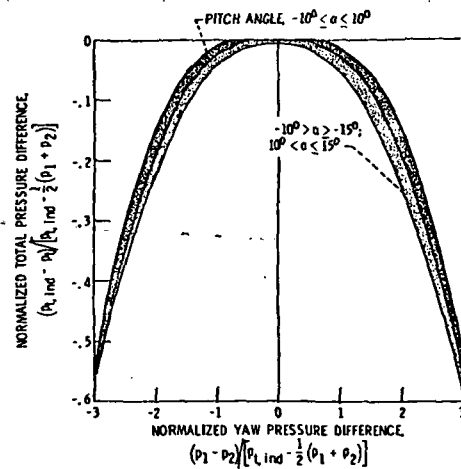
Three-Dimensional Probe



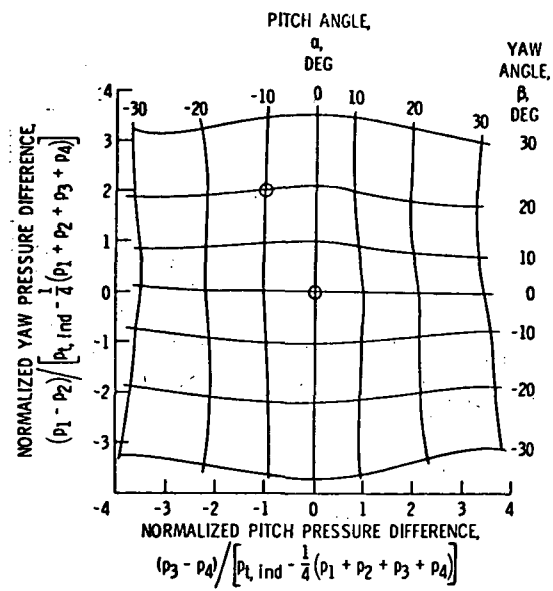
Flow Orientation and Pressure
Tube Nomenclature



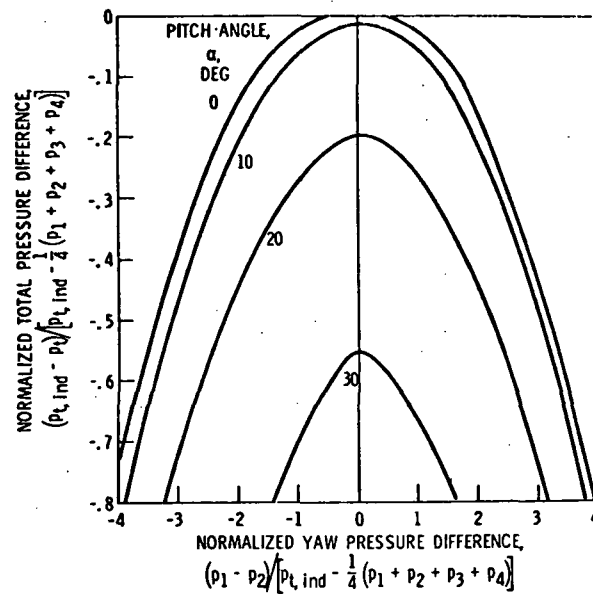
Variation in Yaw Pressure
Difference with Yaw Angle



Variation in Total Pressure
Difference with Yaw Pressure
Difference for 2-D Probe



Pressure Difference Map for 3-D Probe



Variation in Total Pressure Difference With Yaw Pressure Difference for 3-D Probe

TITLE: A Small Combination Sensing Probe for Measurement of Temperature, Pressure, and Flow Direction

AUTHOR: George E. Glawe, Lloyd N. Krause, and Thomas J. Dudzinski

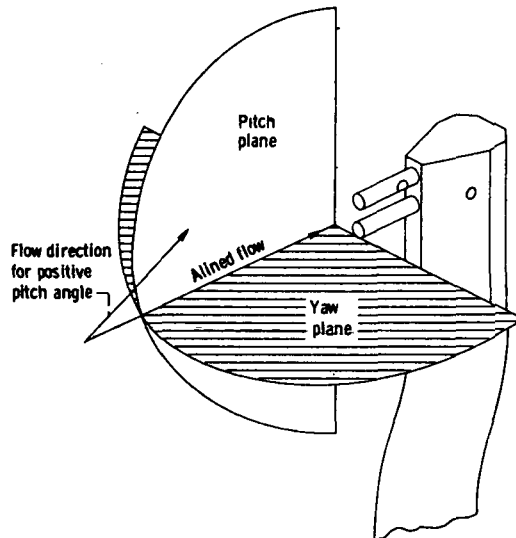
PUBLICATION: NASA TN D-4816, October 1968, N68 36021

SUMMARY: Probe flow characteristics are presented over a range of subsonic Mach numbers as well as for Mach 1.4, and over a range of yaw and pitch angles.

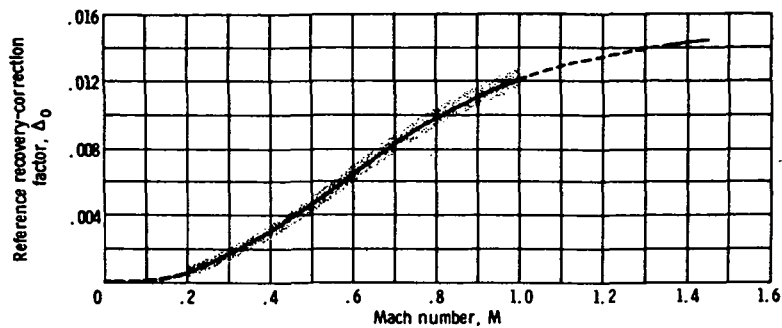
NOTE: Shaded area on graphs is the range of variation for three probes, the solid line is the average for three probes.

TEST CONDITIONS: Mach Number, 0.2 - 0.9 and 1.4; Reynolds Number, $10^6 - 10^7$ per ft., $3 \times 10^4 - 3 \times 10^5$ per cm; yaw angle, $\pm 22^\circ$; pitch angle, $\pm 25^\circ$.

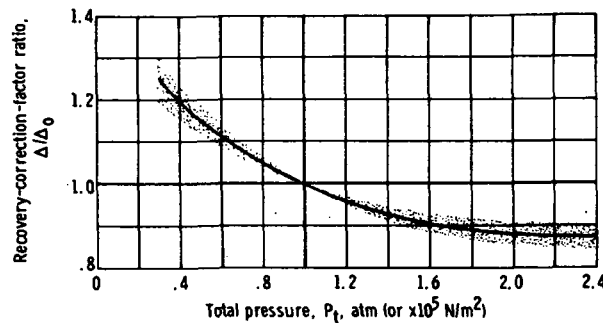
DEFINITION: $\Delta \equiv (T_{TOTAL} - T_{IND}) / (T_{TOTAL})$



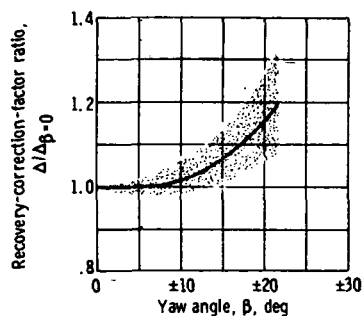
Flow Orientation Nomenclature



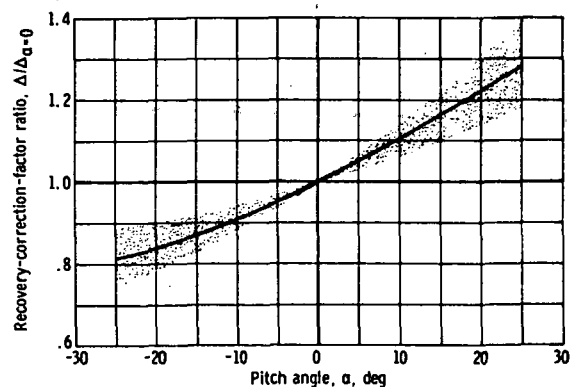
Variation of Reference Recovery-Correction Factor With Mach Number. Δ_0 is the Recovery-Correction Factor at a Total Pressure of 1 Atmosphere ($1 \times 10^5 \text{ N/m}^2$). Alined Flow



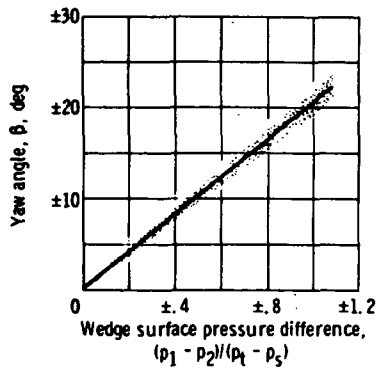
Variation of Recovery-Correction-Factor Ratio With Pressure. Δ_0 is the Recovery-Correction Factor at 1 Atmosphere ($1 \times 10^5 \text{ N/m}^2$). Alined Flow. $0.6 \leq M \leq 0.9$ and $M=1.4$



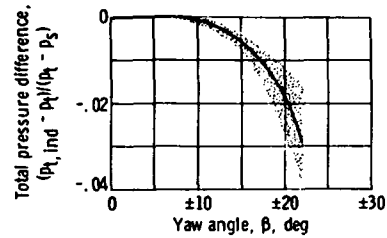
Variation of Recovery-Correction-Factor Ratio With Yaw Angle. $0.3 \leq M \leq 0.9$ and $M = 1.4$



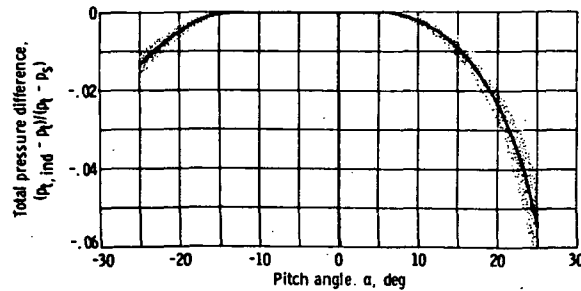
Variation of Recovery-Correction-Factor Ratio With Pitch Angle $0.3 \leq M \leq 0.9$



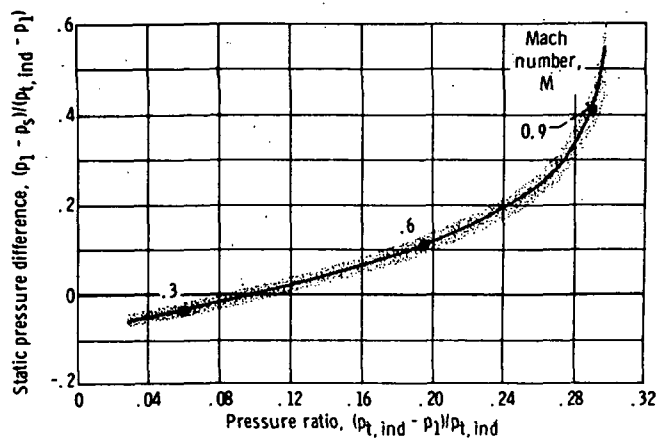
Variation in Wedge Surface Pressure Difference With Yaw Angle. $0.3 \leq M \leq 0.9$



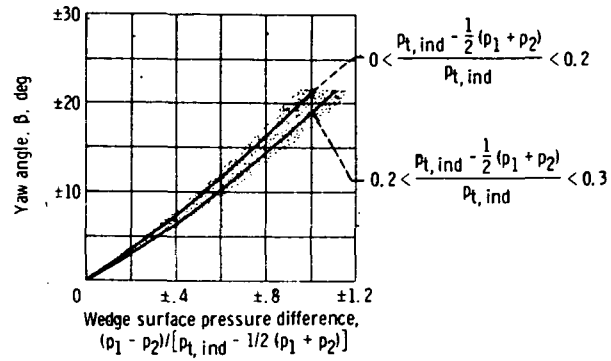
Variation in Total Pressure Difference With Yaw Angle. $0.3 \leq M \leq 0.9$



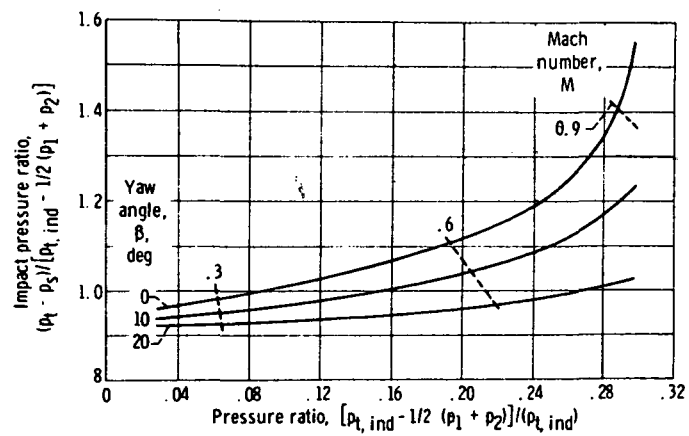
Variation in Total Pressure Difference With Pitch Angle. $0.3 \leq M \leq 0.9$



Variation in Static Pressure Difference With Probe Measurables. Alined Flow



Variation to Wedge Surface Pressure Difference, in Terms of Probe Measurables, With Yaw Angle. $0.3 < M < 0.9$



Variation in Impact Pressure With Probe Measurables

No. 7

TITLE: Characteristics of a 40° Cone for Measuring Mach Number, Total Pressure, and Flow Angle at Supersonic Speeds

AUTHOR: Frank J. Centolanzi

PUBLICATION: NACA TN 3967, May 1957

SUMMARY: Pressure measurements were taken at angles of pitch up to 26° at Mach numbers of 1.72, 1.95 and 2.46 for Reynolds numbers of 3.12×10^6 and 5.41×10^6 per foot. This instrument is capable of measuring Mach number within approximately ± 1.0 percent and the flow angles within $\pm 0.25^\circ$. The total pressure can be measured within ± 0.5 percent at a Mach number of 1.72 and within ± 2.0 percent at a Mach number of 2.46.

NOTE: Subscript 1 refers to upstream side of shock, subscript 2 to downstream side.

TEST CONDITIONS: Mach number, 1.72, 1.95 and 2.46; Reynolds Number, 3.12×10^6 and 5.4×10^6 per ft; yaw angle, $\pm 26^\circ$; pitch angle, $\pm 26^\circ$.

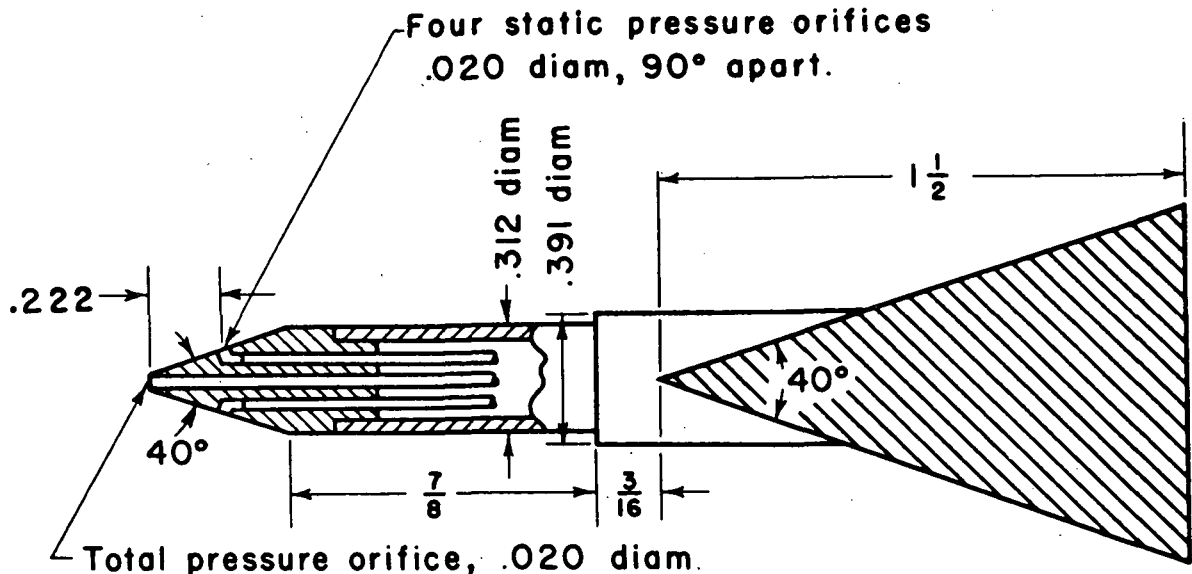
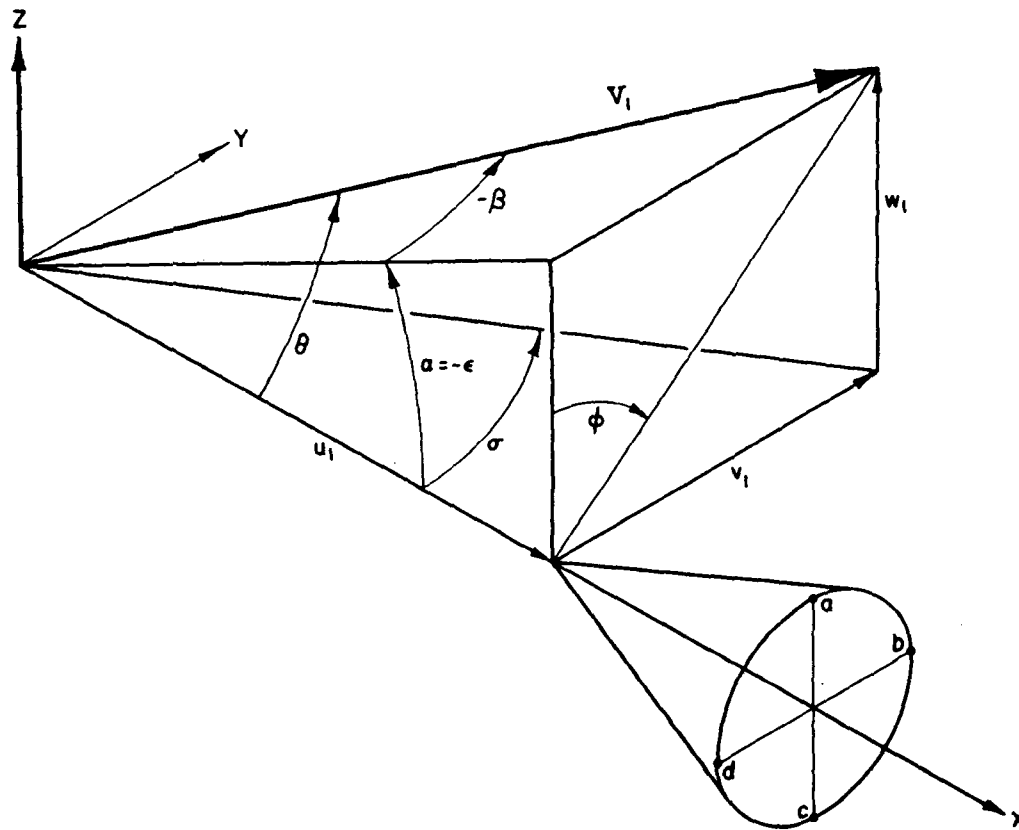
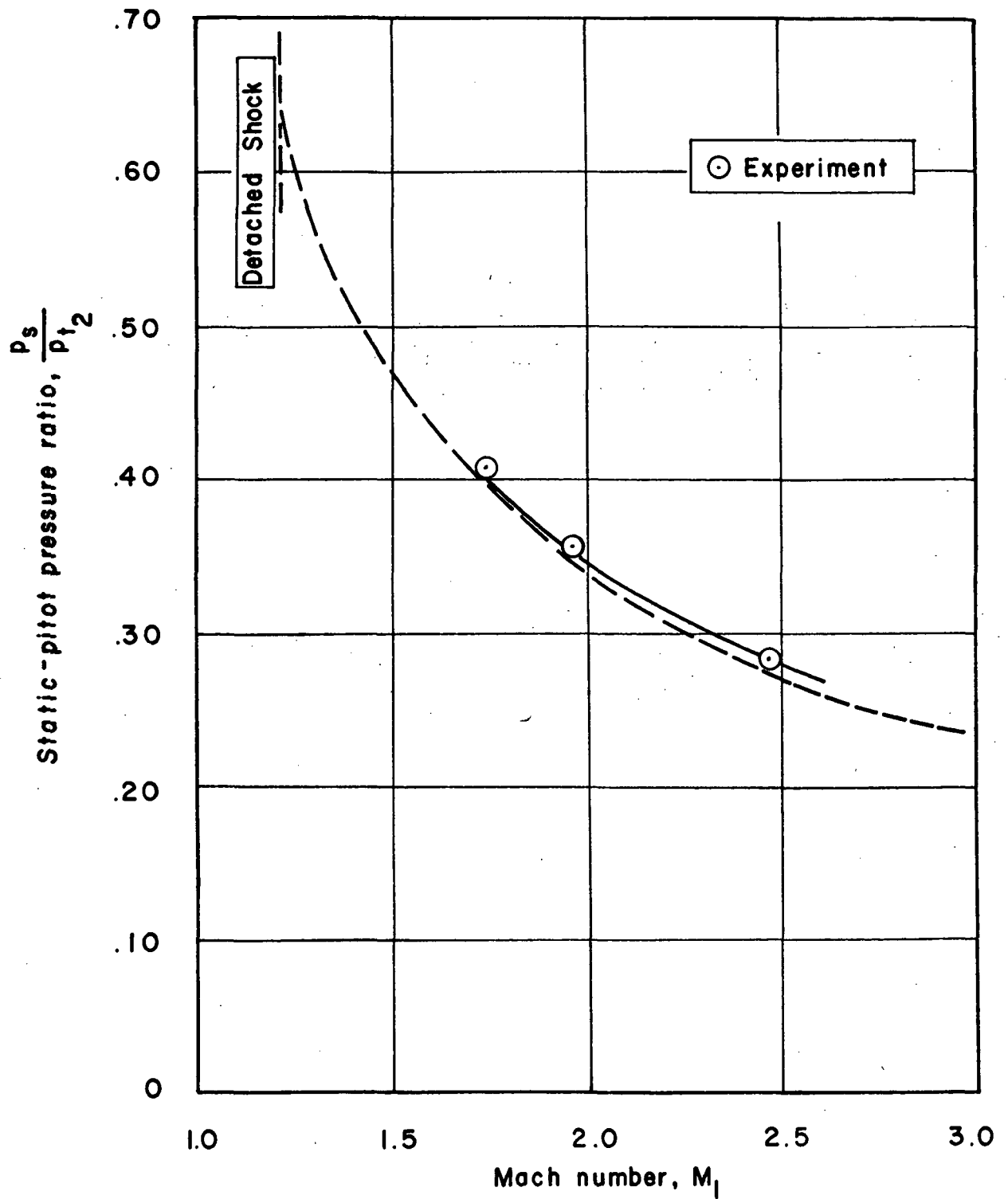


Diagram of the Probe
All Dimensions are in Inches



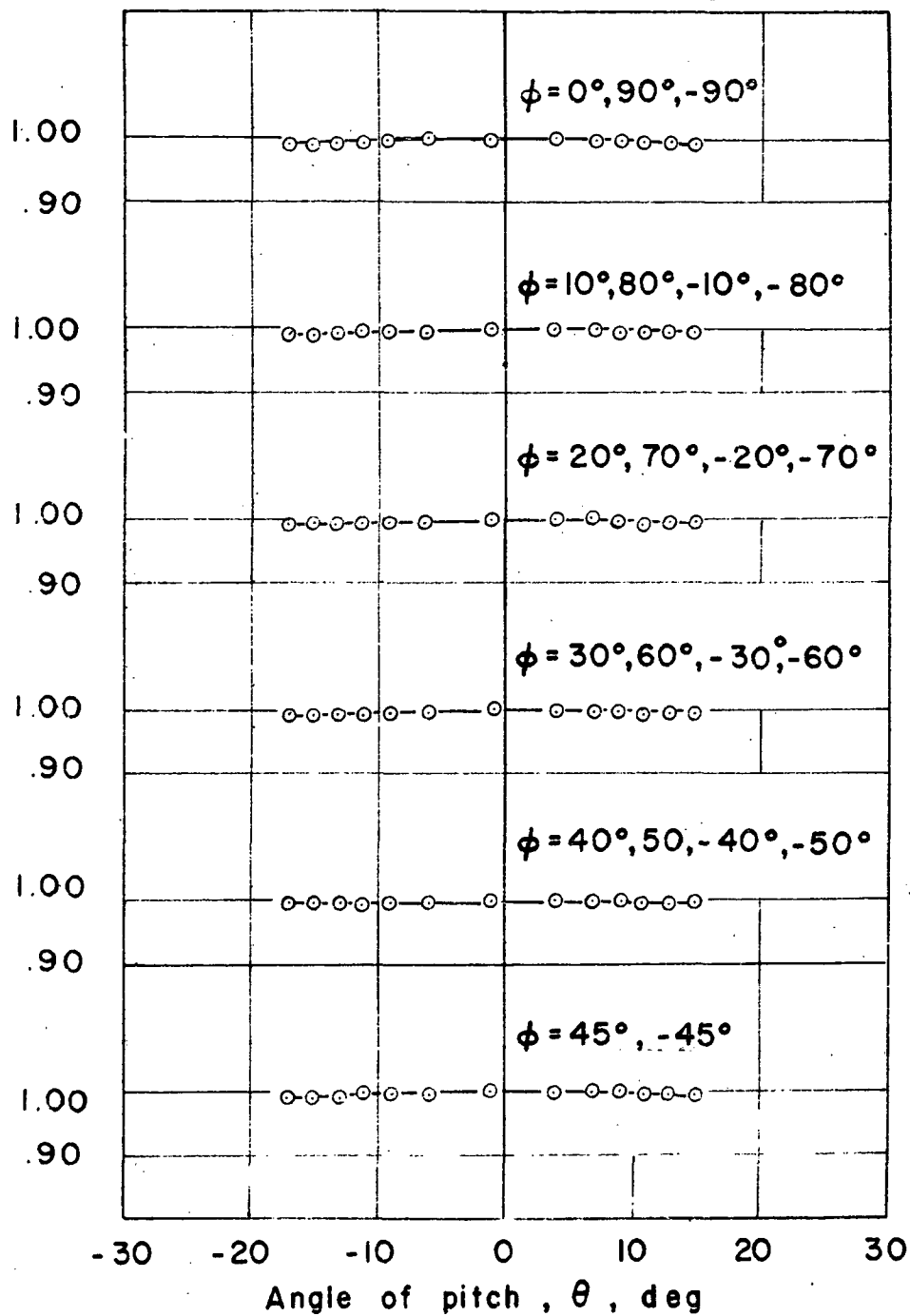
Orifice Designation and Angle Notation



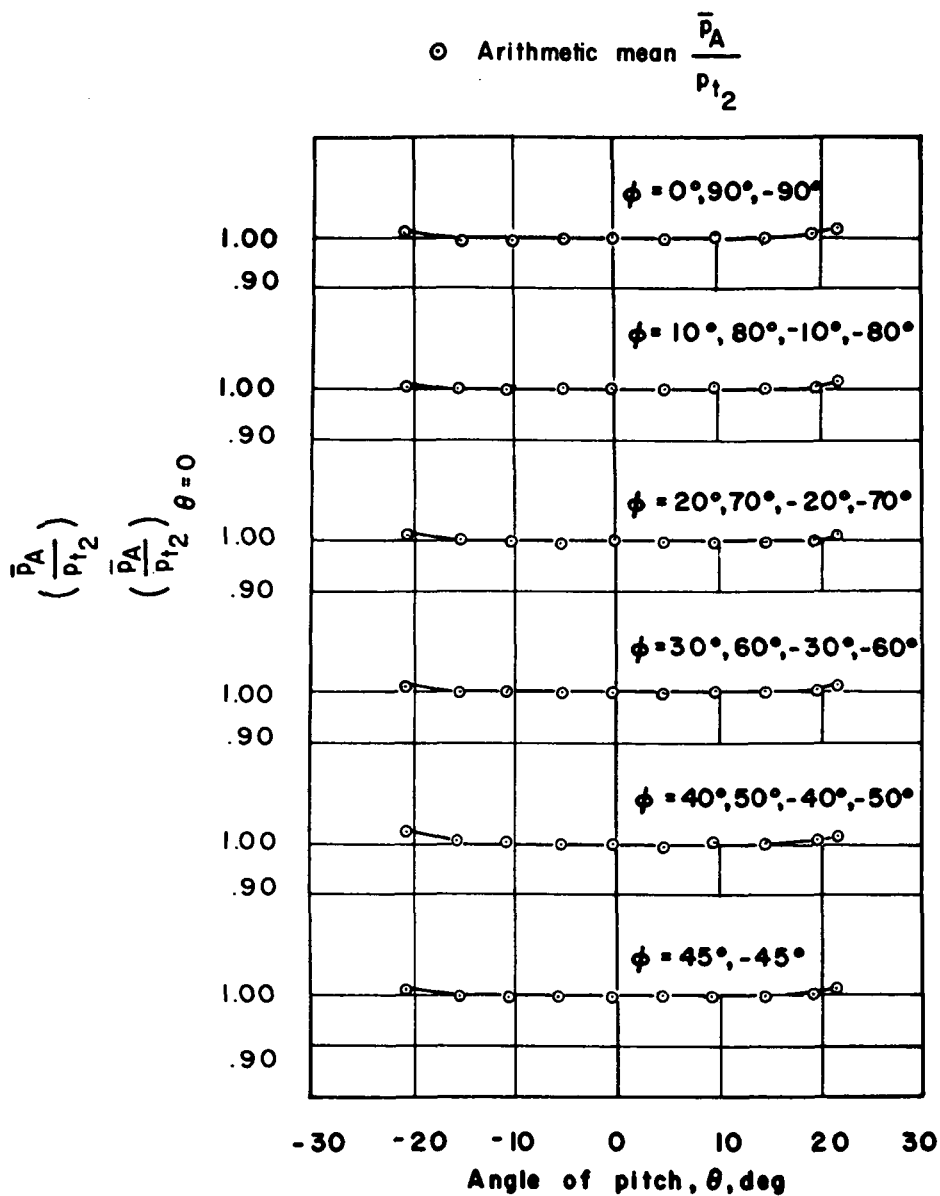
Variation of Static-Pitot Pressure Ratio
With Mach Number at Zero Angle of Pitch

○ Arithmetic mean, $\frac{\bar{p}_A}{p_{t2}}$

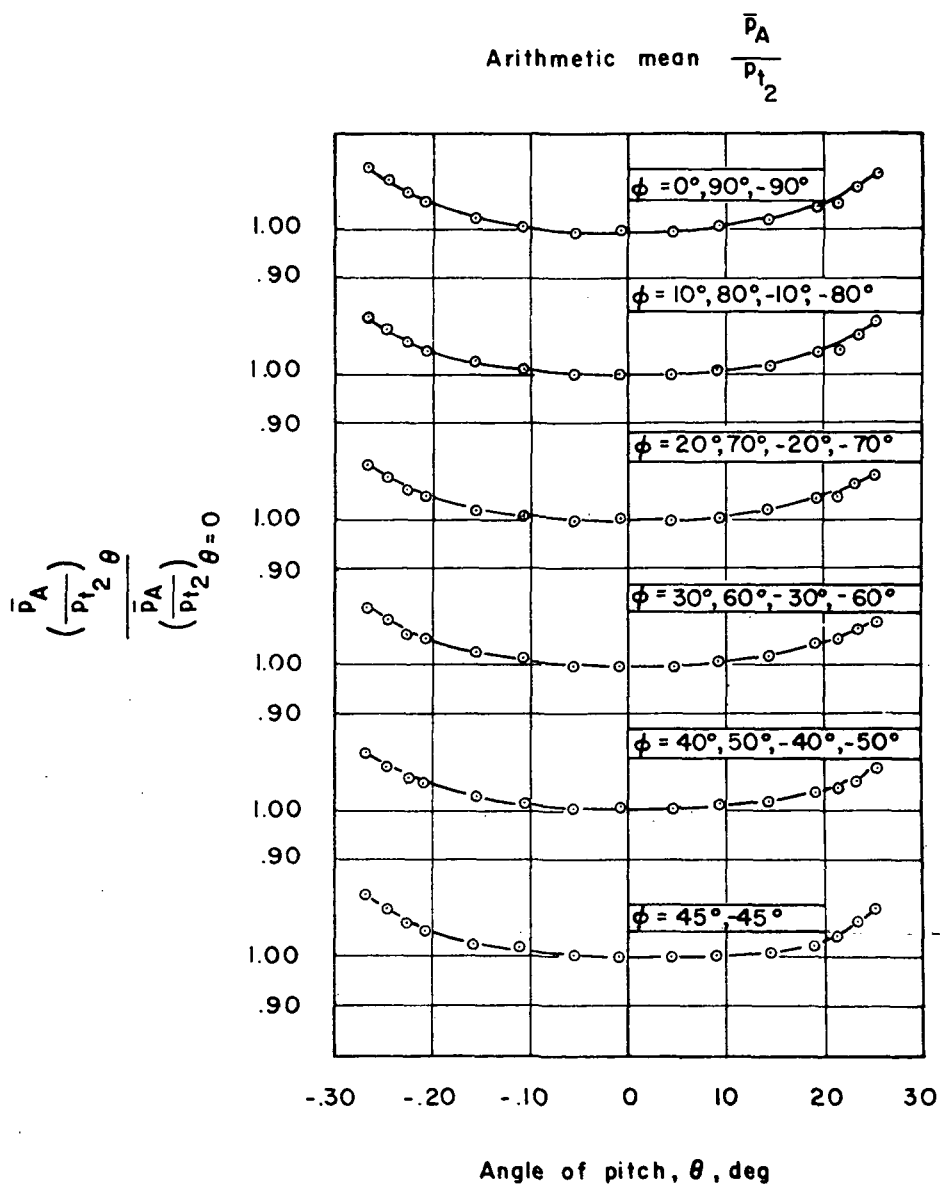
$$\frac{\bar{p}_A}{(p_{t2})} \theta = 0$$



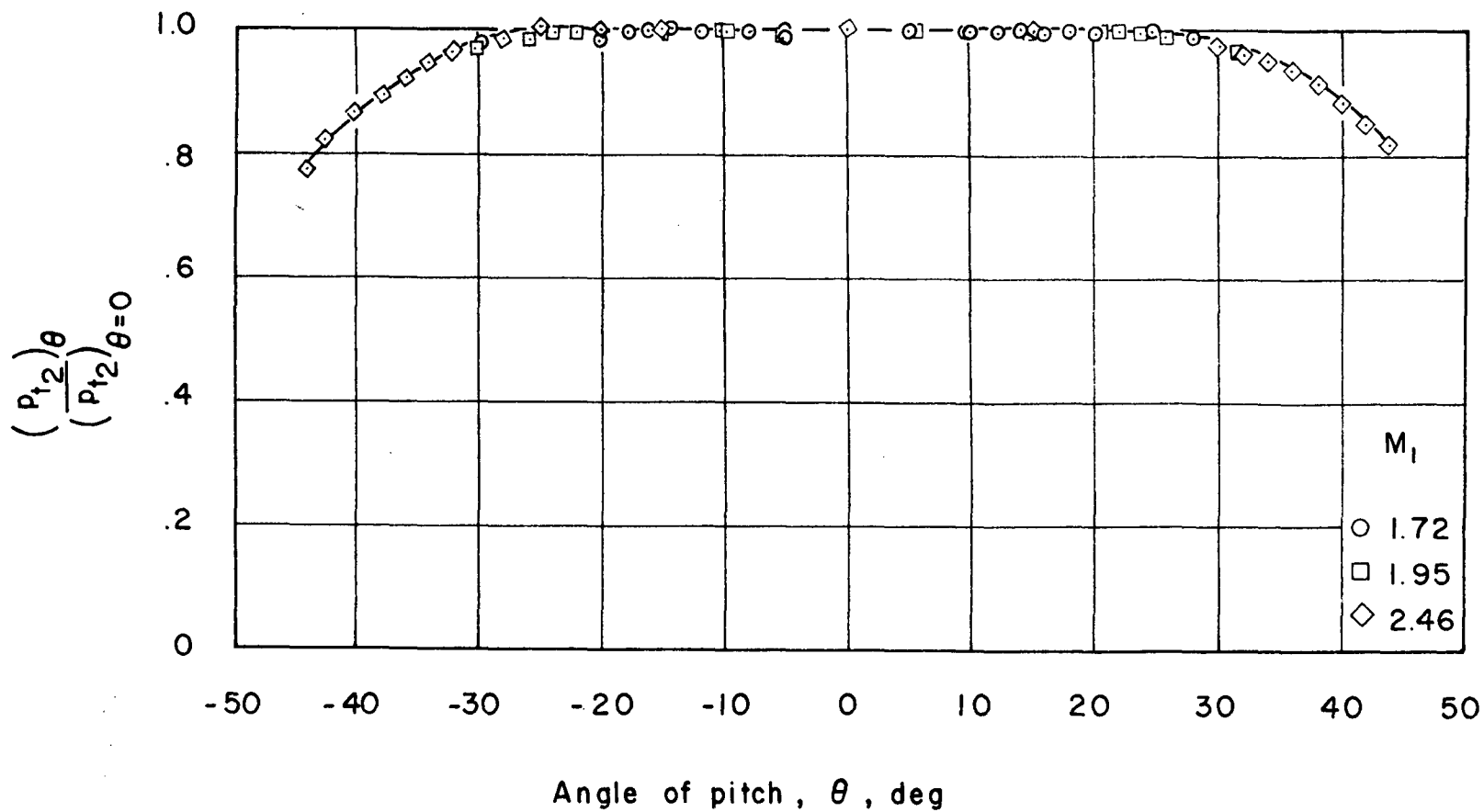
Effect of Angle of Pitch on Ratio of Average Static Pressure to Pitot Pressure, $M = 1.72$



Effect of Angle of Pitch on Ratio of Average Static Pressure to Pitot Pressure, $M = 1.95$



Effect of Angle of Pitch on Ratio of Average Static Pressure to Pitot Pressure, $M = 2.46$



Effect of Angle of Pitch on Pitot Pressure

ϵ Negative σ Negative	ϵ Negative σ Positive
ϵ Positive σ Negative	ϵ Positive σ Positive

$(\frac{\Delta p}{q_1})_\sigma$

Sign convention

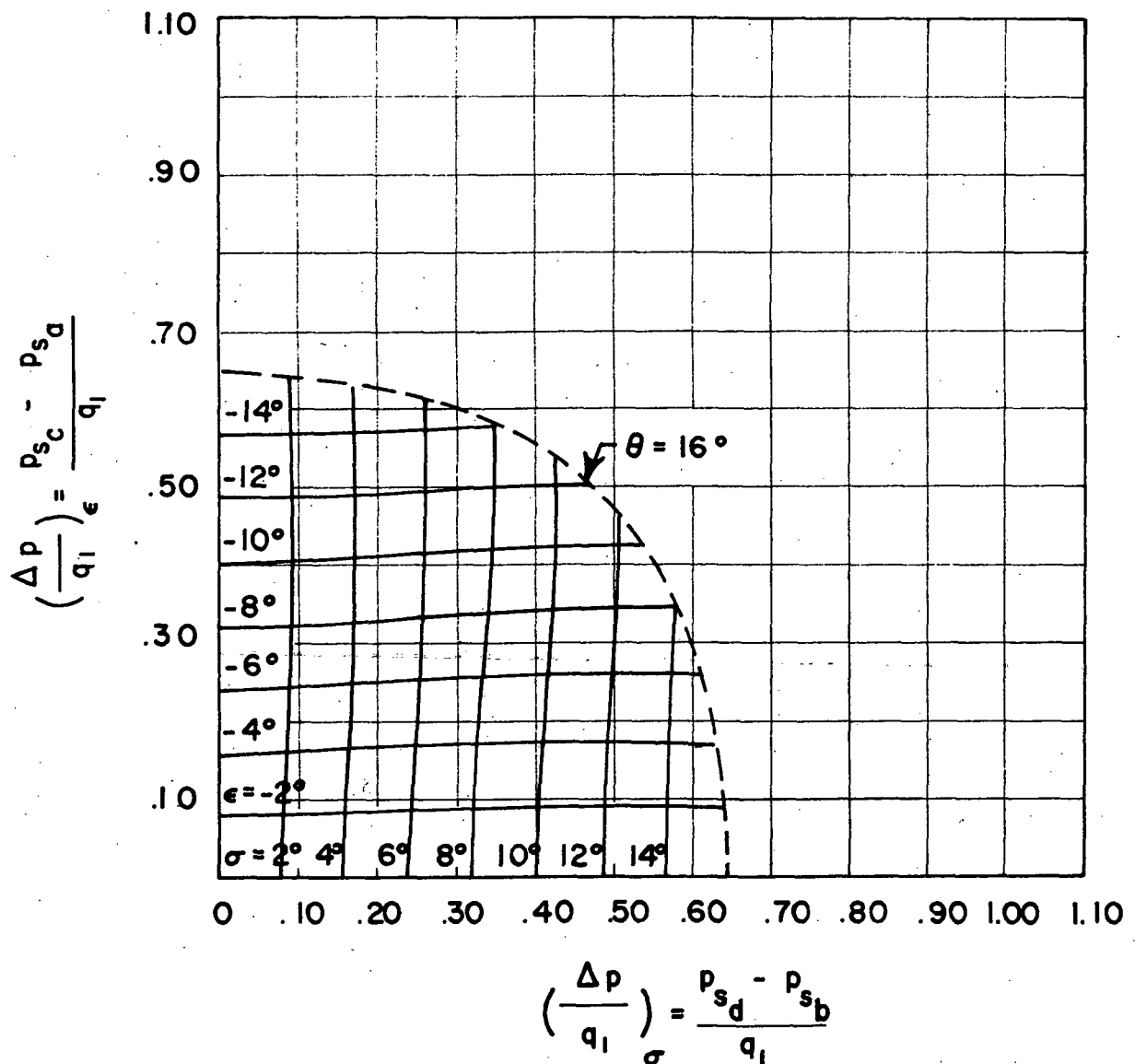


Chart for Determination of Downwash and Sidewash Angles, $M = 1.72$

ϵ Negative
 σ Negative

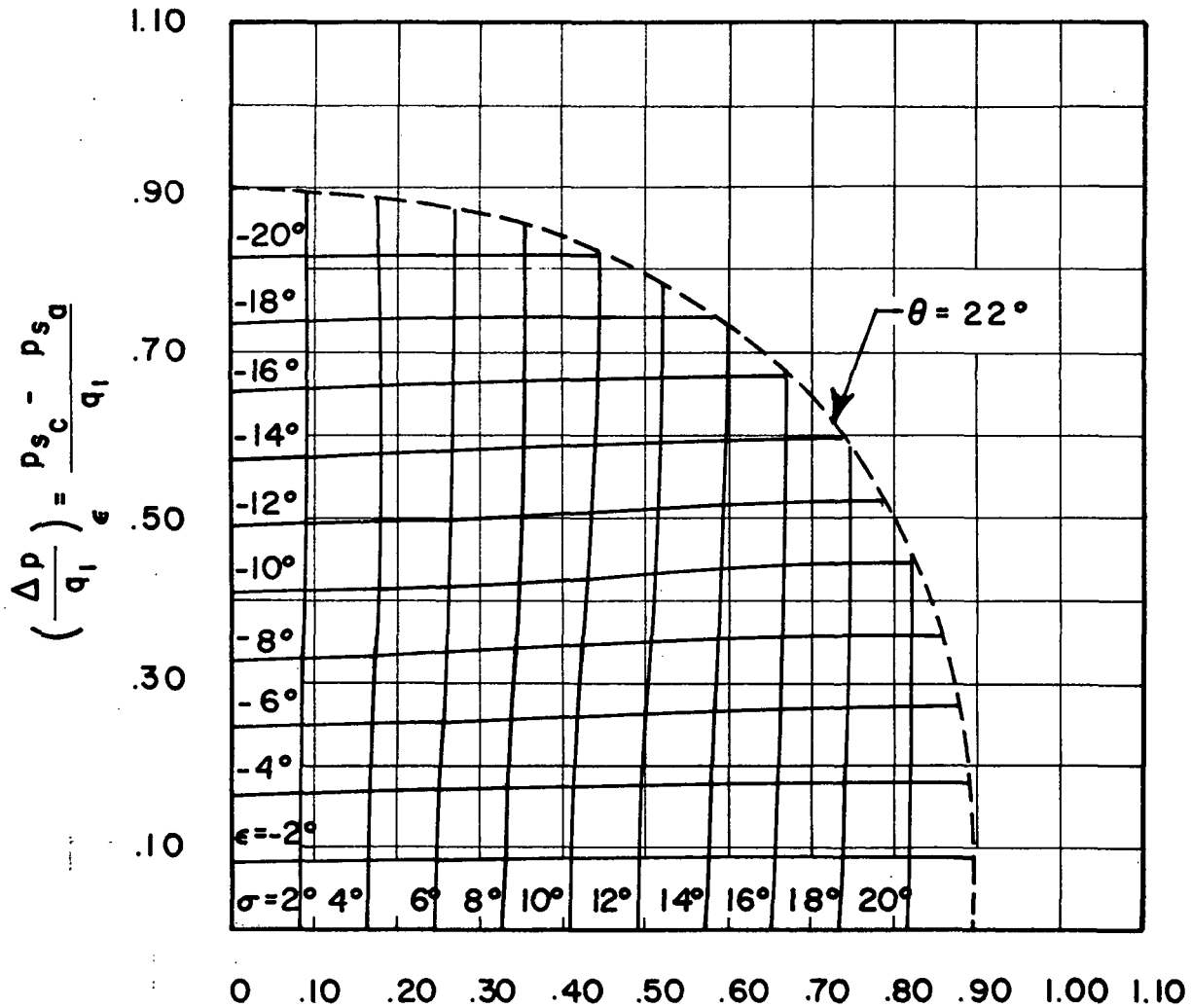
ϵ Negative
 σ Positive

ϵ Positive
 σ Negative

ϵ Positive
 σ Positive

$$\left(\frac{\Delta p}{q_1}\right)_\sigma$$

Sign convention



$$\left(\frac{\Delta p}{q_1}\right)_\sigma = \frac{p_{sd} - p_{sb}}{q_1}$$

Chart for Determination of Downwash and Sidewash Angles, $M = 1.95$

$$\left(\frac{\Delta p}{q_1}\right)_\epsilon$$

ϵ Negative
 σ Negative

ϵ Positive
 σ Negative

ϵ Negative
 σ Positive

ϵ Positive
 σ Positive

$$\left(\frac{\Delta p}{q_1}\right)_\sigma$$

Sign Convention

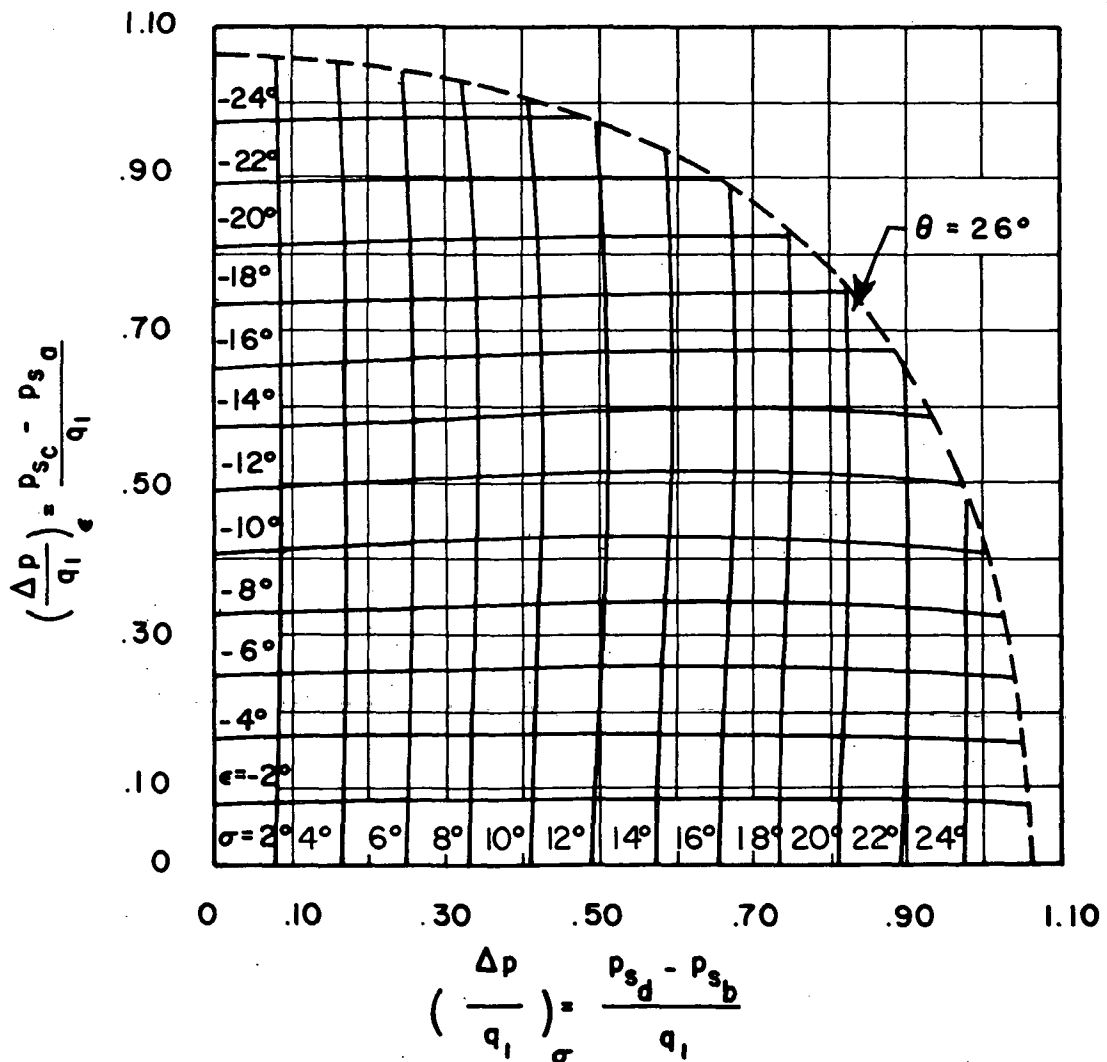


Chart for Determination of Downwash
and Sidewash Angles, $M = 2.46$

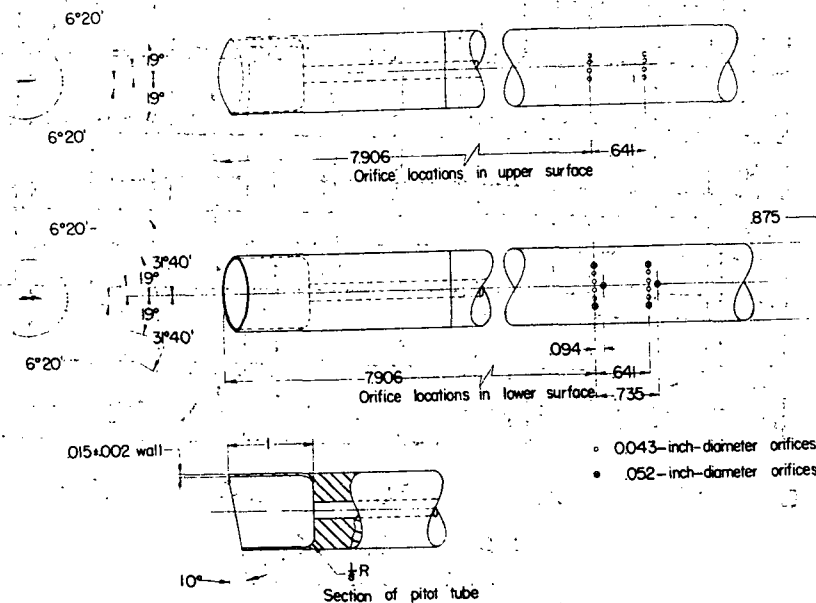
TITLE: Wind-Tunnel Calibration of a Combined Pitot-Static Tube and Vane-Type Flow-Angularity Indicator at Mach Numbers of 1.61 and 2.01

AUTHOR: Archibald R. Sinclair and William D. Mace

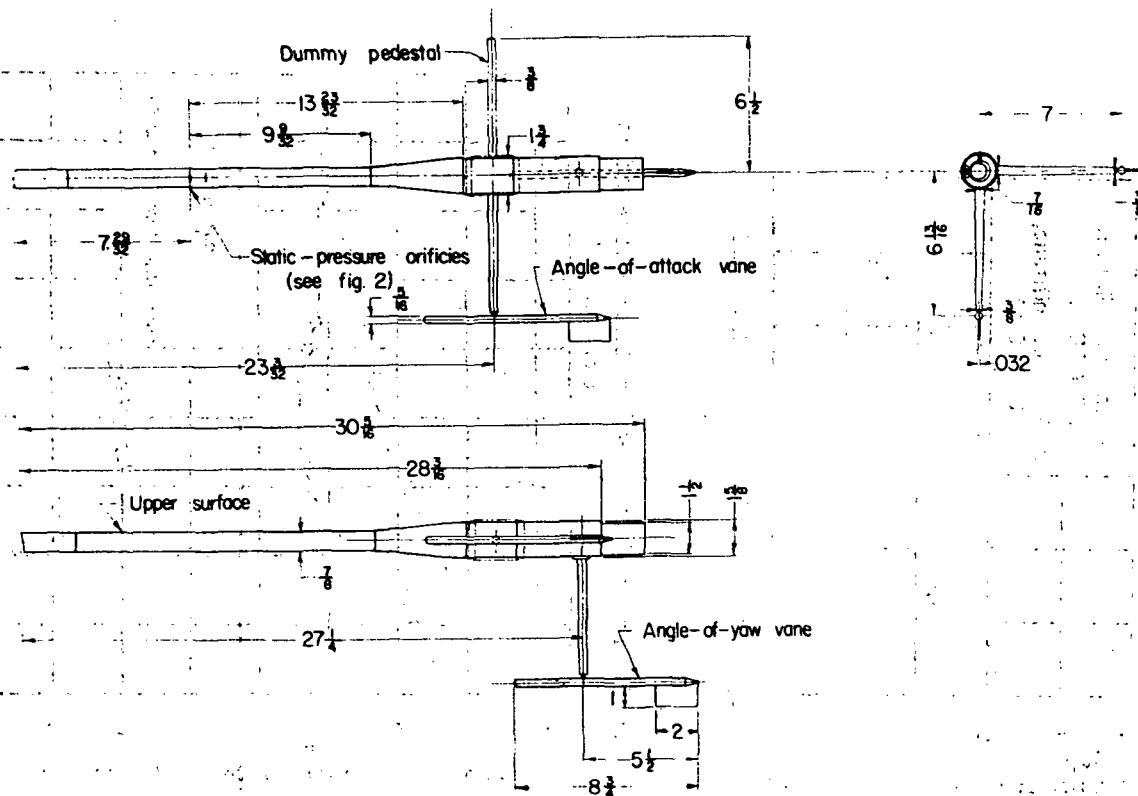
PUBLICATION: NACA TN 3808, October 1956

SUMMARY: A limited calibration of a combined pitot-static tube and vane-type flow-angularity indicator has been made in the Langley 4- by 4-foot supersonic pressure tunnel at Mach numbers of 1.61 and 2.01. The results indicate that the instrument registers too high an angle of attack and gives an error of 0.7° at an angle of attack of 20° for a Mach number of 1.61 and an error of 1.6° at an angle of attack of 24° for a Mach number of 2.01. At zero angle of attack the flow field about the yaw vane was unsymmetrical and caused an error of 1.4° in yaw indication at zero angle of yaw for a Mach number of 2.01. The installation of a dummy vane pedestal to provide a more symmetrical flow field reduced this error to 0.25° . The probe gave static-pressure readings which were too low at angles of yaw.

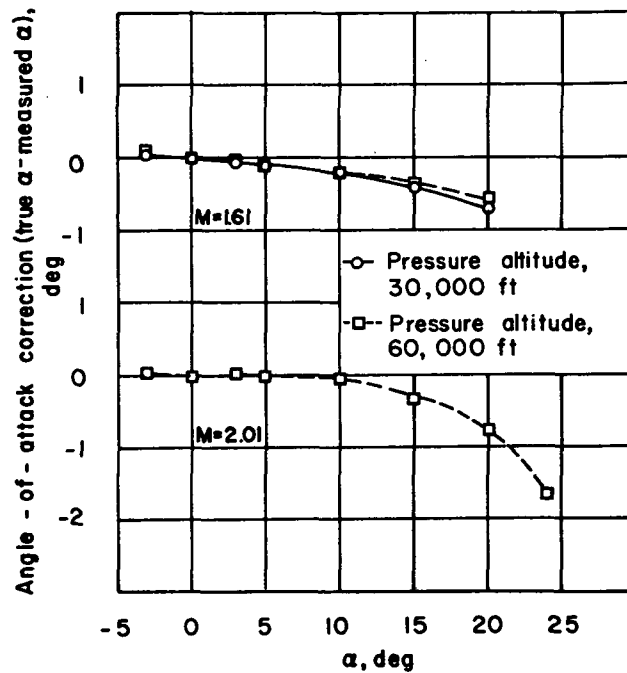
TEST CONDITIONS: Mach number, 1.61 and 2.01; Reynolds Number, 1.2×10^6 , 4.6×10^6 and 5.7×10^6 per ft; yaw angle, (-3°) - 20° ; pitch angle, (-3°) - 24° .



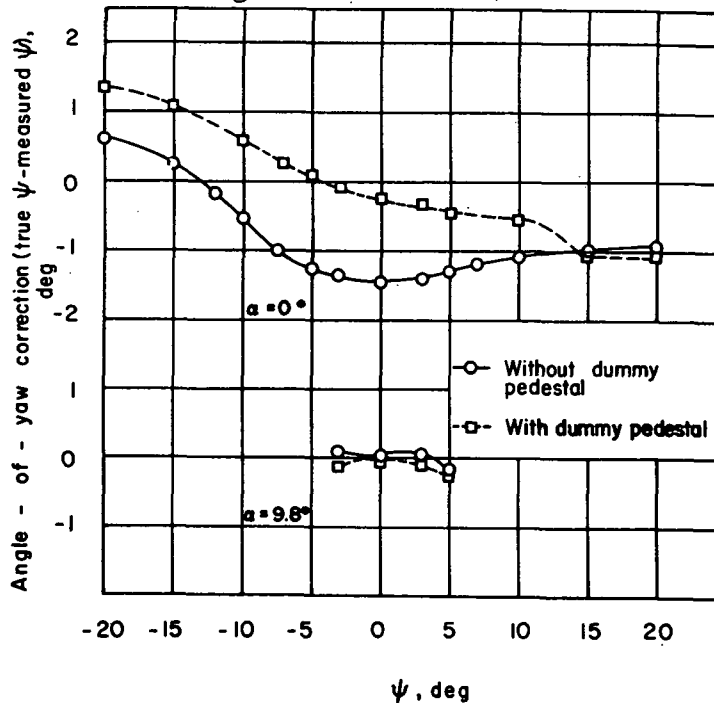
Details of Pitot-Static Tube.
All Linear Dimensions are in Inches



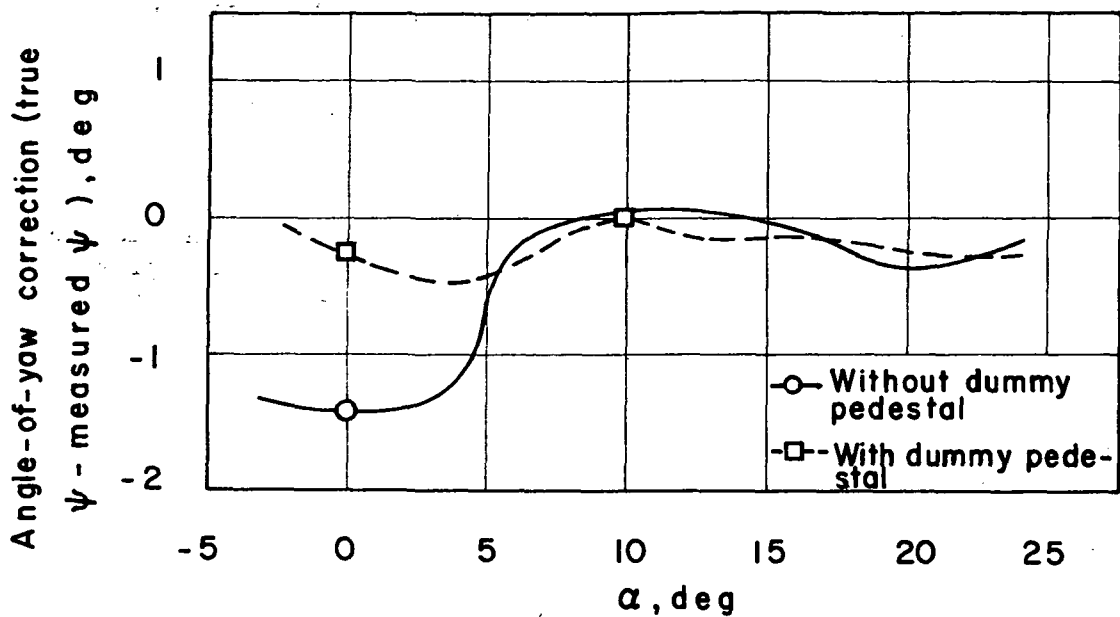
Details of Combined Pitot-Static Tube and Vane-Type
Flow-Angularity Indicator. All Dimensions are in Inches



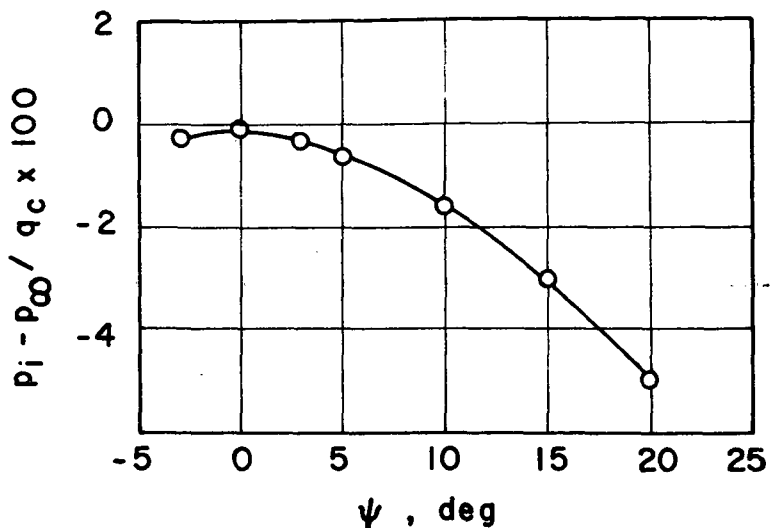
Variation of Angle-of-Attack Correction with Angle of Attack. $\psi = 0^\circ$.



Variation of Angle-of-Yaw Correction with Angle of Yaw. $M = 2.01$; Pressure Altitude, 60,000 Feet.



Variation of Angle-of-Yaw Correction with Angle of Attack.
 $M = 2.01$; Pressure Altitude, 60,000 Feet; $\psi = 0^\circ$.



Variation of Measured Probe Static Pressure with Angle of Yaw. $M = 2.01$; Pressure Altitude, 60,000 Feet; $\alpha = 0^\circ$. Pressures Obtained from Forward Orifices.

TITLE: Several Combination Probes for Surveying Static and Total Pressure and Flow Direction

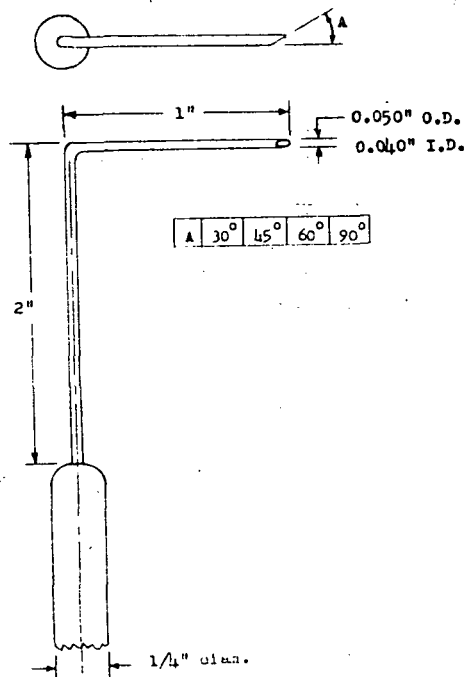
AUTHOR: Wallace M. Schulze, George C. Ashby, Jr., and John R. Erwin

PUBLICATION: NACA TN 2830, 1952

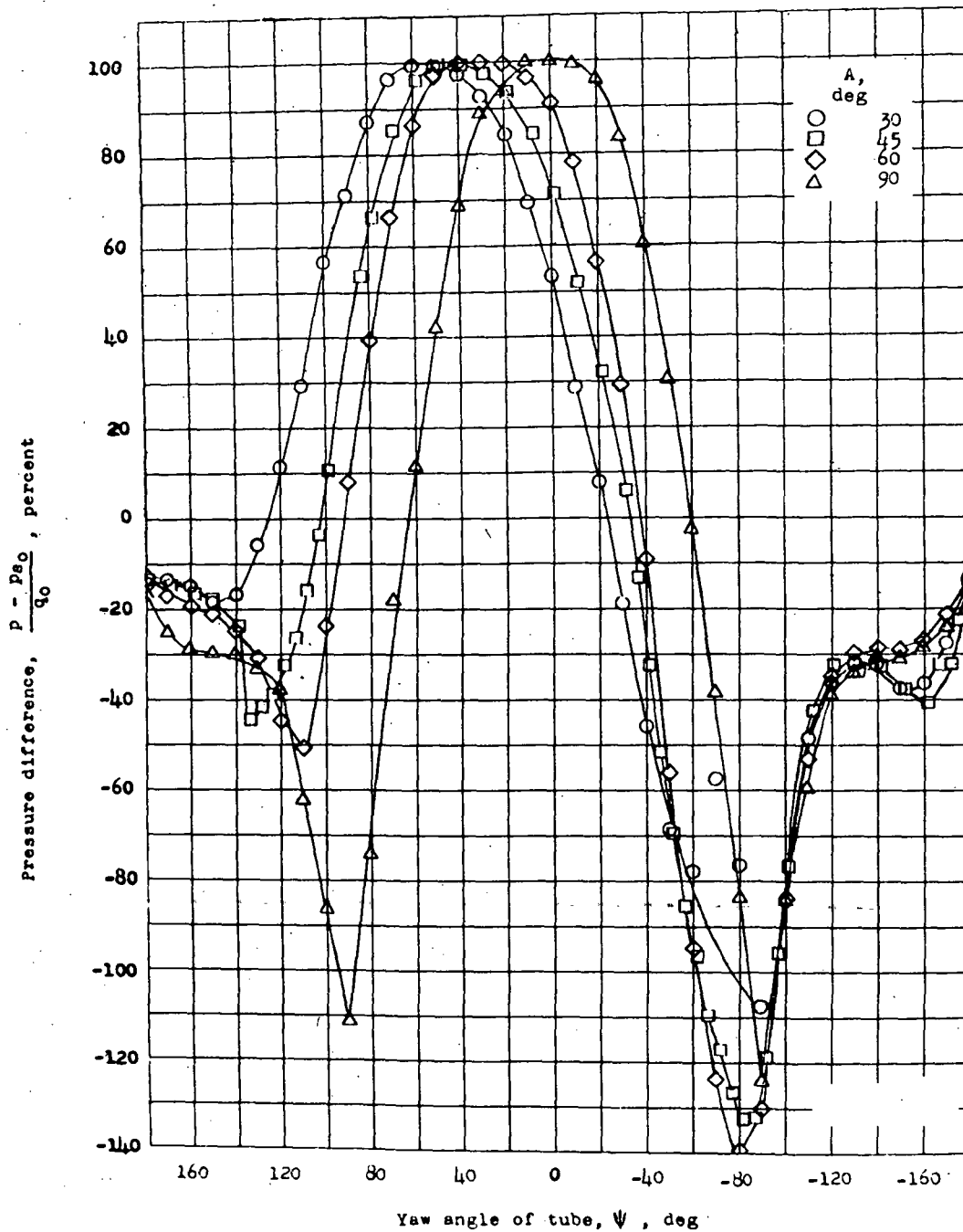
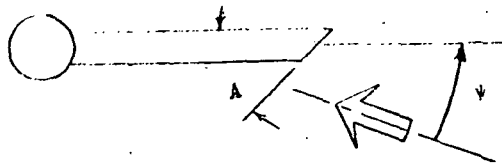
SUMMARY: An investigation has been conducted to provide a basis for the design of combination probes intended to survey the static and total pressure and direction of flow with special reference to subsonic turbo-machine testing. Static-pressure probes, yaw-element probes, claw-type yaw probes, and combination probes were tested in an 8-inch-diameter calibration tunnel at air velocities up to 295 feet per second.

From the results of this investigation, the factors which determine the sensitivity of claw-type yaw probes were determined. Satisfactory combination survey probes for sensing static and total pressure and direction of flow in one or two planes were devised.

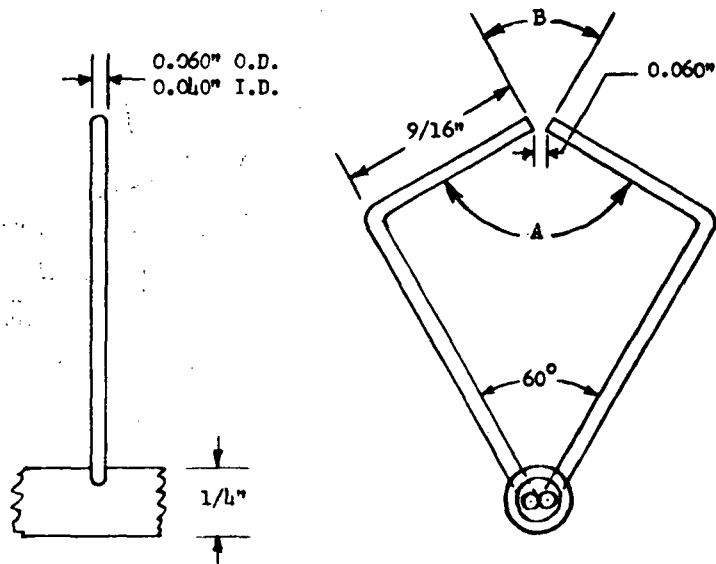
TEST CONDITIONS: Velocity, 0-295 ft/sec; Reynolds Number, not available; yaw angle, 0° - 90° ; pitch angle, assumed 0° .



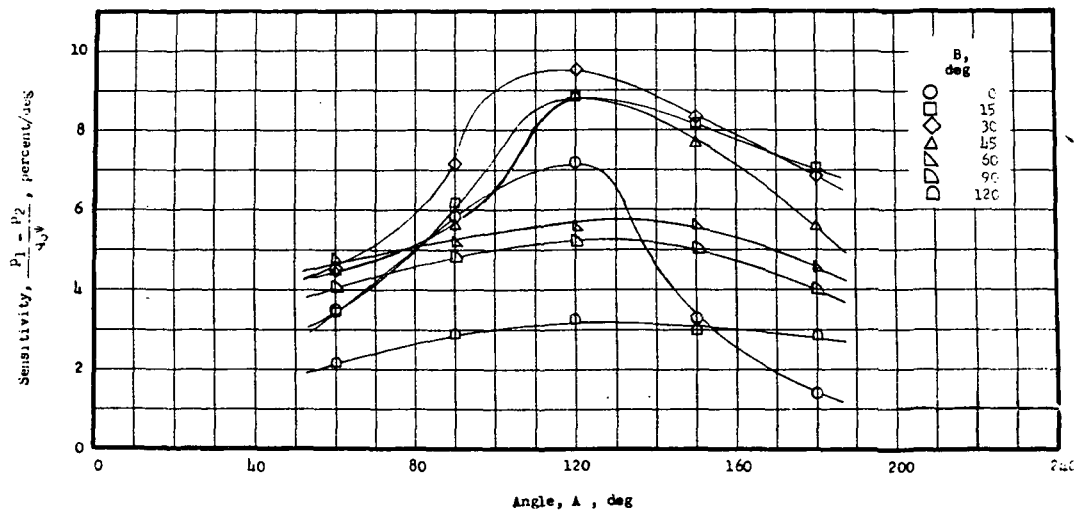
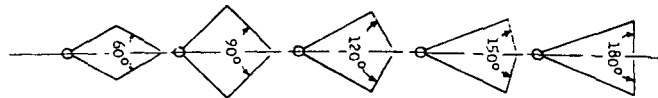
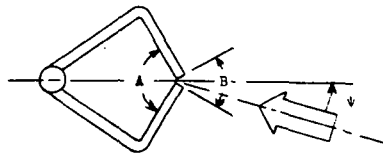
Details of Yaw-Element Probes



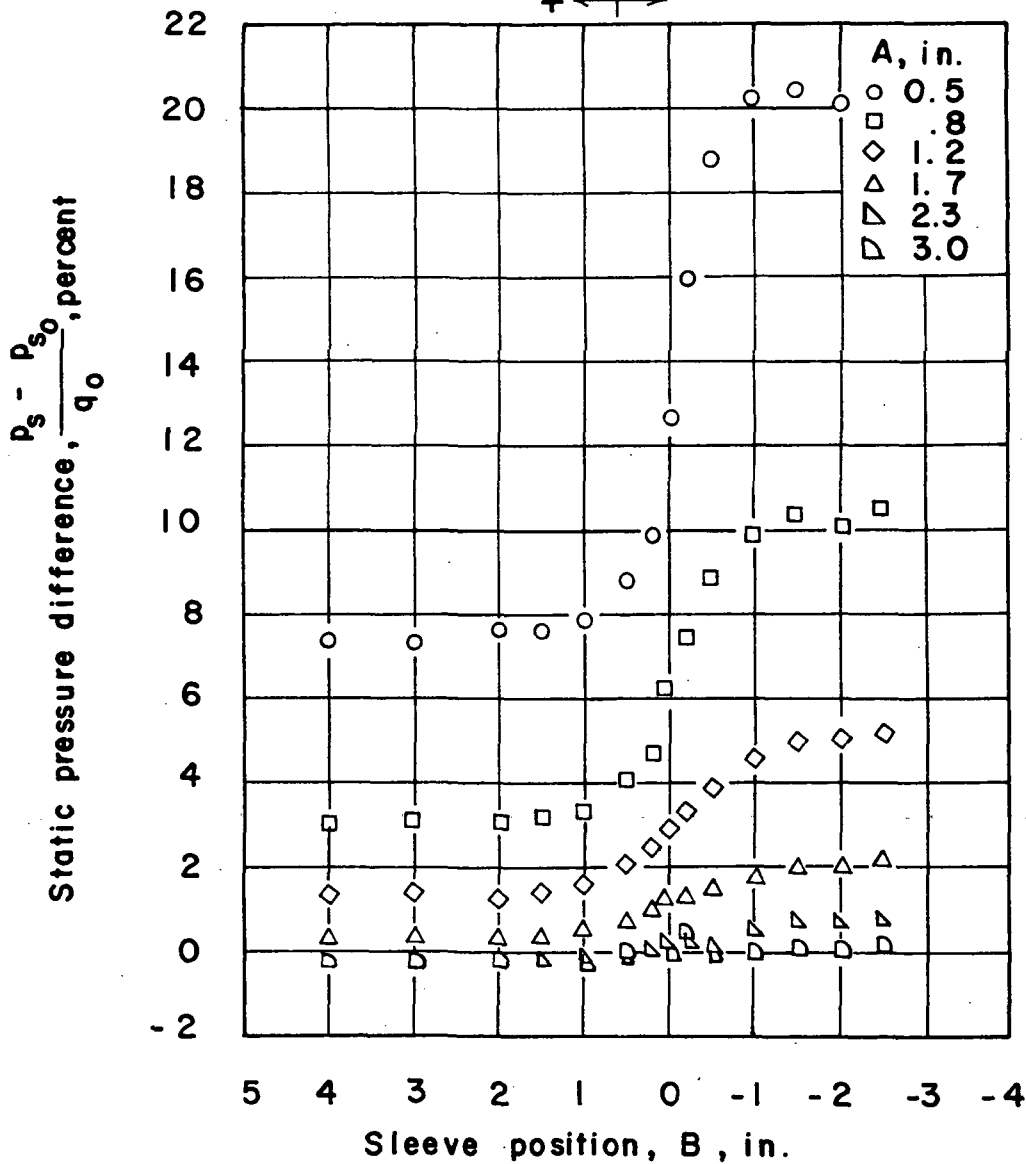
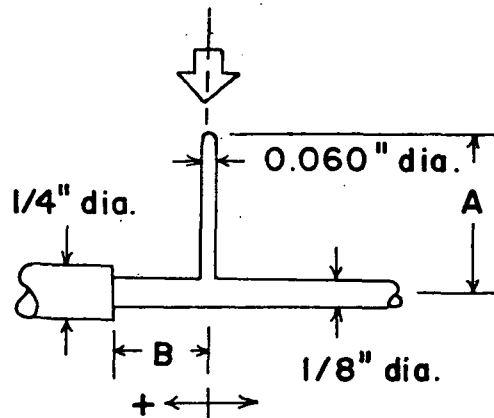
Variation of Difference in the Tube Pressure and Static Pressure for Four Yaw-Element Probes. Flow Velocity, 295 Feet per Second



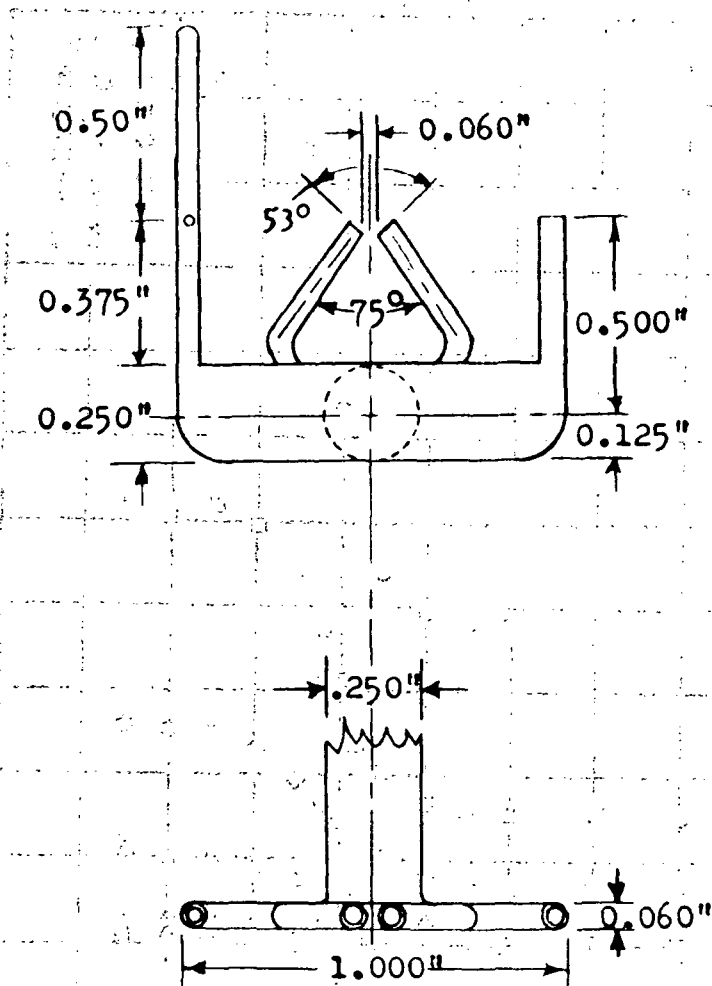
Details of Claw-Type Yaw Probes



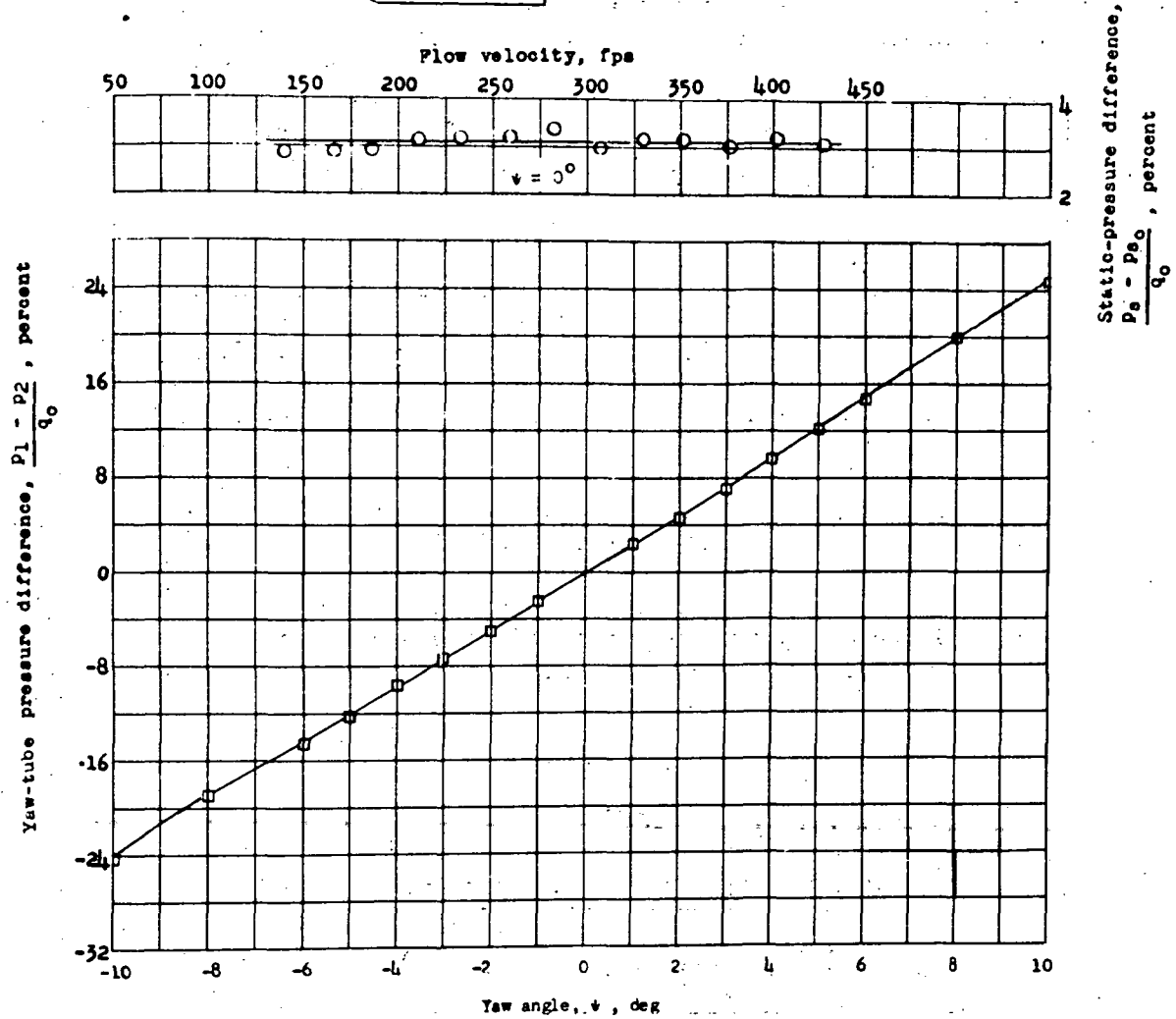
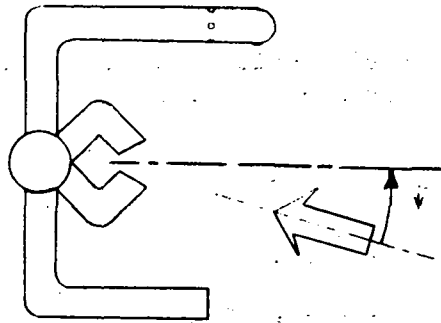
Yaw Sensitivity of Claw-Type Probes. Flow Velocity, 295 Feet Per Second.



Difference in Static-Pressure Reading With Sleeve Position for Six Test Probes. Flow Velocity, 295 Feet per Second



Details of Combination Probe Type A



Variation of Yaw-Tube Pressure Difference With Yaw Angle for Flow Velocity of 285 Feet Per Second and Difference in Static Pressure Reading Between Probe Type A and Standard Static Probe with Flow Velocity.

TITLE: Calibration of a Combined Pitot-Static Tube and Vane-Type Flow Angularity Indicator at Transonic Speeds and at Large Angles of Attack or Yaw

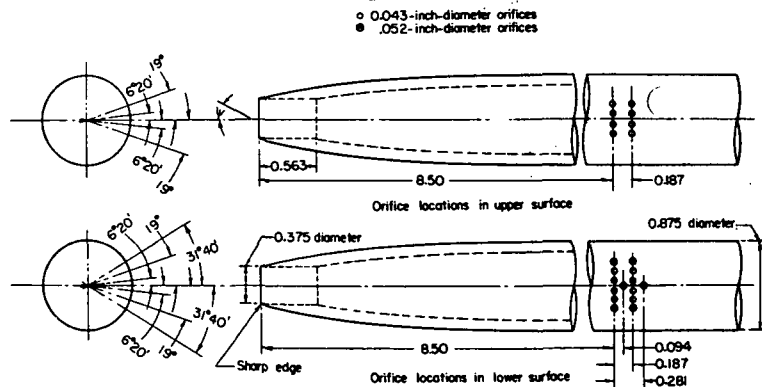
AUTHOR: Albin O. Pearson and Harold A. Brown

PUBLICATION: NACA TM L52F24, September 1952

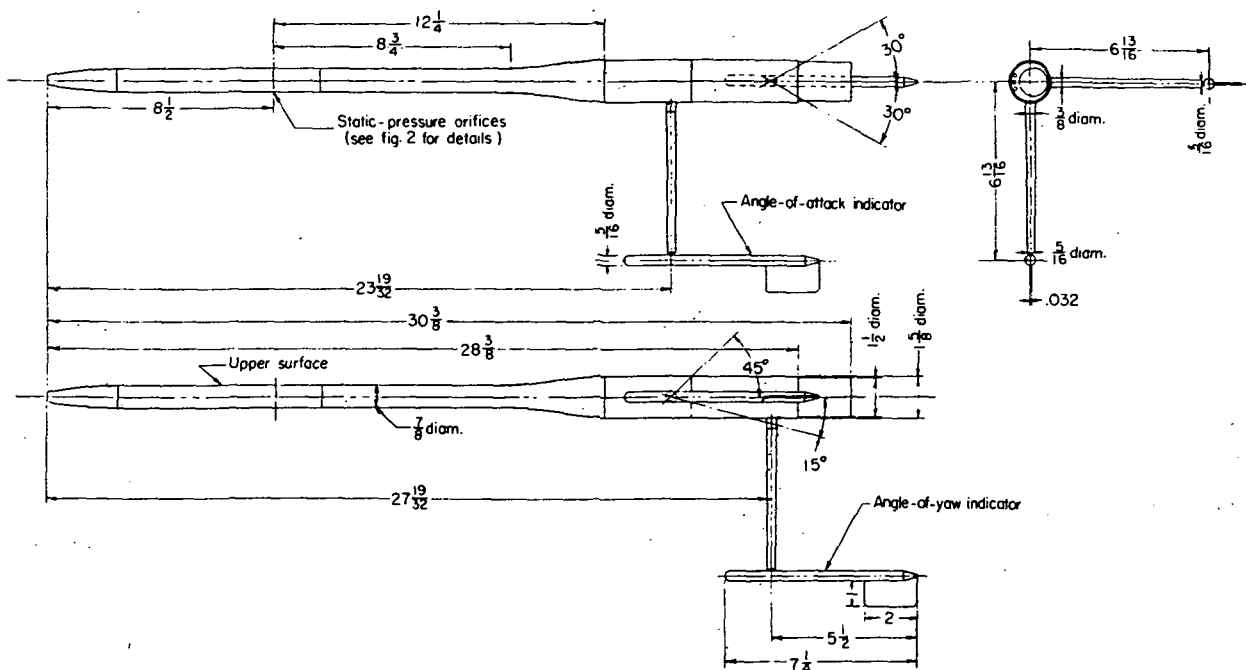
SUMMARY: The results indicate that the angles of attack or yaw at which the total-pressure error is 0.01 (after correction of indicated total pressures for losses due to bow wave ahead of tube at Mach numbers greater than 1.00) vary linearly with Mach number and increase from approximately 21.0° at a Mach number of 0.60 to about 24.3° at a Mach number of 1.10. The static-pressure error is affected more by changes in angle of yaw than by corresponding changes in angle of attack and is positive (measured pressure greater than stream pressure) for angles of attack and negative for angles of yaw. The calibration factor for the pitot-static tube varies linearly with Mach number up to a Mach number of approximately 0.99 followed by an abrupt decrease in value at a Mach number near 1.00. The maximum error in measuring flow angularity by means of the vanes is of the order of 1.5° and occurs at a Mach number of 1.10 and an angle of attack of 25° .

TEST CONDITIONS: Mach Number, 0.6-1.11; Reynolds Number, not available; yaw angle, $\pm 20^\circ$; pitch angle, (-10°) - 25°

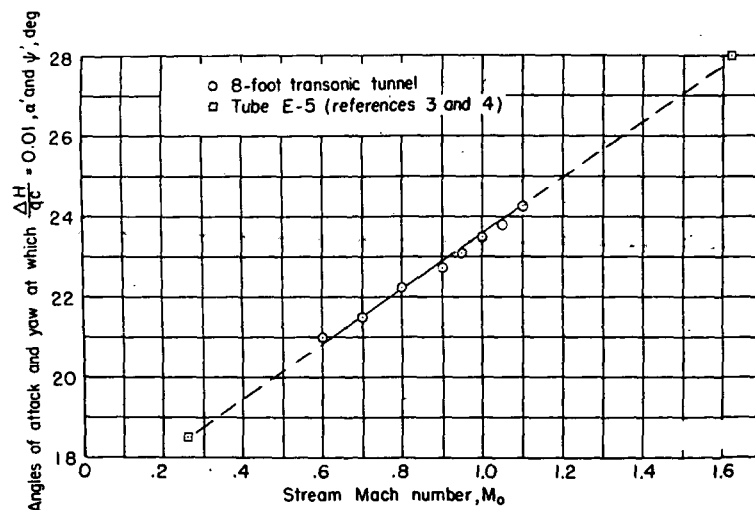
DEFINITIONS: ΔH = Difference between probe and free stream total pressure.
 ΔP = Difference between probe and free stream static pressure.
 q_c = Free stream impact pressure



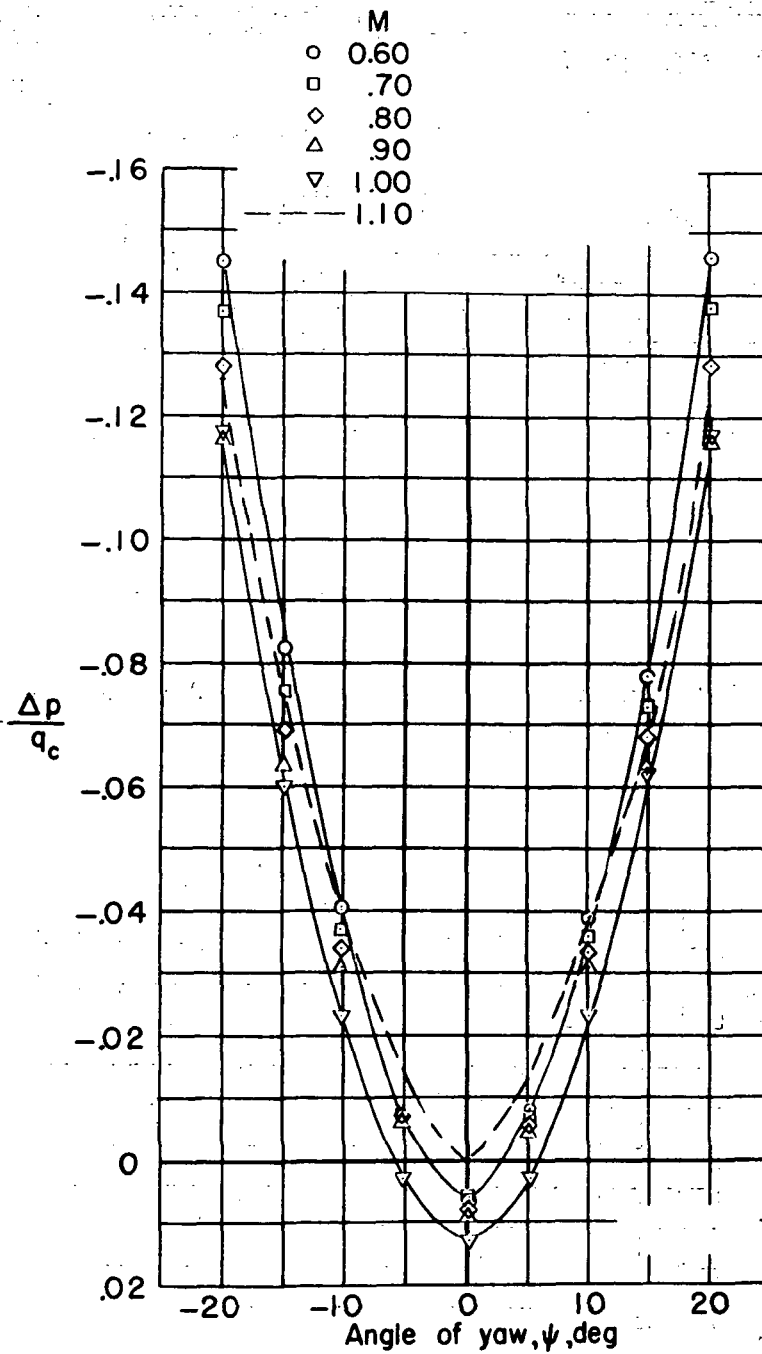
Details of Static- and Total-Pressure Orifices.
 All Linear Dimensions are in Inches.



Details of Combination Pitot-Static Tube and Vane-Type Flow-Angularity Indicator. All Linear Dimensions are in Inches.



Variation With Mach Number of Angles
of Attack and Yaw at Which $\frac{\Delta H}{qc} = 0.01$.



Variation of Static-Pressure Error
With Angle of Yaw. $\alpha = 0^\circ$

2. Pneumatic Probes Suitable for Boundary Layer Measurements

No. 11

TITLE: Accuracy of Pitot-Pressure Rakes for Turbulent Boundary-Layer Measurements in Supersonic Flow

AUTHOR: Carl R. Keener and Edward J. Hopkins

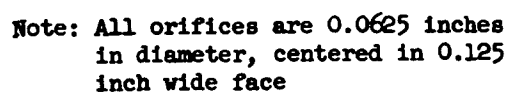
PUBLICATION: NASA TN D-6229, March 1971

SUMMARY: Boundary-layer profiles from three conventional pitot-pressure rakes and a new probeless rake are compared with those from a single traversing probe in a supersonic turbulent boundary layer on the wall of a wind tunnel. Measurements were made at Mach numbers from 2.4 to 3.4 and at momentum-thickness Reynolds numbers from 26,000 to 75,000. The boundary-layer thickness was approximately 6 inches and the rake heights were 5, 8, and 12 inches with different probe size and spacing.

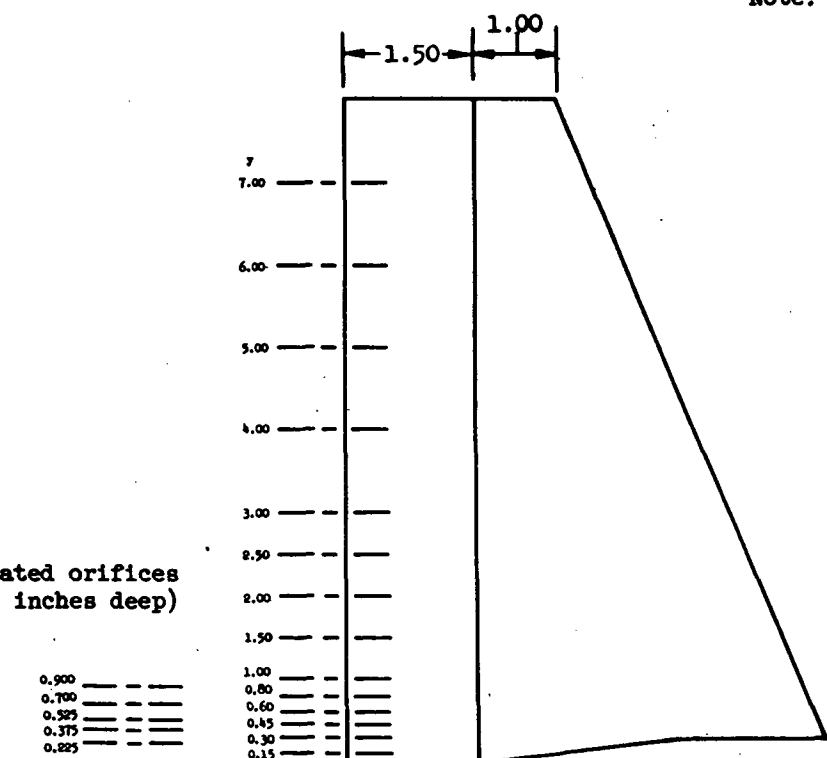
The pitot pressures from both the conventional rakes and the probeless rake agree with the single traversing-probe pressure within 2 percent of the edge pitot pressure. The resulting errors in Mach number and velocity ratios are less than 2 percent; momentum and displacement thickness errors are less than 4 percent. These errors are not excessive and indicate that multiple-probe and probeless rakes can be used in measuring turbulent boundary layer.

The low error of the probeless rake indicates that this type of configuration might have useful application at high temperatures where conventional rake probes might warp or fail.

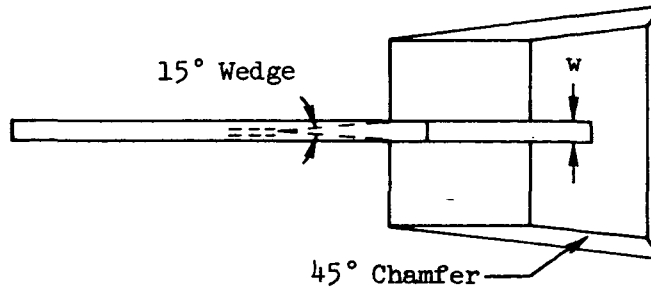
TEST CONDITIONS: Mach Number, 2.4, 2.9 and 3.4; Reynolds Number, 10^6 , 2.5×10^6 and 3.2×10^6 per ft; yaw angle, assumed 0° ; pitch angle, assumed 0° .



**Simulated orifices
(0.25 inches deep)**



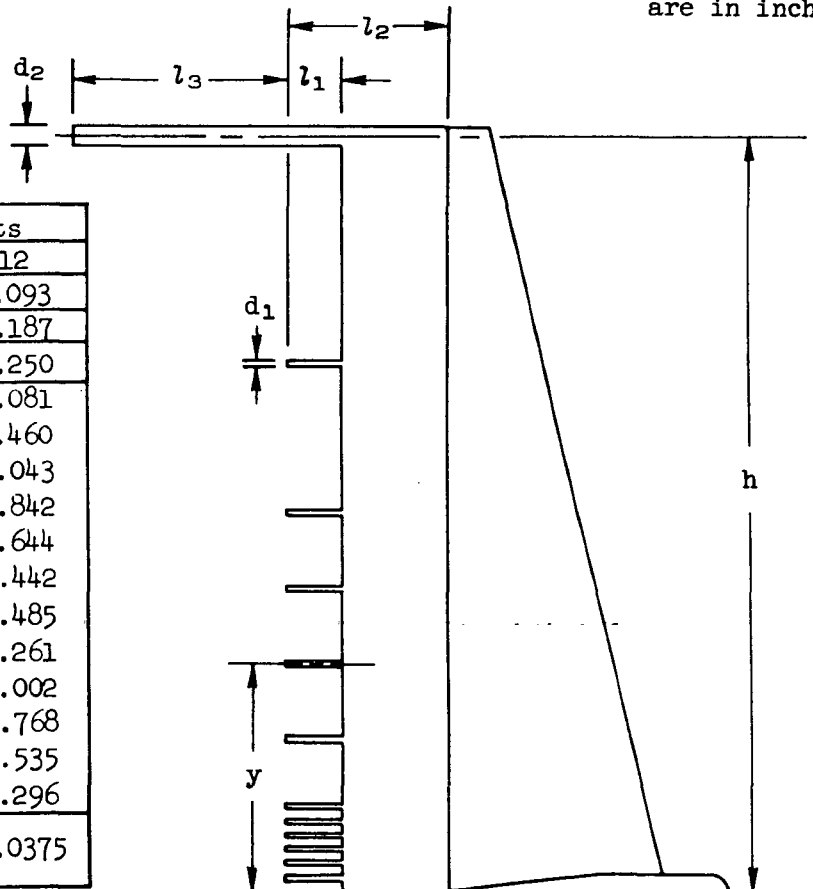
46



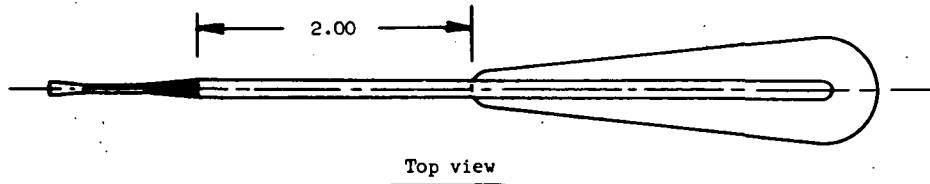
Top probe is pitot-static tube
on 8 and 12-inch rakes

Note: All dimensions
are in inches

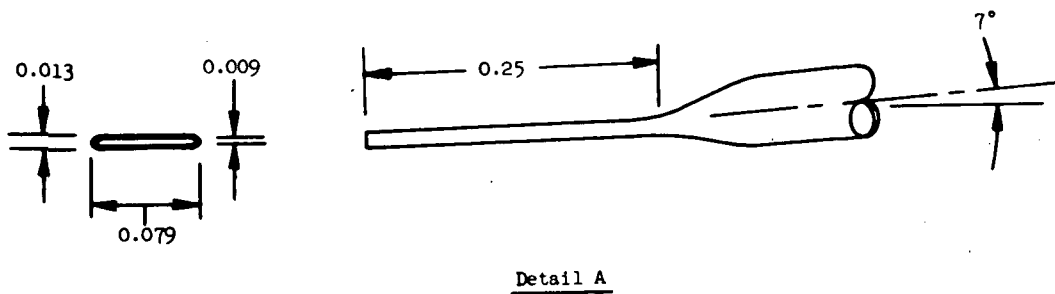
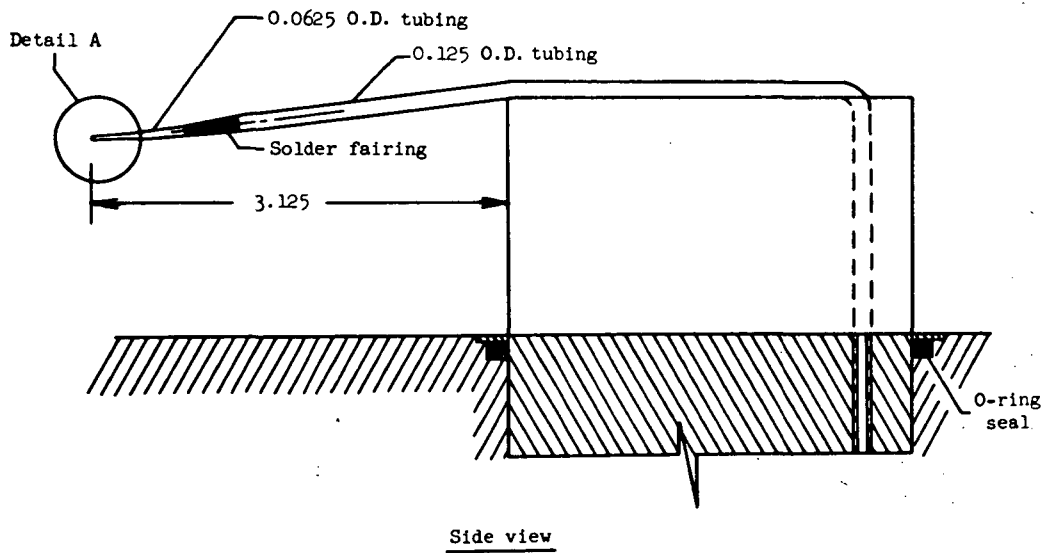
Table of tube heights			
$h =$	5	8	12
$d_1 =$	0.042	0.062	0.093
$d_2 =$	0.042	0.187	0.187
$l_3 =$	0	2.250	2.250
$y =$	4.995	8.019	12.081
	3.496	5.603	8.460
	2.492	4.001	6.043
	1.992	3.200	4.842
	1.490	2.401	3.644
	0.990	1.601	2.442
	0.589	0.955	1.485
	0.490	0.795	1.261
	0.390	0.636	1.002
	0.292	0.473	0.768
	0.193	0.316	0.535
	0.092	0.152	0.296
$\frac{l_1}{d_1} = 10; \frac{l_2}{w} = 6; \frac{w}{h} = 0.0375$			



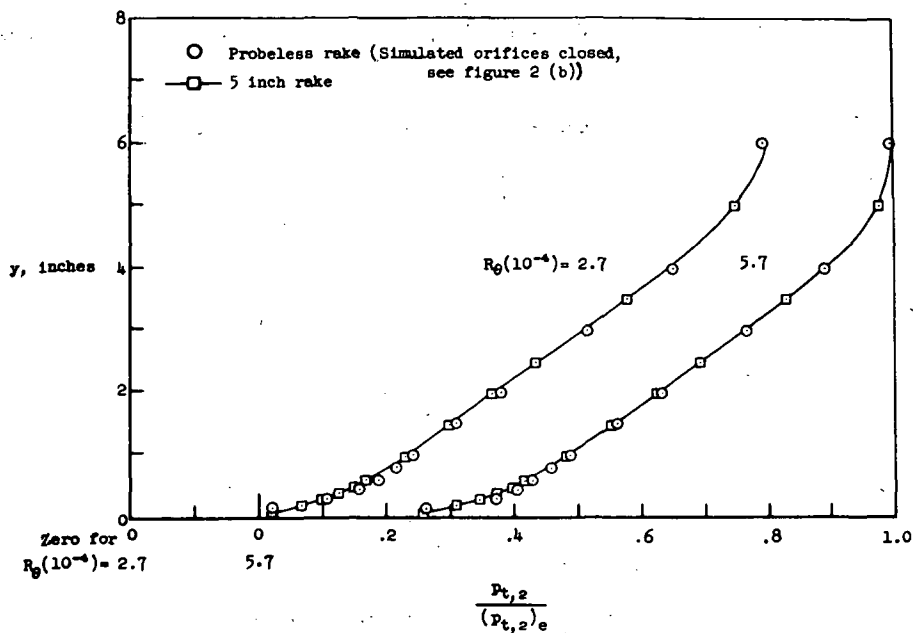
Geometry of Boundary-Layer Pitot-Pressure
Multiple-Probe Rakes



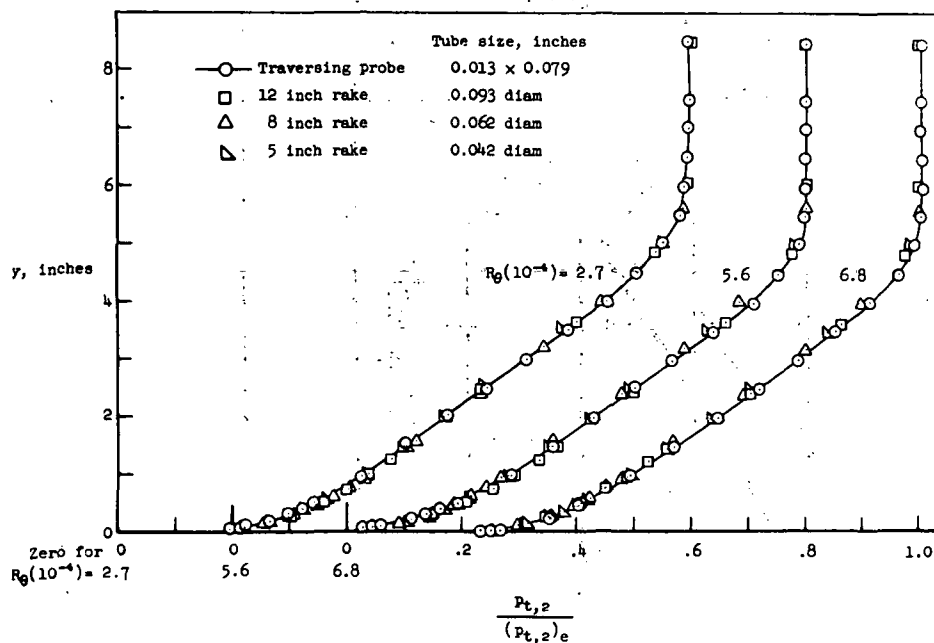
Note: All dimensions are in inches



Geometry of Traversing Probe



Comparison of Probeless Rake and 5 Inch Rake-Pitot Pressures; $M = 2.9$



Comparison of Rake and Traversing-Probe Pitot Pressures, $M = 2.9$. Results are Typical for $M = 2.4$ and $M = 3.4$

TITLE: Boundary Layers on Swept Wings: Their Effects and Their Measurement

AUTHOR: D. Kuchemann

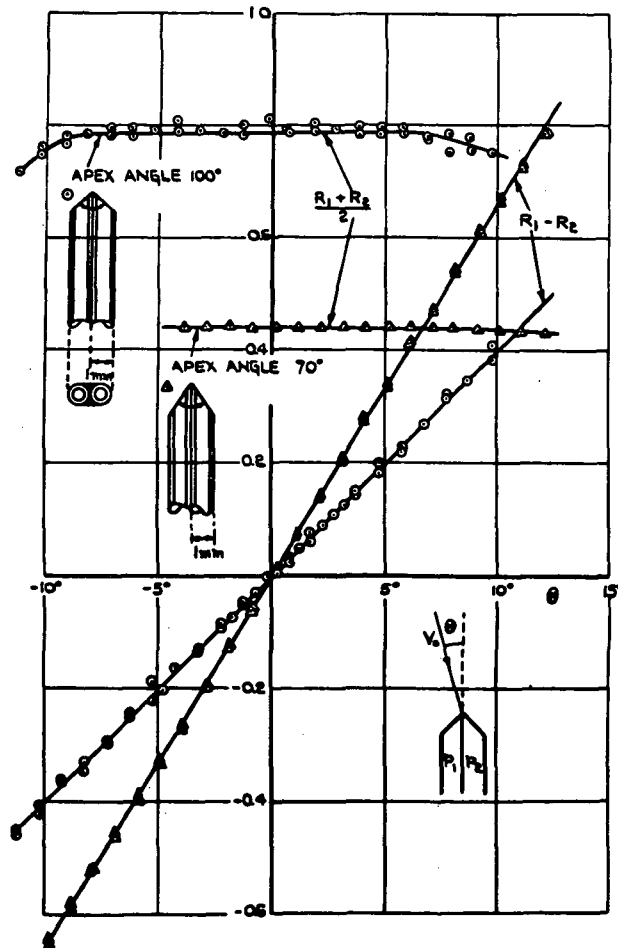
PUBLICATION: Royal Aircraft Establishment Technical Note AERO. 2370, April 1955

SUMMARY: Calibration data for two Conrad yawmeters is presented.

TEST CONDITIONS: Not available.

DEFINITIONS: $R_1 = (P_1 - P_{\text{STATIC}}) / \frac{1}{2} \rho V_o^2$

$R_2 = (P_2 - P_{\text{STATIC}}) / \frac{1}{2} \rho V_o^2$



Calibration Curves of Two-Tube Conrad Yawmeters

TITLE: Equipment Used for Boundary Layer Measurements in Flight

AUTHOR: F. M. Burrows

PUBLICATION: COA Note No. 49, The College of Aeronautics, Cranfield, July 1956

SUMMARY: Wind-tunnel data are presented demonstrating the performance of a transition indicator and a boundary-layer comb (boundary-layer mouse). The effects of yaw angle on the boundary layer comb and comparison with an .030-inch o.d. transversing pitot are included.

TEST CONDITIONS: Velocity, 80 and 130 ft/sec; Reynolds Number, not available; yaw angle, $0^\circ - 30^\circ$; pitch angle, assumed 0° .

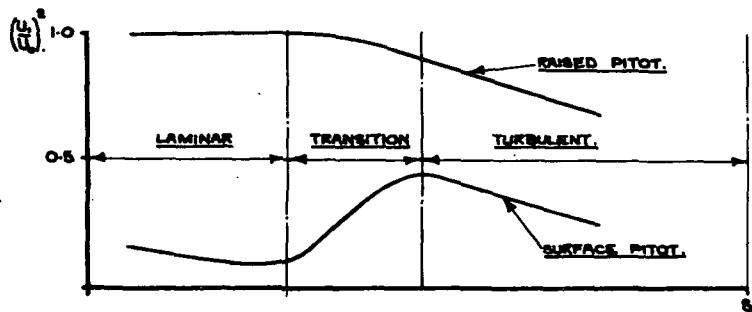
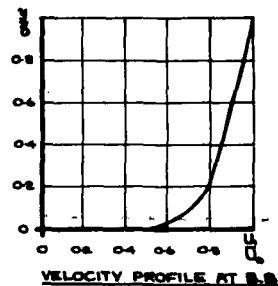
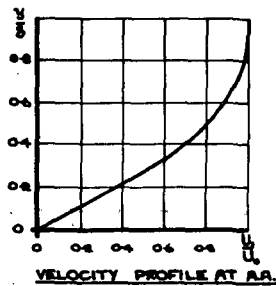
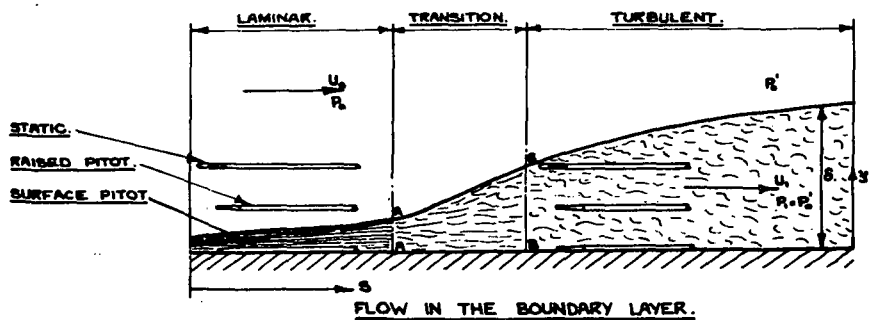
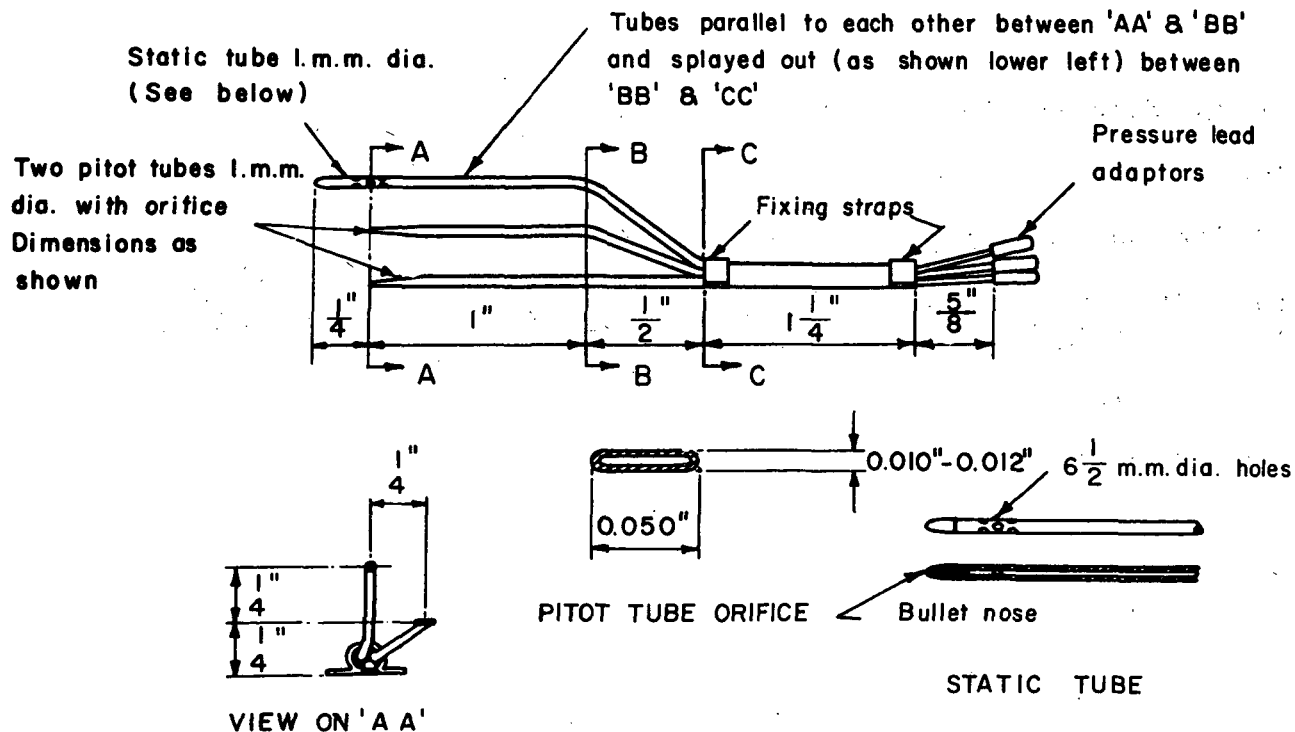
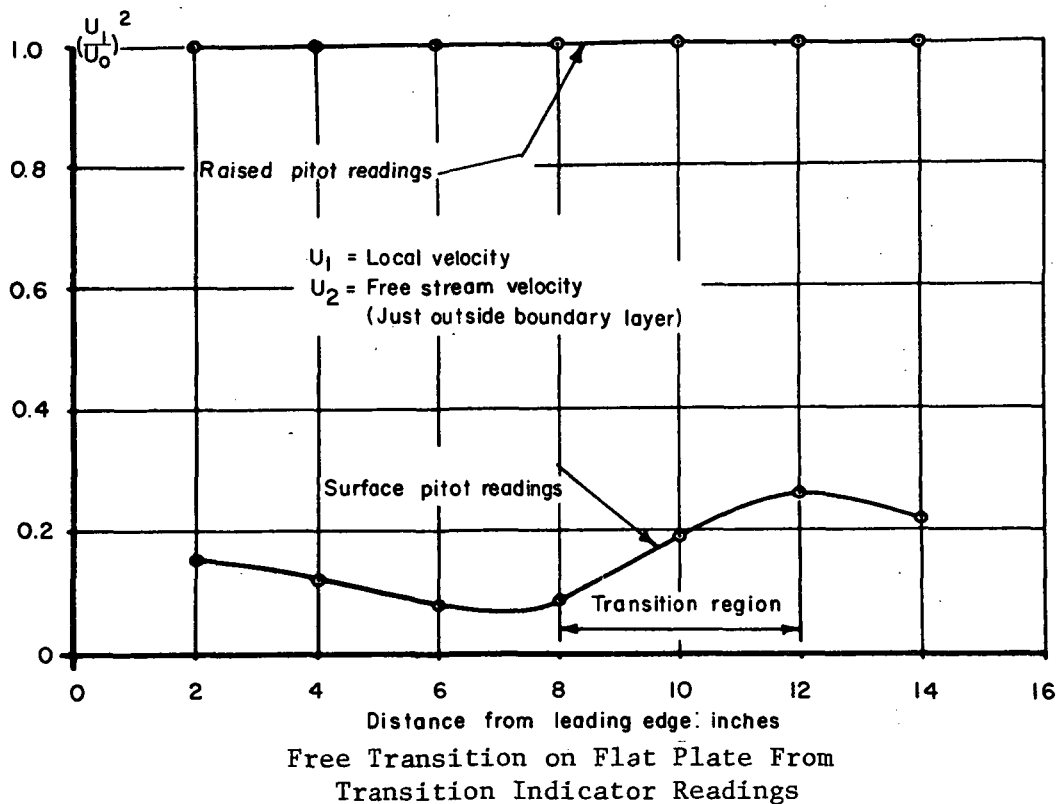
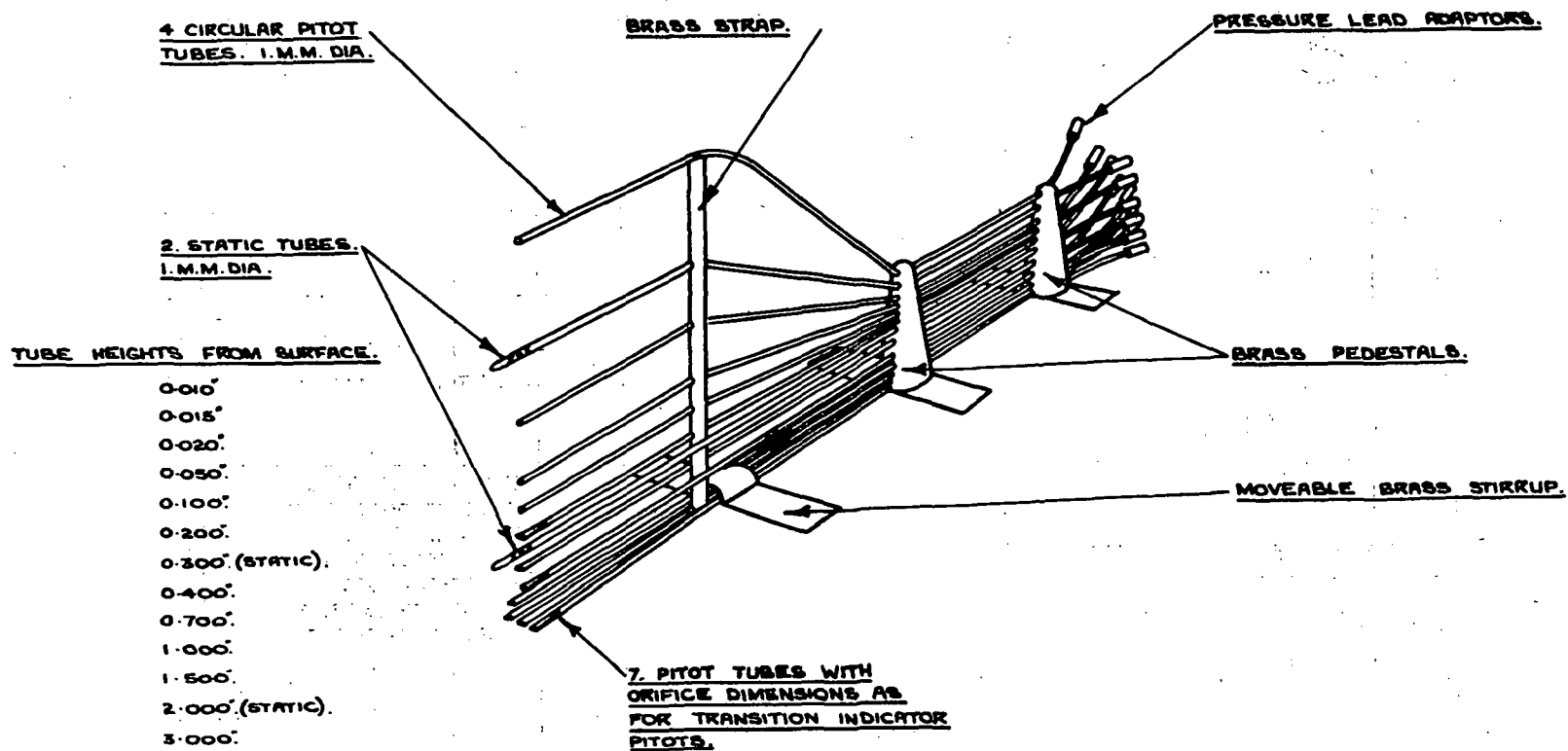


Diagram Showing Principle of Transition Location Using Raised and Surface Pitot Tubes

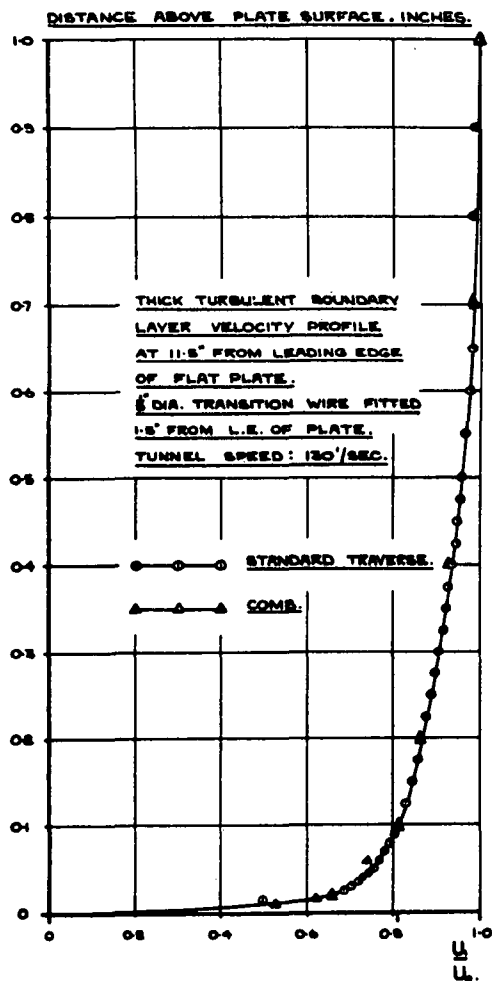


Transition Indicator

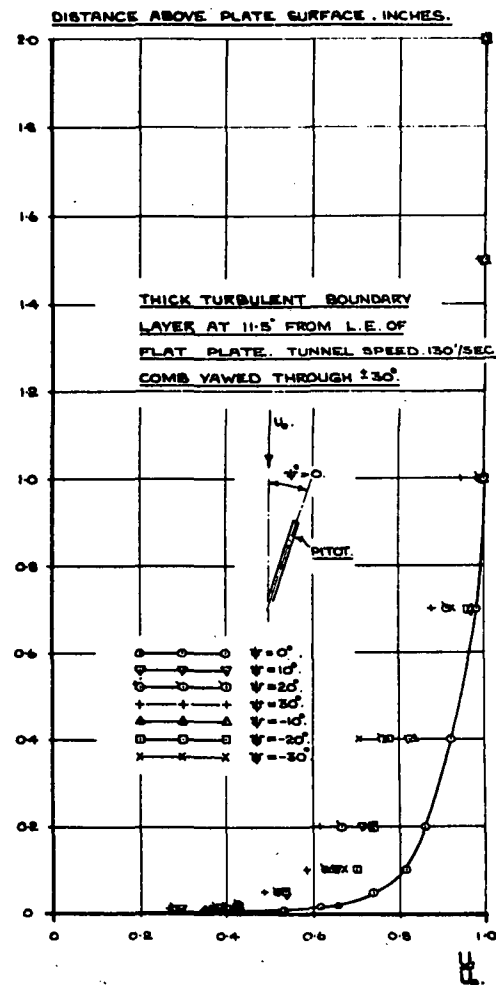




Boundary Layer Comb



Representative Comparison Between Transversing Pitot and Boundary Layer Comb Data



Effect of Yaw Angle on Boundary Layer Comb Data

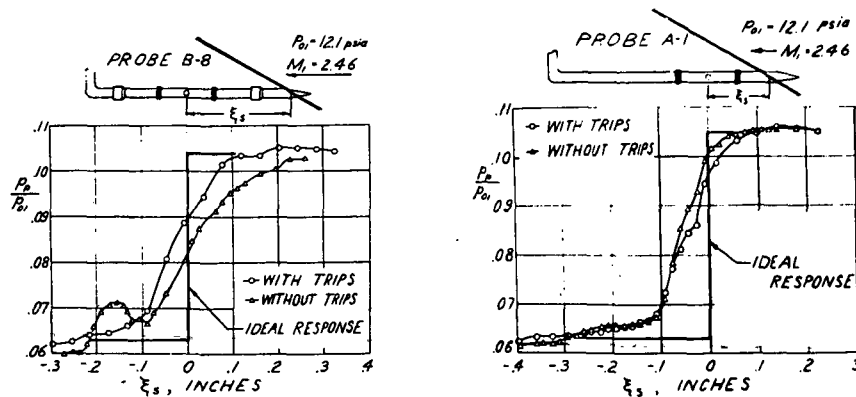
TITLE: A Static-Pressure Probe for the Interaction Region of a Shock Wave and Boundary Layer

AUTHOR: John E. Rode and Victor J. Skoglund

PUBLICATION: Journal of the Aerospace Sciences, vol. 29, October 1962, A62 10866

SUMMARY: Several miniature static-pressure probes were tested in a turbulent, compressible boundary layer without a shock and in an oblique shock outside of the boundary layer where the flow conditions were known. The probes were tested with variations in the location of the pressure ports and with a variety of boundary-layer tips on the probe. All probe tips were made of 0.016-inch o.d. stainless-steel tubing and had 0.004-inch diameter static ports in the side wall. The conical nose angle was 15° . The ports of tube A-1 were 12 diameters from its point, and 17 diameters for tube B-8.

TEST CONDITIONS: Mach Number, 2.46; Reynolds Number, not available; yaw angle, assumed 0° ; pitch angle, assumed 0° .



Response of Probes A-1 and B-8 With and Without Trips in Plain Boundary Layer With an Oblique Shock Wave

TITLE: Yaw Probe Used as Preston Tube

AUTHOR: N. Rajaratnam and D. Muralidhar

PUBLICATION: Aeronautical Journal, vol. 72, December 1968, A69 16396

SUMMARY: The calibration results of a three-tube yaw probe for use as a Preston tube are presented.

TEST CONDITIONS: Velocity, not available; Reynolds Number, not available; yaw angle, $0^\circ - 60^\circ$; pitch angle, assumed 0° .

DEFINITIONS:

$$K_{10} = (P_{\text{STATIC}} - P_1) / (P_{\text{TOTAL}} - P_{\text{STATIC}})$$

$$K_{20} = (P_{\text{STATIC}} - P_2) / (P_{\text{TOTAL}} - P_{\text{STATIC}})$$

$$K_{30} = (P_{\text{STATIC}} - P_3) / (P_{\text{TOTAL}} - P_{\text{STATIC}})$$

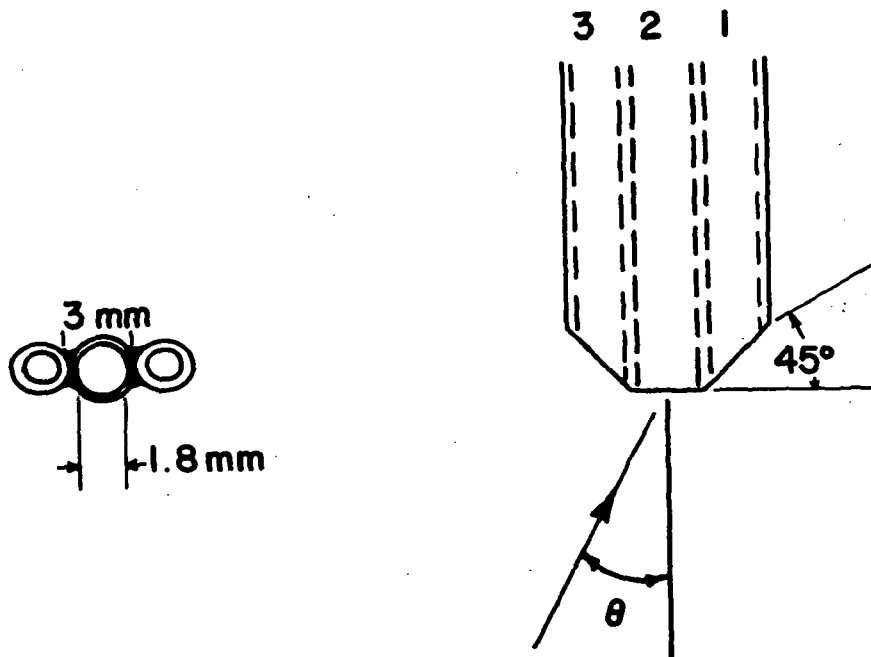
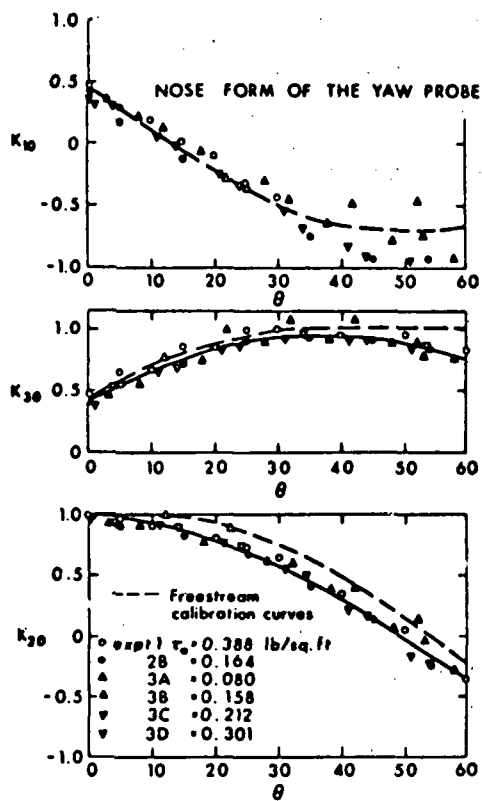
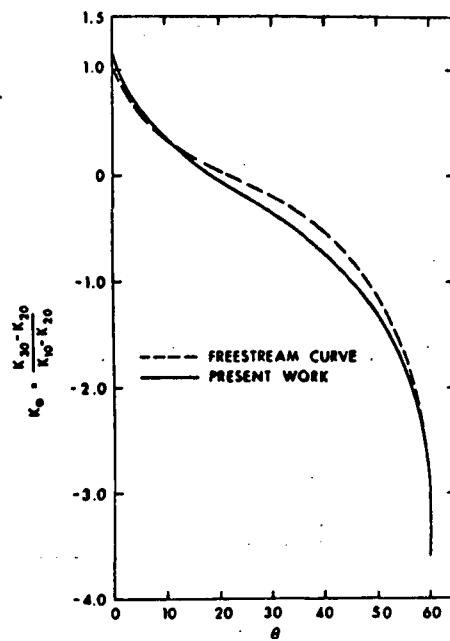
$$K_0 = (P_3 - P_2) / (P_1 - P_2) = (K_{30} - K_{20}) / (K_{10} - K_{20})$$


Diagram of the Probe Tip



Calibration Factors for the
Yaw Probe on the Boundary



Calibration Factor
 K_0 for the Yaw Probe

TITLE: Simplification of the Razor Blade Technique and its Application to the Measurement of Wall-Shear Stress in Wall-Jet Flows

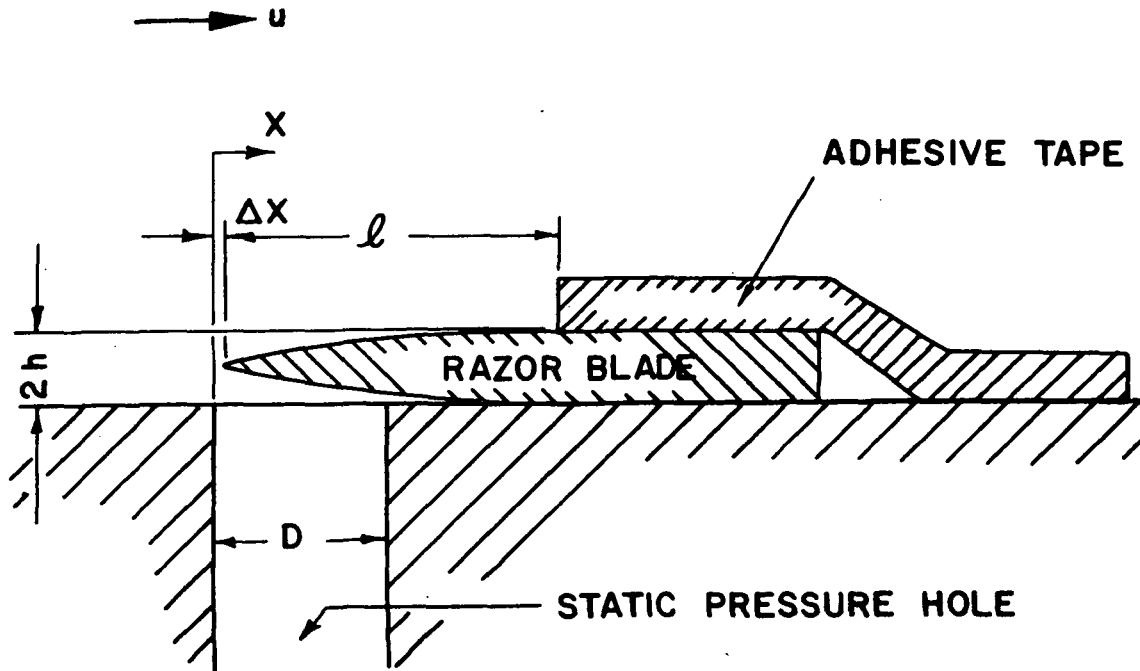
AUTHOR: B. R. Pai and J. H. Whitelaw

PUBLICATION: Aeronautical Quarterly, vol. 20, November 1969, A70 16504

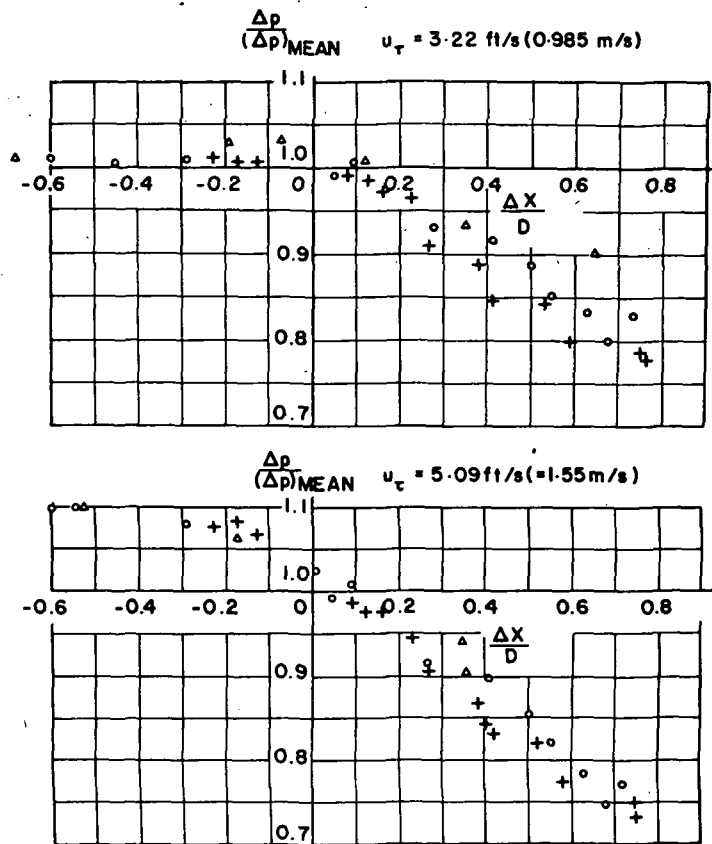
SUMMARY: Adhesive tape or carefully located cement can be used to secure a segment of a razor blade over a static pressure hole. The resulting calibration for shear stress remains valid if the blade is removed and relocated over the same or a different, similar sized hole. Razor blade segments, calibrated in this manner, have been used to measure wall-shear stress in a turbulent boundary layer with tangential, secondary injection.

TEST CONDITIONS: Velocity, 0 - 120 ft/sec; Reynolds Number, not available; yaw angle, assumed 0° ; pitch angle, not applicable.

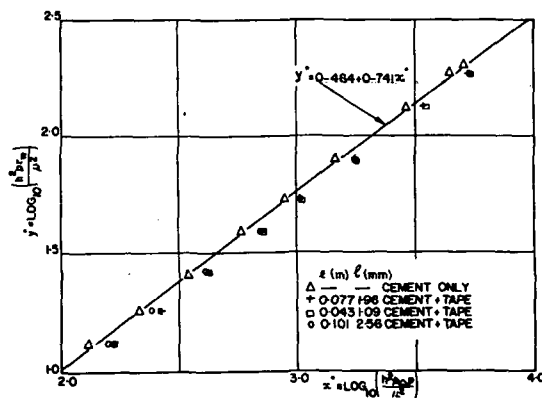
NOTE: $(\Delta p)_{\text{MEAN}}$ is not defined in the publication.



General Arrangement of Razor Blade Technique
For the Measurement of Wall-Shear Stress



Influence of Razor Blade Location and Static-Pressure-Hole Diameter on Measured Dynamic Head



Effect of Adhesive Tape on Razor Blade Calibration

3. Skin Friction Balances

No. 17

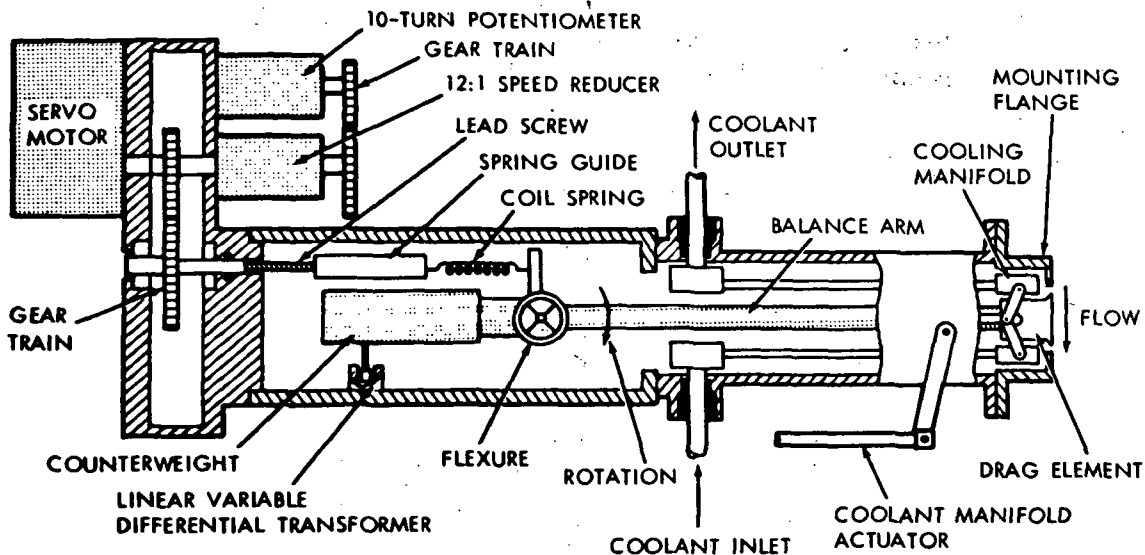
TITLE: Balance for Measuring Skin Friction in the Presence of Heat Transfer

AUTHOR: James R. Bruno, William J. Yanta and Donald B. Risher

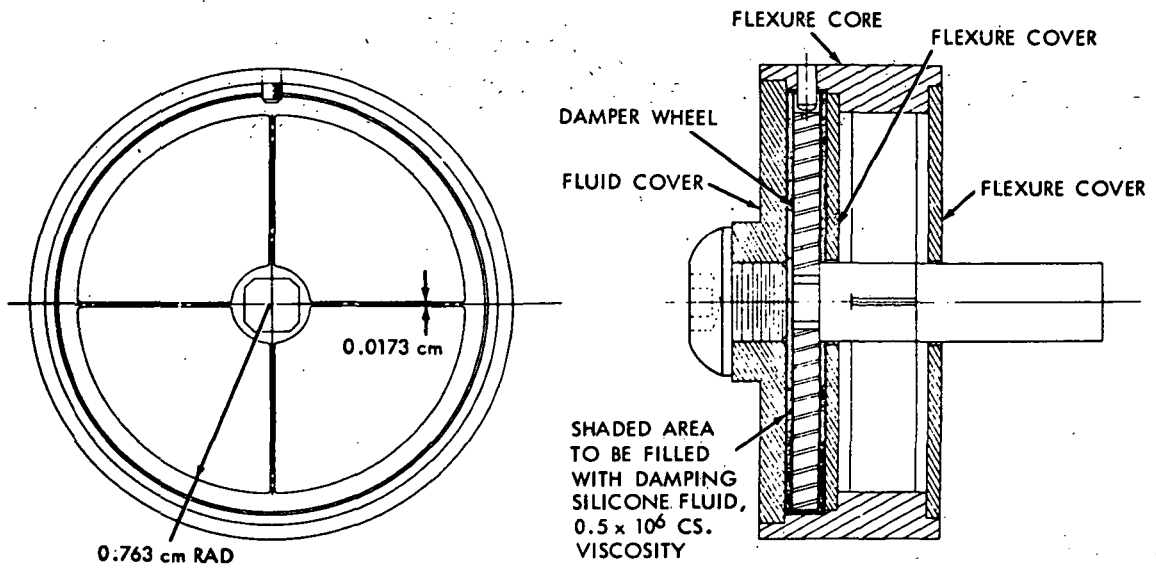
PUBLICATION: Report NOLTR 69-56, Naval Ordnance Laboratory, June 1969

SUMMARY: The development of a skin-friction balance to be used in a wind tunnel with heat-transfer conditions is described. The balance is a null-type device with a floating head element whose temperature can be maintained between 100°K and 345°K. This is accomplished with a cooled or heated jacket that is placed in direct contact with the friction element. At the desired element temperature the jacket is separated from the element and the shear-force data is taken. The balance was used in a Mach 5 supersonic flow with moderate heat-transfer rates. Shear forces ranging from 0.05 gm/cm² have been measured and higher ranges can be obtained by simply changing a coil spring. Calibrations showed that the balance was sensitive to the pressure inside the balance which is equal to the static pressure of the flow. Tests also showed that the pressure effect is constant and repeatable.

TEST CONDITIONS: Mach Number, 4.4 and 4.8; Reynolds Number, not available; yaw angle, not applicable; pitch angle, not applicable.

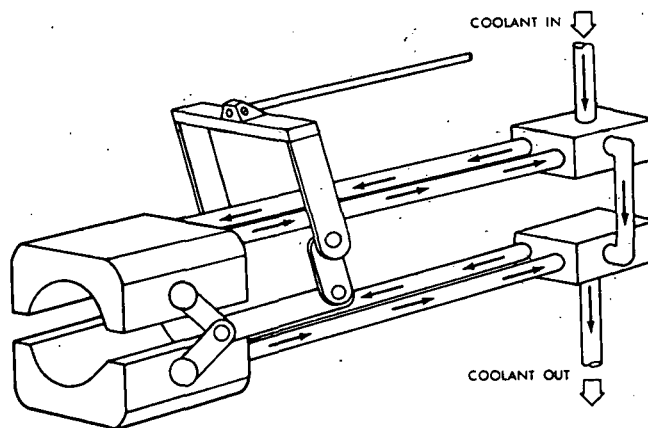


Balance Schematic

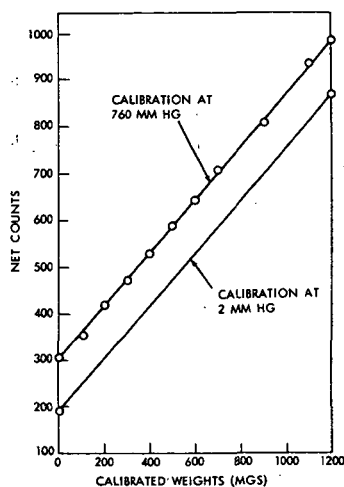


Flexure Core

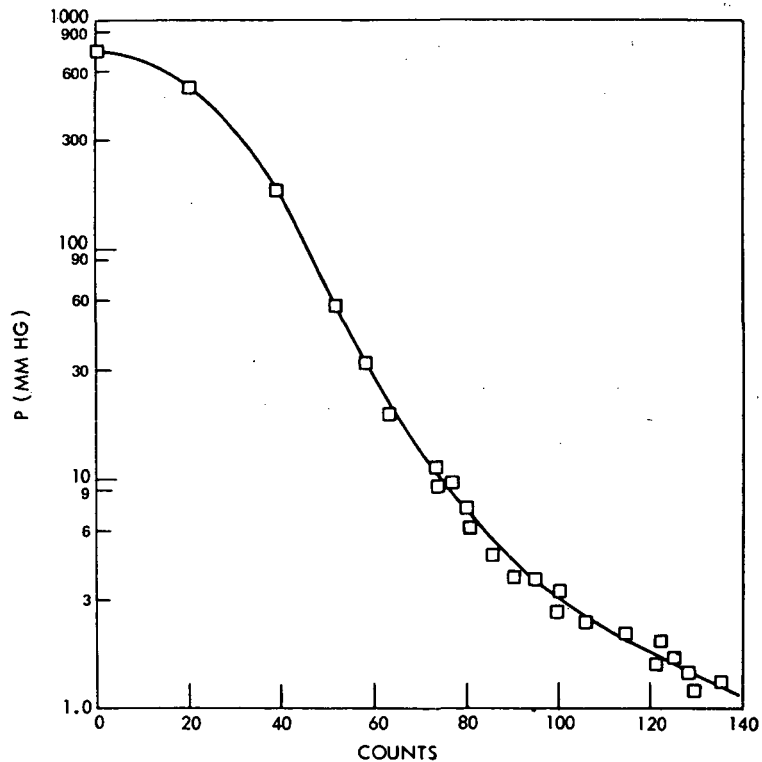
Flexure



Coolant Manifold



Typical Calibration Curve of NOL Balance



Pressure Calibration of NOL Balance

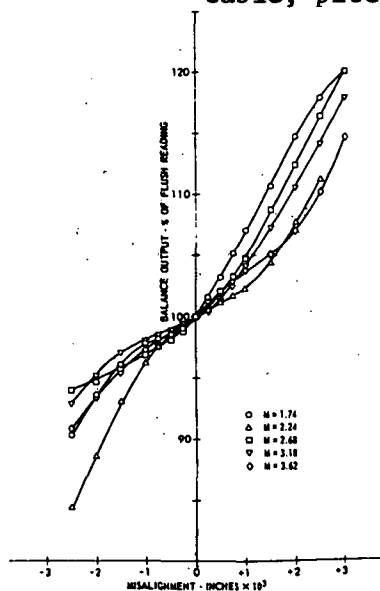
TITLE: A Study of the Effect of Floating-Element Misalignment on Skin-Friction-Balance Accuracy

AUTHOR: Francis B. O'Donnell, Jr.

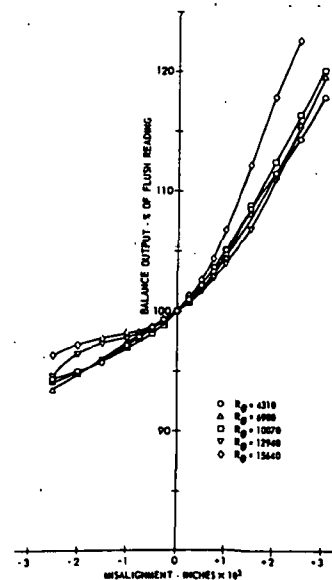
PUBLICATION: Report DRL-515, Defense Research Laboratory, University of Texas, March 1964

SUMMARY: An experimental investigation has been made of the effect of operating a floating-element-type skin friction balance with the element misaligned in the test surface. The misalignment consisted of mounting the element parallel to the test surface, but recessed below or projecting above the surface. A drive mechanism was constructed which permitted traversing the balance and its element through a range of approximately 0.003-inch of recess to 0.003-inch of projection. The effect of misalignment on balance output was determined in a continuous-flow wind tunnel over a Mach number range of 1.73 to 3.55, at several Reynolds numbers. The results indicated that any degree of misalignment resulted in a change of balance output. This error in output was slightly larger for the case of a projecting element than for the recessed-element case. No consistent correlation between misalignment error and either Mach number or Reynolds number was found.

TEST CONDITIONS: Mach Number, 1.74 - 3.62; Reynolds Number based on momentum thickness, $4.31 \times 10^3 - 1.564 \times 10^4$; yaw angle, not applicable; pitch angle, not applicable.



Effect of Misalignment
for Several Mach Numbers
at $R_\theta = 9830$



Effect of Misalignment
for Several Reynolds Numbers
at $M = 2.67$

4. Heat Flux Probes, Devices and Circuits

No. 19

TITLE: A Constant Temperature Hot-Wire Anemometer

AUTHOR: J. C. Wyngaard and J. L. Lumley

PUBLICATION: Journal of Scientific Instruments, vol. 44, May 1967

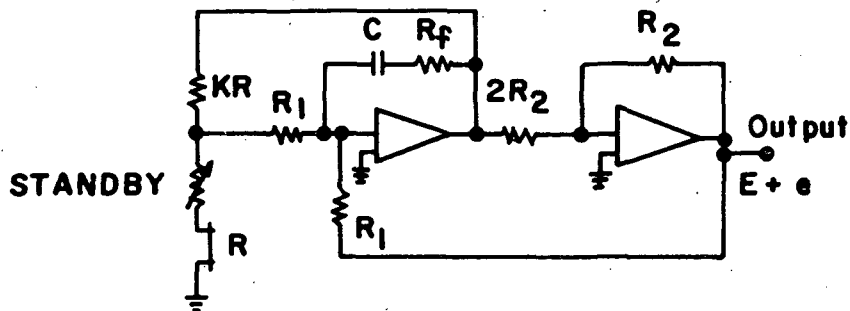
SUMMARY: A compact, inexpensive, constant temperature hot-wire anemometer circuit using solid state operational amplifiers is described. A theoretical expression for frequency response and an experimental check are given. Theoretical and experimental response data for an existing unit are shown.

CIRCUIT DESCRIPTION: Amplifiers are Philbrick P65AU, the first with a P66A booster follower. Maximum sensor current ± 100 ma. Frequency response is flat to 10kHz

$$R_1 = 1.5 \text{ k} \quad R_2 = 5 \text{ k} \quad R_f = 22 \text{ k}$$

$$C = 270 \text{ pf} \quad K = R_1 / (R_1 + R)$$

R = Sensor Operating Resistance



Anemometer Circuit Schematic

TITLE: Self-Aligning Hot-Wire Probe

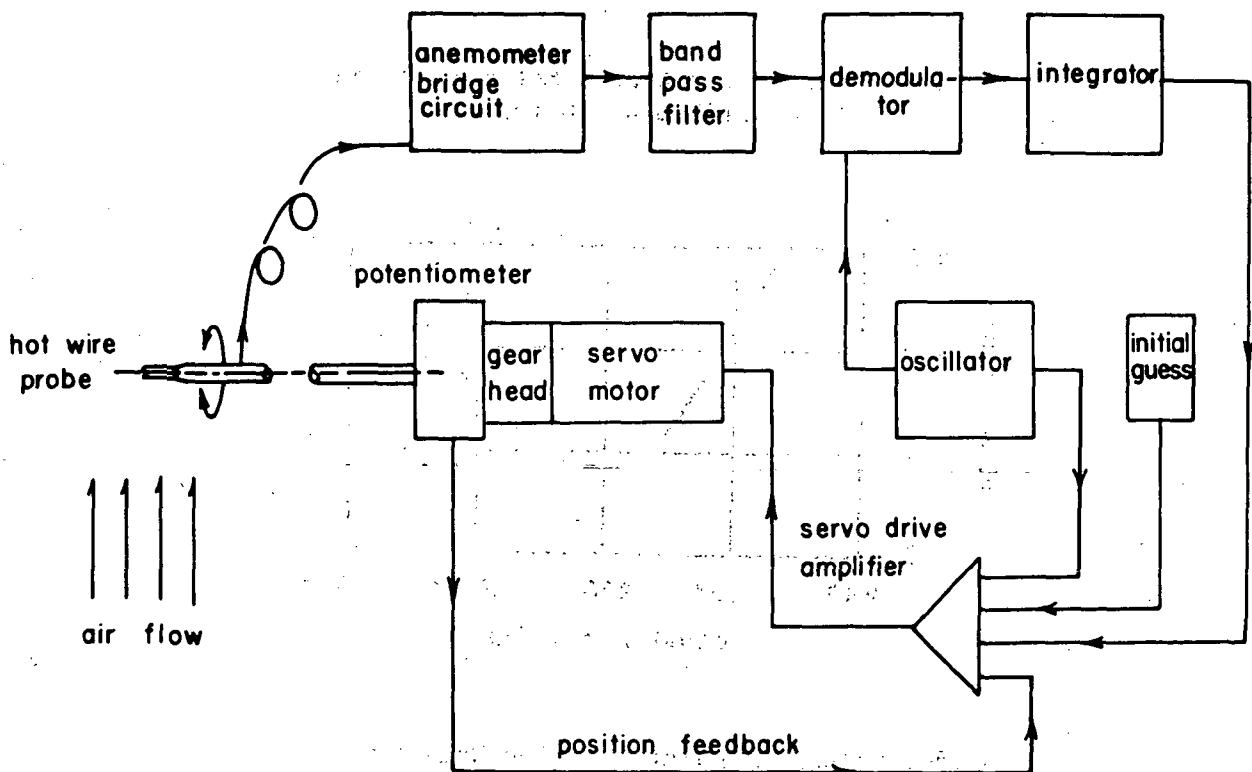
AUTHOR: A. D. Bond and A. M. Porter

PUBLICATION: Journal of the Royal Aeronautical Society, vol. 71, September 1967, A67 40069

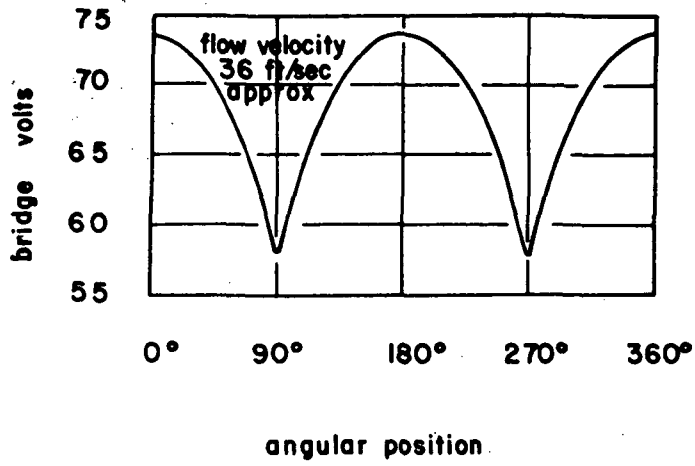
SUMMARY: Flow direction is determined by aligning a hot-wire probe parallel or normal to the flow with a hill-climbing servo-system using injected perturbations for the purpose of gradient computation. With parallel alignment, flow direction is measurable to within $1/40^\circ$, with normal alignment, to within 1° .

TEST CONDITIONS: Velocity, 0 - 36 ft/sec.

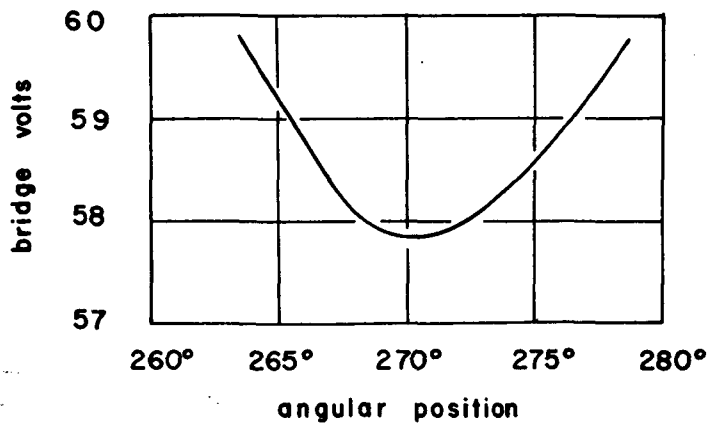
NOTE: Angular positions of 0° and 180° correspond to flow perpendicular to the wire.



The Automatic Control System



The Effect of Rotating a Hot-Wire Probe
in a Uniform Air Stream



Bridge Voltage/Angular Position Relationship
at the 270° Position

TITLE: Simple Miniature Probe Anemometer

AUTHOR: Errol V. Seymour

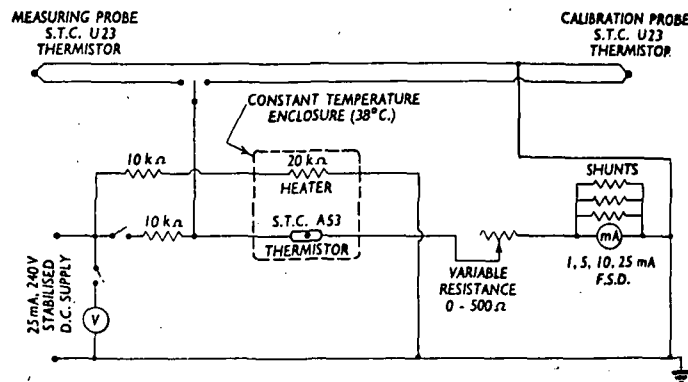
PUBLICATION: The Engineer, vol. 216, November 29, 1963

SUMMARY: Omni-directional velocity measurement in the range 1-300 fps can be measured to accuracies within 1/2%.

The calibration probe is placed in the flow at a point of known velocity. The measuring probe is initially placed at the same location for system calibration and then used to measure the velocity at other points of interest. Relative velocities can be measured by shielding the calibration probe from the flow velocity.

TEST CONDITIONS: Velocity, 1 - 300 ft/sec.

NOTE: No performance or calibration curves are presented in the publication.



Anemometer Circuit Schematic

TITLE: On the Development of a Hot-Wire Probe for Highly Turbulent Flow and Rapidly Reversing Flows

AUTHOR: A. A. Guenkel, R. P. Patel and M. E. Weber

PUBLICATION: Report No. 70-6, Mechanical Engineering Research Laboratories, McGill University, September 1970

SUMMARY: Two hot-wire sensors are mounted within a flow shield such that one is in the thermal wake of the other. The linearized signal of both wires are compared and the small rejected. To give the flow direction sense, the sign of one of the signals is inverted as it passes the amplifier gate.

TEST CONDITIONS: Velocity, 0.3 - 10.0 m/sec; Reynolds Number, not available; yaw angle, 0° - 360° ; pitch angle, assumed 0° .

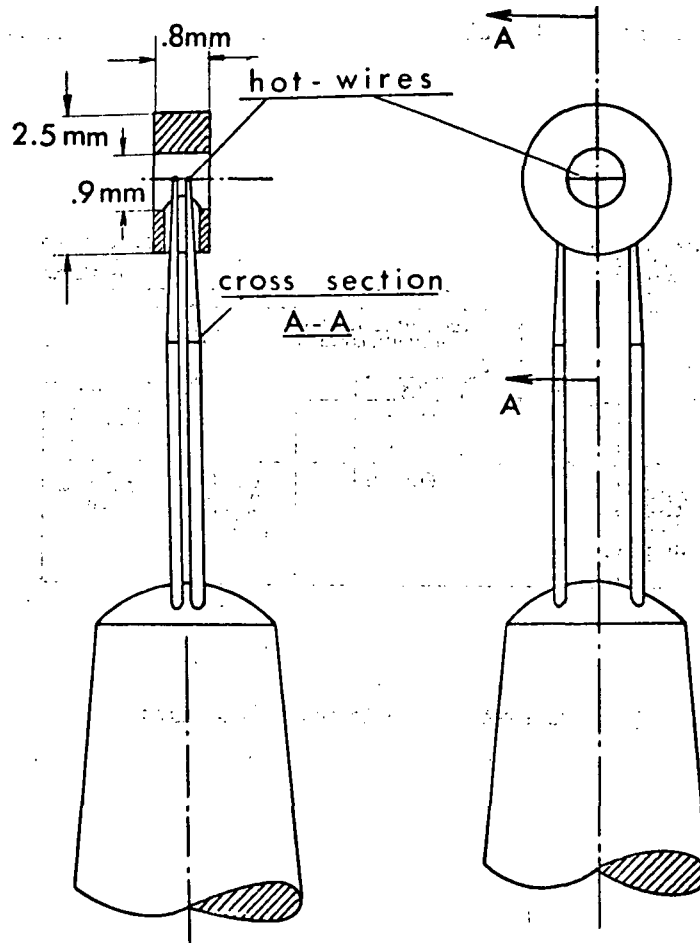
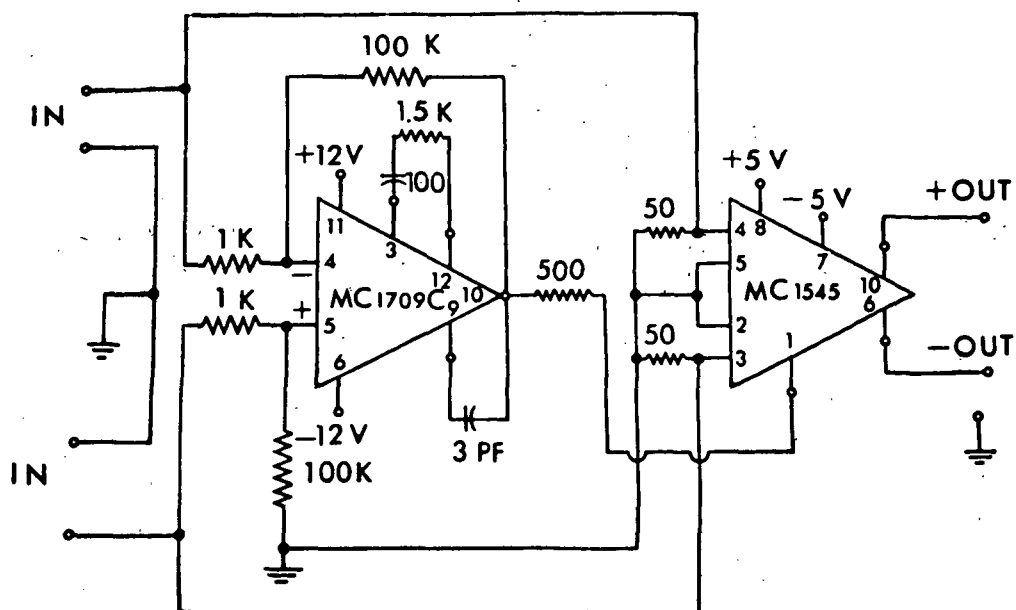
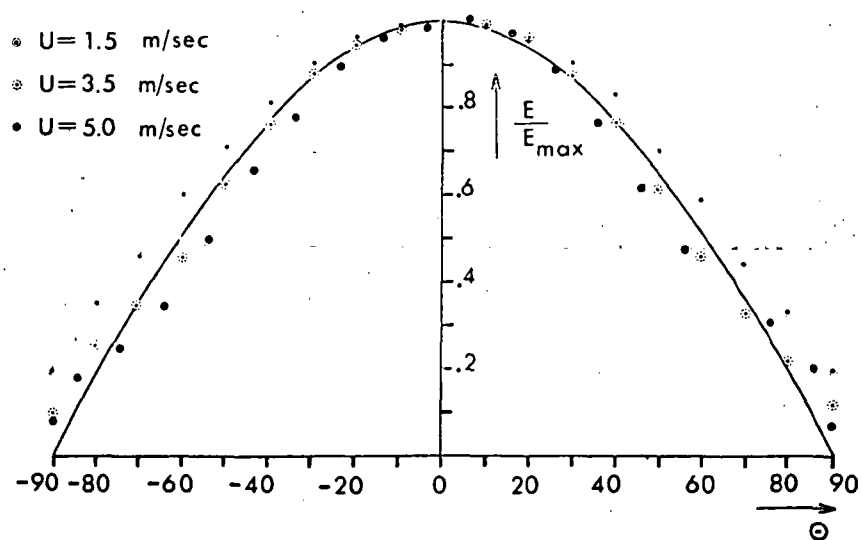


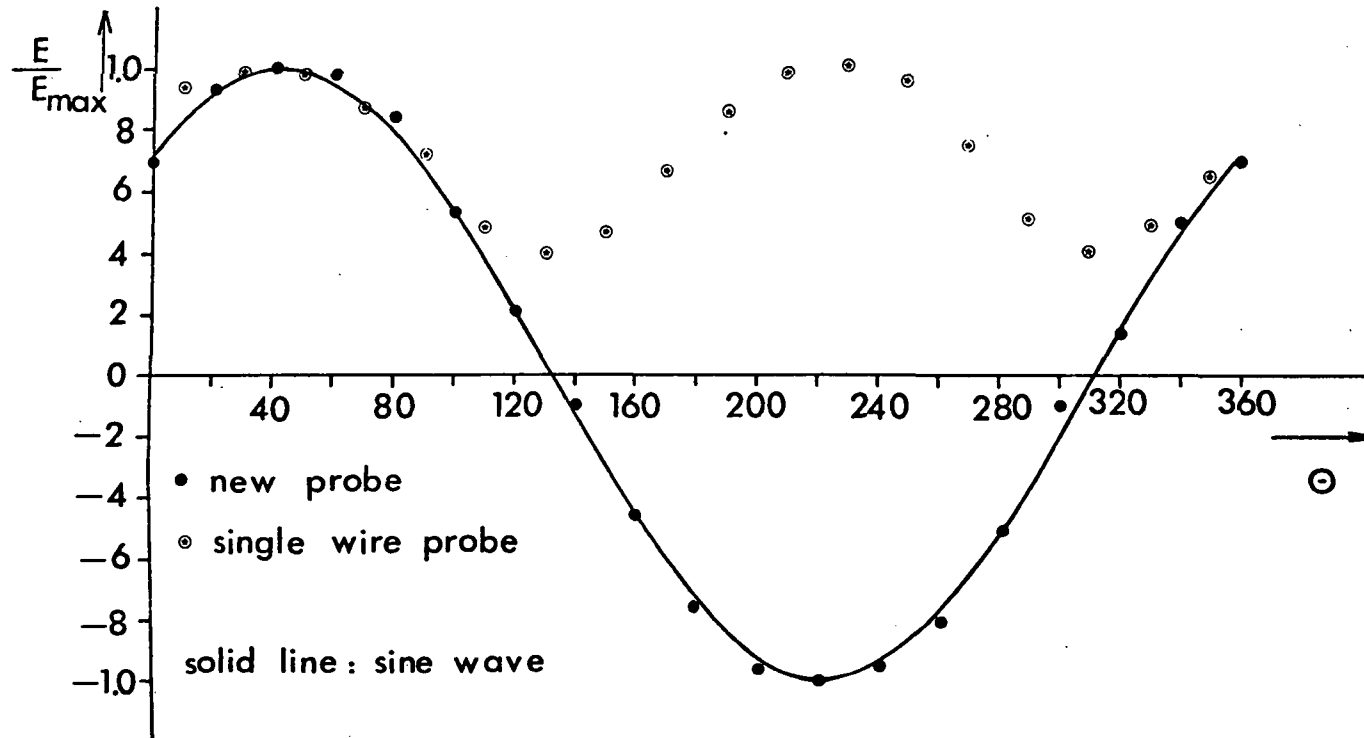
Diagram of the Probe



Switching Circuit Schematic



Directional Sensitivity of the Hot-Wire Probe



Directional Sensitivity of the Probe in a Highly Turbulent (50%) Flow Field

TITLE: A Pulsed Wire Technique for Velocity Measurements in Highly Turbulent Flows

AUTHOR: L. J. S. Bradbury

PUBLICATION: NPL Aero Report 1284, January 1969, N69 40576

SUMMARY: A velocity measuring technique essentially free from non-linear effects and with a well-defined response to variations in flow direction is described. The technique is based on that described by Bauer (1965) in which a "tracer" of heated air is produced by pulsing a fine wire of the sort used in hot-wire anemometry with a short duration pulse of electric current. The time taken for this "tracer" to be convected to another wire operated as a resistance thermometer is measured. Although the technique described in this note is in a very simple form, it is potentially quite powerful and, as an illustration, some measurements of both mean velocity and turbulent intensities in the highly turbulent reverse flow region behind some bluff bodies are described and the results compared with linearized hot-wire measurements.

TEST CONDITIONS: Velocity, 0 - 50 ft/sec; Reynolds Number, not available; yaw angle, $\pm 90^\circ$; pitch angle, $\pm 90^\circ$.

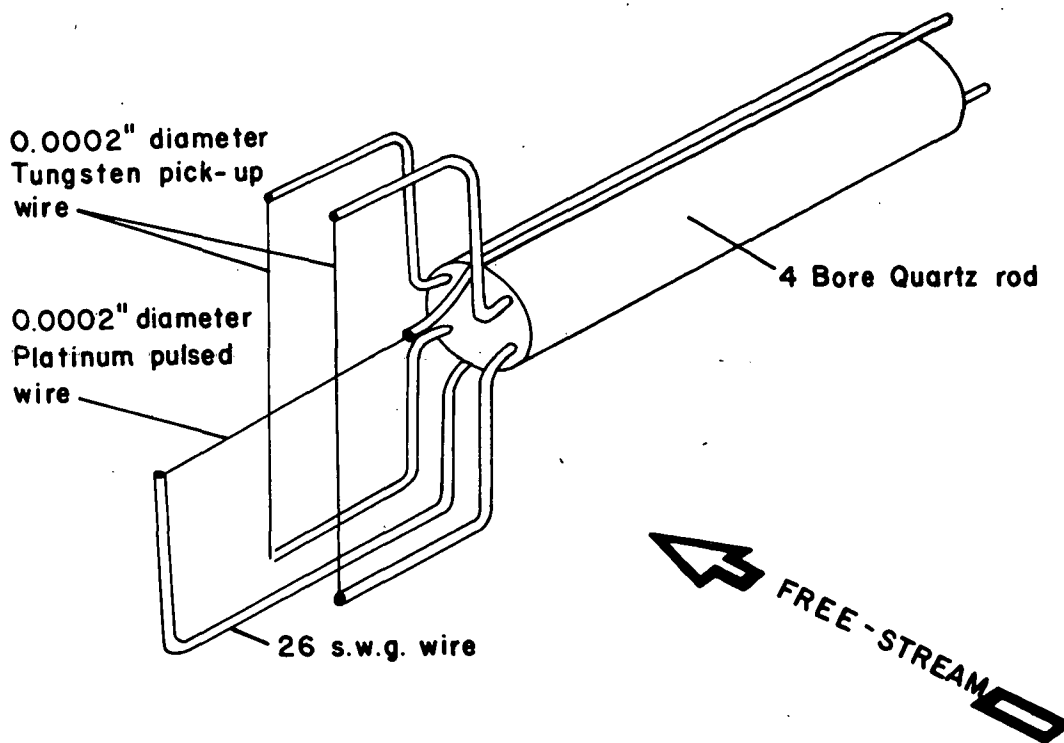
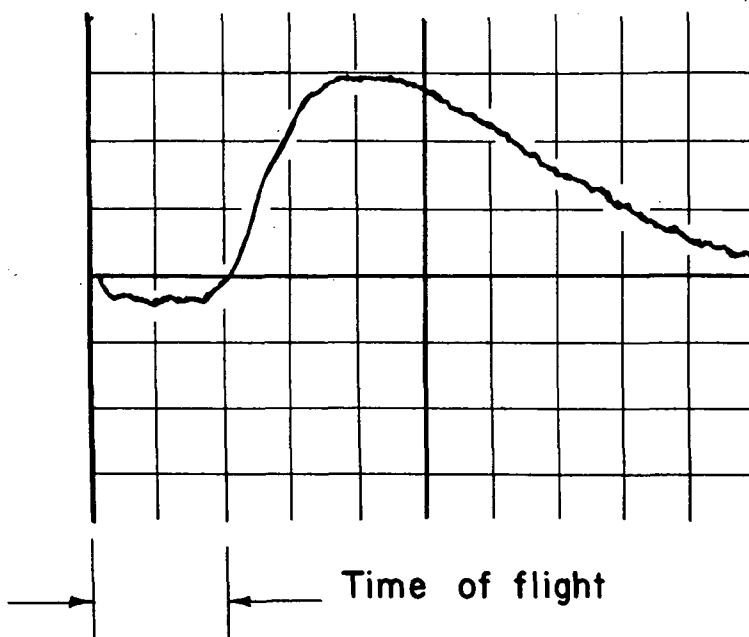
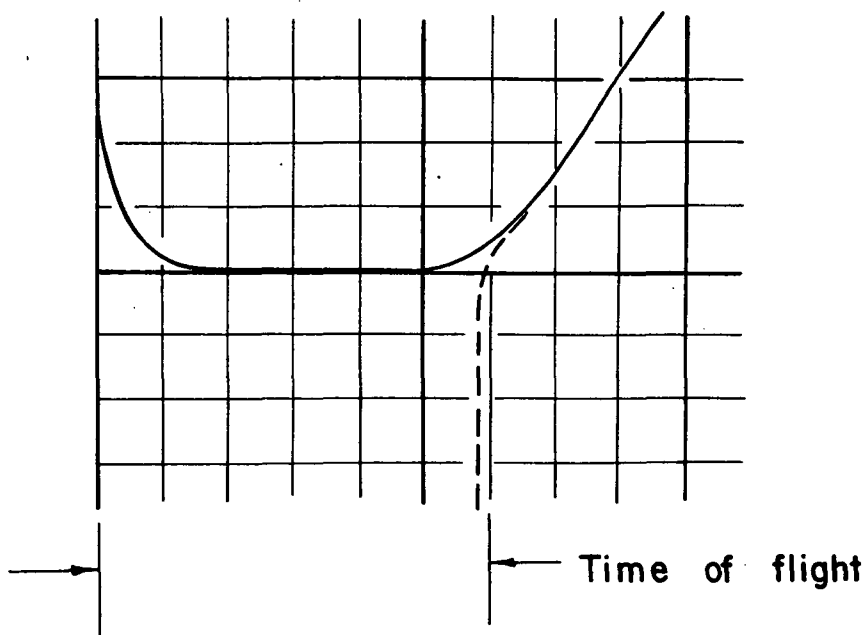


Diagram of Pulsed Wire Probe

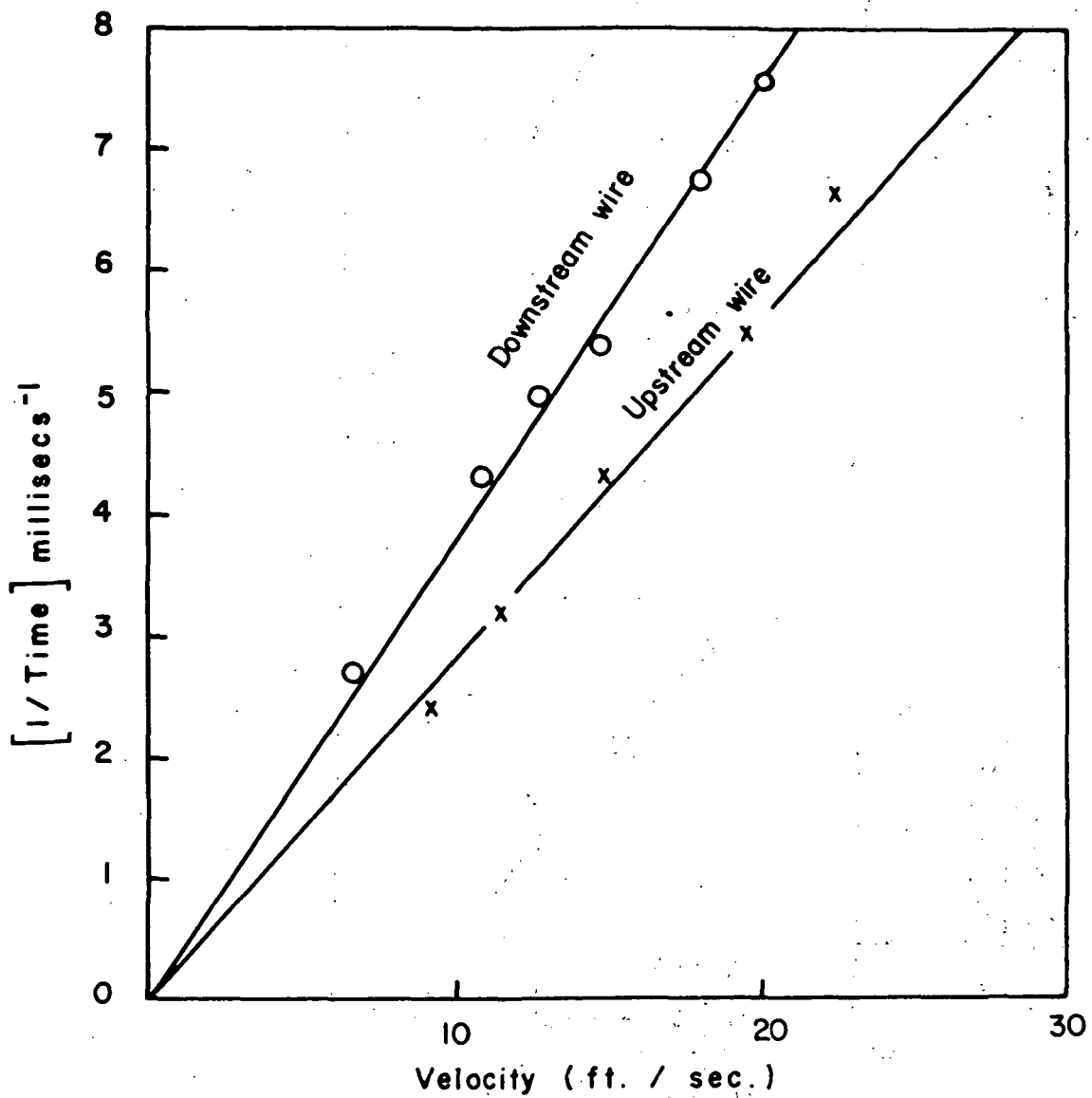


Trace of Pick-up Wire Signal. (200 $\mu\text{s}/\text{div.}$)

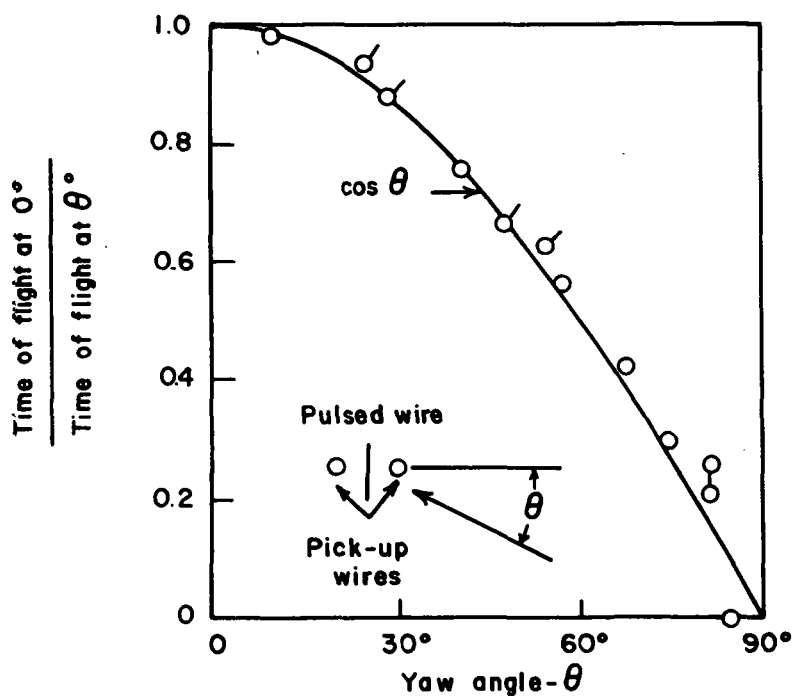
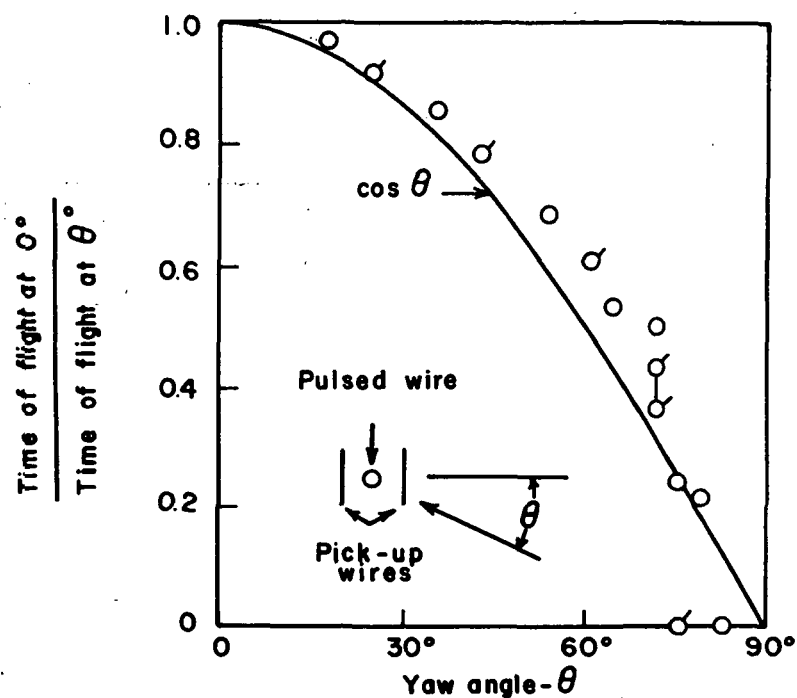


Expanded Trace. (100 $\mu\text{s}/\text{div.}$)

An Example of a Pick-up Wire Signal



Typical Calibration of a Crossed Wire Probe



Yaw Response of a Crossed Wire Probe.
Flagged and Unflagged Symbols are Results
on Opposite Sides of the Axis of Symmetry.

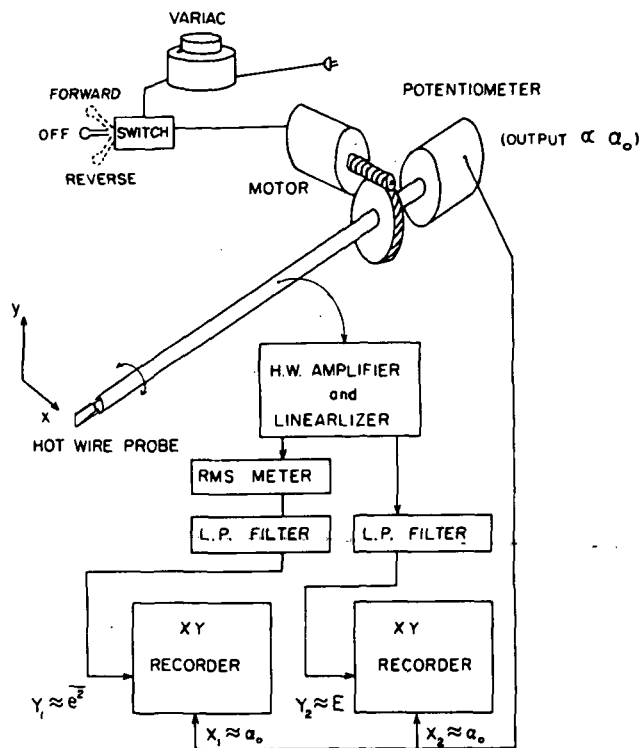
TITLE: Measurement of Reynolds Stress by a Single Rotated Hot-Wire Anemometer

AUTHOR: Hajime Fujita and Leslie S. G. Kovansznay

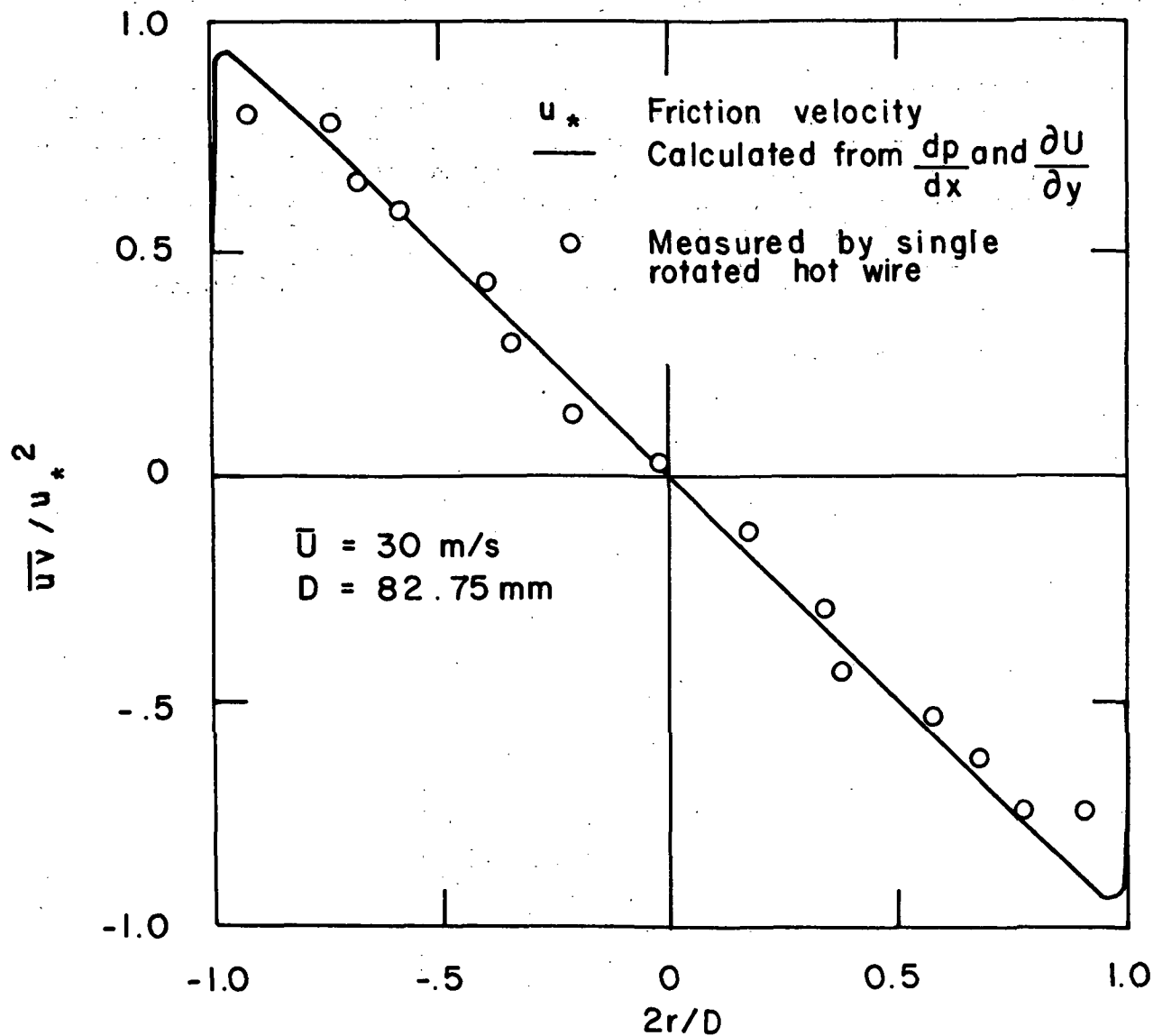
PUBLICATION: The Review of Scientific Instruments, vol. 39, September 1968, A68 41839

SUMMARY: A convenient method for the measurement of turbulence level and turbulent stress (u^2 , v^2 , uv) by a single, continuously rotated, hot-wire anemometer is presented. Emphasis is placed on the simplicity in the measuring procedure and on the accuracy in the results obtained. When comparing the results of Reynolds stress measurement in fully developed turbulent pipe flow, good agreement was found with values calculated from the measured pressure drop (skin friction coefficient) along the pipe. Rotational speeds are on the order of $20^\circ/\text{sec}$.

TEST CONDITIONS: Velocity, 30 m/sec.



Schematic Diagram of Equipment of Turbulence Measurement



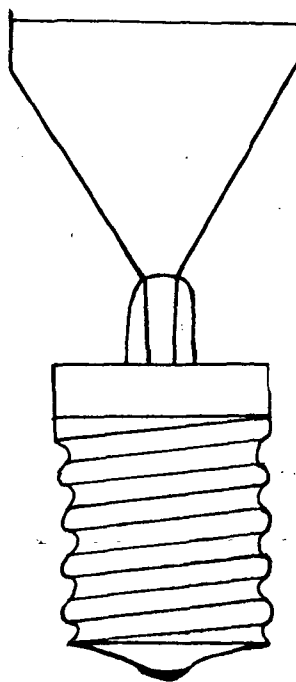
Reynolds Stress in a Fully Developed Turbulent Pipe Flow

TITLE: A Simple Hot-Wire Anemometer Probe

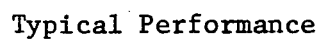
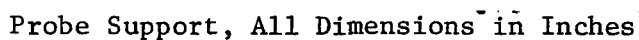
AUTHOR: E. G. Hauptmann

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments,
vol. 2, 1968

SUMMARY: An inexpensive, easily replaceable heat flux sensor can be obtained from miniature pilot lamps by removing the glass envelope and straightening, if necessary, the filament. The best results were obtained for a 24v lamp operated at an overheat of 1.5. The effective diameter is approximately 0.09 inches. The 1.5 overheat yields a time constant of 75 μ sec.



Peanut Bulb Element



TITLE: Directional Heat Meter for Wall Shear Stress Measurements in Turbulent Boundary Layers

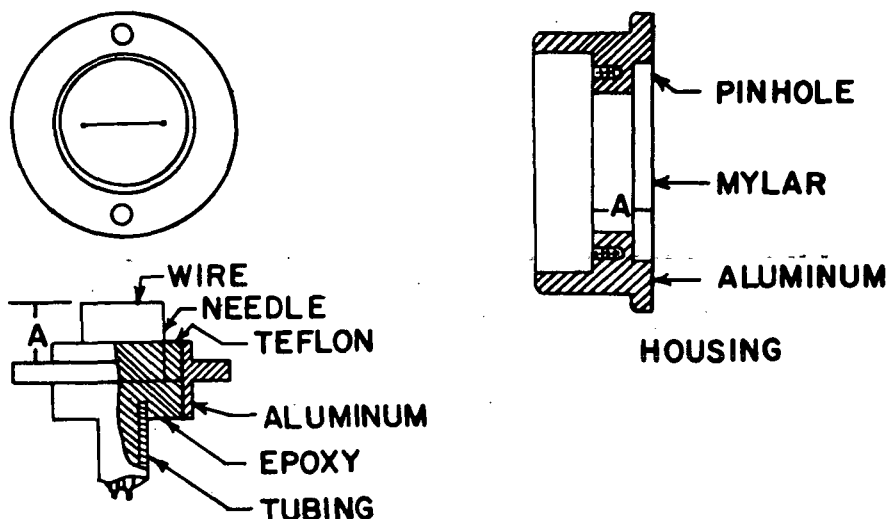
AUTHOR: R. H. Drinkuth and F. J. Pierce

PUBLICATION: Review of Scientific Instruments, vol. 37, June 1966, A66 34242

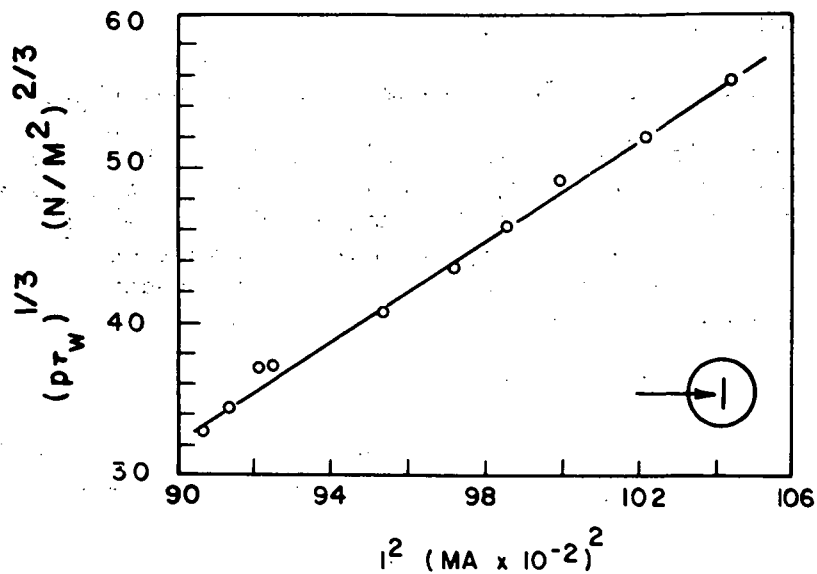
SUMMARY: The wall shear direction is first determined by arbitrarily setting the wire 10° - 20° to the flow, rotating the device to duplicate the reading and bisecting the included angle. Wall shear magnitude is measured by relating the electrical power dissipation I^2 to $(\tau)^{1/3}$. The calibration shown was obtained by using a Clauser chart to predict the wall shear, hence the quantity $p\tau_w$ is used instead of τ_w .

DEVICE: The wire holder is inserted into the housing such that the wire is in intimate contact with the mylar diaphragm (nominally 0.025 mm thick) without any surface distortions. A small hole (pinhole) is located in the diaphragm to relieve the pressure of the entrapped air.

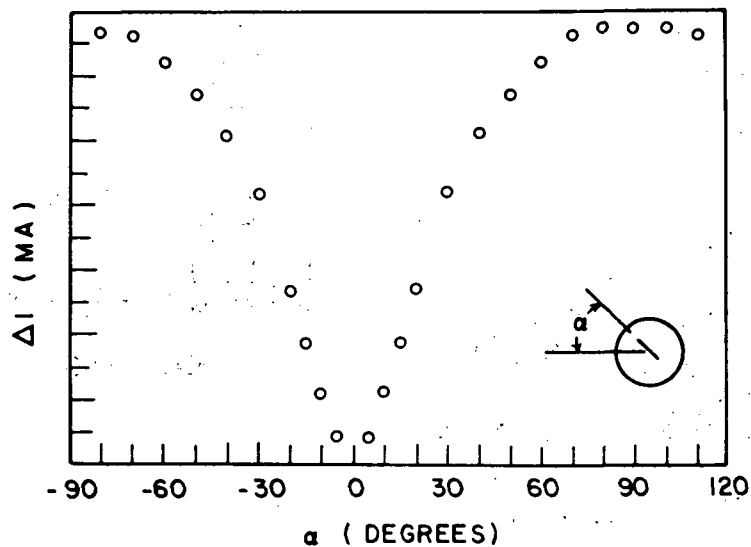
DEFINITIONS: I Wire Current
 τ_w Wall Shearing Stress
 p Local Static Pressure



Schematic of Wire Holder and Housing



Typical Calibration for a Two-Dimensional
Turbulent Boundary Layer Flow



Directional Response of Heat Meter

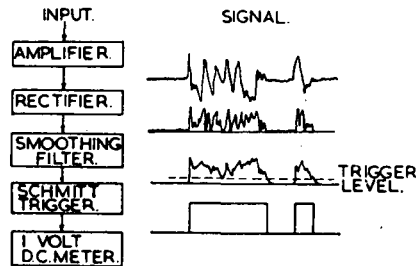
TITLE: Transition Experiments on a Flat Plate at Subsonic and Supersonic Speeds

AUTHOR: F. K. Owen

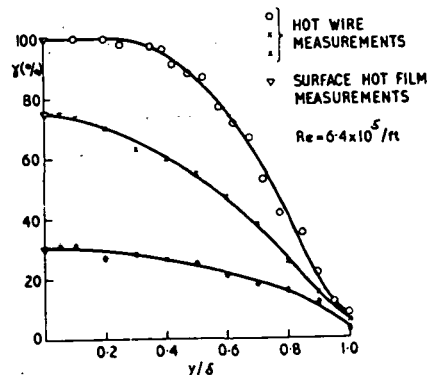
PUBLICATION: AIAA Journal, vol. 8, March 1970, A70 25970

SUMMARY: Thin flush film heat flux sensors are used to measure the intermittency in the transition region. The sensors are operated with constant temperature anemometer amplifiers and the signal is fed into a Schmitt trigger circuit to obtain a representation of the intermittency factor.

TEST CONDITIONS: Mach Number, 0 - 4.5.



Block Diagram of Intermittency Meter



Intermittency Distribution Across Subsonic Boundary Layer

TITLE: Two Techniques for Detecting Boundary-Layer Transition in Flights at Supersonic Speeds and at Altitudes Above 20,000 Feet

AUTHOR: John G. McTigue, John D. Overton and Gilbert Petty, Jr.

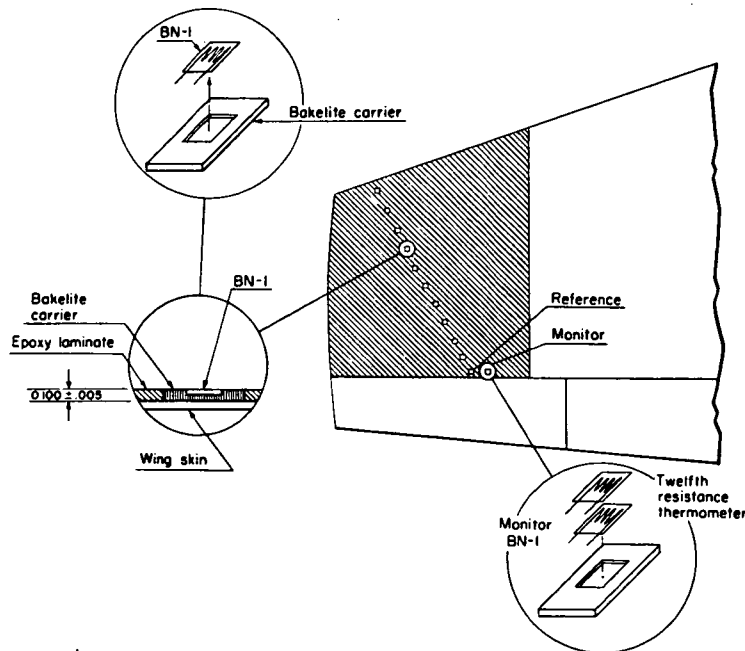
PUBLICATION: NASA TN D-18, August 1959

SUMMARY: Two techniques are presented which make it possible to locate transition on an aircraft during flight. In one method several resistance thermometers were attached to the wing of a supersonic fighter-type airplane. These thermometers were electrically connected to internal-recording circuitry to record a chordwise picture of the boundary-layer conditions on the surface of the wing.

The other method involved the use of sublimable chemicals to obtain a visual indication of the laminar area. Cameras were utilized to record the sublimation process during flight. The sublimation process is ideal for steady-state flight conditions, and also detected protuberances which were causing transition wedges to form ahead of the resistance thermometers used for steady- and maneuvering-flight conditions.

The two techniques proved to be compatible and complementary when used concurrently during steady-state flight conditions.

TEST CONDITIONS: Mach Number, 0 - 2.5; altitude, 0 - 55,000 feet.



Detail Sketch of Installation of BN-1
Resistance-Thermometer Boundary-Layer Detector

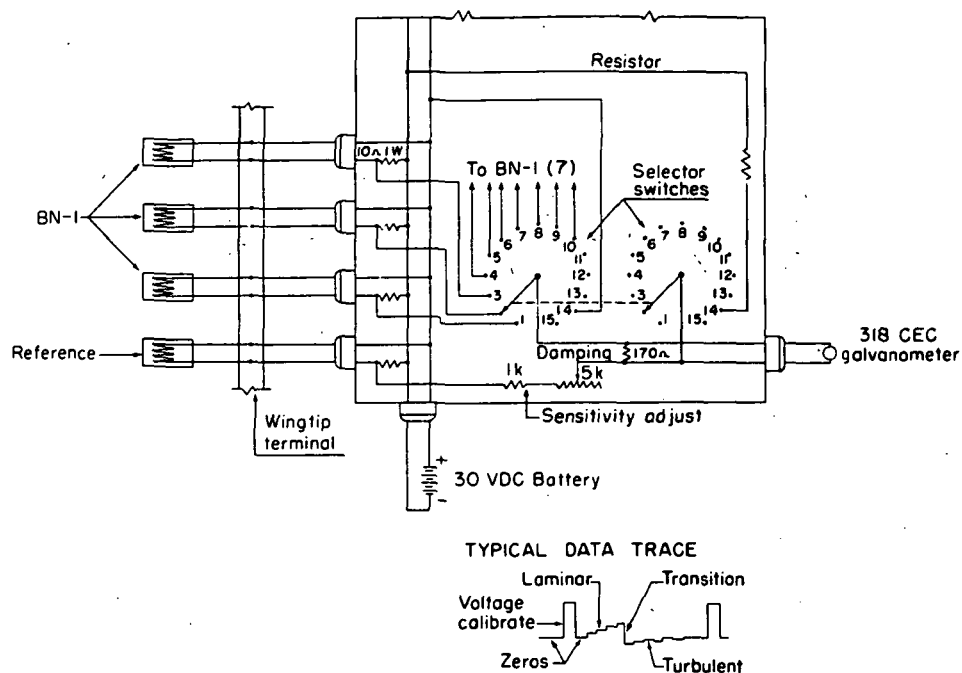
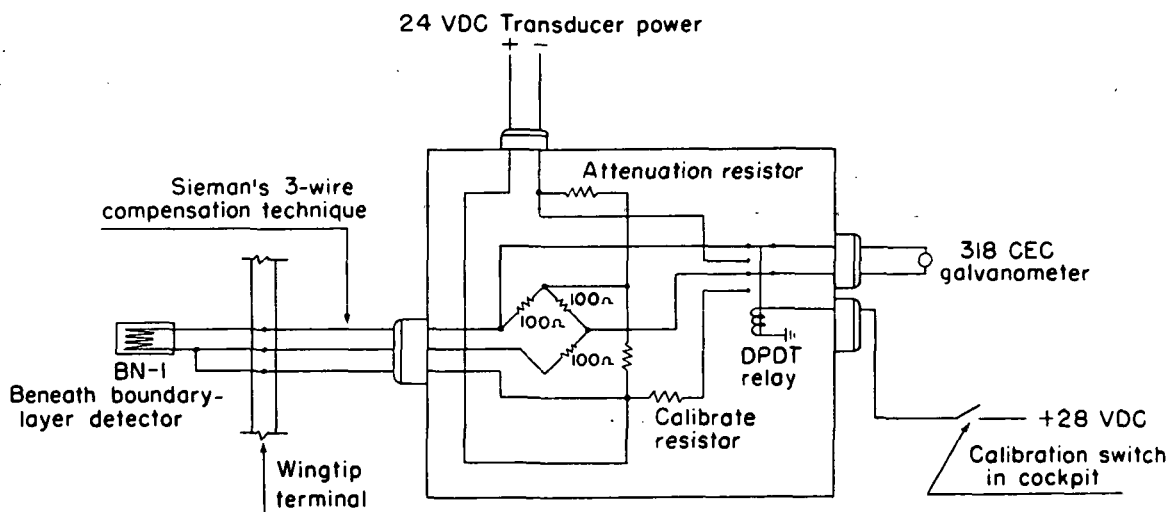
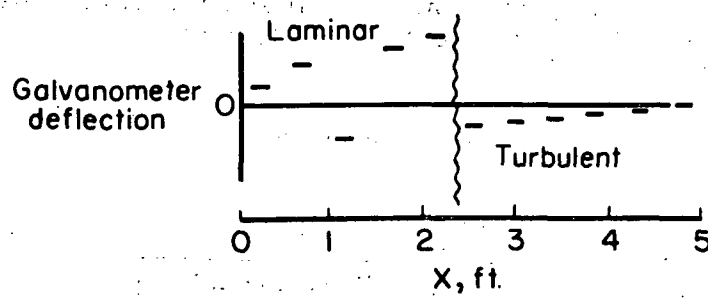
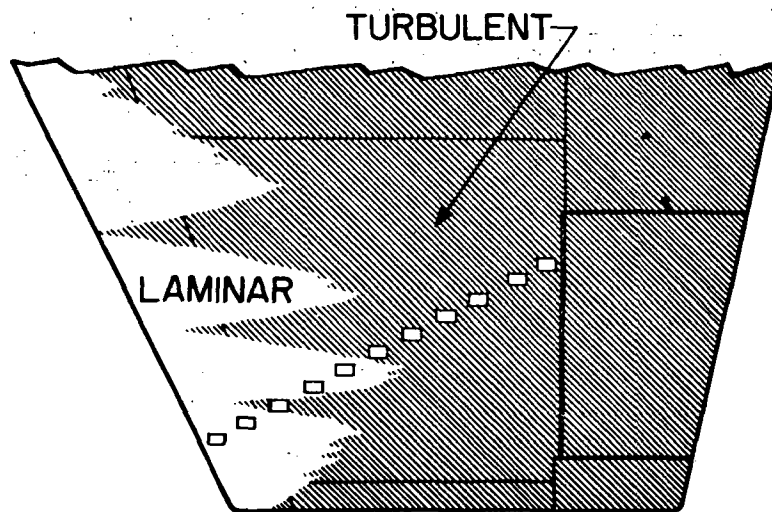


Diagram of BN-1 Resistance-Thermometer Boundary-Layer Transition-Detection Circuit



Drawing of Recording Circuitry of Temperature-Sensing BN-1 Monitor



Sketch Showing Correlation of Sublimation Technique and Resistance-Thermometer Method

B. GENERAL REFERENCES PERTAINING TO MEASUREMENT DEVICES

1. General References

No. 29

TITLE: The Measurement of Air Flow (4th Edition)

AUTHOR: R. C. Pankhurst and E. Ower

PUBLICATION: Oxford, Pergamon Press, Ltd., 1966

SUMMARY: A general theoretical analysis of the pressure-tube anemometer is given, and subsequent chapters discuss various types of air flow meters and measurement methods based on the cooling rates of hot bodies. Also covered are the plate orifice, the shaped nozzle, and the venturi tube; compressible, incompressible and pulsating flow measurements are considered, and a number of practical applications are given.

No. 30

TITLE: Aerodynamic Measurements

AUTHOR: Robert C. Dean, Jr.

PUBLICATION: The M.I.T. Press, 1953

SUMMARY: Various devices and equipment are presented for the measurement of flow temperature, pressure, velocity and direction. The fundamental principles of the different measurement techniques are thoroughly discussed with a particular emphasis on the error involved.

2. Pneumatic Probe Measurements in Boundary Layers and Wakes

No. 31

TITLE: Measurement of Flow Speed and Flow Direction by Aerodynamic Probes and Vanes

AUTHOR: W. Wuest

PUBLICATION: Paper presented at the 30th Flight Mechanics Panel Meeting in Montreal, Canada, 30th May - 2nd June, 1967, N68 23929

SUMMARY: After defining some quantities having the dimensions of pressure, in subsonic and supersonic flow, different types of pitot probes, static

probes, pitot-static probes and flow direction probes including vanes are discussed. Special attention is given to the probes and flow direction sensors which are especially suited for flight tests. The corrections due to deviations of flow direction, pressure gradients, neighborhood of wall and position, low density, Mach number and turbulence are discussed in detail.

No. 32

TITLE: An Evaluation of Four Experimental Methods for Measuring Mean Properties of a Supersonic Turbulent Boundary Layer

AUTHOR: George J. Nothwang

PUBLICATION: NACA Report 1320, 1957

SUMMARY: Surveys were made through a turbulent boundary layer on a flat plate by means of a pitot probe, an X-ray densitometer, and hot-wire and cold-wire probes. Results from these surveys were analyzed to determine (a) the reliability of the basic data and hence the methods by which they were obtained, and (b) how well the actual distributions of properties in the boundary layer compare with those commonly assumed in semiempirical and theoretical analyses. All surveys were made at the same longitudinal station on the flat plate. The tests were conducted in an 8- by 8-inch supersonic nozzle. The free-stream Mach number was 3.03 and the Reynolds number was approximately 210,000 based on the boundary-layer thickness.

Analysis of the data revealed the following points. The values of mean pitot pressure, mean density, and mean total temperature obtained from the pitot probe, X-ray densitometer, and cold-wire probe combined to produce consistent distributions of mean Mach number, mean total temperature, and mean mass flow throughout the boundary layer. However, mean mass flows computed from these data do not include the effects of combined density and velocity fluctuations and, hence, they can be in error for the supersonic turbulent boundary layer. The hot-wire probe indicated values of mean mass flow over the outer portion of the boundary layer that were higher than the values obtained from the pitot, X-ray, and cold-wire surveys. This result was confirmed in an independent test performed at a Mach number of 1.95 in the Ames 1- by 3-foot supersonic wind tunnel. It is suggested that corrections to hot-wire data may be required to obtain the true mass flows in turbulent supersonic boundary layers.

Except for the region very near the plate surface, the assumption of constant total temperature through the boundary layer yielded negligible errors in velocity distribution, and displacement and momentum thickness. The one-sixth power law was found to agree with the experimental velocity distribution within ± 2 percent.

No. 33

TITLE: Some Turbulent Boundary-Layer Measurements Obtained From the Forebody of an Airplane at Mach Numbers up to 1.72

AUTHOR: Edwin J. Saltzman and David F. Fisher

PUBLICATION: NASA TN D-5838, June 1970, N70 29908

SUMMARY: Boundary-layer-profile data were obtained from the smooth under-surface of an airplane fuselage during the demonstration of sensors for measuring boundary-layer characteristics. The data represent Mach numbers from 0.51 to 1.72, angles of attack up to 7° , and Reynolds numbers up to 74 million. The data are interpreted in terms of local skin friction and momentum thickness, and the velocity profiles from which these data are derived are tabulated.

Local transformed friction coefficients obtained from a Clauser type of determination from velocity profiles were close to the incompressible values of Karman-Schoenherr when presented as a function of momentum thickness influenced by angle of attack. The flight values of momentum thickness for angles of attack near 6° to 7° were lower than flat-plate values, approaching the level for slender cones. At angles of attack near 0° to 1° , momentum thickness from flight was higher than flat-plate values. The aircraft nose boom and the protuberances on the boom are believed to be major reasons for the additional thickness at low angles of attack.

No. 34

TITLE: The Behavior of Transverse Cylindrical and Forward Facing Total Pressure Probes in Transverse Total Pressure Gradients

AUTHOR: J. L. Livesey

PUBLICATION: Journal of the Aeronautical Sciences, vol. 23, October 1956

SUMMARY: When a total pressure probe is used for measuring flows with transverse total pressure gradients, a displacement of the effective center of the probe toward a higher total pressure is sometimes observed. This paper gives the results of an investigation of the effect for certain transverse cylindrical and forward facing (or pitot type) total pressure probes. An estimate of the error is given for the transverse cylindrical type of probe, and the errors due to position of hole, depth of hole, and wall proximity with this type of probe are considered incidentally.

A design of a probe of the forward facing type possessing negligible displacement error is suggested.

No. 35

TITLE: Influence of Velocity and Temperature Gradients in Gas Flow on the Accuracy of Flow Direction Measurements With Five-Tube Probes

AUTHOR: R. Ulken

PUBLICATION: Translation B-869:DLR-FB-68-65, Office of International Operations, Boeing Co., Seattle, Washington, February 1969, N69 28074

SUMMARY: After a short survey on flow direction measurement with five-tube probes in gas flows the report describes the influence of gradients of velocity and temperature on the accuracy of this method of measurement. The probes are calibrated in a uniform parallel flow. But if there are gradients in the flow field to be investigated, leading to different dynamic pressures on adjacent streamlines, an apparent flow velocity is obtained by using a probe calibrated in this way. These results can differ considerably from the real values. The influence of these gradients on the accuracy of flow direction measurement is calculated as a function of angle of attack, velocity, and temperature. The theoretical considerations are confirmed by experimental investigation.

No. 36

TITLE: The Evaluation of a Simplified Form of Presentation for Five-Hole Spherical and Hemispherical Pitometer Calibration Data

AUTHOR: M. A. Wright

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments, vol. 3, May 1970

SUMMARY: A theoretical and practical evaluation of a new method of calibration data presentation for five-hole spherical and hemispherical pitometers has shown that it is reliable and an advance over the traditional method in that it has: (i) simplified calibration curves; (ii) reduced the probability of misinterpretation of the data; (iii) become suitable for automatic computation.

Comparisons between the theoretical and practical calibration curves for several pitometer designs and hole configurations have shown that a hemispherical probe is superior to a spherical probe not only because of its simplified construction, but because improved flow conditions around the probe lead to superior calibration characteristics.

No. 37

TITLE: Effects of Pressure-Rake Design Parameters on Static-Pressure Measurement for Rakes Used in Subsonic Free Jets

AUTHOR: Lloyd N. Krause

PUBLICATION: NACA TN 2520, October 1951

SUMMARY: A subsonic-free-jet investigation was conducted to determine the effect of pressure-rake design parameters on static-pressure measurement. The design parameters investigated include the location of the static orifices in relation to the tube nose and supporting strut, the proximity effects of adjacent tubes near the static orifices, and the effect of the ratio of support diameter to jet diameter. The investigation covered a Mach number range of 0.3 to 0.95.

Results of the investigation revealed that the effect of variation in the distance from the leading edge of the static-pressure tube to the static orifices was small compared with the effect of variation of the distance from the static orifices to the supporting strut. The effect of variation in the ratio of support diameter to jet diameter became pronounced at high velocities. Proximity effects of adjacent tubes near static-pressure tubes may be alleviated by proper orientation of the leading edge of the adjacent tube in relation to the static orifices.

Information and recommendations are provided for the design of pressure rakes to be used in subsonic free jets.

No. 38

TITLE: Absolute Measurements of Static-Hole Error Using Flush Transducers

AUTHOR: R. E. Franklin and James M. Wallace

PUBLICATION: Journal of Fluid Mechanics, vol. 42, part 1, 1970, A70 32374

SUMMARY: Experiments are reported in which the error of a sharp-edged, circular static pressure hole normal to the boundary of a moving fluid is deduced from measurements made with pressure transducers, the diaphragms of which were set flush with the boundary surface. An error curve is presented which covers the range of hole Reynolds number up to a value of 2000.

No. 39

TITLE: Measurement of Static Pressure on Aircraft

AUTHOR: William Gracey

PUBLICATION: NACA Report 1364, 1958

SUMMARY: Existing data on the errors involved in the measurement of static pressure by means of static-pressure tubes and fuselage vents are presented. The errors associated with the various design features of static-pressure tubes are discussed for the condition of zero angle of attack and for the case where the tube is inclined to the flow. Errors which result from variations in the configuration of static-pressure vents are also presented. Errors due to the position of a static-pressure tube in the flow field of the airplane are given for locations ahead of the fuselage nose, ahead of the wing tip, and ahead of the vertical tail fin. The errors of static-pressure vents on the fuselage of an airplane are also presented.

A comparison of the calibrations of the four static-pressure-measuring installations indicates that, for an airplane designed to operated at supersonic speeds, a static-pressure tube located ahead of the fuselage nose will, in general, be the most desirable installation. If the operating range is confined to speeds below sonic, a static-pressure tube located ahead of the wing tip may, for some airplane configurations, prove more satisfactory than a fuselage-nose installation. For operation at Mach numbers below 0.8, a static-pressure tube ahead of the vertical tail fin or fuselage vents, properly located and installed, should prove satisfactory.

Various methods of calibrating static-pressure installations in flight are briefly discussed.

No. 40

TITLE: Summary of Methods of Measuring Angle of Attack on Aircraft

AUTHOR: William Gracey

PUBLICATION: NACA TN 4351, August 1958

SUMMARY: Wind-tunnel calibrations of three types of angle-of-attack sensing devices - the pivoted vane, the differential pressure tube, and the null-seeking pressure tube - are presented, the pivoted vane has been used primarily in the flight testing of airplanes and missiles, whereas the null-seeking pressure tube has been used almost exclusively in the service operation of airplanes. The differential pressure tube has not been used to any great extent as a flight instrument.

Flight data on the position errors for three sensor locations - ahead of the fuselage nose, ahead of the wing tip, and on the fore-body of the fuselage - are also presented. For operation throughout the subsonic, transonic, and supersonic speed ranges, a position ahead of the fuselage nose will provide the best installation. If the shape of the fuselage nose is not too blunt, the position error will be essentially zero when the sensor is located 1.5 or more fuselage diameters ahead of the fuselage.

Various methods for calibrating angle-of-attack installations in flight are briefly described.

3. Surface Impact Probe Measurements

No. 41

TITLE: Use of Preston Tubes for Measuring Hypersonic Turbulent Skin Friction

AUTHOR: Earl R. Keener and Edward J. Hopkins

PUBLICATION: NASA TN D-5544, November 1969, N69 10695

SUMMARY: A brief review is made of supersonic Preston tube correlations which account for the effects of Mach number and Reynolds number for adiabatic surfaces. Experimental results are presented which show that two of these correlations are applicable to nonadiabatic surfaces in an airstream at a Mach number of 7. Direct measurements with a skin-friction balance were used for the calibrations. The Reynolds number ranged from 4 to 110 million, and the ratio of wall-to-adiabatic-wall temperature ranged from 0.3 to 0.5. The major conclusion is that the correlation can be simplified and made independent of the boundary-layer-edge condition by basing the correlation factors on only three measurements: wall temperature, wall pressure, and Preston tube pressure.

No. 42

TITLE: An Experimental Investigation of the Surface Pitot Probe Including Effects of Heat Transfer and Compressibility

AUTHOR: D. L. Brott, W. J. Yanta and R. E. Lee

PUBLICATION: Report NOLTR 68-151, Naval Ordnance Laboratory, October 1968

SUMMARY: Preston and Stanton probes have been investigated experimentally in both subsonic and supersonic flow. The Preston probe was tested in subsonic flow at adiabatic wall temperatures from Mach 0.1 to 0.5. The correlation between probe impact pressure and shear stress obtained directly by a skin-friction balance was independent of Mach number and

Reynolds number over the range investigated. The present results are in better agreement with the flat-plate results of Smith and Walker, than the pipe-flow results of Preston or Patel.

Both Preston probes and Stanton probes were investigated at a nominal Mach number of 4.8 at ratios of wall temperature to adiabatic wall temperature from 0.74 to 0.48. The correlation between probe impact pressure and shear stress was independent of Reynolds number over the range investigated, and was displaced from the incompressible results. A small effect due to moderate heat-transfer rates was observed in the correlation of probe impact pressure with shear stress. The Stanton probe results show more scatter than the Preston probe results, and appear to be independent of heat transfer and Reynolds number over the range investigated.

No. 43

TITLE: Measurement of Skin Friction at Low Subsonic Speeds by the Razor-Blade Technique

AUTHOR: L. F. East

PUBLICATION: R. & M. No. 3525, Aeronautical Research Council, August 1966, N68 24792

SUMMARY: The technique consists of forming a surface pitot-tube by placing a small segment of razor blade on the surface with its tapered cutting edge above a static-pressure hole. The effects of limited changes in the razor-blade geometry on the measured pressure have been determined in a two-dimensional turbulent boundary layer and a calibration curve for a particular standardized geometry deduced. The variation of pressure due to yawing the blade segments in three-dimensional boundary layers, similar to those likely to occur on aerodynamic models, is found to be independent of the nature of the boundary layer. From this a method of using razor-blade segments to measure skin friction in three-dimensional boundary layers is proposed, which does not necessitate prior knowledge of the surface-flow direction.

TITLE: Study of Surface Pitots for Measuring Turbulent Skin Friction at Supersonic Mach Numbers - Adiabatic Wall

AUTHOR: Edward J. Hopkins and Earl R. Keener

PUBLICATION: NASA TN D-3478, July 1966, N66 29215

SUMMARY: Two types of surface pitot tubes were investigated for measuring local turbulent skin friction at supersonic speeds. These tubes consisted of a circular tube, called a Preston tube, and a two-dimensional blade over a static orifice, hereinafter called a Stanton tube. They were calibrated in air against the measured skin friction throughout a Mach number range from 2.4 to 3.4 and a Reynolds number range from about 16 to 100 million. The boundary layer in which the surface tubes were mounted varied in thickness from about 5 to 7 inches, and the wall temperature was close to adiabatic. The surface pitot data are presented on the basis of several calibration factors that have been used in the past and on the basis of a newly developed calibration factor.

The new calibration factor collapsed the supersonic Preston tube results onto a single calibration curve, which is invariant with changes in Reynolds number, Mach number, or tube size within wide limits. This curve is the same as Preston's incompressible curve.

It was found that the calibration curve for the Stanton tube was affected by the streamwise position of the tube leading edge relative to the static orifice over which it was mounted.

In addition to the surface tube calibrations, some existing local skin-friction results, measured for turbulent boundary layers on adiabatic flat plates, are compared with theoretical values on the basis of equal momentum thickness Reynolds numbers throughout a Mach number range from 0.06 to 6.7. In general, the theory of Wilson and the T' method of Sommer and Short bracketed all the experimental skin-friction data of both past and present investigations.

No. 45

TITLE: Calibration of the Preston Tube and Limitations on its Use in Pressure Gradients

AUTHOR: V. C. Patel

PUBLICATION: Journal of Fluid Mechanics, vol. 23, part 1, 1965, A65 35455

SUMMARY: Preston's method of measuring skin friction in the turbulent boundary layer makes use of a circular pitot tube resting on the wall. On the assumption of a velocity distribution in the wall region common to boundary layer and pipe flows the calibration curve for the pitot tube can be obtained in fully developed pipe flow. Earlier experiments suggested that Preston's original calibration was in error, and a revised calibration curve has been obtained and is presented here.

From experiments in strong favorable and adverse pressure gradients, limits are assigned to the pressure-gradient conditions within which the calibration can be used with prescribed accuracy. It is shown that in sufficiently strong favorable gradients the "inner-law" velocity distribution breaks down completely, and it is suggested that this breakdown is associated with reversion to laminar flow.

As an incidental result, values have been obtained for the constants occurring in the logarithmic expression for the inner-law velocity distribution.

No. 46

TITLE: Use of a Stanton Tube for Skin-Friction Measurements

AUTHOR: S. S. Abarbanel, R. J. Hakkinen and L. Trilling

PUBLICATION: NASA Memorandum 2-17-59W, March 1959

SUMMARY: A small total-pressure tube resting against a flat-plate surface was used as a Stanton tube and calibrated as a skin-friction meter at various subsonic and supersonic speeds. Laminar flow was maintained for the supersonic runs at a Mach number $M_\infty = 1.33$ and $M_\infty = 1.87$, the calibrations were carried out in a turbulent boundary layer. The subsonic flows were found to be in transition.

The skin-friction readings of a floating element type of balance served as the reference values against which the Stanton tube was calibrated.

A theoretical model was developed which, for moderate values of the shear parameter τ , accurately predicts the performance of the

Stanton tube in subsonic and supersonic flows. A "shear correction factor" was found to explain the deviations from the basic model when τ became too large. Compressibility effects were important only in the case of turbulent supersonic flows, and they did not alter the form of the calibration curve.

The test Reynolds numbers, based on the distance from the leading edge and free-stream conditions, ranged from 70,000 to 875,000. The turbulent-boundary-layer Reynolds numbers, based on momentum thickness, varied between 650 and 2,300. Both laminar and turbulent velocity profiles were taken and the effect of pressure gradient on the calibration was investigated.

No. 47

TITLE: The Determination of Turbulent Skin Friction by Means of Pitot Tubes

AUTHOR: J. H. Preston

PUBLICATION: Aeronautical Research Council Report F.M. 1883, March 1953

SUMMARY: A simple method of determining local turbulent skin friction has been developed which utilizes a round pitot tube resting on the surface. Assuming the existence of a region near the surface in which conditions are functions only of the skin friction, the relevant physical constants of the fluid and a suitable length, a universal non-dimensional relation is obtained for the difference between the total pressure recorded by the tube and the static pressure at the wall, in terms of the skin friction. This relation, on this assumption, is independent of the pressure gradient and surface condition. The truth and form of the relation were first established, to a considerable degree of accuracy, in a pipe using four geometrically similar round pitot tubes - the diameter being taken as representative length. These four pitot tubes were then used to determine the local skin friction coefficient at three stations on a wind tunnel wall, under varying conditions of pressure gradient and surface condition. At each station, within the limits of experimental accuracy, the deduced skin friction coefficient was found to be the same for each pitot tube, thus confirming the basic assumption under a wide range of conditions, and leaving little doubt as to the correctness of the skin friction so found.

Pitot traverses were then carried out in the pipe and in the boundary layer on the wind tunnel wall. The results were plotted in two non-dimensional forms on the basis suggested above and they fell close together in a region whose outer limit represented the breakdown of the basic assumption, but close to the wall the results spread out due to the unknown displacement of the effective centre of a pitot tube near a wall. This again provides further evidence of the existence of a region of local dynamical similarity

and of the correctness of the skin friction deduced from measurements with round pitot tubes on the wind tunnel wall.

The extent of the region in which the local dynamical similarity may be expected to hold appears to vary from about $1/5$ to $1/20$ of the boundary-layer thickness for conditions remote from, and close to, separation respectively.

4. Skin Friction Measurements

No. 48

TITLE: Direct Measurements of Turbulent Skin Friction on a Nonadiabatic Flat Plate at Mach Number 6.5 and Comparisons With Eight Theories

AUTHOR: Edward J. Hopkins, Earl R. Keener and Pearl T. Louie

PUBLICATION: NASA TN D-5675, February 1970, N70 19767

SUMMARY: The local turbulent skin friction was directly measured with a floating-element balance mounted flush in a sharp-edged flat plate. The Mach number was 6.5, the length Reynolds number ranged from about 0.7 to 14 million, and the ratio of wall-to-adiabatic-wall temperature ranged from 0.3 to 0.5 . The skin-friction results were used to evaluate eight theories on a generalized basis by determining which theory gives the best transformation of the data onto an incompressible skin-friction curve. This evaluation indicates that the best predictions of skin friction (within about 5 percent) are given by the Van Driest II or Coles theories when momentum-thickness Reynolds number is used. The uncertainty introduced by theories that require estimates of a virtual origin of turbulent flow is demonstrated.

No. 49

TITLE: Flight Demonstration of a Skin-Friction Gage to a Local Mach Number of 4.9

AUTHOR: Darwin J. Garringer and Edwin J. Saltzman

PUBLICATION: NASA TN D-3830, February 1967, N67 17173

SUMMARY: A small, commercially available skin-friction gage was flight tested on the X-15 airplane. The Reynolds number range investigated extended from 3.3×10^6 to 8.0×10^6 , and local Mach numbers ranged

from 0.7 to 4.9. The ratio of wall-to-recovery temperature varied from about 0.4 to 1.4.

The gage, its cooling system, and the supporting instrumentation performed well. Turbulent skin-friction values measured in flight for a wide range of wall-to-recovery temperature ratios are similar in level to adiabatic flat-plate wind-tunnel results for corresponding Mach numbers and Reynolds numbers. Thus, for the present tests the influence of wall-to-recovery temperature ratio appears to be less than estimated by turbulent theory.

No. 50

TITLE: Free-Flight Measurement of Local Turbulent Skin Friction on an RM-10 Vehicle

AUTHOR: J. C. Westkaemper, D. R. Moore and C. E. Murphey, Jr.

PUBLICATION: Report DRL-461, Defense Research Laboratory, University of Texas, January 1961, AD 652 801

SUMMARY: In-flight skin-friction measurements were made on an RM-10 rocket test missile in a cooperative effort by Defence Research Laboratory and the National Aeronautics and Space Administration. A floating-element, direct-measuring skin friction balance and a surface impact probe were used. The balance performance was questionable because of errors induced by the shock of booster and sustainer rocket ignition. The impact probe method gave satisfactory results including flight times where aerodynamic heating occurred, but the results were at times influenced by angle-of-attack effects. The balance and probe data were found to agree satisfactorily with the Van Driest theory for skin friction on a flat plate with heat transfer except when the missile was at angle of attack, for which times the theory is not applicable. Recommendations for instrumentation refinements are included.

No. 51

TITLE: Skin Friction Measurements in Incompressible Flow

AUTHOR: Donald W. Smith and John H. Walker

PUBLICATION: NASA TR R-26, 1959

SUMMARY: Experiments have been conducted to measure in incompressible flow the local surface-shear stress and the average skin-friction coefficient for a turbulent boundary layer on a smooth, flat plate having zero pressure gradient. The local surface-shear stress was measured by a floating-element skin-friction balance and also by a calibrated total

head tube located on the surface of the test wall. The average skin-friction coefficient was obtained from boundary-layer velocity profiles. The boundary-layer profiles were also used to determine the location of the virtual origin of the turbulent boundary layer. Data were obtained for a range of Reynolds numbers from 1 million to about 45 million with an attendant change in Mach number from 0.11 to 0.32.

The measured local skin-friction coefficients obtained with the floating-element balance agree well with those of Schultz-Grunow and Kempf for Reynolds numbers up to 45 million. The measured average skin-friction coefficient agree with those given by the Schoenherr curve in the ranges of Reynolds numbers from 1 to 3 million and 30 to 45 million. In the range of Reynolds numbers from 3 to 30 million the measured values are less than those predicted by the Schoenherr curve.

The measurement of local surface shear by a calibrated Preston tube appears to be accurate and inexpensive. The calibration as given by Preston must be modified slightly, however, to yield the results obtained from the floating-element skin-friction balance.

No. 52

TITLE: Direct Measurements of Skin Friction

AUTHOR: Satish Dhawan

PUBLICATION: NACA TN 2567, January 1952

SUMMARY: A device has been developed to measure local skin friction on a flat plate by measuring the force exerted upon a very small movable part of the surface of a flat plate. These forces, which range from about 1 milligram to about 100 milligrams, are measured by means of a reluctance measuring device. The apparatus was first applied to measurements in the low-speed range, both for laminar and turbulent boundary layers. The measured skin-friction coefficients show excellent agreement with Blasius' and Von Karman's results. The device was then applied to high-speed subsonic flow and the turbulent-skin-friction coefficients were determined up to a Mach number of about 0.8. A few measurements in supersonic flow were also made.

The paper describes the design and construction of the device and the results of the measurements.

5. Measurement of Unsteady and Turbulent Pressure Fluctuations

No. 53

TITLE: Effects of Noninstantaneous Transducer Response on the Measurement of Turbulent Pressure

AUTHOR: Wayne A. Strawderman

PUBLICATION: Acoustical Society of America Journal, vol. 46, November 1969

SUMMARY: The temporal response characteristics of a transducer limits its ability to resolve the temporal (or frequency) characteristics of the turbulent pressure field, just as the finite size of a transducer limits the ability of a transducer to resolve the spatial details of the turbulent pressure field. The spatial effects have been treated previously under the assumption that the transducer responds instantaneously in time. This analysis treats mathematically the ability of real transducers (i.e., transducers of finite size and noninstantaneous time response) to resolve the pressure statistics of the turbulent field. It is assumed that the spatial and temporal response characteristics of the transducers are known in the form of the impulse response kernel. Mathematical expressions relate the measured statistics of the turbulent pressure field to the actual statistics in terms of the impulse response kernel. It is shown that, where the impulse response kernel is separable in time and space, no temporal distortion of the turbulent pressure statistics occurs.

No. 54

TITLE: Measurements on the Effect of Transducer Size on the Resolution of Boundary-Layer Pressure Fluctuations

AUTHOR: F. E. Geib, Jr.

PUBLICATION: Acoustical Society of America Journal, vol. 46, July 1969

SUMMARY: The response of a flush-mounted transducer to the pressure field in a turbulent boundary layer is known to depend on the spatial and temporal characteristics of the transducer. This paper presents an experimental study of this dependence. The reduced data are presented in a manner similar to that used by Corcos to present his estimation of the response of transducers to a corresponding pressure field.

TITLE: On the Response of Pressure Measuring Instrumentation in Unsteady Flow

AUTHOR: Thomas E. Siddon

PUBLICATION: UTIAS Report No. 136, University of Toronto, January 1969,
AD 682 296

SUMMARY: If a fast-responding pressure probe of classical "static" probe geometry is placed in an unsteady flow it will not register the true instantaneous pressure (i.e., that which would have occurred in the absence of the probe). Interaction between the probe and the unsteady velocity field gives rise to an error between the measured pressure $P_m(t)$ and the true pressure $P_t(t)$. This error is not always small; it can in some circumstances be larger than the difference between $P_t(t)$ and the ambient pressure.

A major objective of the present work has been to explore the possibility of correcting $P_m(t)$ instantaneously, through use of an error-compensating probe. As preliminary steps the fundamental mechanisms of interaction error were examined and a general empirical error function was postulated. A probe configuration was adopted for which the error reduces to the simple form $B\rho V_n^2(t)$; $V_n(t)$ is the instantaneous resultant of the orthogonal velocity components $V(t)$ and $W(t)$ normal to the probe axis; B is a coefficient which typically takes values ranging from $-1/4$ to $-1/2$. A probe was developed to measure simultaneously the unsteady components of $P_m(t)$ and $V_n(t)$ over a wide frequency range. Output signals from the probe were processed by analogue means to correct out the error term $B\rho V_n^2(t)$, providing an improved estimate of the true unsteady pressure.

A series of experiments were conducted in a number of contrived unsteady flows. Major experiments involved a periodic "rotating inclined nozzle flow" and some typical turbulent flows. The prime objectives were to substantiate the assumed form of the error function, to evaluate the error coefficient B , and to attempt the measurement of the correction unsteady pressure by the error-compensating scheme. These goals were realized with reasonable success for a variety of circumstances. The error-compensation scheme was found to be effective, particularly for the specialized rotating inclined nozzle flow. However, the measurements in turbulent flows revealed that the correction to root-mean-square pressure fluctuation level was small, generally amounting to less than 20%.

No. 56

TITLE: Response of Pneumatic Pressure-Measurement Systems to a Step Input in the Free Molecular, Transition, and Continuum Flow Regimes

AUTHOR: Jesse Hord

PUBLICATION: ISA Transactions, vol. 6, no. 3, 1967, A67 42293

SUMMARY: A simplified analysis of the response of pressure-sensing devices and their interconnecting tubing is presented. The results are conveniently presented in the form of time equations, with a fraction of the applied step (usually 63.2%) attained at the sensor as the variable. A single algebraic equation adequately predicts (within $\pm 20\%$) the response of pressure-measuring systems in the free molecule, transition, and continuum flow regimes. Two very simple algebraic equations may be used to predict time response in the free molecule and continuum flow regimes if a maximum error of 2:1 is tolerated in the mid-transition range. The criterion for extending these formulas into the transition flow range is given. The formulas are limited where viscous flow occurs, and they are not restricted for free molecule flow. All of the formulas are suitable for field engineering applications. The literature is synthesized by comparing the results from selected references with those of the analysis. Factors influencing transient pressure measurements are reviewed.

No. 57

TITLE: Effect of Transducer Size, Shape, and Surface Sensitivity on the Measurement of Boundary-Layer Pressures

AUTHOR: Pritchard H. White

PUBLICATION: Acoustical Society of America Journal, vol. 41, May 1967, A67 34964

SUMMARY: The measurement of a random pressure field by a transducer of finite size is subject to error because the transducer senses the spatial average pressure over its face rather than a point value. Methods of correction for this effect under certain conditions have been developed previously; however, many important parameters of the system have been neglected. The present study shows the effect of various boundary-layer parameters, as well as the shape of the transducer, upon the measurement error. The effect of nonuniform surface sensitivity of the transducer is also considered. This factor brings about an effective smaller transducer size, but also can lead to serious errors in cross-correlation analysis.

No. 58

TITLE: Effect of Velocity and Temperature Fluctuation on Pitot Probe Measurements in Compressible Flow

AUTHOR: James E. Danberg

PUBLICATION: Report NOLTR 61-28, Naval Ordnance Laboratory, March 1962,
AD 275 914

SUMMARY: The effect of velocity and temperature fluctuations on the pressure indicated by a pitot probe has been analyzed for the compressible case assuming negligible static pressure fluctuations. This analysis is based on the assumption that the Mach number fluctuation in the free stream ahead of the probe affect the pitot pressure directly. As a result, the measured pitot pressure divided by the static pressure is not just a function of the average Mach number and the ratio of specific heats as it is in steady flow. In order to correctly interpret pitot pressure measurements in a turbulent boundary layer it is necessary to separate from the measurements the effects of the Mach number fluctuations.

The results of the analysis show that the velocity fluctuations directly and also via the temperature fluctuations, cause the indicated pitot pressure to be greater than the pitot pressure associated with the time average velocity and temperature. Velocity fluctuations cause an increase in the pitot pressure in proportion to the mean of the square of the fluctuation as is already known for incompressible flow. The role of the temperature fluctuations is to increase the effect of the velocity fluctuations on the measured pressure. Heat transfer into the wall, acting through the temperature fluctuations, has a reverse effect.

No. 59

TITLE: Lag in Pressure Systems at Extremely Low Pressures

AUTHOR: William T. Davis

PUBLICATION: NACA TN 4334, September 1958

SUMMARY: A theoretical formula for determining time lags in pressure-measuring systems at all pressures, including extremely low pressures where molecular flow occurs, is derived and shown to be accurate to within 10 percent for pressures down to approximately 0.2 millimeter of mercury (0.556 lb/sq ft) for nearly linear pressure changes.

No. 60

TITLE: Averaging of Periodic Pressure Pulsations by a Total-Pressure Probe

AUTHOR: R. C. Johnson

PUBLICATION: NACA TN 3568, October 1955

SUMMARY: Information is presented on the average pressure indicated by a total-pressure probe subjected to a stagnation pressure that alternates periodically between two constant values.

Calculated and experimental data are in good agreement, and errors are reduced when the probe design is such as to ensure laminar-flow pulsations in the probe at all times. The averaging error is minimized when the inside diameter of the probe entrance tube is made as small as possible, and its length as great as possible, consistent with an acceptable time lag.

No. 61

TITLE: A Method for the Determination of the Time Lag in Pressure Measuring Systems Incorporating Capillaries

AUTHOR: Archibald R. Sinclair and W. Warner Robins

PUBLICATION: NACA TN 2793, September 1952

SUMMARY: A method is presented for the determination of the time lag in pressure measuring systems incorporating capillaries; this method is a convenient and systematic means of selecting, designing, or redesigning a pressure measuring system to meet the time requirements of a particular installation. Experimental data are shown and a step-by-step sample application is presented.

Calculated and experimental data are in reasonable agreement and show that response time in a pressure measuring system incorporating capillaries is a function of the orifice pressure, the initial pressure differential, and the system volume, is directly proportional to capillary length and to the viscosity of the gas in the capillary, and is inversely proportional to the fourth power of the capillary inside diameter.

6. Noise Measurements

No. 62

TITLE: Study of Methods for Performing Inflight Measurements of Acoustic Noise

AUTHOR: Frank F. Borriello

PUBLICATION: Report No. NAMC-ASL(12)-R360FR101, Aeronautical Structures Laboratory, Naval Air Material Center, October 1961, AD 266 067

SUMMARY: This study was made to determine the feasibility and methods of measuring the boundary layer, acoustic environment during flight. A list is first presented of the various organizations that have performed work on studies and measurements of acoustic noise for either engine noise or boundary-layer noise. Descriptions are then included of the acoustic environment and of equipment and method for measuring this environment. Important parameters to be measured are stated, and the need for more supersonic data is emphasized.

The results of this study indicate that the measurement of boundary-layer noise is feasible, at least up to sonic speeds. Also, it is more fruitful to run a comprehensive flight-test program with one well-instrumented airplane than to partially instrument a number of service airplanes and perform statistical surveys.

No. 63

TITLE: Flight Measurement of Wall-Pressure Fluctuations and Boundary-Layer Turbulence

AUTHOR: Harold R. Mull and Joseph S. Algranti

PUBLICATION: NASA TN D-280, October 1960

SUMMARY: The results are presented for a flight test program using a fighter type jet aircraft flying at pressure altitudes of 10,000, 20,000, and 30,000 feet at Mach numbers from 0.3 to 0.8. Specially designed apparatus was used to measure and record the output of microphones and hot-wire anemometers mounted on the forward fuselage section and wing of the airplane. Mean-velocity profiles in the boundary layers were obtained from total-pressure measurements.

The ratio of the root-mean-square fluctuating wall pressure to the free-stream dynamic pressure is presented as a function of Reynolds number and Mach number. The longitudinal component of the turbulent-velocity fluctuation was measured, and the turbulence-intensity profiles are presented for the wing and forward-fuselage section.

In general, the results are in agreement with wind-tunnel measurements which have been reported in the literature. For example, the variation of $\sqrt{p^2}/q$ ($\sqrt{p^2}$ is the root mean square of the wall-pressure fluctuation, and q is the free-stream dynamic pressure) with Reynolds number was found to be essentially constant for the forward-fuselage-section boundary layer, while variations at the wing station were probably unduly affected by the microphone diameter (5/8 in.), which was large compared with the boundary-layer thickness.

7. General References on Measurements With Heat Flux Devices

No. 64

TITLE: Thermal Methods of Flow Measurement

AUTHOR: P. Bradshaw

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments, vol. 1, May 1968, A68 29567

SUMMARY: This article is a review of the principles of velocity and mass-flow measurement by means of heat transfer, and especially of the practical difficulties that arise in the application of those principles: it is not intended to be an assessment of the different types of commercial apparatus currently available. The bibliography includes references to literature surveys and to recent research papers as well as the key papers describing the basic principles.

No. 65

TITLE: Advances in Hot-Wire Anemometry

AUTHOR: W. L. Melnik and J. R. Weske, ed.

PUBLICATION: Proceedings of the International Symposium on Hot-Wire Anemometry, University of Maryland, March 20-21, 1967, AD 676 019

SUMMARY: This volume contains the text of the lectures presented at the International Symposium on Hot-Wire Anemometry and comments and discussions that were submitted thereafter in writing. Topics include historical development, signal interpretation, turbulence measurements, recent development and circuit design.

No. 66

TITLE: A Bibliography on Hot-Wire Anemometry

AUTHOR: T. E. Brock and C. J. Moon

PUBLICATION: Report BIB-18, British Hydromechanics Research Association,
October 1965, N66 20985

SUMMARY: A chronological presentation is given to a bibliography on hot-wire anemometry. Abstracts are included for many of the 175 references listed, and both subject and author indices are appended.

8. Hot-Wire Measurement Techniques

No. 67

TITLE: Spatial Resolution of the Vorticity Meter and Other Hot-Wire Arrays

AUTHOR: J. C. Wyngaard

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments,
vol. 2, November 1969, A70 13760

SUMMARY: Analyses of the spatial resolution of the Kovasznay vorticity meter and of arrays for measuring velocity derivatives in isotropic turbulence are presented. Pao's three-dimensional spectrum is used, allowing calculations of the measured values of the one-dimensional vorticity spectrum and mean-square velocity derivatives to be compared with their predicted values. It is found that for accurate measurement the array size should be of the order of the Kolmogorov microscale of the turbulence field.

No. 68

TITLE: Hot-Wire Anemometer Measurements in Flows Where Direction of Mean Velocity Changes During a Traverse

AUTHOR: Keith J. Bullock and Klaus Bremhurst

PUBLICATION: IEEE Transactions on Instrumentation and Measurement, vol. IM-18,
no. 3, September 1969, A70 19710

SUMMARY: Using the relation for the effective cooling velocity C of a hot wire inclined to the mean flow:

$$C = U(\cos^2\alpha + k^2\sin^2\alpha)$$

where α is the angle between the normal to the wire and the mean flow U , relations are derived for the fluctuating terms:

$u, v, \sqrt{u^2/U}, \sqrt{v^2/U}$ and $\overline{uv}/U^2$.

A relation is also derived for a change in the mean flow direction angle. Interference from hot-wire supports was found to limit resolvable mean flow angles to $\pm 30-35^\circ$. This can be extended to $\pm 55^\circ$ by using X probes with smaller values of α .

No. 69

TITLE: Performance of Normal and Yawed Hot Wires

AUTHOR: H. H. Bruun

PUBLICATION: Report ISVR-TR-21, Southampton University, July 1969, N69 39171

SUMMARY: Literature relevant to the measurement of turbulence properties in gas flow by hot-wire anemometers is discussed, and the linearized constant temperature system in use for channel measurements at B.N.L. is described. Special attention is given to possible sources of error in the measurement of Reynolds shear stress and the theoretical estimates of error are compared with results from a smooth pipe. It is concluded that at "low" Reynolds numbers, the present system is capable of predicting shear stress to within 3%.

No. 70

TITLE: The Effects of Flow Inclination on Hot-Film Anemometer Probes

AUTHOR: Robert J. Vidal

PUBLICATION: Report CAL No. VH-2556-A-1, Cornell Aeronautical Laboratory, May 1969, N69 27595

SUMMARY: An existing theoretical description of the hot-film anemometer is reviewed to establish a quantitative basis for examining the effects of flow inclination on the velocity inferred with a typical anemometer configuration. The inclination effects are calculated with this theory and an existing solution for the boundary layer on a yawed cylinder. It is shown that this probe configuration is sensitive to small angles of inclination and a 5° misalignment could produce errors of 30% to 45% in the inferred velocity.

No. 71

TITLE: The Measurement of Reynolds Stress Tensor With a Single Hot-Wire

AUTHOR: Cahit Ciray

PUBLICATION: Middle East Technical University Journal of Pure and Applied Sciences, vol. 2, April 1969

SUMMARY: A method to measure simultaneously velocity and stress components of a turbulent flow using a single hot-wire is presented. Two sets of equations are derived, each of which can be used for this purpose. The formulae for the determination of various constants in these equations are given. The resulting equations form a system of twelve non-linear algebraic equations, from which the unknowns can be found. A method of solution of the system of equations is devised and the reference which contains the associated flow diagram of the computer program is given.

The experimental results are compared with already existing similar data. Possible improvements for the accuracy of the results are suggested.

No. 72

TITLE: Constant Current Anemometer Diagnostics of Flow Fields

AUTHOR: Bernard C. Weinberg and Samuel Lederman

PUBLICATION: PIBAL Report No. 69-13, Polytechnic Institute of Brooklyn, April 1969, N69 35004

SUMMARY: An experimental investigation was undertaken to determine some of the basic characteristics of the constant current hot-wire anemometer. Among these were included effects of size, resistance ratio, dirt accumulation, repeatability, and applicability to certain flow fields. Tests were conducted on a turbulent pipe flow and a three-dimensional turbulent wall jet. Results characterizing these flows were also obtained.

No. 73

TITLE: Turbulence Measurement With Hot Wires at BNL

AUTHOR: C. J. Lawn

PUBLICATION: Report RD/B/M-1277, Berkeley Nuclear Laboratory, England, April 1969, N69 36431

SUMMARY: This report describes the calibration laws for the "standard" 2mm ISVR hot-wire probes placed normal and yawed to the mean flow direction. By using a copper plating procedure the variation in geometry of these hot wires is kept small. The measurements described have shown that these small changes in geometry between different 2mm hot wires will only effect the shape of the calibration laws at low velocity. It has therefore been possible to express the calibration laws in terms of universal functions for velocities greater than 1 m/s.

Using these universal calibration laws a method is given for the determination of the Reynolds stress $u'w'$ and the time mean square values u'^2 and w'^2 in flows with low turbulence intensity.

The reliability of this method is finally shown by comparing hot-wire measurements of the Reynolds stress in a fully developed turbulent pipe flow with the Reynolds stress calculated from the mean pressure drop along the length of the pipe.

No. 74

TITLE: Measurement of Small-Scale Turbulence Structure with Hot Wires

AUTHOR: J. C. Wyngaard

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments, vol. 1, November 1968

SUMMARY: Expressions are derived for turbulence spectra measured with a single hot wire and wires in an X-array, and are applied to the problem of measurement of spectra at small scales. Isotropic turbulence and Pao's form for the three-dimensional spectrum are assumed. Calculated curves for one-dimensional spectra measured with a single wire and an X-array are presented. Experimental data in good agreement with the single wire calculations are shown. The calculations show that both one-dimensional spectra measured with the X-array are contaminated by crosstalk from the other component. The effect is more serious for the longitudinal spectrum and depends on array geometry and the Kolmogorov microscale.

TITLE: Turbulence Measurements With Inclined Hot Wires
Part 1. Heat Transfer Experiments With Inclined Hot Wire

AUTHOR: F. H. Champagne, C. A. Sleicher and O. H. Wehrmann

PUBLICATION: Journal of Fluid Mechanics, vol. 28, part 1, 1967, A67 26913

SUMMARY: The measurement of the turbulent shear stresses and normal and bi-normal intensities with a hot-wire anemometer requires that the directional sensitivity of the hot-wire be known. Normal component or cosine law cooling is generally assumed, although for finite wire lengths the non-uniform wire temperature must cause a deviation from the cosine law.

Careful heat transfer measurements from wires inclined and normal to the flow were taken for several values of the Reynolds number, and the length-to-diameter ratio of the wire, the overheat ratio and for several support configurations. All experiments were performed in air at low subsonic velocities, i.e., $M < 0.1$. The measurements indicate that the heat loss from an inclined wire is larger than that from a wire normal to the flow with the same normal component of velocity. The data were correlated by

$$U_E^2(\alpha) = U^2(0)(\cos^2\alpha + \kappa^2\sin^2\alpha),$$

where $U_E(\alpha)$ is the effective cooling velocity at the angle α between the normal to the wire and the mean flow direction and $U(0)$ is the velocity at $\alpha = 0$. The value of κ was found to depend primarily upon the length-to-diameter ratio (ℓ/d) of the wire. For platinum wires κ is approximately 0.20 for $\ell/d = 200$, decreases with increasing $\ell/d = 600$.

To aid in interpreting the heat transfer data, measurements of the temperature distribution along inclined and normal wires were made with a high sensitivity infra-red detector coupled to a high resolution microscope with reflective optics. The measurements indicate that inclined wires and normal wires have nearly identical end conduction losses, although the temperature distribution on an inclined wire is slightly asymmetrical. Therefore, the deviation from the cosine law is caused by an increase in the convection heat loss, and this increase is attributed to the tangential component of velocity.

No. 76

TITLE: Turbulence Measurements With Inclined Hot-Wires
Part 2: Hot-Wire Response Equations

AUTHOR: F. H. Champagne and C. A. Sleicher

PUBLICATION: Journal of Fluid Mechanics, vol. 28, part 1, 1967, A67 26914

SUMMARY: Hot-wire response equations to include the effects of the tangential velocity component as well as the non-linearities caused by high intensity turbulence are derived for linearized constant temperature operation. For low intensity turbulence similar equations are derived for constant current operation. The equations are applied to an X-wire array to determine the errors in selected turbulence quantities which arise from the assumption of cosine law cooling. The error depends upon the quantity measured, the method of operation, and l/d . For $l/d = 200$ the error ranges from 0 to 17%.

No. 77

TITLE: Instrument Development for the Base Flow Region of a Multinozzle Booster Vehicle

AUTHOR: A. B. Bauer

PUBLICATION: Publication No. U-3663, Aeronutronic Division, Philco-Ford,
June 1966

SUMMARY: Two hot-wire probes are used to measure flow velocity directly by emitting heat pulses from one and sensing the convection time with the other. Flow speeds over 2000 fps can be measured. Flow direction in the plane normal to the wires is determined by moving the sensing wire to locate the wake of the pulse wire.

No. 78

TITLE: Hot-Wire Techniques in Low Density Flows With High Turbulence Levels

AUTHOR: A. R. Hanson, R. E. Larson and F. R. Krause

PUBLICATION: Litton Systems, Aero-Astrodynamic Research Review, No. 2, April 1965, N66 35554

SUMMARY: Static and dynamic calibrations of modern hot-wire systems, which might be used in low density flows with high turbulence levels, are summarized. A review of hot-wire heat loss equations indicated that a time invariant frequency response can be obtained at high turbulence levels only if the probes are operated by constant temperature. The

static calibration of two constant temperature hot-wire systems combined with hot-wire and hot-film sensors showed significant changes in the calibration constants at low densities, a large increase of wall proximity effects at low densities, and considerable changes in the slope of the wire-resistance temperature relation especially at low temperatures. Dynamic calibration indicated that even the most advanced constant temperature hot-wire systems are unable to resolve relatively high turbulent fluctuations in the reattachment region of free shear layers. Further improvement, possible by redesigning both the circuitry and the probes, are suggested.

No. 79

TITLE: Hot-Wire Heat-Loss Characteristics and Anemometry in Subsonic Continuum and Slip Flow

AUTHOR: Frederick W. Boltz

PUBLICATION: NASA TN D-773, February 1961

SUMMARY: An investigation was conducted to determine the heat-loss characteristics of hot wires in subsonic flow and, in particular, to derive a general subsonic heat-loss expression. Heat-loss data were obtained for tungsten wires having nominal diameters of 0.00015, 0.0003, and 0.001 inch at overheat ratios of 0.1, 0.3, 0.6, and 0.9. The Mach number was varied from about 0.1 to 0.98 for low-density operation and from about 0.04 to 0.3 at the highest density. The stagnation pressure was varied from about 1/5 of an atmosphere to 5 atmospheres. With these wire sizes and test conditions heat-loss data were obtained at Reynolds numbers from 1.2 to 146 under conditions varying from essentially continuum flow to those involving a relatively large amount of slip flow.

No. 80

TITLE: Heat-Loss Characteristics of Hot-Wire Anemometers at Various Densities in Transonic and Supersonic Flow

AUTHOR: W. G. Spangenberg

PUBLICATION: NACA TN 3381, May 1955

SUMMARY: An experimental investigation was made of the heat-loss characteristics of heated fine wires suitable for use as anemometers in turbulence research. Speeds ranged from low subsonic to Mach number 1.9. Density and temperature loading were varied over wide limits, and wire diameters ranged from 0.00005 to 0.0015 inch. The effects of each of the several variables on the heat-loss characteristics of both normally oriented and swept wings were measured. The

characteristics were found to depend on Reynolds number, Mach number, temperature loading, and the ratio of wire diameter to the molecular mean free path of the air. Dependence on the last caused marked departures from King's law, even at low speeds. Mach number effects were found to be larger in the subsonic than in the supersonic range. Temperature-loading effects were found to cause large sensitivity differences between constant-temperature and constant-current operation. Possible applications of these instruments, utilizing the displayed characteristics to measure the contributions of each of the various fluctuating quantities contributing to turbulence in compressible flow, are discussed.

No. 81

TITLE: The Constant Temperature Hot-Thermistor Anemometer

AUTHOR: J. L. Lumley

PUBLICATION: ASME Proceedings of Symposium on Measurement in Unsteady Flow, May 21-23, 1962.

SUMMARY: Some general considerations are given which govern the choice of a sensing element in hot-element anemometry, particularly in liquids. Thermistor material is presented as a possible material for such sensing element, and its properties are given. Several of the properties (resistivity, temperature coefficient of resistivity, thermal conductivity) of this material take on unusual values (compared to platinum) and a more careful analysis is made of the effects of these properties on noise level, spatial resolution, frequency response and sensitivity. It is concluded that thermistor material is potentially capable of improvement by at least an order of magnitude in noise level and spacial resolution, but that overcoming conduction effects which produce degeneration in frequency response and sensitivity is a difficult problem. Finally, two probe types utilizing thermistor materials are described.

9. Wall Effects on Hot-Wire Probe Measurements

No. 82

TITLE: The Effect of Proximity of Walls on the Readings of a Hot-Wire Anemometer in Turbulent Boundary Layers

AUTHOR: Ye. U. Repik and V. S. Ponomareva

PUBLICATION: Soviet Research, vol. 2, July 1970, A70 40341

SUMMARY: Corrections are obtained to velocity distributions measured in the turbulent boundary layer by means of a hot-wire anemometer. It is found that the correction for the proximity of the wall in this case is 1-0.5 of that in laminar boundary layers and that the heat flux from the anemometer wire is not function of the wire diameter.

No. 83

TITLE: Importance of the Orientation of the Support of a Hot-Wire Probe in Relation to a Wall on the Determination of Mean Velocities in a Turbulent Boundary Layer

AUTHOR: Pierre Florent and Gerard Thiolet

PUBLICATION: Academie des Sciences (Paris), Comptes Rendies, Serie A - Sciences Maehematiques, vol. 269, September 1969, A70 10665

SUMMARY: The influence of the orientation and the presence of the hot-wire probe support on the determination of the mean velocity in a flat-plate turbulent boundary layer is demonstrated with the aid of a wall probe. The optimal position in which to place the probe support so as to minimize the measurements error is discussed.

No. 84

TITLE: The Correction of Hot-Wire Readings for Proximity to a Solid Boundary

AUTHOR: J. A. B. Wills

PUBLICATION: Journal of Fluid Mechanics, vol. 12, part 3, 1962, A62 5230

SUMMARY: When using a hot wire for velocity measurements close to a solid boundary, errors may be introduced if the effect of the boundary on the rate of heat loss from the wire is ignored. An experimental determination of the effect is described, in which a hot wire was mounted at various distances from a metal surface forming one wall of a two-dimensional channel. The rate of heat loss was determined electrically, and the air velocity at the wire found from the known

laminar velocity profile. The application of the results to turbulent flows is discussed briefly.

10. Skin Friction Measurement with Flush Film Sensors

No. 85

TITLE: Thin-Film Gauges for the Measurement of Velocity on Skin Friction in Air, Water or Blood

AUTHOR: B. J. Bellhouse and F. H. Bellhouse

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments, vol. 1, December 1968, A69 15118

SUMMARY: Platinum film sensors are constructed by applying platinum paint to a polished Pyrex surface and heating to 640°C. The temperature rise takes place over an interval of one hour and is held for 30 minutes. This process is repeated five times to insure a uniform deposit. Electrical connections are made by soldering copper leads or by applying conductive paint.

No. 86

TITLE: Determination of Mean and Dynamic Skin Friction, Separation and Transition in Low-Speed Flow with a Thin-Film Heated Element

AUTHOR: B. J. Bellhouse and D. L. Schultz

PUBLICATION: Journal of Fluid Mechanics, vol. 24, part 2, 1966, A66 23013

SUMMARY: The application of a heated thin metallic film to the measurement of mean skin friction in laminar and turbulent flow on a flat plate, circular cylinder and in an annular tunnel is described. The manner in which transition and separation are detected with this instrument is illustrated by reference to tests on a circular cylinder. The influence of the ambient air temperature and the gauge temperature on the behavior of the instrument is analyzed and it is shown that a reliable element can be constructed, capable of being moved from one location to another, while retaining its calibration. An element, calibrated on an oscillating flat plate, is used to obtain the spectral density of skin friction in a turbulent boundary layer.

No. 87

TITLE: Shearing Stress Measurements by Use of a Heated Element

AUTHOR: H. W. Liepmann and G. T. Skinner

PUBLICATION: NACA TN 3268, November 1954

SUMMARY: Measurements of heat transfer from small elements embedded in the surface of a solid can be used to obtain local skin-friction coefficients. This report discusses the possible range of application of such an instrument in low- and high-speed flow. Experimental data is presented to show that a hot-wire cemented into a groove in a surface can be used to obtain laminar and turbulent skin-friction coefficients with a single calibration.

11. Anemometer Signal Linearizers

No. 88

TITLE: Temperature Compensated Linearizer for Hot-Wire Anemometer

AUTHOR: Leslie S. G. Kovasznay and Rene Chevray

PUBLICATION: The Review of Scientific Instruments, vol. 40, January 1969, A69 17701

SUMMARY: A description of a "linearizing circuit" for low velocity hot-wire anemometry is given and a detailed analysis of the transistor chain used to generate the desired function is presented. The frequency response of the complete circuit proved very satisfactory, but special consideration had to be given to compensate for the temperature sensitivity inherent to solid state devices.

No. 89

TITLE: Linearizer for Constant Temperature Hot-Wire Anemometer

AUTHOR: Francis H. Champagne and John L. Lundberg

PUBLICATION: The Review of Scientific Instruments, vol. 37, July 1966, A66 35806

SUMMARY: Experiments to determine the heat transfer from hot wires with length-to-diameter ratios on the order of 10^2 , which are commonly used in hot wire anemometry, were performed. The Reynolds numbers used were less than 10, the overheat ratio was kept at 0.80, and the fluid was air. The results indicate that the exponent n in the relationship $E_b^2 = A + BU^n$, where E_b is the bridge voltage and U the fluid velocity, was equal to 0.45 for wires operated at constant temperature. A

linearization circuit which provides an accurate means of recovering the velocity from the above relationships for values of n of 0.45 and 0.50 is described. The main component of the circuit is a temperature compensated, ten segment silicon diode function generator manufactured by Philbrick Researches, Inc. The frequency response of the linearizer is from dc to 36 kc or from dc to 180 kc depending on the operational amplifiers used in the circuit.

12. Anemometer Circuit Analysis

No. 90

TITLE: The Dynamic Response of Constant Resistance Anemometers

AUTHOR: M. R. Davis

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments, vol. 3, January 1970, A70 17620

SUMMARY: The dynamic response of constant resistance anemometer systems is related to the thermal equilibrium of the probe during unsteady changes in surface cooling and also to the characteristics of the feedback system which supplies the heating current. This maintains the probe resistance and average temperature at approximately constant values. In this paper the thermal equilibrium and feedback equations are combined, taking account of the inductance of the probe cables, to derive the governing equation for the unsteady probe current. The probe inductance raises the order of this equation from first to second.

No. 91

TITLE: Nonlinear Control Theory for Constant-Temperature Hot-Wire Anemometers

AUTHOR: Peter Freymuth

PUBLICATION: The Review of Scientific Instruments, vol. 40, February 1969, A69 21421

SUMMARY: The linear control theory for constant-temperature anemometers can be extended to the nonlinear case which applies for large velocity fluctuations. The nonlinear theory establishes the domain within which the linear theory is valid and reveals some nonlinear effects for large velocity fluctuations. Calculations using a second order approximation have been carried out for velocity fluctuations as large as $\pm 64\%$ of the mean velocity. They show in case of a sinusoidal input function the generation of a second harmonic at the output which limits the usefulness of the anemometer at high frequencies. In case of a large velocity step, a longer time is

also required to indicate the step correctly than is required for the
first indication of a small step.

reversed, followed by 15 or 20
analogous measurements and
No. 92

TITLE: Noise in Hot-Wire Anemometers

AUTHOR: Peter Freymuth

PUBLICATION: The Review of Scientific Instruments, vol. 37, April 1968,
A68 27173

SUMMARY: An analysis is given of the electronic noise and the signal-to-noise ratio for constant-temperature and constant-current hot-wire anemometers. The analysis shows that both anemometers are equivalent in their signal-to-noise ratio if the noise sources are equivalent and the hot-wires are operated at the same conditions. A discussion of the theoretical results and a comparison of both anemometer types based on these results are given. In addition, some noise measurements on a constant-temperature anemometer and the method to obtain them are reported and compared with theoretical results. The agreement between measured and theoretical results is rather good.

No. 93

TITLE: Feedback Control Theory for Constant-Temperature Hot-Wire Anemometers

AUTHOR: Peter Freymuth

PUBLICATION: The Review of Scientific Instruments, vol. 38, May 1967,
A67 28750

SUMMARY: A feedback control theory for constant-temperature hot-wire anemometers is developed in the form of a linear differential equation of the third order. This theory allows a more detailed and clearer description of the anemometer's behavior than can be obtained by the frequency response method which has been applied until now to constant-temperature hot-wire anemometers. The theory is presented for the case of an equal arm bridge in the feedback system. Although the theory is restricted to small velocity fluctuations, it is assumed that large fluctuations may be measured correctly within approximately the same frequency range as small ones. The theory is applied to the optimization of the frequency response of constant-temperature hot-wire anemometers by means of a square wave test.

No. 94

TITLE: The Signal-to-Noise Ratios of Constant-Current and Constant-Temperature Hot-Wire Anemometers

AUTHOR: I. Kidron

PUBLICATION: IEEE Transactions on Instrumentation and Measurement, IM-16, March 1967, A67 26121

SUMMARY: Expressions are derived for the signal-to-noise ratios of transistorized constant-current anemometers and constant-temperature anemometers. It is shown that the respective signal-to-noise ratios (in the same bandwidth) are not equal, but do not differ appreciably in a practical case.

The minimum detectable signal (defined on the basis of unity signal-to-noise ratio and expressed in terms of equivalent turbulence intensity) of a constant-temperature anemometer, operated with a typical 5 micron diameter tungsten hot-wire transducer at a mean air flow velocity of 50 m/s, is shown to be 0.3 percent turbulence, in a direct current to 100 kHz frequency range.

No. 95

TITLE: Further Studies of the Constant Temperature Hot-Wire Anemometer

AUTHOR: J. C. Wyngaard and C. M. Sheih

PUBLICATION: Journal of Physics, Part E - Journal of Scientific Instruments, vol. 1, February 1968, A68 22946

SUMMARY: An expression is given for the frequency response of a constant temperature hot-wire anemometer using operational amplifiers with finite open loop gain. Criteria for avoiding overshoot and for the negligibility of the effects of input capacitance are given, and the practical limitation on the frequency response is discussed.

C. BIBLIOGRAPHY

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