Introduction to Aerospace

Wind Tunnel Measurement on a Aircraft Model

1 Introduction

Wind tunnel testing is commonly employed during the later stages of the design process to characterise an aircraft's aerodynamic performance. Through use of dynamic similarity, accurate predictions for the aerodynamic forces and moments that the full size aircraft would encounter in flight can be made and the aerodynamic design can be further refined. To achieve dynamic similarity when testing transonic aircraft, cryogenic wing tunnels are often used. They can reach the required Reynolds and Mach numbers by featuring large test sections and operating at high pressure and cryogenic temperatures using nitrogen gas. Use of such facilities, while producing very accurate results, comes at a significant operating cost.

The objective of this experiment is to use a wind tunnel to characterise the aerodynamics of a business jet airplane by measuring the lift, drag and pitching moment acting on a scale model. The model will be tested at several incidences and three different tail configurations, in order for various aerodynamic quantities relating to its longitudinal static stability to be experimentally derived.

2 The Model

The model represents a 1:24 scale model of a twin-jet executive aeroplane, the full size version of which you will be flight testing using the Department's Flight Simulators. It has an all-moving tailplane without a separate elevator, so movements of the pilot's control column would alter the tail setting angle i_H. On the model, the tailplane is attached by a single pin whose location in a series of holes fixes the tail setting angle. The range of movement is from $+2°$ (most forward hole) to $-5°$ at intervals of $1°$. These angles are taken relative to fuselage datum line, not to the zero-lift line.

The relevant characteristics for the model can be found in table 1. The horizontal datum of the aircraft is the nose of the fuselage and the vertical datum is the fuselage centreline. The quarter-chord point of the tail's mean aerodynamic chord is 0.810 m aft and 0.2 m above the datum. The load-cell is situated 0.541 m aft and 0.11 m below the datum.

Table 1: Model Characteristics

	Unit	Wing	Tailplane
Span	m	0.873	0.388
Root chord	m	0.203	0.108
Chord at aircraft centre-line	m	0.215	0.108
Tip chord	m	0.081	0.057
Geometric mean chord	m	0.147	0.082
Mean aerodynamic chord	m	0.158	0.085
Aspect ratio		5.93	4.74
Gross area	m^2	0.128	0.032
Dihedral (at trailing edge)	°	2	0
Geometric Twist	°	-3	0
Aerodynamic Twist	°	-3.5	0
Root sections		NACA 64_2-415	NACA 63_1-A012
Tip sections		NACA 64_2-412	NACA 63_1-A012

3 Experimental Setup & Procedure

The model is mounted on a sting forward of a 6-axis force/moment balance in the manner shown in figure 1. Its roll, pitch and yaw of the model can be varied by tilting the model about the point of the load-cell. The model is mounted onto the string at a small positive incidence, hence the geometric angle of attack of the model

$$\alpha_g = \theta + 0.8^\circ \tag{1}$$

where θ is the sting's pitch angle.

The force balance measures the forces and moments generated by the model about the centre of the load-cell. For this laboratory we will only be considering the load-cell horizontal (F_x, taken as positive backwards) and vertical (F_z, taken as positive upwards) axial forces as well as the pitching moment (M_c). Note that these forces are the sum of aerodynamic and gravitational forces and the orientation of the load cell is defined by the sting pitch angle (relative to the wind tunnel).

The tunnel is of the closed-section, closed-return type, driven by a 30 kW motor. The working section is 1.6m x 1.1m, at atmospheric pressure and the maximum speed is about 40 m/s. The tunnel speed is controlled by a crude controller which will set the approximate motor driving frequency for any requested target velocity. For this experiment, it will be run at about 25 m/s. The latter quantity is derived from the measurements of a Pitot-static tube attached on the tunnel floor, to the side of the model. Readings for the wind-tunnel speed are available through the digital interface provided.

To estimate the air density in the tunnel, the pressure and temperature of the air in the tunnel can be also be read from the micromanometer. The readings are provided by a

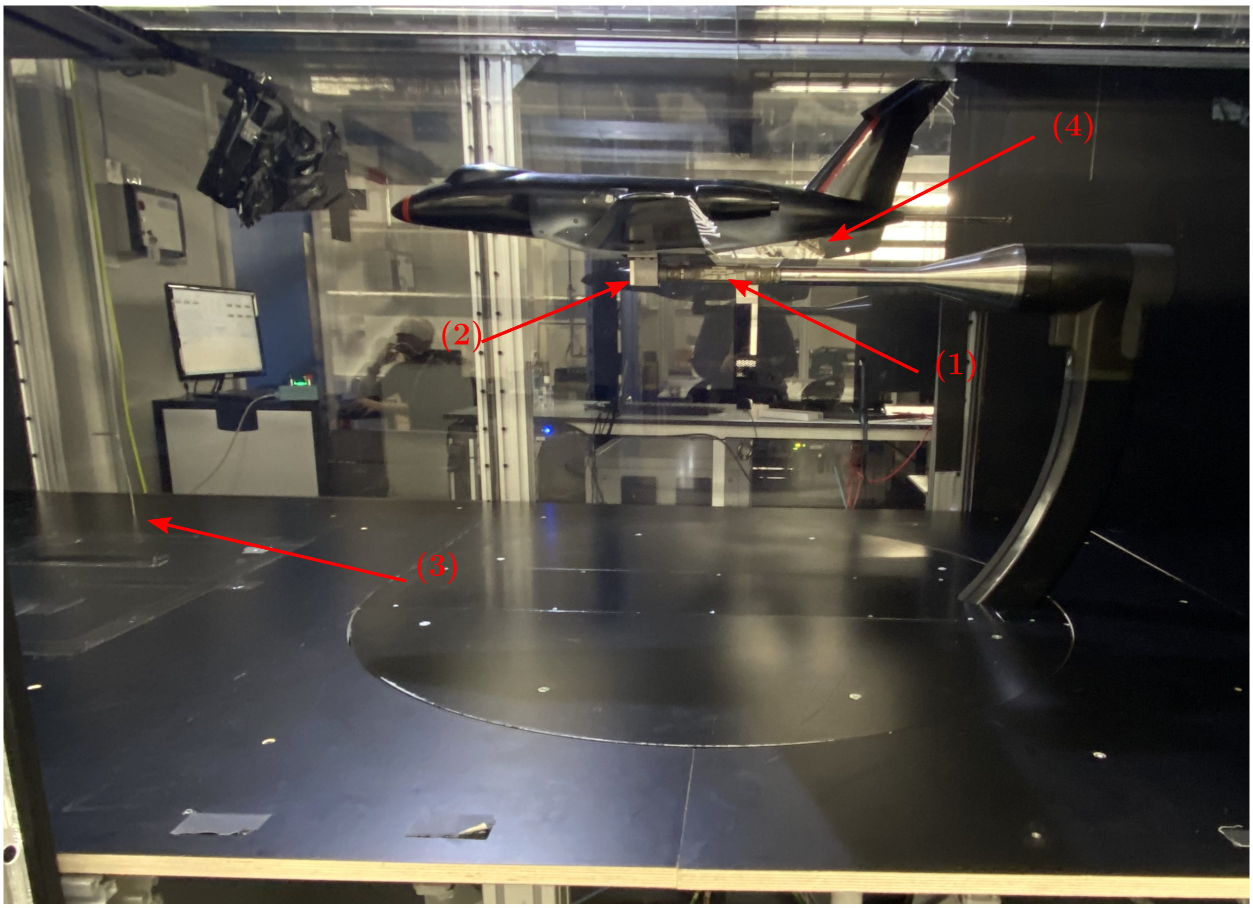

Figure 1: Picture of the experimental setup showing (1) the location of the load-cell, (2) the supporting strut (placed inside a fairing for the experiment), (3) the Pitot-static tube, and (4) the tailplane angle setting pins.

temperature probe inside the tunnel's working section and an atmospheric pressure sensor (barometer) located just outside the tunnel.

Procedure

1. Record the tunnel air temperature and the barometric pressure. Possible variation of temperature and pressure should be monitored throughout the lab session.

2. With the tailplane removed, check that the tunnel doors are closed and set the sting incidence to 0 degrees ($\theta = 0°$).

3. Ensuring that the airspeed in the tunnel is null, set the balance readings to zero (tare), in order to record the correct measurements for every model configuration.

4. Turn on the wind tunnel and set the fan frequency such that a tunnel airspeed of 25 m/s is achieved. Slowly increase the motor frequency setting so as not to exceed required speed setting.

5. Set your model to the first angle of attack to be tested ($\alpha_g = -2°$ or $\theta = -2.8°$). When the wind speed is correctly adjusted, record the wind-on lift, drag and pitching moment. The measurement is given as a 5 second average of the readings collected by the load cell. The standard deviation associated with that average is also provided as an indication of the variation/fluctuation of the force & moment readings, and can be used to estimate the precision of your readings.

6. Set the model incidence to the value required for the next measurement.

7. When you have taken measurements over the entire geometric angle of attack range (-2° to 12°, in steps of 2°) with the tailplane off, turn off the tunnel. Once the airspeed has reduced enough, open the tunnel door and fit the tailplane at $i_H = +1°$ (second hole counting from the front). Make sure you do not apply excessive force on the model, as that might damage the load cells. Repeat the previous procedure from step 2 onwards.

8. After taking the measurements over the same incidence range with $i_H = +1°$, stop the tunnel and fit the tailplane at i_H = -2° (fifth hole from the front). Repeat the measurement procedure for all incidences between -2° and 12°.

9. For the $i_H = -2°$ case you should increase the incidence beyond the maximum of 12° to attempt to measure lift, drag and pitching moment as the aircraft approaches the stall. To do so it is necessary to go up to higher model incidences, about 17°. Make sure that the tail strut attachment on the force balance does not touch the floor at these incidences. If the model starts to vibrate, turn off the wind tunnel and lower the incidence to avoid damaging the load cell.

10. Finally return the incidence to 0° and tell the demonstrator you have completed the experiment.

Throughout your tests observe, by means of the wool tufts attached to the aircraft, and significant features of the flow about the model such as:

- The onset of the wing stall and its spread across the wing
- The location of the wing wake in relation to the tail
- Stalling of the tail
- Other significant events

4 Data Analysis

4.1 Estimating Aerodynamic Forces

The force and moment measurements taken represent the combined effect of aerodynamic loads and the weight of the model. Furthermore they have all been measured in the sting's frame of reference. This section provides guidance on how to correct for the model's self-weight and present aerodynamic forces in the wind-tunnel's frame of reference and moments about a nominal centre of gravity position $x_{cg} = 0.426$ m aft of the model datum and along its centreline.

Model Self-Weight

If the model mass is W_m, and its centre of gravity is located a horizontal distance Δx_m and vertical distance Δz_m from the centre of the load-cell, then the contributions of self-weight to the forces and moments measured by the load-cell will be

$$\begin{aligned} F_{x,m} &= W_m \sin\theta \\ F_{z,m} &= -W_m \cos\theta \\ M_{c,m} &= \Delta z_m F_{x,m} + \Delta x_m F_{z,m}. \end{aligned} \tag{2}$$

The mass properties of the model are dependent on whether the model is being tested tail-on or tail-off, and can be found in Table 2. Assuming that the load-cell has been zeroed at $\theta = 0°$, the aerodynamic loads $(F_{x,a}, F_{z,a}, M_{c,a})$ at the load-cell can therefore be estimated by

$$\begin{aligned} F_{x,a} &= F_x - W_m \sin\theta \\ F_{z,a} &= F_z + W_m(\cos\theta - 1) \\ M_{c,a} &= M_c - W_m \left[\Delta z_m \sin\theta + \Delta x_m(\cos\theta - 1)\right]. \end{aligned} \tag{3}$$

Table 2: Mass properties for the wind-tunnel model

	Tail-off	Tail-on
Weight (W) N	70.26	74.65
x-offset (Δx_m) m	-0.0667	-0.0529
z-offset (Δz_m) m	0.0690	0.0763

Changing Frames of Reference

For further analysis, aerodynamic forces must be expressed in the wind frame of reference, as lift and drag, and aerodynamic moments relocated to the aircraft's nominal centre of gravity position. Based on the dimensions given above, that nominal position is 0.115 m forward and 0.11 m above the load cell. Therefore the pitching moment is given by

$$M = M_{c,a} - 0.115F_{z,a} - 0.11F_{x,a} \tag{4}$$

and the Lift and Drag forces are given by

$$L = F_{z,a}\cos\theta - F_{x_a}\sin\theta \tag{5}$$

$$D = F_{x,a}\cos\theta + F_{z_a}\sin\theta. \tag{6}$$

Aerodynamic Coefficients

The lift, drag and pitching moments coefficient may then be derived in the usual way:

$$C_{L_{tu}} = \frac{L}{qS}; \qquad C_{D_{tu}} = \frac{D}{qS}; \qquad C_{M_{tu}} = \frac{M}{qS\bar{c}} \tag{7}$$

where q is the dynamic pressure recorded, S is the aircraft's reference area, and the suffix "tu" means "tunnel values", which will have to now be further corrected for the aerodynamic effects of the tunnel walls.

NOTE: It is strongly recommended that you prepare an Excel sheet to use for data entry that applies the methods outlined in section 4.1 to your measurements of F_x, F_z and M_c, such that you can sanity check your measurements based on your knowledge of how lift, drag and pitching moment should vary with angle of incidence.

4.2 Wind Tunnel Corrections

Solid Blockage

These values first have to be corrected for "solid blockage". The presence of the model and its supporting struts reduces the local cross-sectional area of the air-stream. The local velocity is therefore greater than the "far upstream value" and the "tunnel values" of the coefficients are correspondingly exaggerated. A correction factor is therefore required. The correction factor is a function of the overall blockage ratio and of the cross-sectional shape of the model, and it has been computed to be approximately 0.94 in this case, by the methods of [1]. The correction factor is to be applied to all three measured quantities, i.e. lift, drag and pitching moment.

Drag due to support

The drag measured in the tunnel further include the drag of sting upstream of the load cell and interference effects between the model and the sting fairing. Strictly, a separate experiment should be carried out to determine the appropriate correction. In the present case an estimated correction can be applied.

$$\Delta C_{D,\text{sting}} = 0.022. \tag{8}$$

This correction was derived based on the theoretical drag of the fairing and is quite large. Given more time it would be essential to establish a more accurate figure through further experimentation.

Wall Effects on Lifting Bodies

Since the tunnel working section has solid walls, the streamlines of the flow at some finite distance from the model are constrained to be straight, whereas this would not be the case in free air. In addition to solid blockage, an important effect is that the downwash at the model is altered. To analyse the problem, a flow must be devised in which the streamlines coincide with the tunnel walls: such a flow is produced by the actual trailing vortex system of the model wing plus a system of image vortices outside the tunnel. The image vortices produce an upwash at the model, thus reducing the induced downwash relative to the free-air value for a given lift coefficient. In free air, the incidence corresponding to a given C_L would therefore be greater than the tunnel value and the induced drag would be higher, due to the rotation of the lift vector. Depending on the method used (Refs [1], [2] or other computational methods), various estimates of the corrections can be obtained. A compromise figure is used here:

$$\alpha = \alpha_g + 0.83C_L \tag{9}$$

where α is the angle of attack which would have produced these results if there were no tunnel wall, and all angles are in degrees. The corresponding increase in drag coefficient in free air would be:

$$\Delta C_{D_i} = 0.015C_L^2 \tag{10}$$

The upwash at the tail due to the system of image vortices is greater than that at the wing, in the present case by about 50%. Since an incidence correction has already been applied to the whole model, only the additional upwash at the tail influences the pitching moment. This is equivalent to an increase in tail setting angle and hence the tail lift coefficients recorded in the tunnel are larger with respect to free-air conditions. Consequently, the pitching moments measured in the tunnel are also larger than in free air (in the wind tunnel reference frame) by an amount proportional to the wing lift coefficient. The model in the tunnel therefore appears to be more stable than in free air.

The additional upwash has an effect on the tailplane C_L that can be quantified as:

$$\Delta\varepsilon = 0.5 \times 0.83C_L = 0.415C_L \tag{11}$$

where ε is expressed in degrees. The correction to the pitching moment coefficient is thus approximately

$$\Delta C_M = -\Delta\varepsilon \frac{\partial C_M}{\partial i_H} = -0.415 C_L \left(\frac{\partial C_M}{\partial i_H}\right) \tag{12}$$

where the negative sign is present beacuse of the downwash. This correction requires a value of $\frac{\Delta C_M}{\Delta i_H}$ (substantially the same as $\frac{\partial C_M}{\partial i_H}$) from the curves to be plotted later. Note that this latter quantity must be per degree and is negative.

There are various other corrections which could be applied, described in [1] and [2]. However, they are treated as negligible in the present case and have not been computed.

4.3 Reduction of Results

The following properties of the aircraft can be determined through this experiment:

(a) The trimmed lift coefficient at the two tail settings with the aircraft CG assumed to be in a realistic position, and the corresponding static margins and neutral points.

(b) The aerodynamic centre position of the tail-less aircraft (x_W).

(c) The lift curve slope of the wing-fuselage combination, without the tailplane, a_W, and the complete aircraft (with tail), a, where

$$a = a_W + \eta_H \frac{S_H}{S_{ref}} a_H \left(1 + \frac{\mathrm{d}\varepsilon}{\mathrm{d}\alpha}\right). \tag{13}$$

(d) The effective tail lift curve slope, $\eta_H a_H$.

(e) The variation of the downwash at the tail with incidence, $\mathrm{d}\varepsilon/\mathrm{d}\alpha$.

(f) The aircraft's Oswald efficiency factor for each tail configuration.

Note that in order to derive (e) at the two trimmed lift coefficient, (b), (c) and (d) must also be obtained at these values of C_L, since the various curves you will obtain are not necessarily the neat, straight lines shown in the diagrams below. However, the fit should be computed in the linear region.

To derive these parameters proceed as follows:

1. Plot C_M against C_L for the three configurations. In this and subsequent plots, draw faired curves through the points, do not just draw mean straight lines. If, at a given C_L, the values of C_M at the two tail settings $i_{H,1}$ and $i_{H,2}$ are $C_{M,1}$ and $C_{M,2}$ respectively, then:

$$\frac{\partial C_M}{\partial i_h} = \frac{C_{M,2} - C_{M,1}}{i_{H,2} - i_{H,1}} \tag{14}$$

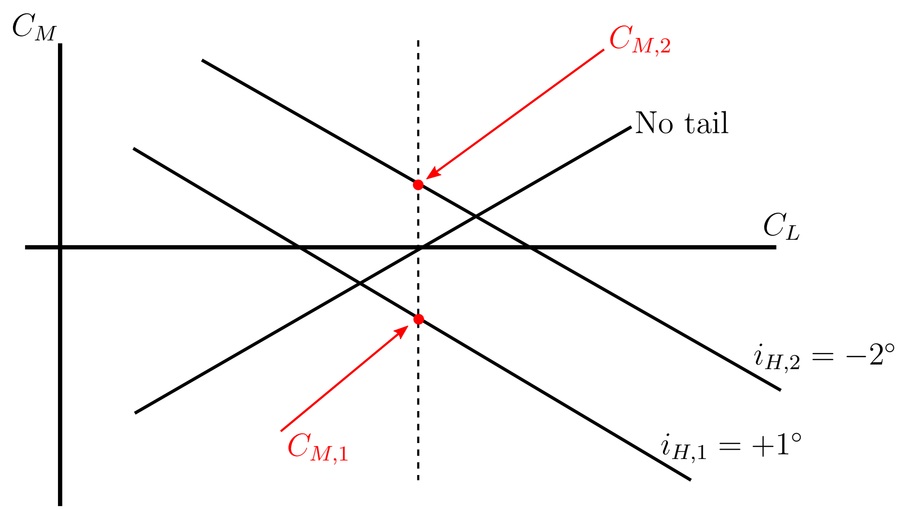

Figure 2: C_M against C_L.

See figure 2 for reference. A mean value for a C_L of about 0.5 will suffice for this computation. Then, from (14), the correction ΔC_M to be added to all the "tail-on" values of C_M can be found as

$$C_{M,cor} = C_M - 0.415 C_L \left(\frac{\partial C_M}{\partial i_h} \right) \tag{15}$$

Now tabulate $C_{M,cor}$ as a function of C_L for the two tailplanes settings and plot the corresponding curves. Use these curves to identify the C_L at which each tail setting angle would result in trimmed flight, i.e. $C_{M,cor} = 0$. Since

$$K_n = -\frac{\partial C_{M,cg}}{\partial C_L} \tag{16}$$

the slopes of the curves at these two points give the corresponding values of the stick-fixed static margin K_n.

The corresponding neutral point position x_{np} can be found from the above values of K_n, as

$$K_n = \overline{x}_{np} - \overline{x}_{cg}. \tag{17}$$

2. Find the slopes of the C_M - C_L curve (tail off) at the incidences corresponding to the trimmed C_L's tail-on. Since moments are taken about the nominal cg point, the pitching moment coefficient can be expressed as the sum of a zero lift moment (C_{M_0}) and moment generated by the lift force generated at the aerodynamic centre,

$$C_M = C_{M_0} + C_L \frac{x_{cg} - x_W}{\overline{c}} \tag{18}$$

Hence, differentiating and solving for x_W we find

$$x_W = x_{cg} - \overline{c}\,\frac{\partial C_M}{\partial C_L} \tag{19}$$

so the position of the wing/fuselage aero centre x_W is easily determined from the measured slopes.

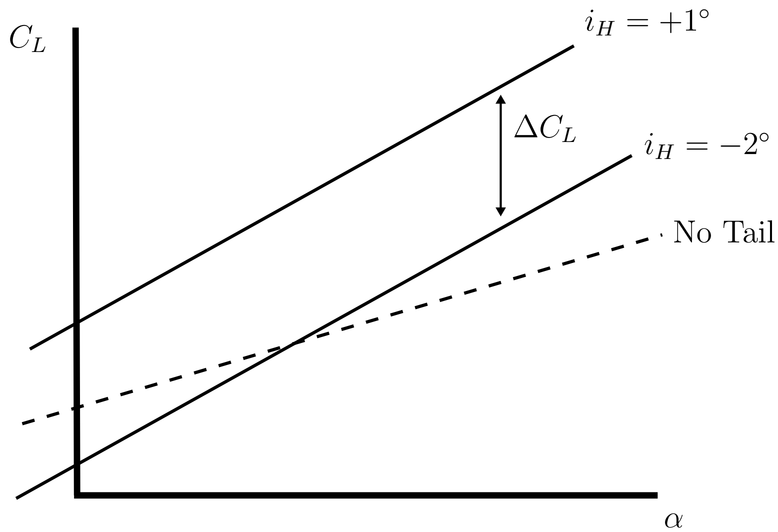

Figure 3: Trimmed C_L curves.

3. Plot C_L against true (not geometric) incidence α for the three configurations. Find the lift coefficient at stall $C_{L_{max}}$ for the i_H = -2° case. Further find the slope of each "tail-on" curve at the lift coefficient for which $C_{M,g}$= 0 at the tail settings. These slopes are the appropriate values of a. Also, find the slope of the "tail-off" curve, a_W, at the two trimmed lift coefficients. See figure 3 for reference.

4. Since

$$C_L = C_{L_{0W}} + a_W\alpha + \frac{S_H}{S}C_{L_H} \tag{20}$$

and

$$C_{L_H} = \eta_H a_H \left[\alpha\left(1 - \frac{d\varepsilon}{d\alpha}\right) + i_H\right] \tag{21}$$

the difference in C_L at a given α for the two tail settings is

$$\Delta C_L = \eta_H a_H \left(\frac{S_T}{S}\right)(i_{H1} - i_{H2}) \tag{22}$$

Hence a_H can be found at each trimmed C_L.

5. In order to find the tail volume coefficient $\overline{V}_H$ the distance $\bar{l}_H$ between the aerodynamic centre of the aeroplane-less-tail x_W and the aerodynamic centre of the tail ($x_H = 0.810$ m) is required. If the aerodynamic centre is x_W aft of datum then, from the dimensions of the model:
$$l_H = x_H - x_W \tag{23}$$
Hence
$$\overline{V}_H = \frac{S_H l_H}{S_{ref}\bar{c}} \tag{24}$$
Now, from the definition of the neutral point,
$$x_{np} = x_W + \eta_H \overline{V}_H \frac{a_H}{a}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\bar{c}, \tag{25}$$
with all other parameters having been calculated, $\frac{d\varepsilon}{d\alpha}$ can be found for each case.

6. Plot C_D against ${C_L}^2$ for the three configurations. According to the simplest theories, the plots should be straight lines. The intercept of a line on the C_D axis will be the zero-lift drag coefficient $C_{D,0}$ and its slope will be $\frac{1}{\pi ARe}$. Find the mean slopes at moderate lift coefficients and hence calculate the model's Oswald efficiency e for each configuration. Also, estimate the values of C_{D_0}.

References

[1] Rae, W.H. and Pope, A. "Low speed wind tunnel testing", Wiley, 1984

[2] Pankhurst, R.C. and Holder, D.W. "Wind tunnel technique", Pitman, 1952