

كتاب تجارب معمل

ديناميكا الغازات

أولاً: بيانات المعمل الأساسية

إسم المعمل: معمل ديناميكا الغازات

القسم العلمي: ميكانيكا قوى

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1	Air Flow Instruments	To measure the sensitivity of a Pitot-static tube to misalignment. To calibrate a cylindrical yaw probe
2	Pressure Distribution Over A Circular Cylinder.	To calculate the pressure distribution resulting from the flow of an ideal fluid about a circular cylinder and to compare with the measured flow of a real fluid.
3	Wind Tunnel Tests On Airplane	To obtain a Drag Polar of the F-16 Airplane. To measure the static longitudinal and directional stability of the airplane
4	Measuring The Drag of Simple Shapes & The Turbulence of The Air Stream	To measure the drag of simple shapes at various Reynolds numbers. To measure turbulence of the air stream of the AERO LAB Wind Tunnel.
5	Characteristic Curves of an Airfoil with a Leading Edge Slat & Split Flap	To measure the Lift, Drag and Pitching Moment of an airfoil with slat and with flap.
6	Pressure Distribution over an Airfoil	To measure the pressure distribution over an Airfoil at various angles of attack.
7	Boundary Layer on a Flat Plate	To measure the velocity profile in the boundary layer at several longitudinal stations on a flat plate and to estimate the location of transition.

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Experiment 1

▪ بيانات عامة:

إسم التجربة: .. AIR FLOW INSTRUMENTS.....

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثانى

الأدوات المطلوبة للتجربة: جهاز قياس معدل السريان

▪ الأساس النظرى للتجربة:

OBJECTIVES:

- I. To measure the sensitivity of a Pitot-static tube to misalignment
- II. To calibrate a cylindrical yaw probe

DISCUSSION:

- I. A Pitot-static tube is widely used for measuring unknown velocities in fluids. Usually the direction of flow can be estimated within a few degrees at any point in a flow pattern and the Pitot-static tube pointed in the assumed direction. Our objective is to determine how large the discrepancy may be between assumed and actual flow directions before appreciable errors appear in the measurement.
- II. The cylindrical yaw probe has three static pressure openings spaced at 45° intervals. The center opening indicates total pressure and is not directly used for angular measurements. The two outer holes are at 90° to each other. The static pressure read

at each outer hole should be the same if the flow bisects the angle between the holes. The difference in pressure Δp would then be zero. If the flow does not bisect the angle, Δp may be expected to be a function of the angle θ (which is defined as the angle between the flow direction and the bisector). Δp should be expressed non-dimensionally by dividing by the dynamic pressure $q = 0.5\rho V^2$. The triangular block may not be set exactly perpendicular to the bisector of the angle formed by the outside holes giving rise to an "instrument error". In addition, the air velocity in the tunnel is usually aligned a fraction of a degree from the horizontal direction. This deviation from the horizontal may be termed the "airstream error". It is clear that the instrument error is inverted when the instrument is inverted but not the airstream error. Erect and inverted runs may, therefore, be used to separate and evaluate these errors. The instrument error can be reduced to zero by re-adjusting the alignment block until the average between the erect and inverted curves passes through the origin. The reasoning underlying this procedure should be investigated by the student with the help of sketches.

PROCEDURE:

1. Insert the Pitot-static tube into one of the threaded fittings in the side of the test section so that the sensing head is in the center of the test section pointing upstream. Connect the leads to two ports on the 24-tube multi-manometer. Using an inclinometer, adjust Pitot-static tube for zero degree angle of attack. Run tunnel at 140 mph then measure and record the total pressure and static pressure at the Pitot-static tube. Adjust angle of attack plus and minus in 1° increments until reading changes dramatically. Record all results.
2. Insert yaw head in the same manner as the Pitot-static tube. Attach the two outside tubes to the 24-tube multi-manometer. Adjust yaw head angle of attack to zero using an inclinometer on the slotted side of the base. Run tunnel at 140 mph then measure

and record pressure at P1 and P2. Adjust Angle of Attack plus and minus in 2° increments for a total of ±20°. Record all results. Insert yaw head through opening on other side of test section thereby inverting the instrument. Repeat the procedure.

DATA REDUCTION:

1. Plot the dynamic pressure read by the Pitot-static tube as a function of angle to the airstream.
2. Plot $\Delta p/q$ for yaw head versus angle of attack for the erect and inverted position. Assume q is equal (in magnitude) to the depression of the test section pressure below atmospheric.
3. From the foregoing plots determine (a) the maximum angular misalignment for a Pitot-static tube to maintain the dynamic pressure reading within 1/2 percent of its value when aligned with the flow.
4. From the plots of the yaw head data determine (a) the instrument error, (b) the airstream inclination, (c) the calibration factor of the yaw head $K = d(\Delta p/q)/d\theta$ taken near zero yaw.

ADDITIONAL ASSIGNMENTS:

1. Plot a theoretical curve for $\Delta p/q$ versus angle based on the equation for the pressure coefficient derived from perfect fluid theory for the flow around a circular cylinder.

$$C_p = 1 - 4 \sin^2 \beta$$

(where β is the angle between a hole and the airstream). Compare with p measured values.

2. For maximum accuracy the static, total and dynamic pressures measured by a Pitot-static tube should be corrected. Determine the correction to static, total and dynamic pressures for the Pitot-static tube (modified Prandtl type) using information from sources such as Ref. 1.

3. Using the corrections to P_s , you found in step 2, measure the true static pressure in the test section (with the Pitot-static tube) and divide by the pressure measured in the static pressure ring (used by the DAC to determine "q" and airspeed). If the ratio is appreciably different from unity assess the error in the yaw head calibration.

Experiment 2

▪ بيانات عامة:

إسم التجربة: .. PRESSURE DISTRIBUTION OVER A CIRCULAR
.CYLINDER

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثاني

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات تحت الصوتية بملحقاته

▪ الأساس النظري للتجربة:

OBJECTIVES:

To calculate the pressure distribution resulting from the flow of an ideal fluid about a circular cylinder and to compare with the measured flow of a real fluid.

PROCEDURE:

Remove the top window of test section to install the pressure cylinder. Mount the cylinder to the yaw table inside the test section. Replace the top cover. Connect tubes 1 24 to tubes 1 - 24 on 24-tube multi-manometer. Run the tunnel at 60 mph. Measure and record all 24 pressures around cylinder (note - pressures are referenced to prevailing atmospheric pressure because the manometer reservoir is open to atmospheric pressure). Repeat procedure at 40 mph.

DISCUSSION:

As derived in several standard textbooks, the pressure coefficient in ideal flow theory at any angular position θ on the surface of a circular cylinder is expressed by $C_p = 1 - 4 \sin^2\theta$.

The pressure coefficients resulting from the flow of air about a cylinder at 15° intervals will be measured and compared with theory. Since these real flow coefficients depend to some extent on Reynolds number it is recommended that measurements be taken at airspeeds of 40, 60, and 70 mph. Since the positions of the separation points are determining factors in the flow patterns, it is suggested that wool tufts be fastened to the cylinder near the equatorial diameter to indicate flow reversal and separation. (A wand with a single wool or thread tuft might also be used).

PLOT:

Plot the variation of Pressure Coefficient versus angular position for the ideal flow and the three real flows. Locate the position of separation points on the curves, if independently measured. Indicate the magnitude of Reynolds number for each curve.

NOTE:

The large size and drag of the model limits the top speed with the model in place to approximately 75 mph.

Experiment 3

■ بيانات عامة:

إسم التجربة: WIND TUNNEL TESTS ON AIRPLANE.

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثانى

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات تحت الصوتية بملحقاته

■ الأساس النظرى للتجربة:

OBJECTIVES:

- I. To obtain a Drag Polar of the F-16 Airplane.
- II. To measure the static longitudinal and directional stability of the airplane

DISCUSSION:

Many thousands of hours of wind tunnel tests, ranging from overall performance characteristics to stability and control are conducted on a new military airplane. In a single laboratory session, the student can only touch upon such a program.

WIND TUNNEL CORRECTIONS:

An airplane flies in practically a limitless volume of air (except when near the ground) whereas an airplane model "flies" in a wind tunnel test section in a volume of air which is more confined. This difference gives

rise to a number of corrections which are applied to wind tunnel data (Ref. 1). Many are quite small and in this experiment we will consider only the correction arising from the streamlines being forced straight by the flat walls of the wind tunnel whereas they are curved at corresponding distances from the full scale airplane, the so-called wind tunnel wall correction. The correction depends upon the shape and sweep of the wing as well as its span relative to the test section width but as an approximation we suggest the following additive corrections to angle of attack and to the drag coefficient for any given value of C_L :

$$\Delta\alpha = 1.5 C_L \text{ (in degrees)}$$
$$\Delta C_D = 0.02 C_L^2$$

Another consideration in reducing the data is that the Sting Balance reads Normal Force and Axial Force which coincide with Lift and Drag at zero angle of attack and otherwise are related, as follows:

$$L = N \cos(\alpha) - A \sin(\alpha)$$
$$D = A \cos(\alpha) + N \sin(\alpha)$$

The balance reads moments about an axis through the center of the moment strain gage bridge. The exact location of the balance moment axis can be determined by moving a weight along the calibration barrel (with the barrel horizontal) and noting the point where the moment changes sign and passes through zero. To be useful in the analysis of an airplane the moment axis should pass through the aerodynamic center (a. c.) of the airplane, which must be given or assumed. The moment read by the balance must consequently be adjusted by the Normal Force times the horizontal distance between the balance moment axes and the airplane a.c. and may be plus or minus depending on the geometry. (The two axes do not ordinarily coincide in a vertical direction either but this relatively short distance times the Axial Force may be neglected in this experiment).

PROCEDURE:

1. Zero balance components and airspeed at zero angle of attack. Make wind-off pitch run to obtain corrections for model weight.
2. Measure Normal Force, Axial Force and Pitching Moment at 2° intervals from $\alpha = -4^\circ$ to $\alpha = 16^\circ$,
3. Remove the airplane model from the balance, rotate the balance 90° within its holder as described in the EWT Operations Manual, reinstall the model (upright as usual) and measure Side Force, Yawing Moment and Axial Force at angles of yaw of $0^\circ, 5^\circ, 10^\circ, 15^\circ, 20^\circ, 25^\circ, 30^\circ$.

PLOT:

- (1) C_D vs. $C_{L(polar)}$
- (2) C_M vs. α
- (3) C_N vs. Ψ

ANALYSIS:

1. What is the minimum landing speed for the model as configured?
2. What is the thrust required for $M = 0.5$ at 20,000 ft. altitude (neglecting compressibility effects)?
3. What is the static longitudinal stability $dC_M / d\alpha$ at $\alpha = 0$?
4. What is the static directional stability $dC_N / d\Psi$ at $\Psi = 0$?

Suggestion: Conduct the test at 130 mph to cover the high drag range just beyond the stall.

Experiment 4

▪ بيانات عامة:

إسم التجربة: .. MEASURING THE DRAG OF SIMPLE SHAPES & THE
TURBULENCE OF THE AIR STREAM

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثاني

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات تحت الصوتية بملحقاته

▪ الأساس النظري للتجربة:

OBJECTIVES:

- I. To measure the drag of simple shapes at various Reynolds numbers.
- II. To measure turbulence of the air stream of the AERO LAB Wind Tunnel.

DISCUSSION:

- I. One would expect little scale effect (variation of Drag Coefficient with Reynolds number) on the blunt shapes and considerable scale effect on the rounded shapes. Investigate at airspeeds of 60, 80, 100, 120, and 130 mph.
- II. The drag coefficient of a sphere experiences a marked reduction when the flow in the boundary layer upstream of the separation point undergoes transition from laminar to turbulent flow. The Reynolds number associated with this transition is known as the

critical Reynolds number. Its value is lower for high turbulence air streams and its magnitude can be related to the turbulence of the air stream. To define its magnitude more precisely the critical Reynolds number is taken at the point on the drag curve where $CD = 0.3$.

The student should obtain the complete variation of drag coefficient with Reynolds numbers over an airspeed range from 50 mph to the top speed of the tunnel using the larger diameter sphere. (The smaller sphere is not large enough to undergo critical conditions even at the top speed of the tunnel).

The per cent turbulence may then be determined by referring to curves available from sources such as Ref. 1. The value of the critical Reynolds number may also be affected by the surface finish of the sphere. Hot wire anemometers are generally considered more accurate indicators of per cent turbulence.

ADDITIONAL EXPERIMENT:

Fabricate a ring of bare copper wire or fishing line with a diameter about $\frac{1}{2}$ inch less than that of the turbulence sphere. Mount the sphere on the balance and operate the tunnel at an airspeed of 10 or 20 mph below the critical Reynolds number. Note the drag of the sphere. Then fasten the ring on the sphere (with an adhesive or tape) and repeat. The drag should be reduced with the ring in place. Explain the seeming paradox where adding roughness decreases drag.

Experiment 5:

▪ بيانات عامة:

CHARACTERISTIC CURVES OF AN AIRFOIL WITH A . إسم التجربة:
LEADING EDGE SLAT & SPLIT FLAP

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثاني

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات تحت الصوتية بملحقاته

▪ الأساس النظري للتجربة:

OBJECTIVES:

To measure the Lift, Drag and Pitching Moment of an airfoil with slat and with flap.

DISCUSSION:

The airfoil profile is unusual in that its lower surface is straight and is taken as its chord line. The angle of attack can consequently be checked with an inclinometer contacting the lower surface.

It is recommended that tests be conducted at a dynamic pressure corresponding to about 120 mph (about 20% below top speed) to allow for the additional power required for the model in a high-drag attitude.

The student should review the DISCUSSION of Experiment 3 regarding tunnel wall corrections, relations between Lift, Drag, Normal Force and Axial Force and the transfer of Pitching Moment to parallel axes.

The tunnel wall corrections for this test may be approximated by the following additive connections:

$$\Delta\alpha = 1.7 C_L$$
$$\Delta C_D = 0.03 C_L^2$$

In correcting the results to infinite aspect ratio, assume elliptic lift distribution even though this condition is only an approximation, especially for the flap-deflected runs.

PROCEDURE:

1. Start out with slat and flap retracted (in the airfoil-alone configuration).
2. Set angle of attack reading to zero with the chord line horizontal as determined by inclinometer.
3. Zero balance components and airspeed at zero angle of attack. Make wind-off pitch run to obtain corrections for model weight.
4. Starting at an angle of attack of -4° with the wind on, take readings of Normal Force, Axial Force and Pitching Moment at 2° intervals to 4° beyond the stall.
5. Deploy the slat to about 3/16 inch slot and repeat items (3) and (4).
6. Deploy the flap and set at 45° to the chord line. Repeat (3) and (4).

DATA REDUCTION:

1. Convert Normal Force and Axial Force to Lift and Drag and transfer Pitching Moment to an axis through the quarter chord point of the airfoil.
2. Correct for tunnel wall effect and Aspect Ratio. Calculate Reynolds number.
3. Plot C_L , C_D and C_M versus α for (a) the wing alone (b) wing with slot and (c) wing with slot and flap.
4. Compare the measured results with NACA curves (available online) and discuss any differences. Estimate the possible combined reading error in balance and dynamic pressure measurements.

ADDITIONAL ASSIGNMENTS:

1. Measure influence of slat gap in increasing the maximum lift coefficient.
2. Measure influence of flap angle on the maximum lift coefficient.
3. Does the increase of lift coefficient of (1) plus the increase of (2) equal the increase due to both optimum slat and flap settings?

Experiment 6

▪ بيانات عامة:

..... PRESSURE DISTRIBUTION OVER AN AIRFOIL.. اسم التجربة:

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثاني

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات تحت الصوتية بملحقاته

▪ الأساس النظري للتجربة:

OBJECTIVES:

To measure the pressure distribution over an Airfoil at various angles of attack.

DISCUSSION:

An airfoil develops Lift at a positive angle of attack through generally lower pressures above the wing and higher below with respect to the pressure of the approaching air. The overall pressure distribution can be measured with small tubes embedded in the wing leading to a suitable pressure transducer.

The model is equipped with 18 pressure openings. The openings are located 0, 7.5, 10, 20, 30, 40, 50, 60 and 70% chord on both upper and

lower surfaces and there is an additional opening at 80% chord on the upper surface.

PROCEDURE:

1. Install the pressure wing vertically in the wind tunnel and connect the pressure tubes, in order, to the 24-tube multi-manometer. Dividing the pressure measured at any point on the airfoil by q provides the pressure coefficient at that point, $(p - p_{ref})/q = C_p$
2. Operate the tunnel at an airspeed of 130 mph and make pressure measurements at angles of attack of 0° and 4° (or as assigned).

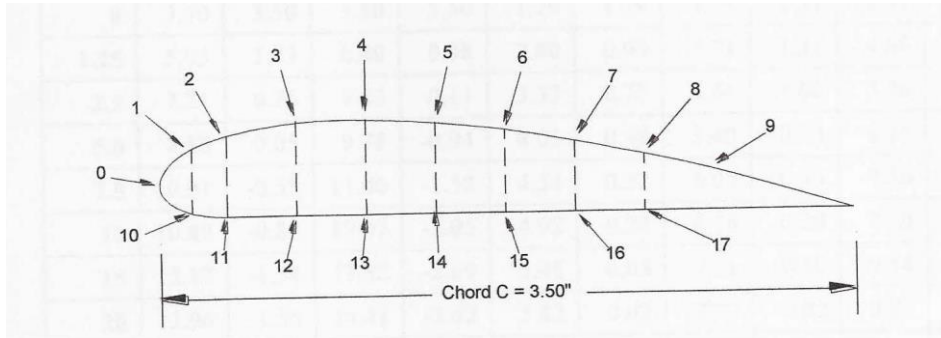
RESULTS:

1. Plot pressure coefficient to a suitable scale on the airfoil plots included in this manual (copies can be made for each student).
2. Plot pressure coefficients along the chord line and along a normal line (Ref. 1) and integrate to give Normal and Chordal Forces. (Use a planimeter, count squares on quadruled paper, or use Simpson's rule to integrate).
3. Determine Lift and Draft coefficients from (2) and compare with results of Experiment 5.
4. Note that this method does not measure drag viscous forces.

DATA SHEET FOR EXPERIMENT 6

Date.....

Test Section Airspeed.....



Station Number	% f chord	PRESSURE DATA			
		$\alpha_1 =$	$\alpha_2 =$	$\alpha_3 =$	$\alpha_4 =$
0	0				
1	5				
2	10				
3	20				
4	30				
5	40				
6	50				
7	60				
8	70				
9	80				
10	5				
11	10				
12	20				
13	30				
14	40				
15	50				
16	60				
17	70				

Experiment 7

■ بيانات عامة:

BOUNDARY LAYER ON A FLAT PLATE. اسم التجربة:

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثاني

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات تحت الصوتية بملحقاته

■ الأساس النظري للتجربة:

OBJECTIVES:

To measure the velocity profile in the boundary layer at several longitudinal stations on a flat plate and to estimate the location of transition.

DISCUSSION:

Because of viscous dissipation the total pressure P_t varies within the boundary layer, whereas the static pressure, p , remains closely equal to its value in the flow just outside the boundary layer. At any level in the boundary layer the dynamic pressure q is equal to $P_t - P$ so by measuring P_t with a total head tube (and using the constant value of p) q and, therefore, the velocity can be determined at that level. There are various

definitions of boundary layer thickness which are roughly equivalent. We suggest that the thickness be considered the distance from the plate to the point where the velocity reaches 99% of its free stream value.

The leading edge of the plate is curved in a parabolic manner rather than sharpened as in illustrations in many text books. The parabolic nose can be shown experimentally to increase the region of laminar flow.

PROCEDURE:

1. Total head readings are taken with a pressure probe with 10 open-ended tubes extending into the flow. Such a probe is often used for this purpose and is known as a boundary layer mouse. Fasten the mouse to the plate with the forward set of tapped holes in the plastic plate. The tubes are supported on small projections at approximately 0.018, 0.025, 0.030, 0.040, 0.060, 0.080, 0.100, 0.120, 0.160 and 0.200 inches from the plate. Connect the 10 tubes of the mouse (in order) to the pressure inlet nipples and connect the pressure from the static pressure ring of the tunnel to the transducer reference pressure connection.
2. Maintain airspeed of 20 mph and take total head readings at each of the positions defined by the tapped holes. Record the distances from the leading edge of the plate to the tube openings for each of these positions. Record the temperature of the air passing through the test section.

DATA REDUCTION:

1. Make individual plots of velocity versus height from the plate for each station and determine the boundary layer thickness, δ , at each station.
2. Plot velocity profiles in dimensionless form, y/δ versus u/V_∞ . Calculate local Reynolds number for each station. Estimate location and Reynolds number for transition based on the plots.
3. Assuming zero pressure gradient along the plate and uniform external velocity, the shearing stress is given by the Von Karman integral momentum equation:

$$\tau_o = \frac{d \left[\int_0^\delta \rho (Vu - u^2) dy \right]}{dx}$$

4. Approximate the derivative by taking the value of the integral (determined graphically) at successive stations and dividing by the distance between the stations (check units). Do this for the forward two stations and the last two stations.

Note: A low airspeed is recommended to form boundary layers of reasonable thickness in spite of the short distance from the leading edge of the plate.

Experiment 8

▪ بيانات عامة:

إسم التجربة: DRAG BY THE WAKE SURVEY METHOD.

الفرقة المقرر عليها التجربة: الفرقة الثالثة

الفصل الدراسي: الثانى

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات فوق الصوتية بملحقاته

▪ الأساس النظرى للتجربة:

OBJECTIVES:

To measure the drag coefficient of a wing by the wake survey method.

DISCUSSION:

Application of momentum principles indicates that the drag force on an airfoil in a flow should equal the reduction in linear momentum of the flow (in the drag direction) provided that the measuring stations are taken where the static pressures are substantially equal. Since the flow approaching the airfoil is uniform, the drag coefficient may be written in terms of a downstream wake survey (as developed in Ref. 1):

$$C_D = \frac{Y_w}{c} - \frac{c \int q dy}{q_o}$$

where Y_w , = width of wake

q = dynamic pressure in wake

q_o = upstream dynamic pressure

It is recommended that the wake survey be made as far from the trailing edge as possible to render static pressure effects negligible. These assumptions preclude the use in highly rotational flows such as those in the wake of an airfoil near the stall or with flaps deflected.

This method of drag measurement is often used on portions of airplane wings in flight to test special drag profiles or surface treatments.

PROCEDURE:

Install pressure wing vertically in the test section and the 18-hole total pressure rake downstream of the airfoil. Make wake measurements at $\alpha = 0^\circ, 2^\circ$ and 4° , or as assigned.

COMPARE:

The values obtained with corresponding readings from balance readings on the plain airfoil with proper consideration of aspect ratios with the organization.

APPENDIX I

SUPPLEMENTARY INFORMATION FOR REPORT WRITING

The SUMMARY Section:

NACA papers always begin with a summary, which is concise and accurate condensation of the entire paper. Although this section is characterized by briefness, the following items should be included: The object and scope of the work, the information obtained, and the main conclusions reached. Only essential ideas should be given and specific statements are of more value than generalities.

From the summary readers, indexers, the bibliographers should be able to obtain a fairly complete idea of the material contained in the paper. For this reason, the summary should be an independent unit; therefore, mention of equations, tables, and figures by number must be avoided, symbols must be adequately defined, and references must be identified other than by the number given in the list.

The INTRODUCTION Section:

The introduction serves as a preparation for the material to follow and relates the current work on the subject to the field. As much of the following material as is applicable should be included in any logical order:

1. The status of the problem prior to the present research
2. The purpose of the investigation precisely defined
3. The conditions under which the work was done and the procedure, if unusual

4. The scope of the present work and its connection with the general problem.
5. Recognition of similar work on the subject
6. Significance of the material treated

In addition, it may be desirable to state where and when the work was done. Such mention should occur in the introduction unless it is specifically included in a following section.

If valuable help has been received from a person not connected with the investigation, a brief courteous acknowledgment can be made at the end of the introduction; the person should be mentioned by full name and title. The organization with which the person is associated should be added after his name if the contribution was a result of his affiliation with the organization.

The DESCRIPTION OF APPARATUS Section:

For papers presenting experimental data this section is usually a brief but adequate discussion of the apparatus used, the material employed, the models tested, and the experimental setup. Unless the equipment is new or modified, suitable reference to a published description is satisfactory. Dimensions and descriptions of unmodified, permanent equipment should be kept in the present tense. Trade names of equipment or material, including aircraft and engines, may be used, if necessary, for identification and if no evaluation is presented. Trade names and designations should be carefully checked for correct form.

Any sketches or photographs of the equipment, setup, and tests in progress should be referred to in this section

The RESULTS Section:

A well-organized and objective presentation of the results should be given. Not only the results, but also the method of computation or derivation used to obtain them should be presented unless it is described in another section, for example, "Procedure." If the method is involved, one complete example may be included; however, if this example entails a lengthy computation or derivation, it may be put in an appendix and only the main steps indicated under "Results".

Tables and figures that show the results should be referred to in this section. A tabular form for the results is more useful if many readers might want to plot the results in a variety of forms; graphs are preferable in showing trends and comparisons. All statements about the results and any numerical values cited from them should agree precisely with the tables and figures.

In short papers the presentation of results may be combined with other sections, such as "Procedure" or "Discussion." The heading should be altered accordingly, for example, "Results and Discussion".

The DISCUSSION Section:

Discussion of the results, together with their analysis, to show that the conclusions are warranted is one of the most important parts of the paper. Each major conclusion should be clearly explained and comparisons should be made with results of similar work by other

investigators. If the results have an immediate application, this should be pointed out in the discussion and, if possible, an example should be worked out to show the method of application. All statements should be clear to readers who are in other fields of aeronautical science and may not be so well acquainted with the subject as the author.

The APPENDICES Section:

Related material desirable as supporting evidence but not essential to the development of the paper itself - material such as derivations - may well be placed in an appendix following the concluding section. Only in rare cases are groups of tables or figures presented as an appendix.

Appendices should be referred to in the text. This is because only material related closely enough to the text to warrant mention should be appended. For reference each appendix of a group should be identified as appendix A, appendix B, and so forth, but a single appendix need not be thus identified. In addition, an appendix should be given a title (e.g. "Section Characteristics of Miscellaneous Airfoils) whenever possible. Appendices should preferably be arranged in the order of their mention in the text. This order, however, may not be feasible if it is desirable to maintain a certain logical reference to a list of symbols, which is one of a group of appendices, is either first or last.

In typed papers each appendix begins on a new page. In order to save space in printed reports, however, an appendix follows continuously after the signature. The heading is set full measure and the text is set in two columns beneath it.

Experiment 9

■ بيانات عامة:

إسم التجربة: Supersonic Flow and Shockwaves

الفرقة المقرر عليها التجربة:

الفصل الدراسي:

الأدوات المطلوبة للتجربة: نفق هوائي ذو سرعات فوق الصوتية بملحقته

■ الأساس النظري للتجربة:

INTRODUCTION

The AEROLAB 1"x1" supersonic tunnel is of the sliding block type to provide a stepless variation of Mach number. With storage pressure of 200 psi (1379 kPa), the available pressure ratio limits the maximum Mach number to about 3.8.

The electrical power requirement 220V/3/50 Hz for the wind tunnel unit is approximately 300 W. The air compressor and dryer unit requires about 6 kW.

DEVICE MAIN COMPONENTS

The supersonic wind tunnel device consists of three main components:

- a) Air supply system,
- b) Tunnel,

c) Schlieren system.

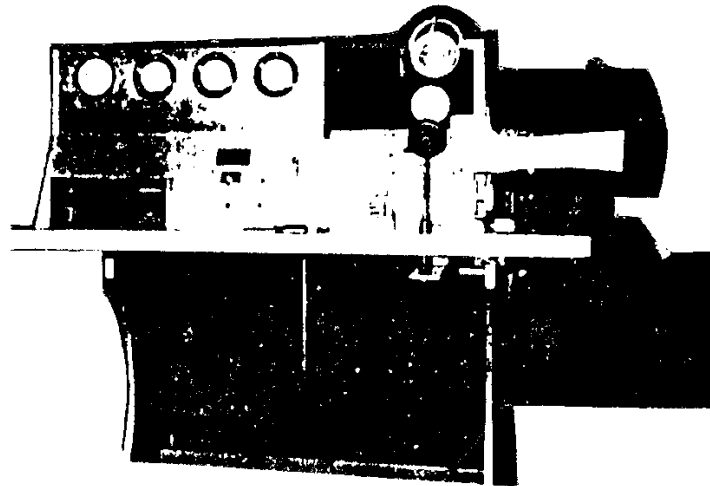
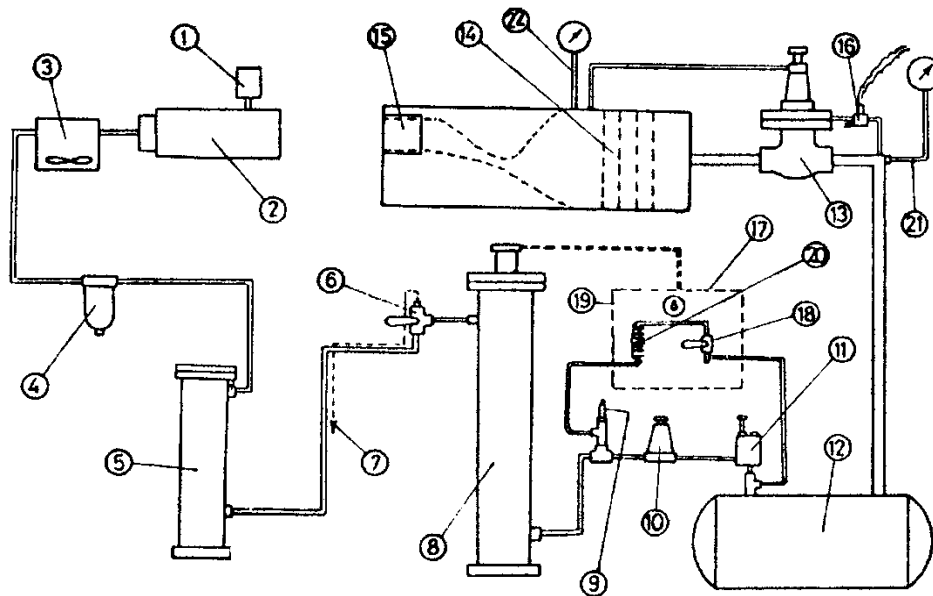


Fig. 1 Supersonic wind tunnel device

Air supply system

The air supply system consists of:

1. One 5 hp two stage Ingersoll rand type T-30 model 242 Air compressor (220V/3/ 50 Hz power requirement).
2. One air-cooled after-cooler.
3. One dryer tower with heating elements.
4. CASH back pressure valve FR-10 (set at 75 psi).
5. One compressor unloading valve.
6. One filter with automatic drain.
7. One pre-filter for oil.
8. Safety valves, fittings, dryer control panel and compressor starting unit and switches.
9. Two 50 ft³ air tanks (200 psig ASME code) interconnected with 2 inch pipe fittings.



- | | |
|--|---|
| 1- Motor 5 HP. | 2- Compressor ingersol rand model 242 (200 PSIG). |
| 3- After cooler. | 4- Filter & Automatic drain. |
| 5- Pre-Filter. | 6- Bleed valve. |
| 7- Steam bleed. | 8- Dryer tower. |
| 9- Safety Valve (200 PSI). | 10- Back pressure valve 0.5" (75 PSI). |
| 11- Unload valve for constant speed operation. | 12- Air reservoir. |
| 13- Leslie LAEKO regulating valve. | 14- Flow distribution plate & screen. |
| 15- Wind-tunnel test section. | 16- Solenoid valve. |
| 17- Dryer control panel. | 18- Purge valve on dryer control panel. |
| 19- Timer. | 20- Capillary tube coil for restricted flow. |
| 21- Storage pressure. | 22- Stagnation pressure. |

Fig. 2 Air supply and drying system

Tunnel

The tunnel, Fig. 3, consists of:

A) Stagnation Chamber:

The stagnation chamber contains a perforated steel plate in front of the air inlet to distribute the incoming flow. The air passes through two 20 x 20 screens and into the supersonic nozzle.

B) Test Section

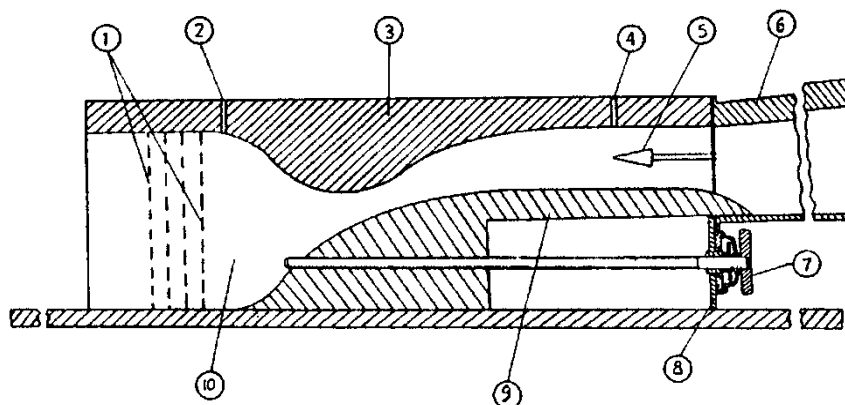
The test section is formed of two chord members and two 1 inch thick side plates. The side plates are surface ground on the inside, Fig. 3.

C) Nozzle:

The nozzle blocks are constructed of solid aluminum. The contours are machined and carefully hand finished. The upper block is sealed with a neoprene gasket, the lower with a teflon tube in the outer part of the groove and a rubber tube beneath to furnish resiliency. The teflon tubing provides a low coefficient of friction and good durability.

D) Model:

The center of the test section is located at the center of rotation of the angle of attack quadrant. The exact axial setting of the model is immaterial since the flow is uniform.



- 1- Screens.
- 3- Steel plate.
- 5- Model.

- 2- Stagnation pressure tap.
- 4- Static pressure tap.

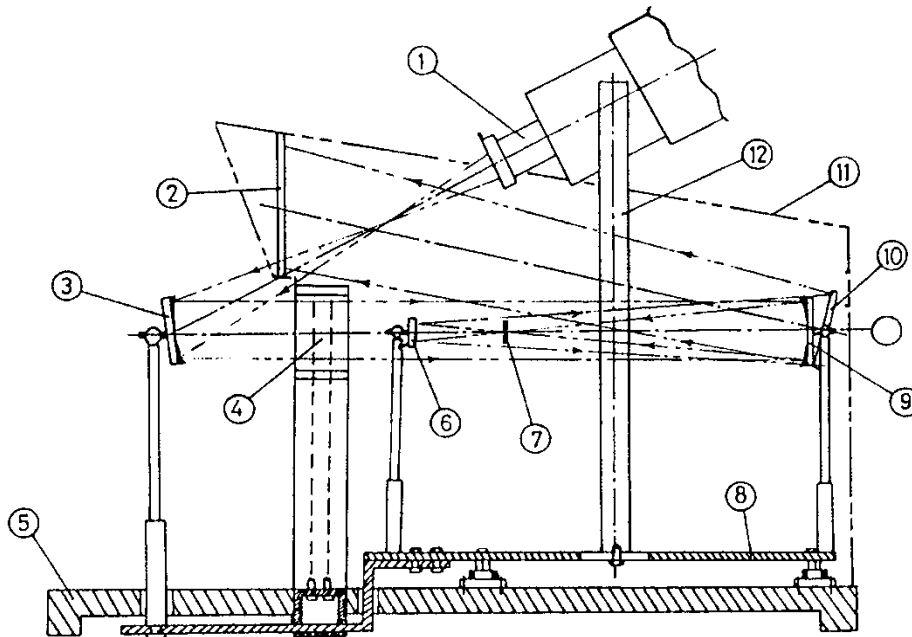
7- Hand wheel.
9- Slide block.

6- Diffuser.
8- Gear to counter.
10- Stagnation chamber.

Fig. 3 Tunnel

Schlieren System

The Schlieren system includes a high pressure mercury arc source, two three-inch diameter spherical mirrors, a knife edge, two flat first-surface mirrors and a viewing screen. The light source is the PEK type 112-5145 high pressure mercury arc lamp. The light source required a DC power supply with a high initial voltage for starting. The lamps have a nominal life of 100 hours.



1- Light source mercury vapor.
3- 3" mirror-spherical.
5- Counter top of 1" supersonic console table.
7- Knife edge.
9- 3" spherical mirror.
11 Schlieren system fiber glass

2- Viewing screen.
4- Test section.
6- First surface mirror 22 x 30 mm.
8- Schlieren system shock mounted plate form.
10- First surface mirror 39 x 48 mm.
12- Light source support post.

enclosure.

Fig. 4 Schlieren system

ADJUSTING THE OPTICS

The first step in adjusting the optics of the system is to focus an image of the light source on a hand-held card to locate the local point (which should be on the axis of the cylindrical light housing). If the image is too high or low, the mercury vapor lamp (after cools of) can be moved in its spring clips to position the light source on the axis of the condensing lens.

To obtain the necessary sensitivity of the Schlieren system for observation of shock waves and density gradients, the knife edge must be located at the focal point mirror C, Fig. 4.

OPERATION OF THE DRYER

Air is supplied from the compressor through the after-cooler to the filter water separator unit. The air supply line between the pre-filter and the drying tower is equipped with a bleed cock to make it possible to slowly reduce the inlet pressure to atmospheric prior to reactivation of the dryer. A slow reduction of the pressure in the dryer is desired because a rapid pressure change might blow the silica gel desiccant out of the dryer into the pipelines when the actuating handle is moved. The maximum air pressure supplied to the system not to exceed 200 psi (1379 kPa) and may be low as 100 psi (689 kPa) for satisfactory

performance. The storage tank is equipped with a safety valve, which is set up to open at 200 psi (1379 kPa). Two other pressure relief valves, one on the compressor and one just before the (red) Cash Back Pressure valve should be checked periodically. The AEROLAB air drying system is designed to remove moisture and oil from the air as efficiently as possible. Most of the oil and water is removed by compressing air and thereby condensing the water vapor content in approximately direct ratio with the absolute pressure. Since the absolute pressure is low in the storage tank directly after a run, a back pressure valve is placed between the air receiver and the compressor to maintain a minimum pressure of about 80 psi on the dryer at all times. After passing through the centrifugal separator, the air enters the top of the pre-filter (the short tank) which is filled with a desiccant, which absorbs oil in preference to water. The oil-free air then enters the top of the dryer (the long tank) and the water vapor is removed by passing through the silica gel bed. Finally, the air passes through the Cash Back Pressure valve and into the storage tank. After about a year of operation the desiccant in the pre-filter should be removed through the lower opening and then refilled through the upper diagonal pipe.

OPERATION OF THE DEVICE

The Mach number is set by turning the hand wheel to a counter reading determined from prior calibration as presented in Fig. 5.

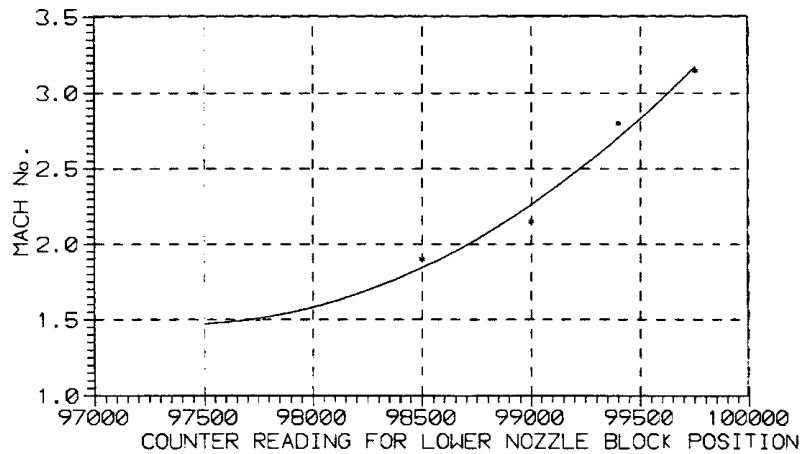


Fig. 5 Calibration of the Mach number by prior counter reading of the hand wheel

This moves the lower block in an axial direction to change the geometry of the nozzle. Stagnation pressure control, Fig. 6 is accomplished with a Leslie LAEKO regulating valve. The valve is located directly behind the front control panel with its pilot valve on top. An aluminum hand wheel on the pilot valve is used for pressure setting. In practice, the desired stagnation pressure for a run is determined from the enclosed curve of minimum stagnation pressure versus Mach number. The Veeder Root Counter which indicates axial position is set to 0 when the lower block is at the extreme inner (high Mach number) limit of travel. The tunnel has been tested at stagnation pressure up to 180 psi (1241 kPa). A small leakage of high pressure air past the seals of the lower block is normal considering the necessity of keeping seals free enough to permit sliding of the block.

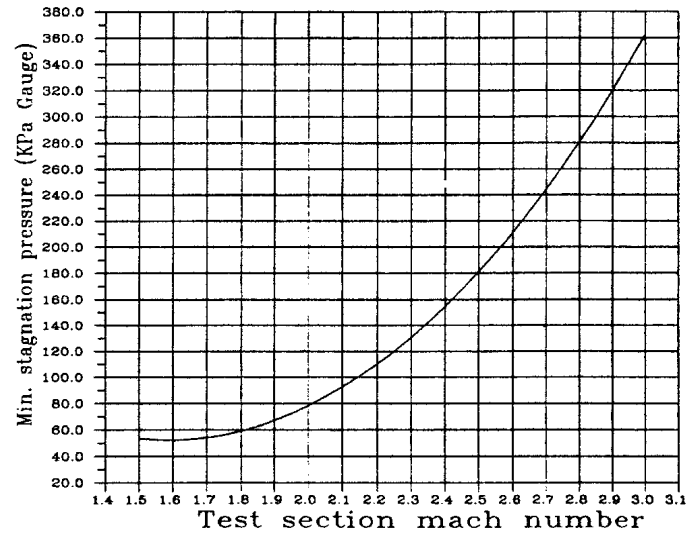
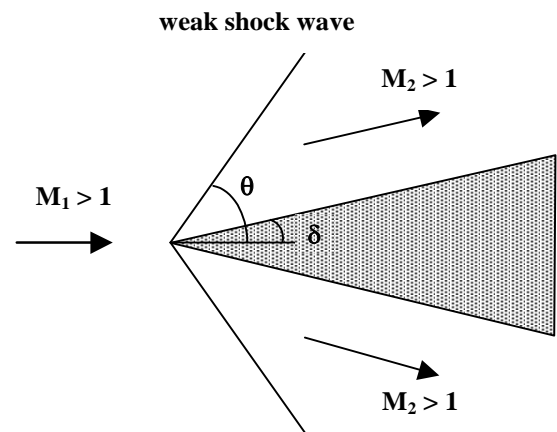


Fig. 6 Variation of the test section Mach number

OBLIQUE SHOCK WAVE

The downstream Mach number and the pressure ratio across the oblique shock wave, Fig. 5 are given respectively by the following equations:



$$M_2^2 = \frac{2 + (\gamma - 1)^2 M_1^2 \sin^2 \theta}{[2\gamma M_1^2 \sin^2 \theta - (\gamma - 1)] \sin^2(\theta - \delta)}$$

Fig. 7 Oblique shock wave

$$\frac{p_2}{p_1} = \frac{2\gamma}{\gamma + 1} M_1^2 \sin^2 \theta - \frac{\gamma - 1}{\gamma + 1}$$

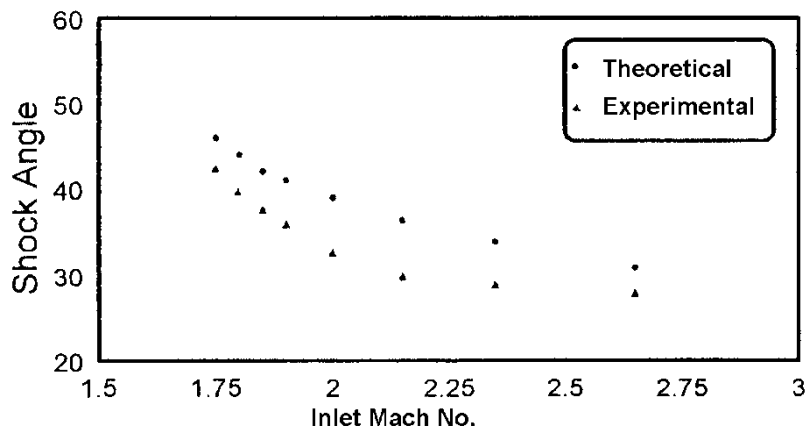
where γ : the specific heat ratio (= 1.4 for air),

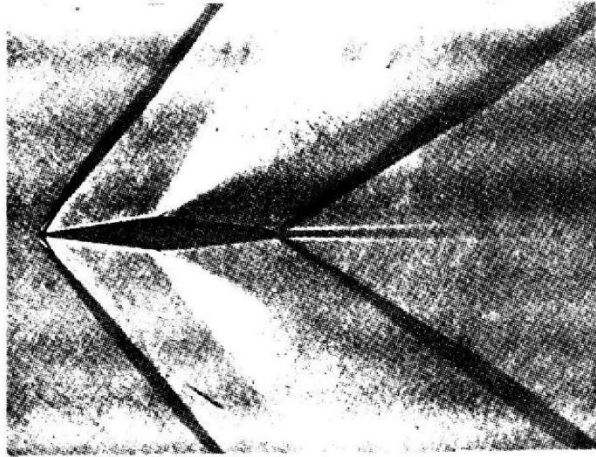
δ : the wedge angle,

θ : the shock angle.

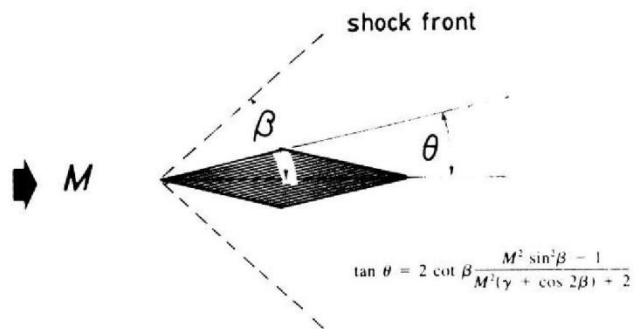
EXPERIMENTAL RESULTS

Run	EXPERIMENTAL						THEORETICAL	
	Counter reading	M_1	P_x	P_y	Shock angle θ	P_y / P_x	P_y / P_x	θ
1	97000	1.5	128	180	55	1.40	1.6	57
2	97750	1.75	56	74	44	1.31	1.56	46
3	98000	1.8	52.5	71.5	42	1.3	1.513	44
4	98250	1.85	46	57.2	38	1.25	1.77	42
5	98500	1.9	42	55.5	36	1.32	1.80	41
6	98750	2.0	30	55	32	1.8	1.86	39
7	99000	2.15	27	50	30	1.85	1.95	38
8	99250	2.35	24	45	29	1.875	1.98	34
9	99500	2.65	27	55	28	2.04	1.99	31





M = 1.6



Shock Waves with a Wedge