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By

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Preliminary Measurements in a Shock Tunnel of Shock Angle and Undersurface Pressure related to a Nonweiler Wing

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SUMMARY

Undersurface pressures and shock angles have been measured at one station on a Nonweiler wing at $M_\infty = 8.8$ and compared with the theory for two-dimensional flow about a wedge. The measurements have been made in a closed jet of a shock tunnel driven by nitrogen or helium, using nitrogen as the test gas.

A miniature high pressure quartz transducer is buried into the wing, and used to measure pressures up to 1 p.s.i.a. The pressure gauge is calibrated statically by gauged gas pressure, and dynamically in a calibration shock tube.

Shock angles and pressures have been measured in the incidence range of $-5.5°$ to $+20°$ for a Nonweiler wing of design incidence 10° and found to agree closely with the solution for the flow about a two-dimensional wedge.

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Replaces N.P.L. Aero. Report 1037 - A.R.C.23,945.
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List of Symbols

P_{res}	pressure of shocked gas brought to rest at the entrance to the hypersonic nozzle
T_{res}	temperature of shocked gas at entrance to nozzle
p_W	free stream pressure
p_L	local pressure on model
C_p	pressure coefficient $= \dfrac{p_L - p_W}{\frac{1}{2}\gamma p_W M_\infty^2}$
γ	ratio of specific heats of nozzle gas
M_∞	free stream Mach number
dB	decibels
ζ	angle of shock wave
α	angle of incidence, (angle of inner ridge line to flow direction)
ρ_W	free stream density
ρ_o	2.37×10^{-3} slugs ft^{-3} (standard atmospheric density)
M_s	incident shock Mach number
mS	milliseconds

2. Introduction

Nonweiler[1] has focussed attention on the use of inverted-V delta wings which when at the design incidence and Mach number, have an oblique attached shock wave lying in the plane of the leading edges of the delta. If the body sweep back angle is such as to give a supersonic leading edge, then the upper and lower surfaces of a Nonweiler wing may be regarded as independent, and for the moment it is relevant to study effects referred to the undersurface.

Such a wing could well be used as an integrated high speed vehicle using the uniform two-dimensional flow between the shock and the wing undersurface as precompressed air for an engine intake, particularly as the flow direction is relatively insensitive to wing incidence.

Because there is a wide variation of attached shock wave angles and Mach number for a given geometry of wing Peckham[3] points out that it may be possible to arrange a re-entry trajectory that by change of attitude will enable the desired lift/drag ratio to be obtained.

With a Nonweiler wing the angle through which the flow is turned, and the shock angle ζ, is fixed by the geometry of the wing; so that for a specific geometry there is a unique Mach number for shock attachment at any one incidence.

Measurements have been made by Squire and Treadgold (reported in Ref. 3) on some Nonweiler wings in continuous wind tunnels at a flow Mach number of 4.0 and 4.3, and extremely close agreement is found with two-dimensional theory for shock angle and pressure coefficient with change of incidence.

The dimensions of the Nonweiler wing used in this investigation were such as to give a "closing" oblique shock wave of $\zeta = 15.5°$ for an incidence $\alpha = 10°$ at $M_\infty = 8.0$. When this design is tested at $M_\infty = 8.8$ the shock solution should hold at $\alpha = 5°$ and $15°$, with only a slight departure in this range. It is found experimentally that the shock angle does not alter by more than $0.5°$ in this range.

The aspect ratio of the body is 4/3. The length is 8 in., the wing span is 5-1/3 in., the planform sweepback angle is 36.7° and the angle of intersection of the two undersurfaces is 147.8°.

3. Brief Description of N.P.L. 6 in. Shock Tunnel

The 6 in. shock tunnel at the N.P.L. has been in operation since December, 1960. The internal diameter of the driver and driven sections is nominally 6 in. The length of the driver chamber is 18.5 ft and the driven tube is 39 ft. The nozzle is designed for straight-through operation with provision for boundary-layer bleed, but is used principally in the reflected shock manner. At present a conical nozzle is used having an exit diameter of 16 in. followed by a closed jet working section which exhausts into a 370 cu ft vacuum chamber.

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When used as a reflected shock tunnel, with a cold gas driver the available "running times" are 6 mS with hydrogen driver at $M_s = 6.0$, 7 mS with helium driver at $M_s = 4.10$ and 4 mS with nitrogen driver at $M_s = 2.4$. The time durations quoted refer to the quasi-steady pressure step occurring in the reservoir gas.

The maximum pressure level that can be used in the driver and driven sections is 7,500 p.s.i.

Pressurising of the driver chamber can be directly from a mobile-rechargeable trailer (15,000 standard cu ft) at 3,600 p.s.i. or from 9 cu ft storage bottles at 15,000 p.s.i. which contain the appropriate gas compressed previously by a Hofer compressor.

The shock tunnel is fired by pressurising the chamber until a previously weakened metal diaphragm bursts rapidly, and the shock wave is driven down the channel. Mild steel diaphragms of 1/4 in. thickness and having 0.085 in. Vee milling burst repeatably at 2,850 p.s.i., ±50 p.s.i. The channel pressures for tailoring with room-temperature hydrogen and helium respectively are 8 p.s.i.a. and 22 p.s.i.a. The equalising pressure in the nozzle and vacuum chamber after a run is typically 40 p.s.i.a.

The shock wave speed is determined from 6 fast-response platinum thin-film thermometers spaced at 2 ft intervals. The reservoir pressure is measured by a SLM PZ6 quartz transducer, followed by a PV16 electrometer amplifier, and displayed on a Tektronix oscilloscope type 535. The model local pressure is sensed by a SLM PZ6M quartz transducer, followed by a PV16 electrometer amplifier and is displayed on a Tektronix oscilloscope type 502.

The shock tunnel nozzle has a changeable throat. Using a 1 in. diameter throat the flow Mach number deduced from the ratio of pitot to reservoir pressure (as in Fig. 6) lies between 8.0 and 9.0 according to the calculated temperature of the reflected shock wave. With hydrogen driver gas at $M_s = 6.0$, the reflected shock temperature is 4,016°K and $M_\infty = 8.00$. With helium driver gas at $M_s = 4.15$ the reflected shock temperature is 2,100°K and $M_\infty = 8.50$. With nitrogen driver gas at $M_s = 2.4$, the reflected shock temperature is 920°K and $M_\infty = 8.80$ at centreline.

Models in the closed jet working section are supported on a quadrant mounted sting which is isolated from the tunnel by vacuum bellows and rigidly attached to the ground.

4. Shock Angle Measurements

A schlieren system using a 10 ft focal length, 15 in. diameter spherical mirror on the spark side and a 20 ft focal length, 15 in. diameter spherical mirror (on the camera side) was used with two 22 in. diameter plane mirrors to take the model flow photographs. A single unit argon-jet stabilised spark light source was used in conjunction with a horizontal graded cut-off. The image was recorded on Ilford LN (blue sensitive) plates.

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The Nonweiler model had a mechanical register for a flat plate to be attached, and the roll angle of the model was adjusted to zero by observing the flat plate image in the direct shadow beam. The flat plate was removed for the tests. The incidence of the model was checked before and after each run.

The free stream density was $2.3 \times 10^{-2} \rho_0$ for Fig. 4 and $1.4 \times 10^{-2} \rho_0$ for Fig. 5. The Reynolds number was 2×10^6/ft for Fig. 4 and 0.8×10^6/ft for Fig. 5.

Measurements of shock angle were made with a simple protractor from 8 in. × 6 in. photographs. The error in interpretation was thought to be not greater than 0.5°.

5. Pressure Measurements on Models in High Mach Number Expanded Flows

Pressure measurements in cold gas driven shock tunnels have not been commonly reported. Cornell Aeronautical Laboratories (Buffalo N.Y.) have achieved excellent pressure plotting of models in their 24 in. exit-diameter conical nozzle on their heated helium driven shock tunnel at reservoir pressures of 6,000 p.s.i. Their ability to do this is directly related to the several years in which a small group of technicians have treated this as a real problem to be solved, and have constructed and and designed transducers on an 'ad hoc' basis, until the essential physical features required have been realised and incorporated. A recent paper by Martin[6] shows that pressure levels of order 10^{-3} p.s.i.a. can be measured after a response time of about 2 mS.

General Electric (Valley Forge, Pa.)[5] using a large combustion heated shock tunnel at reservoir pressures of order 4,000 p.s.i. have possibly contributed the most useful information on pressure measurements about models in high Mach number short duration flows, at pressures down to 10^{-3} p.s.i.a.

At AVCO (Wilmington Mass.) similar pressure levels are measured, again using an independently developed surface pressure gauge.

It is clear that the success of these groups has resulted because of directing research and development (quite independently) to this problem.

Stevens[7] at R.A.E. has evaluated some commercial gauges and R.A.E. designed gauges, and shown in a detailed manner how suitable model surface pressure gauges may be constructed. He has tested his design of gauge in a shock tube and found the design to be most satisfactory for freedom from vibration and self-oscillation, but has not yet used them in a shock tunnel.

At the N.P.L. the only experience previously gained was in the measurement of pitot pressure in various shock tunnels; and in the design of a seismic model support system for removing tunnel firing vibrations which has enabled pressures to be measured on models down to 0.05 p.s.i.a. Because this model support system is attached to an 8 in. nozzle[8], and the

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present work is a preliminary to the simultaneous measurement of pressure at several stations on the model, it has been necessary from considerations of model size to use the 6 in. diameter channel shock tunnel having a 16 in. diameter nozzle for these tests, and mount the instrument model on a conventional sting. An unfiltered oscilloscope trace obtained in this tunnel on the model is shown in Fig. 7(b) showing some fluctuations of pressure having frequency components of 4 to 5 kc/s. These may be stress waves in the tunnel, or actual flow perturbations. The pressure gauge assembly is not at present shock-isolated from the model, and when implemented this should improve the steadiness of the trace.

6. Pressure Transducer Calibration

The quartz gauge may be statically calibrated because of the high leakage resistance of specially treated quartz crystals, the use of special high resistance cable $(> 10^{12}\ \Omega)$ and the very high input impedance of the electrometer amplifier. Special cable connectors must be used to maintain the charge insulation.

It has been variously reported[5,6] that static calibration and shock-tube dynamic calibration curves do not agree. Since shock tunnel model testing involves dynamic pressure changes of short duration, taking place at low ambient pressure levels, then to calibrate by applying a step pressure signal of known amplitude more closely simulates the actual test environment. The dynamic calibration has been carried out in a uniform rectangular shock tube with a fixed driving pressure of 1 atmosphere, and the shock wave speed measured by observing the time for the shock wave to pass two platinum thin-film resistance thermometer stations. The actual thin-film signals were displayed on a calibrated oscilloscope time base for the time difference measurement. This was done rather than using microsecond chronometers because the thin-film signals were typically < 1 mV at these low initial channel pressures and Mach numbers. It has been reported[7,9] that experimental normal shock wave results are in agreement with shock tube theory up to Mach 4.0. The present design of calibration shock tube only permitted using the reflected shock pressure for calibration. The experimental agreement with theory has not been reported at low Mach numbers. The measured pressure levels have been correlated with theory and a calibration constant deduced.

The static calibration utilises the complete gauge and associated electronic circuits as used for the shock tunnel run. Pressures have been read on a Wallace and Tiernan gauge and the DC electrical signal at the oscilloscope noted for various pressures in the required range. Remarkable linearity and repeatability has been found when calibrated in this manner (Fig. 11).

7. Undersurface Pressure Measurement

7.1 Cavity and gauge

The expected pressure level behind the oblique shock wave on the model was in range of 0.1 to 1.0 p.s.i. for a nitrogen driver gas at 3,000 p.s.i. and a channel pressure of 30 p.s.i.

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Of the gauges that could be statically calibrated only SLM quartz piezo-electric transducers and Statham 0-10 p.s.i.a. flush-diaphragm strain-gauge transducers were available. Of the former only the PZ6M was of a size that would enable several gauges to be fitted within one model. The PZ6M gauge had its vent sealed at atmospheric pressure. Previous use of this gauge in short duration facilities has shown it to be acceleration and temperature sensitive. It has an unnecessarily high natural frequency for test section measurements, and being intended for use at high pressure levels has a low charge sensitivity which necessitates the gauge being followed by the shortest possible lead length (i.e., minimum lead capacity) if a sensitivity comparable with a strain gauge transducer is to be obtained (nominally 2 mV/p.s.i.). The great advantage of using the quartz gauge in this application is that tunnel pressures may exceed the required working range of the Statham transducer and cause permanent damage to the linkage system.

Special high insulation low noise coaxial cable having a length of approximately 6 ft was taken from the gauge on the model through a pressure-vacuum tight screened electrical connection in the door of the nozzle chamber. The total of the lead capacitance and internal capacitance of the electrometer unit was 230 pFs. The PV16 electrometer unit was used with its minimum input capacitor selected, and the output signal taken directly from the electrometer stage. The white-noise level in this condition was 50 μv peak-peak.

The effect of cavity size on the performance of a gauge in short duration flows has been overlooked by the majority of experimenters. Ref.5 shows a simple criteria that has been derived empirically. The present authors have used this as an aid in designing for a pressure response time of 0.1 mS. The reason for choosing this response time is that in an available steady measuring time of 5 mS, as displayed on an oscilloscope over 10 cm, then signal excursions having a period of 0.2 cm would be of little consequence at this stage of the investigation, and would be better removed acoustically by design of the pressure hole, since this provides a measure of crystal shock isolation (Ref.5), than by an electronic filter after the pressure gauge which usually results in the depletion of the measured signal voltage. The response time of the PZ6 gauge and its associated detachable cavity is about 0.1 mS as measured in a calibration shock tube (Fig.7(c)). It is to be noted that the rise to a steady local pressure (as in Fig.7(b)) is not a characteristic of the cavity and the gauge, but is a feature of the nozzle starting process. In view of this it would clearly be possible to have a longer pressure sampling tube which would immediately improve the signal to noise ratio.

The pressure sampling tube may give rise to resonance effects at higher ambient pressure levels. Davis[10] shows that the dynamic response amplitude indicated by a pressure gauge in a cavity for 3 ambient pressures ranging from 1 atmosphere to 1/3 atmosphere, changes from that of an acoustic resonance associated with the tube length to a normal second-order circuit response of -6dB/octave at the lower ambient pressure. For this reason it was thought that the shock tube initial environment pressure (10^{-3} mm Hg) might assist the required measurement. It should be noted that the frequency response characteristic of a pressure transducer

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and cavity is dependent upon the absolute value of the ambient pressure to which it is subjected, and for a transducer experiencing a varying static pressure there is no unique frequency response characteristic.

The internal bore of the sampling tube was 0.078 in. (2 m.m.) and the total length of the tube was 0.15 in. The total volume ahead of the PZ6M gauge was 0.25×10^{-2} cu ins.

7.2 Reduction of data

The flow Mach number (M_∞) is defined for these experiments by the measured ratio of the pitot pressure and the reservoir pressure at a corresponding time. This ratio is tabulated in Ref. 4 for flow Mach number in increments of 0.01. The ratio of free stream pressure to reservoir pressure is similarly tabulated.

The pitot pressure was measured by an SLM PZ38 quartz transducer mounted in a 2 in. diameter cylinder end-on to the flow. The form of pressure response is shown in Fig. 6(b). This gauge, which is essentially a differential pressure gauge, was calibrated with a reference back pressure of 10^{-3} mm Hg and used in the tunnel under the same conditions.

Thus for an assumed flow Mach number, and a measured reservoir pressure, a value of free stream pressure p_W may be deduced. The measured local pressure on the body (p_L) may then be used to define a pressure coefficient as follows:

$$C_p = \frac{p_L - p_W}{\frac{1}{2}\rho_W U_\infty^2} = \frac{p_L - p_W}{\frac{1}{2}\gamma p_W M_\infty^2} = 0.0185\left(\frac{p_L}{p_W} - 1\right) \quad \text{for} \quad \begin{aligned} M_\infty &= 8.80 \\ \gamma &= 1.40 \end{aligned}$$

Measured and theoretical values of C_p, and p_L/p_W against angles of incidence are plotted in Figs.10 and 9. The theoretical values are for a two-dimensional wedge at the same incidence.

8. Discussion of Results

8.1 Shock wave angle

The shock angle has been measured for angles of incidence in the range $-5.5°$ to $+20°$ referred to the inner ridge line of the inverted delta. It can be seen in Figs. 4 and 5 that the shock remains closely attached to the arms of the inverted delta over a wide range of incidence. Flow pictures have been taken for reservoir temperatures of 920°K and 2,100°K, and show the same effect. The measured angle is compared in Fig. 8 with the theory for a two-dimensional wedge at the same Mach number.

The actual angle of the inner-ridge line to the plane containing the swept leading edges was 5.50°, and at $M_\infty = 8.0$ and at an incidence of 10°, the shock wave should have been coincident with this same plane. At a flow Mach number of $M_\infty = 8.80$, Peckham[3] shows that two shock angles exist for the condition of shock attachment at low incidence.

Reference/

Reference to Fig. 8 shows that the measured shock attachment angle only varies through about 1° for an incidence change of 5° to 20°.

8.2 Local pressure

Measurements of local pressure have been made at model incidence angles of 0°, 5°, 8°, 10° and 16°. The static gauge calibration constant has been used to determine the local pressure, and the free stream pressure deduced from measured reservoir pressure.

At each angle of incidence, two or three runs have been made and the results plotted in an attempt to indicate repeatability of the shock tunnel conditions. The scatter is about 10%. Since the two relevant pressures can be measured to 5% from the oscilloscope traces, the width of the scatter may be indicative of slight variations in flow Mach number from run to run. If the flow Mach number changes from 8.80 to 8.70 then p_W changes by 7%, and since the pressure ratio across the oblique shock should be a function of incidence only, the local pressure measured will be of order 7% in error. It would clearly be desirable to measure the pitot pressure during every run to be sure of defining the flow Mach number.

In view of the fact that the measured shock angle was greater than predicted by theory, it is not unexpected that the measured pressure-ratio across the actual shock wave is greater than for the theoretical case.

The pressure ratio p_L/p_W has been calculated using the measured shock angle and the variation is plotted in Fig. 9. At low angles of incidence the agreement is better, but at higher angles of incidence where the theory and measurement almost agree, the pressure measurement is 20% high on theory. Further experiments are needed to explain this discrepancy, since in tests reported in Ref. 3 at $M_\infty = 4.3$ no large departures were evident.

9. Conclusions and Future Work

This preliminary investigation has shown that a plane shock is formed close to the leading edges of a Nonweiler wing and is fairly insensitive to angle of incidence. Measured local pressures agree reasonably with theoretical pressures behind oblique shocks on a two-dimensional wedge.

The use of a miniature high pressure gauge to measure very low pressures has proved to be possible, and the use of a detachable cavity to permit a shock tube calibration and hence a dynamic time constant to be evaluated has been justified.

Future work will include the pressure plotting of the undersurface with several gauges simultaneously to see if it is isobaric, and heat transfer rates near the leading edges of the body will be measured for various leading edge bluntness.

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10. Acknowledgements

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References/

References

No.	Author(s)	Title, etc.
1	T. Nonweiler	Delta wings of shapes amenable to exact shock-wave theory. A.R.C.22,644. March, 1961.
2	D. M. Sykes	A note on the hypersonic flow past plane and inverted-V delta wings. A.R.C.22,848. May, 1961.
3	D. H. Peckham	On three-dimensional bodies of delta planform which can support plane attached shock waves. A.R.C. C.P. No.640. March, 1962.
4	Ames Research Staff	Equations, tables and charts for compressible flow. NACA Report 1135. 1953.
5	C. J. Harris and E. M. Kaegi	The application of pressure and force transducers in shock tunnel aerodynamic studies. General Electric (MSVD). 1959.
6	J. F. Martin, G. R. Duryea and L. M. Stevenson	Instrumentation for force and pressure measurements in a hypersonic shock tunnel. Advanced Hypervelocity Techniques Symposium. Denver, Col. March, 1962.
7	D. R. Stevens	The evaluation of some commercial and development pressure gauges in a laboratory type shock tube with a view to their suitability for use in shock tunnels. A.R.C.23,803. March, 1962.
8	L. Pennelegion, P. Gough and P. J. Peggs	NPL Aero Note in preparation.
9	J. Lukasiewicz	Shock tube theory and applications. N.A.E. Canada Report 15, 1952.
10	E. L. Davis	The measurement of unsteady pressures in wind tunnels. AGARD Report 169, March, 1958. Presented to Wind Tunnel and Model Testing Panel of the Advisory Group for Aeronautical Research and Development, London, England, March 24-28, 1958.

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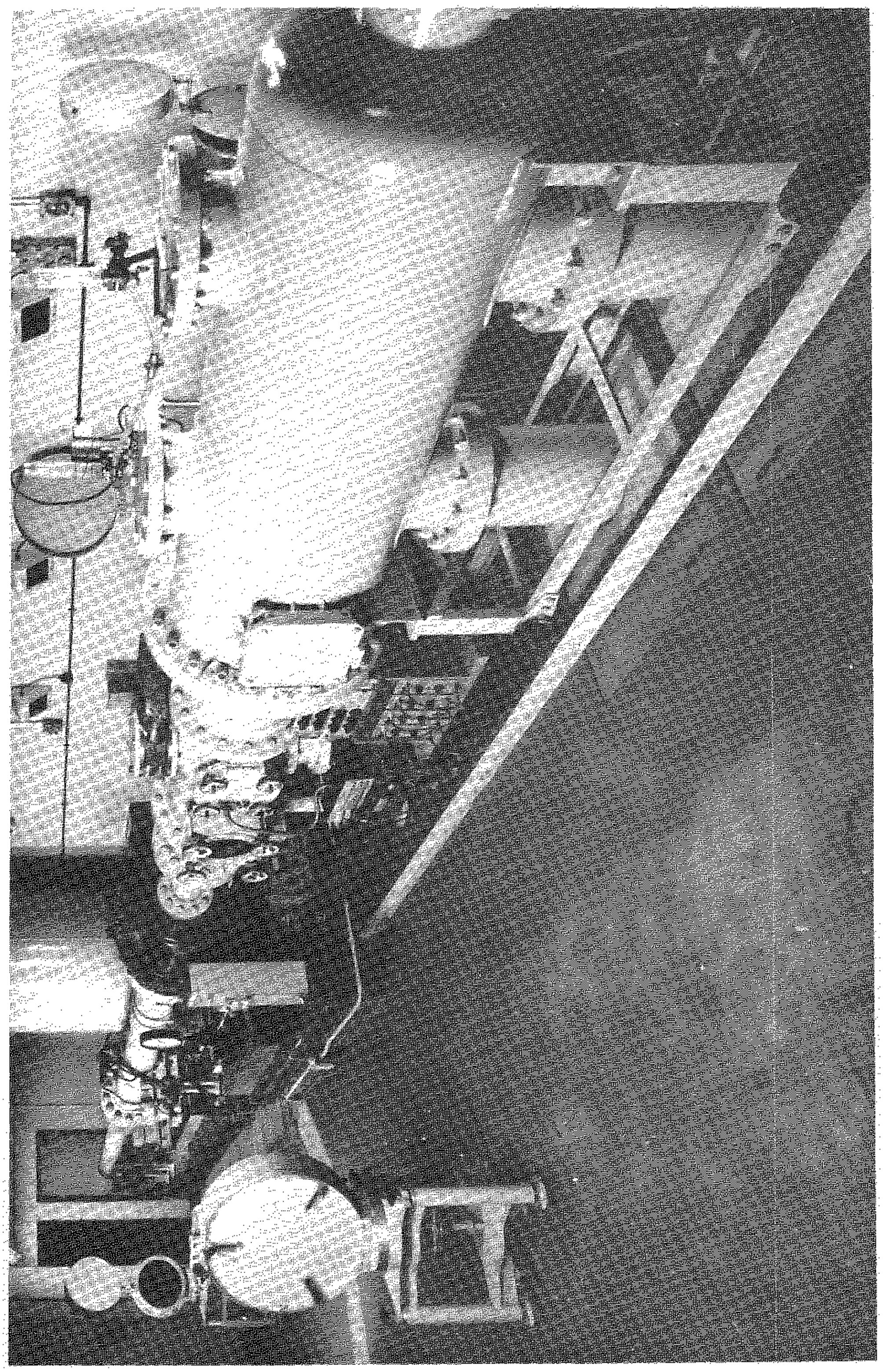
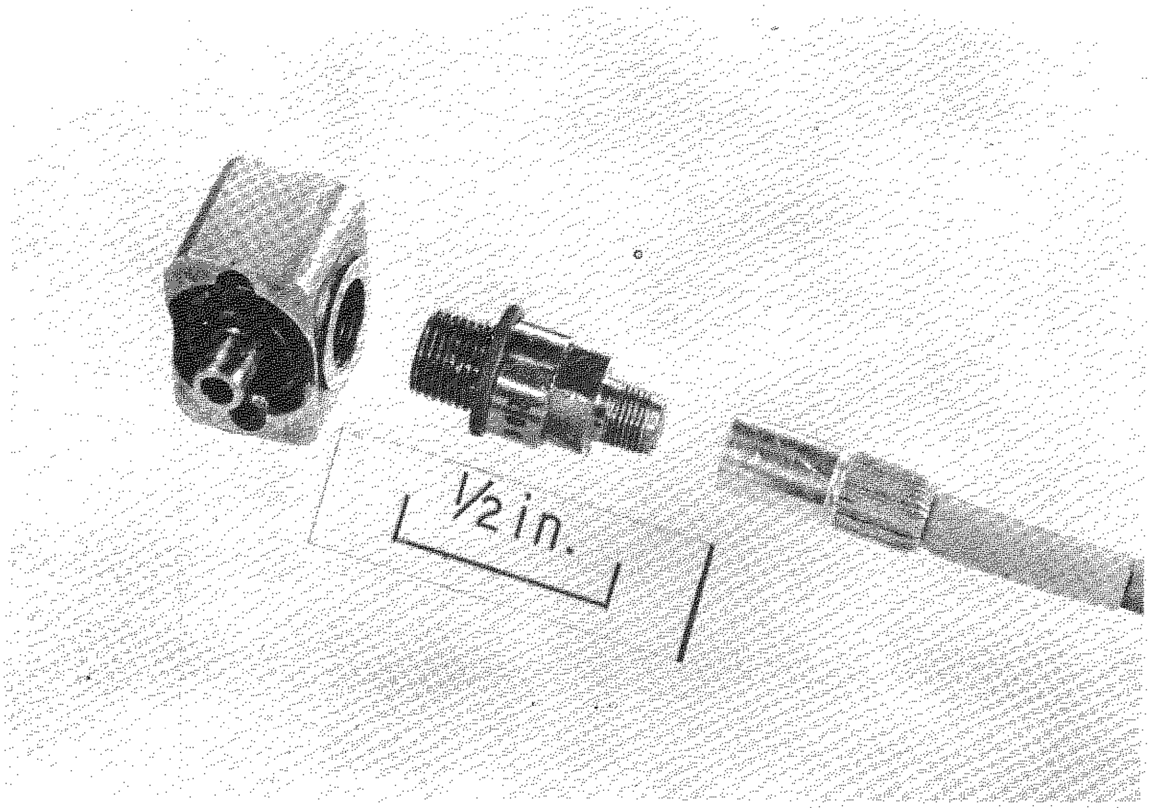
FIG. 1



NPL 6" Hypersonic shock tunnel.

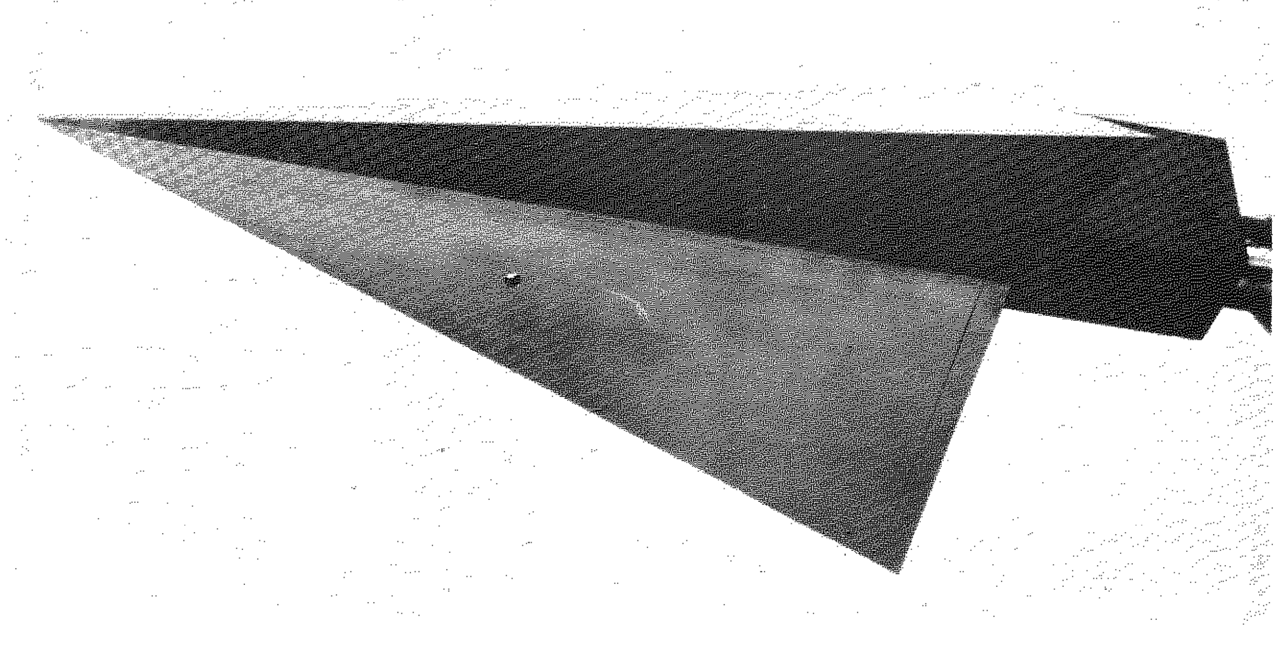
(80843)

FIG. 2



Transducer and detachable cavity

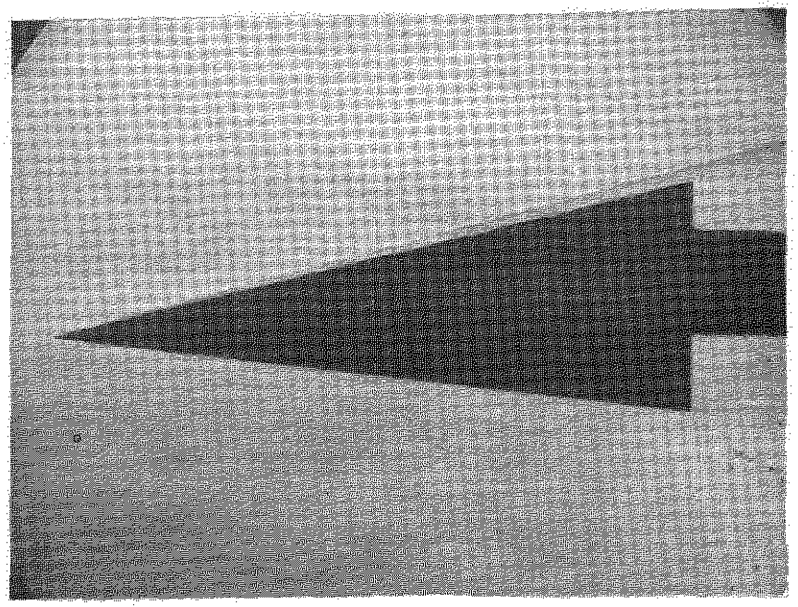
FIG. 3



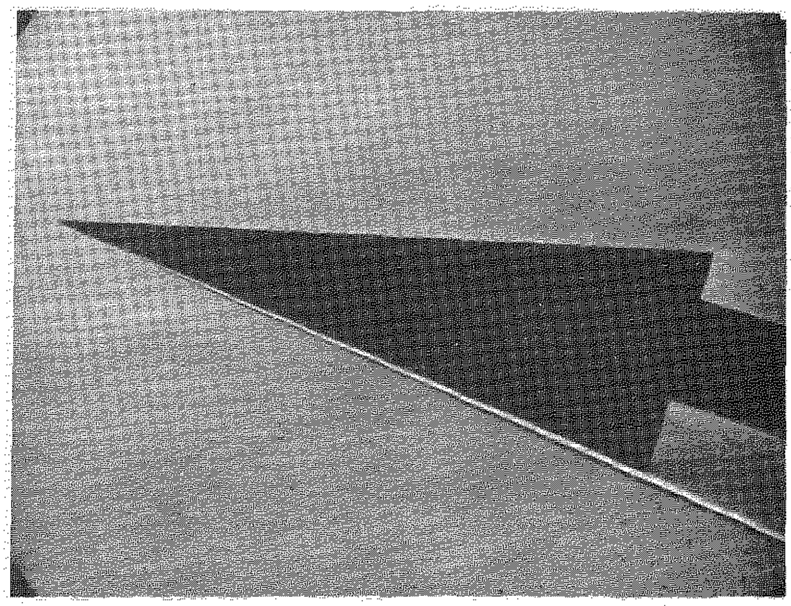
Nonweiler wing showing pressure station

FIG. 4

$M_{\infty} = 8{\cdot}80 \quad T_{res} = 920^{\circ}\,K \quad p_{res} = 720$ p.s.i.a.



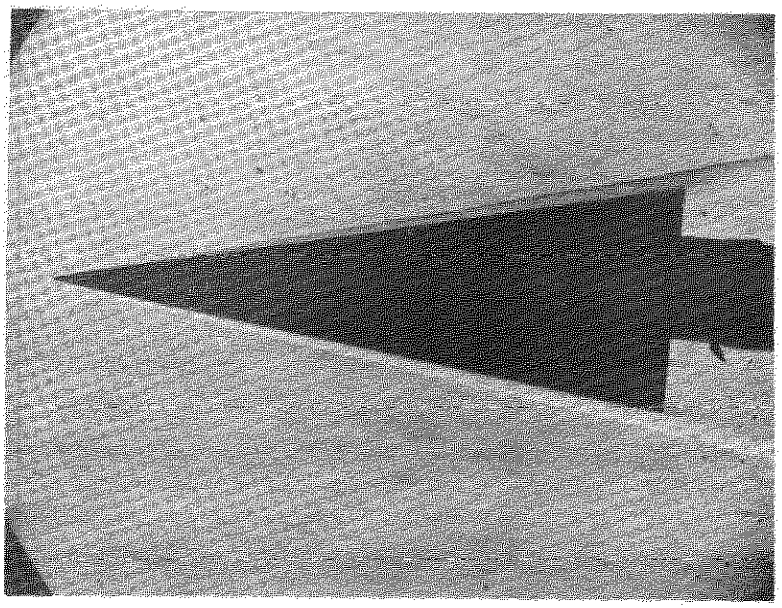
$\alpha = 0^{\circ}$



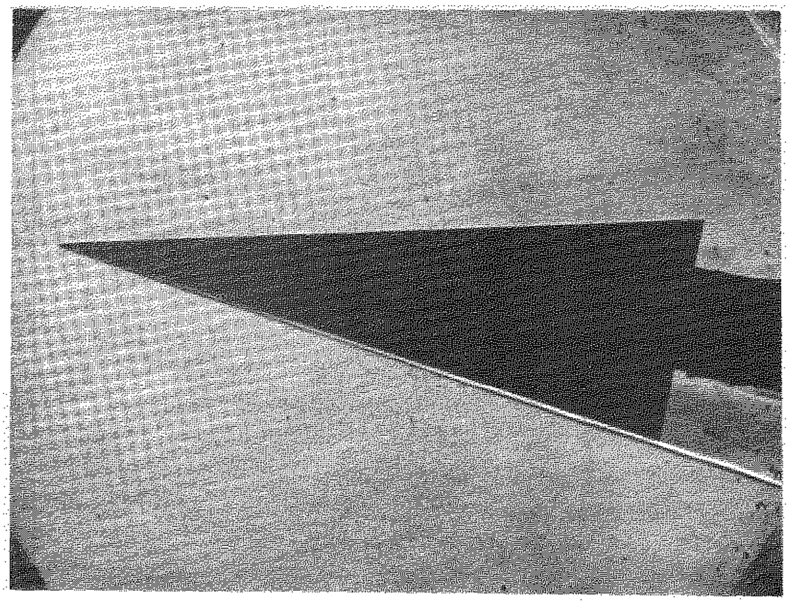
$\alpha = 16^{\circ}$

FIG. 5

$M_\infty = 8{\cdot}50 \quad T_{res} = 2{,}100^\circ K \quad p_{res} = 1725$ p.s.i.a.

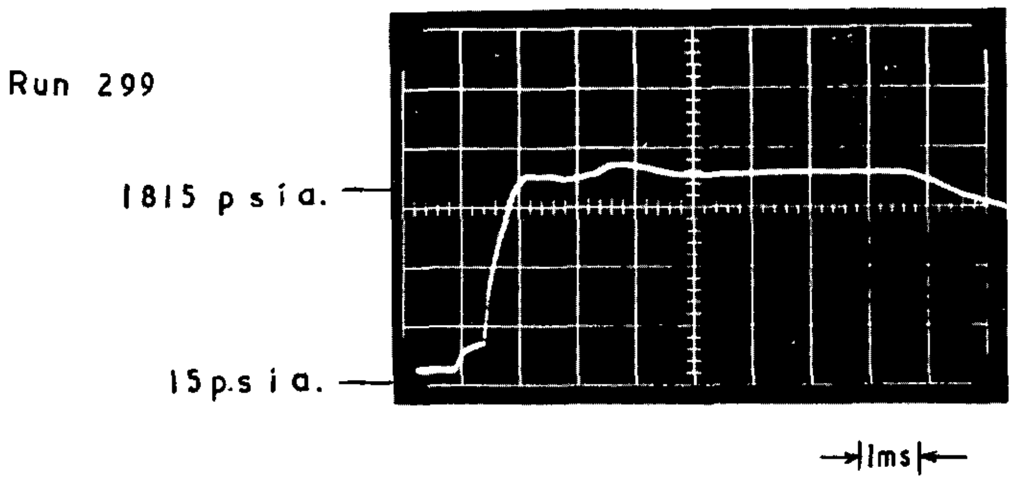


$\alpha = 2^\circ$



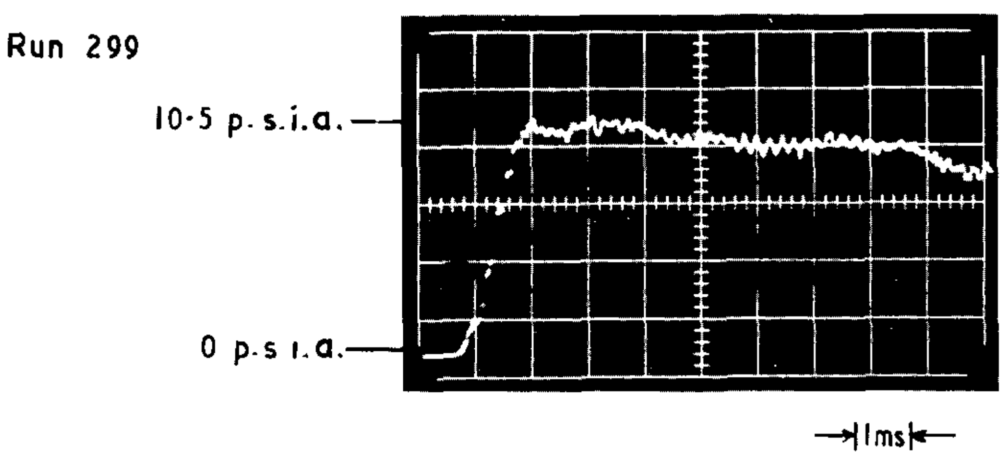
$\alpha = 10^\circ$

FIG 6



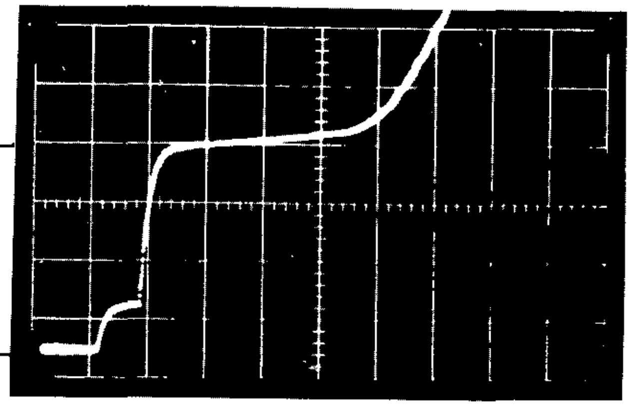
(a) Nozzle reservoir pressure

$M_s = 4\ 10 \quad T_{res} = 2100^{\circ}K$

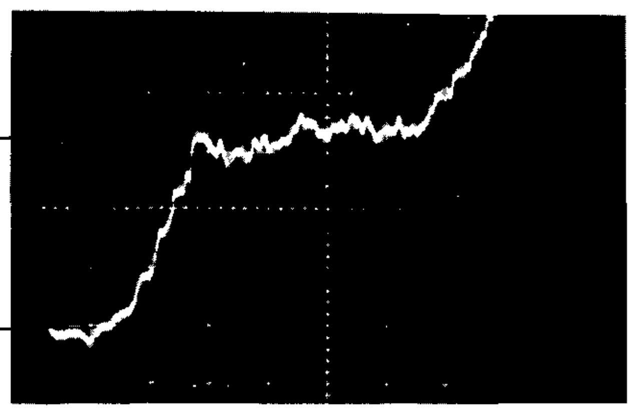


(b) Pitot pressure $M_{\infty} = 8{\cdot}70$

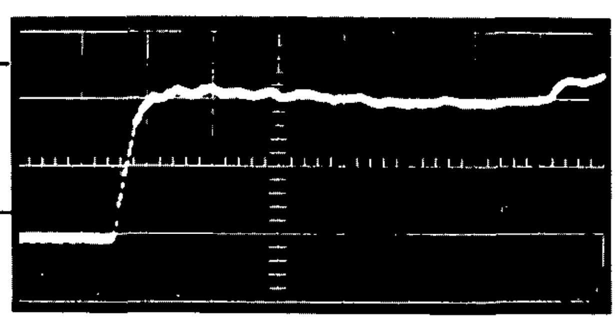
FIG. 7



(a) Nozzle reservoir pressure
$M_s = 2 \cdot 40$ $T_{res} = 920°$ K

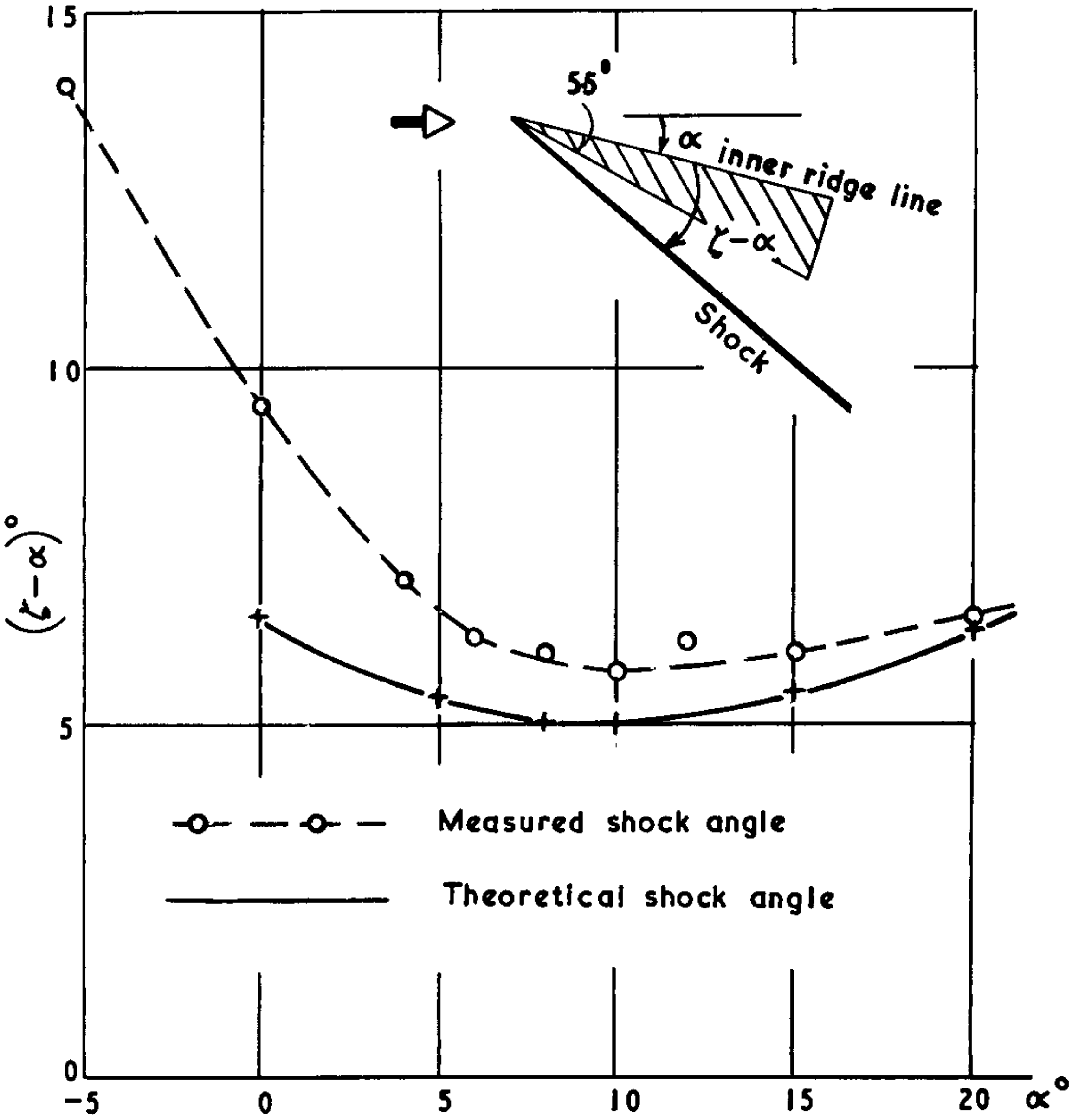


(b) Under surface local pressure
$M_\infty = 8 \cdot 8$ $\alpha = 10°$



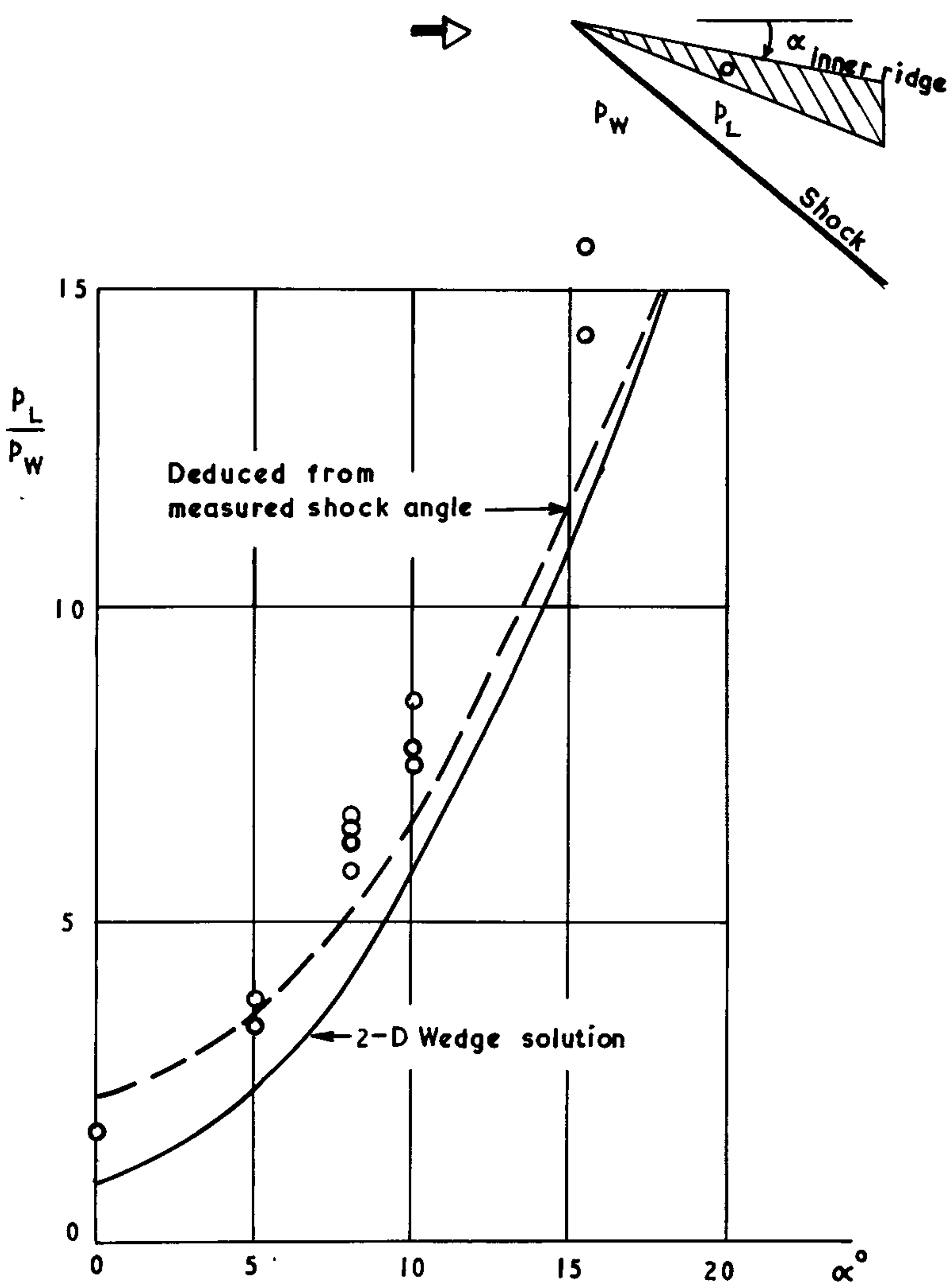
(c) Dynamic calibration response

FIG. 8



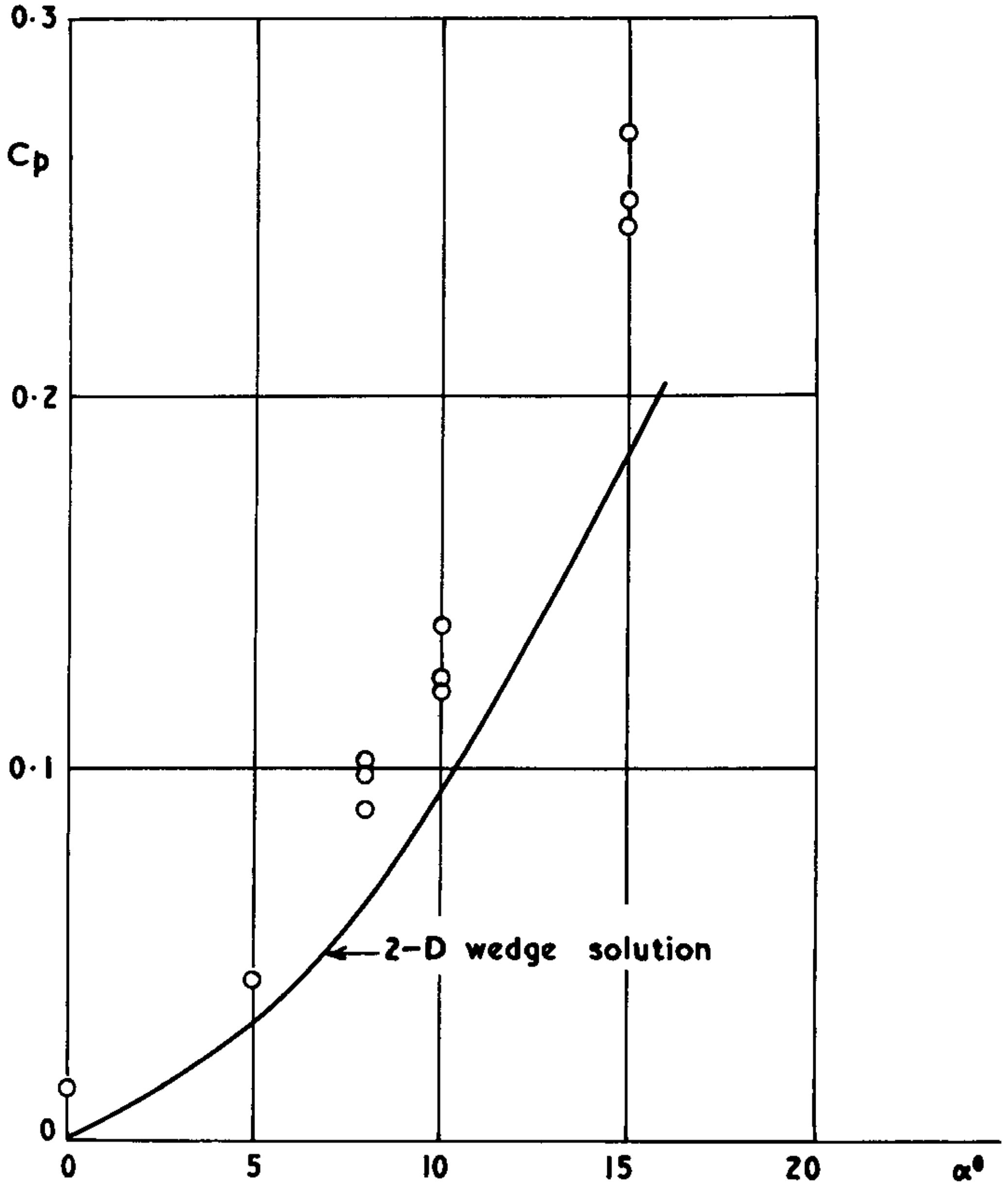
Theory and measured shock angle v. model incidence ($M_\infty = 8{\cdot}8$)

FIG. 9



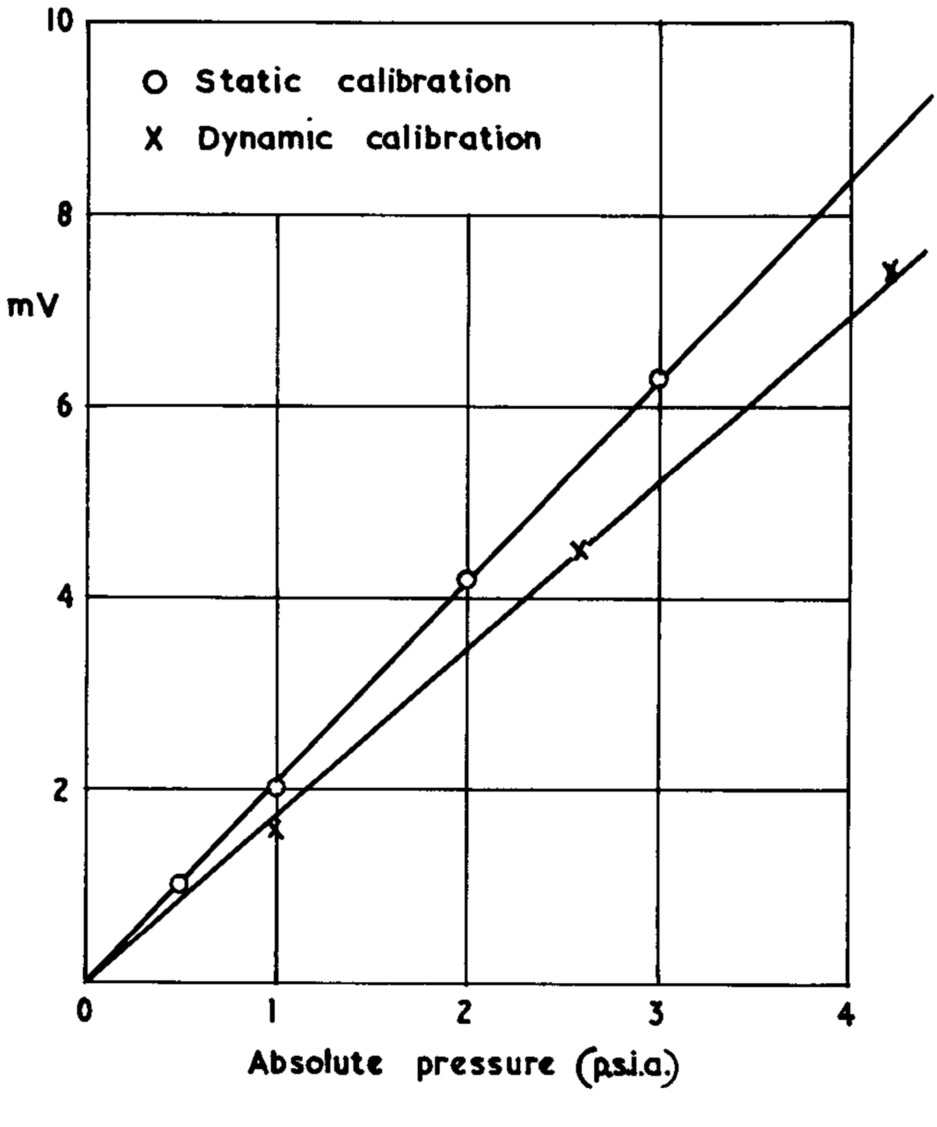
Variation of pressure ratio p_L/p_W with incidence at $M_\infty = 8\cdot8$

FIG.10.



Variation of pressure coefficient C_p with incidence at $M_\infty = 8{\cdot}8$

FIG. 11.



Static and dynamic calibration of local pressure.

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July, 1962.
Pennelegion, L. and Cash, R. F.

PRELIMINARY MEASUREMENTS IN A SHOCK TUNNEL OF SHOCK ANGLE AND UNDERSURFACE PRESSURE RELATED TO A NONWEILER WING

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Pennelegion, L. and Cash, R. F.

PRELIMINARY MEASUREMENTS IN A SHOCK TUNNEL OF SHOCK ANGLE AND UNDERSURFACE PRESSURE RELATED TO A NONWEILER WING

Measurements of shock angle and undersurface pressure of a Nonweiler wing have been made in a closed jet of a shock tunnel at $M_\infty = 8.8$. Over an incidence range of $-5.5°$ to $+20°$ the agreement with the flow about a two-dimensional wedge is within 15%.

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