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FLIGHT AND WIND-TUNNEL TESTS ON AN AERODYNAMICALLY COMPENSATED PITOT-STATIC HEAD FOR THE BAC 221 AIRCRAFT

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SUMMARY

The BAC 221 aircraft is fitted with a special pitot-static head, insensitive to incidence, and aerodynamically compensated on similar principles to the head fitted to the Concorde, to minimise aircraft pressure field effects. Tests have been made on the head covering the Mach number range 0·2 to 1·4 in two wind tunnels and at three altitudes in flight.

The compensation significantly reduces the static pressure errors at high subsonic and transonic speeds but small errors remain at low subsonic and supersonic speeds. The sensitivity to incidence and sideslip, of the pitot and static pressures sensed, is satisfactorily low.

The manufacturer's prediction of the head performance agrees reasonably well with tunnel results but poorly with flight results. It appears that the prediction of the aircraft pressure field is inaccurate.

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* Replaces RAE Technical Report 69013 - ARC 31370

CONTENTS

1 INTRODUCTION

Accurate knowledge of airspeed and altitude (and of the corresponding Mach number for high performance aircraft) is fundamental to the safe operation of aircraft. These parameters are normally derived from measurements of the ambient static pressure at the aircraft altitude, and the pitot pressure. It is essential not only that these pressures should be sensed as accurately as possible, but that the magnitude of any sensing errors remaining should be known. This is particularly important for research aircraft such as the BAC 221 where the pitot and static pressures are used both to derive information for presentation to the pilot, and must also be recorded separately as part of the test information.

It is essential that for flight test use, the pressure sensing errors should cause only small errors in indicated altitude at approach speeds at sea level, certainly less than say 40 ft. Larger errors can be tolerated at other flight conditions but it is desirable that altitude errors should be less than ±100 ft. These limits, which also satisfy flight safety requirements imply that errors in indicated airspeed will always be well within ±1%, particularly at high Mach numbers.

It is inherently difficult to obtain accurate results from flight tests designed to determine pressure sensing errors, since it is necessary to derive the small difference between the true and the sensed pressures, which are measured independently, and are both large. In addition, a level of accuracy which is perfectly adequate to enable determination of the pressure sensing errors within satisfactory limits for both flight tests and flight safety, may not be sufficient for the fundamental determination of the performance of a sophisticated pitot-static system such as that on the BAC 221. The accuracy of pressure error measurements must be at least sufficient to enable definition of the errors to better than the limits given above.

The BAC 221 slender ogee-wing research aircraft, Figs.1 and 2, was converted from one of the two Fairey Delta 2 research aircraft. Pitot and static pressures were sensed on the latter aircraft by a Mk.9 pitot-static head mounted on a boom extending forward from the aircraft nose. The Mk.9 head[1,2,3] is a relatively old design and more modern pitot-static heads have a better performance, particularly over the wide speed and incidence range of which the BAC 221 is capable; at low speeds of about 110 kt the aircraft has been flown at incidences above 20°, and Mach numbers in excess of 1·5 have also been achieved[4]. A special head, with a low sensitivity of pressure error to flow incidence, and incorporating aerodynamic compensation to minimise errors induced by the presence of the aircraft, was designed for the BAC 221 by the Rosemount Engineering Company, using the original nose boom mounting.

This Report describes the comprehensive tests made on the head, both in flight on the BAC 221 at R.A.E. Bedford, and also in low and high speed wind tunnels at R.A.E. Bedford and the Aircraft Research Association (A.R.A.) respectively. The results are compared with the manufacturer's predictions. The principles of aerodynamic compensation are discussed briefly in section 2.

Heads designed on a similar principle to that for the BAC 221 are fitted to the Concorde prototypes, and another has been tested on a Lightning aircraft at the Aircraft and Armament Experimental Establishment, Boscombe Down[5].

2 SOURCES OF PRESSURE ERROR AND PRINCIPLES OF COMPENSATION

The main sources of error are the pressure field induced by the aircraft's volume distribution, and incorrect alignment of the pressure sensing orifices with the local flow direction. The latter source is due both to the variation of the aircraft's incidence and sideslip angles and to the flow angles induced by the presence of the aircraft, which may vary with incidence[6,7]. The aircraft's pressure field, which affects only the static pressure for a head mounted on a nose boom, is a function of Mach number, aircraft configuration, which changes with undercarriage operation and drooped nose position on the BAC 221, and may also vary with incidence. The dependence of the pressure errors on flow direction angle normally varies with Mach number. In addition, changes in engine air mass flow may alter the local flow direction at the orifices and may also be considered as an effective change in aircraft volume distribution, thus altering the pressure field. These sources of error may be dependent on Reynolds number, but this is not normally significant.

At transonic and supersonic speeds, shock waves generated by the aircraft can lead to large pressure sensing errors. This is particularly so at transonic speeds, when the rapid movement of shocks across the static orifices causes large and sudden changes in measured static pressure. A nose boom mounting for a pitot-static head, as on the BAC 221, is considered to be the best pressure sensing location on a supersonic aircraft since only one shock wave, the fuselage bow shock, then passes across the static orifices.

At supersonic speeds, the pitot-static head has its own bow shock, and it is the pitot pressure downstream of this shock which is sensed. In the absence of interference, the static pressure downstream of the pitot shock decreases back to a value close to the free stream value in about 10 head diameters[6], usually by the time the static orifices are reached.

The error due to the aircraft pressure field may be reduced by shaping the head to induce an approximately equal and opposite pressure error at its static orifices[8,9]. This technique is known as aerodynamic compensation*, and may be achieved in two ways.

The compensation may be either forward facing, when the nose shape of the head forward of the static orifices is specially designed, or rearward facing, when the shape of the head aft of the orifices provides the compensation[9]. At supersonic speeds, when the aircraft bow shock is downstream of the static orifices, the aircraft pressure field no longer influences the pressure sensed on a nose probe since disturbances cannot travel upstream in supersonic flow. Therefore, no compensation is required at supersonic speeds and it follows that rearward facing compensation is preferable, since forward facing compensation, being upstream of the static orifices, can leave a large residual error at supersonic speeds. A head with rearward facing compensation is designed so that the part of the head forward of the static orifices will induce zero error at supersonic speeds, and that the part downstream will induce an error approximately equal and opposite to that induced by the aircraft's pressure field at subsonic speeds. In practice there is usually a small residual error with rearward facing compensation, since the compensation profile is very close to the static holes, allowing some disturbances to propagate upstream in the boundary layer, and also probably because the design of the nose section may not be perfect. A head having rearward facing compensation was designed for the BAC 221.

It should be emphasised that a compensated head is designed for use at a particular position on a particular type of aircraft and cannot in general be used on other types of aircraft.

The use of a compensated pitot-static head mounted on a nose boom should cancel the effect of pressure errors due to the aircraft's pressure field, but it is still necessary to design the head for minimum sensitivity of the

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* Aerodynamic compensation may be regarded as any aerodynamic design feature of a pitot-static head which induces intentional and useful non-zero pressure errors when the head is tested in isolation and which are intended to oppose errors due to some other source.

sensed pitot and static pressures to flow direction. This was particularly important for the BAC 221 which has a very large range of flight incidence.

Current British pitot-static heads, such as the Mk.9 [1,2,3] and Mk.10 [10], have hemispherical noses containing the pitot entry. This geometry is insensitive to flow direction angles up to about 5° in any plane, but more modern heads have better pitot entry geometry with considerably wider insensitive ranges[11]. A conical nose giving a very wide insensitive range was chosen for the BAC 221 head.

The design of an adequate static pressure orifice configuration is more difficult. The static pressure orifices of the Mk.9 head are slots which are virtually axi-symmetric. The sensitivity of the static pressure sensing of this head to flow direction angles in any plane is thus fairly constant[2]. However, it is possible to choose a static orifice configuration which is very much less sensitive to flow direction angles in one plane, such as the incidence plane, than in the plane at right angles[6]. The Mk.10 head has such a configuration[10], having 22 circular holes disposed in a complicated pattern at several axial positions. This head has extremely low sensitivity of static pressure sensing to incidence, at the expense of an increased sensitivity to sideslip, but on balance it is a very great improvement over the Mk.9 head. Unfortunately this complicated static hole configuration is not suitable for use on a compensated head, since the static orifices are not at a single axial position on the head, but are spread over a considerable distance, which would make the design of accurate compensation extremely difficult.

A simple asymmetric two orifice configuration was chosen for the BAC 221 pitot-static head. This configuration was considered to be the best compromise over the wide range of incidence and Mach number of the aircraft. The head was designed and manufactured by the Rosemount Engineering Company and is described in section 3.

3 DESCRIPTION OF THE PITOT-STATIC HEAD

General views of the pitot-static head mounted on the nose of the BAC 221 are shown in Fig.3. A general arrangement drawing of the head and wind vane assembly is shown in Fig.4*.

The nose of the head is conical and the pitot hole has a conical inlet. There is a small drain hole on the bottom generator of the head which allows any water, which enters the pitot line through the pitot inlet to escape.

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* The wind vanes which sense aircraft incidence and sideslip angles are located on a mounting integral with the pitot-static head assembly. The complete assembly was wind tunnel tested by the Aircraft Research Association[12].

There are two static pressure orifices, 0·070 inch in diameter, disposed at ±37·5° to the downwards vertical through the axis of the head. The head compensation is provided by the 1° change of taper just aft of the orifices.

In order to design the necessary compensation profile on the head, the manufacturer was supplied with details of the shape of the aircraft fuselage, and with limited unpublished data on the pressure errors of the Fairey Delta 2, from which the BAC 221 was converted, and which had closely the same nose shape. Due to the very slender fuselages of these aircraft, their uncompensated pressure errors are relatively small. A change of head taper was thus considered to provide adequate compensation, rather than the sharply waisted compensation profile required for such a large aircraft as the XB-70[9].

Details of the head, which also incorporates an improved de-icing heater, compared to that in current British heads, are given by the manufacturer in Ref.13.

4 TESTS MADE

4.1 13ft x 9ft tunnel, R.A.E. Bedford

Three sets of tests were made in this low speed tunnel and are summarised in Table 1. The first set was for confirmation that the head performance was safe for use in flight and also investigated errors due to head misalignment in roll. The two further sets of tests investigated the head performance in detail. Except for a limited investigation of the effect of Reynolds number, all the tests were made at a wind speed of 200 ft/sec. The pitot and static pressures were read on Betz manometers. No tunnel interference corrections were necessary.

Incomplete heads were tested in Set 1 and Set 2 of the tests. Both heads were complete back to the 0·82 inch diameter station shown in the general arrangement of the head assembly in Fig.4. In Set 1 the head terminated in a 0·82 inch diameter section which was mounted coaxially in the end of a circular tube 1·5 inches in diameter. The front of this tube tapered forward, over a length of 2 inches, to a diameter slightly larger than the head diameter of 0·82 inch. The front of this taper was about 10 inches aft of the static orifices. The mounting arrangements were similar in Set 2 but, in order to minimise the influence of the 1·5 inch diameter mounting tube, a longer 0·82 inch diameter section was included so that the front of the mounting tube was 23 inches downstream of the static orifices. The Set 2 arrangement was also tested with the conical fairing in its flight position relative to the static holes, Set 2b.

Set 3 were the most comprehensive tests made on the head in the 13ft × 9ft tunnel. The head assembly was complete back to the front of the 2·80 inch diameter parallel section on which the wind vanes are normally mounted (Fig.4). This section was replaced by a long 2·80 inch diameter tube to which the head was bolted coaxially.

The centres of rotation in the incidence and sideslip planes were 50 inches downstream of the static orifices in Sets 2 and 3 but in Set 1 the centre of rotation in the incidence plane was about 16 inches behind the static orifices, and about 25 inches in the sideslip plane. In all the tests the assemblies were mounted on wire rigs.

4.2 8ft × 9ft tunnel, Aircraft Research Association

These were very comprehensive tests on both the pitot-static head and the wind vanes and the results are reported in Ref.12. For completeness it is necessary to include brief details here.

The complete assembly shown in Fig.4 was bolted to a 5 inch long conical adaptor, with a downstream diameter of 4 inches, which was in turn attached to a conical sting with a total taper angle of 8°. The sting could be rolled about its axis, and rotated in a vertical plane, thus enabling any desired combination of incidence and sideslip to be set. The centre of rotation in the vertical plane was at the static orifice station, so that the orifices remained at the same point on the tunnel centre-line.

The pitot and static pressures were measured by means of Statham differential pressure transducers, with tunnel settling chamber and tunnel plenum static pressure respectively as references. All signals during the tests were recorded digitally on punched cards. Corrections were applied for non-alignment of the flow with the tunnel axis and for the differences between the reference pressures and the true pitot and static pressures in the empty tunnel.

At subsonic speeds, the pressure field of the sting support system is felt at the head static orifices and the static pressure error induced at Mach numbers less than about 0·9 was estimated to be equivalent to a measured pressure coefficient of $C_{ps} = 0.002$ where

$$C_{ps} = (p - p_o)/\tfrac{1}{2}\rho V_T^2 \qquad (1)$$

p = measured static pressure

p_o = true static pressure

ρ = air density

V_T = true airspeed.

The error is likely to be slightly greater at transonic speeds but is of course zero at speeds above which the flow past the static orifices is sonic.

The tests were made in the Mach number range $0{\cdot}26 \leqslant M \leqslant 1{\cdot}4$ at values of incidence and sideslip shown in Table 2.

4.3 Flight tests

Before discussing the tests made, it is convenient to describe the instrumentation used.

4.3.1 Test instrumentation

The only instrument fitted specifically for these tests was one of three sensitive aneroids, of limited range, suitable for use as static pressure measuring instruments at nominal aircraft altitudes of sea level, 9000 ft and 35000 ft respectively. For a given flight at a particular altitude, the correct aneroid was connected to the aircraft static pressure system. A camera, fitted to photograph the dial face of the aneroid, was operated by the same switch as two photographic trace recorders upon which the following quantities were recorded.

Aircraft indicated static pressure

Aircraft indicated dynamic pressure

Outside air total temperature

Normal acceleration

Fuel used.

Prior to the commencement of the tests, the static pressure lag inherent in the static system of the aircraft due to pipe lengths and instrument volumes was measured using the method described by Smith[14]. This measurement was made both with an aneroid fitted and also with the normal aircraft system alone. The static pressure lag time constant for the former configuration was about twice that for the latter. For example, at 35000 ft the time constant with the aneroid fitted was 1·65 sec compared with 0·89 sec for the normal system with less volume. The latter figure is itself about twice that of the Fairey Delta 2 system which had a comparable, although not identical, volume and, as may be expected, it appears that the two hole static orifice configuration of the BAC 221 pitot-static head, introduces rather more lag than the relatively large area slot configuration of the Mk.9 head[1] fitted to the Fairey Delta 2.

4.3.2 Flight tests made

The flight tests to determine the aircraft pressure errors were made in two parts, at low altitude and high altitude respectively, using the normal aneroid method[15].

The low altitude tests consisted of constant speed fly-pasts, at an altitude of approximately 100 ft, during which the aircraft was photographed from an elevated position to one side of the flight path. A photograph was taken as the aircraft flew over ground marker boards, a known distance apart, which also appeared in the photograph. The height of the aircraft could thus be determined. Simultaneously with the ground based photograph, a photograph was taken in the aircraft of the aneroid, and the paper traces in the recorders were automatically marked.

Immediately prior to take-off, and after landing, the aircraft was parked between the marker boards, and photographs and trace records were taken to confirm that no significant changes in instrument zeroes had occurred, and to provide a static pressure datum for use during the analysis. Using this datum, the ambient static pressure at the aircraft altitude during each fly-past was calculated for comparison with the static pressure measured in the aircraft.

Most of the high altitude tests were made in company with the calibrated Javelin aircraft operated by A.& A.E.E. Boscombe Down. The pressure errors of this aircraft have been accurately determined[15] and it is used as a standard for direct comparison. The test technique required either flying the two aircraft in formation, at speeds where their flight envelopes overlapped, or flying past the Javelin at the higher speeds of which the BAC 221 is capable. In either case, a photograph of the BAC 221 was taken, using a fixed sideways looking camera in the Javelin, simultaneously with photographs taken in both aircraft of aneroids connected to their respective static pressure systems, and with running of the photographic trace recorders. Such tests were made at nominal aircraft altitudes of 9000 and 35000 ft.

From the measured Javelin static pressure, and from the small height difference between the aircraft determined from the photograph, the true static pressure at the BAC 221 altitude was calculated for comparison with the static pressure measured in the aircraft.

Data to assist in defining the aircraft pressure errors at transonic speeds were also obtained from continuous recordings of aircraft static and dynamic pressure during unaccompanied accelerations and decelerations through the transonic region. At transonic and supersonic speeds the BAC 221 cannot be flown at constant speed in level flight due to the non-variable reheat system on the aircraft. Small corrections for the lag in the pressure sensing systems of the aircraft under such conditions were applied using the method of Ref.14.

5 WIND-TUNNEL RESULTS

The measured pitot and static pressures are presented in the form of pressure coefficients. The pitot and static pressure coefficients, C_{pp} and C_{ps} respectively, are defined as the difference, or error, between the measured pressure and the true free stream pressure, divided by the kinetic pressure, (see equation (1), section 4.2). It will be convenient to discuss the dependence of these coefficients on Mach number and on flow direction angle (or incidence and sideslip angles) separately.

The estimated accuracy of the 13ft × 9ft tunnel results is ±0·001 in in both C_{pp} and C_{ps}; the corresponding A.R.A. tunnel accuracy is thought to be ±0·002 except at Mach numbers of 1·03 and 1·06, where there may be tunnel wall interference, and $M = 1·3$ and 1·4, where the tunnel longitudinal static pressure distribution is not very smooth; in all these conditions the accuracy in C_{ps} is degraded by an unknown amount.

Manufacturer's predictions of the performance of the head in isolation, with none of the adaptors for mounting it on the aircraft nose attached, are available[13]. Such a head is equivalent to that tested in Set 2 of the 13 × 9 tests but without the mounting tube needed for those tests. There is thus necessarily a difference between the head configuration to which the manufacturers prediction applies, and the configuration tested in wind tunnels, but the resulting discrepancy in the measured static pressure coefficient should be extremely small. There should be no difference at supersonic speeds where changes in shape have no influence on the pressure field upstream. Broadly speaking, due to the increased cross-sectional area introduced by attachments or mounts, the static pressures measured at subsonic speeds would be expected to be slightly more positive (or less negative) than those predicted for the head alone. This point should be borne in mind when comparing the manufacturer's predictions with the wind-tunnel test results presented in this Report. However, changes in pressure due to changes, for example, in incidence or sideslip should be directly comparable between tunnel results and predictions.

5.1 Effect of Mach number on the static pressure measured

The static pressure coefficients measured during the A.R.A. and the 13ft × 9ft tunnel tests are shown as a function of Mach number, at zero incidence and sideslip in Fig.5; unfortunately different heads were available at the time of each set of 13ft × 9ft tunnel tests and of the A.R.A. tests so that a direct comparison of results from the same head in different tunnels is not available. The A.R.A. results at Mach numbers less than unity have been corrected for the pressure field of the sting support by subtracting 0·002 from the measured pressure coefficients (see section 4.2). Due to the uncertainty, mentioned earlier, in the A.R.A. results at $M = 1{\cdot}4$, the results above $M = 1{\cdot}3$ have been extrapolated on the basis of results from a similar pitot-static head[5,16]. The manufacturer's prediction is also shown for Mach numbers less than unity.

Throughout the Mach number range, the measured static pressure coefficient is negative. At subsonic speeds the negative value is designed to compensate for the positive pressure field induced ahead of the aircraft; at supersonic speeds the measured negative value is undesirable and leads to a residual error in flight, although the error tends to zero quite rapidly with increasing Mach number.

At low speeds there are significant differences between the static pressure coefficients measured in the A.R.A. tunnel and in the 13ft × 9ft tunnel, and also between the various 13ft × 9ft results. The most valid comparison between the results from the two tunnels uses Set 3 of the 13ft × 9ft tests which relate to the test geometry most similar to that in the A.R.A. tunnel. It is seen that the variation of C_{ps} with Mach number, or more properly Reynolds number at these low speeds, in the two tunnels is in the opposite sense, although the variation in Set 3 may not be experimentally significant. There is also a very significant and unexplained difference in the measured level of compensation. The change with Reynolds number in Set 2a of the 13ft × 9ft tests may be significant but is in the opposite sense to that predicted by the manufacturer and the A.R.A. results. It is possible that the head used may have a defect such as burrs on the static holes or a manufacturing error.

The difference in Reynolds number between the two tunnels is not a likely cause of the discrepancies. The Reynolds number was about $1{\cdot}3 \times 10^6$ per ft at $M = 0{\cdot}18$ in the 13ft × 9ft tunnel and about 2×10^6 per ft at $M = 0{\cdot}3$ in the A.R.A. tunnel (see Table 3). This small difference is unlikely to be significant and in any case the variation of the results with Reynolds number in the 13ft × 9ft tunnel is in the opposite sense to that required to bring the results from the two tunnels together. The possibility remains of errors in tunnel technique but no satisfactory explanation has been found.

The differences in measured pressure between the various configurations tested at a constant Mach number of 0·18 in the 13ft × 9ft tunnel show the small effects of the different mounting configurations. The nearest configuration to an isolated head, Set 2a, has the most negative coefficient, and, Set 1, the configuration with the largest cross-sectional area increase downstream of, and close to, the static holes, has, as would be expected, the least negative coefficient. The difference is 0·0025 in C_{ps}. The configurations for Sets 2b and 3, which are similar, and intermediate in area change between Sets 1 and 2a, give consistent results within the experimental accuracy. However, the variation, discussed earlier, of the results from Set 2a with Reynolds number, suggests that the results from this head should be viewed with caution.

The manufacturer's prediction of the variation with Mach number of the compensation provided is seen from Fig.5 to be in good agreement with the A.R.A. tunnel results, although there is an approximately constant small difference between theoretical and measured values. The difference is in fact slightly greater than is shown since the tunnel results have not been corrected for the small positive pressure increment due to the influence of the head fairing and the wind vane assembly. However, the 13ft × 9ft tunnel results suggests that the effect of the fairing is no greater than an increment of $C_{ps} = 0{\cdot}0015$.

The head used in the flight tests is that which was tested in the A.R.A. tunnel. Since unexplained differences in head performance occurred it is considered that the A.R.A. results should be used as the basic data for interpreting the results of flight tests.

5.2 Effect of flow direction angle on the static pressure sensed

For a pitot-static head with a finite number of static orifices, two parameters are required to specify the flow direction relative to the holes. These parameters may either be the angle between the flow vector and the head axis (the flow direction angle) plus the orientation in roll of the flow vector relative to the plane of symmetry through the head axis, or the angles of incidence and sideslip of the head. The former pair has more fundamental significance but in flight the flow direction is derived directly from the wind vanes in terms of incidence and sideslip angles. Thus the test ranges were based on flight test requirements and are therefore better conditioned when expressed in terms of incidence and sideslip. For these reasons the wind-tunnel results showing the effects of flow direction angle are presented in terms of incidence and sideslip.

Fig.6 shows the variation of measured static pressure with incidence at zero sideslip. At each Mach number the value of C_{ps} at zero incidence has been taken as a datum and subtracted from the measured coefficients at finite angles of incidence to yield C_{pso}. Most of the results are from the A.R.A. tunnel but the 13ft × 9ft tunnel results and the manufacturer's predictions are included for comparison. At subsonic speeds there is in general no significant effect of Mach number except at $M = 0{\cdot}5$ where the behaviour of the head is rather different from that at lower or higher Mach numbers; the reason for this is not known. The results from the 13ft × 9ft tunnel are slightly different from the A.R.A. results but the magnitude of the difference is small. The manufacturer's prediction tends to be slightly optimistic, particularly at negative incidence at $M = 0{\cdot}3$ where zero error is predicted, but comparison with the prediction for $M = 0{\cdot}6$ suggests that the $M = 0{\cdot}3$ prediction may reflect the experimental accuracy of the tests upon which the prediction is based.

There is a definite change in the behaviour of the head in the transonic speed range. At subsonic speeds, the errors in measured pressure are predominantly negative, but at transonic and supersonic speeds the errors are positive, and increase positively with incidence. This change in character is probably due to the pressure rise through the head bow shock wave at supersonic speeds, and it seems that a longer nose section ahead of the static orifices is desirable. Gracey[6] recommends that the static orifices should be about 10 head diameters downstream of the nose, but on the BAC 221 head the orifices are about 7 diameters downstream.

Throughout the Mach number range the compensated head is rather more sensitive to incidence than the Mk.10 head[10,16], particularly at supersonic speeds. This is not surprising since the static orifice configuration on the compensated head is very simple compared with that on the Mk.10 head which would not be suitable for use with aerodynamic compensation.

The variation of measured static pressure with sideslip at zero incidence is shown in Fig.7. As in Fig.6, the values of C_{pso} are referred to the value of C_{ps} at zero flow direction angle. Since the static orifices are symmetrical about the vertical plane through the head axis, the sensitivity to positive and negative sideslip should be the same. Within the expected experimental accuracy the head performance is considered to be reasonably symmetrical. The results at $M = 1{\cdot}06$ are considered to be unreliable due to tunnel wall interference. The low speed results from the 13ft × 9ft tunnel show some evidence of an asymmetry of about $\beta = 1^\circ$, which is probably due to misalignment of the head with the tunnel flow. Correction for this asymmetry would bring the results from the two tunnels into very good agreement, both with themselves and with the manufacturer's low speed prediction. The head is rather less sensitive to sideslip than the Mk.10 head[10] and about the same as the Mk.9 head[1,2].

The performance of the head at combined angles of incidence and sideslip is shown in Fig.8 for the 13ft × 9ft tunnel and Fig.9 for two Mach numbers in the A.R.A. tunnel. The results are again referred to a datum C_{ps} at zero flow direction angle. It is at once obvious from the 13ft × 9ft tunnel results that the head performance at combined angles of incidence and sideslip could not have been predicted by compounding the results at zero incidence and zero sideslip in some simple fashion. This should not of course be unexpected for a head with an asymmetric orifice arrangement. The sideslip angle achieved in flight rarely exceeds 6° and it is seen from the 13ft × 9ft tunnel results that, throughout the incidence range investigated, the pressure errors due to flow direction angle will be very small. Comparison of the 13ft × 9ft results with the A.R.A. results at $M = 0{\cdot}26$ in Fig.9 shows detail differences in the head performance but the pattern is broadly similar within the limited range of incidence of the A.R.A. tests. The A.R.A. results at $M = 1{\cdot}3$ are typical of the high Mach number results.

The effect of the datum error of 3° in the setting of the head in roll, which was investigated during Set 1 of the 13ft × 9ft tests, was found to be of the same order as the experimental accuracy up to the nominal angles investigated of $\alpha = 20^\circ$ with $\beta = 0^\circ$, and $\beta = 10^\circ$ with $\alpha = 0^\circ$. This error in roll is equivalent to introducing a sideslip error of slightly less than 1° at $\alpha = 20^\circ$ and $\beta = 0^\circ$, nominally, and the result is consistent with the data presented in Fig.8. This test was made because the manufacturer specifies a very tight tolerance of ±0·1° in roll alignment and it appears from the results that such accuracy in alignment is unnecessary.

5.3 Pitot pressure

Since the pitot inlet is symmetrical about the head axis the measured pitot pressure should be equally sensitive to incidence and sideslip. In Fig.10 the pitot pressure results from the 13ft × 9ft tunnel are presented as a function of the flow direction angle, γ, which is the vector sum of the angles of incidence and sideslip.

At $\gamma = 0^\circ$ there is a very small measured pitot pressure error of $C_{pp} = -0{\cdot}001$ which is considered to be just experimentally significant. The error is probably due to flow through the drain hole in the pitot line. There is no increase in error until about $\gamma = 10^\circ$. The sensitivity of the sensed pitot pressure to large flow direction angles is extremely low; at $\gamma = 20^\circ$ the error is only about $C_{pp} = -0{\cdot}01$ which agrees well with earlier results for a similar pitot entry[11]. Current British pitot-static heads such as the Mk.9 and Mk.10 with hemispherical noses have very large pitot errors of the order of $C_{pp} = -0{\cdot}1$ at $\gamma = 20^\circ$ [1,2].

There is some evidence of asymmetry in the results from the 13ft × 9ft tunnel, since at the higher flow direction angles the experimental points do not quite collapse as a function of γ only. This could be due either to some slight asymmetry in the tunnel flow, to misalignment of the plane of the pitot orifice relative to the head axis or to an increase, with flow direction angle, of the error due to the pitot drainhole. The asymmetry in the results is evident only at the higher flow direction angles which were investigated in Set 3 of the 13ft × 9ft tests; comparison between the various heads tested, which should reveal any head asymmetry, is not therefore possible.

The tests in the A.R.A. tunnel[12] were made over a smaller range of flow direction angle than those in the 13ft × 9ft tunnel and the point at which the pitot pressure became significantly sensitive to flow direction angle was not reached. The A.R.A. results have not been presented here. At supersonic speeds where there is a normal shock ahead of the pitot inlet the pitot pressure sensed was closely that expected from theoretical normal shock relationships.

The manufacturer's prediction[13] of the pitot performance up to $\gamma = 18^\circ$ at subsonic speeds is included in Fig.10. The possibility of an additional error of up to $C_{pp} = -0{\cdot}001$ at supersonic speeds due to flow through the pitot drain hole is acknowledged. When this error is included, the manufacturer's prediction is seen to be slightly optimistic.

6 FLIGHT RESULTS

It is most convenient to present the flight pressure error results, corrected to zero incidence, as a function of Mach number. This allows easier comparison with wind-tunnel data and with the manufacturer's prediction of the performance of the pitot-static head. The aircraft incidence was estimated from low and high speed wind-tunnel data[17,18] for the normal force coefficient appropriate to the known aircraft weight and flight conditions, since the incidence vanes fitted at the time of the flight tests were unreliable. A small correction for the upwash induced at the static orifices by the wing lift distribution, was applied using a modification of the analysis of Jones[19], as developed by Berndt[20] for wind-tunnel interference on small aspect ratio wings. The wind-tunnel data[12], presented in Fig.6, were then used to subtract the effect of incidence on the pitot-static head from the measured flight pressure errors. Since the pitot-static head is relatively insensitive to incidence, errors introduced by using wind-tunnel data to estimate the aircraft incidence, and by the wing upwash correction, are thought not to be significant. A considerable portion of the aircraft nose boom was simulated in the wind tunnel and no correction for boom upwash is considered necessary. No correction was applied for the variation of the aircraft's pressure field with incidence. This effect, which appears as a change of pressure error due to change of altitude at constant Mach number, will be shown not to be important for the BAC 221.

The flight static pressure errors, corrected to zero incidence, are shown in Fig.11 as a function of Mach number for the three measuring altitudes, with the aircraft in the clean and the approach configurations. In the latter configuration the aircraft nose is drooped by 8° and the probe incidence is reduced by this amount. A single mean curve has been drawn through all the flight results; at Mach numbers less than 0·9, this curve is the best least squares quadratic fit through the experimental points.

The level of uncertainty introduced by the scatter in the calibration of the pressure errors of the pacer Javelin aircraft was found to be a function of altitude only[15] and is equivalent to about ±19ft at 9000 ft and ±45ft at 35000 ft. The resultant possible pressure coefficient error bands, with respect to the mean flight curve for the BAC 221, are defined by the dashed curves in Fig.11. The curves in the transonic region have been omitted for clarity.

At subsonic speeds, the majority of the results obtained using the calibrated Javelin at altitudes of 9000 ft and 35000 ft are within the uncertainty of the Javelin's calibration, but the scatter is rather greater than would be expected from this source alone at supersonic speeds. Additional errors of course arise from the other measurements taken during the present tests and these errors may be greater at supersonic speeds at which the BAC 221 flew past the Javelin rather than formated on it. The scatter on the sea level results is equivalent to altitude errors of about ±20 ft at low speeds. The accuracy obtained from the flight results is considered to be adequate to define the performance of the pitot-static head.

At speeds where results are available at three test altitudes, and with two aircraft configurations, it is considered that, within the experimental accuracy, there are no significant effects of altitude or aircraft configuration. Such effects have been found on a Lightning aircraft[5] and are thought to be due to changes in the pressure field of the aircraft with incidence, and in engine mass flow, both changes occurring with changes of altitude at constant Mach number. The compensation on the Lightning head was very good at low altitude but much too large at high altitude. The BAC 221 is a much more slender and less bulky aircraft than the Lightning and the pitot-static head of the former is relatively remote from the engine intakes compared with the installation on the latter. Altitude effects would, therefore, probably be less on the BAC 221.

The definition of the behaviour of the pressure error in the transonic region was assisted by the measurements taken during transonic accelerations and decelerations shown in Fig.12. These results were obtained from time histories of aircraft indicated static and dynamic pressure measurements. Smooth curves were faired through the experimental records after lag corrections were applied, and the departure from these curves provided a measure of the pressure error in the transonic range. Since the aircraft height was not necessarily constant during the relevant periods, the data do not define the pressure error absolutely, but rapid changes in pressure, such as those shown in Fig.12 at an indicated Mach number of about 1·01 are attributable to changes in pressure error only. The magnitude of the 'transonic jump' in pressure error is seen to be reasonably defined. The differences between the four curves are probably due to the aircraft having a finite rate of climb or descent in some cases and to insufficiently accurate pressure lag corrections during the rapid pressure changes.

At Mach numbers greater than unity, where the pressure field of the aircraft is not felt at the static orifices of the head, the flight results shown in Fig.11 should be directly comparable to the tunnel results shown in Fig.5. Unfortunately the scatter on both sets of results precludes a direct comparison but it is clear that there is a significant residual error which decreases with increasing Mach number.

The manufacturer's prediction[13], included in Fig.11, of the flight static pressure error with the compensated head fitted, is in poor agreement with the flight results, although the prediction of the performance of the head alone is seen from Fig.5 to be in reasonable agreement with the A.R.A. tunnel results on the same head. It appears that the procedure for predicting the aircraft pressure field at subsonic speeds is inadequate. At supersonic speeds, the error is rather greater than that predicted by the manufacturer; this was also the case for the compensated head on the Lightning aircraft[5]. Inspection of Table 3 suggests that variation of Reynolds number is unlikely to be a factor in the disagreement.

Although the head does not perform as well as had been predicted, particularly at low subsonic speeds, it still represents a considerable improvement on many current heads. Fig.13 shows the residual altitude error corresponding to the mean flight curve in Fig.11. The altitude error throughout most of the flight range is seen to be within the limits specified in section 1. The improvement over current heads, however, is most significant in the transonic range, where the jump in pressure error is equivalent to a change in indicated height of about 370 ft, at an altitude of 35000 ft, disposed symmetrically about the true height. At the same altitude, unpublished data on the Fairey Delta 2 aircraft which had the same nose shape, and hence similar aircraft induced pressure errors, as the BAC 221, give a transonic jump of about 720 ft from -640 to +80 ft. The improvement due to the compensated head is thus considerable, both in terms of the transonic jump in indicated altitude and in the absolute altitude error at high subsonic speeds, but there is the penalty of a slightly increased error at low supersonic speeds.

It should be noted that presenting the head performance as a function of Mach number at zero incidence does mask one favourable feature in flight. At subsonic speeds the static pressure errors introduced by incidence are in general negative, and at supersonic speeds they are positive. In both ranges, the effect of incidence is thus to reduce the absolute static pressure error, although this is a small effect since the head is relatively insensitive to incidence.

No data on the pitot pressure performance have been extracted from the flight results since this is not affected by the presence of the aircraft and is covered adequately by the wind-tunnel results in section 5.3.

7 CONCLUSIONS

Following the flight and wind-tunnel tests on the aerodynamically compensated pitot-static head for the BAC 221 aircraft, the following conclusions can be drawn.

(i) The compensation is successful in reducing the static pressure errors significantly at high subsonic and transonic Mach numbers. The transonic jump in indicated altitude is only 370 ft, disposed symmetrically about the true altitude. At low supersonic Mach numbers the head introduces a small residual pressure error, always equivalent to less than 190 ft in altitude, and falling rapidly to zero with increasing Mach number. A significant but small error, equivalent to less than 100 ft, remains at moderate subsonic speeds.

(ii) The static pressure sensed is relatively insensitive to incidence and moderately so to sideslip.

(iii) The sensitivity of the pitot pressure to flow direction is extremely small.

(iv) There is a significant difference of about 0·008 between the static pressure coefficients measured at low speeds in the A.R.A. and 13ft × 9ft wind tunnels. There is also a difference in the dependence of the measured pressure on Reynolds number. These differences are unexplained, but it may be significant that the tests were made on different heads.

(v) The manufacturer's prediction of the head performance agrees reasonably well with wind-tunnel results but at low subsonic speeds the prediction of the flight static pressure error is in poor agreement with the flight results. The discrepancy is equivalent to a pressure error coefficient of about 0·017. It appears that the manufacturer has predicted the performance of the pitot-static head reasonably well but that the procedure for predicting the aircraft pressure field at subsonic speeds is inadequate.

Acknowledgements

The authors gratefully acknowledge both the co-operation of the personnel of A. & A.E.E. Boscombe Down who operated the Javelin pacer aircraft, and also the contribution of Dr. L. A. Wyatt who made the tests in the 13ft × 9ft tunnel at R.A.E. Bedford. The manufacturers, Resemount Engineering Co. Ltd., are agreeable to the publication of this paper.

Table 1

TESTS IN THE 13ft × 9ft TUNNEL

Set	Mounting	α	β	Test points
1	Complete back to 0·82 inch diameter section. Mounting tube 10 inches aft of static holes	$-5^\circ \rightarrow +10^\circ$	$-10^\circ \rightarrow +10^\circ$	α range at $\beta = 0^\circ, -5^\circ, -10^\circ$) 2·5° steps β range at $\alpha = 0^\circ$) Tests at $\alpha = 0^\circ$ and $\beta = 0^\circ$ repeated with head rolled 3°
2a	As Set 1 but with mounting tube 25 inches aft of static holes	$-2^\circ \rightarrow +2^\circ$	0°	1° steps at $\beta = 0^\circ$ plus effect of Reynolds No.
2b	Repeat 2a but with flight fairing added	0°	0°	1 point
3	Flight assembly complete back to forward end of wind vane mounting section	$-6^\circ \rightarrow +30^\circ$	$0^\circ \rightarrow +10^\circ$	All combinations of incidence and sideslip in steps of 2° in α and 1° in β. $\beta = 7^\circ$ and 9° omitted. Also effect of Reynolds No. at $\alpha = \beta = 0^\circ$

Table 2

TEST POINTS IN THE A.R.A. 8ft × 9ft TUNNEL

M	α	β	Test points
0·26	$-5° \to +15°$	$-10° \to +10°$	All combinations of incidence and sideslip in 1° steps. No tests at $\alpha = -4°, +11°, +13°$
0·5, 0·8	$-5° \to +15°$	$-10° \to +10°$	1° steps. α tests at $\beta = 0°$ only, β tests at $\alpha = 0°$ only
0·7	$-5° \to +15°$	$-5° \to +10°$	All combinations of incidence and sideslip in 1° steps. No tests at $\alpha = -4°, +11°, +13°$ or $\beta = -4°, -2°, +7°, +9°$
0·9, 0·97 1·0, 1·03 1·1, 1·3 1·4	$-2° \to +10°$	$-5° \to +5°$	All combinations of incidence and sideslip in 1° steps
0·93 1·06 1·2	$-2° \to +10°$	$-5° \to +5°$	1° steps α tests at $\beta = 0°$ only, β tests at $\alpha = 0°$ only

Table 3

COMPARISON OF FLIGHT AND WIND-TUNNEL REYNOLDS NUMBERS

Test conditions		Mach number	Reynolds number per foot
13 ft × 9 ft		0·18	1·3 × 10^6
A.R.A.		0·3	2·0 × 10^6
		1·0	4·4 × 10^6
		1·4	4·5 × 10^6
Flight	Ground level	0·2	1·4 × 10^6
		0·7	5·1 × 10^6
	9000 ft	0·3	1·7 × 10^6
		0·7	4·0 × 10^6
	35000 ft	0·5	1·2 × 10^6
		1·0	2·4 × 10^6
		1·4	3·3 × 10^6

SYMBOLS

C_{pp}	pitot pressure coefficient, $(p_T - p_{TO})/\frac{1}{2}\,\rho\,V_T^2$	
C_{ps}	static pressure coefficient, $(p - p_o)/\frac{1}{2}\,\rho\,V_T^2$	
C_{psi}	C_{ps} based on indicated values	
C_{pso}	changes in C_{ps} due to incidence or sideslip	
M	Mach number	
M_i	indicated Mach number	
p	measured static pressure	lb/ft^2
p_o	true static pressure	lb/ft^2
p_T	measured pitot pressure	lb/ft^2
p_{TO}	true pitot pressure	lb/ft^2
V_T	true airspeed	ft/sec
α	incidence angle	deg
β	sideslip angle	deg
γ	total flow direction angle	deg
ρ	air density	$slug/ft^3$

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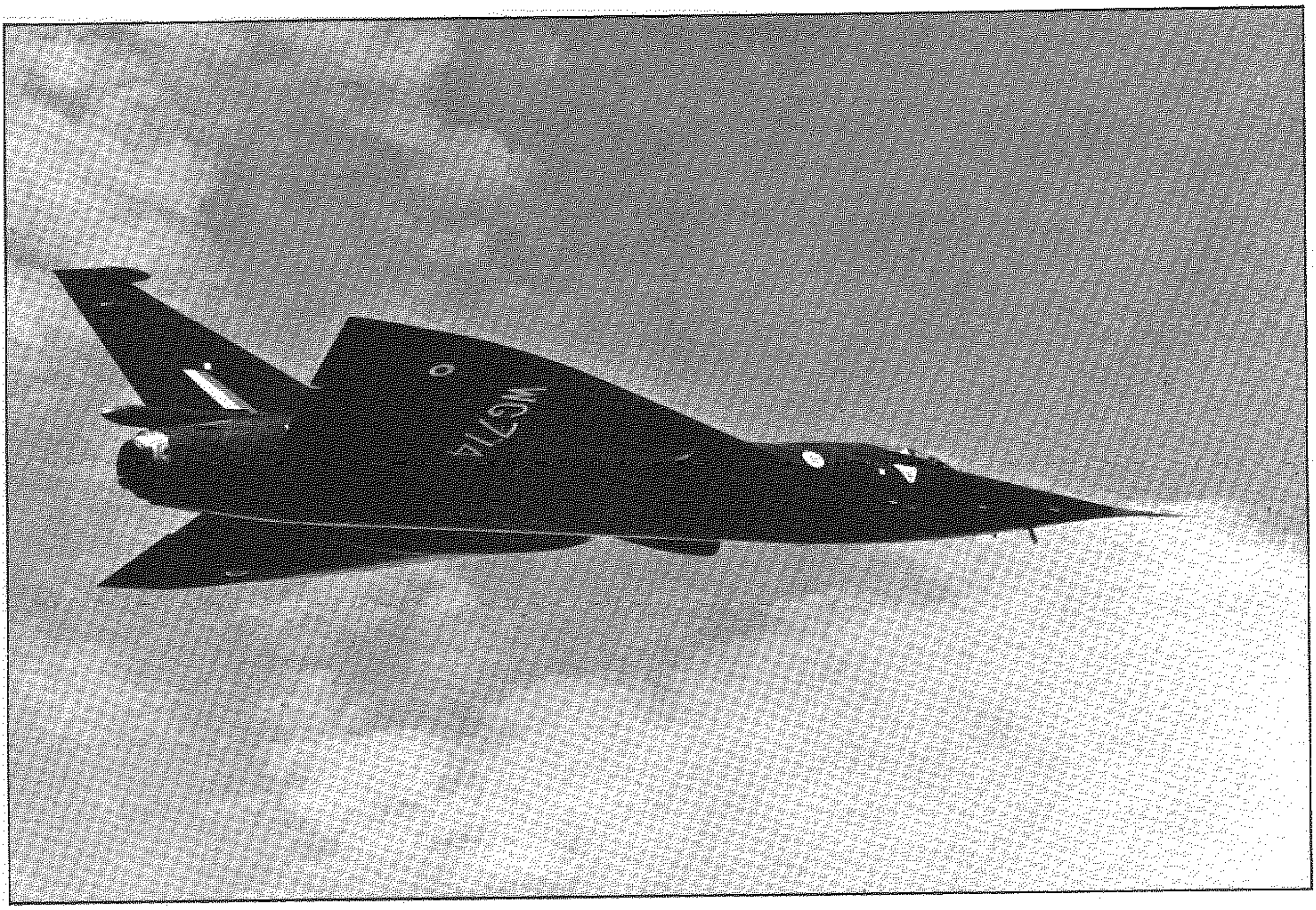


Fig.1. BAC 221

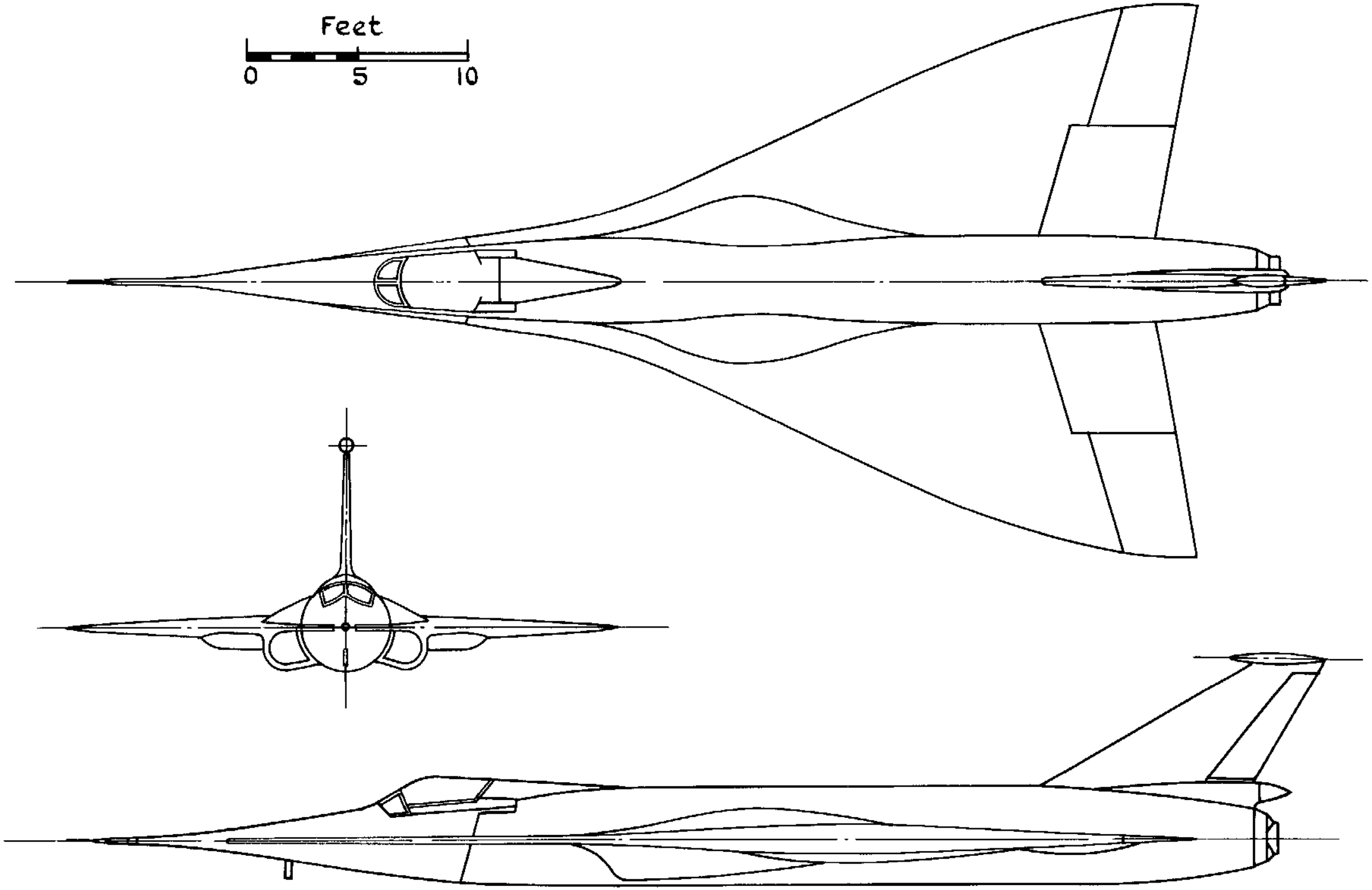
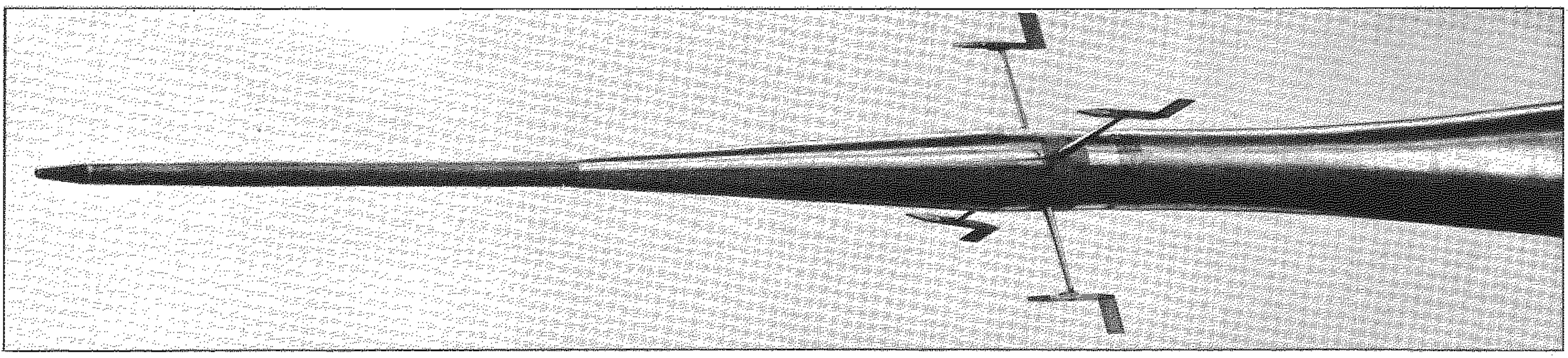
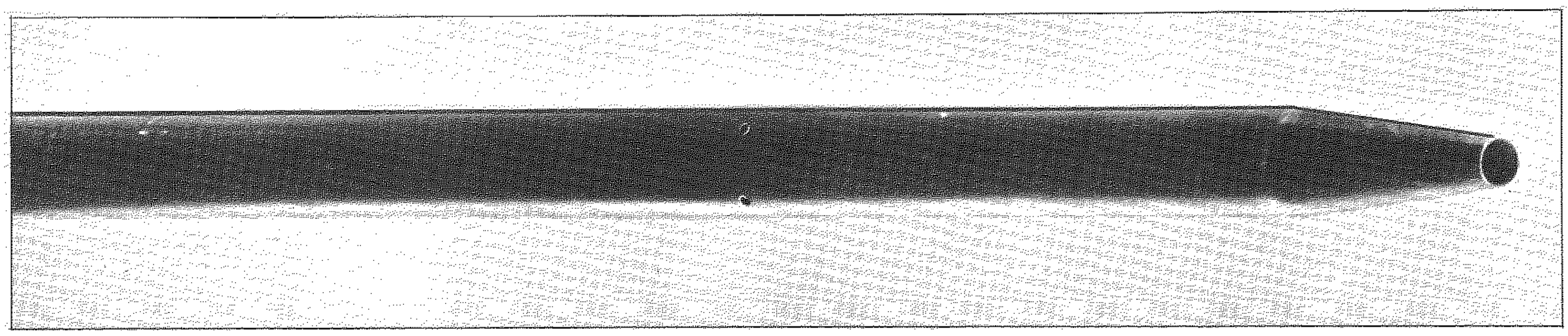


Fig. 2 BAC 221 General arrangement

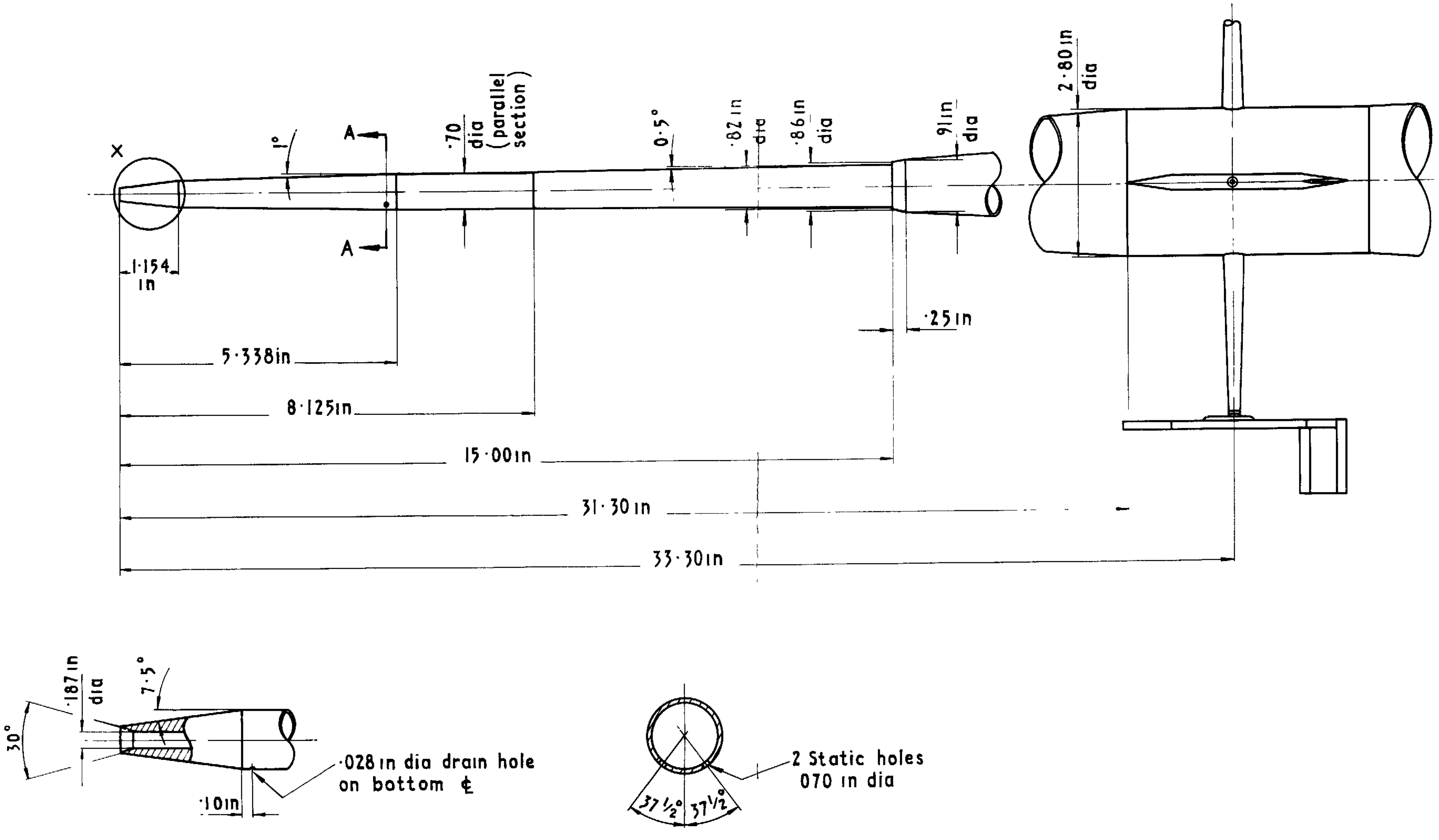


a. Complete nose boom assembly



b. Underside of pitot-static head showing static orifices

Fig.3. BAC 221 pitot-static head and nose boom assembly (from ref.12)



Enlarged detail at X

Enlarged section AA

Scale : $\frac{1}{2}$ full size

Fig.4 BAC 221 Pitot static head and wind vane assembly

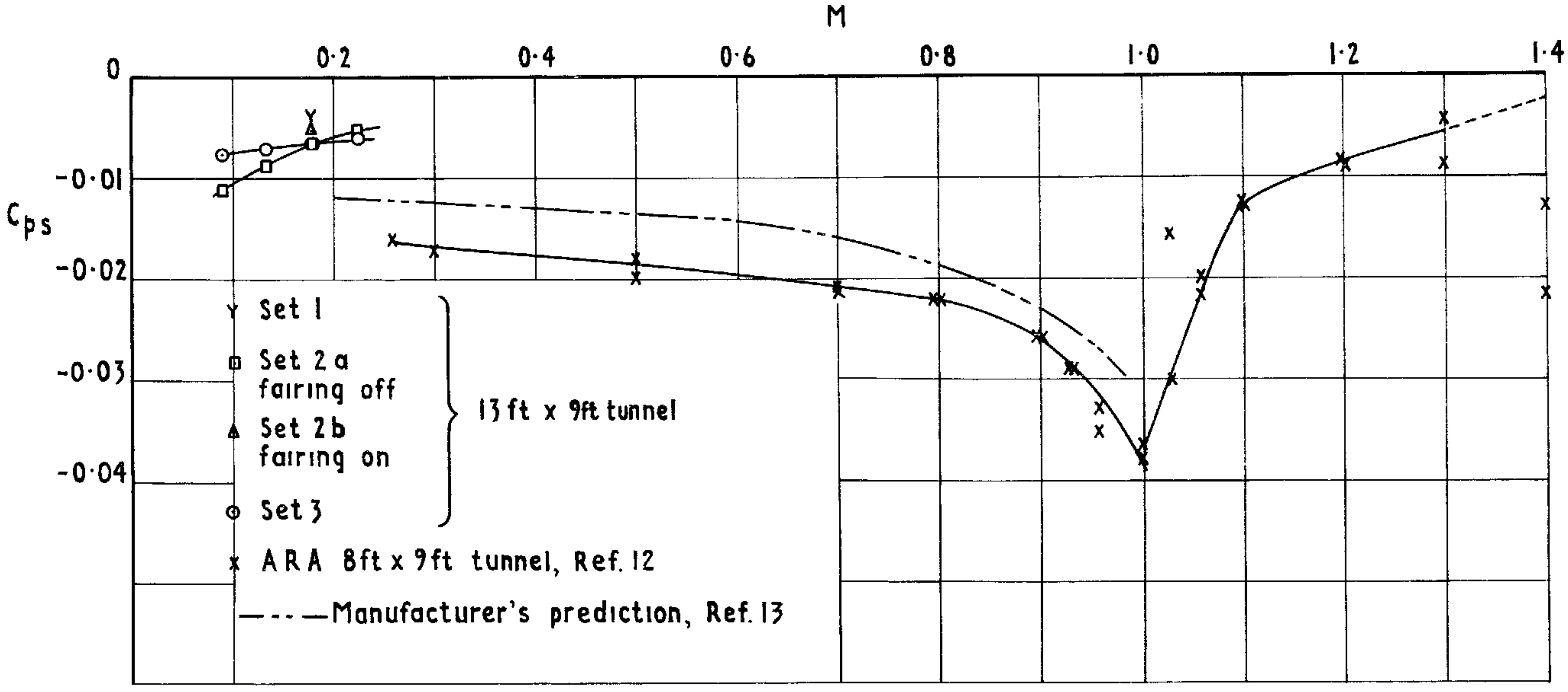


Fig. 5 Variation of head static pressure with Mach number, $\alpha = \beta = 0$

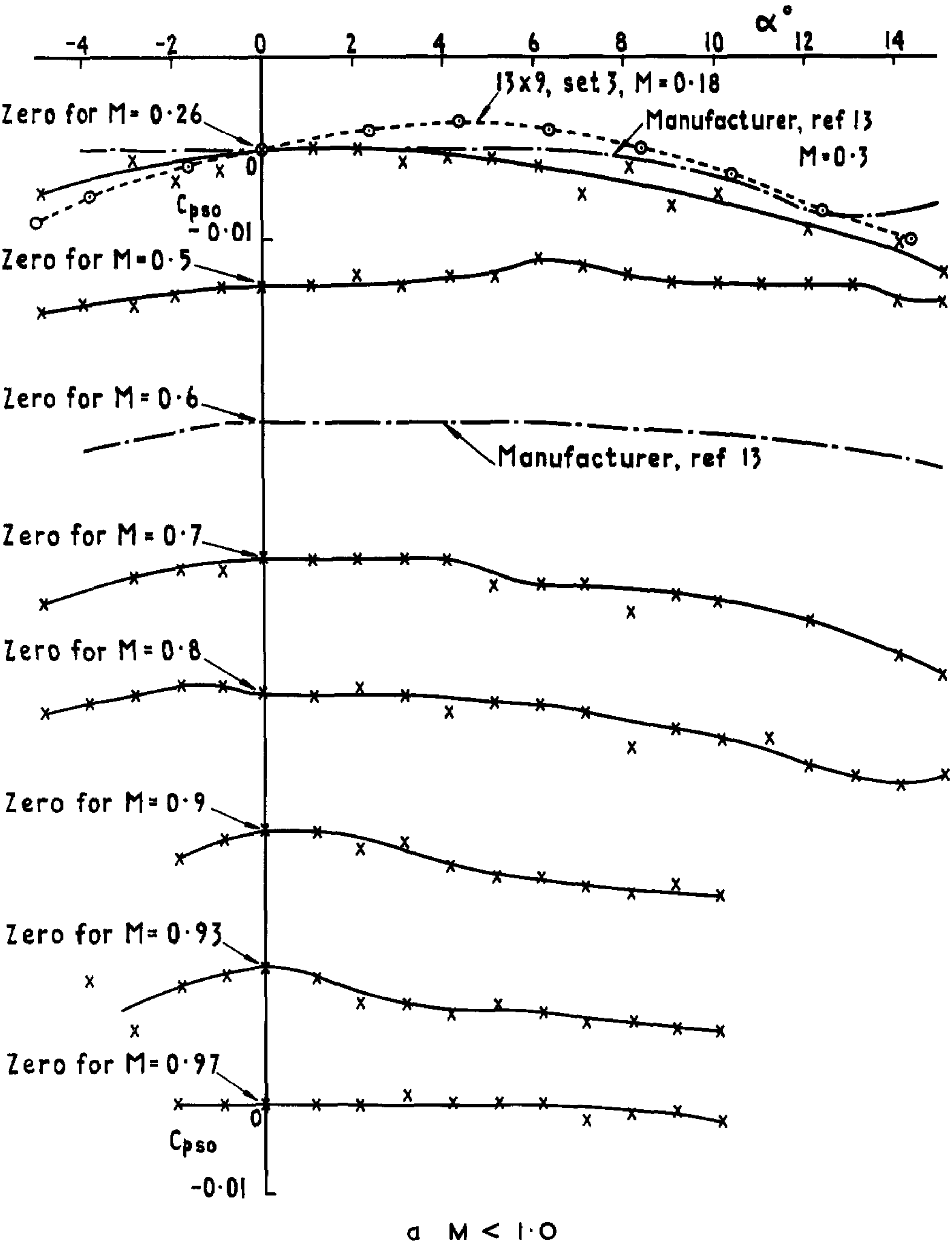
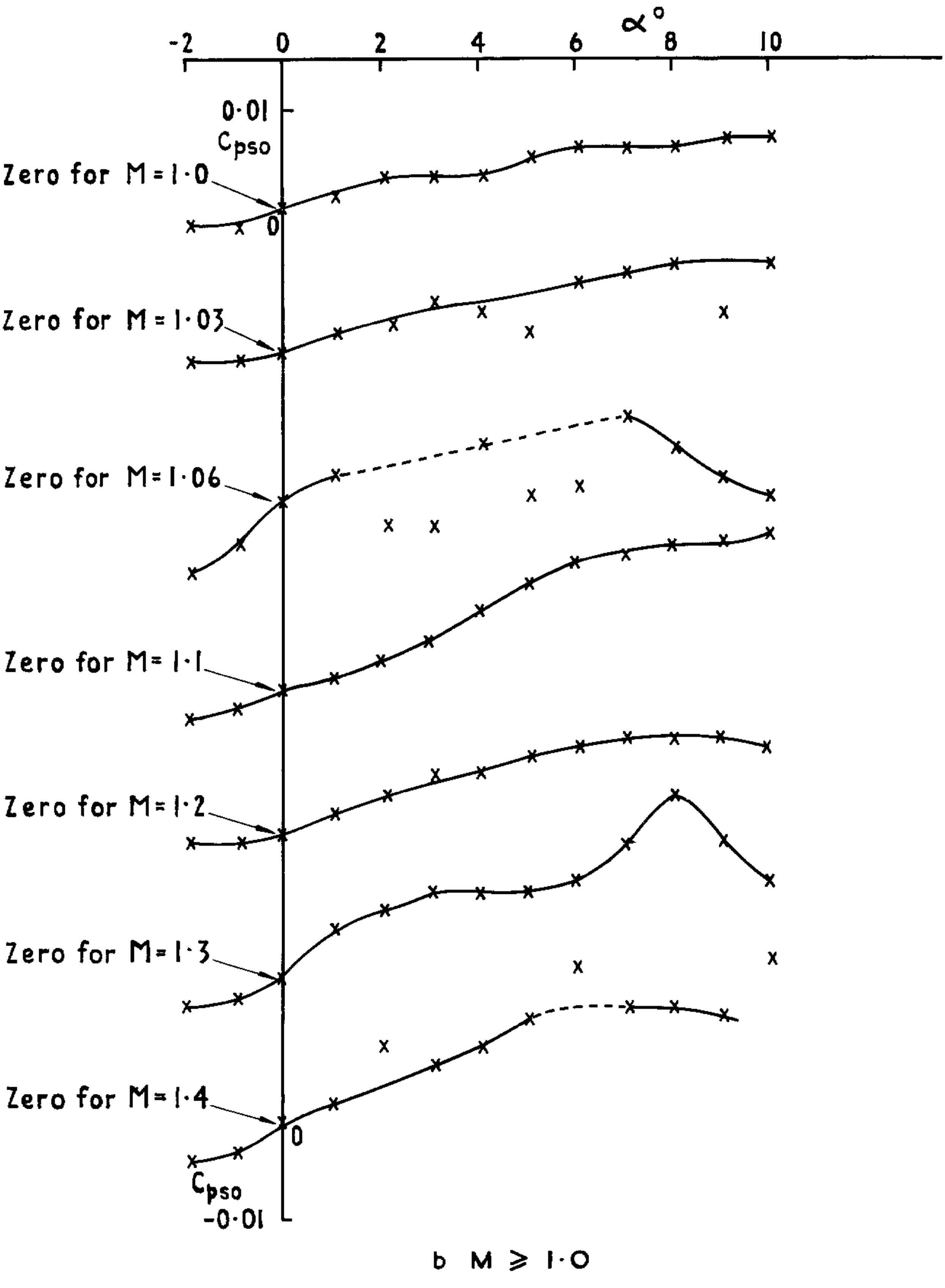
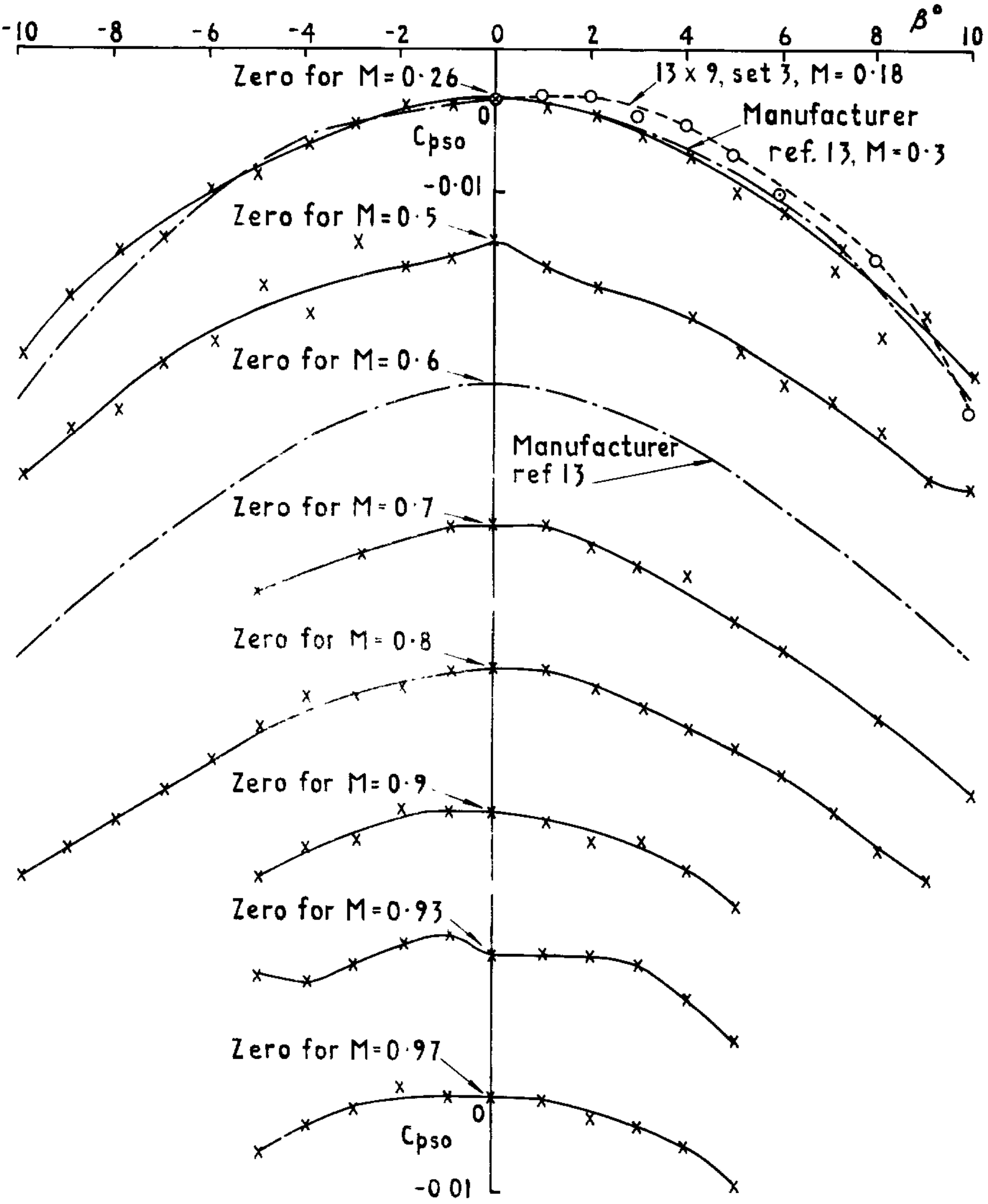


Fig. 6a Variation of head static pressure with incidence in the A.R.A. tunnel, $\beta = 0$



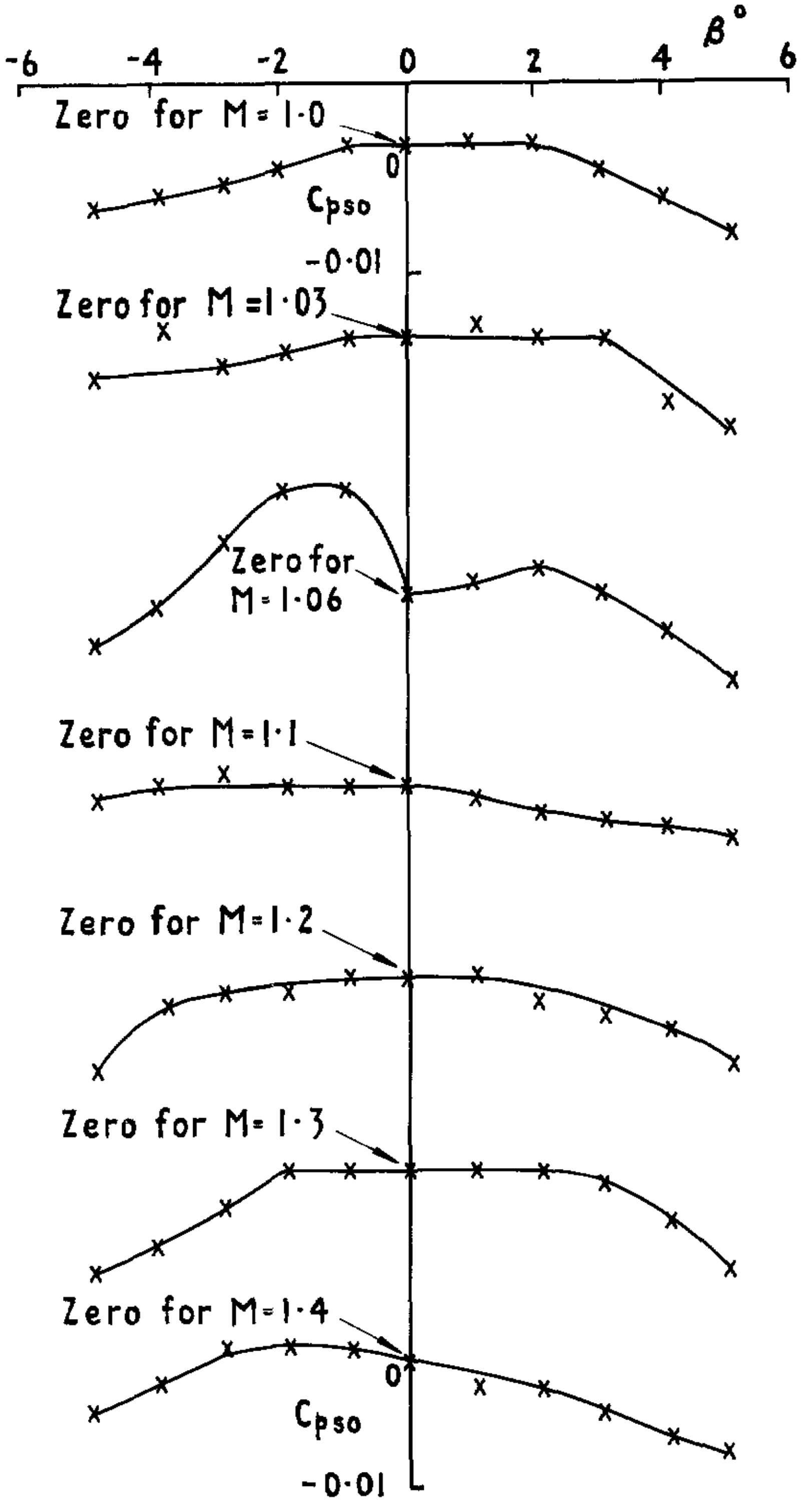
b $M \geqslant 1{\cdot}0$

Fig. 6 b Variation of head static pressure with incidence in the ARA tunnel, $\beta = 0$



a M < 1·0

Fig 7a Variation of head static pressure with sideslip in the A.R.A tunnel, $\alpha = 0$



b M ⩾ 1·0

Fig. 7b Variation of head static pressure with sideslip in the ARA tunnel, $\alpha = 0$

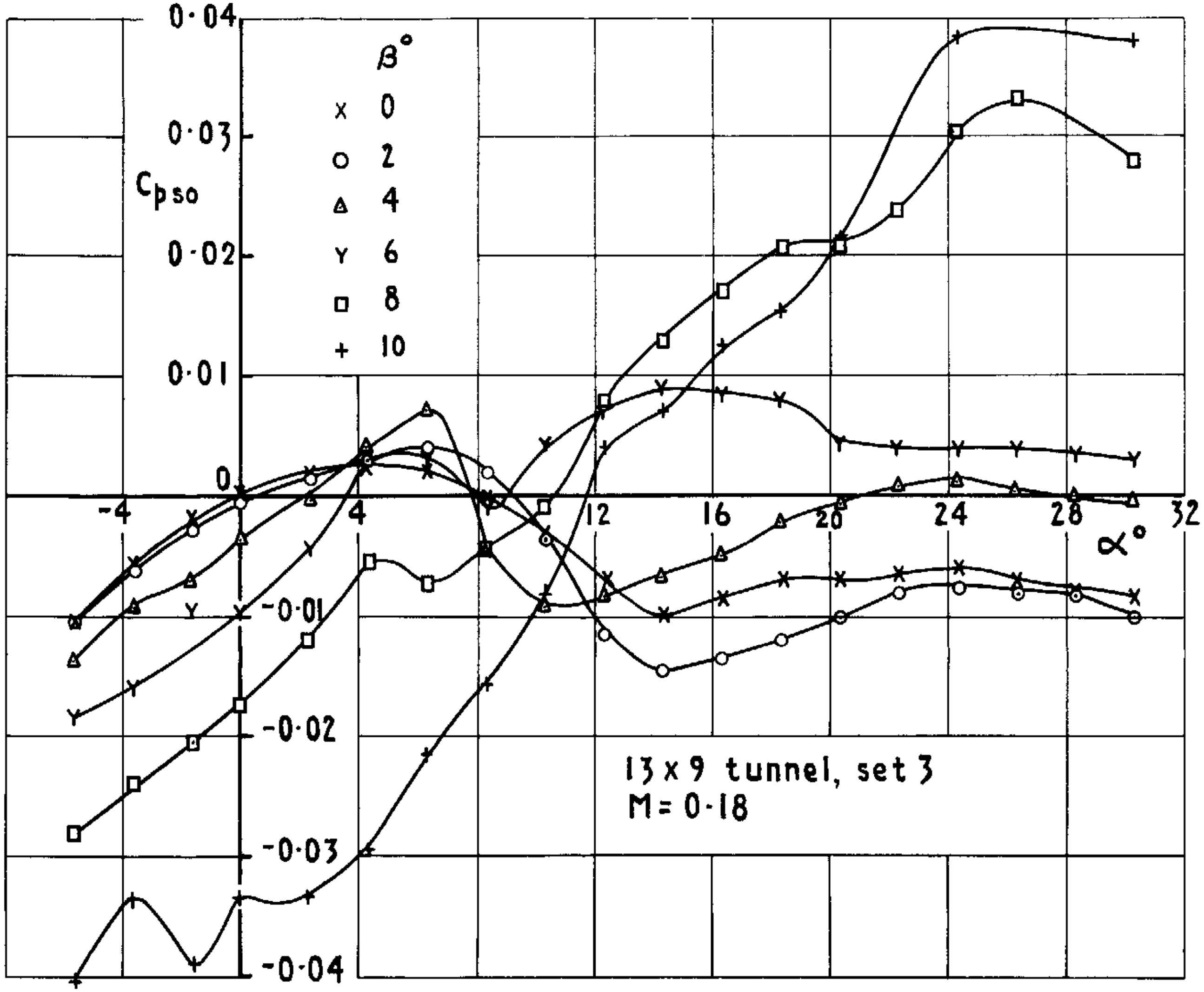
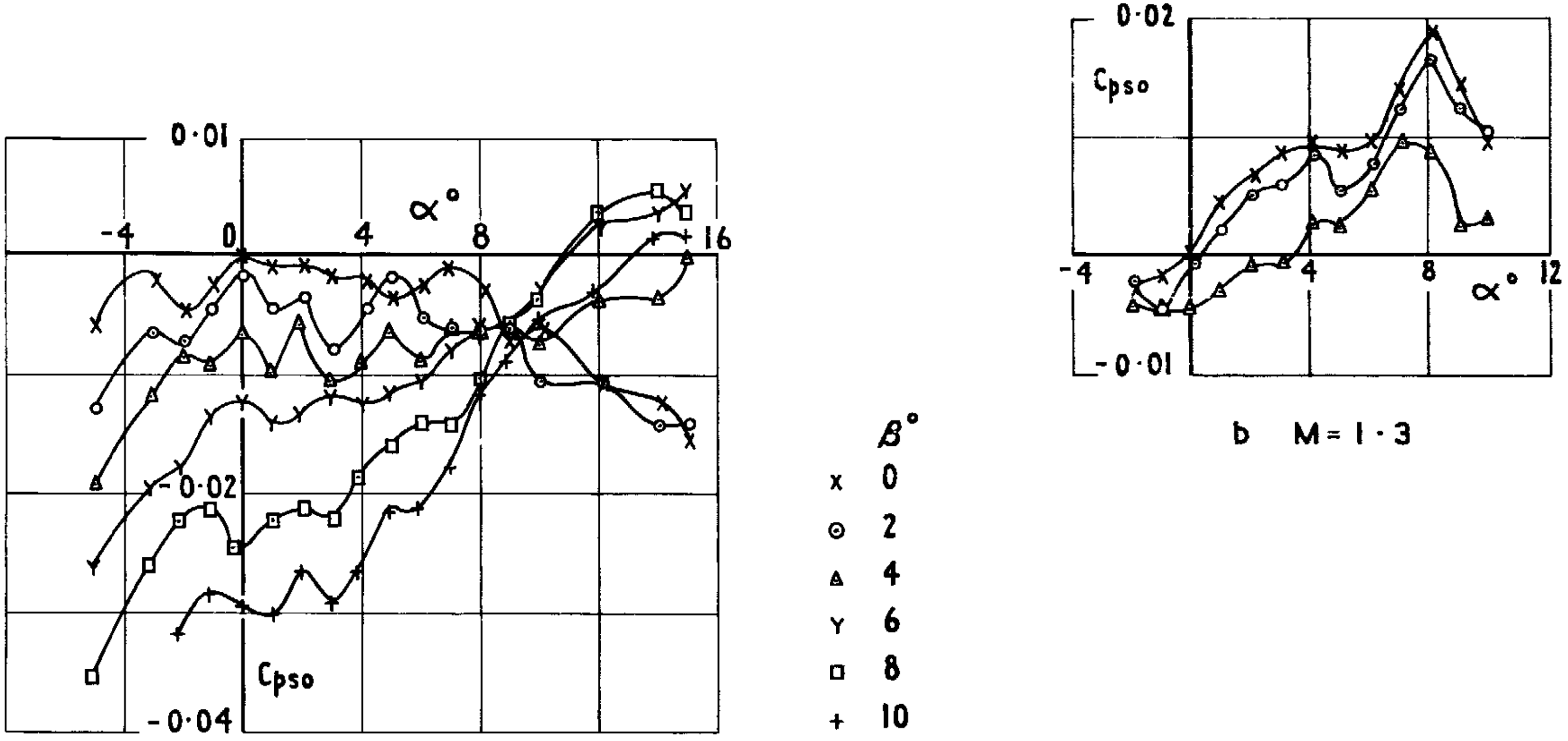


Fig. 8 Variation of head static pressure with incidence and sideslip in the 13 x 9 tunnel



a M = 0·26

b M = 1·3

Fig.9 Variation of head static pressure with incidence and sideslip in the A. R.A. tunnel

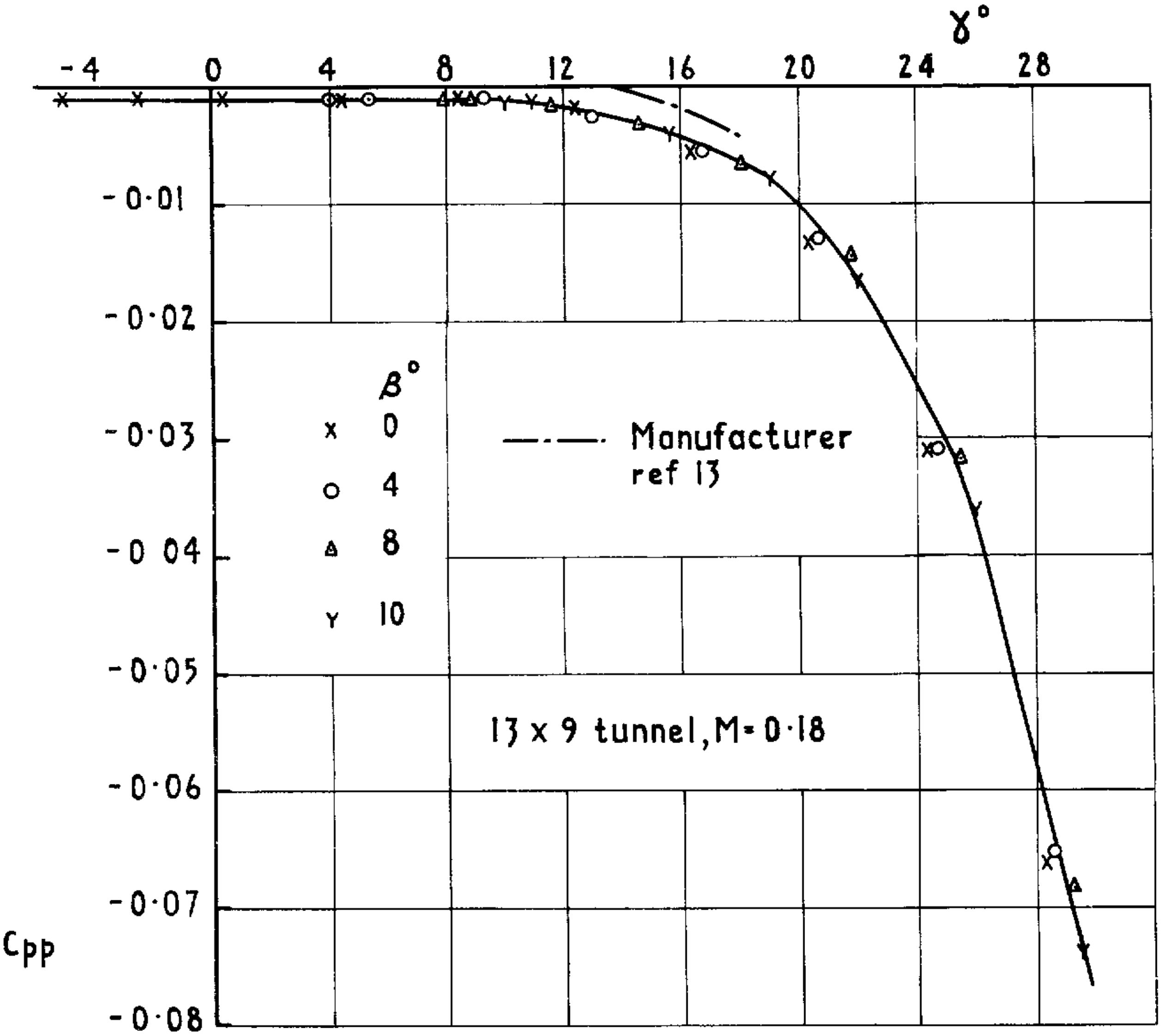


Fig. 10 Variation of head pitot pressure with flow direction angle in the 13 x 9 tunnel

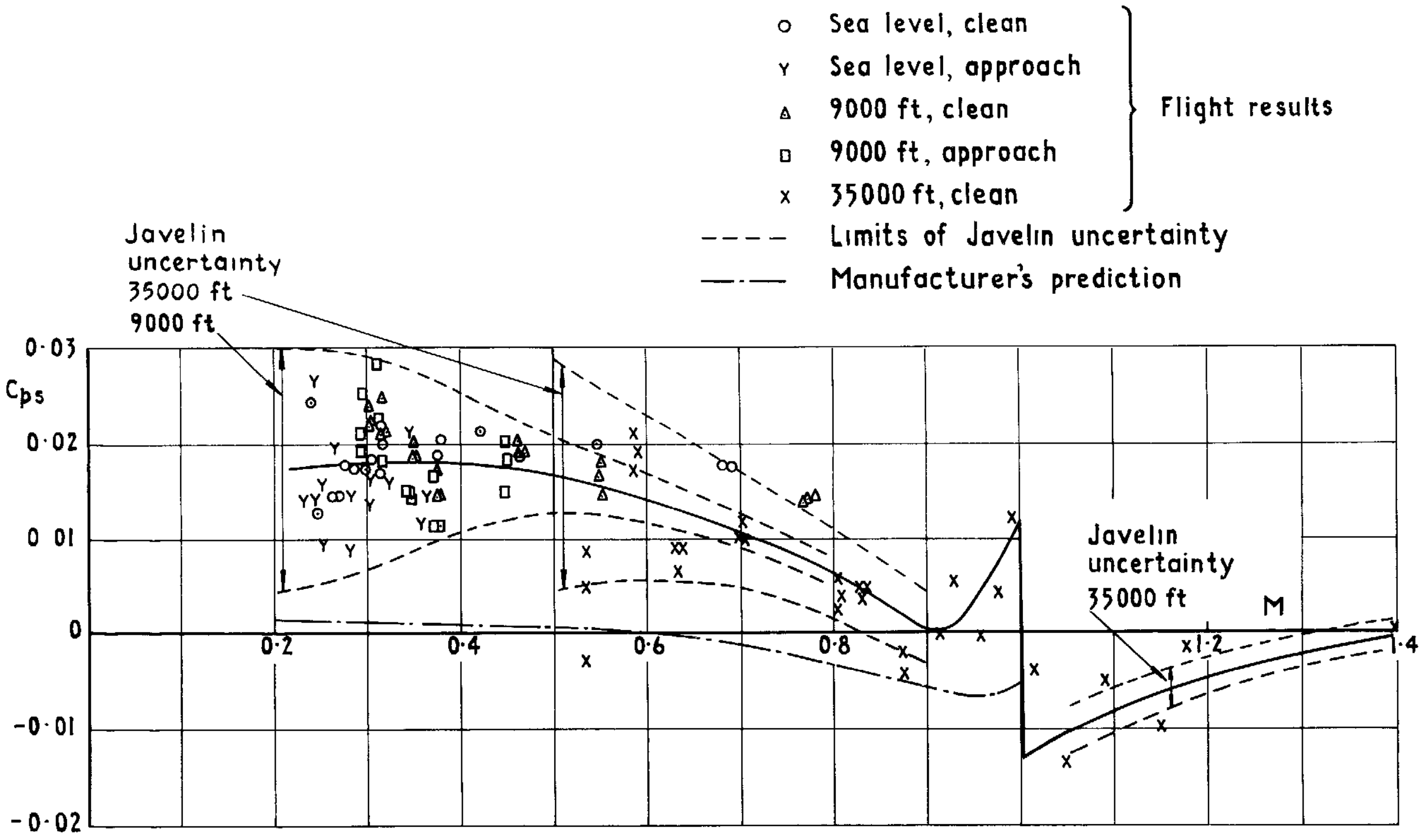


Fig 11 Aircraft pressure error versus Mach number at zero incidence

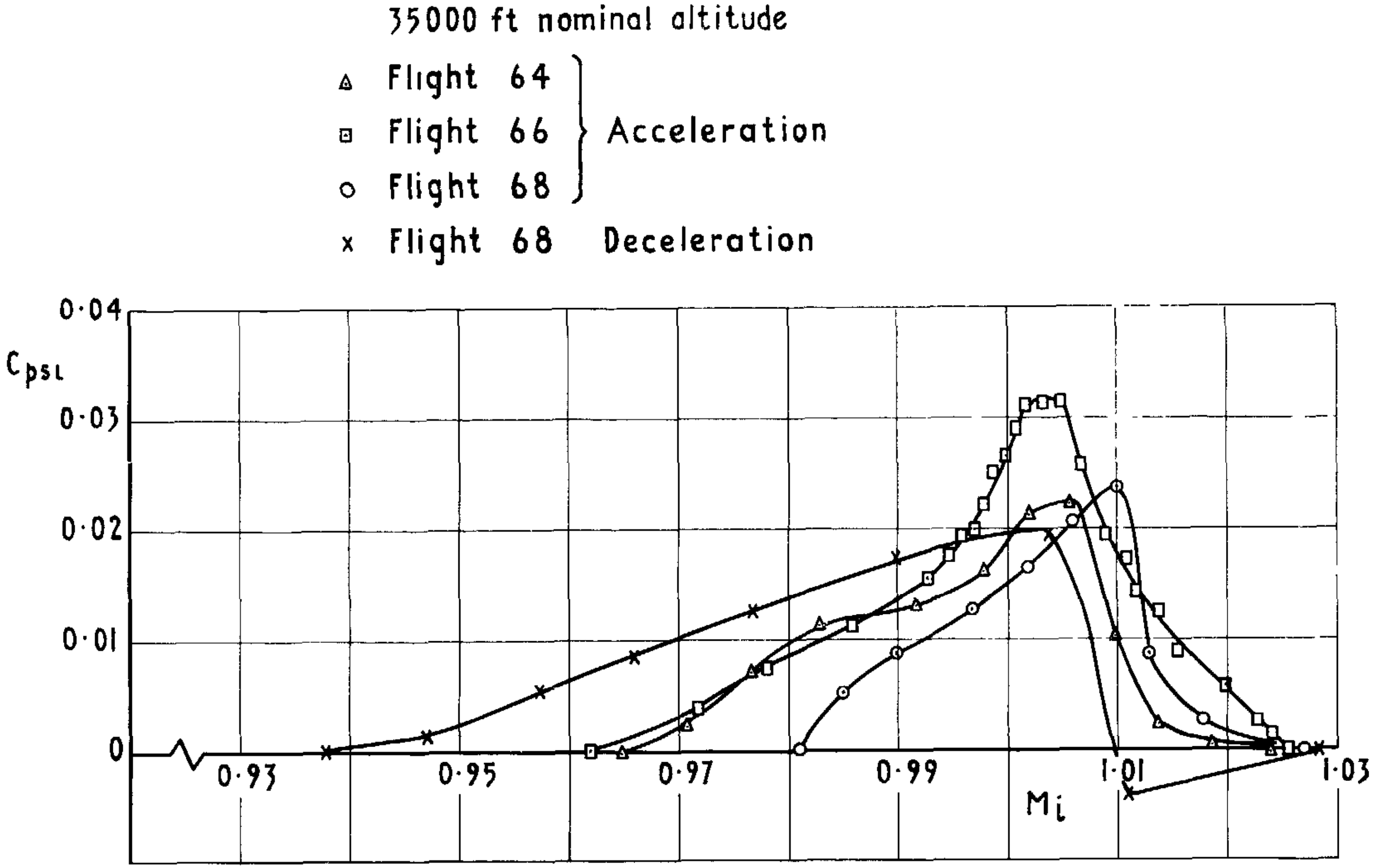


Fig.12 Pressure errors during transonic flights

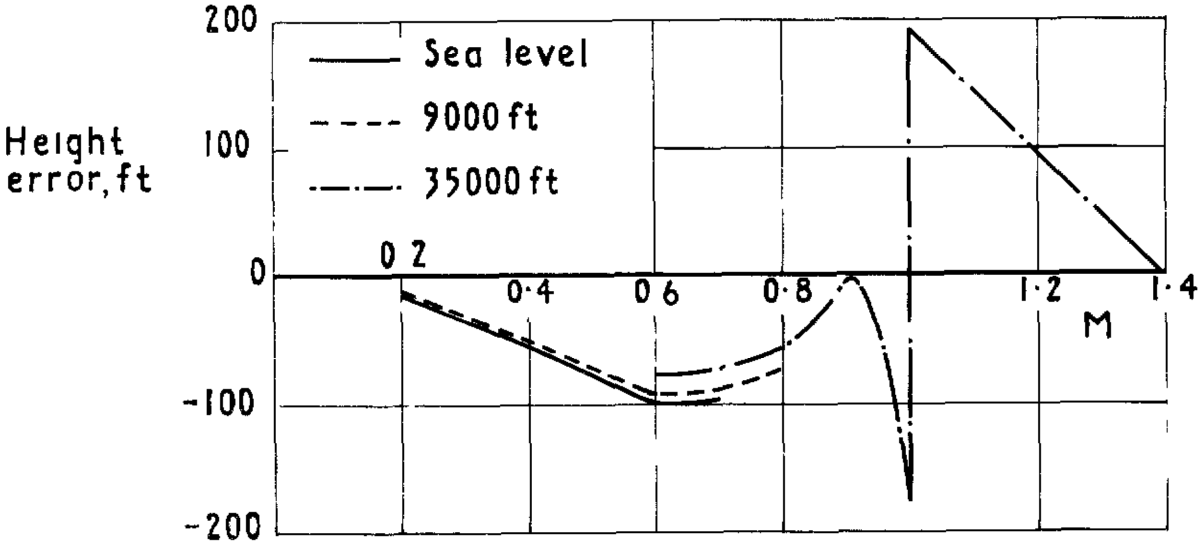


Fig 13 Height errors versus Mach number at zero incidence

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Nicholas, O P

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The manufacturer's prediction of the head performance agrees reasonably well with tunnel results but poorly with flight results It appears that the prediction of the aircraft pressure field is inaccurate

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