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Measurements of Skin-Friction Using Surface-Pitot Tubes in Free Flight at Supersonic Speeds

by

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MEASUREMENTS OF SKIN-FRICTION USING SURFACE-PITOT TUBES IN FREE FLIGHT AT SUPERSONIC SPEEDS

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SUMMARY

Surface-pitot tubes have been used to measure local skin-friction in free-flight. Measurements were made at 10 stations on a simple delta-winged model flying at zero lift at Mach numbers between 2.2 and 1.2.

This Note describes the design and development of the instrumentation and the methods employed to analyse the experimental data. Good agreement between the measured and estimated values of skin friction exists over most of the Mach number range.

Replaces R.A.E. Technical Note No. Aero.2891 - A.R.C. 24,715.

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1 INTRODUCTION

Much recent research work on the aerodynamic design of slender wings for supersonic transport aircraft has been devoted to the reduction of wave drag to as low a level as possible whilst retaining adequate volume and without sacrificing other aerodynamic characteristics. This has been so successful that the situation now exists where the skin-friction drag may be more than half the total drag at zero lift so that the accurate determination of skin-friction drag is of great importance. Hitherto no measurements have been made in free flight of the skin-friction drag on slender wings although such measurements would be of considerable value in extending the data obtained from wind-tunnel tests to the higher Reynolds number obtained in free flight.

A further need for skin-friction measurements arises from the desire to obtain experimental verification of the various theoretical methods used to achieve minimum wave drag. The usual experimental method for deriving the wave drag in free flight has been to measure the total drag of the model and subtract the estimated values of the non-wave-drag components from the total drag. The skin-friction component is always difficult to estimate with precision since it depends on such parameters as the local heat-transfer conditions, pressure gradients and laminar-to-turbulent transition points which may be ill-defined. Furthermore the correction of total drag measurements to full-scale Reynolds numbers is invariably necessary, which in turn introduces further uncertainties.

In a recent experiment[1] on the drag of a slender wing, the accuracy to which the wave drag could be evaluated was greatly dependent on the accuracy of the skin-friction estimate. Such experience has clearly demonstrated the need for a means of measuring skin friction in order to remove the uncertainty which exists at the present time.

An experiment was designed to develop a satisfactory method for measuring local skin friction in free flight and two models each carrying ten measuring stations have been flown. This Note summarises the reasons for choice of method and gives details of the design and development of the instrumentation, the reduction of data and a discussion of the results obtained. The results show that the surface-pitot tube is a satisfactory device for measuring the local skin friction on free-flight models.

2 CHOICE OF METHOD

A number of methods are available for skin-friction measurements in wind tunnels but not all of them lend themselves to free-flight work. For example, direct force measurements on a moving surface element and pitot traverses to obtain the velocity profile of the boundary layer are most difficult to use in flight, the first because of the need to balance out inertia forces which are very large compared with the frictional forces and the second because of the mechanical complexity involved. Two methods that can be used are the measurement of skin friction from the heat-transfer conditions using Reynolds analogy, and pressure measurements in the boundary layer by means of surface-pitot or Preston tubes.

At first sight the heat-transfer method seems preferable since the measuring technique is fully developed and the instrumentation does not interfere with the flow in any way, but the assumption that Reynolds analogy holds may not be justified for all kinds of flow; obviously a method which allows an independent check is desirable. Hence it was decided to use one of the boundary-layer pressure measuring techniques and surface-pitot tubes were preferred to Preston tubes because of their robustness and ease of manufacture.

The surface-pitot tube, sometimes called a Stanton tube or half-pitot tube, was first used for investigating conditions in a boundary layer by Stanton[3] in 1920, but its use as a skin-friction measuring device was not fully appreciated until work by G.I. Taylor[4] and Ludwieg and Tillmann[5] established that there exists a region in a boundary layer near to the wall where the shearing stress, τ, is directly related to the pressure measured by a surface-pitot tube. Preston[6] suggested a simple non-dimensional relationship of the form:

$$\frac{\tau\, h^2\, \rho_W}{\mu_W^2} = F\left(\frac{\rho_W\, h^2\, \Delta p}{\mu_W^2}\right) \qquad (1)$$

where h is the height of the surface pitot tube,

Δp is the pressure difference between surface pitot pressure and local static pressure,

ρ_W is the density of the fluid at the wall,

μ_W is the viscosity of the fluid at the wall,

and F represents a function which has to be determined by calibrating a surface-pitot tube in a flow where skin-friction can be determined by other means.

Such a calibration has recently been done in a supersonic wind tunnel at R.A.E. Bedford using pitot traverses to measure the momentum thickness from which the local skin friction was evaluated. The function was determined over a Mach number range from 1.8-2.7 and a Reynolds number range from 6-25 million based on the length of the boundary-layer run. This work is reported in Ref.7, which also gives a more comprehensive history of the method and of other calibrations. Further details of the calibration are given in Section 6.

3 PRELIMINARY EXPERIMENTS

Some preliminary experiments[2] were made to investigate the practicability of the technique in free flight and difficulties were revealed which have since been overcome. Details of these are given in the following sections.

3.1 Response lag

One surface-pitot tube was flown on a free-flight model and it was found that the variation in recorded pitot pressure did not follow the expected trends because of the lag in response of the system to changing pressures. The major effect was traced to the constricting effect of the very small blade height.

Laboratory tests were made to investigate ways of reducing lags to acceptable levels and to measure the response of the system to a known pressure change so that corrections could be applied to the flight measurements if necessary. As expected, the blade height was the most serious cause of lag but since this was already being set at the maximum permissible value (see Section 6.3) no reduction in lag could be obtained from changing it. However, reducing the volume of air that has to pass through the tube also reduced the lag and a great improvement in response was obtained by reducing the volume of air in the transducer chamber. All the transducers subsequently used with the pitot tubes were thus modified and care was taken when installing them in the model to keep the length of pipe between transducer and tube as short as possible; the average length was about one and a half inches.

After installation of the surface-pitot tubes and associated instrumentation in the present model, the response to an applied step-change in pressure was measured for each station and the time constant from the exponential response curve evaluated. The presence of dirt and dust in the aperture of the tube was found to cause additional lag, but when cleared, consistent response times were obtained. The tubes were thereafter kept covered and sealed from dust until a few minutes before the model was fired. The measured time constants were about 0.05 seconds. This is the time taken for the instrument to reach 63% of the steady reading following a step change in applied pressure. The change in Mach number of the model during such a time interval is about 0.005 which can certainly be considered as small. The implications of this amount of lag are discussed in Section 6.2.

No measurable lag was found in the response of a static pressure transducer.

3.2 Measurements of static pressure

The second difficulty revealed by a preliminary flight test[2] was the measurement of static pressure. The static tappings were located 0.1" upstream of the pitot hole. This was equivalent to 30 blade heights and, on the basis of wind-tunnel results, should have been sufficiently far forward to avoid interference effects from the pitot entry while still being close enough for the measured static pressure to be assumed equal to that at the pitot tube. In any event the distance appears to have been too small as all the static pressure measurements were higher than theory predicted. The solution to this problem is discussed in Section 4.2 where the location of the static holes on the present model is given.

4 DESCRIPTION OF THE MODEL

4.1 Geometry

The model chosen for these experiments was one of a series being flown to investigate the aerodynamic characteristics of narrow delta wings in free flight. The configuration is shown in Fig.1 and details are given in Table 2.

It was basically a narrow delta planform of diamond cross-section modified at the aft end to give streamwise tips. The maximum thickness chord ratio was 0.065. The surface-pitot tubes were located on the unmodified part of the wing so for the purposes of this work it can be considered as a true delta planform of diamond cross section. The leading edges were sharp and no roughness bands for fixing boundary-layer transition were applied.

4.2 Arrangement of measuring stations

The choice of positions for the tubes was influenced by the space available in the model since the transducers had to be positioned as close as possible to the tube. The arrangement finally chosen is shown in Fig.1 and the co-ordinates are given in Table 1. All were on the underside of the wing to avoid disturbances produced by imperfectly fitting hatches and interference effects from the fin. Since the model was symmetrical and designed to fly at zero lift the choice of the underside is of no aerodynamic significance. The photographs of the model and the measuring stations in Fig.2 show the arrangement clearly. It was hoped that this array would show up any chordwise or spanwise variations in skin friction if present. The repeat stations on the starboard wing (stations 1 & 6) were intended to reveal, by comparison, any effects of yaw or asymmetries and provide a measure of the overall consistency of measurement.

The static-pressure tappings were located to avoid the interference effects from the pitot tubes which had been met in the preliminary experiments, except for stations 1 & 6 where the holes were again 1/10" upstream to check on the interference supposition. (Fig.1 and Table 1.) The remaining static holes were placed on the same spanwise lines as the pitot tubes spaced roughly at the mid-points between the tubes for both row 1; stations 2,3 and 4 and row 2, stations 7,8,9 and 10. It was hoped by doing this to establish the spanwise static pressure distribution along the rows so that the static pressure at the tubes could be deduced by interpolation. Station 5, which is not a member of a row, was treated differently; its static hole was located $\frac{1}{2}$" upstream of its pitot tube, the distance being chosen rather arbitrarily to give an interference-free location. In practice the $\frac{1}{2}$" gap was sufficient to avoid interference but may of course be rather larger than is necessary.

4.3 Construction of the surface-pitot tubes

Fig.3 shows the construction of a surface-pitot tube. The tube is formed from a pressure tapping, identical to those used for static pressure measurement, with a wedge-shaped blade over it, the edge of the wedge being vertically over the leading edge of the hole. The wedges were made from small portions of an ordinary safety-razor blade, 0.004" in thickness. These were attached to the skin of the model with a cold-setting epoxy resin adhesive used in sufficient quantity to bring the edge of the blade up to the desired height.

4.4 Telemetry

The static pressure tappings were each connected to separate $\pm 2\frac{1}{2}$ p.s.i. transducers and the surface-pitot tubes to 0-15 p.s.i. transducers, the reference side of the instruments being connected to a common pressure chamber which was sealed immediately prior to the flight at ground-level ambient pressure. Three linear accelerometers were carried monitoring lateral, normal and longitudinal accelerations at the centre of gravity. These were used to obtain a measure of the angles of incidence and yaw of the model and for checking the velocity history. The readings of all 23 instruments - 10 pitot-tube pressures, 10 static pressures and 3 accelerations - were telemetered during flight by means of an R.A.E. 465 Mc/s multi-channel telemetry set.

The model also carried a Doppler transponder used to obtain velocity and trajectory data.

5 FLIGHT DETAILS

The model was mounted in pick-a-back fashion (Fig.2b) on a solid propellant rocket motor which boosted it to a maximum Mach number of just under 2.3 in a little over 3 seconds. The model then separated from the boost motor and continued in coasting flight while the measurements were made. Trajectory and velocity measurements were obtained from kine-theodolites and radio-Doppler data. The velocity, Mach number and dynamic-pressure time histories are shown in Fig.4 together with a plot of the trajectory. The Reynolds number per foot is plotted against Mach number in Fig.5 and based on the centre-line chord c_o = 6.667 ft covers a range from 50-100 million.

The distance between the pitot-tubes and the leading edge in the stream direction are given in Table 1 and hence the local Reynolds numbers can be evaluated; most of them lie in the range 10-30 millions.

Wing flutter occurred during the flight between M = 2.3 and 1.8. It was possible to determine the modes of motion from the accelerometer records since this phenomenon has been investigated on similar models. The principal mode present was a high frequency (400 c/s) movement of the wing tips and although it affected the instruments to some extent, satisfactory pressure readings could still be obtained. When the flutter died out at M = 1.8 no unusual changes in pressure occurred; it is therefore reasonable to assume that the tip movements did not appreciably affect the flow over the surface as far forward as the pitot tubes and static pressure holes.

6 ANALYSIS OF DATA

The calibration equation which was derived from the wind-tunnel measurements reported in Ref.7 is

$$\log_{10} \frac{\rho_W h^2 \tau}{\mu_W^2} = 0.764 \log_{10} \frac{\rho_W h^2 \Delta p}{\mu_W^2} - 0.684$$

the terms being defined in Section 2.

The constants in this equation were determined by fitting a best straight line using a least-squares method to the data acquired from a large number of measurements in the tunnel covering a Mach number range from 1.8-2.7 at Reynolds numbers from 6-25 million and blade heights of 0.0014" to 0.010". This calibration clearly is satisfactory at the higher Mach numbers for the Rynolds numbers and blade heights; (0.0035") on the present model, (apart from those farthest from the leading edge e.g. station 7) but is not altogether satisfactory at the lower Mach numbers, 1.2-1.6.

In the following paragraphs the derivation of the various terms in the equation is outlined.

6.1 Static pressures

The static pressures measured at stations 1 and 6 (holes 1/10" upstream of the pitot) and at station 5 (hole $\frac{1}{2}$" upstream of the pitot) were assumed to be occurring at the surface-pitot tube. Apart from possible interference between the pitot tube and the static hole this assumption is fully justified on a model of this scale.

The remaining stations were arranged in two spanwise rows and the results were plotted out as a spanwise pressure distribution; the values appropriate to the pitot-tube position were obtained by interpolation. This is illustrated in Fig.7 for both rows for a number of Mach numbers. The distributions for row 1 appear satisfactory but those for row 2 were not reliable since the values at station 10 are consistently very low and have had to be disregarded. The distributions drawn in for row 2, ignored the results from station 10 and were therefore based to some extent on the evidence of row 1 and theory. The interpolated static pressures for each station are shown in Fig.8 and the results are discussed more fully in Section 7. The poor results from static hole 10 were probably due to a faulty transducer.

6.2 Pressure difference Δp

This is the difference between the pressure obtained from the surface-pitot tubes and the interpolated or measured static pressure derived as above. An assessment was made of the correction to the pitot pressure for lag in the pipe and transducer, using the previously determined response time. This was done, for the instrument which had shown the greatest lag, by assuming that at the lowest Mach number, where the pressure is changing most slowly, the measured pressure was equal to the true pressure. Then, by a step-by-step process, the true pressure was determined at each instant of time up to the time of the highest Mach number. The difference between true and measured pressure never exceeded 2%; for the majority of the tubes it would probably be less than 1% and therefore no connections have been applied.

6.3 Blade height h

The blade heights which could be used were determined by the conditions appropriate to the calibration equation. This was for blade heights between 1 and 6 times the laminar sub-layer thickness or about 1/10 of the boundary-layer thickness. In the present experiment it was necessary to work to the upper limit to obtain a large enough pressure difference to be measured with accuracy, to keep lags to a minimum and to ease the manufacturing and measuring problems. There is great difficulty in mounting a blade accurately to the required height and the actual heights vary slightly from blade to blade. The specified height was 0.0035" and all but one were within ±0.0003". The values are given in Table 1. Height measurements were made with a depth microscope and dial gauge, several readings by more than one operator being taken and a mean value computed. Measurements were also taken at several stations across the span of the blade to check on whether any unevenness or slope was present. A second measuring technique was also tried. This was by traversing the blade edges with a "Talysurf" surface roughness measuring machine; the diameter of the stylus tip of this device prevented a really accurate measurement being made, but the results were all within 0.0002" of the depth microscope results. The overall accuracy of the final height measurements, judged from repeatability is thought to be about 0.0002" or about 6%.

6.4 Viscosity, μ_W

The viscosity of the air at the wall is required, and since this varies with temperature, some knowledge of surface temperature is needed. The surface of the model was made of $\frac{1}{4}$" thick resin-impregnated glass-fibre which is an excellent heat insulator, this makes direct measurement of surface temperature extremely difficult since a temperature sensing device in the surface produces a discontinuity in thermal properties and consequently unrepresentative heating rates. Calculating the surface temperature from extrapolation of a measurement some distance under the surface is again difficult because of the very high temperature gradients that occur. These problems led to a check on the errors in the final answer that would result from errors in the value of surface temperature. A convenient way of expressing the surface temperature is by means of the expression $T_W - T_o/T_R - T_o$, where suffix W refers to wall conditions, suffix o to free-stream conditions and suffix R to recovery conditions. This expression has a value of zero for full heat transfer and unity for zero heat transfer. Since the wall material is a good thermal insulator the heat-transfer conditions are expected to be near zero ($T_W - T_o/T_R - T_o \approx 1$) for the majority of the flight and some measure of the effect of an error in temperature measurement may be obtained from the fact that an error in $T_W - T_o/T_R - T_o$ of 0.1 produced an error in the skin-friction results equivalent to an error of only 1% in the pitot and static pressure measurements. The expected surface temperatures were computed by a high-speed electronic computer (Mercury) using Eckerts "intermediate-enthalpy" method[8] for flat-plate conditions together with the actual trajectory data measured from the flight of the model and taking into account heat conduction and heat capacity properties of the wall material. It is difficult to assess the accuracy of this estimate but available experimental evidence on the validity of Eckerts intermediate-enthalpy method suggests that the error in the heat-transfer parameter should never exceed 0.1.

The viscosity of the air under wall conditions was then evaluated from Sutherland's formula using the estimated temperatures. Fig.6 shows the estimated surface temperature parameter plotted against flight time with Mach number marked on the curve. The value rises above 1 at Mach numbers below 1.5 because the surface temperature is higher than the boundary-layer recovery temperature.

6.5 Density, ρ_W

The density was evaluated from the local free-stream density corrected to wall conditions by means of the measured static pressures* and the surface temperatures estimated as in Section 6.4 above.

* Theoretical static pressures were used here when the measured values were considered unreliable as described in Section 7.

6.6 Skin-friction evaluation

All the necessary information for evaluating the shearing stress at the wall, τ, is therefore available; the skin friction is then given by

$$C_f = \frac{\tau}{\frac{1}{2}\rho_o V_o^2}$$

where ρ_o and V_o are the free-stream density and velocity.

The skin-friction coefficient, C_f was computed from the calibration equation for all ten stations for the Mach number range 1.2-2.2. These results are shown in Fig.9, together with three theoretical estimates for three different heat-transfer conditions. The estimates were derived by the Prandtl-Schlichting methods, assuming a turbulent boundary layer originating at the leading-edge ahead of the station in the free-stream direction. The most probable estimate of the friction is shown by a heavy line and corresponds to that for the heat transfer conditions calculated in Section 6.4 and shown in Fig.6.

The most probable value lies for the most part close to the zero heat transfer $(T_W - T_o/T_R - T_o = 1)$ for the reasons already described. The two lightly drawn curves for full and zero heat transfer are not intended to indicate an uncertainty band on the estimate but are given to provide an indication of the errors that can arise if the heat-transfer conditions are not known.

Points are also plotted on Fig.9 derived from the measured pitot pressure and a theoretical estimate of static pressure. This is discussed fully in Section 7.

The records of the normal and longitudinal accelerations were analysed and the angles of incidence in pitch and yaw during the flight deduced, using previously measured values of the lift-curve slope and the side-force derivative. The angle of yaw was at all times less than 1/10 degrees and is thus negligibly small. The presence of the fin and other asymmetries cause the model to trim at a small negative incidence in pitch, about -0.2° at M = 2.2 increasing to -0.4° at M = 1.2. This small incidence should not invalidate the flat plate flow assumptions and will cause increments in C_p of about -0.003.

7 DISCUSSION OF RESULTS

The static pressures interpolated from the two spanwise distributions and as measured from stations 1,5 and 6 are shown in Fig.8 plotted against Mach number together with estimates based on linearized thin-wing theory[9,10]. The results from stations 1 and 6 with their static holes 1/10" upstream of the pitot tube clearly give too high a pressure confirming the suspicions from the first model[2] that the pressure rise at the pitot tube was giving a high pressure region at the static hole. Station 5, whose static hole was ½" upstream, does not show this effect; its pressure readings agree very well with the theoretical values.

The pressures interpolated from the two spanwise distributions, for stations 2,3 and 4 and 7,8,9 and 10, shown in Fig.8, agree reasonably well with theory at the higher Mach numbers where the accuracy of measurement is higher. But at the lower Mach number they tend to diverge from the theoretical curves. The discrepancies here are thought to have come from experimental inaccuracies, rather than from any inadequacy of the theoretical method; because very good agreement between results from this theory and experimental results has been obtained on a similar wing in the Bedford 8 ft tunnel over the Mach number range M = 1.4 to 2.2.

The interpolation method for determining the experimental pressures at the tubes for row 1 appears satisfactory when good readings were obtained from all the instruments; but it is not so good for row 2 where an unreliable value for station 10 was obtained. In general, having the static hole some distance upstream of the pitot tube (e.g. station 5) seems to be the most satisfactory solution for a body of very small curvature; it has been shown to give reliable results and a faulty instrument affects only the one measuring station rather than several as in the spanwise distribution method.

The skin-friction coefficients evaluated from the measured pitot pressures and both measured and theoretical static pressures are shown for each station in Fig.9, together with the estimated values. The agreement between the estimated values and measured values is good from a Mach number of 2.2 down to 1.5 but for most stations the measured coefficients then drop below the "most probable" estimate curve.

The discrepancy between results computed from measured static pressures and those from theoretical static pressures is not large, except at the lower Mach numbers and for stations 1 and 6 which suffered from interference.

The measured values agree more closely with the estimate for zero heat-transfer than the "most probable" value, but these two estimates are only significantly different at the highest Mach number and even there the difference is only some 8%, which is within the overall accuracy of the experimental results (see Section 7.1).

Stations 6 and 9 at idential positions but on opposite wings both gave a slightly higher skin friction than expected; there is nothing that suggests they are faulty in any way and the agreement between the two clearly points to some real effect that is being measured.

The agreement betweem this pair, stations 6 and 9 and the similar pair, stations 1 and 3 show there is little or no asymmetry in the flow, as one might expect from the very small measured yaw angles ($< 0.1°$). The differences between them are about 5% at high Mach numbers increasing to 10 or more at M = 1.2. This can be taken as a likely measure of the repeatability of the method.

Low values of skin friction were obtained at M = 1.2-1.3 on the majority of measuring stations. The accuracy of measurement at this end of the speed range is expected to be poor but the behaviour of the various stations is too consistent to be explained by random errors.

The most likely cause of the low values is the calibration equation not being valid at M = 1.2-1.3 since it was established only over the range M = 1.8 - 2.7.

While the calibration equation was found to be independent of Mach number between M = 1.8 and 2.7, a different calibration has been obtained by Bradshaw and Gregory[11] for subsonic flow. It is unlikely that the conditions in the boundary layer change abruptly as the free stream becomes sonic, in which case one might expect the calibration equation to tend towards the subsonic value for low supersonic Mach numbers of the free stream. Using the subsonic calibration for the present results at M = 1.2 brings the measured skin friction much closer to the estimated value. Clearly the need arises for a calibration to be made to link up between the subsonic one of Bradshaw and Gregory[11] and the supersonic one of Smith, Gaudet and Winter[7].

A further possible, but less likely cause of the low readings at M = 1.2 is that the tubes are recording a transitional flow in which the boundary-layer is not fully turbulent. Laminar flow regions are likely close to the leading edge since the Reynolds number per foot is about 8 millions at these speeds, and transitional flow, between laminar and turbulent, has been observed and measured by surface-pitot tubes in Ref.7. However nearly all the measuring stations exhibited the low values of skin friction despite a wide variation in their local Reynolds numbers; which does not appear consistent with the transition assumption.

Apart from the measurements at low Mach numbers the remaining results from M = 2.2 to 1.4 are most reasonable. This can clearly be seen in Fig.10 where the measured and estimated spanwise variations of skin friction (for rows 1 and 2) are compared. The expected increase in C_F towards the leading edge is apparent for the experimental data and the general levels are in good agreement. The measured values are those evaluated with theoretical static pressures.

The results as a whole suggest that it is practicable to measure local skin-friction coefficients in free flight but that care in location of the static holes is required to avoid interference from the pitot tubes. Reliable results are obainable only within the limits of the calibration equation.

For Mach numbers near M = 1.2 a further calibration is required. If accurate measurements are required at low supersonic speeds a separate model covering only the required Mach number range would be needed so that appropriate ranges of instruments could be fitted.

7.1 Accuracy

The accuracy of the method depends largely on the accuracy to which the two pressures can be measured, on the accurate determination of blade height and on the applicability of the calibration equation to the conditions of the test. The pressure measurements will be most accurate at the high Mach numbers where most of the instrument range is used on the pitot measurements; the expected uncertainty should be 3% - 4% deteriorating to perhaps three times this figure at the lowest Mach number for the present test.

The uncertainty in blade height stated in Section 6.3 was ±0.0002" in about 0.0035" which is about 6%.

The surface pitot tube is sensitive to the angle of flow to the blade. The true reading is obtained when the flow is normal to the blade and the pressure drops as $\cos^2\beta$, where β is the angle the flow makes to the normal. Thus the error is only $2\frac{1}{2}$% for angles up to 10°. Expected flow angles for the present model at zero lift in the vicinity of the measuring stations are less than 5°.

The overall uncertainty of the various measurements is expected to be about ±10% but the accuracy of measured skin-friction coefficients is also dependent on the accuracy of the calibration equation.

The scatter of results from which the calibration was deduced in Ref.7 was less than 10% and therefore the equation accuracy - the best straight line - should be very much better, perhaps 2-3%. This accuracy cannot of course be guaranteed outside the limits of Mach number and Reynolds number, where roughly half the free-flight results lie.

8 CONCLUSIONS

(1) The surface-pitot tube method for measuring local skin-friction can be used in free flight. The permissible blade heights give adequate pressure differences and the lag in the blade-transducer system can be reduced to a level such that the deceleration of the model does not introduce additional errors.

(2) Ideally, a measure of static pressure at the place of the surface-pitot tube is needed but this is not possible. Two ways of overcoming this have been used; (a) establishing the spanwise static pressure distribution using a series of measurements and interpolating for the value at the tube; (b) measuring the static pressure a little way upstream of the tube. The second of these methods is preferred since only a single measurement is required for each station and a false reading by one instrument does not invalidate a whole row of measuring stations. The choice of distance upstream has been shown to be important if interference is to be avoided; for the present model 1/10" is inadequate and $\frac{1}{2}$" satisfactory. In the present experiment the difficulty of determining the static pressure adequately introduces the more serious error.

(3) The measured values of local skin friction coefficient on a slender delta wing agree with estimated flat-plate values to an accuracy of about ±10% at Mach numbers between 1.4 and 2.2. Below M = 1.4, the measured values are low; the most likely explanation being the inapplicability of the calibration equations. The present results are in good agreement with those from a preliminary model.

(4) If the method is to be used widely, an extension of the range of validity of the calibration equation is needed to cover a lager range of Mach and Reynolds numbers.

LIST OF SYMBOLS

c_o = centre-line chord

$\bar{c}$ = geometric mean chord

C_f = skin-friction coefficient $\left(= \frac{\tau_W}{\frac{1}{2}\rho_o V^2} \right)$

C_p = static pressure coefficient

F = a function in equation (1)

h = surface-pitot tube blade height

M = Mach number

Δp = differences between pitot and static pressures

q = dynamic pressure ($\frac{1}{2}\rho_o V^2$)

s = local wing semi-span

t = maximum thickness of wing

T = temperature

V = velocity

x = chordwise co-ordinate

y = spanwise co-ordinate

β = angle of local flow to the pitot tube

ρ = density

μ = viscosity

τ = shearing stress

Suffixes

o = free-stream conditions

R = recovery conditions

W = wall conditions

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TABLE 1

Surface-pitot tube positions, blade heights and static hole positions

Surface-pitot tube					Static hole	
Station No.	Position aft apex in.	Spanwise posn. +ve to starboard in.	Blade height in.	Distance from L.E. in stream direction ft	Aft apex in.	Spanwise position in.
1	35	-3.0	0.0036	2.083	34.9	-3.0
2	35	+0.5	0.0038	2.777	35	+1.25
3	35	+3.0	0.0036	2.083	35	+4.75
4	35	+5.5	0.0038	1.389	35	+6.5
5	41	+6.0	0.0033	1.75	40.5	+6.0
6	52	-6.25	0.0032	2.597	51.9	-6.25
7	52	+0.5	0.0037	4.194	52	+2.25
8	52	+4.0	0.0027	3.222	52	+5.125
9	52	+6.25	0.0035	2.597	52	+7.375
10	52	+8.50	0.0038	1.972	52	+9.5

TABLE 2

Model details

Wing planform area	12.813 sq. ft
Aspect ratio	0.867
Planform factor, P	0.57
Geometric mean chord, $\bar{c}$	3.843 ft
Maximum centre-line thickness/chord ratio	0.065
Gross fin area	1.296 sq. ft
Weight	210 lb
Centre of gravity position	0.502 c_o
Newby area distribution (zero camber and twist)	$S(x) = 4\ s(x)\ t\ x(1-x)$

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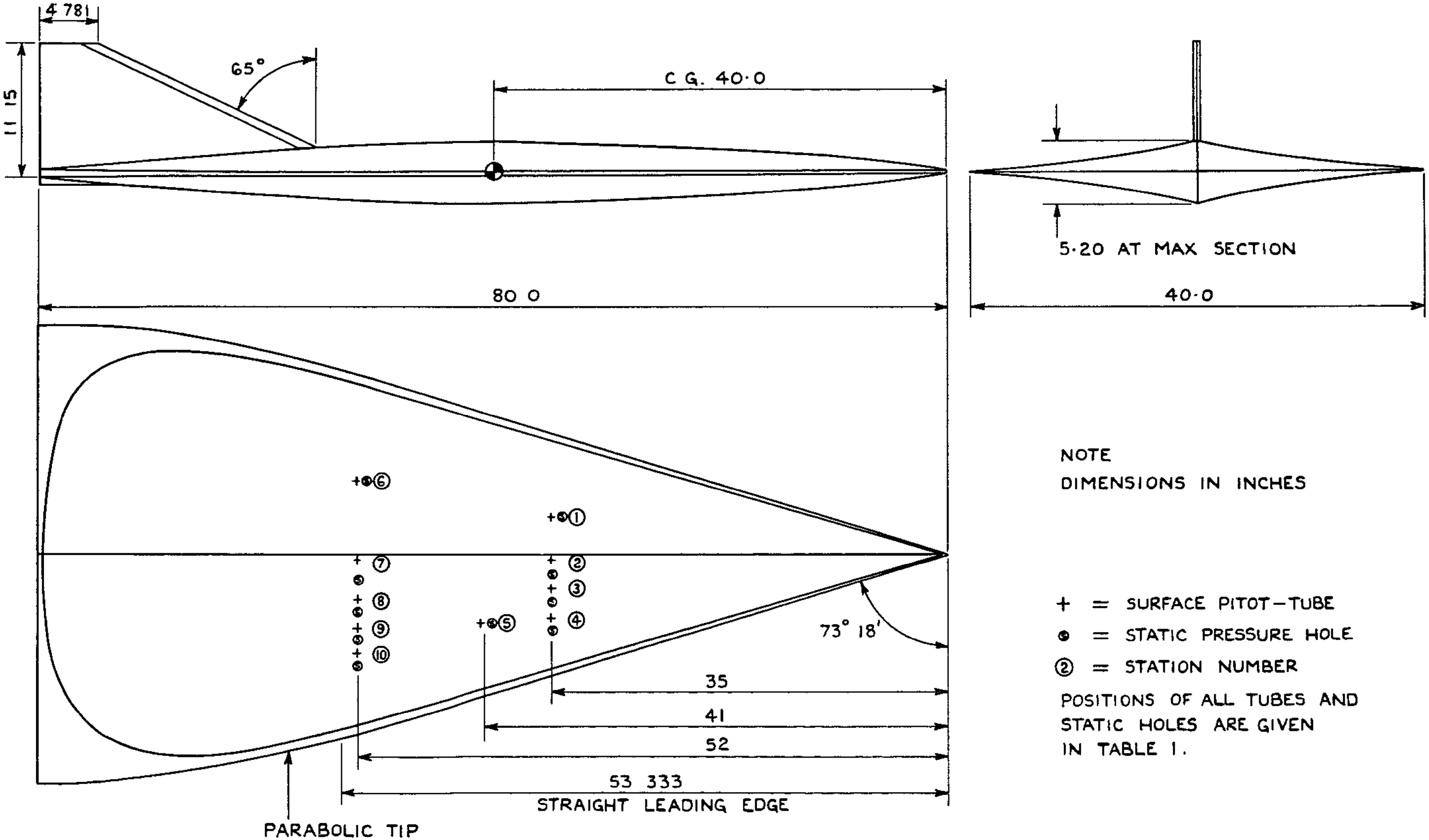


FIG. 1. GENERAL ARRANGEMENT OF THE MODEL.

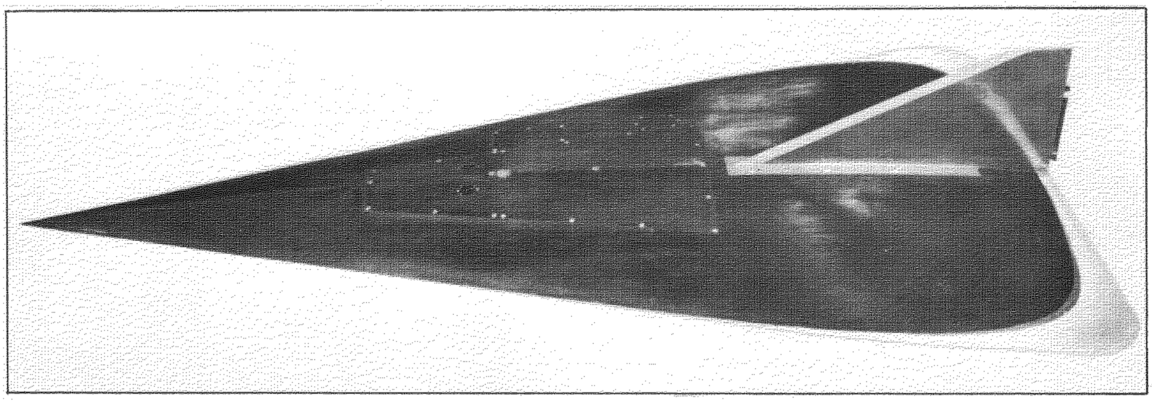


FIG.2a. MODEL UPPER SURFACE

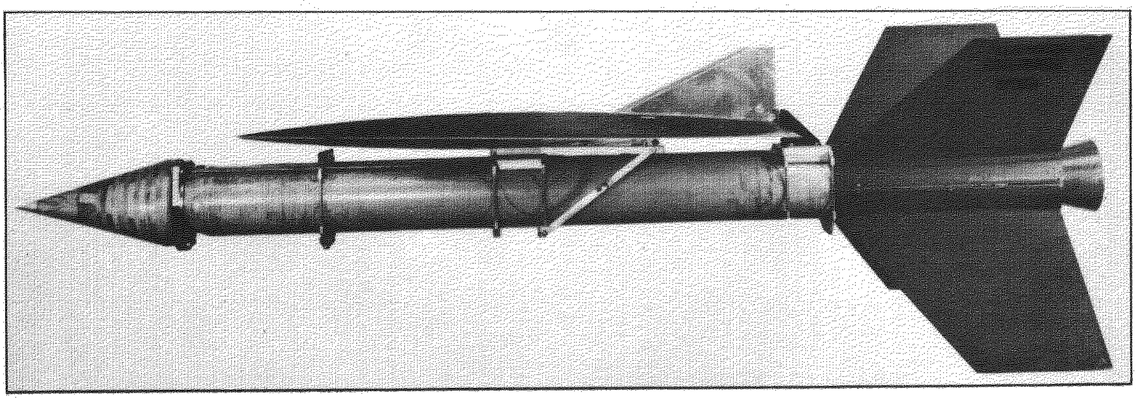


FIG.2b. MODEL PLUS BOOST ASSEMBLY

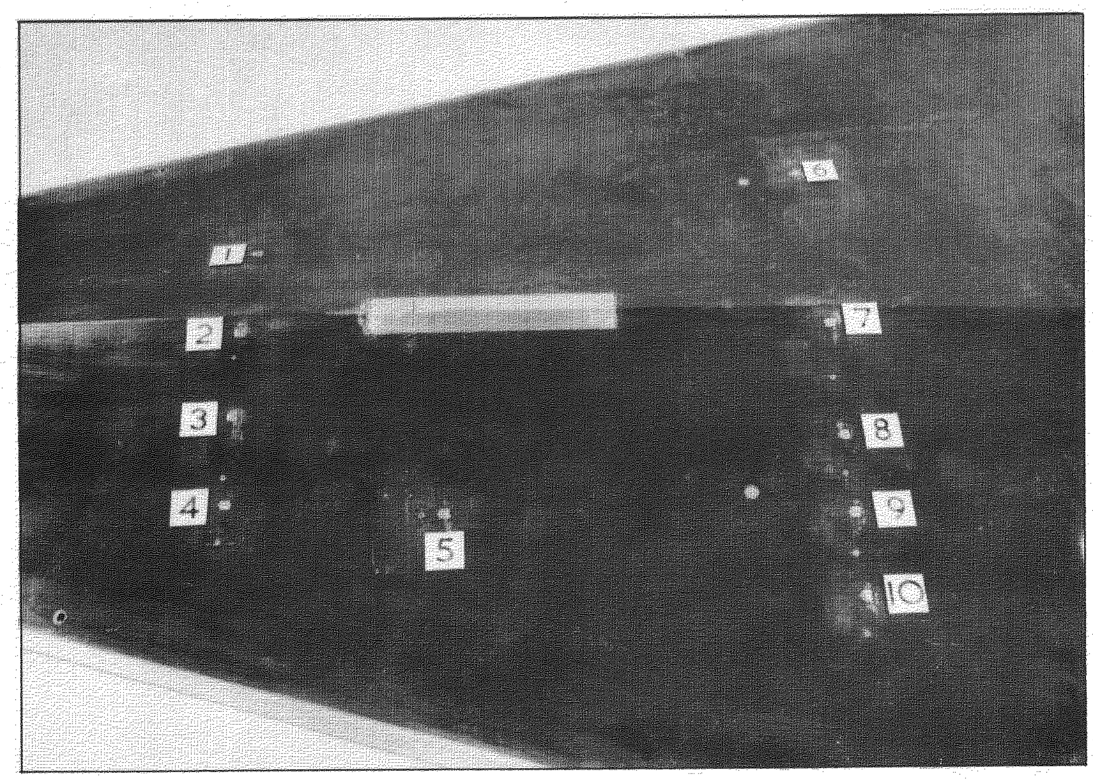


FIG.2.c. MODEL LOWER SURFACE SHOWING DETAILS OF MEASURING STATIONS

FIG.2a, b, & c

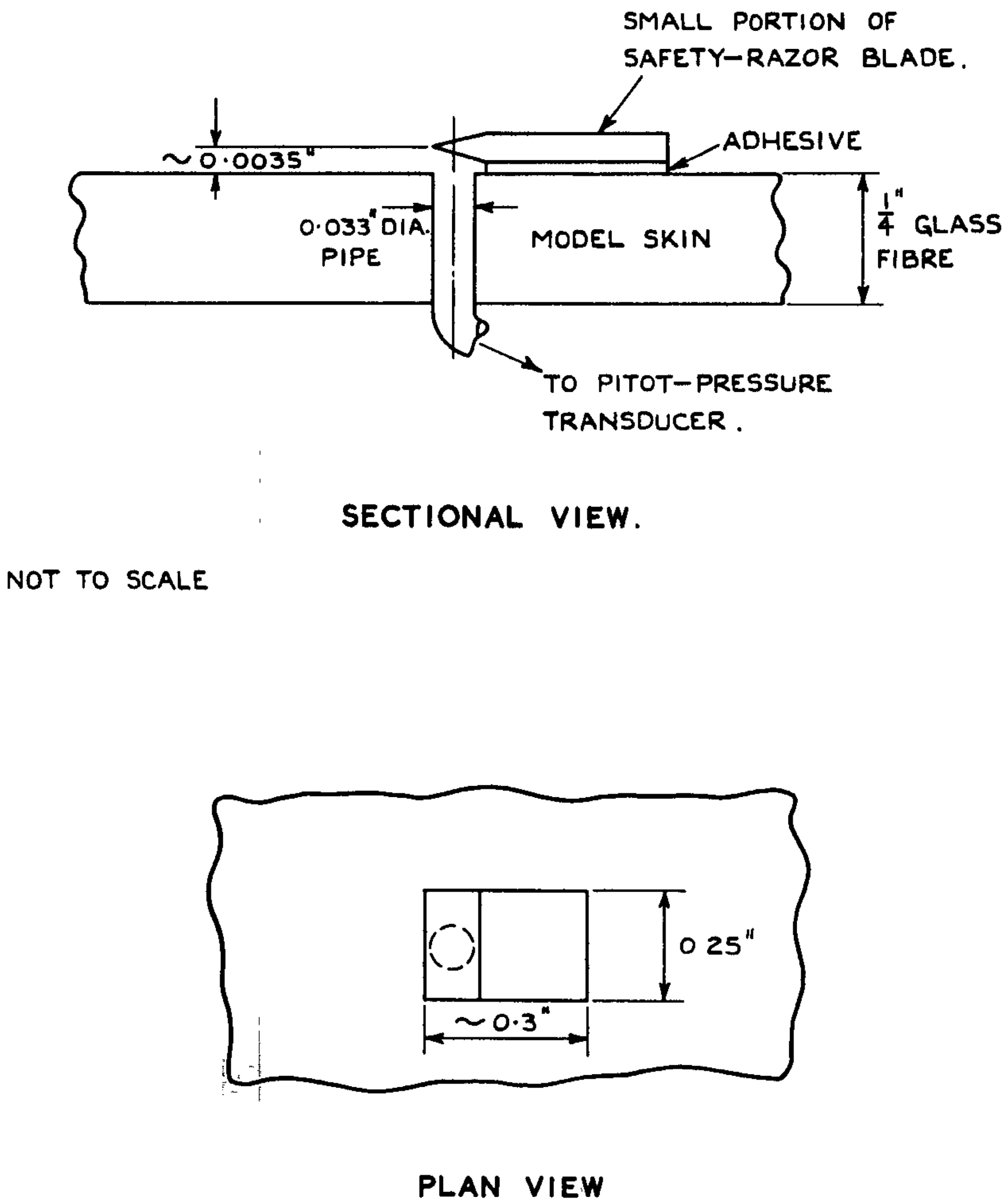
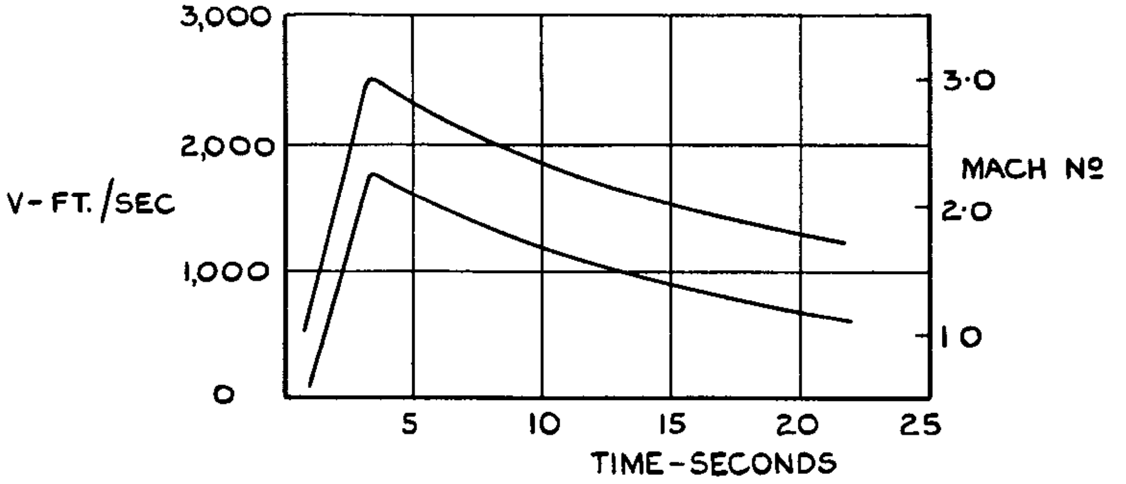
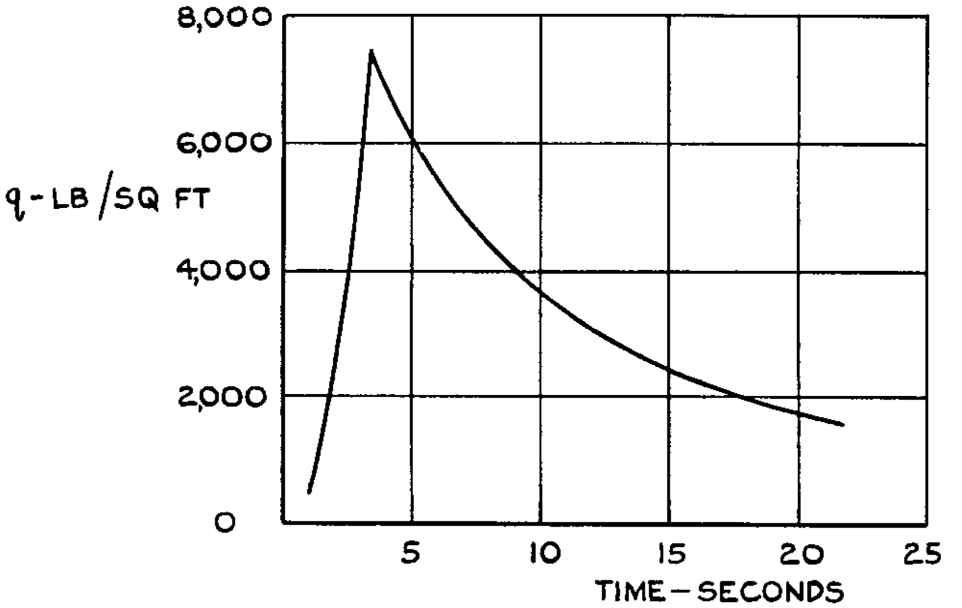


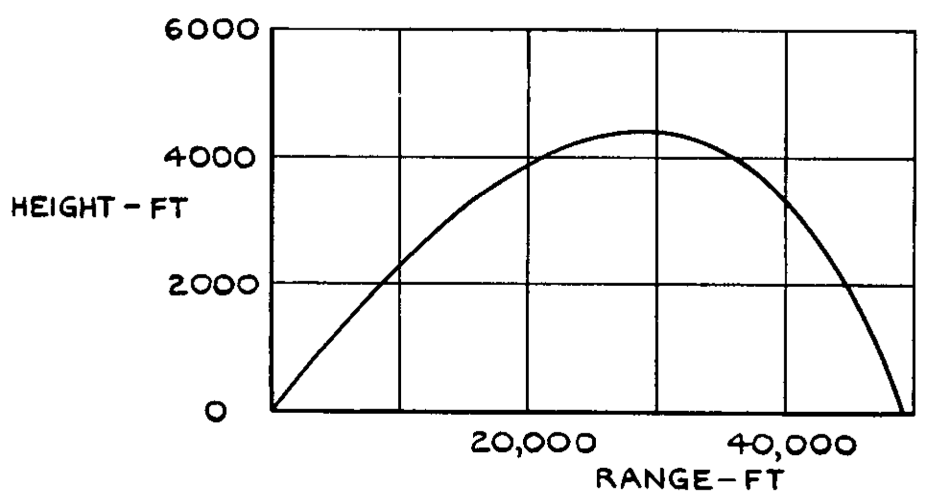
FIG. 3. ARRANGEMENT OF A SURFACE–PITOT TUBE.



(a) VELOCITY AND MACH Nº



(b) DYNAMIC PRESSURE



(c) TRAJECTORY

FIG.4 (a-c). TRAJECTORY AND VELOCITY DATA.

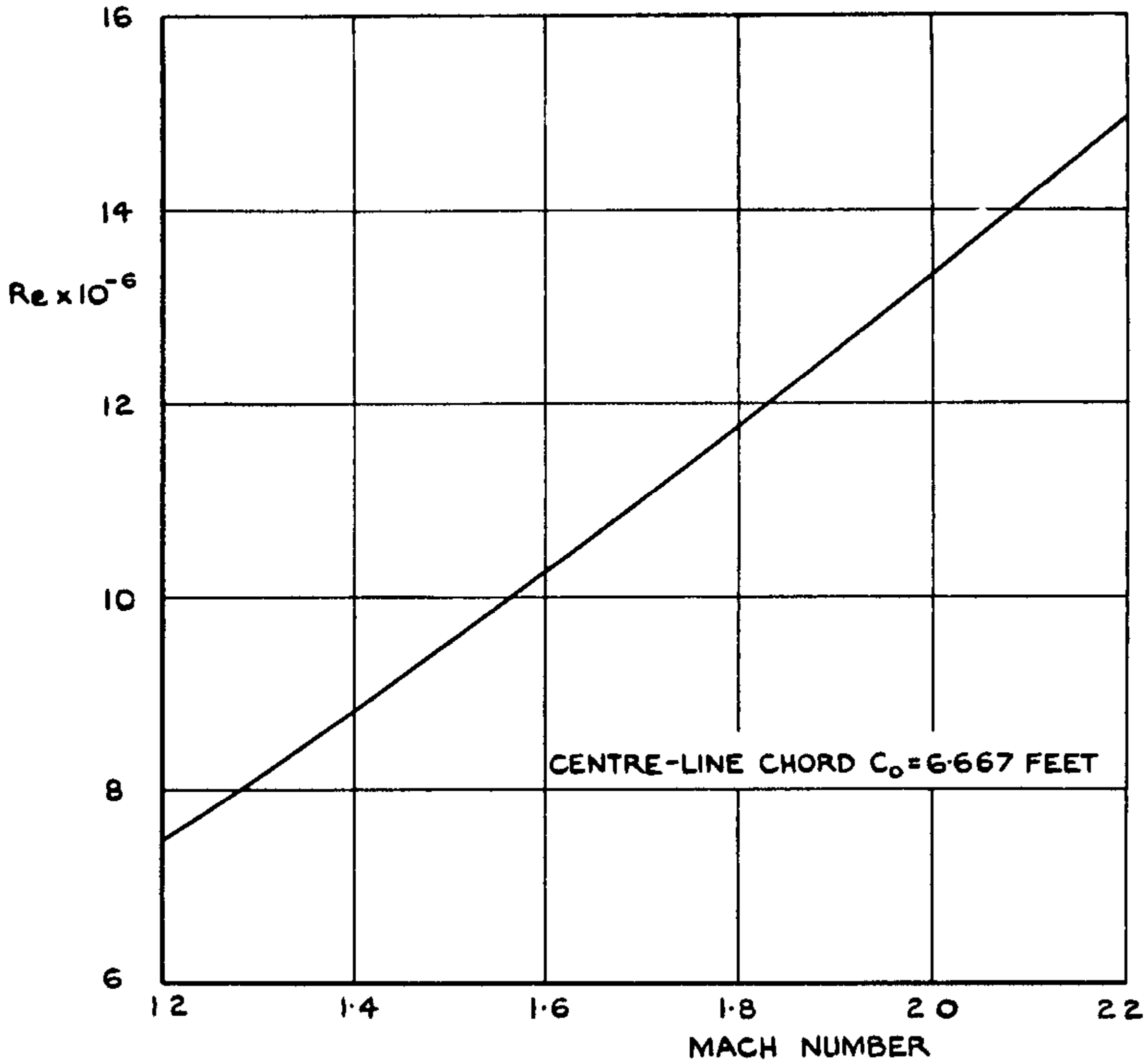


FIG. 5. REYNOLDS NUMBER PER FOOT.

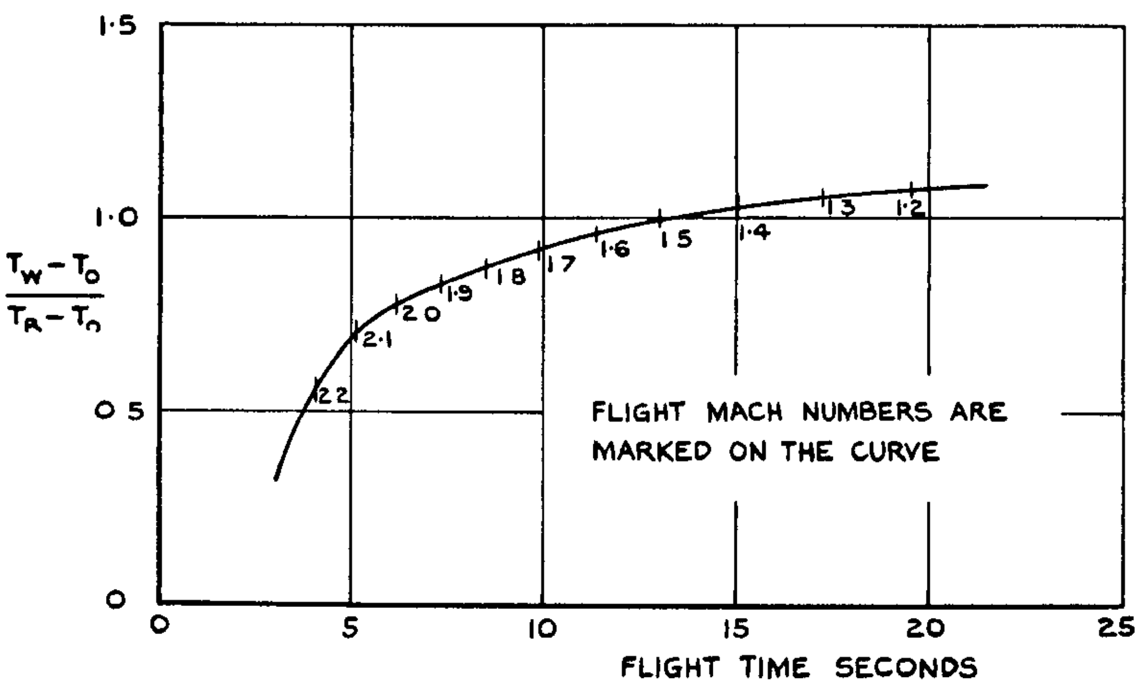
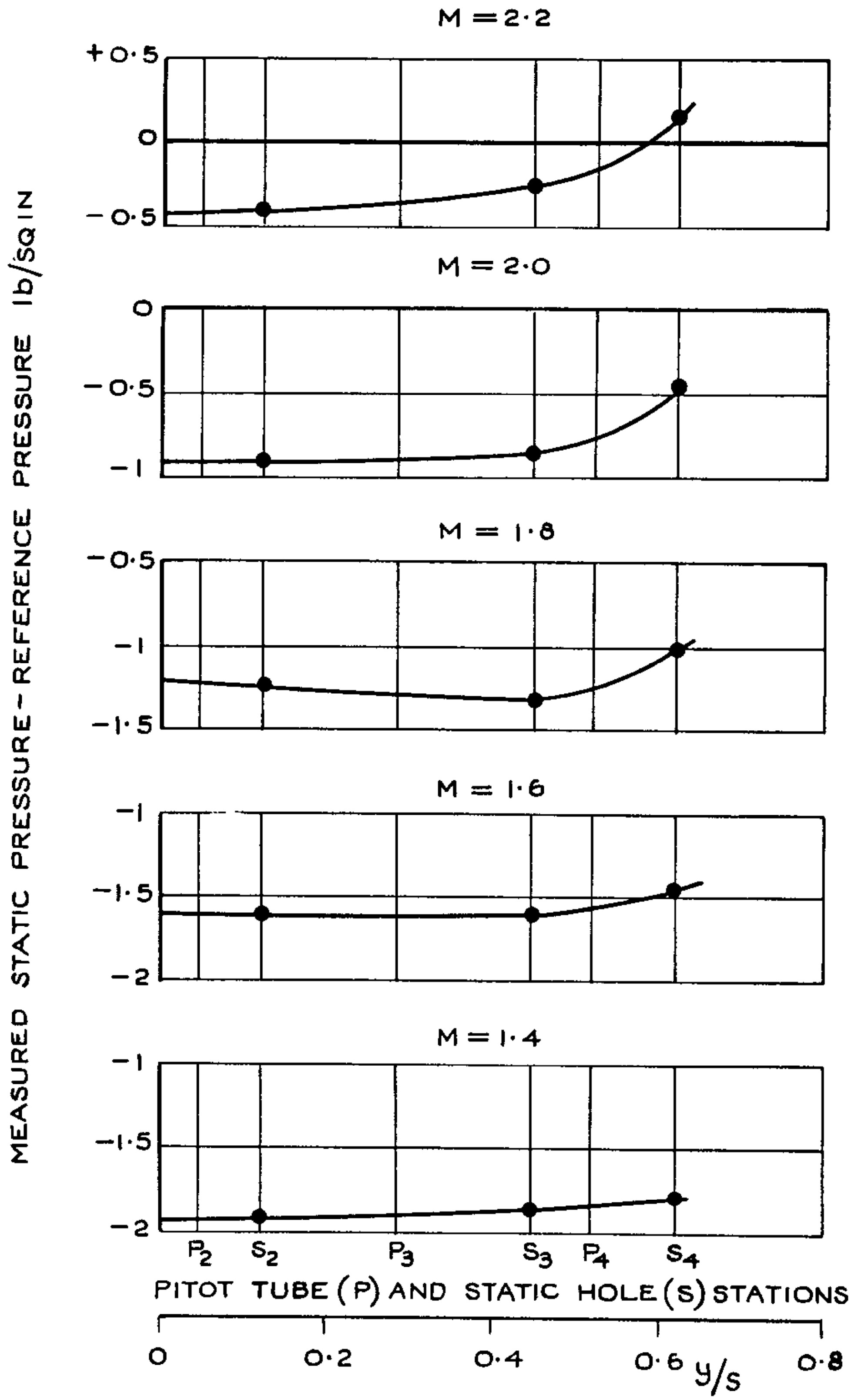
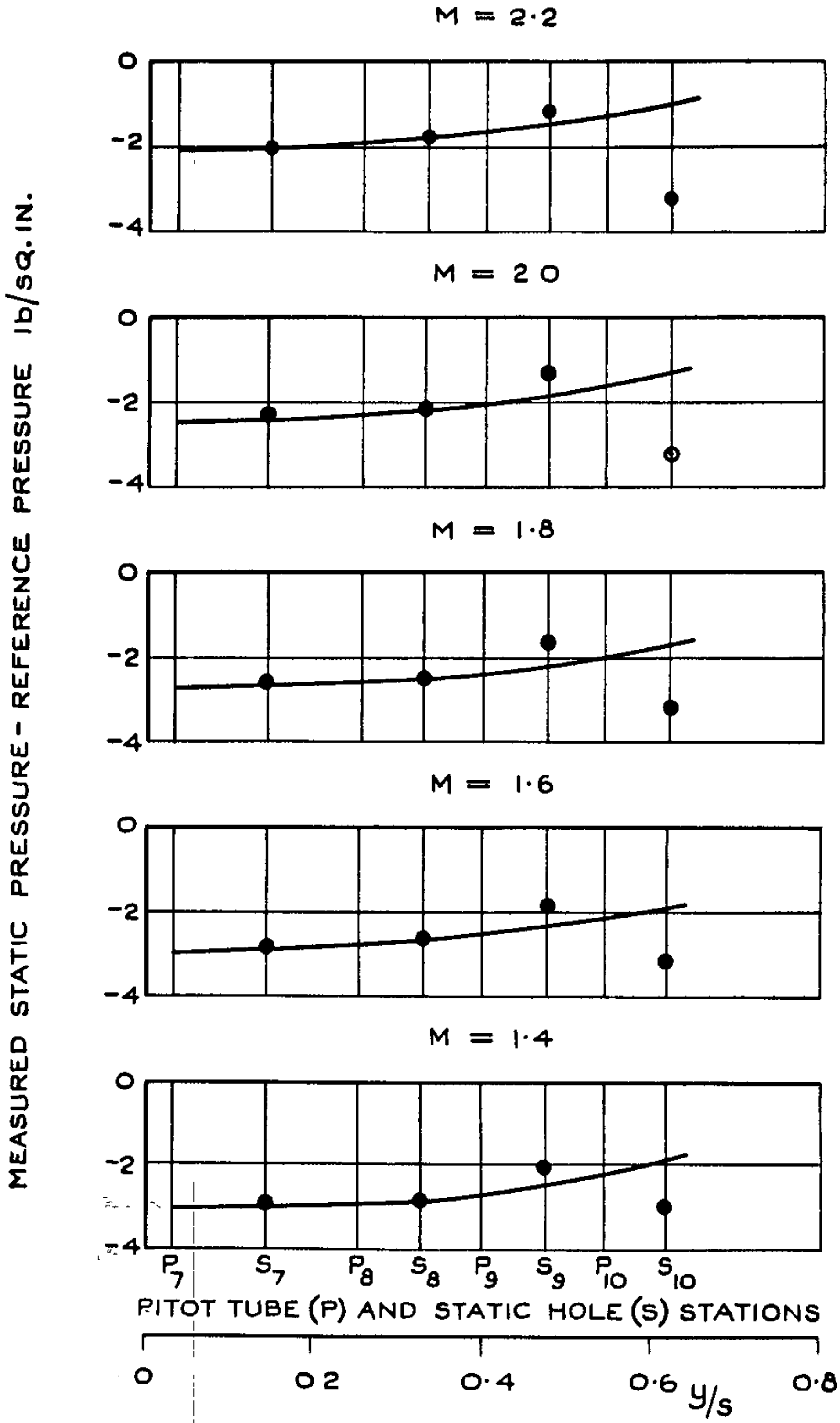


FIG. 6. ESTIMATED MODEL SURFACE-TEMPERATURE PARAMETER, $\frac{T_W - T_O}{T_R - T_O}$



(a) ROW 1 $x/c_0 = 0{\cdot}4375$

FIG. 7. SPANWISE STATIC PRESSURE DISTRIBUTION.



(b) ROW 2 $x/c_0 = 0{\cdot}65$

FIG. 7. (CONTD.) SPANWISE STATIC PRESSURE DISTRIBUTION.

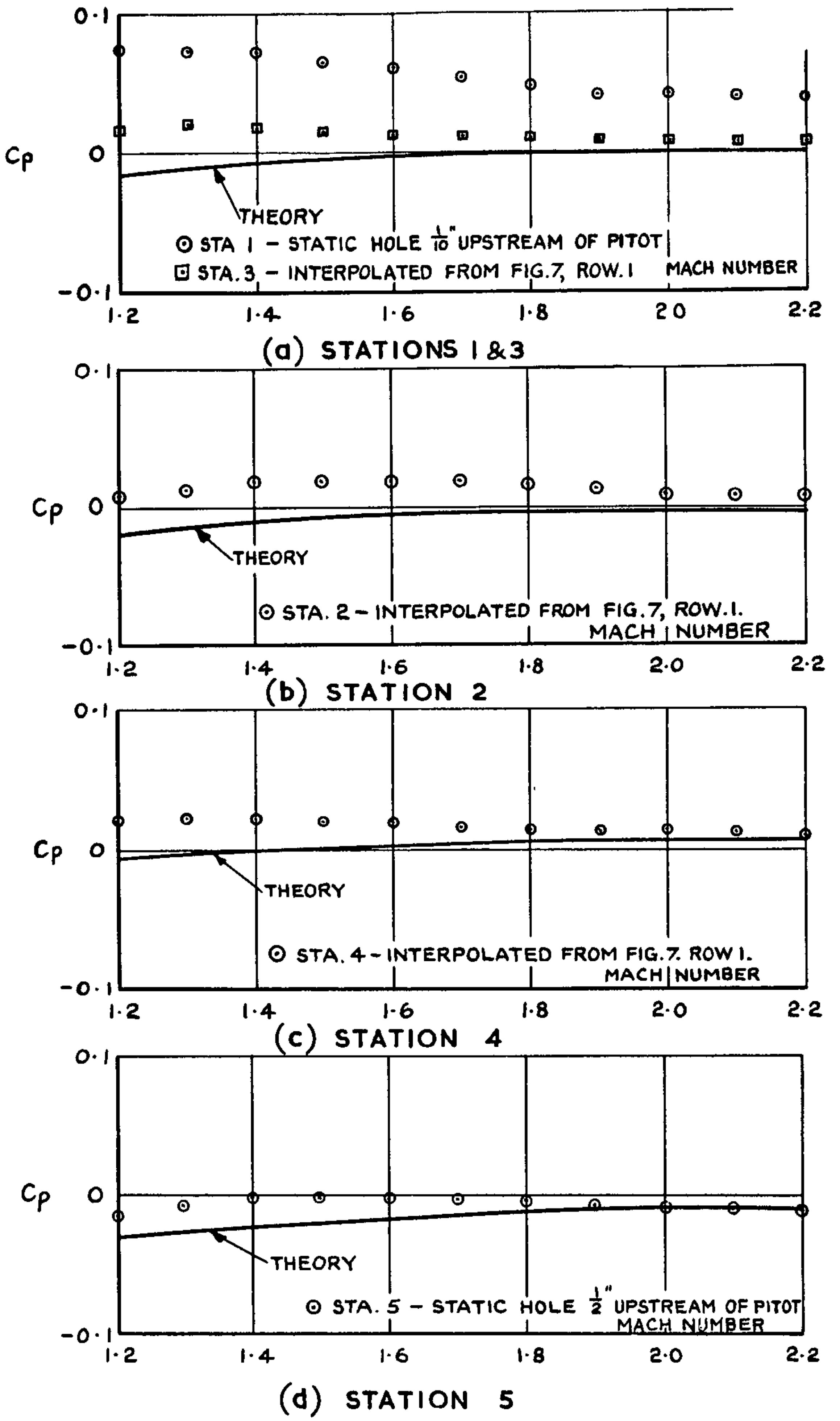
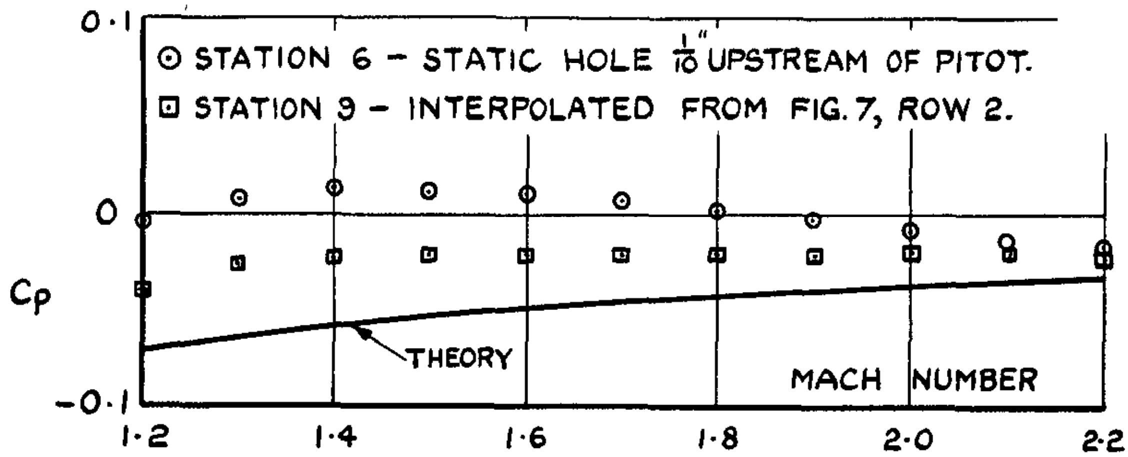
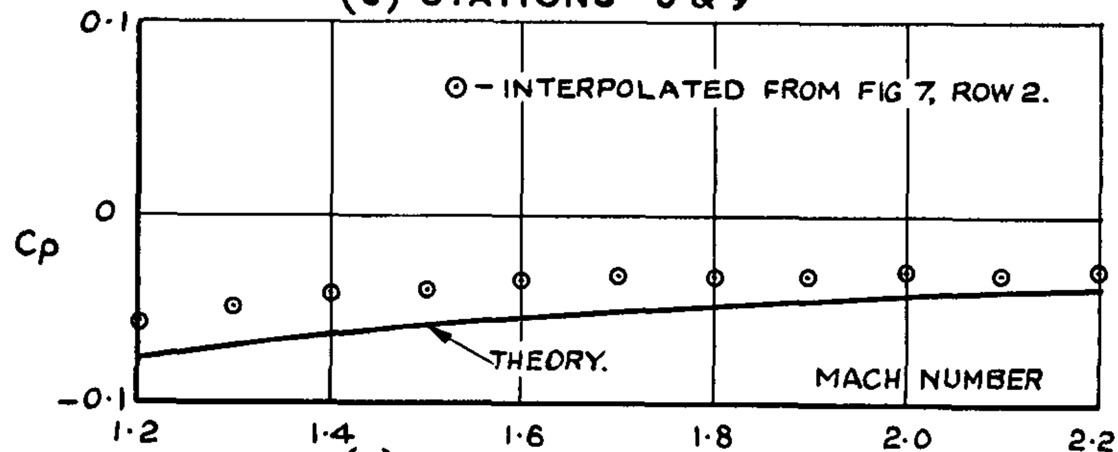


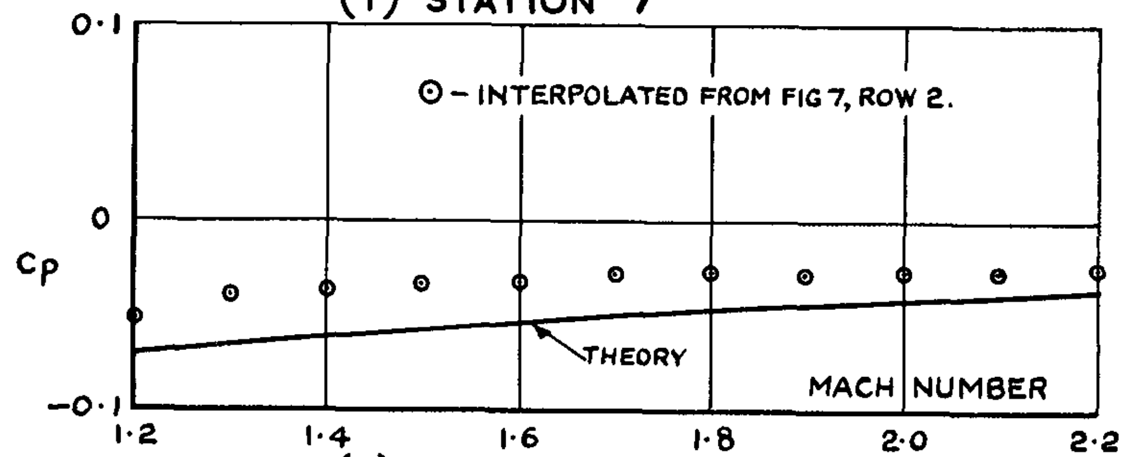
FIG. 8. STATIC PRESSURES, THEORY & EXPERIMENT.



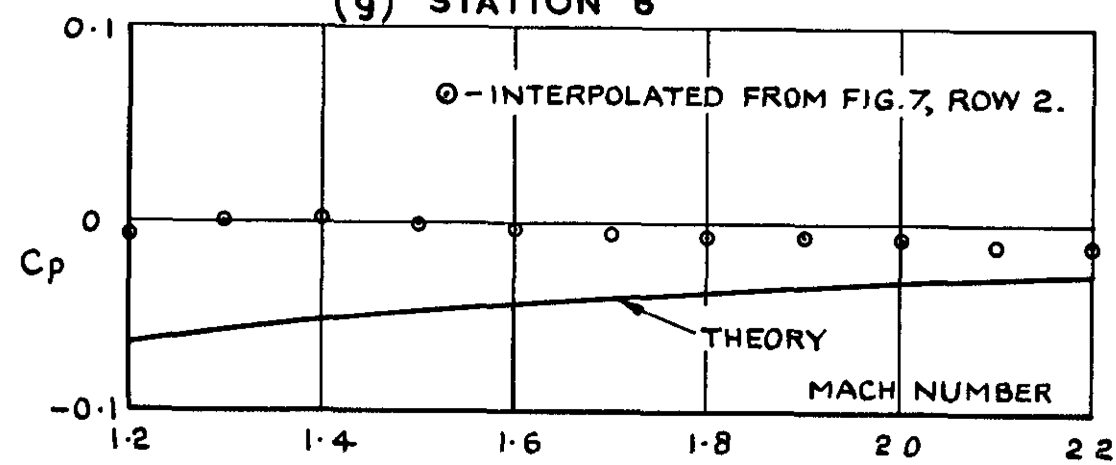
(e) STATIONS 6 & 9



(f) STATION 7



(g) STATION 8



(h) STATION 10

FIG. 8. (CONT.) STATIC PRESSURES, THEORY & EXPERIMENT.

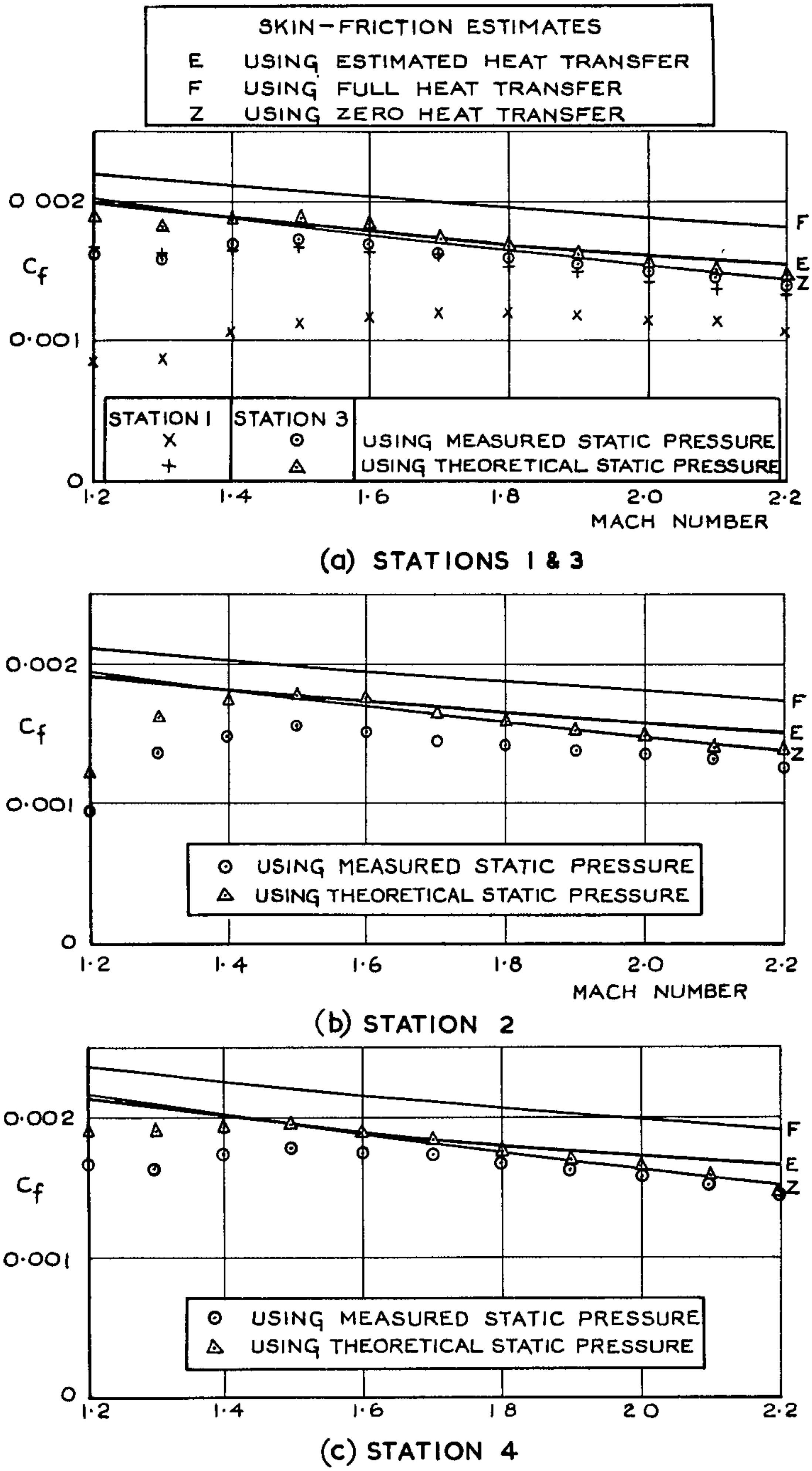
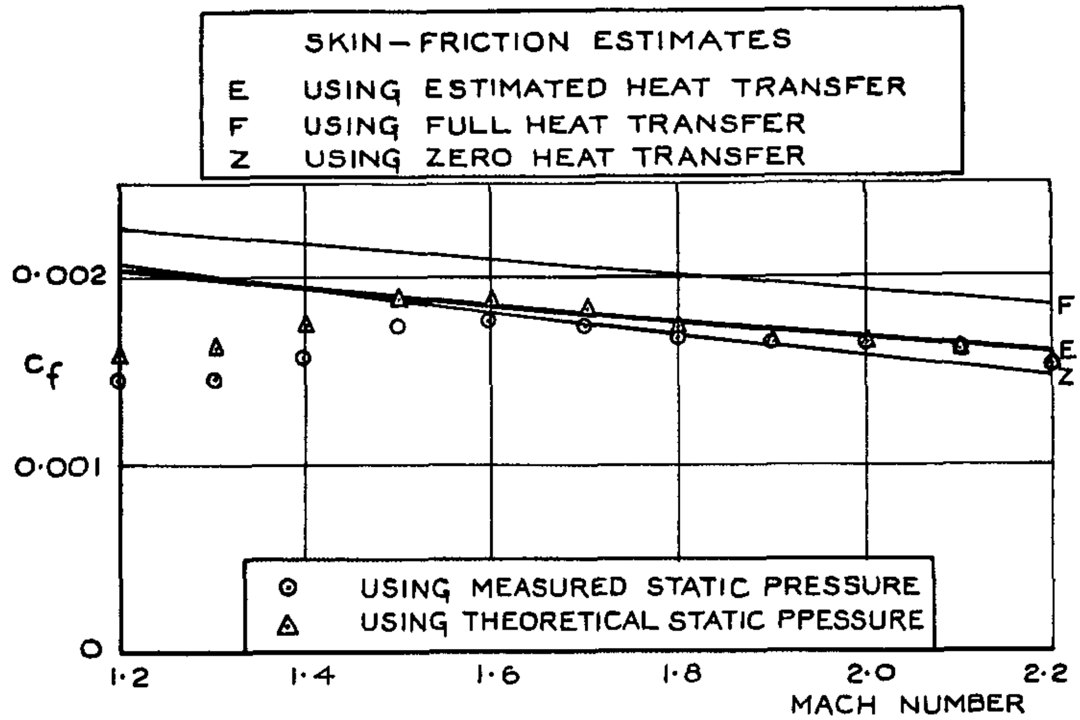
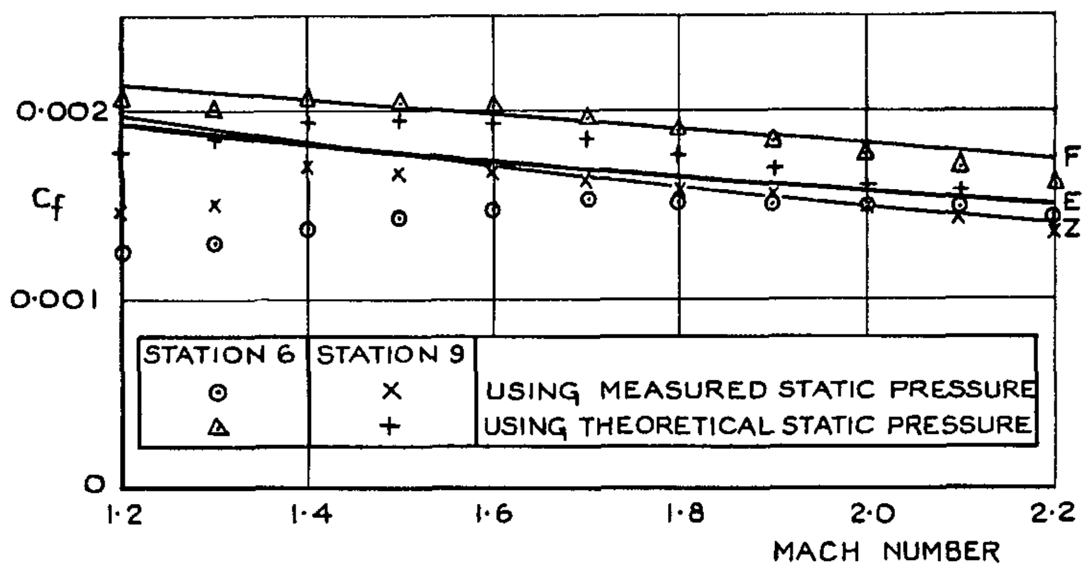


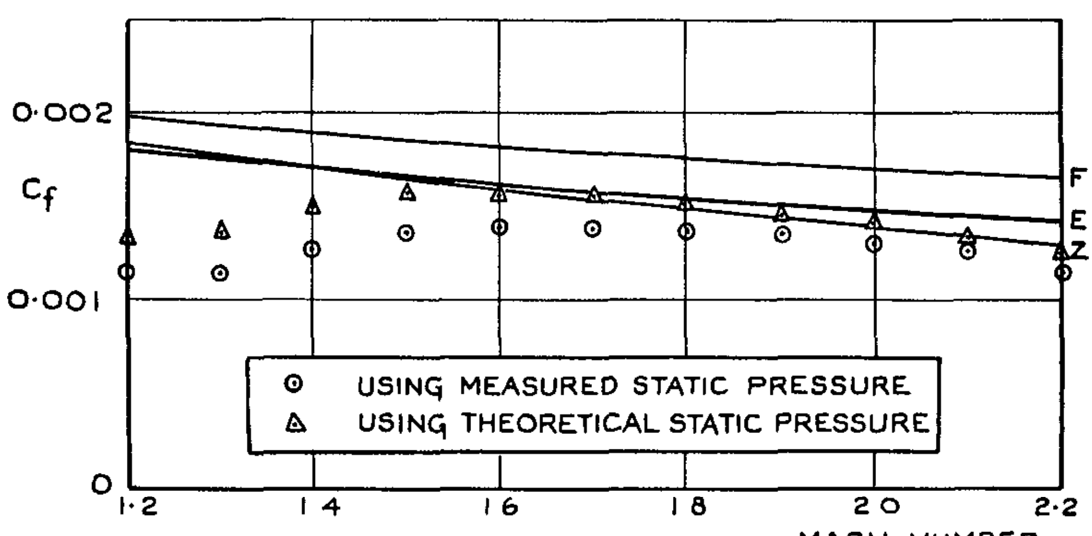
FIG. 9. SKIN-FRICTION COEFFICIENTS.



(d) STATION 5

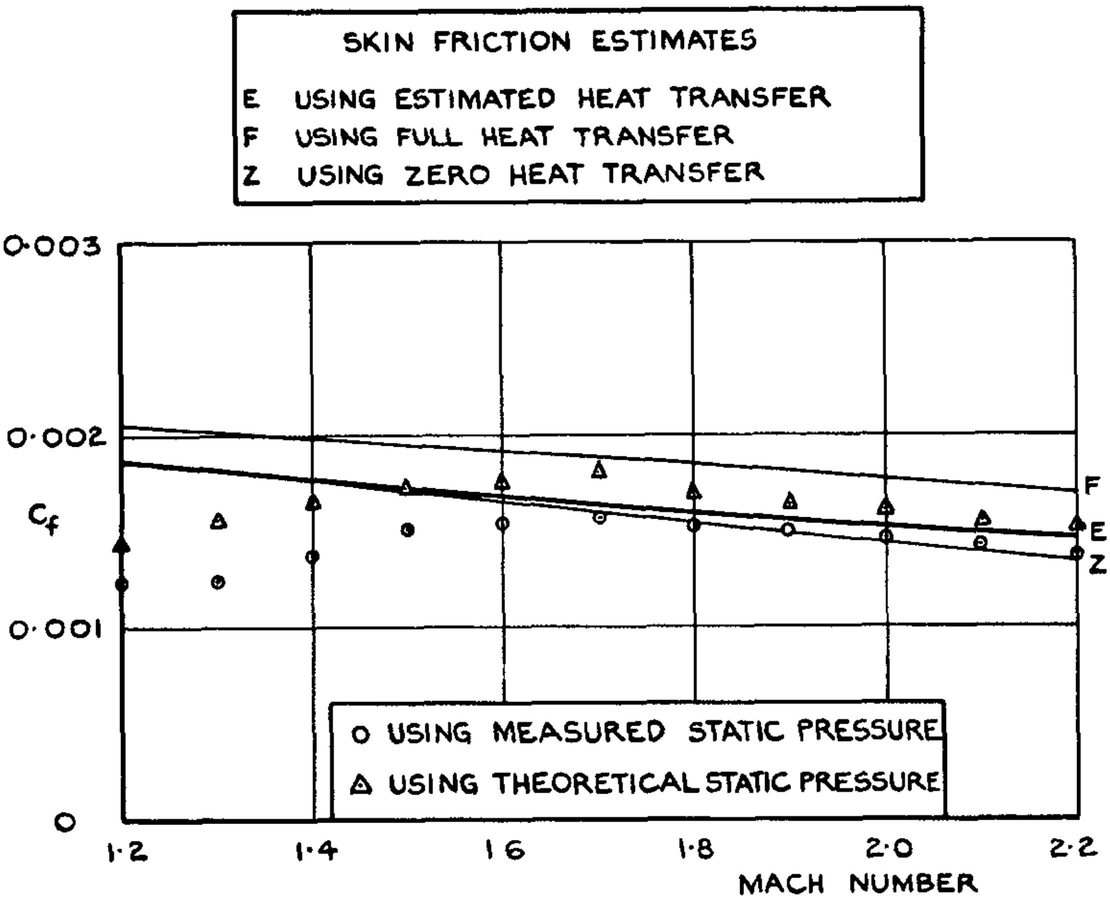


(e) STATIONS 6 & 9

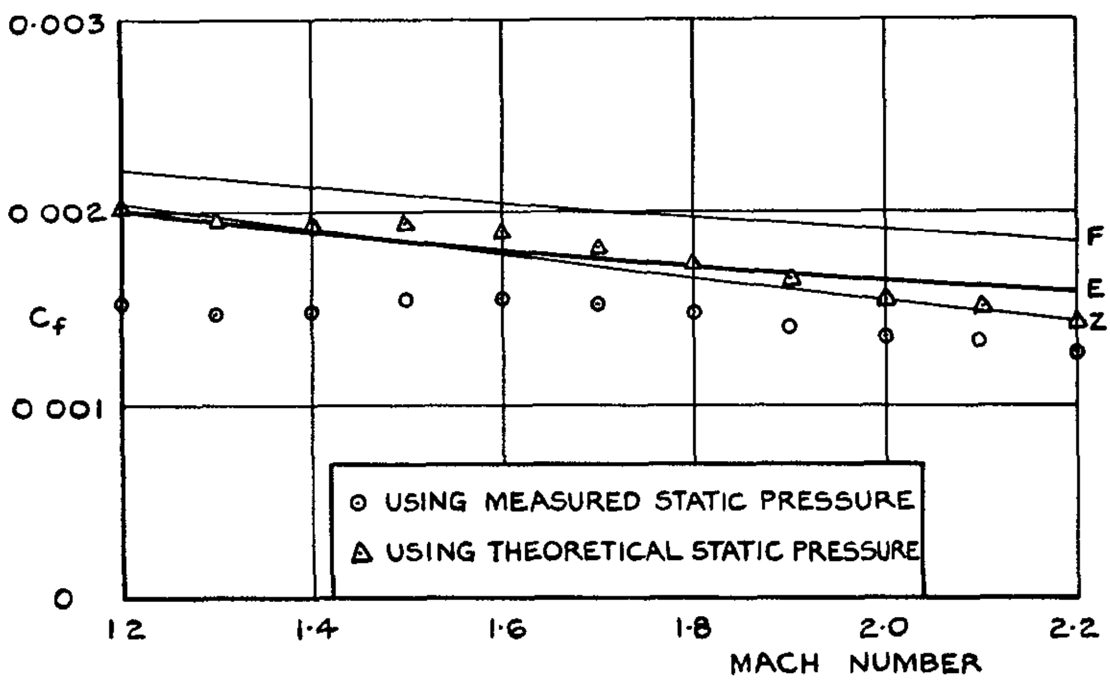


(f) STATION 7

FIG. 9. (CONTD.) SKIN-FRICTION COEFFICIENTS.



(g) STATION 8



(h) STATION 10

FIG.9 (CONCLD.). SKIN-FRICTION COEFFICIENTS.

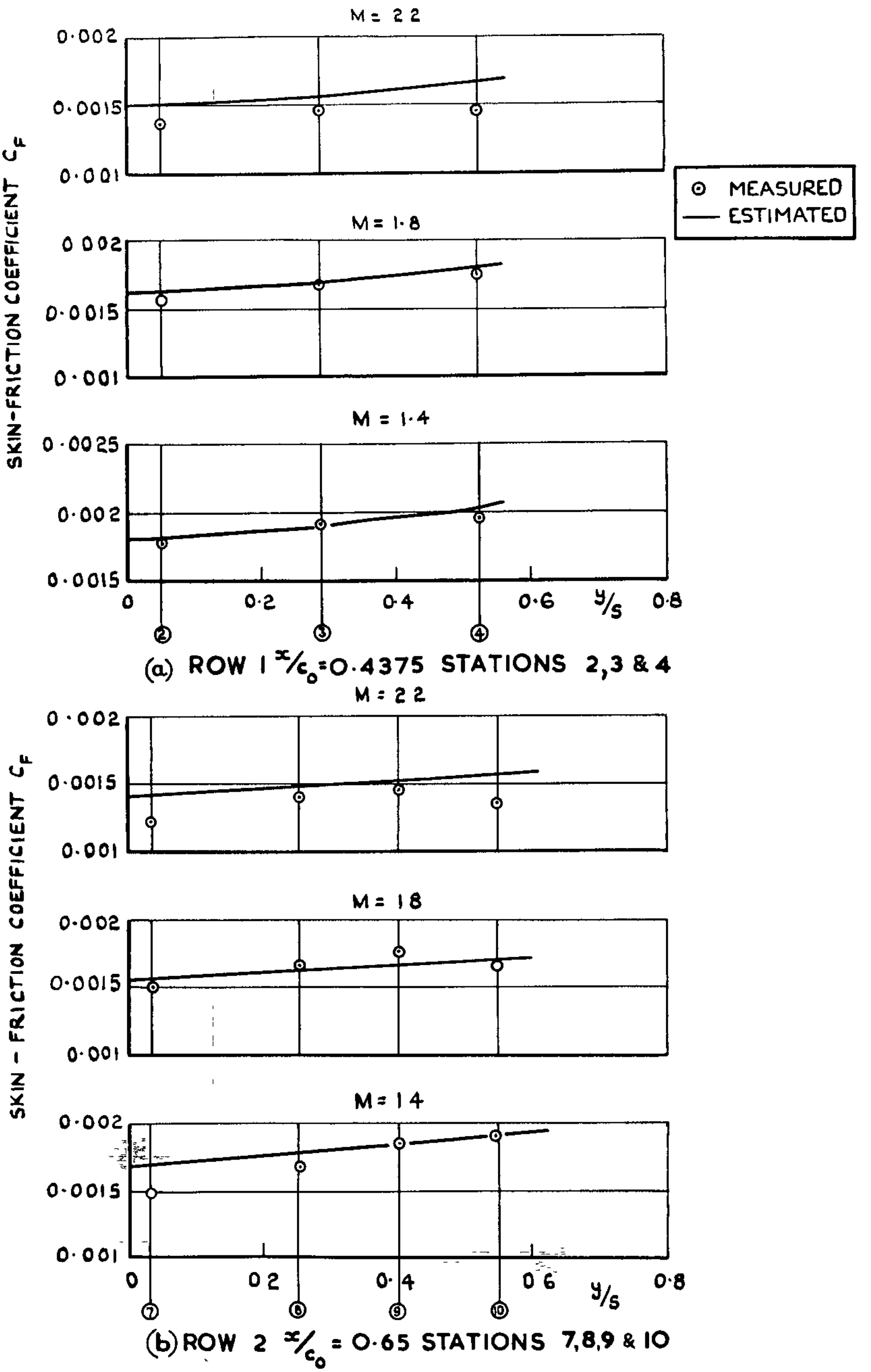


FIG.10. (a & b) SPANWISE VARIATION OF SKIN FRICTION.

A.R.C. C.P. No.711

533.693.3:
533.6.013.12:
533.6.082.3:
533.6.055

MEASUREMENTS OF SKIN-FRICTION USING SURFACE-PITOT TUBES IN FREE FLIGHT AT SUPERSONIC SPEEDS. Edwards, J. B. W. April, 1963.

Surface-pitot tubes have been used to measure local skin-friction in free-flight. Measurements were made at 10 stations on a simple delta-winged model flying at zero lift at Mach numbers between 2.2 and 1.2.

This Note describes the design and development of the instrumentation and the methods employed to analyse the experimental data. Good agreement between the measured and estimated values of skin-friction exists over most of the Mach number range.

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533.6.082.2:
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