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by

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SUMMARY

The AEDC 10° cone has been tested in a large number of American transonic wind tunnels to investigate the influence of free stream pressure fluctuations on transition Reynolds numbers. Measurements made on the AEDC cone in three RAE wind tunnels are described in this Report.

In the RAE 8ft × 8ft subsonic/supersonic tunnel the cone transition Reynolds numbers were relatively high and the pressure fluctuations low. In the RAE 3ft × 4ft high supersonic speed tunnel the transition Reynolds numbers were high and there were strong unit Reynolds number effects. In the RAE 8ft × 6ft tunnel transition was controlled by an effective roughness, either at the cone tip or the microphones; the tunnel pressure fluctuations varied appreciably with Mach number and were high at subsonic speeds.

The results show that surface pressure fluctuation measurements at supersonic speeds are sensitive to the degree of smoothness of the surface, particularaly in the transition region of the boundary layer.

A tentative classification of the aerodynamics facilities used for transition tests is suggested.

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* Replaces RAE Technical Report 76077 - ARC 37395

CONTENTS

1 INTRODUCTION

There is considerable interest in defining the characteristics required for the new generation of transonic tunnels[1], and this has inevitably involved some re-appraisal of the characteristics of existing facilities, both in Europe and America. One characteristic of great importance in a facility is the rms level and the frequency spectrum of the free stream pressure fluctuations, for these can influence both dynamic and static measurements at transonic speeds (see the evidence provided in Refs.2 and 3, or in chapter 1 of Ref.1). A comprehensive set of measurements on a single model tested in a large number of facilities would be of considerable intrinsic interest.

In a collaborative experiment, initiated by the staff of AEDC Tullahoma, the pressure fluctuations close to the centre line of many American transonic tunnels have been measured on the surface of a highly polished 10^{o} cone. Although few spectra have yet been published, some of the rms pressure fluctuations, $\bar{p}/q$, are available in a preliminary survey[4]. It has been found that an increase in the coefficient of the rms pressure fluctuation generally reduces the transition Reynolds numbers.

It is hoped that the same cone will be attached to a USAF aircraft and test flown at subsonic, transonic and supersonic speeds. From previous experiments in flight on other models the pressure fluctuation level should be much smaller than in the wind tunnels (see Fig.4 of Ref.5) and thus higher transition Reynolds numbers are expected. Thus the results of the flight tests should provide an interesting comparison with the extensive tunnel measurements.

The cone was tested in several European tunnels in 1973 (see Table 1) with the twin objectives of defining the level of pressure fluctuations and determining the transition Reynolds numbers. The present paper outlines the main results of the tests in three RAE wind tunnels, comments on their significance and suggests a tentative classification of aerodynamic facilities used for transition tests.

2 EXPERIMENTAL DETAILS

2.1 AEDC 10^{o} transition cone

Fig.1 shows the AEDC 10^{o} transition cone rigged in the RAE 8ft × 6ft transonic tunnel during the final test of the European tour. The surface of the cone was polished to 0.2 to 0.3μm rms waviness and was maintained without difficulty during the test series. In contrast the tip of the cone was fragile, and its maintenance posed serious problems. The tip was already slightly bent on its arrival at Bedford for the first series of tests, but because the representative from AEDC considered that the tip had the same small distortion during the tests in American tunnels, no attempt was made to correct it.

Standard Brüel and Kjaer 6.36mm (0.25in) diameter microphones with a frequency range from 30Hz to 100kHz were provided on different generators ($\phi = 180^{\circ}$ and 225°) at $x = 457$mm (18in) and $x = 660$mm (26in) to measure the surface pressure fluctuations (Fig.2). It was difficult to ensure an absolutely smooth surface across the cone, the microphones and their surrounding sleeves, when the microphones were inserted.

Full details of the microphone calibration were given in the section of Ref.4 entitled 'background noise'. The microphone signals were recorded on magnetic tape at typical conditions, and the total broadband signals required to calculate the rms pressure level, $\bar{p}/q$, were obtained from rms meters. The time constant of these meters was varied between 3s and 10s to ensure steady readings, the larger time constant being essential when turbulent bursts were occurring in the transitional boundary layer above the microphones. 20 to 30s samples of the microphone signals were also recorded on magnetic tape* for typical conditions, including where possible:

laminar boundary layers at both microphones,

a transitional boundary layer at the forward microphone with a turbulent boundary layer at the aft microphone,

and

turbulent boundary layers at both microphones.

2.2 Detection of transition

Throughout these tests the microphones were used to detect the onset and completion of transition (called respectively R_t and R_T). In this unusual method of detecting transition the microphone signals were measured as a function of unit Reynolds number (R) at constant Mach number. For laminar boundary layers the measured rms pressure fluctuations, $\bar{p}/q$, correspond with the tunnel noise level, because the pressure fluctuations generated by a very thin laminar boundary layer are small[6], and outside the frequency range of the microphones. For transitional boundary layers the rms pressure fluctuations increase rapidly

* Copies of these tapes are held at RAE Bedford for subsequent analysis.

to a peak (as turbulent spots occur intermittently within the laminar signal), and then fall to an almost constant level for a fully turbulent boundary layer (Fig.3). At subsonic and transonic speeds this constant level was generally at about the same level as the laminar signal (Fig.3), probably because the pressure fluctuations caused by the thin turbulent boundary layers were still outside the frequency range of the microphones. However at supersonic speeds the level with a turbulent boundary layer was considerably higher than the laminar level (Fig.4). We shall see that a subsequent investigation (section 3.4) suggested that some of this increase at supersonic speeds might be attributed to the increased sensitivity of the microphone signals to how badly the microphone and its surrounding sleeve matched the cone surface.

For tests to evaluate the technique, the microphone measurements may be compared with measurements from a small surface pitot tube. Thus Fig.3 includes transition data obtained at transonic speeds from a surface pitot fixed on a generator of the cone at $x = 457$mm, $\phi = 0^o$ (i.e. the same streamwise distance as the forward microphone). This surface pitot was made from 1.2mm hypodermic tube which was flattened at the nose. The measurements of onset and complete transition with the fixed surface pitot agree broadly with those from the forward microphone on the generator at $x = 457$mm and $\phi = 225^o$. Some differences may be attributed to the lack of symmetry of the flow on the cone (see discussion of Fig.10).

Originally it had been intended to detect transition with the traversing surface pitot used during the American tests[4]. However, this traversing surface pitot did not perform well in the initial tests in the RAE 8ft × 8ft tunnel (the difficulties are described briefly in the Appendix) or in subsequent tests during the European tour*. Hence the traversing surface pitot was not fitted for tests in either the RAE 3ft × 4ft or 8ft × 6ft tunnels.

2.3 Test conditions

All the measurements were made at constant free stream Mach number (M) over a range of unit Reynolds number, at wall temperatures close to the adiabatic recovery temperature.

3 RESULTS

3.1 RAE 8ft × 8ft tunnel

Fig.5 shows the transition Reynolds numbers measured in the RAE 8ft × 8ft tunnel (from Tables 2 and 3), plotted against the free stream rms pressure

* Similar difficulties were previously encountered in the NASA Langley 8ft transonic pressure tunnel in the forward position of the traverse gear[4].

fluctuations, $\tilde{p}/q$, taken from under the laminar boundary layer. These free stream pressure fluctuations were independent of unit Reynolds number. The microphone measurements (Fig.5a) are limited and insufficient to establish if there is a simple correlation between the transition Reynolds number for onset, R_t , and $\tilde{p}/q$, as suggested in Ref.4, although they were consistent with such a correlation. The traversing surface pitot measurements (Fig.5b) are restricted to $M = 0.20$ and 0.80. These transition Reynolds numbers also correspond reasonably well with those measured in American transonic tunnels[4].

The transition measurements obtained from the microphone and the traversing surface pitot were not sufficiently detailed to establish if there is a unit Reynolds number effect on the AEDC cone in this tunnel. Such an effect had been observed at supersonic speeds on another cone previously tested in this tunnel (section 4).

These tests confirm previous measurements that indicated that the level of pressure fluctuations in the working section is low at subsonic speeds (Fig.46, Ref.3).

3.2 RAE 3ft × 4ft tunnel

Fig.6a shows the variation of the transition Reynolds numbers R_t, R_p and R_T with Mach number and unit Reynolds number derived from curves such as those shown in Fig.4. In general the variations in R_p and R_T with Mach number are comparatively small, whereas at a given Mach number the variation with unit Reynolds number is large. Thus over the Mach number range from $M = 3.0$ to 4.5, at a unit Reynolds number of 10×10^7/m, we have:

for peak pressure fluctuations

$$R_p = 4.4 \times 10^6 \quad \text{to} \quad 5.4 \times 10^6$$

and for complete transition

$$R_T = 5.2 \times 10^6 \quad \text{to} \quad 5.8 \times 10^6 \ .$$

The variation in transition onset is much less well defined but at a unit Reynolds number of 5×10^7/m it varies from about

$$R_t = 1.4 \times 10^6 \quad \text{at} \quad M = 3.0$$

to

$$R_t = 2.7 \times 10^6 \quad \text{at} \quad M = 4.5 \ .$$

Fig.6a also includes some measurements (shown by the dashed curves) of transition made with surface hot films on the flat plate used for boundary layer experiments in this tunnel[7]. The measurements on the flat plate vary strongly with unit Reynolds number, just as on the cone. However the Reynolds numbers are not the same on both models and change relative to each other with Mach number. Some differences may arise in defining transition in both experiments. Some differences must also be anticipated because of the differences in surface finish (the flat plate is not highly polished) and between the boundary layer development in two dimensions on the plate and three dimensions on the cone. The table inserted in Fig.6a also shows that as Mach number increases so apparently does the level of the rms pressure fluctuations, $\bar{p}/q$, for a laminar boundary layer. However these measurements are probably too high owing to the protuberance of the microphones from the surface of the cone (see section 3.4). These free stream pressure fluctuations were independent of unit Reynolds number, just as in the RAE 8ft × 8ft tunnel.

Fig.6b shows that the ratio of the transition Reynolds numbers between the cone and the flat plate falls as Mach number increases both for transition onset, $(R_t)_c/(R_t)_p$, and complete transition, $(R_T)_c/(R_T)_p$. This behaviour is similar to that observed in previous tests in supersonic wind tunnels[8]. The ratio for transition onset is within the range of 3 to 1 calculated for incompressible boundary layers[9]. A recent review[10] based on stability theory shows that the cone to plate transition ratio can lie in the range from 3 to somewhat less than 1, and is dependent on the free stream disturbance level, both at subsonic and supersonic speeds. Although the flat plate has a small leading edge radius (which would slightly increase the transition Reynolds numbers by lowering the effective unit Reynolds numbers) its surface is not highly polished, which would reduce the transition Reynolds numbers.

Fig.6c shows that the measured Reynolds numbers for complete transition are about 70% of the predictions* given by the Pate and Schueler correlation[11], possibly because of the tip distortion. Kendal showed[12] that at supersonic speeds from M = 2.2 to 4.8, a small offset of only 0.05mm on the tip of a 20° total angle cone 360mm long, reduced the transition Reynolds number on a generator well behind the rearward facing step to 64% of that measured on a generator with

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* The values of C_F and δ^* required for the predictions according to equation (2) (section 4) were estimated using Ref.13. This reference gave good estimates of C_F and δ^* for the RAE 8ft × 8ft tunnel at M = 1.4 and 2.4.

no surface discontinuity. Hence tip distortion could well have been a significant factor in the present tests.

A brief investigation was made to establish the variation in transition Reynolds number with angle of incidence at supersonic speeds. The microphone measurements, used to define the transition region and made at a Mach number of 4.5, showed that quite small angularities could significantly alter the shape of the curves of $\bar{p}/q$ as a function of R_x. Thus the Reynolds number for peak pressure fluctuations was reduced by about 0.4×10^6 from $\alpha = 0^\circ$, to $\alpha = -0.09^\circ$ (Fig.7a). The peak pressure fluctuations were used to assess the incidence effects because R_p was more clearly defined than either R_t or R_T (onset or complete transition). Fig.7b shows a comparison of the pressure fluctuations $\bar{p}/q$ as a function of the angle of incidence at constant Reynolds number. The first set of measurements is taken close to transition onset ($R_x = 3.6 \times 10^6$ at $\alpha = 0^\circ$) and has a peak at $\alpha = -0.5^\circ$. The second set of measurements is taken in the middle of the transition region ($R_x = 4.5 \times 10^6$ at $\alpha = 0^\circ$) and has a peak at $\alpha = -0.20^\circ$. Thus both sets of data given in Fig.7 show that a decrease in the angle of incidence decreases the Reynolds number for peak pressure fluctuations, and hence decreases the transition Reynolds number along the generator considered.

Similar tests over the Mach number range from M = 4.5 to 3.0 may be represented for $-0.8^\circ \leqslant \alpha \leqslant 0^\circ$ by the approximation

$$R_p\alpha/R_{p_0} \triangleq 1.0 - 0.6(\alpha^\circ)$$

as shown in Fig.8, or expressed in terms of the cone-semi angle θ_c

$$R_p\alpha/R_{p_0} = 1 - 3(\alpha/\theta_c) \quad . \qquad (1)$$

Equation (1) agrees reasonably well with transition measurements[12] on a 20° total angle cone at M = 2.2; in all these tests relatively large changes in tip radius altered the transition Reynolds number at zero incidence (R_{p_0}) without significantly altering the slope of the $R_p\alpha/R_{p_0}$ v. α/θ_c curves (Ref 12, Fig.14).

Fig.8 includes a prediction of the effect of incidence on transition on cones at supersonic speeds[14] which agrees fairly well with the measurements. This approximate theory is based on the hypothesis that the angle of incidence effects boundary layer transition only through its influence on the thickness of the laminar boundary layer. This approximate theory gave good agreement with experiments on a 25° total angle cone at a free stream Mach number of M = 3.0.

For completeness Fig.8 also includes two points from Fig.11 measured at M = 0.80 and 1.19 in the RAE 8ft × 6ft tunnel, but these points will be discussed later.

3.3 RAE 8ft × 6ft tunnel

Fig.9a shows that the free stream rms pressure fluctuations on the cone increase from $\bar{p}/q = 0.0075$ at M = 0.3 to a maximum of about $\bar{p}/q = 0.018$ close to M = 0.60. Most of the increase occurs at two peaks in the spectrum of pressure fluctuations. These peaks are caused by the rotation of the fan, which is immediately downstream of the working section. The levels of these peaks in terms of $\sqrt{nF(n)}$ at the primary* and secondary frequencies are also plotted in Fig.9a but these levels are taken from previous measurements on a slender body (see Fig.43a of Ref.3). At Mach numbers of 0.70 and above the tunnel spectra are much flatter (see Fig.43b of Ref.3). This is reflected in the present cone measurements by the fall from $\bar{p}/q = 0.009$ at M = 0.80 to $\bar{p}/q = 0.005$ at M = 1.2.

Fig.9b summarises the transition data derived from the microphone measurements in the RAE 8ft × 6ft tunnel over the Mach number range from 0.30 to 1.19. The rapid forward movement of the transition front observed suggests that surface imperfections may act as an effective roughness, and that for these tests there is manifestly no correlation between the transition Reynolds numbers and the rms pressure fluctuations.

Some idea of the roughness height, k , required may be obtained by assuming a critical roughness Reynolds number[15,16].

$$R_k = 600 \quad .$$

The critical unit Reynolds number is about

$$R = 5.5 \times 10^6/\text{m}$$

at M = 0.80 and 1.19 so that if the roughness is close to the apex

$$k = 600/5.5 \times 10^6 = 0.11 \text{ mm} \quad (0.004\text{in}) \quad .$$

No imperfections of this height could be observed at the nose although there was a definite asymmetry as mentioned previously (section 2.1).

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* Primary frequency = number of blades × fan revolutions per second.
Secondary frequency = 2 × primary frequency.

Another possibility considered was that the microphones acted as roughness elements. A photograph taken after the test showed that the forward microphone projected about 0.25mm from the cone surface. With $R_k = 600$, this would be estimated[16] to cause transition at this microphone at $R_x = 2.5 \times 10^6$, which is of the correct order. Another photograph showed that although the rear microphone was flush with the cone surface, the sleeve around it was recessed up to about 0.50mm and this could have influenced transition. This microphone was a replacement installed before these tests. The original microphone had failed towards the end of the tests in the RAE 3ft × 4ft tunnel, and may not have had such a large recess.

To investigate if the tip or the microphones were fixing transition, flow sublimation tests were made with atmospheric static pressure in the working section, so that observers in the plenum chamber could monitor the development of the sublimation patterns in the naphthalene, which was sprayed on the model. The Mach numbers selected were M = 0.18 and 0.40; the time taken to develop the patterns was about 5min and 1min respectively. The contrast of the patterns developed against the highly polished surface of the cone was not sufficiently good to be photographed satisfactorily but Fig.10 is based on sketches.

At both test conditions the natural transition front was about 125mm from the apex. A turbulent wedge extended through the laminar region from a tiny piece of cotton (used for cleaning the model), which adhered about 60mm downstream of the apex. As expected from the forward position of the natural transition front, no turbulent wedges were observed from either microphone, from the surface pitot or from a discrete roughness element applied just upstream of the forward microphone. Hence it appeared that the tip was fixing transition. Interpolating between these two Mach numbers we may infer that at M = 0.30 and atmospheric static pressure in the working section

$$R \simeq 8 \times 10^6/\text{m}$$

and

$$R_T \quad \text{about} \quad 0.9 \times 10^6 \ .$$

If this point is added to the curve for M = 0.30 in Fig.9, we find a similar rapid forward movement of transition as observed at M = 0.80 and 1.19.

A third sublimation test was made with acenapthene (in which the pattern developed more slowly compared to naphthalene), at a Mach number of 0.80 and a unit Reynolds number for which the transition front swept rapidly forward. The forward microphone was still indicating peak pressure fluctuations similar to those in the previous run, so it is likely that the acenapthene deposit itself did not alter the transition front significantly. The transition front viewed from the TV monitor was ragged and ill-defined, and is not sketched here, but it extended between the microphone positions at $x = 457$mm and $x = 660$mm . Natural transition was observed about 25mm downstream of the forward microphone, which projected about 0.25mm from the cone surface and a turbulent wedge was formed behind a discrete roughness element just upstream but on a different generator*.

The other side of the cone could not be seen on the TV monitor and was photographed by a remotely controlled camera. This revealed a turbulent wedge from the surface pitot (immediately opposite the forward microphone), and a ragged transition front, which along one generator appeared to extend about 900mm from the apex. Hence although this sublimation test is also broadly consistent with the transition data derived from the microphone measurements, it also suggests that the transition front was not uniform because of either the bent tip of the cone or a small flow inclination. In retrospect, it might have been wiser to provide a small finite radius on the nose, thus ensuring a less fragile tip without greatly altering the boundary layer development. It may prove difficult to maintain the sharp tip during the flight tests.

The transition Reynolds numbers measured on this cone in American wind tunnels at transonic speeds were sensitive to flow angularity; hence a brief investigation of this effect was included in the tests in the RAE 8ft × 6ft tunnel. The Reynolds number for peak pressure fluctuations, R_p , was selected as the prime indicator of the effects of flow angularity, as in the RAE 3ft × 4ft tunnel at supersonic speeds. Test Mach numbers of 0.80 and 1.19 were chosen because at these speeds the peak pressure fluctuations were well defined at both microphone positions.

Fig.11 shows that at $M = 0.80$ a change in flow angularity from 0^o to -1.0^o has only a small influence on the peak pressure fluctuations. At the forward microphone the Reynolds number for peak pressure fluctuations falls from $R_x = 2.15 \times 10^6$ to 2.00×10^6 $(R_{p\alpha}/R_{p0} = 0.93)$. Similarly at the aft microphone (data not shown) the Reynolds number falls from $R_x = 3 \times 10^6$ to

* The development of these wedges was clearly visible on the TV monitor; at this condition the working section static pressure was sub-atmospheric and precluded direct viewing by observers in the plenum chamber.

2.85×10^6 (R_{p_α}/R_{p_0} = 0.95). This reduction is thus about 6% for 1^o, and is of the same order as measured in American wind tunnels.

A remarkable contrast is shown by the measurements at M = 1.19 (Fig.11) where a change in the flow angularity from 0^o to -1.0^o has a much larger influence. Thus at the forward microphone the Reynolds number for peak pressure fluctuations falls from $R_x = 3.1 \times 10^6$ to 2.45×10^6 (R_{p_α}/R_{p_0} = 0.79). Similarly the Reynolds number for transition onset falls from $R_x = 2.25 \times 10^6$ to 1.80×10^6 (R_{t_α}/R_{t_0} = 0.80). At the aft microphone (data not shown) the Reynolds number for peak pressure fluctuations falls from $R_x = 4.1 \times 10^6$ to $R_x = 3.1 \times 10^6$ (R_{p_α}/R_{p_0} = 0.74). Similarly the Reynolds number for complete transition falls from $R_x = 5.1 \times 10^6$ to 4.0×10^6 (R_{T_α}/R_{T_0} = 0.79).

It should be recalled that on this model during these tests at M = 0.80 and 1.19 transition is characterised by a rapid forward movement at a critical unit Reynolds number. Over the frequency range from 20Hz to about 10kHz the tunnel unsteadiness spectrum does not change much between M = 0.80 and 1.19 (from previous sidewall measurements, given in Ref.3, Fig.43b), so that there is probably no essential change in the nature of the disturbances entering the boundary layer. Hence the radically different behaviour of the transition with respect to variations in flow angularity between M = 0.80 and 1.19 must be attributed to some other mechanism, and this behaviour may be observed in other transonic facilities.

The points taken from Fig.11 and plotted in Fig.8, together with the previous measurements, indicate that there is a strong effect of Mach number on the variation of transition with angle of incidence within the range from M = 0.80 to 2.5. This has been confirmed by more recent measurements in the NASA Langley 4ft supersonic wind tunnel (Ref.17, Fig.4).

3.4 Peak pressure fluctuations caused by transition

A comparison of Figs.3 and 4 shows much higher peak pressure fluctuations on the cone at supersonic speeds than at subsonic speeds indicating that this method of determining transition became more sensitive as Mach number increased. However the increase in peak pressure fluctuations with Mach number is so large as to cause doubts about the validity of the measurements. A level of pressure fluctuations as high as $\bar{p}/q = 0.10$ would normally only be found in the presence of a strong perturbation in the flow, e.g. in the reattachment region of a bubble or under a shockwave at transonic speeds[18] and, if genuine, it would be important for the design of aircraft structures. Hence a brief review was made of other measurements of peak pressure fluctuations in the transition region, and these measurements are shown in Fig.12. (It has been assumed that the peak pressure fluctuations are uncorrelated with the pressure fluctuations measured with a laminar boundary layer.) Thus Fig.12 shows

$$(\bar{p}/q) = \sqrt{(\bar{p}/q)_p^2 - (\bar{p}/q)_t^2}$$

where $(\bar{p}/q)_p$ = uncorrected peak pressure fluctuations
and $(\bar{p}/q)_t$ = pressure fluctuation at onset.

Previous measurements on a much larger cone at AEDC[19], with a large microphone at $x = 1.15$m show that the peak pressure fluctuations increase rapidly from

$$\bar{p}/q = 0.02 \quad \text{at} \quad M = 3.0$$

to

$$\bar{p}/q = 0.10 \quad \text{at} \quad M = 4.0$$

thus corroborating the increase measured by both large microphones between $M = 3.0$ and 4.0 in the present tests (Fig.12). Similarly flight measurements with a large microphone on the fin of the X-15 aircraft[20] gave peak pressure fluctuations of about

$$\bar{p}/q_e = 0.04 \quad \text{at} \quad M_e = 3.7$$

and

$$\bar{p}/q_e = 0.05 \quad \text{at} \quad M_e = 4.1 \; .$$

(However it was noticed after these tests that the microphones were not flush with the surface on the fin.)

In marked contrast, measurements on a cone fitted with small microphones[21] gave levels of only

$$\bar{p}/q = 0.004$$

from $M = 4$ to 8. Similarly, measurements with a large microphone on a flat plate at low speeds[22] with thick boundary layers gave only

$$\bar{p}/q = 0.005$$

in fair agreement with the present tests in the RAE 8ft × 8ft tunnel at subsonic speeds ($\bar{p}/q = 0.005$).

This review showed no clear trend for the variation of peak pressure fluctuations with Mach number, but suggested that the higher values measured might well be associated either with the diameter of the microphone, relative to the boundary layer thickness, or with how well the microphones were fitted. To verify these possibilities, some brief tests were made with small Kulite pressure transducers (1.6 and 2.4mm diameter) on a 7° total angle cone in the ARA 2.25ft × 2.50ft supersonic tunnel[23]. The tests covered the Mach number range from M = 1.8 to 3.0 where a rapid increase in peak pressure fluctuations might have been expected from the present tests and the review of previous measurements.

The new measurements, included in Fig.12 show a small reduction from only

$$\bar{p}/q = 0.005 \quad \text{at} \quad M = 1.8$$

to

$$\bar{p}/q = 0.003 \quad \text{at} \quad M = 3.0 \ .$$

These levels are consistent both with the supersonic measurements of Martelluci *et al.*[21] and the low speed measurements of Blackman[22], and thus suggest that the microphone diameter relative to the boundary layer thickness, or the degree of flushness, may influence the measured pressure fluctuations. The microphone diameter relative to the boundary layer thickness did not appear to be very important, because the peak pressure fluctuations, quoted above, were virtually identical with natural and fixed transition. Hence the large pressure fluctuations within the transition region should be attributed to the degree of flushness of the surface, which would create localised shock waves and separations around the protuberances.

Within the ARA experiment it was not easy to vary the diameter of the pressure transducer or to vary the degree of protuberance from the surface. However, to establish the order of magnitude of the effect of a badly fitted pressure transducer or microphone, small rectangular strips of sellotape 0.1mm thick were stuck to the cone just downstream of the pressure transducers, leaving the transducers otherwise unaltered. Fig.13 shows that at M = 2.4 these steps caused a progressive increase in the pressure fluctuations for laminar, transitional and turbulent boundary layers. Although the increase is fairly small for the laminar boundary layers, the increase is large in the transition region, being a factor of about 3.7 at M = 2.4. Fig.14 shows similar results for M = 3.0, but here the peak pressure fluctuations have increased by a factor of about 4.5. These results illustrate the extreme sensitivity of the measurements at supersonic speeds to how well the pressure transducers are fitted, and thus the high peak pressure fluctuations shown in Fig.12 between M = 2.5 and 4.5 can be readily explained. A recent investigation[24] shows that large errors in surface pressure fluctuations at supersonic speeds are also caused by transducers protruding into thick turbulent boundary layers.

The measurements given in Figs.13 and 14 suggest that the particular surface imperfections tested do not generally alter the streamwise Reynolds number, R_x , for peak pressure fluctuations. Hence the indications of transition given in Figs.13 and 14 with the steps are probably correct, although the peak pressure fluctuations are incorrect, and considerably higher than they would be for an unperturbed, transitional boundary layer. Thus there is a good chance that the transition Reynolds numbers deduced from the microphones in the tests of the AEDC 10° cone (Figs.5a, 6 and 9) are correct, even if some of the measured peak pressure fluctuations at supersonic speeds are incorrect (Fig.12). Fig.13 also suggests that the level of $\bar{p}/q$ shown in Fig.5a for the laminar boundary layer at M = 2.4 could be too high.

4 SOME REMARKS ON POSSIBLE CORRELATIONS OF TRANSITION MEASUREMENTS

The correlation between the transition Reynolds numbers on the cone and the rms pressure fluctuations achieved in the American tests in perforated transonic tunnels[4] deserves careful consideration because of the possible implications for tests of other models. The correlation is somewhat surprising because of the known differences between the pressure fluctuation spectra of different tunnels and the differing directions of wave propagation (e.g. some waves may propagate downstream from the settling chamber, while others may propagate upstream from the diffuser or transversely from the sidewalls). However, the authors of Ref.4 suggested that if the amplitude of the pressure fluctuations greatly exceeds the limits implied by the linear stability theory (which predicts the growth of infinitesimally small disturbances in particular ranges of frequency), then the frequency content and the direction of wave propagation might be relatively unimportant, at least for the main part of the transition process, the nonlinear growth of turbulent spots. Some evidence from low and high speed flows which might support this hypothesis is now presented.

McDonald[25] has developed a method to predict the boundary layer development on a flat plate at low speeds from the laminar to the turbulent flow as a function of the rms free stream turbulent level, u^1/u , without any specification of the turbulence spectrum. McDonald comments "This analysis places little emphasis on the frequency of the disturbance and only the mean disturbance energy is considered important. This is obviously an oversimplification in certain instances. However, it is noted experimentally that, for instance, when the free stream turbulence intensities are greater than about ¼%, the resulting transition locations from a wide range of tunnels, measured by various experimenters over many years, depends solely on the free stream turbulence energy level and apparently not on its frequency content. At turbulence levels of less than ¼%, it appears that a maximum transition Reynolds number for a given tunnel can be achieved (due to an acoustic phenomenon?) and further reduction in the free stream turbulence level is ineffective. The present analysis does not reflect this cut off phenomenon and the predictions at these low turbulence levels must be regarded as upper limits of the transition Reynolds number." It is reasonable to argue that a rather similar phenomenon may occur when transition is provoked by pressure fluctuations, and turbulence levels are extremely small. This method of predicting boundary layer transition has recently been extended to hypersonic speeds[26].

Wygnanski *et al.*[27,28] have recently repeated Reynolds classic pipe flow transition experiment in air at a low Mach number of M = 0.06. Fig.15, based on Fig.2b of Ref.27, shows that at a free stream turbulence level above about 5% the mode of transition is by 'puffs', which propagate from the free stream towards the walls of the pipe. In contrast for free stream turbulence levels below about 0.5% the mode of transition is quite different, namely the development of an instability region (slugs) close to the walls which propagate towards the centre line of the pipe. For this mode of transition the shape of the turbulence spectrum is important. For the mode of transition typified by the 'puffs' the shape of the turbulence spectrum is relatively unimportant (see the discussion of Fig.13 of Ref.27). This change in the mode of transition is compatible with the correlation of the transition measurements on the cone in terms of rms pressure fluctuations above certain levels without any specification of the spectrum.

The work of Wygnanski *et al.* suggests a further problem. If the modes of transition for a pipe flow experiment are so different for low and high turbulence levels, a similar phenomenon may occur within boundary layers*. Hence the so called 'unit Reynolds number effect' might well be completely different at low

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* It is interesting to recall that theoretically the pipe flow is stable to all small disturbances, whereas in contrast a boundary layer is unstable to small disturbances of particular frequencies (see the discussion in Ref.29).

turbulence levels from what it is at high turbulence levels. This hypothesis of at least two different modes for transition, one appropriate to low turbulence levels (or low pressure fluctuation levels) the other to high turbulence levels (or high pressure fluctuation levels) may well explain some of the inconsistencies between flight or range experiments and wind tunnel experiments at transonic speeds. In a range the level of pressure fluctuations is extremely small and the first mode of transition must be appropriate. In contrast, in most transonic wind tunnels the level of pressure fluctuations will generally be about three orders of magnitude higher ($\bar{p}/q > 0.01$) so that the second mode of transition may be appropriate. Hence in any attempt to correlate range and wind tunnel experiments at transonic speeds it would be essential to specify the mode of transition as well as the level and spectrum of the pressure fluctuations. Any attempt to extrapolate wind tunnel measurements to ranges is doomed to failure unless the same mode of transition is duplicated in both facilities. This observation recalls Morkovin's warning about the possibilities of 'multiple responsibility' and 'dominant responsibility' of flow features promoting transition[30,31].

The Pate and Schueler empirical correlation of transition measurements on sharp slender cones in wind tunnels has been widely used at supersonic speeds[11,8] over the Mach number range from M = 3 to 14. The transition Reynolds number for complete transition measured on a sharp cone on the tunnel centre line is given[8] by

$$R_T = \frac{48.5(C_F)^{-1.4}\left[0.8 + 0.2\left[\frac{C_1}{C}\right]\right]}{\left[\frac{\delta^*}{C}\right]^{0.5}} \qquad (2)$$

where C_F = mean turbulent skin friction coefficient on the tunnel wall

δ^* = boundary layer displacement thickness on the tunnel wall

C = circumference of tunnel working section

and C_1 = circumference of the reference working section = 1.22m (4ft).

Equation (2) expresses the hypothesis that the measured transition Reynolds numbers are influenced by the turbulent boundary layer on the walls of the tunnel. Thus the parameters C_F and δ^* are implicitly related with the level and scale of the pressure fluctuations generated by the turbulent boundary layer. For an adiabatic turbulent boundary layer on the walls of the 8ft tunnel the average skin friction coefficient C_F , and the local skin friction coefficient C_f , are related over a wide range of Reynolds number and Mach number by the approximation (Table 3, Ref.32)

$$C_F = 1.2C_f \quad . \tag{3}$$

A similar relation should apply for other supersonic nozzle boundary layers. Hence the rms pressure fluctuations generated by these turbulent boundary layers may be related to the local skin friction coefficient by the approximation

$$\bar{p}/q = \text{constant} \times C_f \tag{4}$$

with the constant roughly established[33] as about 2.5 at moderate supersonic and subsonic speeds. Hence combining equations (3) and (4) we find

$$\bar{p}/q \doteq 2C_F \quad . \tag{5}$$

Now the previous experiments on cones at supersonic speeds show (Fig.9, Ref.8) that the Pate and Schueler correlation applies over the range in mean skin friction coefficient from

$$0.5 \times 10^{-3} < C_F < 2.5 \times 10^{-3}$$

which corresponds with

$$1 \times 10^{-3} < \bar{p}/q < 5 \times 10^{-3} \tag{6}$$

according to equation (5). This range of predicted rms pressure levels corresponds quite well to the pressure fluctuations measured in Pate and Schueler's original paper at Mach numbers of $M = 3.0$ to 5.0 in which the correlation between radiated pressure fluctuations from the wall and transition was first convincingly demonstrated (see the curves marked 'shroud removed' in Figs.5 and 6 of Ref.11). The boundary layer thickness ratio δ^*/C appears in equation (2) to represent, in a general way, the scale of the turbulent boundary layer eddies relative to the circumference of the tunnel. Alternatively, we may regard δ^* as the parameter controlling the frequency content of the turbulent boundary layer pressure fluctuation spectrum, which is broad band in character[34] but has a maximum at about $\sqrt{nF(n)} = 0.003$ at a frequency parameter

$$f\delta^*/U = 0.2 \tag{7}$$

at low speeds (Fig.16). There is some uncertainty as to whether equation (7) applies at supersonic speeds, but it should still give the order of magnitude of the peak frequency, which will thus be about several k cycles. Equation (4) does not, of course, include the low frequency pressure fluctuations characteristic of the tunnel design rather than the wall turbulent boundary layers. These low frequency pressure fluctuations are included in the rms pressure fluctuations measured by the microphones, which are accordingly larger (see Table 2) than the values given by equation (4).

The values of the parameters C_F and δ^* used in equation (2) often have to be estimated for supersonic wind tunnels. The estimates can be somewhat uncertain because of the powerful effects of the strong, favourable pressure gradients in the nozzle on the turbulent boundary layer development, or even the possibility of reverse transition from turbulent to laminar flow for a region close to the throat. However these uncertainties do not occur in the RAE 8ft × 8ft tunnel because the local skin friction coefficient C_f, and boundary layer displacement thickness δ^*, have been measured in the working section for a wide range of Mach numbers and unit Reynolds numbers[32]. These values, together with the approximation given by equation (3), have been used to estimate the Reynolds number for complete transition, R_T according to equation (2) for both subsonic and supersonic speeds. The results are tabulated below and illustrated in Fig.17.

Mach number M	Unit Reynolds number $R/m \times 10^{-6}$	Skin friction $10^3 C_f$	Displacement thickness ratio $10^3 \delta^*/C$	Complete transition $R_T \times 10^{-6}$ Predicted	Complete transition $R_T \times 10^{-6}$ Measured
0.2	7.2	1.77	1.84	5.2	4.7
0.6	6.9	1.76	1.80	5.3	4.6
0.8	7.9	1.65	1.82	5.7	5.1
1.4	9.8	1.54	1.85	6.2	6.4
2.4	8.4	1.28	2.28	7.2	5.7

Fig.17 shows that the measured and predicted values of R_T are in fair agreement, except at M = 2.4 where the measured value is significantly lower than the predicted value just as in the RAE 3ft × 4ft tunnel from M = 3.0 to 4.5 (Fig.6c). The agreement is surprisingly good at subsonic speeds and could imply that even in this speed range the pressure fluctuations radiated from the wall boundary layer still control transition, although it could be fortuitous. The wall pressure fluctuations may control transition because the turbulence level in the working section is low (only about $u^1/U = 0.2\%$ at M = 0.2 according to unpublished hot wire measurements by T.B. Owen). It is surprising that an empirical correlation derived from measurements at supersonic speeds should work so well at low subsonic speeds, in view of the strong effects of compressibility on the stability of a laminar boundary layer to small disturbances. However, these effects are extremely large even within the range of the supersonic experiments (M = 3 to 14) and the correlation may describe the later stages of the transition process, rather than the initial growth of small disturbances.

Difficulties with the transition sensors prevented the measurement of unit Reynolds number effects during the tests of the AEDC cone (see Tables 2 and 3). However the shaded areas at M = 0.2, 1.4 and 2.2 in Fig.17 indicate the large variation in transition Reynolds number with unit Reynolds number predicted from the measured variation of C_f and δ^*. Unpublished tests on another highly polished 10^o cone showed that large unit Reynolds number effects did occur in the RAE 8ft × 8ft tunnel at low supersonic speeds[35]. These measurements (Fig.18) are based on long exposure schlieren photographs of the boundary layer on the cone; they correspond with a position intermediate between transition onset R_t and complete transition R_T, but biased towards transition onset. The bars in Fig.18 indicate the range of reading on one or more photographs, whereas the arrows indicate fully laminar flows. The most significant feature of the measurements is that at the lowest Mach number, M = 1.4, the unit Reynolds number effect is large and reasonably well defined; it also agrees surprisingly well with the predictions for complete transition given by Pate and Schueler (equation (2)) for the profiles available (these predictions for particular values of C_f and δ^* are shown by the circles in Fig.18). Hence there is a possibility that equation (2) is valid for lower supersonic Mach numbers than considered previously.

Finally, it should be noted that if Pate and Schueler's equation is valid at subsonic speeds in the RAE 8ft × 8ft tunnel (as suggested by Fig.17) it should also be valid for well designed transonic tunnels. Thus in a carefully designed transonic tunnel with a low turbulence level and with a low level of high frequency pressure fluctuations satisfying the condition

$$1 \times 10^{-3} < \bar{p}/q < 5 \times 10^{-3} \qquad (6)$$

we might expect transition Reynolds numbers still to depend on the pressure fluctuations radiated from the turbulent sidewall boundary layers, and hence to find strong unit Reynolds number effects. The measurements at M = 1.2 in the slotted NASA Langley 8ft transonic pressure tunnel (Fig.12, Ref.4) nearly satisfy the condition stated as equation (6), for $\bar{p}/q$ is then only about

6×10^{-3}. At this low level of pressure fluctuations there is a unit Reynolds number effect on the end of transition, which might well be in accordance with equation (2). In contrast, at subsonic Mach numbers in this tunnel, the pressure fluctuations are much higher ($\bar{p}/q > 1 \times 10^{-2}$) and there are apparently no unit Reynolds number effects.

If we accept the transition measurements in perforated transonic tunnels with higher levels of pressure fluctuations[4], satisfying the conditions

$$\bar{p}/q > 1 \times 10^{-2}$$

the transition Reynolds numbers are determined primarily by the value of $\bar{p}/q$ and will have no dependence on a length scale, such as the displacement thickness of the wall boundary layer associated with the Pate and Schueler correlation. Hence in this situation we should expect no unit Reynolds number effects, unless a variation in unit Reynolds number changes $\bar{p}/q$. Thus we may draw a sketch (Fig.19) indicating the general trend of transition Reynolds numbers on a model within a wide variety of facilities. Fig.19 represents the general trend of transition measurements on the AEDC 10° cone at M = 0.80 (most of which correspond with adiabatic conditions) in the absence of surface roughness and vibration effects. The slopes of the R_T-$\bar{p}/q$ curves are drawn differently for $\bar{p}/q > 1 \times 10^{-2}$ and for $1 \times 10^{-3} < \bar{p}/q < 5 \times 10^{-3}$ to indicate that the modes of transition may be different in character between these two regimes. Although no flight experiments are yet available, we may expect that the transition Reynolds numbers will be higher, unless some other effect supervenes (such as flow angularity, surface roughness or vibration). For the flight experiment the slope of the R_T-$\bar{p}/q$ curve is drawn differently, because we may not assume that the mode of transition will be the same as in a wind tunnel with disturbances at least two orders of magnitude higher. A transition curve close to the flight experiment might be observed in small transonic wind tunnels with laminar side-wall boundary layers, for there the pressure fluctuations should be of the same low magnitude as in a flight or range experiment[36,37]. We may expect unit Reynolds number effects in flight, the length scale being provided by, say, the radius of curvature of the tip*.

* Potter has shown that reliable transition data on cones at transonic speeds may be obtained from range tests[38]. However, there is not yet sufficient data to establish the magnitude of the unit Reynolds number effect. Transition tests on cones at supersonic speeds (M = 2.2 and 5.1) in ranges show large unit Reynolds number effects[39,40] such that

$$Re_t \alpha (R)^{0.65} .$$

(In a range experiment it is generally difficult to ensure adiabatic conditions.)

This tentative classification of facilities by rms pressure fluctuations is perhaps oversimplified and somewhat arbitrary*, both in the curves suggested for transition Reynolds number and the areas marked as uncertain.

However Fig.19 has important implications for the design and operation of the new generation of high Reynolds number wind tunnels (which must attain low levels of flow unsteadiness[1,2]), and for attempts to predict the performance of aircraft or missiles at full scale. In particular, Fig.19 suggests that many wind tunnel models may have to be tested with devices to ensure a fixed transition point, if scale effects on turbulent boundary layer development are to be established. In addition the correct simulation of full-scale transition position will have to take account of the differing levels of flow unsteadiness and roughness on the aircraft and the model.

5 CONCLUSIONS

Tests of the AEDC 10° cone in three RAE wind tunnels suggest the following conclusions:

(1) The level of pressure fluctuations in the RAE 8ft × 8ft subsonic/supersonic tunnel is low, and the transition Reynolds numbers are relatively high (Fig.5).

(2) The transition Reynolds numbers in the RAE 3ft × 4ft supersonic tunnel are relatively high, and comparable with those measured on a flat plate in the same facility (Fig.6a), although they are somewhat lower than the predicted values from the Pate and Schueler correlation (Fig.6c).

(3) The level of pressure fluctuations in the RAE 8ft × 6ft transonic tunnel varies appreciably with Mach number (Fig.9a). In these tests surface imperfections, either at the tip or the microphones (Fig.9b), controlled the transition position.

(4) Surface pressure fluctuations measured at supersonic speeds are sensitive to how well the transducers match the surface (Fig.13 and 14) particularly in the transition region of the boundary layer (Fig.12).

(5) Extrapolation of transition data from wind tunnel to flight tests is a difficult task because of the high pressure fluctuations in most transonic tunnels (section 4).

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* Some of the degree of arbitrariness may be removed when the results of flight tests of the AEDC cone and additional transition tests in transonic tunnels with low levels of $\bar{p}/q$, become available.

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Appendix

NOTES ON RAE 8FT × 8FT TUNNEL TESTS

Microphone M2 (Table 2) $\phi = 180^\circ$

The level of pressure fluctuations in the RAE 8ft × 8ft tunnel is extremely low and hence microphone M2, x = 660mm (26in), could be used to detect the forward movement of the transition region as unit Reynolds number increased (Table 2). Microphone M1, x = 457mm (18in), was not used because near transition the signal displayed large low frequency fluctuations, possibly associated with an unsteady laminar separation downstream of the microphone, which projected slightly from the surface.

Compared with the limited data available from the traversing surface pitot at subsonic speeds (M = 0.2 and 0.8), microphone M2 shows an earlier transition onset (smaller R_t) and a later completion of transition (larger R_T). These anomalies may be attributed to a lack of symmetry in the transition front between $\phi = 0^\circ$ and $\phi = 180^\circ$.

Traversing surface pitot (Table 3) $\phi = 0^\circ$

Convincing indications of the onset of transition were difficult to obtain from the traversing surface pitot, despite the adjustments made. Initially the pitot pressure increased steadily as the probe was traversed forward. No break indicating completion of transition, or onset was noticed. However, many of these traces indicated progressively increasing pitot pressure fluctuations for distances closer to the apex than 406mm (x < 16in). These fluctuations were tentatively ascribed to lateral vibrations of the pitot. (Although no lateral vibration could be observed on the TV screen with the wind on, a lateral mode could be excited by striking the probe support for x < 406mm.) Hence the contact pressure between the cone and the surface pitot was increased, and two possibly valid traces were obtained at a Mach number of M = 0.20 (Table 3).

The pitot probe was then bent downwards to ensure a nose down pitching moment. Three possibly valid traces were then obtained at M = 0.80, but the results for $R = 8.2 \times 10^6$/m (2.5×10^6/ft) may be unreliable because the variation with angle of incidence ($\pm 1^\circ$) was unprecedented (Table 3).

Table 1

EUROPEAN TUNNELS

Tunnel	Test section	Configuration	Mach numbers
RAE (B)	8ft × 8ft	Closed	Subsonic and supersonic
ARA (B)	9ft × 8ft	Perforated (normal holes)	Subsonic and transonic
ONERA S2 (Modane)	1.8 × 1.8m	Perforated (60° inclined holes)	Subsonic and transonic
S3	0.56 × 0.78m	Slotted	Subsonic and transonic
NLR (Amsterdam)	2.0 × 1.6m	Slotted	Subsonic and transonic
RAE (B)	3ft × 4ft	Closed	Supersonic
RAE (F)	8ft × 6ft	Slotted	Subsonic and transonic

B = Bedford
F = Farnborough

Table 2

MICROPHONE M2
$\phi = 180^{o}$, $x = 660mm$

Mach number	Unit Reynolds number $\times 10^{-6}$		Transition $\times 10^{-6}$			rms
			Onset	Peak	Complete	$\bar{p}/q$
M	(R/m)	(R/ft)	R_t	R_p	R_T	%
0.8	5.9	1.8	-	3.9	-	0.52% fully turbulent
0.8	7.9	2.4	-	-	5.1	
0.6	3.3	1.0	2.2	-	-	0.31 to 0.43% laminar
0.6	5.3	1.6	-	3.4	-	
0.6	6.9	2.1	-	-	4.6	
0.2	2.9	0.9	1.8	-	-	0.5% fully turbulent
0.2	5.2	1.6	-	3.4	-	
0.2	7.2	2.2	-	-	4.7	
1.4	4.5	1.5	3.3	-	-	0.8% laminar
1.4	7.9	2.4	-	5.2	-	
1.4	9.8	3.0	-	-	6.4	
2.4	4.3	1.3	2.9	-	-	0.26% laminar
2.4	7.5	2.3	-	4.9	-	
2.4	8.5	2.6	-	-	5.7	

Table 3

TRAVERSING SURFACE PITOT
ϕ = 0, x varies

Mach number M	Unit Reynolds number $\times 10^{-6}$ R/m	Unit Reynolds number $\times 10^{-6}$ R/ft	Transition $\times 10^{-6}$ Onset R_t	Transition $\times 10^{-6}$ Complete R_T	Remarks
0.8	13.1	4.0	-	-	Invalid
0.8	6.6	2.0	-	-	Invalid
0.8	4.9	1.5	0.8	1.3	Doubtful: too low
0.6	6.6	2.0	-	-	Invalid
0.6	3.3	1.0	-	-	Invalid
0.2	4.9	1.5	1.1	1.8	Doubtful: too low
Contact pressure increased					
0.2	4.9	1.5	2.9	3.9	Possibly valid of traverse above
0.2	6.6	2.0	2.9	4.0	Possibly valid
0.2	9.8	3.0	-	-	Invalid
0.2	8.2	2.5	-	-	Invalid
0.2	6.6	2.0	2.9	4.0	'Elastic' behaviour of pitot
Probe bent downwards					
0.8	4.9	1.5	3.3	4.5	Possibly valid
0.8	6.6	2.0	3.3	4.4	Possibly valid
0.8	9.8	3.0	-	-	Invalid
0.8	8.2	2.5	3.3	4.1	Possibly valid
Attempt to assess sensitivity to changes in angle of incidence					
0.8	8.2	2.5 = +1°	-	-	Invalid
0.8	8.2	2.5 = −1°	1.0	1.2	Extremely low

SYMBOLS

C	circumference of tunnel working section } equation (2)
C_1	circumference of reference working section = 1.22m (4ft) } equation (2)
C_F	mean turbulent skin friction coefficient on the tunnel wall } equation (2)
C_f	local skin friction coefficient in working section (equation (3))
f	frequency (Hz)
k	roughness height
L	typical length
M	free stream Mach number
M_e	local Mach number at edge of boundary layer
$\sqrt{nF(n)}$	level of excitation at particular frequency = $\Delta p/q(\varepsilon)^{\frac{1}{2}}$
Δp	pressure fluctuation in a band Δf at frequency f
$\bar{p}$	rms pressure fluctuation
q	$= \frac{1}{2}\rho U^2$ kinetic pressure
R	free stream unit Reynolds number
R_k	roughness Reynolds number $R \times k$
R_t, R_p, R_T	transition Reynolds numbers at onset, peak and complete transition based on x and R
U	free stream velocity
x	streamwise length measured from cone apex
α	angle of incidence (defined in Fig.2)
δ^*	boundary layer displacement thickness
ε	analyser bandwidth ratio $\Delta f/f$
θ_c	cone semi angle
ρ	free stream density
ϕ	roll angle (defined in Fig.2)

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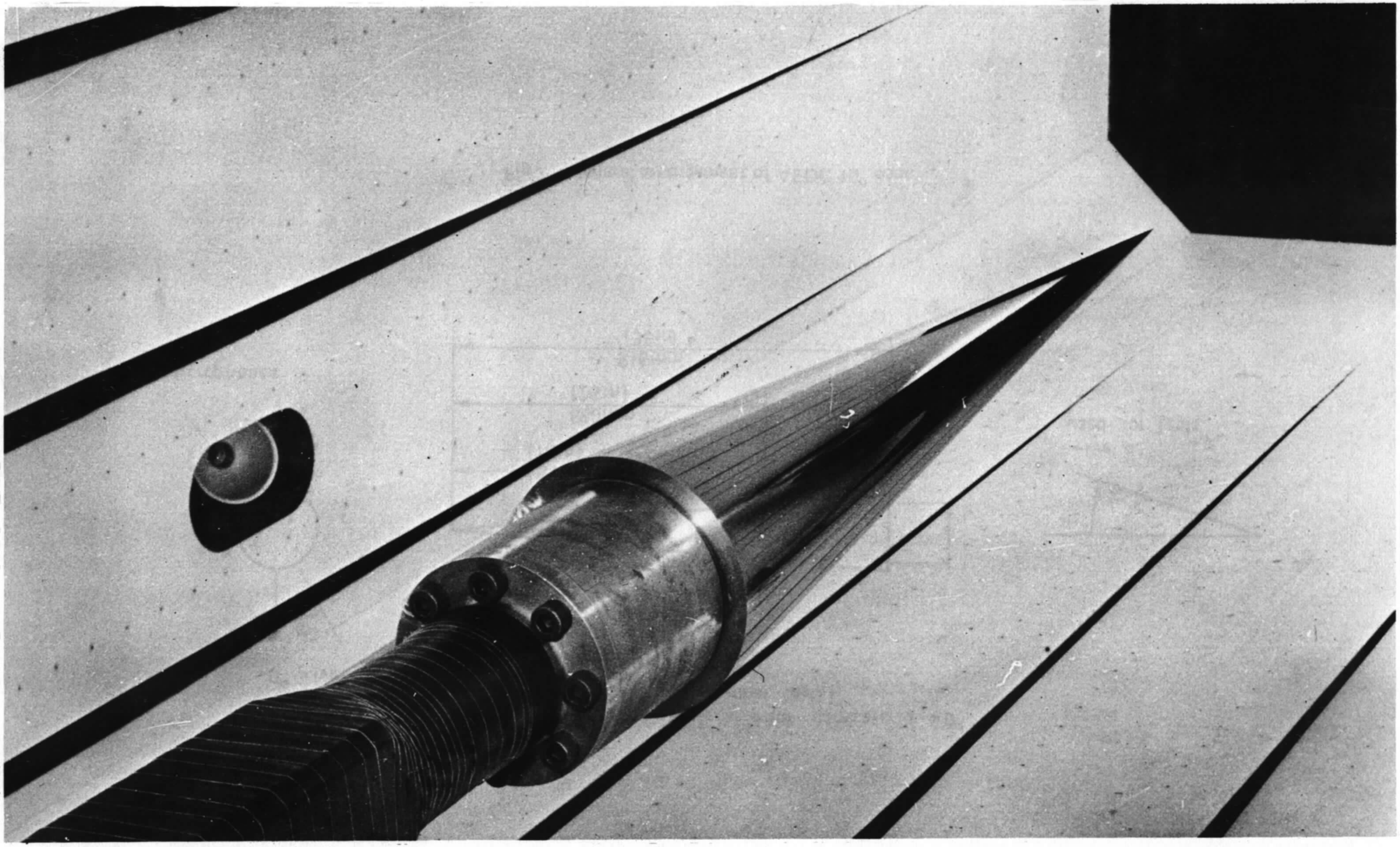


Fig.1 AEDC 10° cone in RAE 8ft x 6ft tunnel

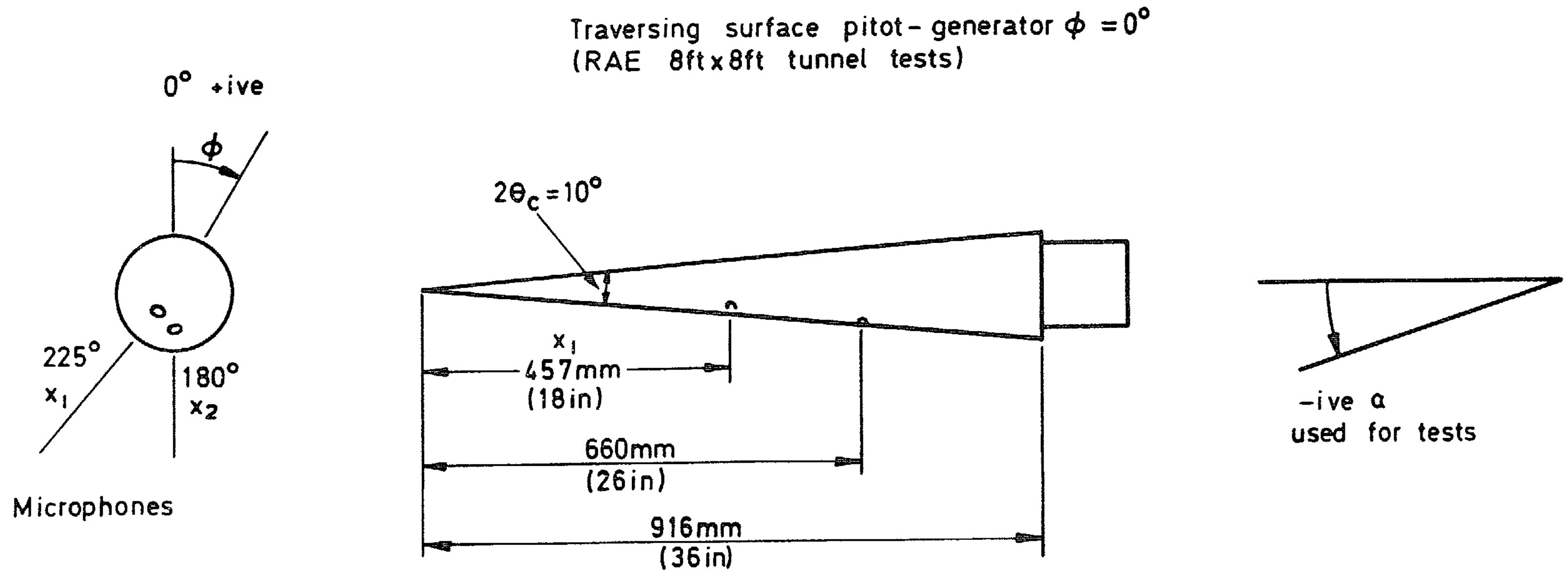


Fig 2 General arrangement of AEDC 10° cone

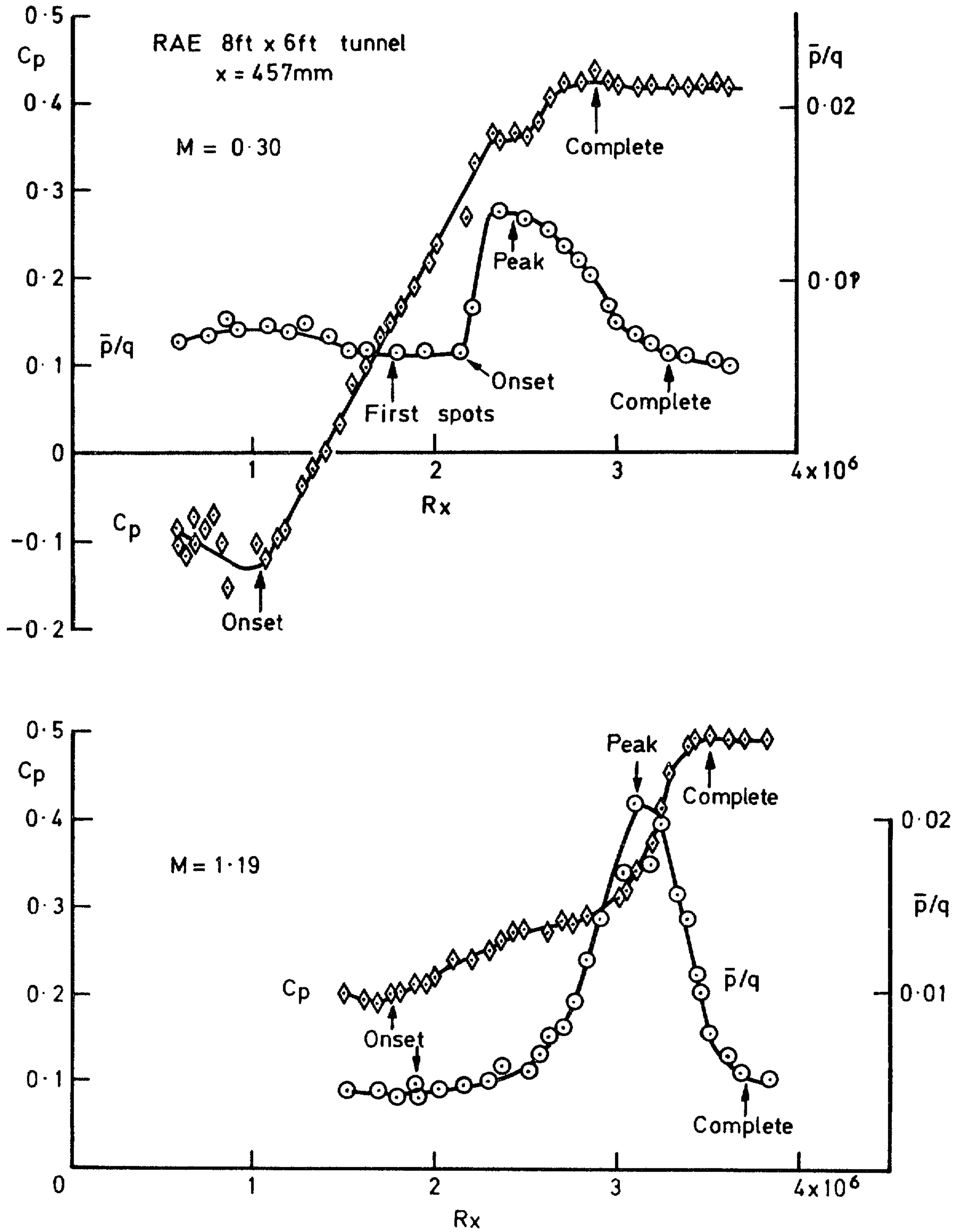


Fig 3 Comparison of static pressure fluctuations and fixed surface pitot pressure

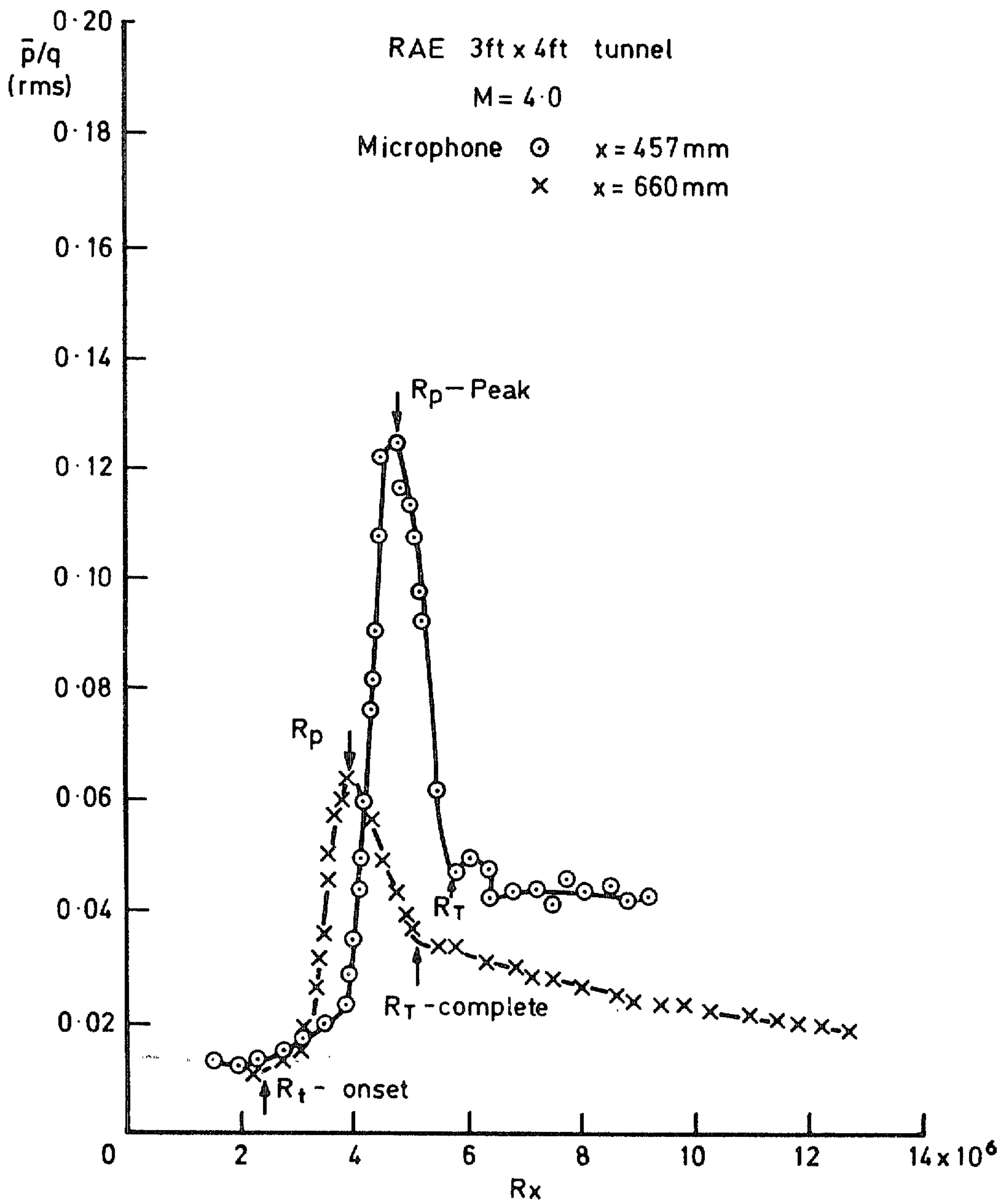
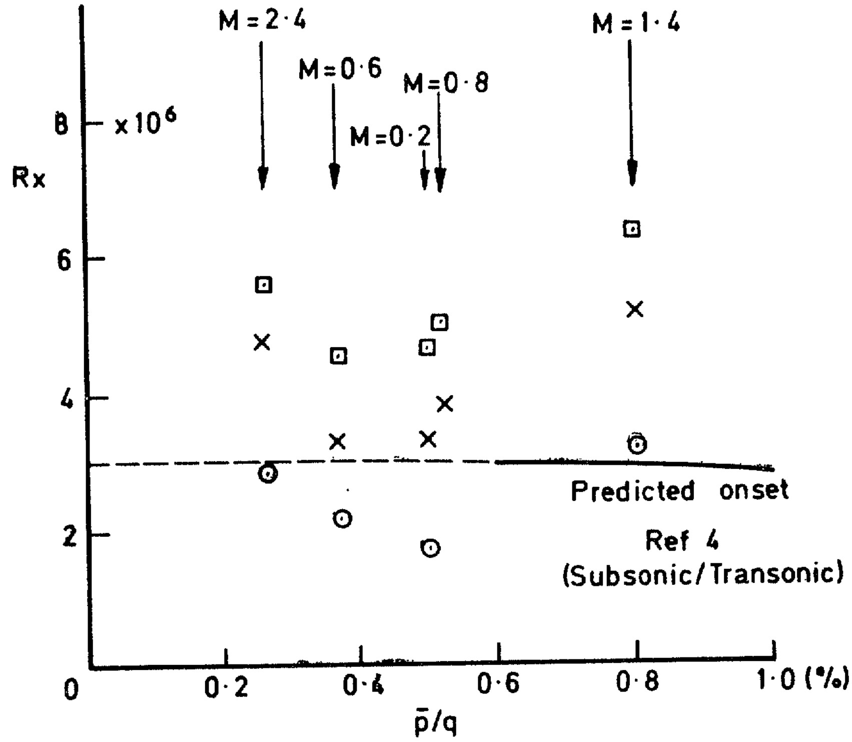
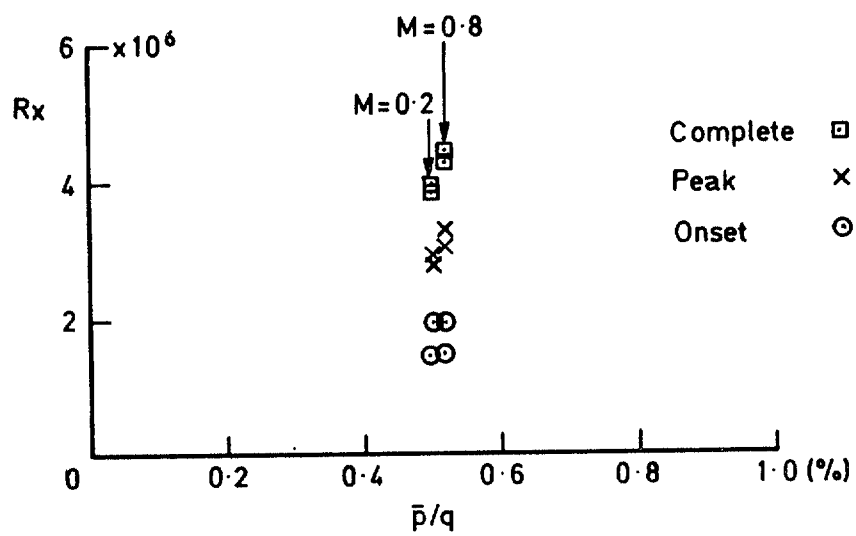


Fig 4 **Variation of static pressure fluctuations with Reynolds number — AEDC 10° cone**

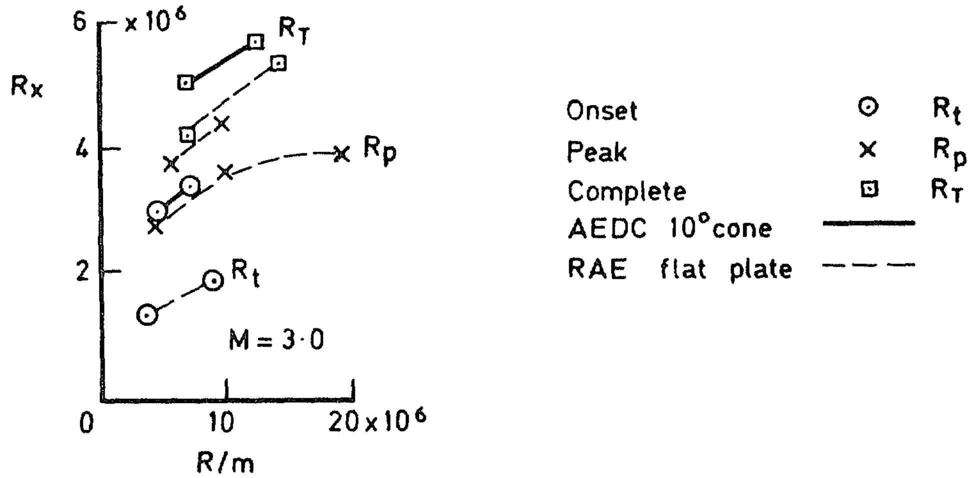


a Microphone (x = 660mm)



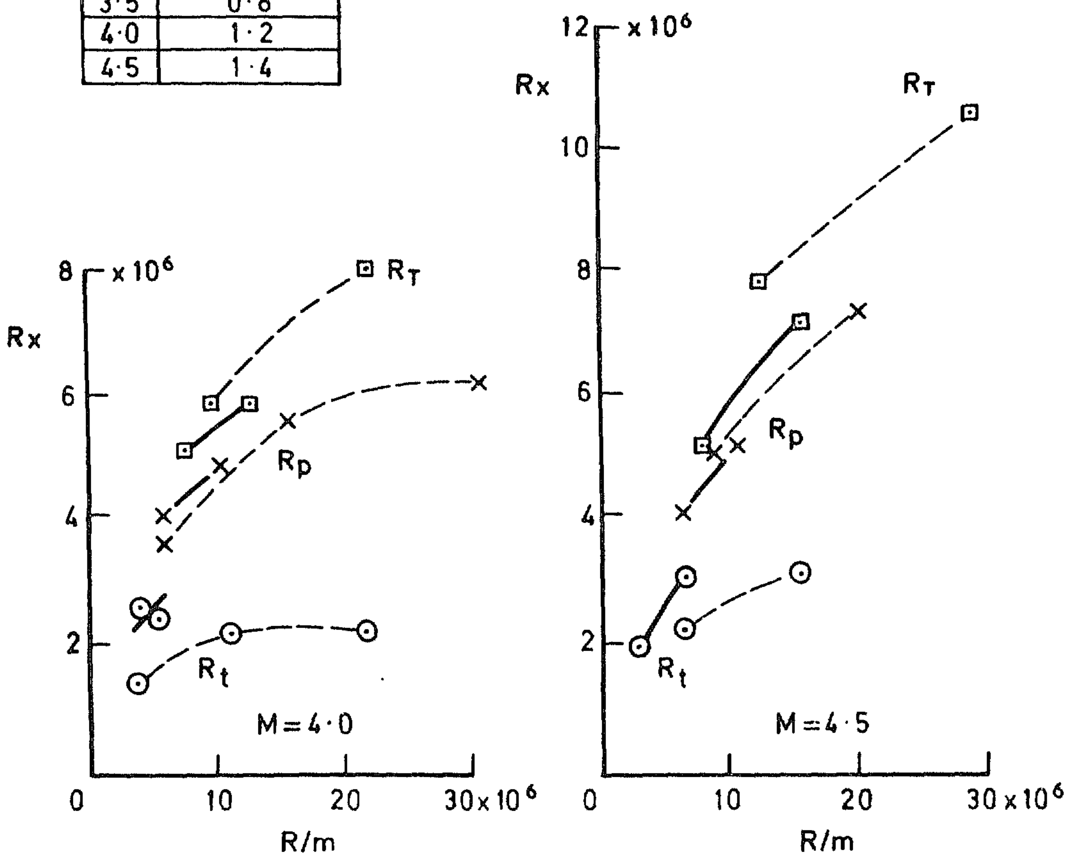
b Traversing surface pitot

Fig 5 RAE 8ft x 8ft tunnel – transition Reynolds numbers v rms pressure fluctuations in free stream



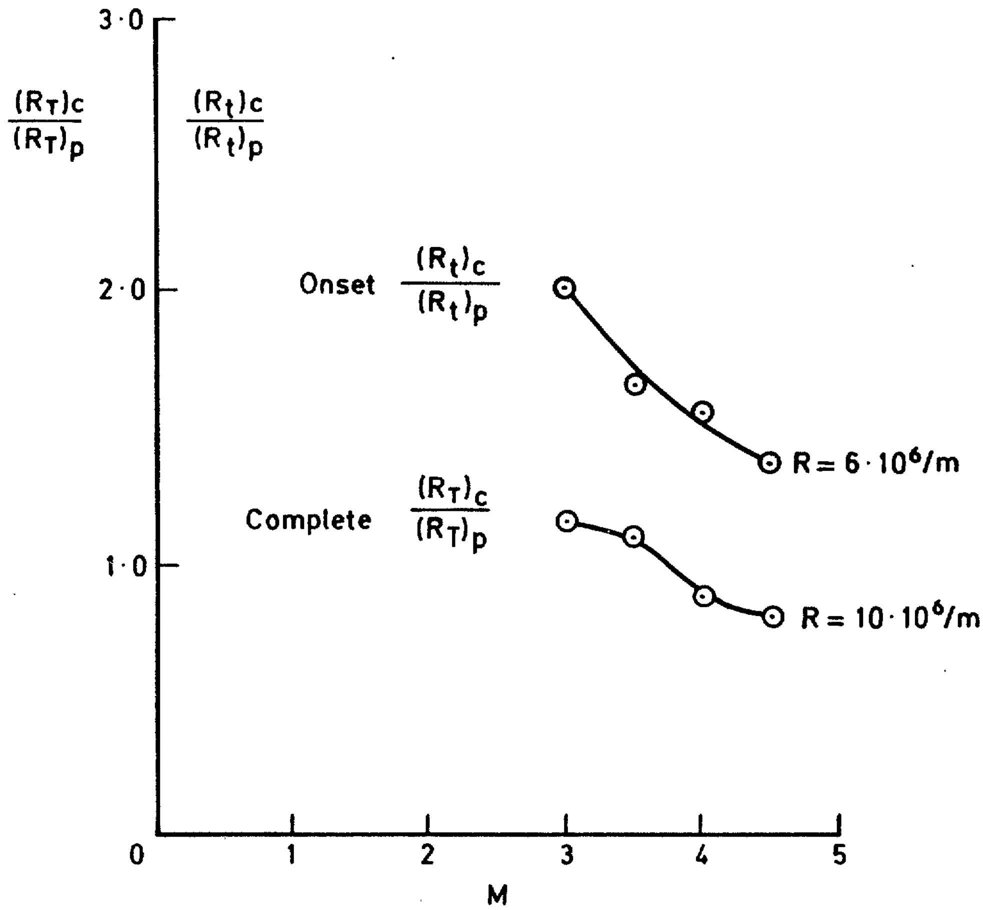
Indicated free stream pressure fluctuations (laminar boundary layer)

M	$\bar{p}/q$ (%)
2·5	0·4
3·0	0·6
3·5	0·8
4·0	1·2
4·5	1·4



a Measurements

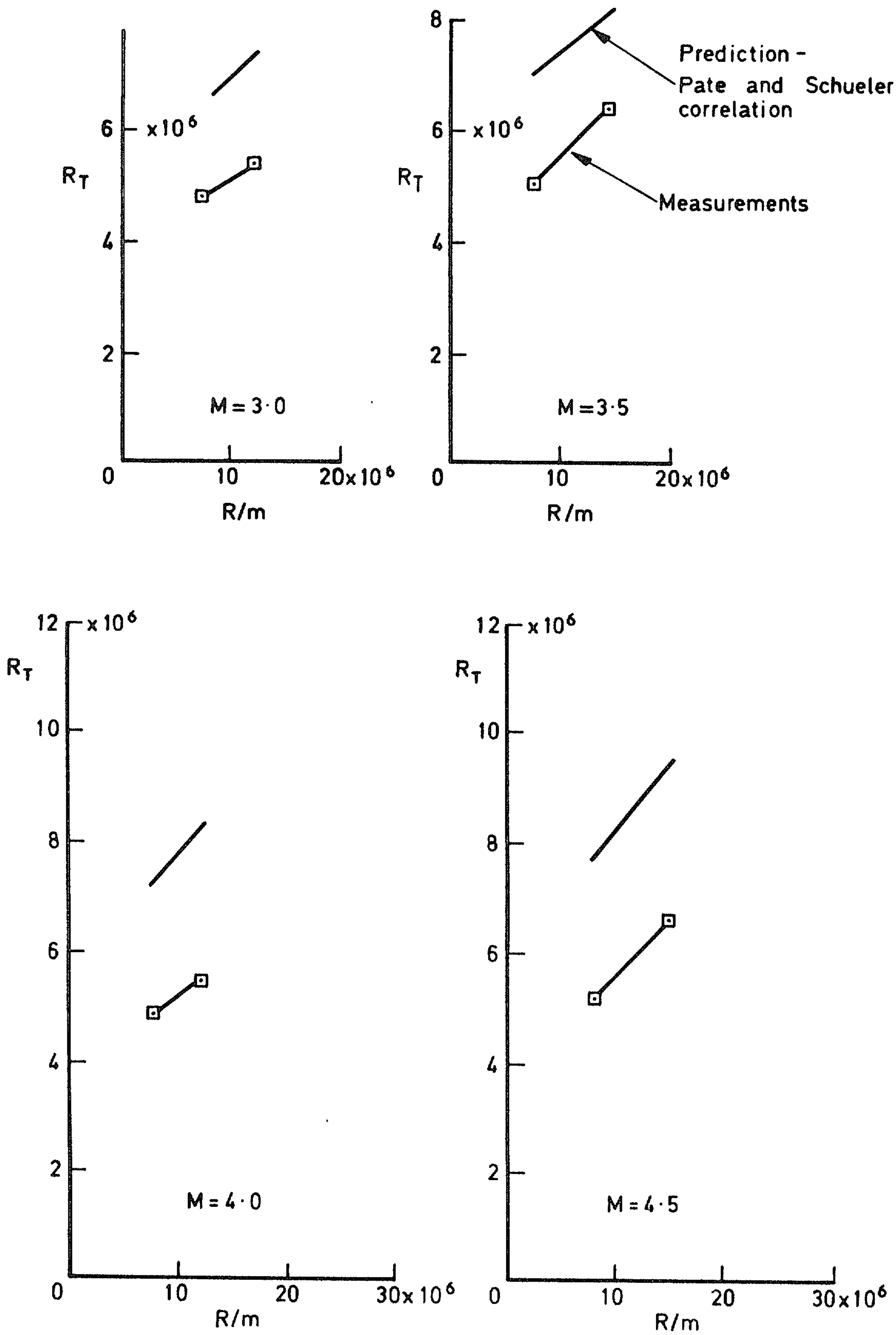
Fig 6a RAE 3ft x 4ft tunnel – variation of transition Reynolds number with Mach number and unit Reynolds number



b Ratios for cone and flat plate

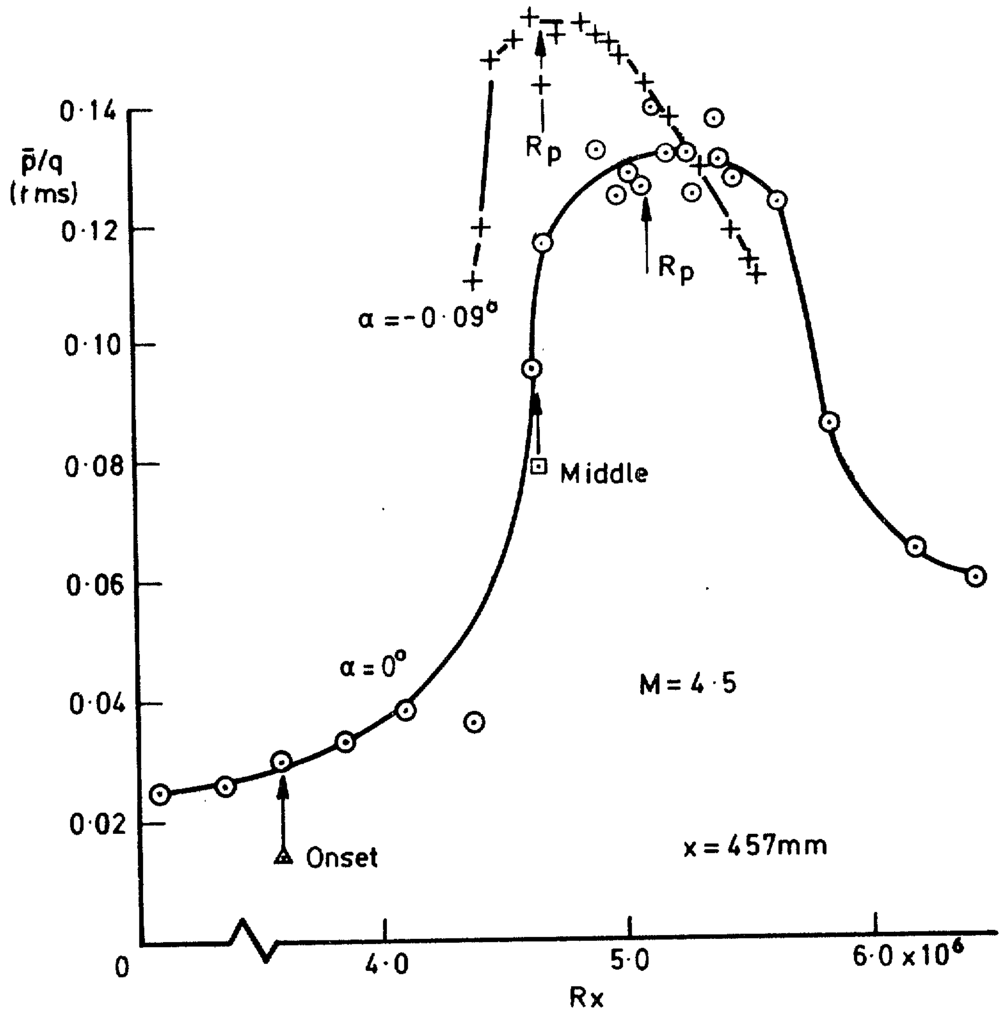
Fig 6b RAE 3ft x 4ft tunnel – variation of transition Reynolds number with Mach number and unit Reynolds number

Fig 6c

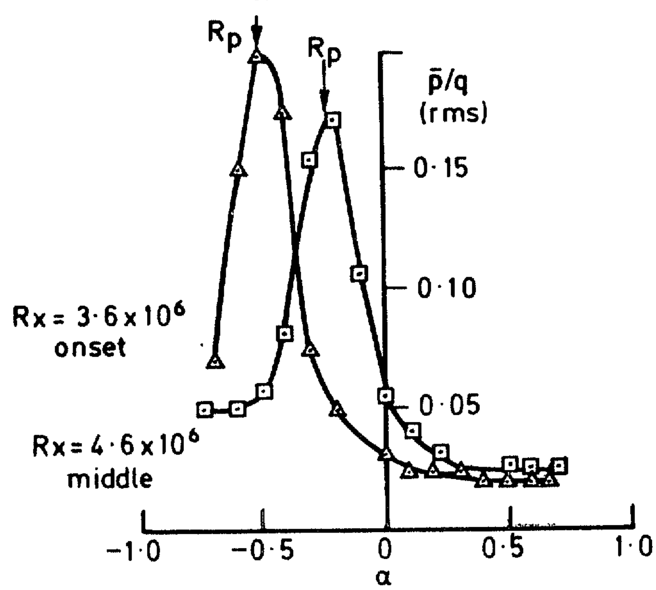


c Comparison with predictions

Fig 6c RAE 3ft x 4ft tunnel — variation of transition Reynolds number with Mach number and unit Reynolds number



a α constant R varies



b R constant α varies

Fig 7a&b RAE 3ft x 4ft tunnel — variation of transition front pressure fluctuations with Reynolds number and angle of incidence

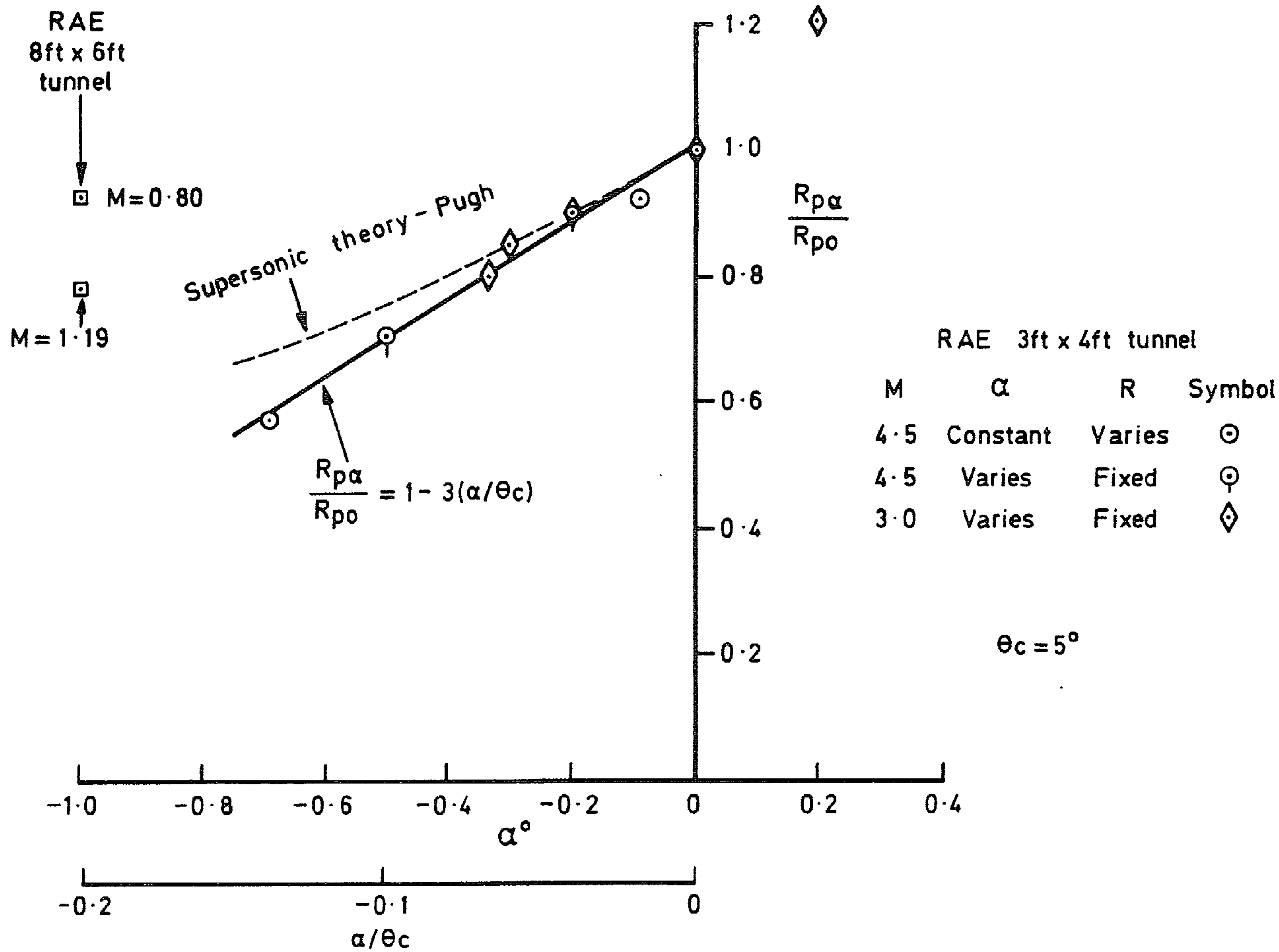


Fig 8 Variation of transition with angle of incidence and Mach number

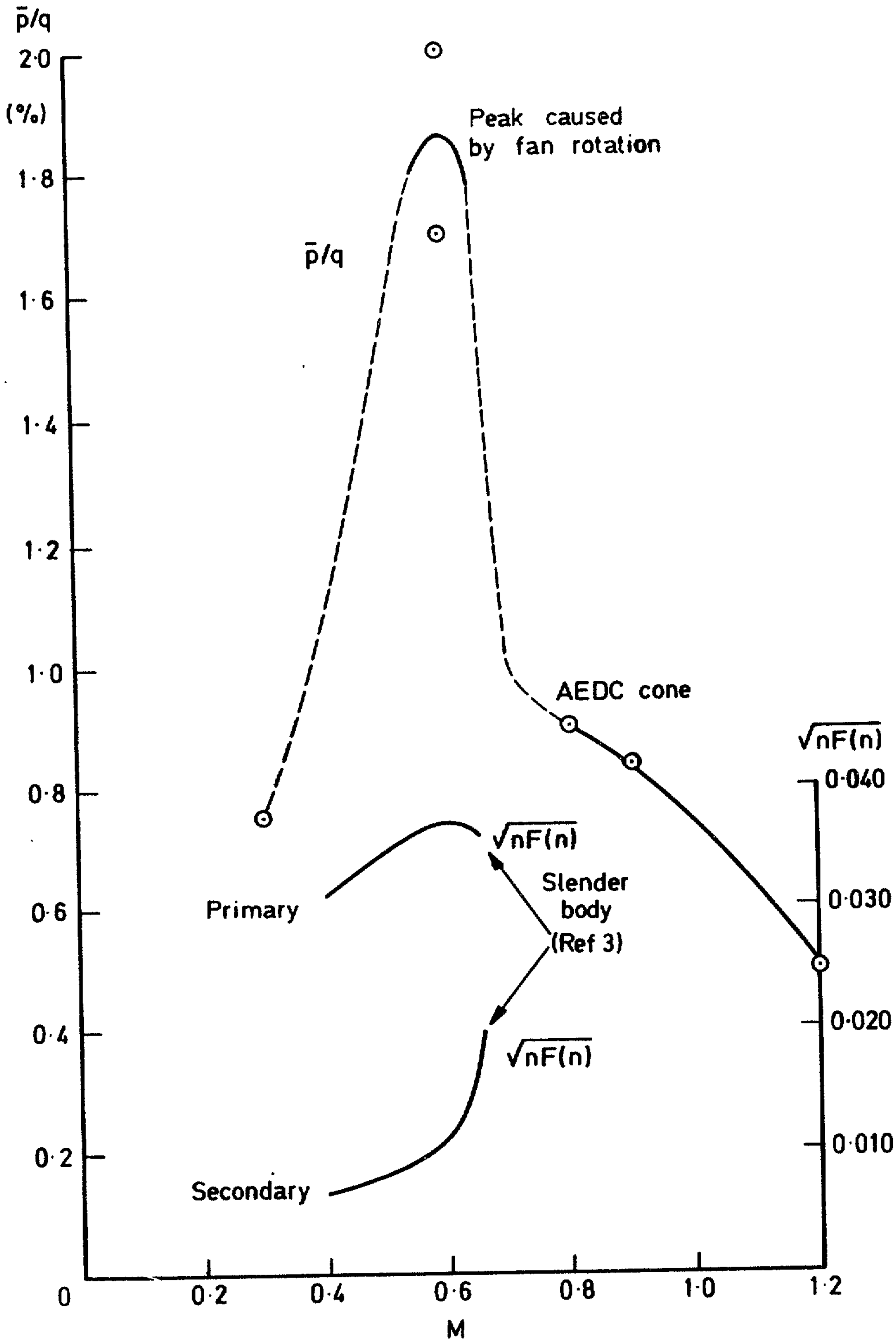


Fig 9a RAE 8ft x 6ft tunnel — variation of tunnel pressure fluctuations with Mach number

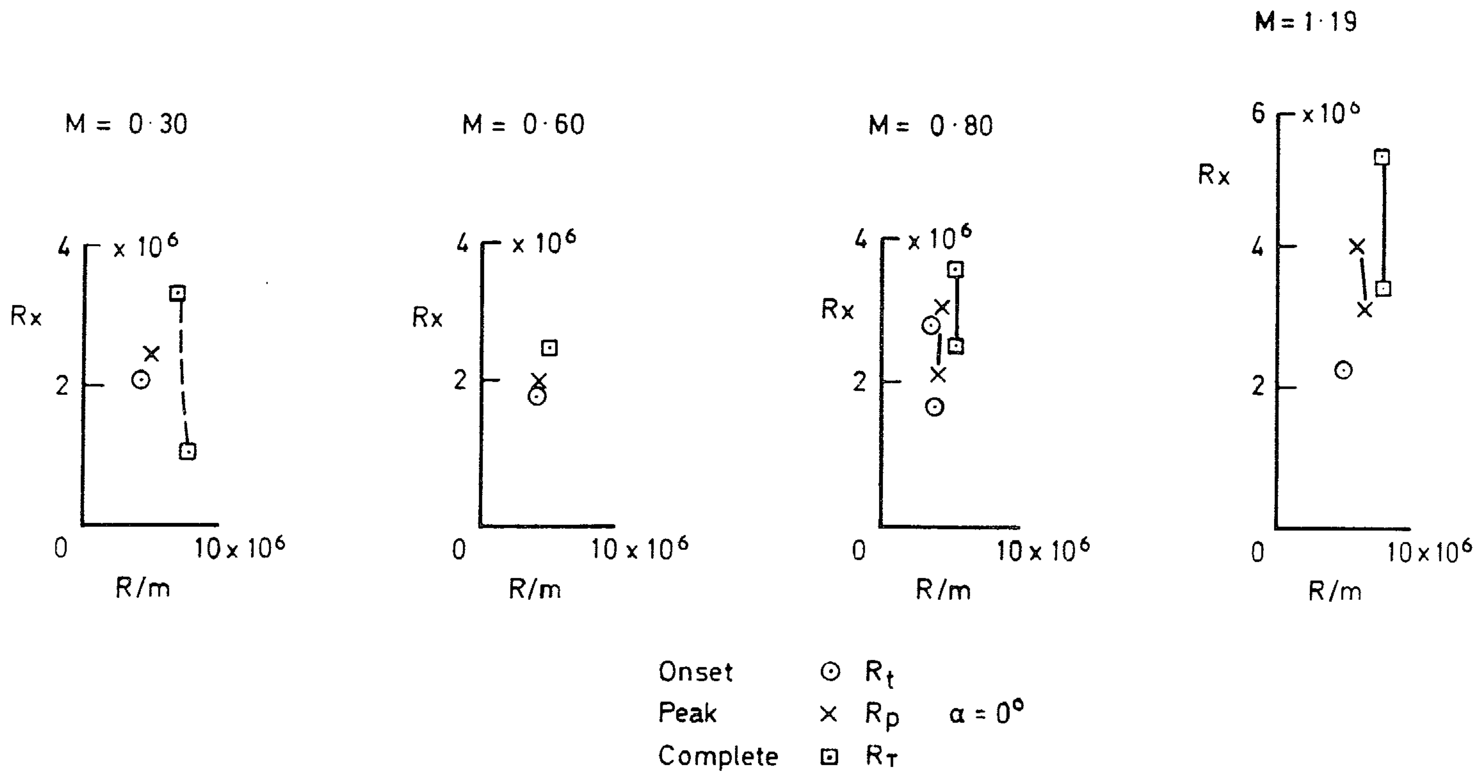
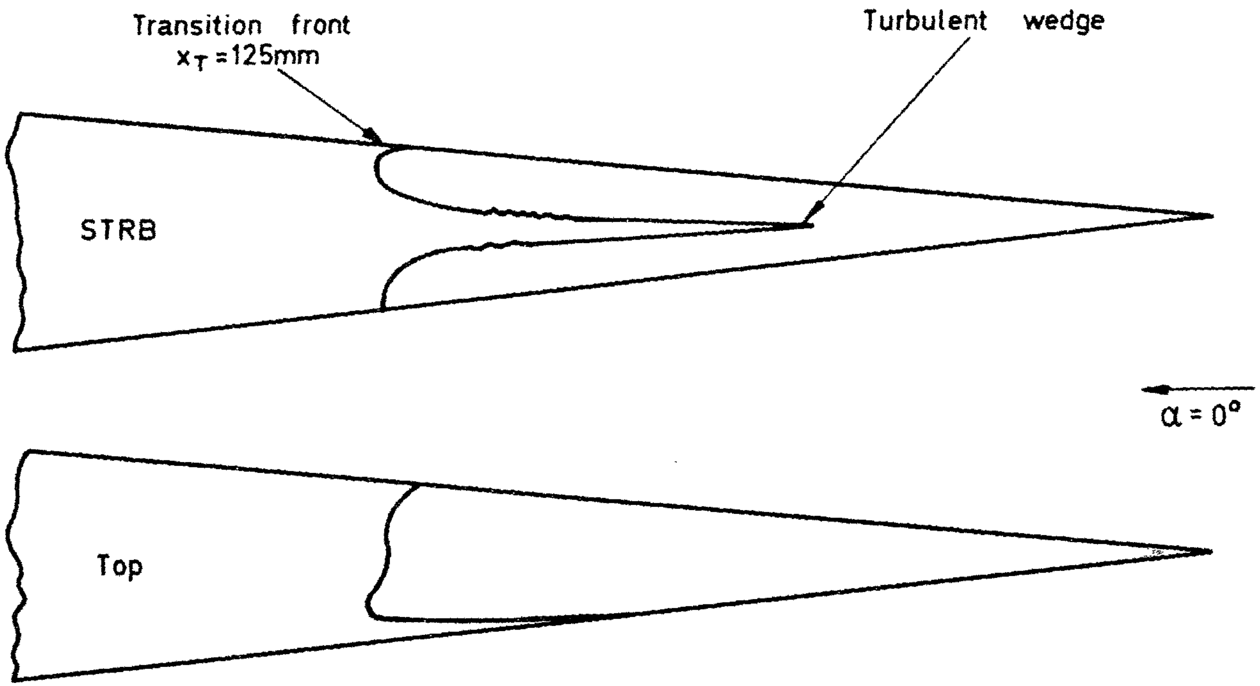


Fig 9b **RAE 8ft x 6ft tunnel — variation of transition Reynolds numbers with Mach number and unit Reynolds number**

RAE 8ft x 6ft tunnel



M	R_T	Indicator
0·18	0.5×10^6	Napthalene – as above
0·40	1.0×10^6	Napthalene – as above

Fig 10 Sublimation patterns of transition

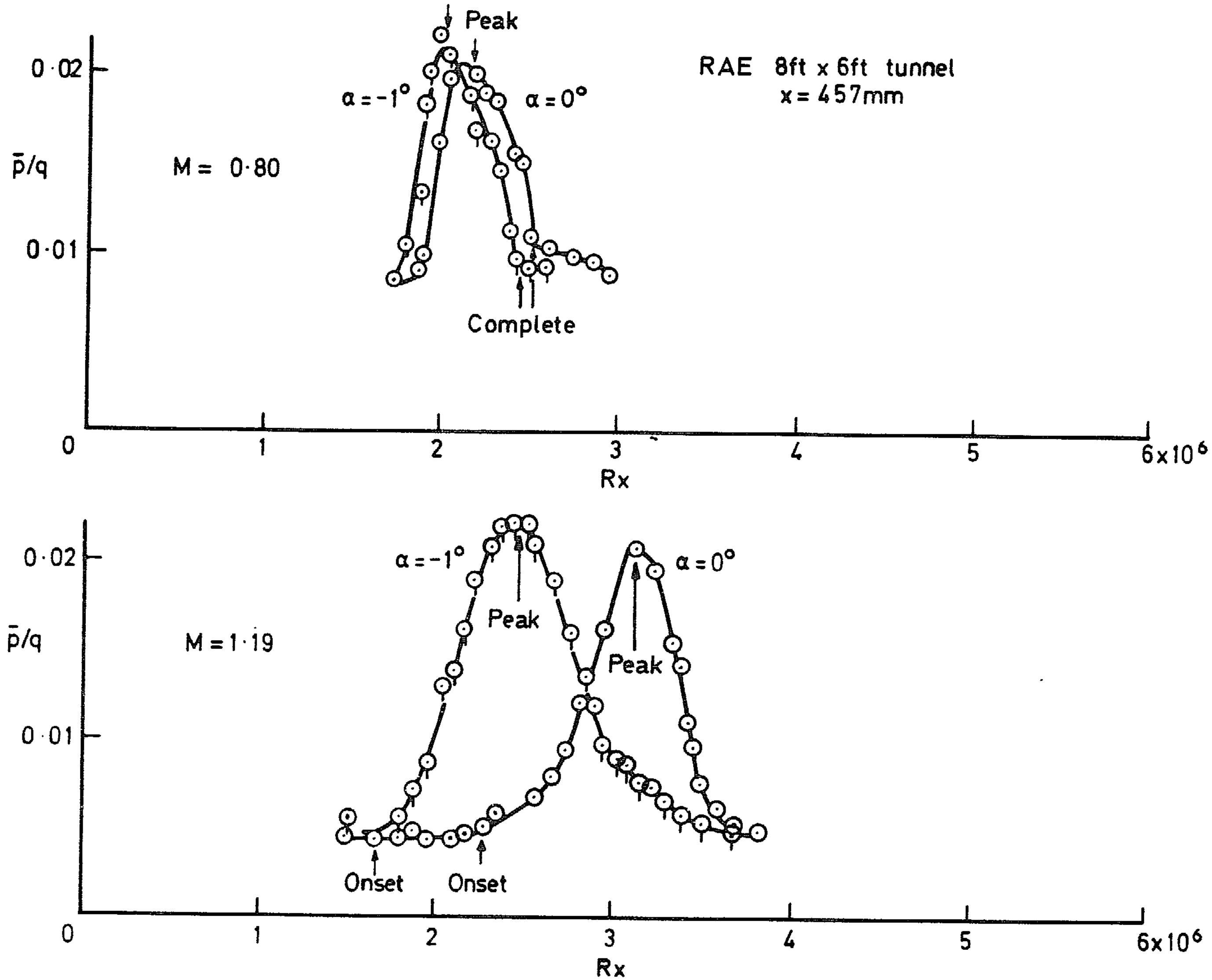


Fig 11 Influence of flow angularity on transition

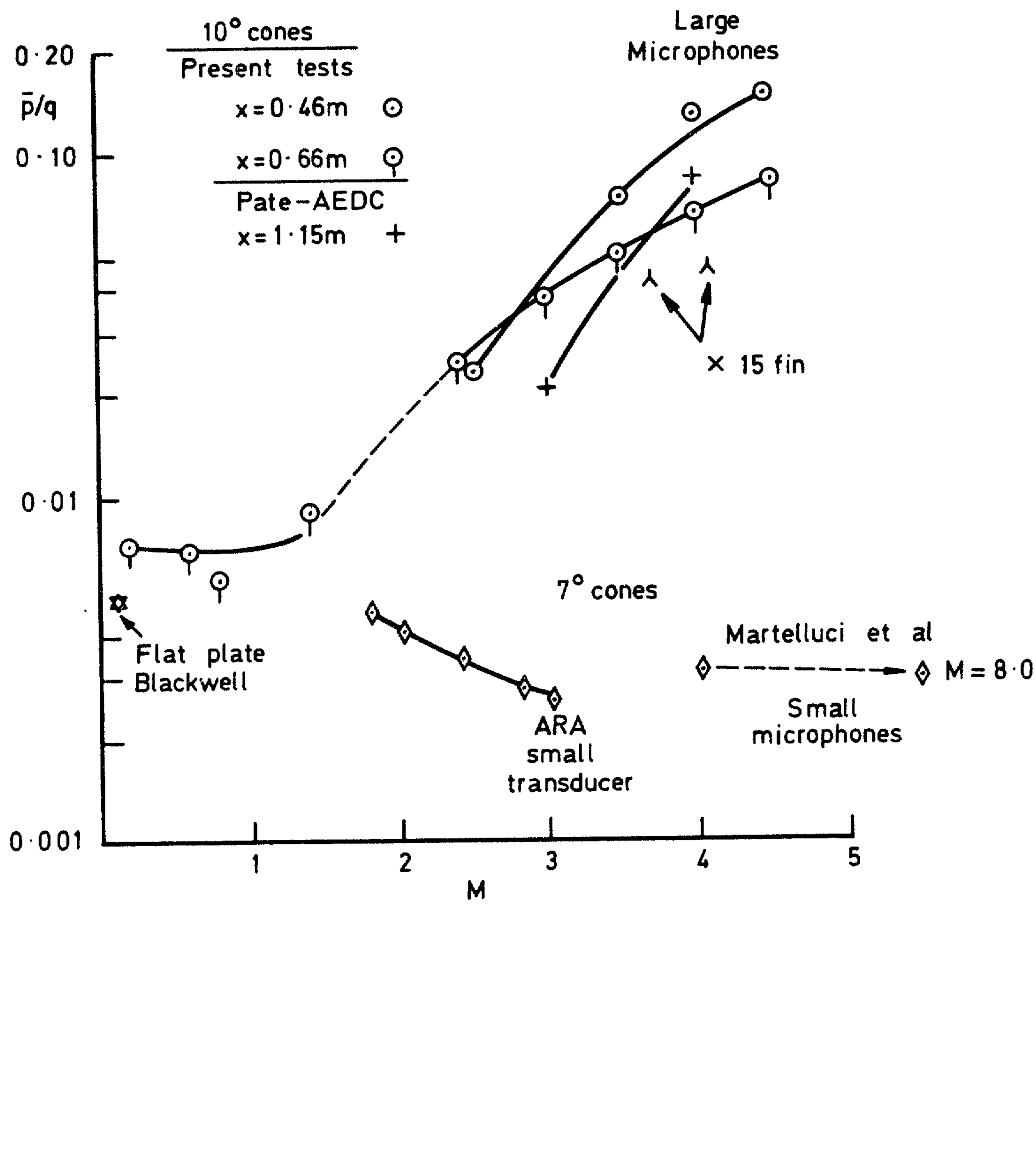
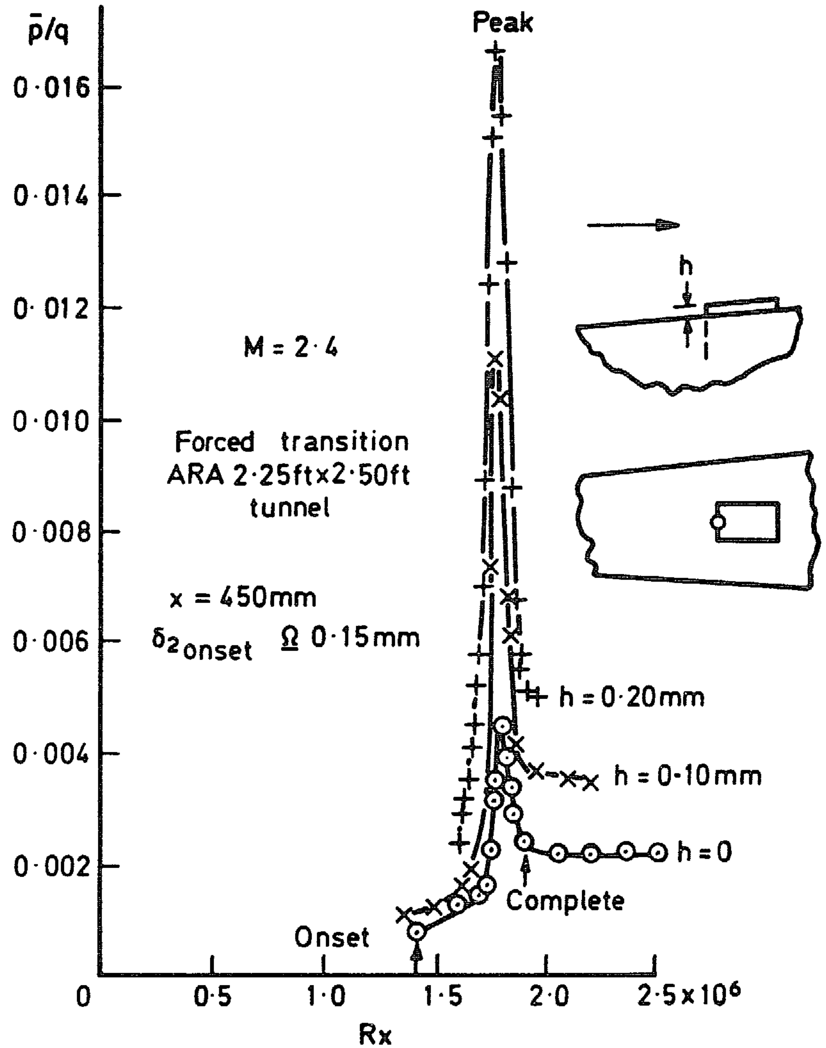


Fig 12 Variation of peak pressure fluctuations with Mach number



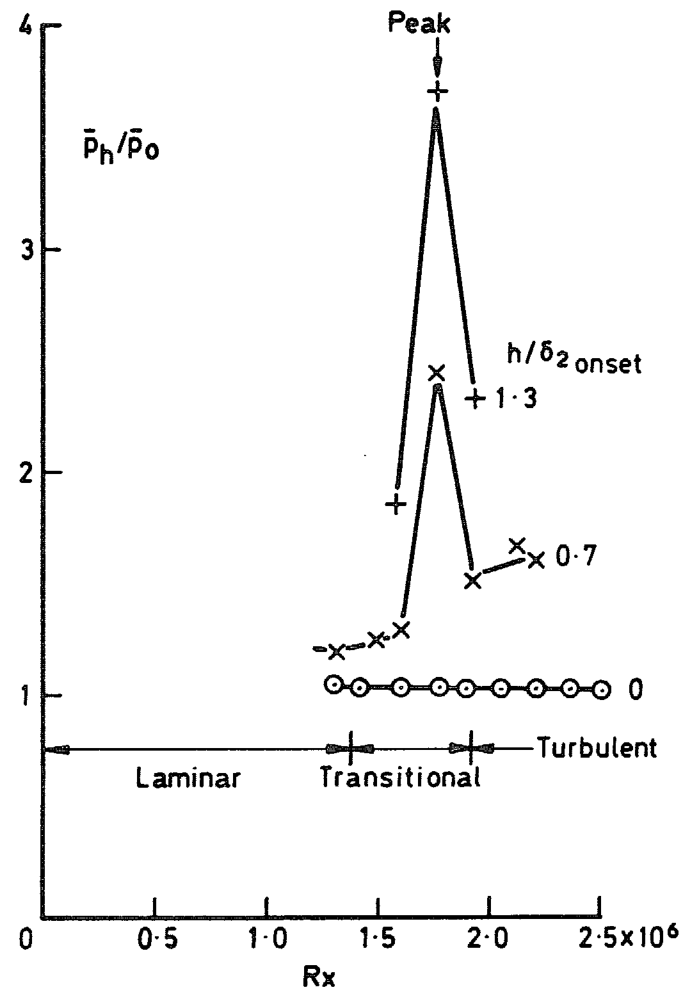


Fig 13 Influence of step height on static pressure fluctuations

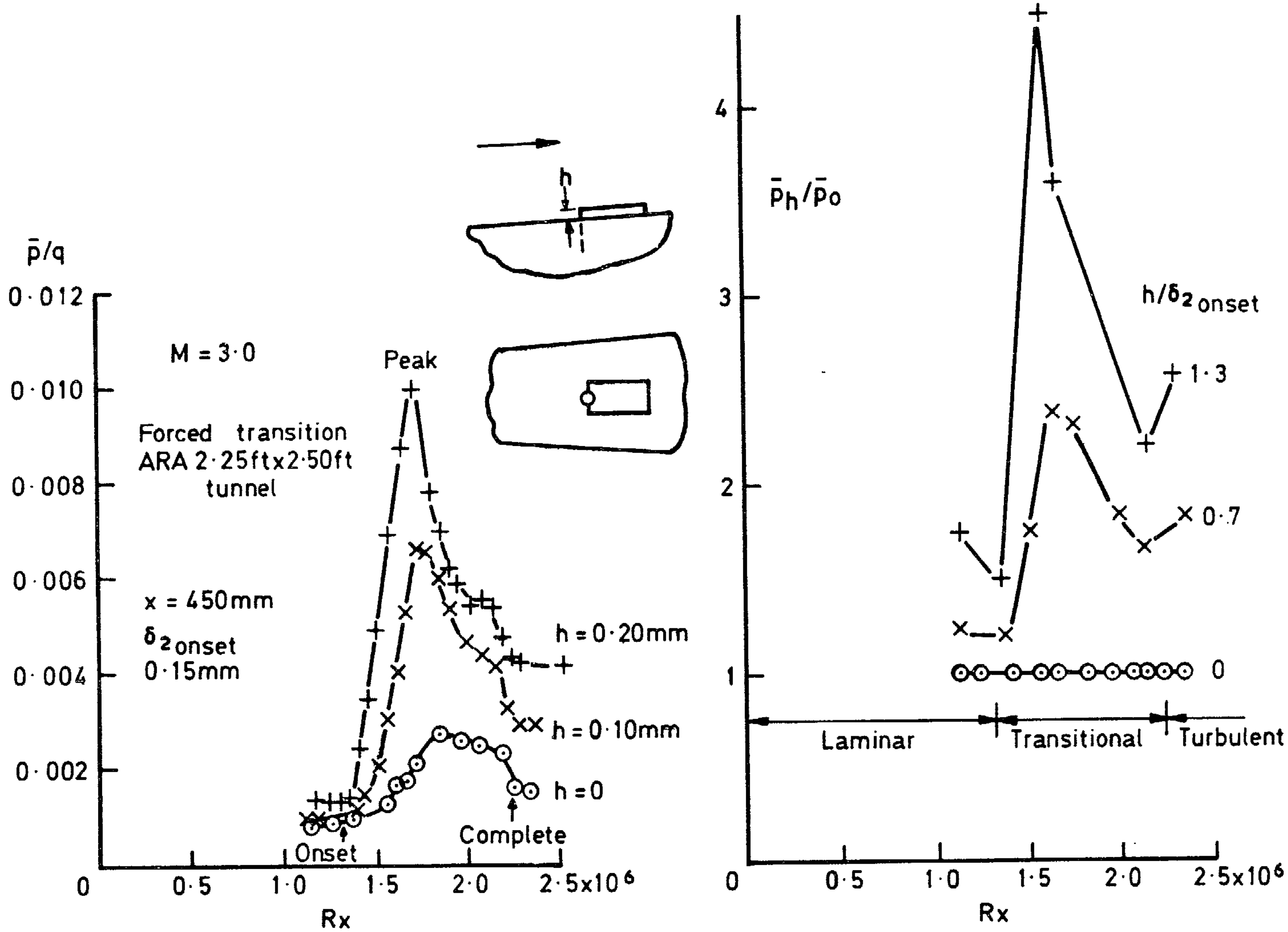


Fig 14 Influence of step height on static pressure fluctuations

Fig 15

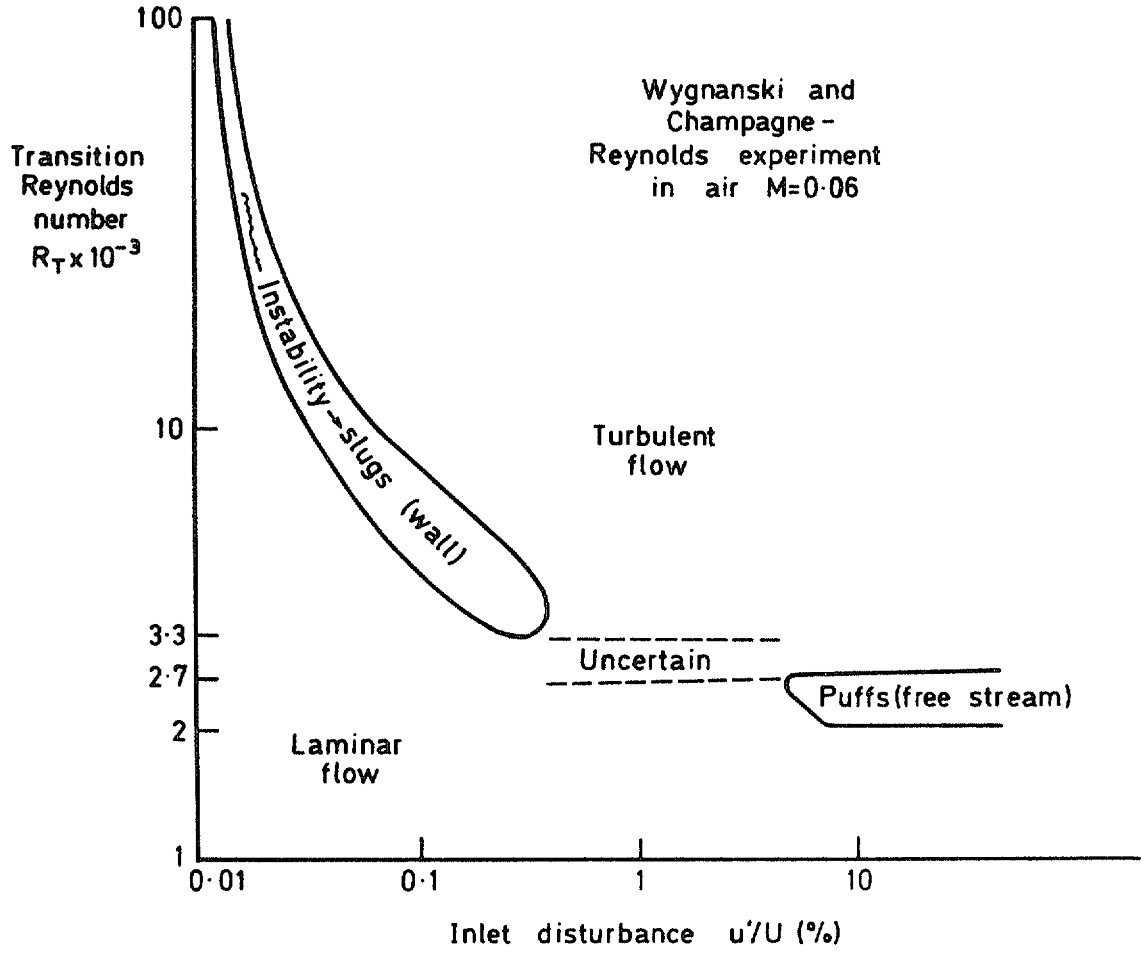


Fig 15 Transition modes in a pipe determined by inlet turbulence

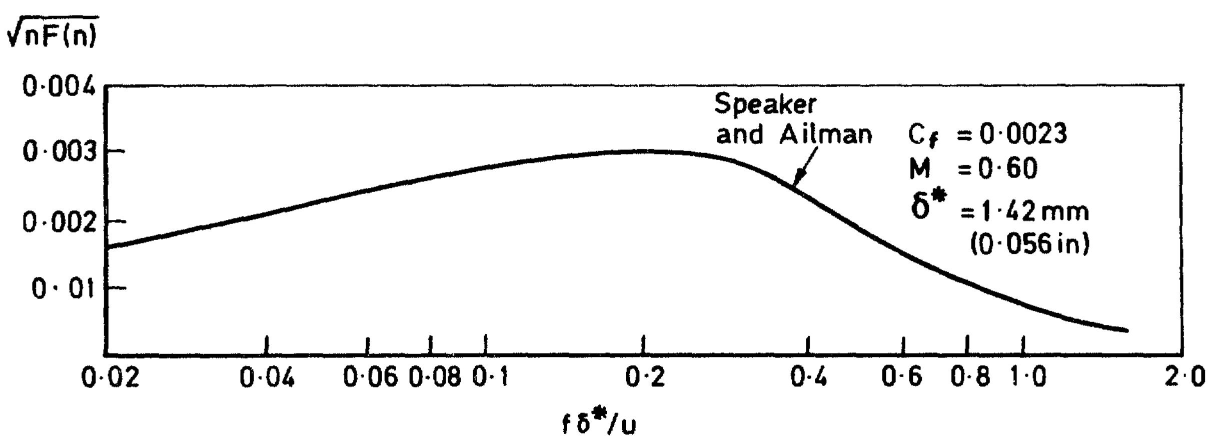


Fig 16 Spectrum of pressure fluctuations under an attached turbulent boundary layer

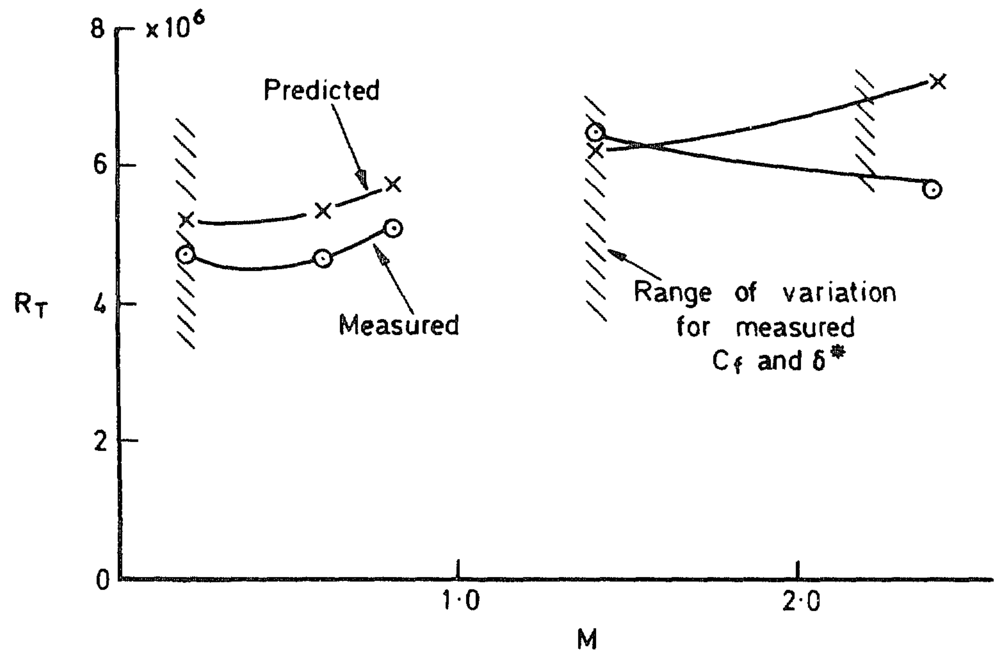


Fig 17 RAE 8ft x 8ft tunnel – comparison of measured and predicted Reynolds numbers for complete transition

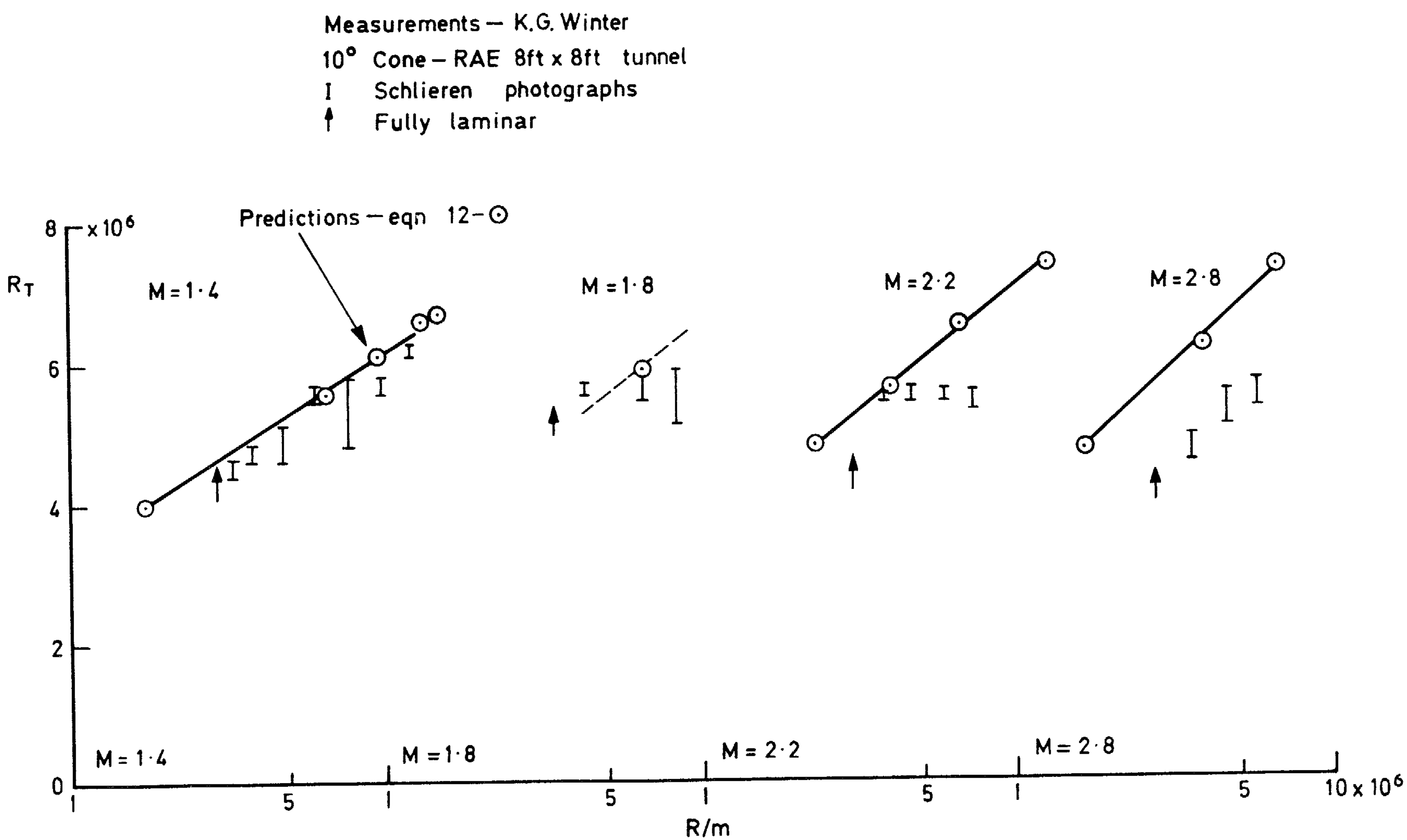


Fig 18 Comparison of measured and predicted effects of unit Reynolds number on transition in the RAE 8ft x 8ft tunnel

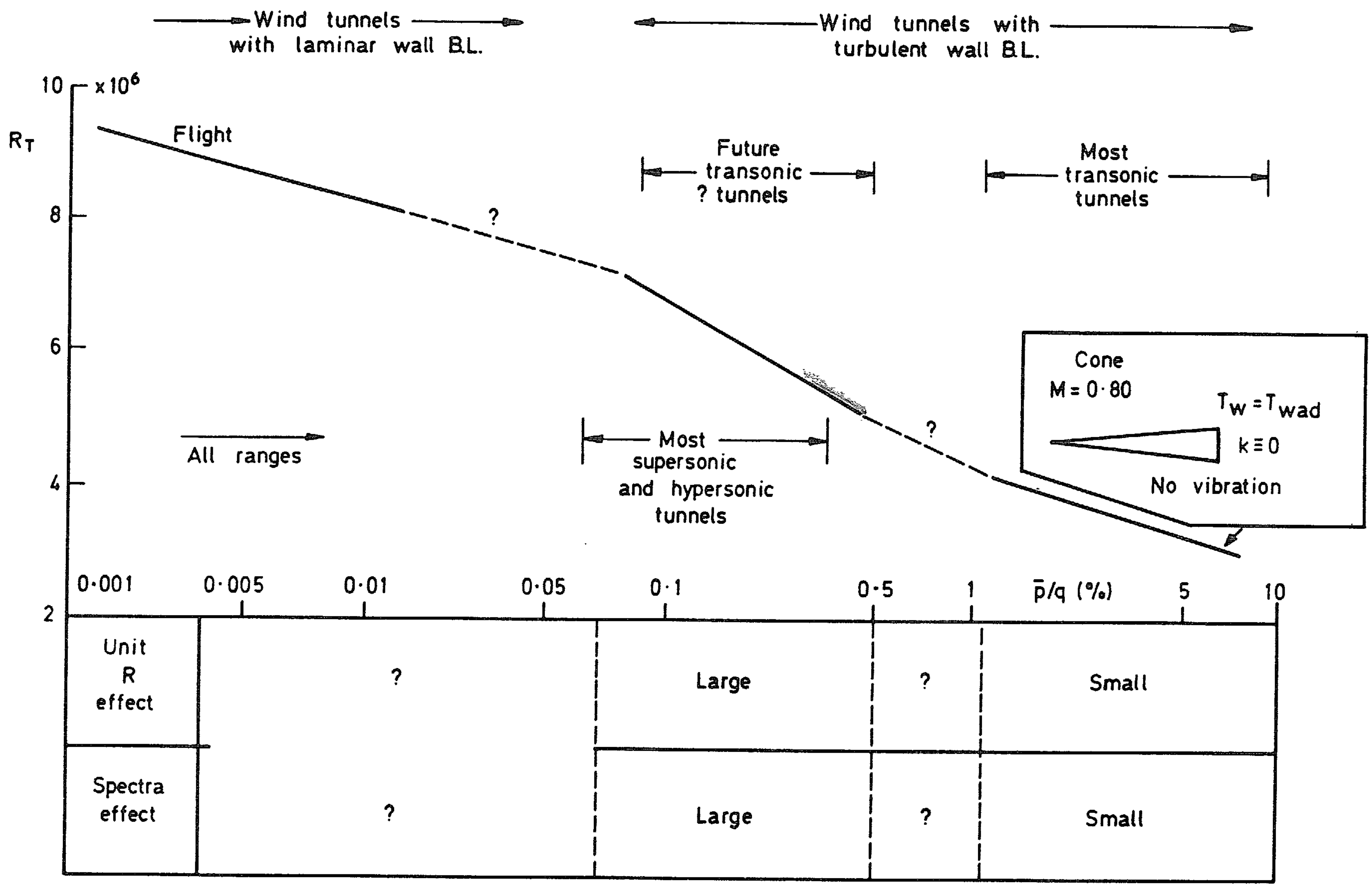


Fig 19 Test facilities for transition experiments

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