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An Investigation of Transition Fixing Technique for a 10.5% Thick, 28° Sweptback Wing at High Subsonic Speeds and $R \simeq 3 \times 10^6$

By

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AN INVESTIGATION OF TRANSITION FIXING TECHNIQUE FOR A 10.5% THICK, 28° SWEPTBACK WING AT HIGH SUBSONIC SPEEDS AND $R \simeq 3 \times 10^6$

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SUMMARY

The results are given of force and pressure measurements on a $10\frac{1}{2}$% thick 28° sweptback wing of modern design mounted on a simplified body with fin. Boundary layer transition on the wing upper surface was left free or fixed by one of two alternative roughness bands.

The development of regions of separated flow and associated shock wave positions are shown to be very sensitive to transition conditions, particularly when the pressure gradient ahead of the main shock wave is small.

The applicability of the results to full scale is discussed. The "forward band" results generally give the best guide to drag, incidence and pitching moment for a given low C_L, but the lift and pitching moment breaks are too low at the higher Mach numbers ($M \geq 0.78$); the breaks are also too gradual. The "aft band" results should give a good indication of C_L for lift and pitching moment breaks for $M \geq 0.84$ but the deviations from low C_L behaviour become too gentle from C_L values very near the breaks for $M = 0.84$ and $M = 0.86$. At $M = 0.80$, C_L values at the breaks are too high with aft bands, as they generally are with no band, because of a laminar boundary layer interaction with the main shock.

In the general context of transition fixing technique, the results and analysis provide a good illustration of the care needed under conditions with an incipient rear separation (class B flow) and of the value of detailed pressure plotting. There are examples of where

(i) the results are invalidated as a result of a laminar boundary layer-shock wave interaction,

(ii) the results are invalidated by a major interaction between a rear separation and the boundary layer-shock interaction,

(iii) the results are spuriously optimistic, probably because the boundary layer is in a transitional state at the shock, i.e., with the roughness band just ahead of the shock.

To obtain reliable predictions of full-scale behaviour for wings similar to that tested here, tests will be needed with several (i.e., three or more) different roughness bands.

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* Replaces A.R.C.32 220 (ARA Report No. 16)

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1. Introduction/

1. Introduction

The results of the first series of tests on a model known as M.29/1 were presented in Ref.1. This showed a need for further tests to help in the interpretation of the data and its extrapolation to full scale conditions.

Consequently, the wing was modified by the installation of three chordwise rows of pressure plotting holes and was retested in August, 1968. In addition to measuring the pressure distributions for the conditions of the earlier tests, forces and pressures were measured using different transition fixing techniques.

The development of the pressure distributions has been described with some reference to results from section tests but detailed comparisons are outside the scope of this note.

The applicability of the results to full scale is discussed in some detail.

The reader is advised to obtain a brief overall appreciation of the results from the conclusions before reading the detailed discussion.

2. Details of Model and Tests

Details of the model geometry are given in Ref.2 where it is described as the "first configuration of model M.29/1" and in Ref.1. The wing - fuselage - fin configuration was used. The planview of the wing is shown in Fig.1, which also shows the layout of the main pressure plotting holes and of the alternative transition fixing strips. The three rows of pressure plotting holes added after the first test series were at $\eta = 0.438$, $\eta = 0.600$ and $\eta = 0.800$, where η is the proportion of the gross semi-span. The pressure points were on the upper surface of the port wing and the lower surface of the starboard wing.

For all of the tests described here the lower surface of the wing had a 0.05" wide transition strip of 0.004" - 0.005" dia. ballotini stuck to the wing with a thin film of Araldite with its leading edge on a straight line joining 5% root chord to 5% tip chord. The upper surface of the wing had either no band or an identical band (in planform position, ballotini size and strip width) to that on the lower surface or a 0.05" wide band of 0.006" - 0.007" dia. ballotini with its leading edge on a series of straight lines linking the following points:-

% nett semi-span	0	50	85	100
% c'	40	40	30	10

where c' is the local chord on an imaginary wing with a trailing edge extension to a straight line joining the trailing edge of the body side chord to the trailing edge of the tip chord. For convenience, the upper surface bands are referred to in the text and figures as the forward or aft roughness bands. Oil flow tests (in which the oil was not applied very close to the roughness) showed that there were no significant areas of laminar flow behind the forward roughness bands at $M = 0.75$, $C_L = 0.57$; $M = 0.78$, $C_L = 0.46$ and $M = 0.84$, $C_L = 0.59$ or behind the aft bands at $M = 0.84$, $C_L = 0.32$, 0.38 and 0.50.

The leading edge of the aft band intersects the pressure plotting stations at 0.49c for $\eta = 0.438$, 0.47c for $\eta = 0.60$ and 0.37c for $\eta = 0.80$.

The/

The tests with the forward band covered a Mach number range from $M = 0.50$ to $M = 0.92$ at a tunnel stagnation pressure, H, of 1.00 atmosphere giving a Reynolds number, based on the wing aerodynamic mean chord, varying from about $R = 2.30 \times 10^6$ at $M = 0.50$ to $R = 2.90 \times 10^6$ at $M = 0.80$ and $R = 3.00 \times 10^6$ at $M = 0.92$. The tests with the aft band were at $M = 0.70$ and $H = 1.00$ atmosphere and from $M = 0.80$ to $M = 0.92$ at $H = 1.20$ atmospheres with additional tests at $M = 0.84$ with $H = 0.80$ atmospheres and $H = 1.00$ atmosphere. With no roughness band on the upper surface the range covered was from $M = 0.80$ to $M = 0.92$, all at atmospheric stagnation pressure.

The incidence range for force and pressure measurements normally extended from near zero lift to beyond the pitch up or to model buffet. All tests were at zero sideslip. A few flow visualisation tests, using titanium oxide in oil, were done, mostly at $M = 0.84$.

3. Reduction of Results

The overall data were reduced to non-dimensional coefficient form using the model geometry as specified in Ref.2, viz,

gross wing area $= 3.452\ \text{ft}^2$.

aerodynamic mean chord, $\bar{\bar{c}} = 8.931$ in.

The pitching moments were referred to a point at $0.3\bar{\bar{c}}$ of the gross wing and 1.263 in. below the fuselage centre line.

Corrections were applied for the sting and balance deflection under load, the tunnel flow pitch angle, empty-tunnel buoyancy, tunnel constraint, blockage, blockage buoyancy and sting interference. Also, the drag data were converted to a constant Reynolds number of $R_{\bar{\bar{c}}} = 3 \times 10^6$, independent of test Mach number. This conversion assumed that the change in drag with Reynolds number was the same as the estimated change in profile drag with transition at 5% chord. It takes no account of any changes of separation drag with Reynolds number nor of any changes of transition position with Reynolds number. With the aft roughness band on the upper surface and laminar flow back to the band right across the span the variation of profile drag with Reynolds number would be about 5% less than with transition at 5% chord, so the use of the same correction for the aft band as for the forward band introduces errors of less than 0.0001 in C_D, very much less than the effects of unknown movements in transition position which must occur under some conditions with the aft band.

Further details of most corrections are given in Ref.1

4. Presentation of Results

The bulk of the figures are a systematic record of results from the tests with the three different upper surface transition conditions. The list of figures at the beginning of the note gives the order in which the results appear. There are clearly gross differences in the pressure distributions and consequently in the forces and moments measured with the different boundary layer conditions. These differences and the likely full-scale behaviour will be discussed later in order of increasing Mach number

The only aspect of the presentation which need be mentioned here is the layout of Figs.6-15 giving the pressure distributions. The normal layout of a half sheet of these figures shows chordwise distributions at the three stations on the wing for the three different transition conditions at the same Mach number and at similar values of incidence and lift coefficient. The origins for the three stations are staggered by increments of 0.10 in x/c and either 0.20 0.40 or 0.60 in C_p, but the distributions at a given station for the different conditions are not staggered. In a few cases two sets of distributions for one transition fixing condition are included, either because interpolation is needed to obtain comparable conditions, or because different sets are needed for comparisons near constant C_L than near constant incidence. In some other cases results from only one or two of the transition fixing conditions are shown.

5. Discussion of Results and Possible Extrapolation to Flight Conditions

The main discussion of the results will consider one test Mach number at a time in ascending order. The pressures and the forces and moments are so inter-related that they will be discussed together. It will often be necessary to use evidence from a higher Mach number than the one under discussion.

One feature that occurs repeatedly is the marked sensitivity of pressure distributions and hence of lift and pitching moment to the upper surface transition conditions. To avoid repetition later in the text, it is worth stating here the philosophy (based largely on Refs.3 and 4) that, to represent full scale conditions at which there is an interaction between a shock-induced separation and a rear separation, the boundary layer must be turbulent ahead of the shock, and the boundary layer thickness and profile at the trailing edge must be correctly scaled. For less extreme conditions, where there would be no rear separation at full scale, the boundary layer must be turbulent ahead of the shock, but the other requirement may be relaxed so long as the thickness and profile ahead of the shock are not so altered in relation to full scale that after interaction with the shock a rear separation is provoked. With natural transition, the requirement for correct thickness scaling at the trailing edge implies laminar flow to about 0.45c in these tests. If transition is provoked artificially, the trip causes additional thickening of the boundary layer so that the thickness at the trailing edge corresponds to that with natural transition somewhere ahead of the roughness band. Hence, the ideal position for the roughness band, from the boundary layer thickness criterion, is further aft than one would deduce by ignoring this effect. However, with the narrow roughness bands used in these tests the effect should be small and it has been ignored in the following discussion. In general for the Reynolds numbers of these tests, a laminar boundary layer-shock wave interaction will give an optimistic result and too thick a boundary layer at the trailing edge will give a pessimistic result. The terms "optimistic" and "pessimistic" are used here in the narrow context of lift and pitch breaks and divergences from linear curves and not in relation to stressing conditions.

The drag results will not generally be discussed in detail with the other results. The drag data from the first test series have been presented in Ref.1 and the results from the second series with the forward roughness band are in good agreement with these earlier results. For Mach number and C_L below the onset of significant flow separation, these forward band results should, with allowances for Reynolds number changes and items not represented on the model, give the best available guide to full scale drag. For some other conditions the flow outside the boundary layer is far better represented in tests with other transition fixes, but, in these cases, the extent of laminar flow on the wing

is/

is unknown and may be varying rapidly with M and C_L. It is therefore dangerous to try and draw general conclusions about drag from any tests without forward bands. It would be possible in principle, to use a sublimation method to measure the extent of laminar flow for specified conditions and to calculate the effects by strip methods but this would be very costly and could only be justified for a small number of important conditions.

5.1 $\underline{M = 0.50}$

The pressure distributions (Fig.6) develop with C_L in much the same way at the three pressure plotting stations until the onset of separation. By $C_L = 0.772$ the trailing edge pressure has diverged at all three stations and the leading edge suction peak has almost collapsed at the outer station. At this C_L the roughness band (only the forward band was used at this Mach number) is in a very steep adverse pressure gradient and introduces quite noticeable local distortions into the pressure distributions.

The $C_L - \alpha$ curve (Fig.2a) is linear up to about $C_L = 0.60$, $\alpha = 4°$, and then loses slope gradually. The $C_m - C_L$ curve (Fig.3a) breaks slowly in a pitch-up sense from a somewhat lower C_L. The local $C_L - \alpha$ curves (Fig.5a) derived from integration of the pressures show a gradual loss of lift curve slope at $\eta = 0.438$ and $\eta = 0.80$ from $\alpha \simeq 4°$ and a maximum local C_L on the outer station at $\alpha \simeq 6°$. Except at the extreme root, trailing edge pressures (Fig.16) show a gradual divergence from $C_L \simeq 0.60$ i.e., $\alpha \simeq 4°$.

All this suggests that the loss of overall lift curve slope is due to a combination of a trailing edge separation across most of the span and a leading edge separation spreading in from near the tip. The boundary layer at the trailing edge is, of course, too thick to give an accurate representation of full scale conditions. There may be some extra thickening due to the roughness band but, apart from this, it is impossible to get a thin enough boundary layer by alternative transition fixing techniques as the adverse gradient behind the leading edge suction peak will cause transition. There is some evidence at $M = 0.70$ of a laminar separation near the leading edge at high C_L and this situation is also likely to exist at $M = 0.50$. Thus significant scale effects are likely on both contributions to the stall at $M = 0.50$. The trailing edge separation would obviously be delayed at higher Reynolds number but the scale effects on the leading edge separation are less certain. The techniques necessary to improve the representation of full scale at this Mach number were not investigated in this case.

5.2 $\underline{M = 0.70}$

Figure 7a shows pressure distributions over a wide range of C_L with the forward roughness band. With increase in incidence a suction peak develops near the leading edge, becomes a supersonic region ending in a shock wave and widens as the shock moves back. The roughness band is generally near the top of the peak, rather than in the steep adverse gradient as at $M = 0.50$, and its local effect on the pressure distributions is not so obvious. However, Fig.7b shows some comparisons with tests in which an aft band was used, i.e., as far as the suction peaks are concerned, with no interference from a roughness band.

It/

It is seen from these comparisons that without the interference of the band the boundary layer separates, presumably as a laminar layer, over a small region near the "peak". The forward roughness band, whether or not it is far enough ahead of the shock to ensure a fully turbulent shock wave-boundary layer interaction, appears to suppress the tendency for a laminar separation and the associated lift loss (thickening of the boundary layer and lowering of suctions ahead of the shock), particularly at the outer stations. At $C_L \simeq 0.57$ the shock has probably moved back far enough for the forward roughness band to fix transition fully ahead of it. It seems therefore, that at $M = 0.70$ in contrast to higher Mach numbers, the aft band results with a laminar shock wave boundary layer interaction are pessimistic. The results with the forward band are to be preferred and some favourable scale effect is to be expected with the change to full scale.

5.3 <u>$M = 0.78$</u>

The development of the pressure distributions at the three stations (Fig.8) is basically similar. At low C_L, a leading edge suction peak forms as incidence is increased. A separate "hump" also forms behind the peak at 20 - 30% chord. This is similar to the distributions found in two dimensional tests on the corresponding aerofoil at equivalent conditions. With increase in C_L a shock is formed, apparently from the amalgamation of the leading edge peak and the "hump". The distributions for $C_L = 0.529$ show the shock about 0.06c further forward at $\eta = 0.438$ than at the other stations, where the middle of the steep pressure rise is at about 0.33c. With further increase in C_L the shock wave moves back to about 0.40c at $\eta = 0.438$ and about 0.36c at $\eta = 0.60$ but forward at $\eta = 0.80$ with a corresponding divergence of trailing edge pressure. With still further increase in C_L the forward movement of the shock and the divergence of the trailing edge pressure (Fig.16) progress inboard. The force and moment results show a loss of lift curve slope and a progressive pitch up as the flow separation spreads inboard, indicated on the pressure distributions by forward movement of the shock (Fig.8) and trailing edge pressure divergence (Fig.16) and on Fig.5a by maxima in the local C_L values. The curves of Fig.5a for $M = 0.78$ are rather different in character from those at lower M, breaking sharply (for the outer two stations) at maximum local C_L with little previous curvature. The conditions for maximum local C_L at the two outer stations correlate roughly on a "Mach number normal to shock" basis, the lower maximum C_L at $\eta = 0.80$ resulting from a further forward, less swept shock at that station.

With the transition band at 5% chord, the only position tested at this Mach number, the boundary layer is too thick to represent full scale conditions at the shock and at the trailing edge. With a further aft transition band it might have been possible to get nearer to full scale conditions but the shock is too far forward to avoid a laminar interaction and still keep the boundary layer thin enough at the trailing edge. In addition, even if a band were placed as far back as possible to keep ahead of the shock wave-boundary layer interaction there is a risk that the adverse gradient behind the leading edge suction peak might fix transition ahead of the band.

There/

There would certainly be an increase in the C_L for lift break and pitch up with a change to full scale Reynolds number. There is evidence that at higher Mach numbers the corresponding C_L improvement is about 0.10, but it is likely to be less than this at $M = 0.78$. The changes after the break would probably be more rapid at full scale.

5.4 $\underline{M = 0.80}$

The pressure distributions at low C_L (Fig.9a) develop rather differently on the inner and outer stations. At $\eta = 0.438$ two shocks develop, one from the "hump" in the pressure distribution around 20% chord as at $M = 0.78$ and the other from the back of the roof top near 60% chord. At $\eta = 0.80$ a single shock develops from the "hump". At the intermediate station the behaviour could be classed as somewhere between the others with depending on C_L and transition conditions. At $C_L = 0.437$ with the forward band there are two shocks about 15% chord apart but at $C_L = 0.461$ with no transition band there is either a single shock with the pressure rise on the surface spread over about 20% chord, as might occur with a laminar boundary layer ahead of the shock, or perhaps two shocks about 10% chord apart. The line on that Figure has been drawn to suit the first interpretation.

The pressure distributions at C_L values between 0.56 and 0.70 (Fig.9b) include several features of importance to the interpretation of the forces and moments. The basic shock pattern on the upper surface has now developed into that which applies, except at low C_L, for all higher Mach numbers. Inboard are two converging shocks, a "forward" one sweeping back from near the root leading edge and a "rear" one at the end of the "roof top", with a single "outboard" shock from their intersection. For the conditions of the left hand side of the figure, the intersection of the shocks is outboard of $\eta = 0.438$, whereas, on the right hand side of the figure, a single shock at $\eta = 0.438$ shows that the intersection is inboard of that station.

Considering the left hand side of Fig.9b, at $\eta = 0.438$ the aft roughness band is between the shock waves. Oil flow tests on this and other wings have shown that, in this sort of condition, the forward shock would regularly provoke transition, assuming the boundary layer was laminar up to the shock; hence the aft band is unnecessary at this station in this case. At $\eta = 0.60$ the aft transition band is at 0.47c, too far back to fix transition ahead of the shock and the pressure distribution just ahead of the shock appears to have the rounded form typical of a laminar interaction. At $\eta = 0.80$ the aft transition band is at 0.37c, which may be just far enough ahead of the shock but this is doubtful. The shape of the pressure distribution immediately ahead of the shock does not suggest a laminar interaction but is more likely to be the combined result of a transitional boundary layer interaction and the local perturbations due to the rougness band than to a turbulent boundary layer interaction.

With the shock pattern present under these conditions (see Fig.9b) the position of the forward shock on any station which it crosses is very sensitive to incidence and to pressure distribution ahead of it on other sections. Detailed comparisons for the different transition conditions could therefore be very confusing in this respect. There are appreciable differences between the different

roughness/

roughness configurations in the positions of the rear shock at $\eta = 0.438$ and the single outboard shock at $\eta = 0.60$ even before any significant divergence of trailing edge pressure at these stations. However, it is not clear whether this is an indication of the sensitivity of sectional behaviour to boundary layer conditions or whether this, too, is a three dimensional effect, with the shock position at $\eta = 0.60$ being influenced by that at $\eta = 0.80$ (where the trailing edge pressure has started to diverge with the forward band) and the rear shock position at $\eta = 0.438$ depending on the spanwise position of the intersection point of the three shock system.

For the higher C_L values on the right hand side of the Figure, the roughness band is too far back at all three pressure plotting stations to fix transition ahead of the shock. With the forward band the trailing edge pressure has started to diverge at all three stations. At $\eta = 0.438$ the lower surface pressure shows a further divergence than that on the upper surface. The technique of pressure plotting the upper surface of one wing and the lower surface of the other can probably be blamed for a misleading result here. The lower surface pressure distributions are not severe enough to cause separation so the greater divergence must be due to an asymmetry with a more severe upper surface separation on the wing which is pressure plotted on the lower surface.

The similarity of pressure distributions and forces for the aft band and no band supports the suggestion that the aft band is too far back to give any reliable data at this Mach number and leads to a conclusion that at full scale the lift break and pitch up would probably occur at some C_L between those given by forward and aft band results.

At $C_L \simeq 0.70$ the comparison of pressure distributions at $\eta = 0.80$ with the aft roughness band and with no roughness band on the top surface (Fig.9b) shows a more rearward shock position and better trailing edge pressure recovery with the aft band. This suggests that a roughness band on the surface beneath the interaction region between a shock wave and a laminar boundary layer can delay the forward movement of the shock in some way. Even with similar results at $M = 0.82$, 0.86, the evidence here is hardly strong enough to prove that this is a genuine effect and not just scatter resulting from variations in the extent of transition induced well forward by some means (e.g., dirt). However, there is sufficient unpublished evidence on another model to show that the effect is genuine. The results with an aft band just a little too far back can therefore be more optimistic than for transition free tests.

5.5 $\underline{M = 0.82}$

At low C_L there is quite a marked difference in the development of the pressure distributions at $\eta = 0.438$ and at the other stations (Fig.10a). At $\eta = 0.438$ a shock forms from the end of the "roof top" before the leading edge peak and the "hump" behind it begin to amalgamate into a forward shock. On the other two stations there is no sign of a shock forming from the end of the roof top but separate shocks form from the leading edge peak and the "hump" just behind it and move well back before amalgamating. Both these forms of development can be seen in the results of section tests on corresponding aerofoils but at different Mach numbers. Hence, at $\eta = 0.438$ under these conditions, the planform kink, the root effects on the subcritical pressure distributions and the three dimensional shock system developing from them produce a result that looks qualitatively like the effect of a reduction in effective sweep in relating the pressures over the three-dimensional wing to two-dimensional data.

The/

The pressure distributions for $C_L \simeq 0.45$ in Fig.10a show the sensitivity of the three-dimensional shock pattern to comparatively small changes ahead of the forward shock. Compared with those with the aft band, the distributions ahead of the forward shock with the forward band (admittedly at a slightly higher C_L) show higher suctions corresponding to higher local Mach numbers with a consequent increase in the sweep angle of the forward shock. Hence at $\eta = 0.60$ the forward shock is much farther aft with the forward band.

For lift coefficients between about 0.40 and 0.60 it is just possible that the aft band is far enough ahead of the outer wing shock to give a turbulent boundary layer interaction. It would require considerable extra testing to establish whether or not this is so. If the shock-boundary layer interaction is turbulent, then the C_L for lift break and pitch up given by this configuration should be representative of full scale conditions because the boundary layer thickness at the trailing edge would be approximately correctly scaled. If not, then the tunnel measurements could be optimistic.

Whatever the correct interpretation of data for $0.40 < C_L < 0.60$, there is no doubt that at this Mach number the results with the aft band or with no band on the upper surface become optimistic relative to full scale, because of a laminar interaction, immediately the lift and pitching moment curves for these transition conditions start to break, i.e., above $C_L \simeq 0.60$. Comparison of results with the aft band and with no band was the subject of further comment in the discussion of results for $M = 0.80$.

Considering absolute levels of lift and pitching moment curves as distinct from the positions of breaks, it is necessary to consider differential boundary layer growth on the two surfaces. With the aft band on the upper surface and a forward band on the lower surface (as in the tests referred to briefly as "aft band") the differential growth must be very different from full scale so the actual values of α and C_m at a given C_L from the aft band tests will not be representative of full scale. For low enough C_L, i.e., before the onset of separation with the forward transition band, α and C_m at a given C_L from the forward band tests may well be nearer to the full scale values, since these quantities tend to be more sensitive to differential transition movements on the upper and lower surfaces than to Reynolds number changes with transition near the leading edge on both surfaces. However, there is a small effect of the roughness band position, and presumably of Reynolds number, on the shock position prior to the onset of separation which for a given C_L would give a lower and probably a lower C_m at full scale.

5.6 <u>$M = 0.84$</u>

Again, as at $M = 0.82$, there are marked differences in the development of the pressure distributions at the three stations looking qualitatively like the result of a smaller effective sweep angle inboard. It appears that the effects of the planform kink and the root extend to at least $\eta = 0.60$ since this station now develops a shock from the end of the roof top whereas at $\eta = 0.80$ the main shock appears to grow from the middle of the roof top region.

Pressure/

Pressure distributions are presented for the aft position only of the top surface roughness band at this Mach number, except for $C_L \simeq 0.46$ (Fig.11b). At this condition the forward shock at $\eta = 0.438$ is ahead of the aft roughness band. With the aft band, the pressure rise through the forward shock is spread over about 0.15 of the chord, suggesting a laminar interaction and that the adverse gradient between about 0.05c and 0.10c is insufficient to trip the boundary layer. Similarly at $C_L = 0.588$ (Fig.11c) the boundary layer appears to remain laminar to the forward shock.

At the other stations, apart from near $C_L = 0.30$ there is no adverse gradient sufficient to fix transition ahead of the aft roughness band or the main shock (whichever is further forward). Even at $C_L = 0.35$, where the adverse gradients at the rear of the leading edge peak are quite steep, an oil flow test with transition free on the upper surface showed laminar flow back to the shock over significant parts of the span although over other parts there was turbulent flow from near the leading edge. For C_L less than about 0.60 the aft band is far enough forward to fix transition ahead of the shock and, since it is in about the right place to give a boundary layer thickness at the trailing edge equivalent to full scale conditions, the outer wing behaviour should be representative of full scale. Since the start of the lift break and pitch up are due to the outer wing the tunnel tests with the aft band should give reliable results for the C_L at which these phenomena occur. However, very little of the curves for $C_L > 0.60$ satisfy the condition of turbulent flow ahead of the shock as the forward movement of the shock associated with lift break and pitch up rapidly brings it close to the roughness band. An oil flow test at $C_L = 0.57$ showed several small regions of laminar interaction scattered across the outer wing; at this C_L, none of these are large enough to have much influence on the overall position of the shock but, to judge from the pressures, they must become much more pronounced with a small further increase in C_L. Since a laminar interaction usually gives an optimistic result and a laminar interaction complicated by interference from a roughness band can give even more optimistic results (see $M = 0.80$), the full scale lift break and pitch up would be expected to be sharper than the tunnel (aft band) results and consequently any useable C_L derived from these tunnel results would be optimistic. The forward band results are of course very pessimistic in this context.

Remarks made for $M = 0.82$ about full scale values of α and C_m at a given C_L apply equally to $M = 0.84$. The hump in the $C_m - C_L$ curve for $0.25 < C_L < 0.45$ and the corresponding dip in the $C_L - \alpha$ curve for the aft fix (and for transition free on the top surface) are probably due to the variations in transition position on the outer wing mentioned earlier.

The effect on the pressure distributions of the comparatively small changes in Reynolds number achieved by testing at three values of stagnation pressure cannot be isolated because the varying dynamic pressure results in different sting and balance deflections and consequently different corrected angles of incidence for nominally comparable data points. It is clear, however, from the example shown in Fig.11b that the effects are very small. The effects

of/

of the Reynolds number variation on the lift (Fig.2b) and pitching moment (Fig.3b) at C_L for which the results are considered applicable to full scale are merely changes in the level of the curves, presumably due to different boundary layer displacement effects. The differences are more marked at high C_L under conditions where this is a laminar boundary layer-shock wave interaction.

5.7 $M = 0.86$

At this M, the main shock is formed well back on the section at each station. A forward shock forms at the inner station eventually and moves back quite slowly with increase in C_L, being near the crest when the flow starts to separate at the trailing edge, even with the aft transition band.

In contrast to lower speeds, the trailing edge pressure at $\eta = 0.438$ diverges to $C_p \simeq -0.10$ while that at the other stations remains positive. It then becomes slightly less negative with increasing C_L before diverging further after divergence at the other stations.

The first divergence at $\eta = 0.438$ occurs with the aft roughness band at a condition where the transition is induced by the forward shock well ahead of the roughness band and of the transition position needed for a correct representation of the full scale boundary layer thickness at the trailing edge. The boundary layer at this station is therefore unavoidably too thick and this trailing edge pressure divergence will not be completely representative of full scale. The partial recovery from the divergence as C_L is increased and as the forward shock and transition position move back is an indication of the sensitivity to boundary layer thickness and suggests that at full scale, trailing edge separation at this station would not occur at much lower C_L than it does outboard.

The pressure distributions for $C_L \simeq 0.64$ (Fig.12b) show that with the aft roughness band, the shock at $\eta = 0.80$ has moved forward sufficiently for the roughness to be in the shock wave-boundary layer interaction region. This appears to be just true at $\eta = 0.60$ as well. The shapes of the $C_L - \alpha$ and $C_m - C_L$ curves beyond about $C_L = 0.60$ are therefore liable to be more favourable than at full scale.

At $C_L \simeq 0.75$, the comparison of pressure distributions with the *aft roughness band and with no roughness band on the top surface* (Fig.12c) shows a more rearward shock position at both $\eta = 0.60$ and $\eta = 0.80$ with the aft band. This effect is referred to in the discussion of results for $M = 0.80$

5.8. $M = 0.88$

Pressure distributions shown in Fig.13 are for the aft upper surface roughness band only. At $C_L = 0.206$ (Fig.13a) there are suction peaks near the fron of the upper surface and the adverse gradients behind these at $\eta = 0.438$ and $\eta = 0.60$ are probably sufficient to fix transition*. A poorer trailing edge pressure recovery at these stations than at $\eta = 0.80$ and a convergence of $C_L - \alpha$ curves for the various transition bands around this C_L tend to

support/

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* Similar peaks were present at $M = 0.86$, $C_L \simeq 0.27$ (Fig.12a) but, to avoid excessive repetition, they were not discussed.

support this suggestion. However, at this C_L, the adverse effects, relative to full scale, of the upper surface boundary layer being too thick are probably small compared with corresponding effects on the lower surface. Here, the usual distribution over the rear of the section associated with "rear loading" has been lost by boundary layer separation effects.

With a lower surface boundary layer representative of full scale, a condition which could be approached but not reached by testing with a further back roughness band, this effect would be expected only at lower C_L. However, for C_L below about 0.05, an aft band on the lower surface would be no use since transition would be fixed well forward by the adverse gradient behind a suction peak (similar to those shown for $M = 0.92$ in Fig.15a).

At $C_L = 0.465$ (Fig.13a) the results should in general be very close to full scale. The leading edge peaks appear too small to fix transition so, over most of the upper surface, the conditions specified earlier for good simulation are approximately satisfied. (The exceptions are small areas very near the root and tip where the boundary layer will be too thick. These are likely to affect only the precise shape of curves very near to the breaks. Transition will be farther forward at these positions than over most of the span because of the forward shock near the root and the position of the roughness band near the tip). The lower surface boundary layer will be too thick but there is no rear separation here and the pressure distribution ahead of the shock makes the shock position insensitive to boundary layer conditions. The experimental values of α and C_m for C_L near this condition would be influenced by the excess lower surface boundary layer thickness, so more positive values of both would be expected full scale.

At $C_L = 0.573$ and 0.663 (Fig.13b) the same remarks apply as for $C_L = 0.465$ except that there may be a small part of the span near $\eta = 0.6$ where the top surface boundary layer is still laminar or transitional at the start of the shock wave-boundary layer interaction region. Since on either side of this narrow area there is a thin turbulent layer ahead of the shock and the critical area of the wing is further outboard this laminar interaction, if present, probably has little overall effect.

5.9 $\underline{M = 0.90}$

At this Mach number the features discussed at $M = 0.88$ are mostly present in a more extreme form, except that the shocks are far enough back to avoid any laminar interactions in the test range with the aft band.

The lower surface separations at low C_L result in low lift curve slope and severe static instability (tail off at least) but it must be remembered that the lower surface boundary layer is "overfixed" so that C_L below which these phenomena occur would be reduced at full scale.

5.10 $\underline{M = 0.92}$

At low C_L the pressure distributions (Fig.15a) show separated flow at the rear of both wing surfaces. The forward movement of the lower surface shock on the outer wing as C_L is reduced appears to "freeze" between 40%c and 50%c

presumably/

presumably because of the rapid reduction in Mach number ahead of it as it moves forward. For the range over which the shock position is sensitive to C_L it is presumably also sensitive to the boundary layer condition. However, at least for C_L less than about zero, tests with a farther back roughness band on the lower surface would be rendered misleading and of little additional value because transition would be fixed at the outboard sections by the adverse gradient behind the leading edge peak.

Most of the upper surface pressure distributions show about 5% chord difference in shock position with the different roughness bands, i.e., less sensitivity to boundary layer thickness than results at lower M with a rear separation. In addition the shock movement with C_L is slow. The main exception is on the outer station at high C_L when the pressure distribution ahead of the shock is almost a roof top.

Otherwise the flow is accelerating as it approaches the shock and the relative insensitivity of shock position to both C_L and boundary layer thickness has the same cause as the lower surface "freeze" discussed above.

6. Conclusions

The pressure distributions measured at three stations on the wing develop with Mach number and incidence in a similar manner to those measured in corresponding section tests. The main features of the development occur at lower M at the planform kink station than farther outboard, showing that for this station on this wing the combined effects of the planform kink and the root correspond qualitatively to a lower effective sweepback.

At conditions where shock strength or adverse pressure gradients are sufficient to induce separation, the development of the regions of separated flow and the associated lift and pitching moment behaviour are very sensitive to the transition fixing technique. In order to predict full scale lift and pitching moment break boundaries and the behaviour beyond them, it is important where possible to satisfy the conditions that the boundary layer is turbulent ahead of the main shock and that the boundary layer at the trailing edge is correctly scaled. At the Reynolds numbers of these tests, the latter implies laminar flow to about 0.45c.

The two upper surface roughness bands used in these tests are the extremes of a useful range. At least for $M \geq 0.78$, the forward band gives lift and pitching moment breaks at too low a C_L and the breaks are generally too gentle. It is, however, needed for determination of drag at C_L below separation boundaries and for absolute values of C_m and α over a large part of the M, C_L range.

Apart from being farther forward at the tip to keep ahead of a curved shock front, the rear band was about in the right place to satisfy the above conditions for good representation of full scale behaviour, except that

(i) for Mach numbers below $M = 0.82$ and perhaps at $M = 0.82$, the main shock wave was too far forward for the band to fix transition ahead of it and, for the same reason, one cannot reach far beyond the lift and pitching moment breaks at $M = 0.84$ and $M = 0.86$,

(ii)/

(ii) under some conditions a leading edge peak or a forward shock prevents a long run of laminar flow.

Where (i) only, above, applies, but the main shock is fairly well back, it would be possible to do a series of tests with varying transition band positions, all sufficiently forward to fix transition ahead of the main shock and to make a reasonable extrapolation to full scale. The bands on both surfaces should be moved together to reduce effects of differential transition movements on C_m and α for a given C_L.

Where transition occurs naturally ahead of a roughness band any further movement of the band is generally useless. However, on this wing the separations tend to spread in from near the tip so transition ahead of the band on the inner wing has little effect on lift break boundaries.

References/

No.	Author(s)	Title, etc.
1.	P. G. Hutton, D. Morton, R. W. A. Berrington and A. B. Haines	ARA Model Test Note M.29/1/1
2.	P. G. Hutton	ARA Model Test Note M.29/1
3.	H. H. Pearcey, J. Osborne and A. B. Haines	The interaction between local effects at the shock and rear separation - a source of significant scale effects in wind-tunnel tests on aerofoils and wings. AGARD Conference on Transonic Aerodynamics, Paris. September, 1968.
4.	J. A. Blackwell, Jr.	Effect of Reynolds number and boundary-layer transition location on shock-induced separation. AGARD Conference on Transonic Aerodynamics, Paris. September, 1968.

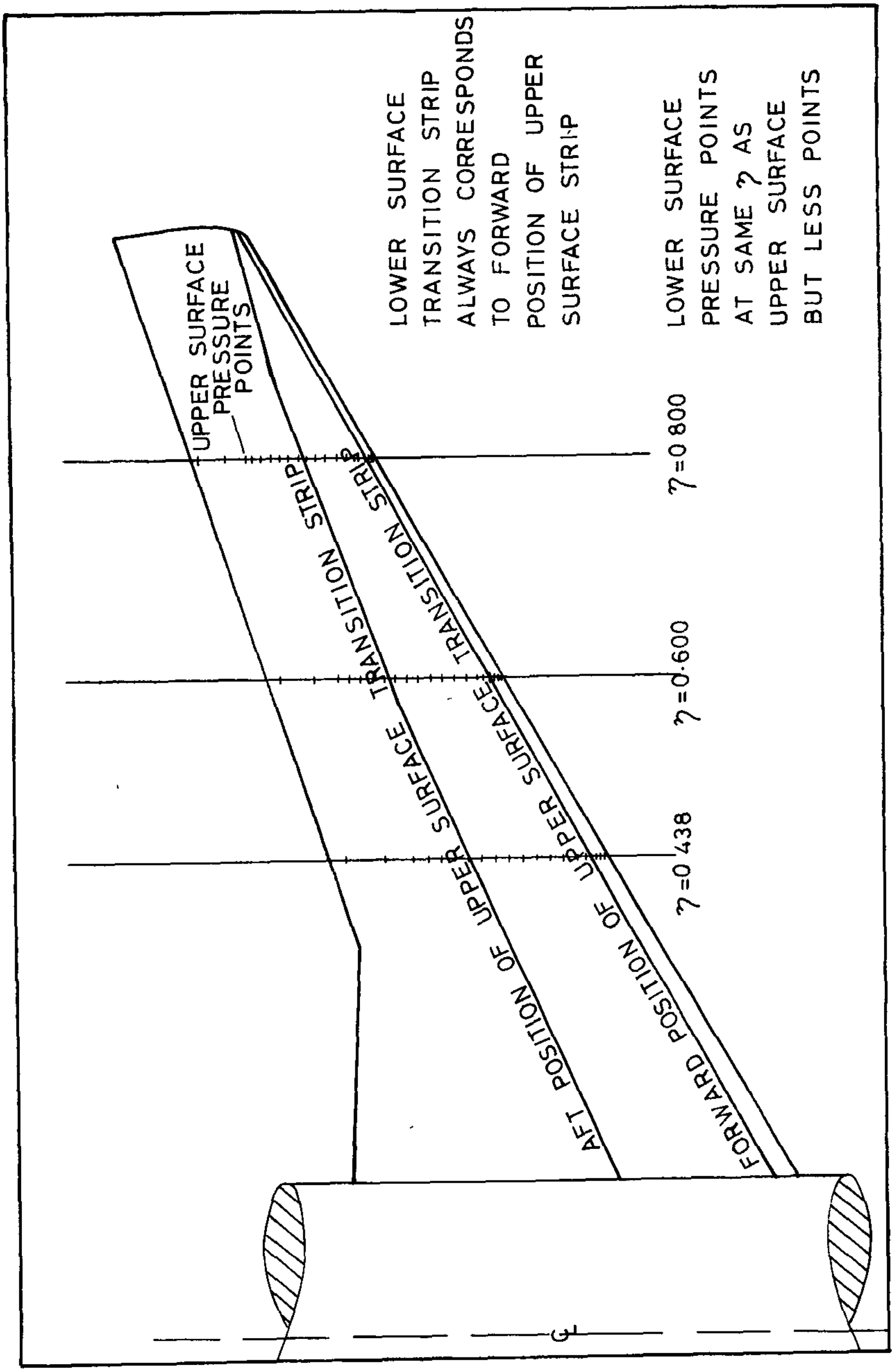


FIG. 1. WING PLANFORM, PRESSURE PLOTTING AND TRANSITION FIXING

FIG.2a

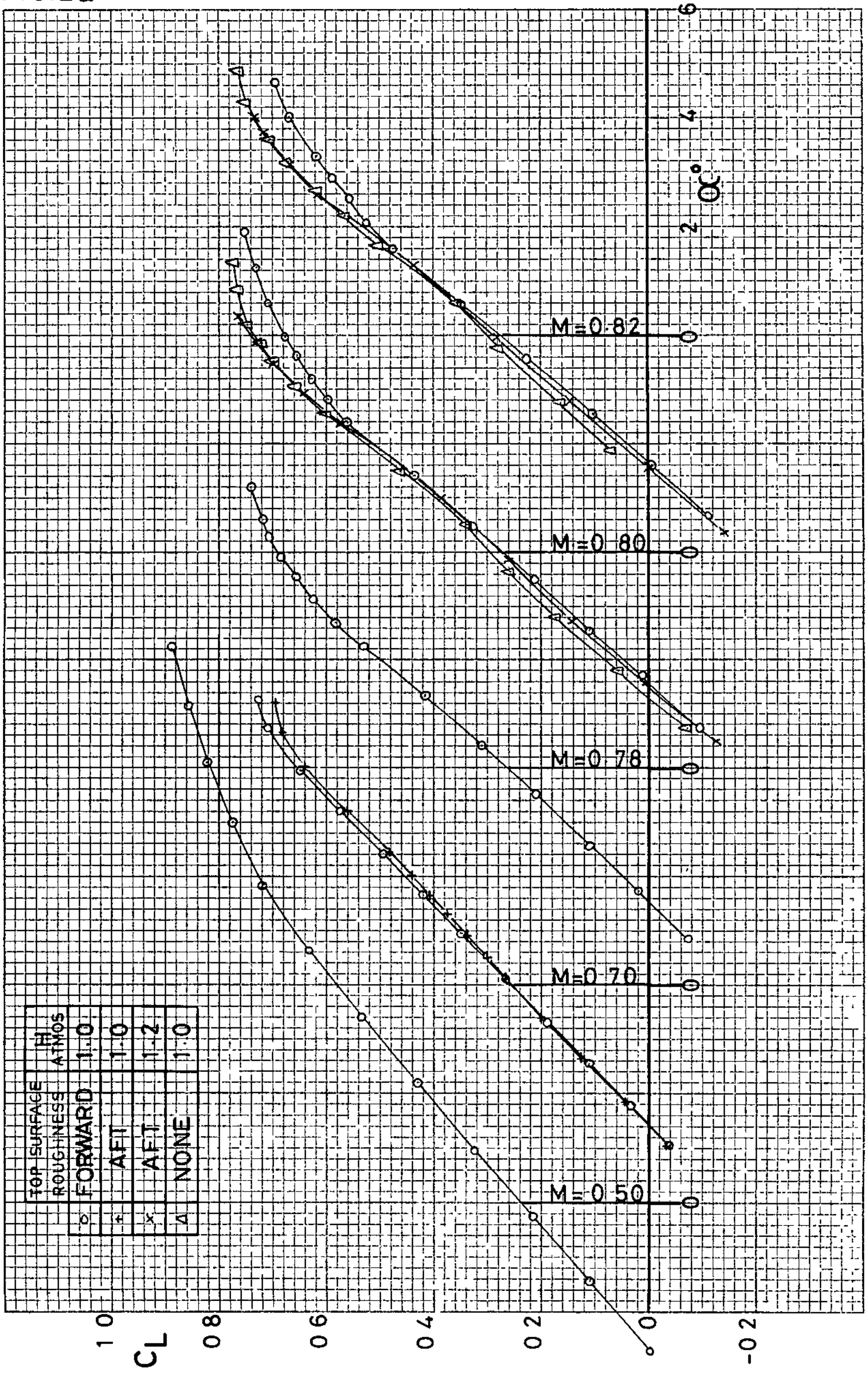


FIG. 2a. $C_L \sim \alpha$ WITH VARIOUS TRANSITION FIXES

FIG. 2b.

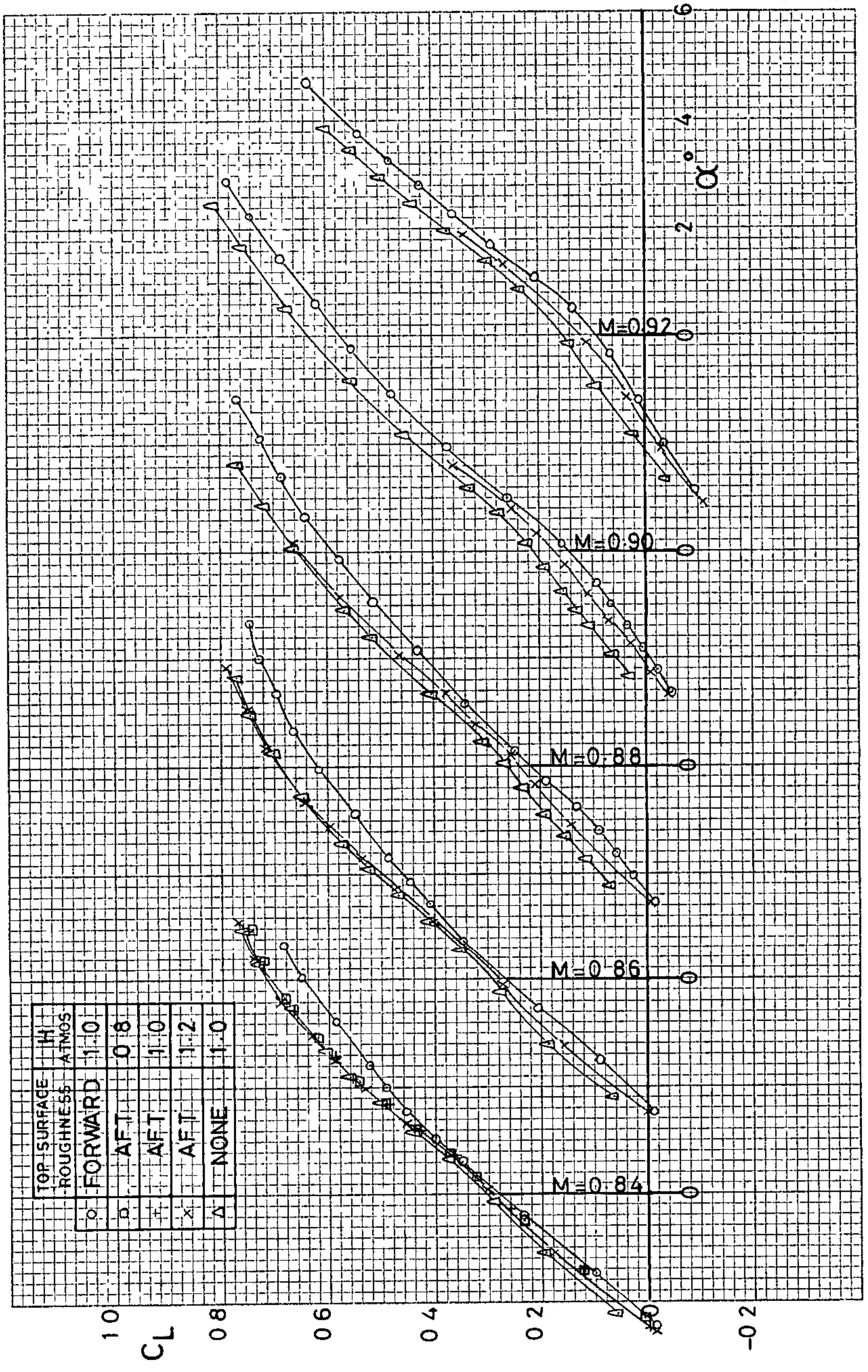


FIG. 2b. $C_L \sim \alpha$ WITH VARIOUS TRANSITION FIXES.

FIG. 3a

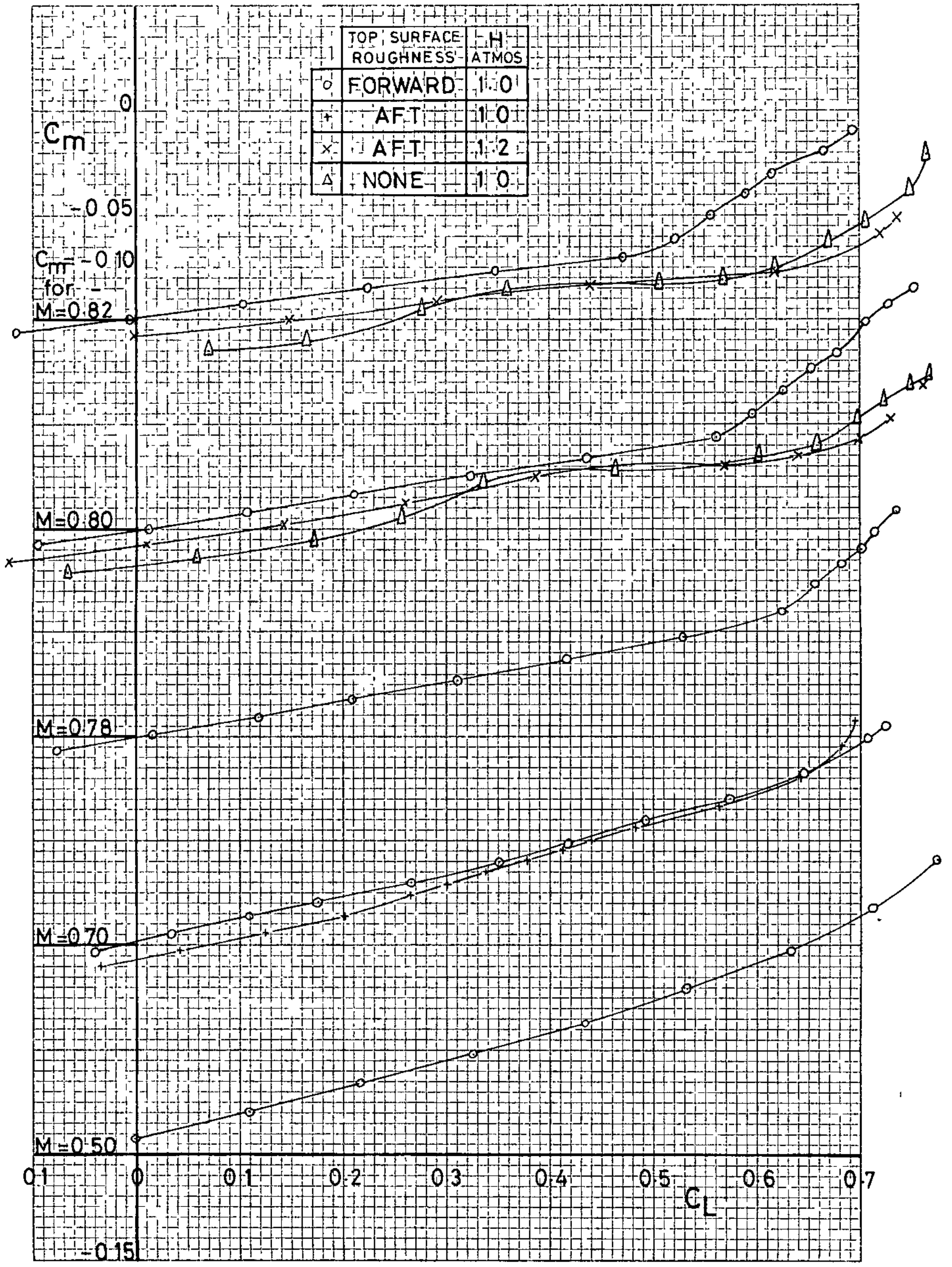


FIG 3a. C_m-C_L WITH VARIOUS TRANSITION FIXES.

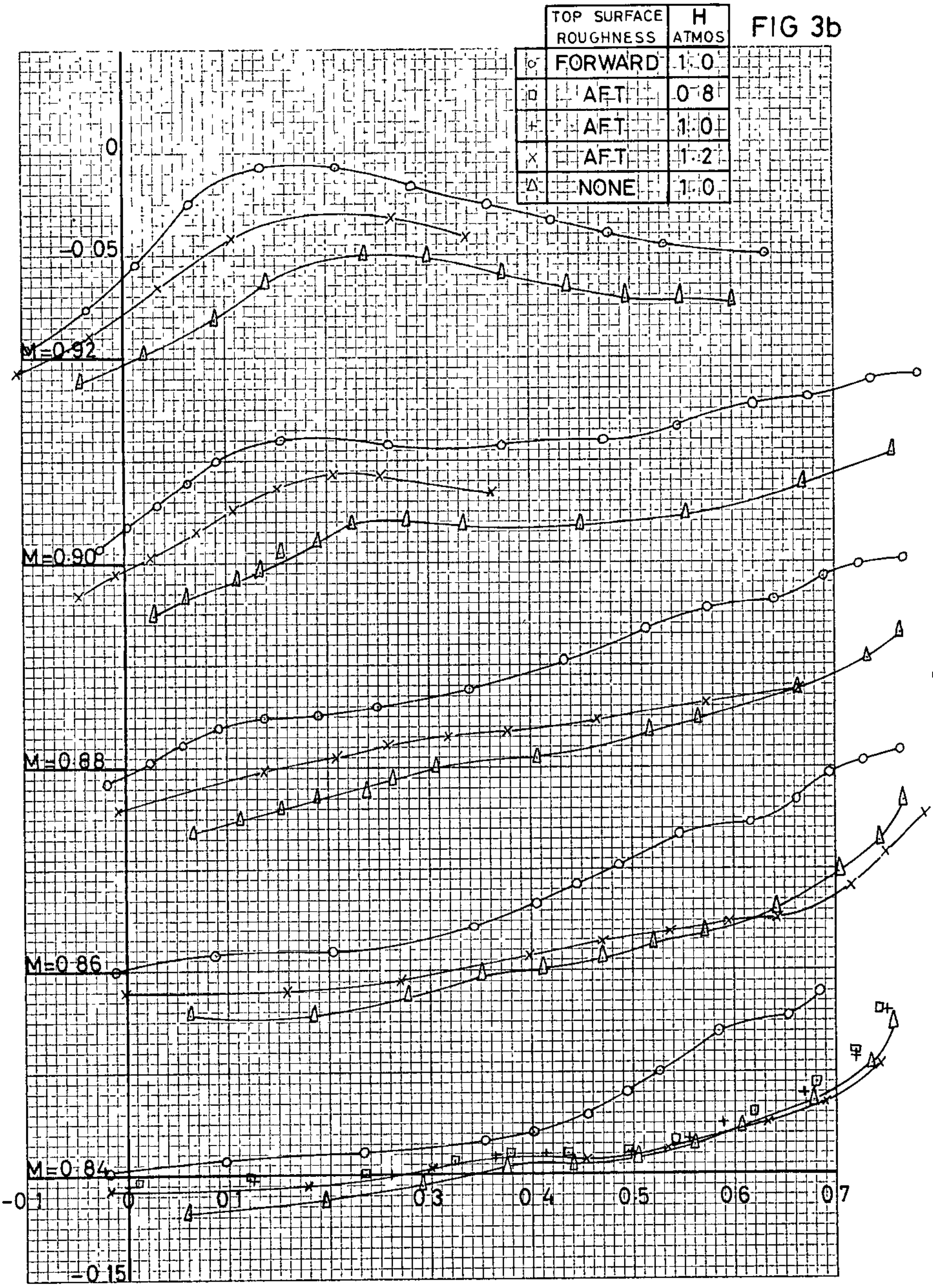


FIG. 3b. C_m ~ C_L WITH VARIOUS TRANSITION FIXES.

FIG 4a

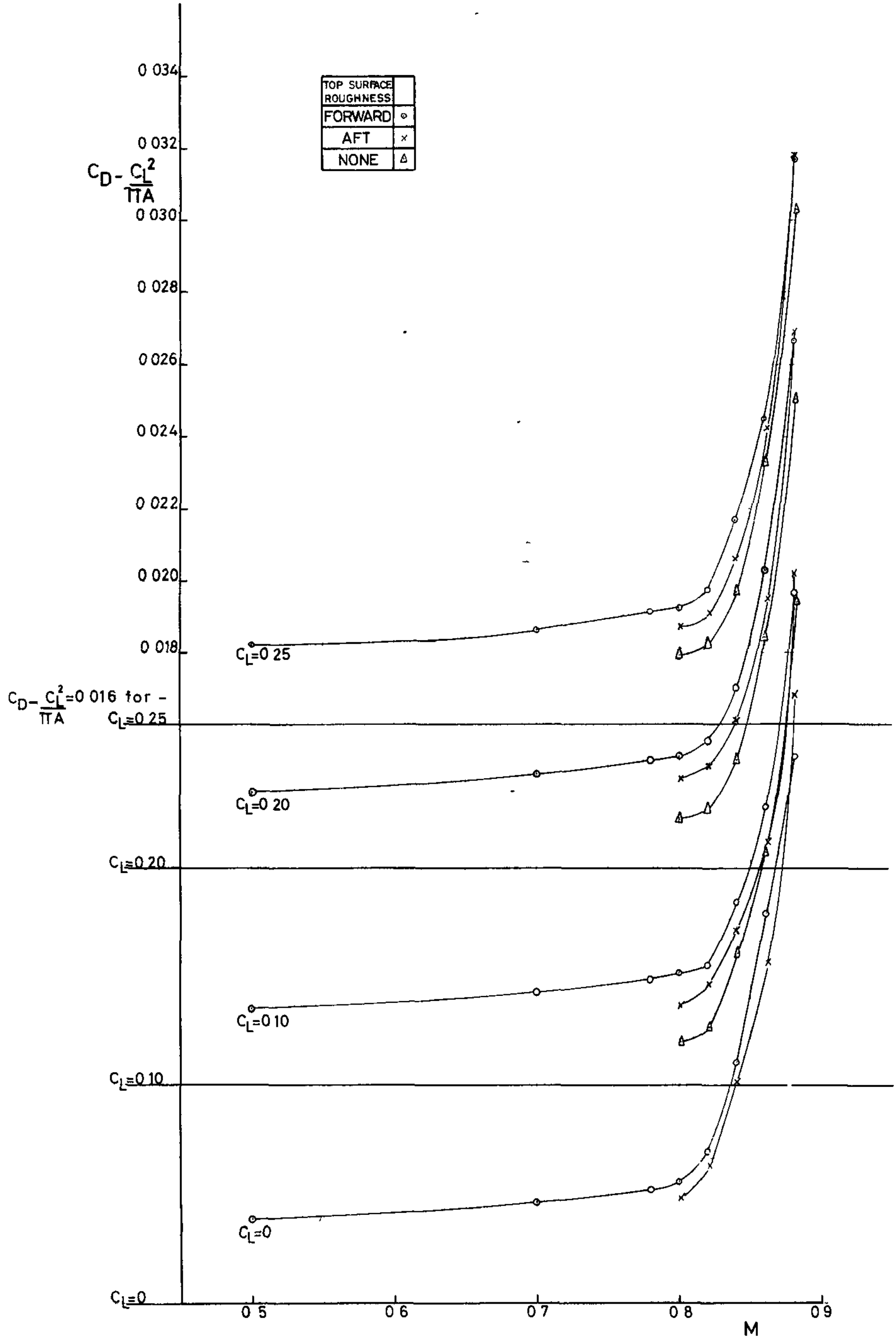


FIG 4a DRAG WITH VARIOUS TRANSITION FIXES

FIG 4b

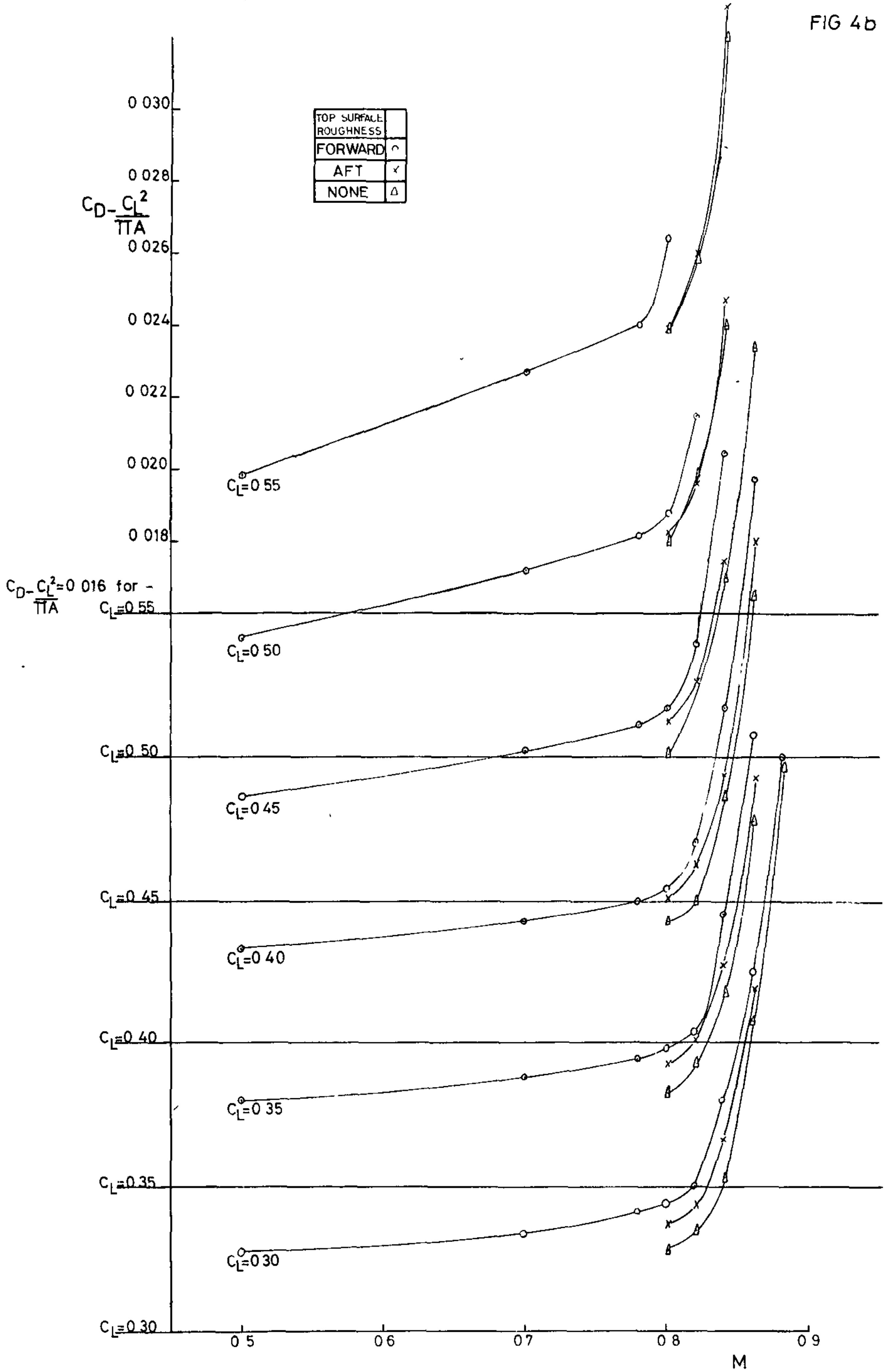


FIG 4b DRAG WITH VARIOUS TRANSITION FIXES

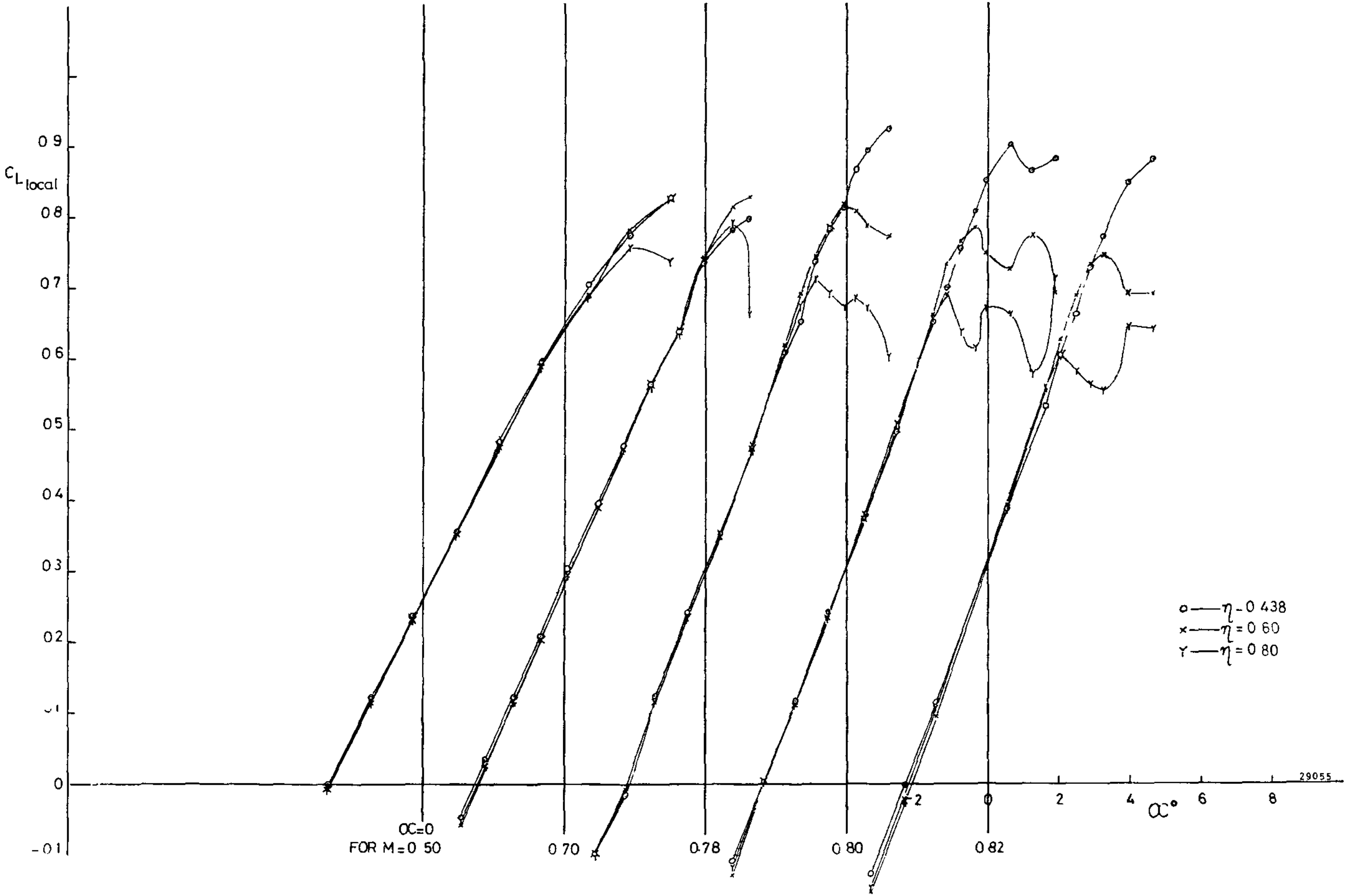


FIG 5a $C_{L_{local}}$ - α FORWARD UPPER SURFACE ROUGHNESS BAND

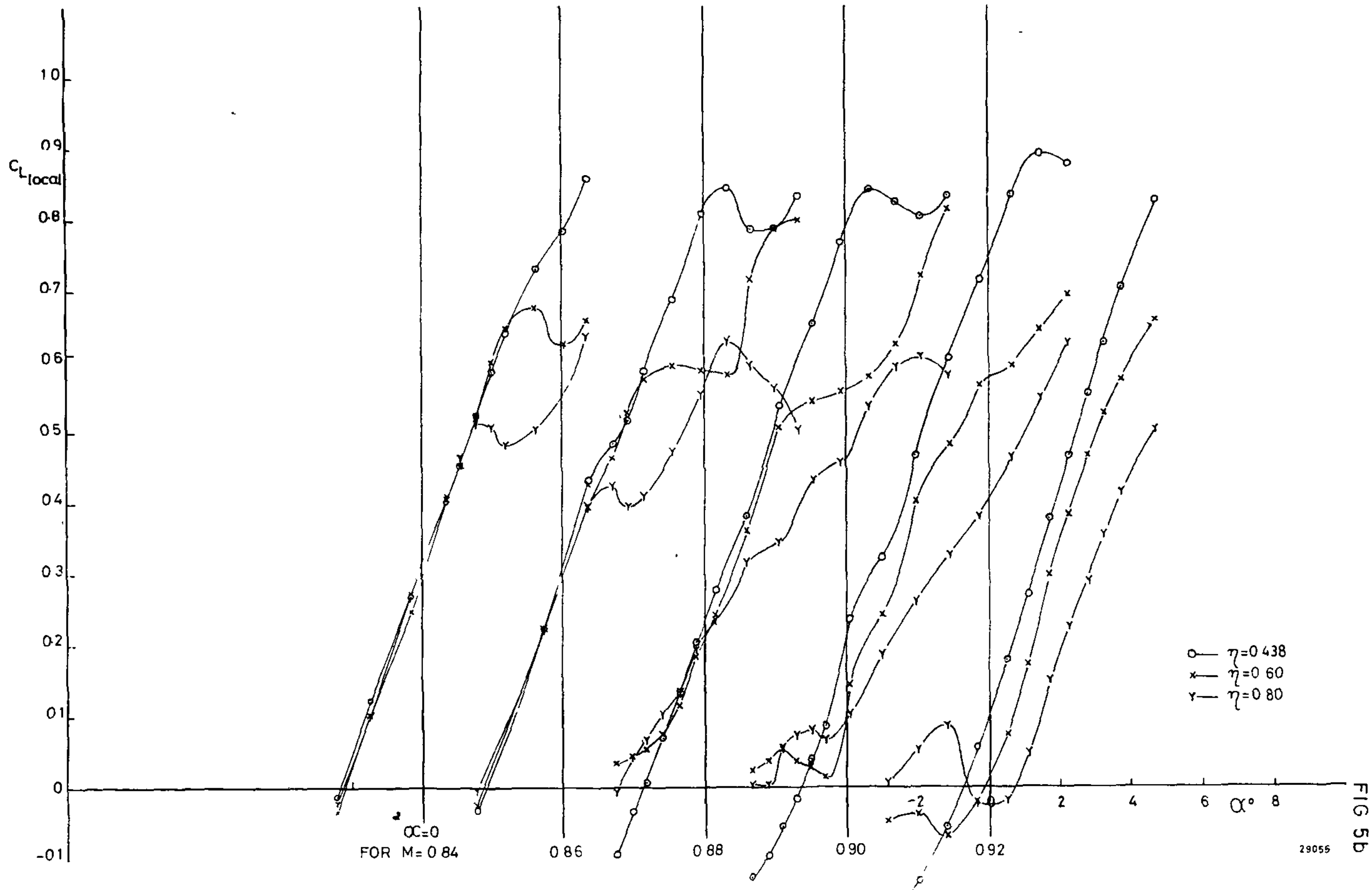


FIG 5b $C_{L_{local}}$ - α FORWARD UPPER SURFACE ROUGHNESS BAND

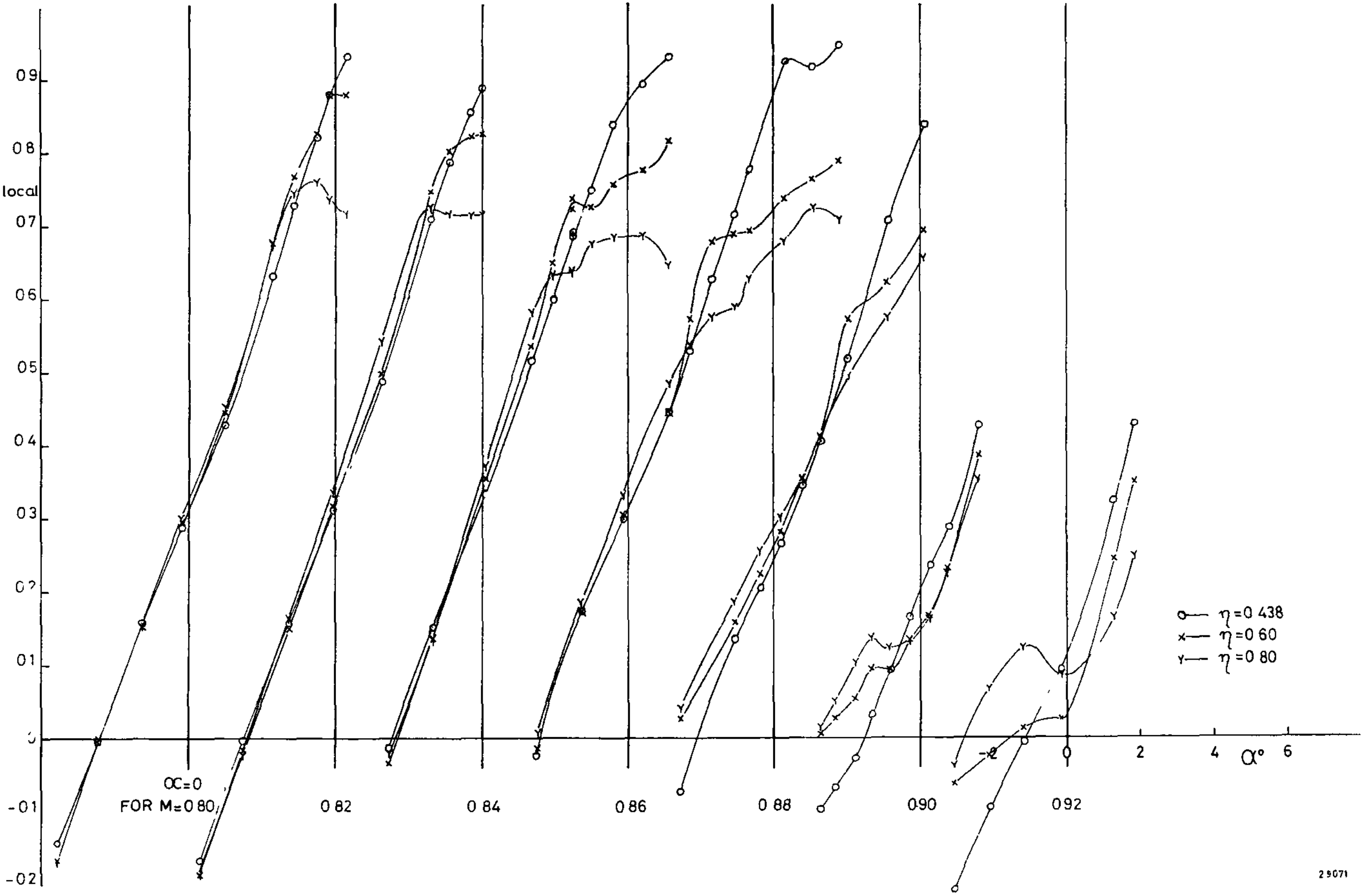


FIG 5c $C_{L_{local}}$ ~ α AFT UPPER SURFACE ROUGHNESS BAND

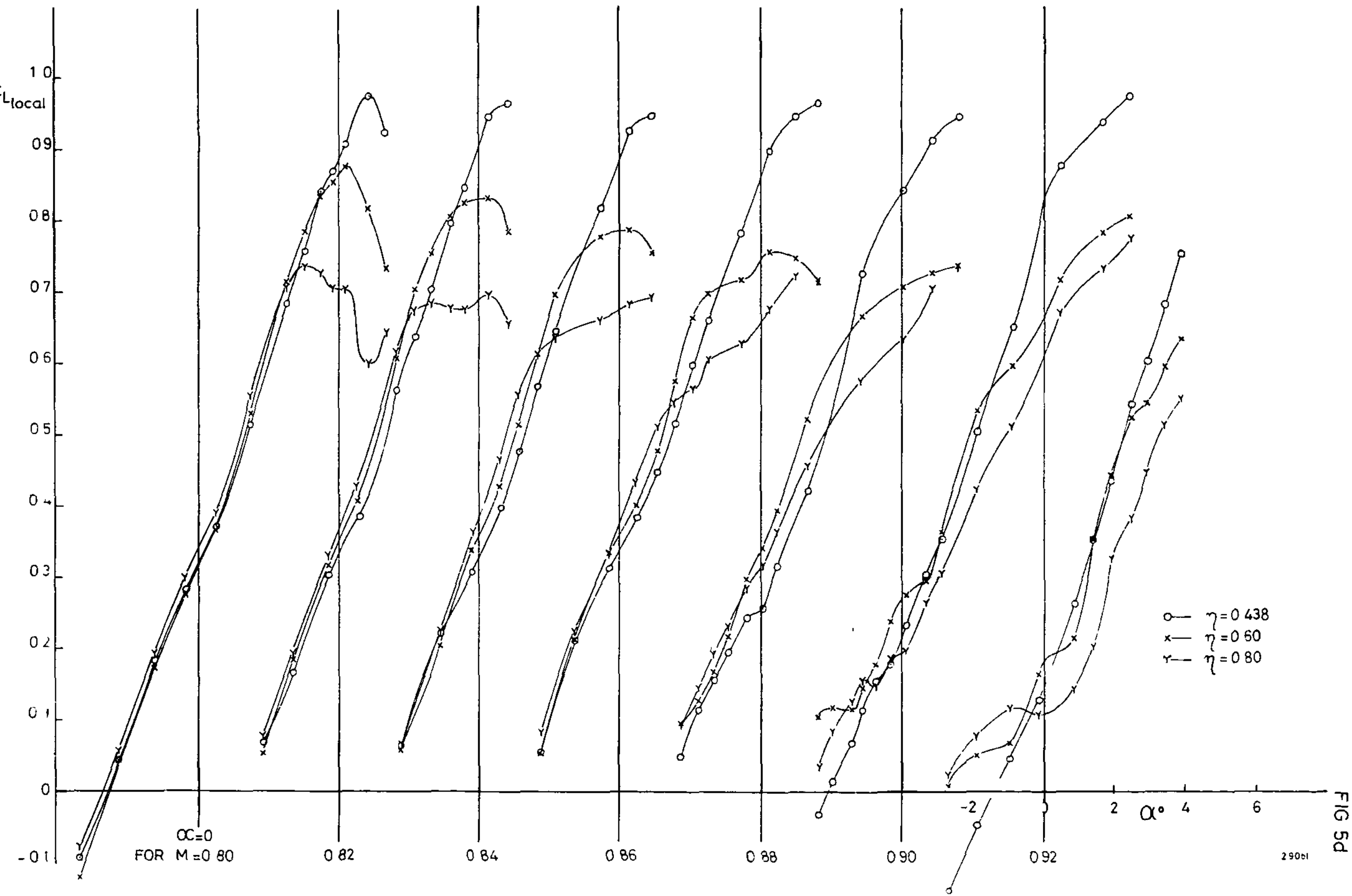


FIG 5d $C_{L_{local}} \sim \alpha$ NO UPPER SURFACE ROUGHNESS BAND

FIG 6

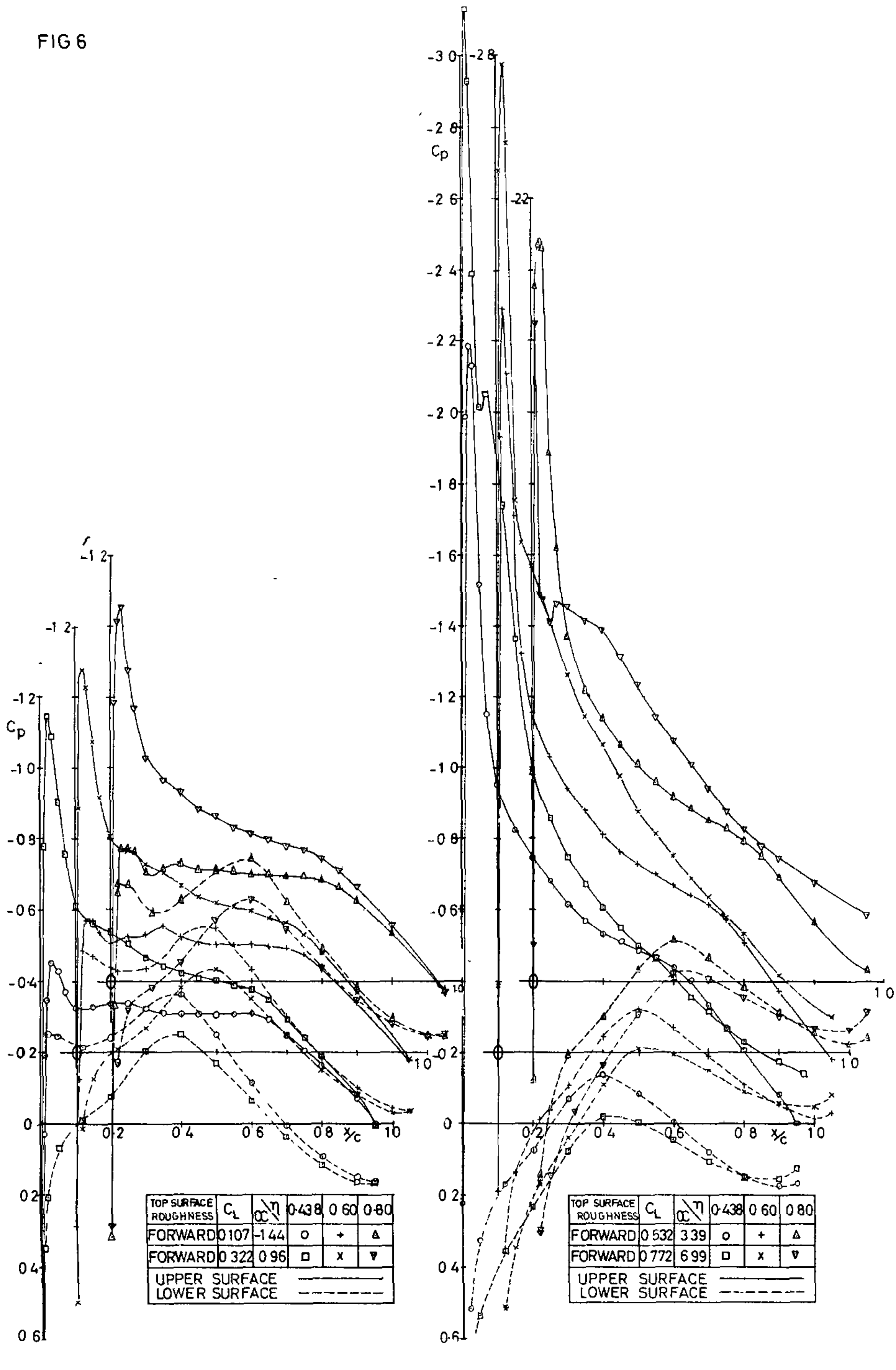
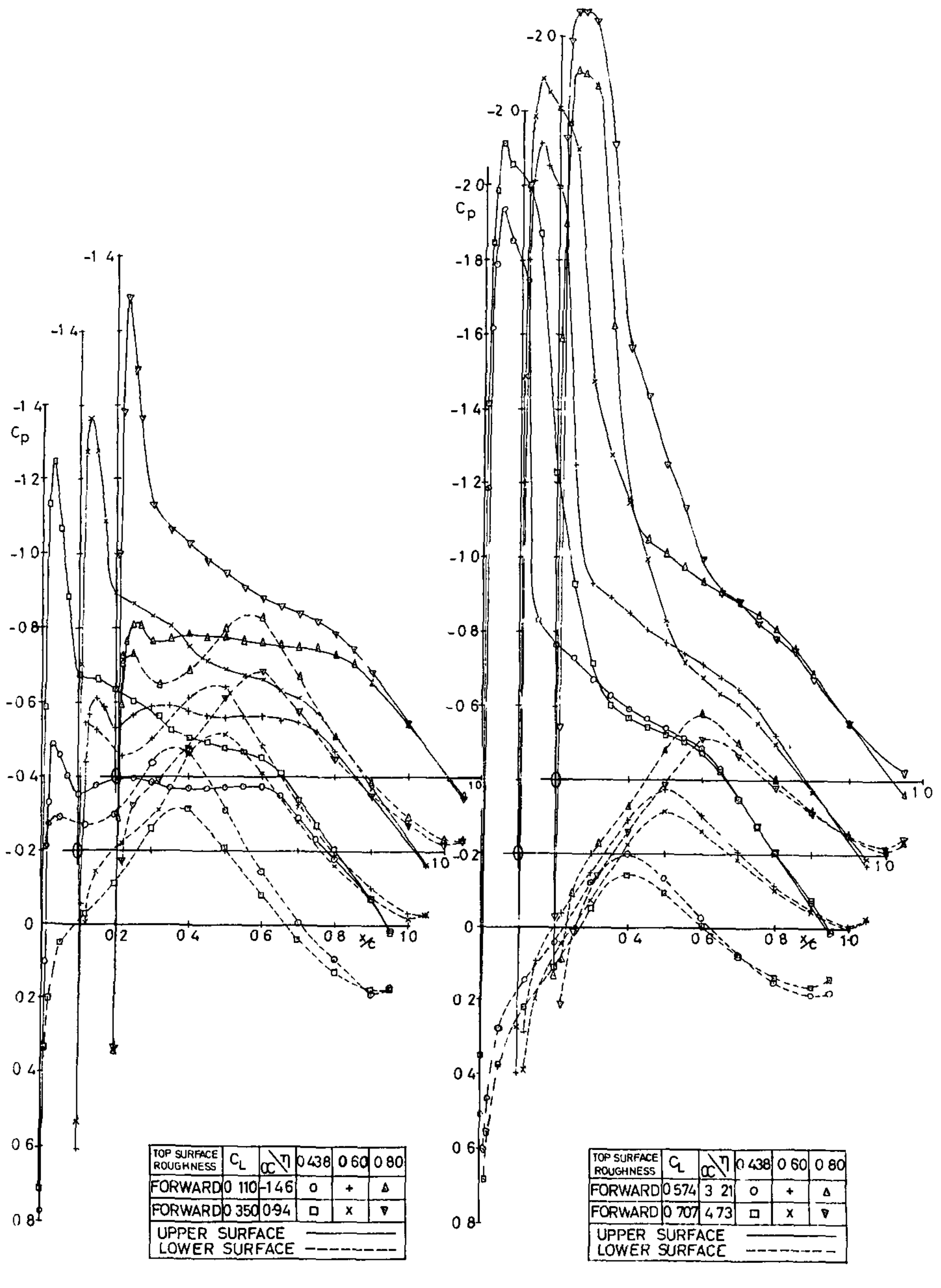


FIG 6 PRESSURE DISTRIBUTIONS AT M = 0·50



TOP SURFACE ROUGHNESS	C_L	$\alpha°$ \ η	0 438	0 60	0 80
FORWARD	0 110	-1 46	o	+	Δ
FORWARD	0 350	0 94	□	x	∇

UPPER SURFACE ————
LOWER SURFACE - - - - - -

TOP SURFACE ROUGHNESS	C_L	$\alpha°$ \ η	0 438	0 60	0 80
FORWARD	0 574	3 21	o	+	Δ
FORWARD	0 707	4 73	□	x	∇

UPPER SURFACE ————
LOWER SURFACE - - - - - -

FIG 7a PRESSURE DISTRIBUTIONS AT M = 0 70

FIG 7b

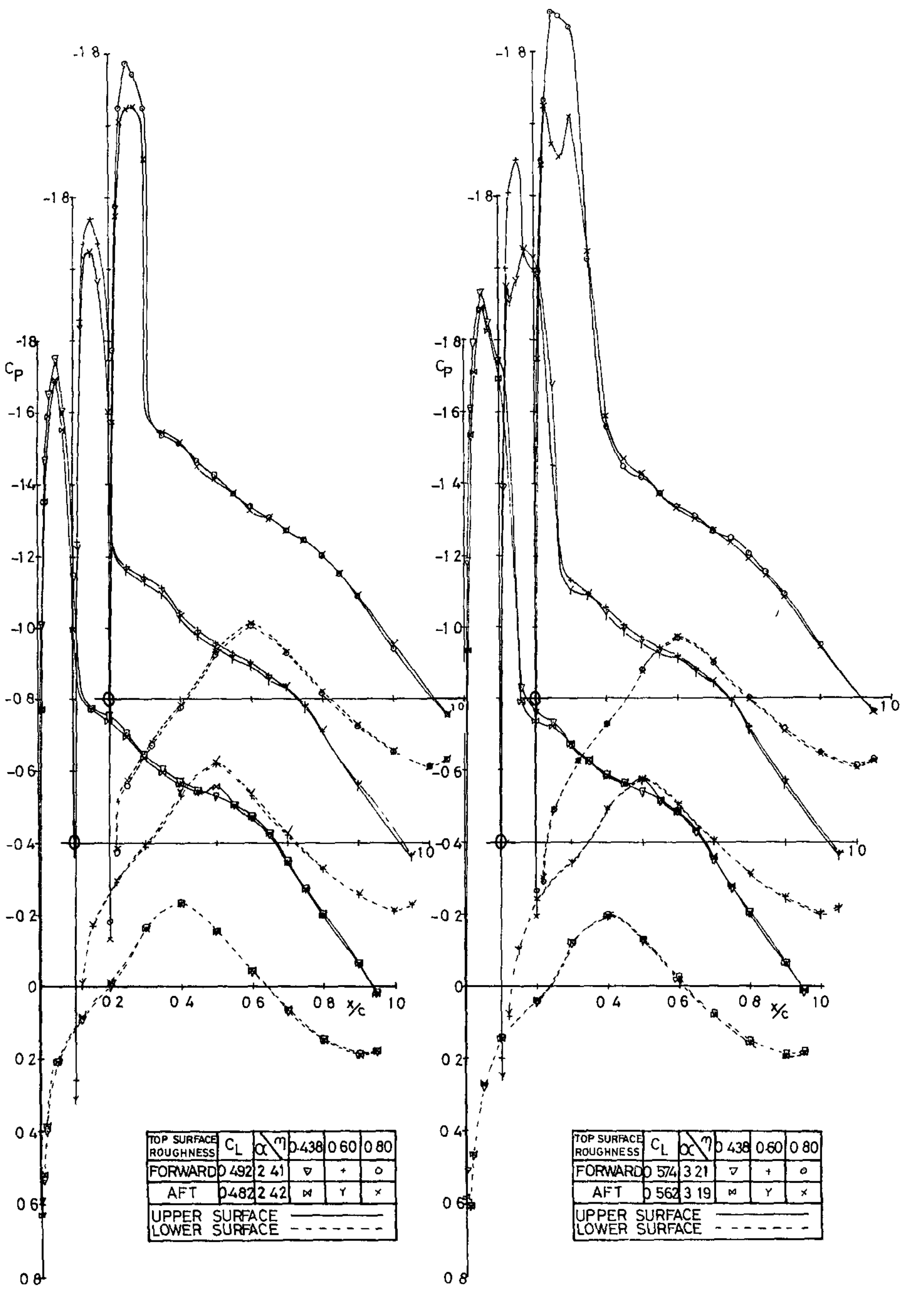


FIG 7b PRESSURE DISTRIBUTIONS AT M=0.70

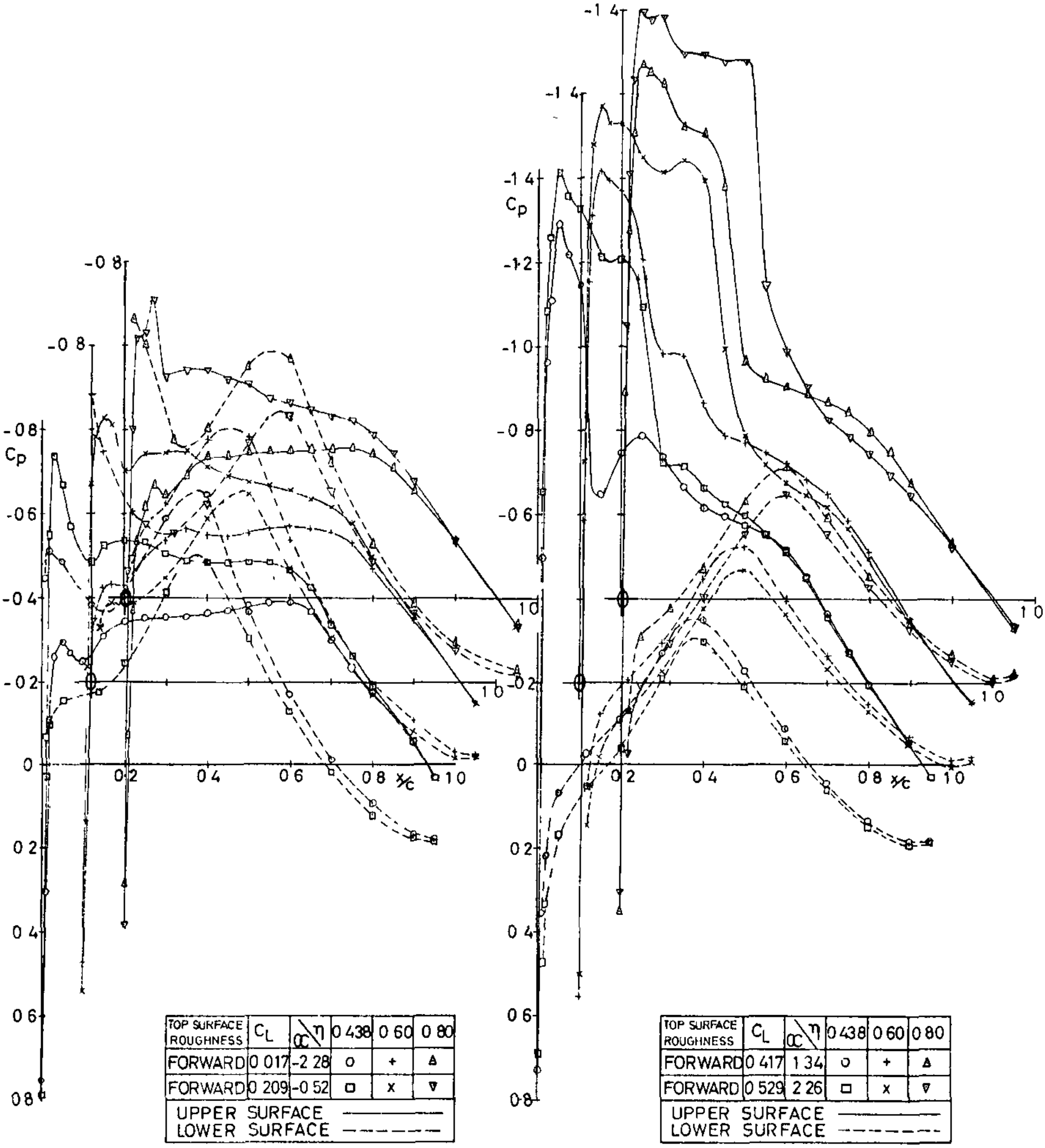


FIG 8a PRESSURE DISTRIBUTIONS AT M=0 78

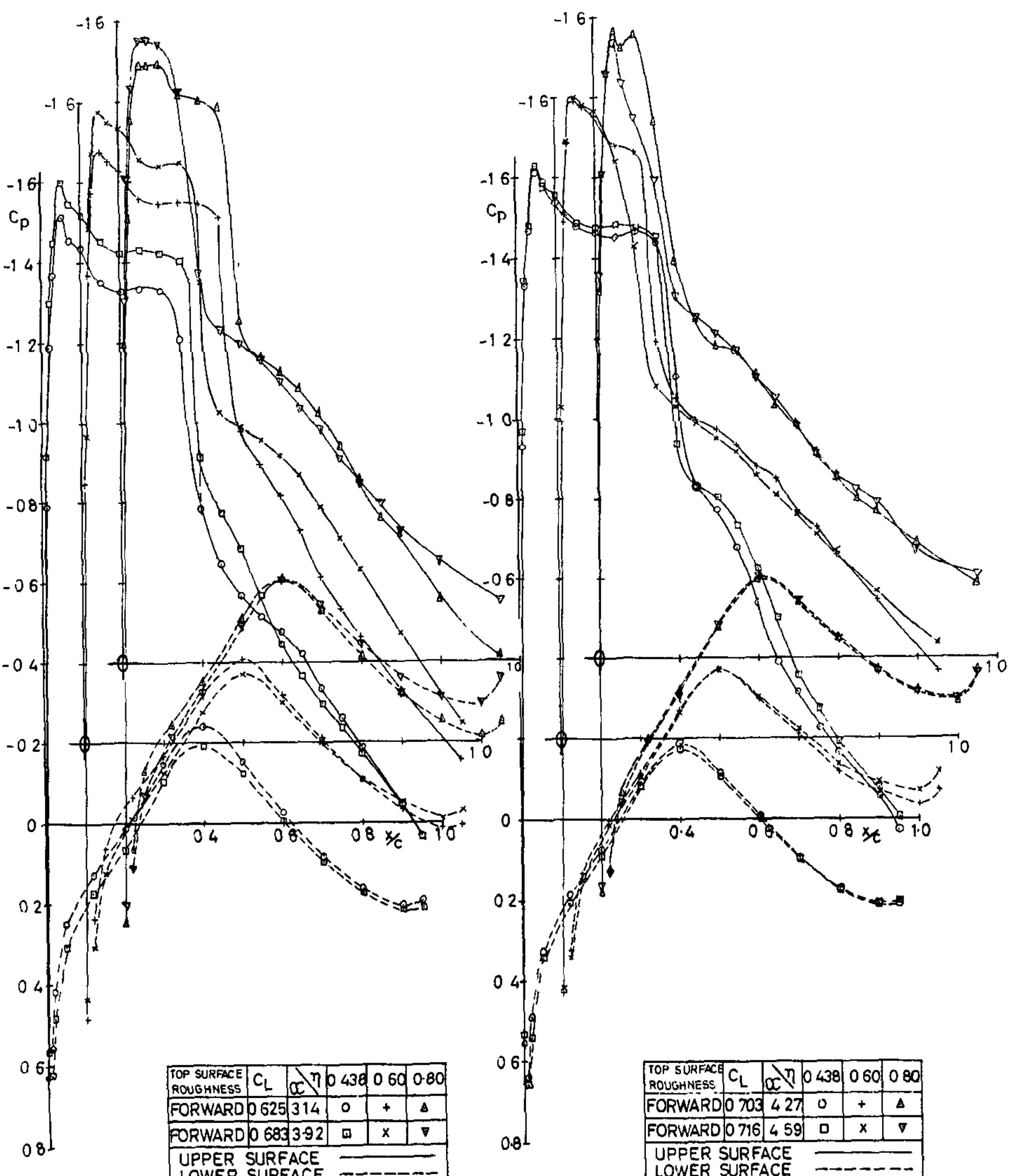
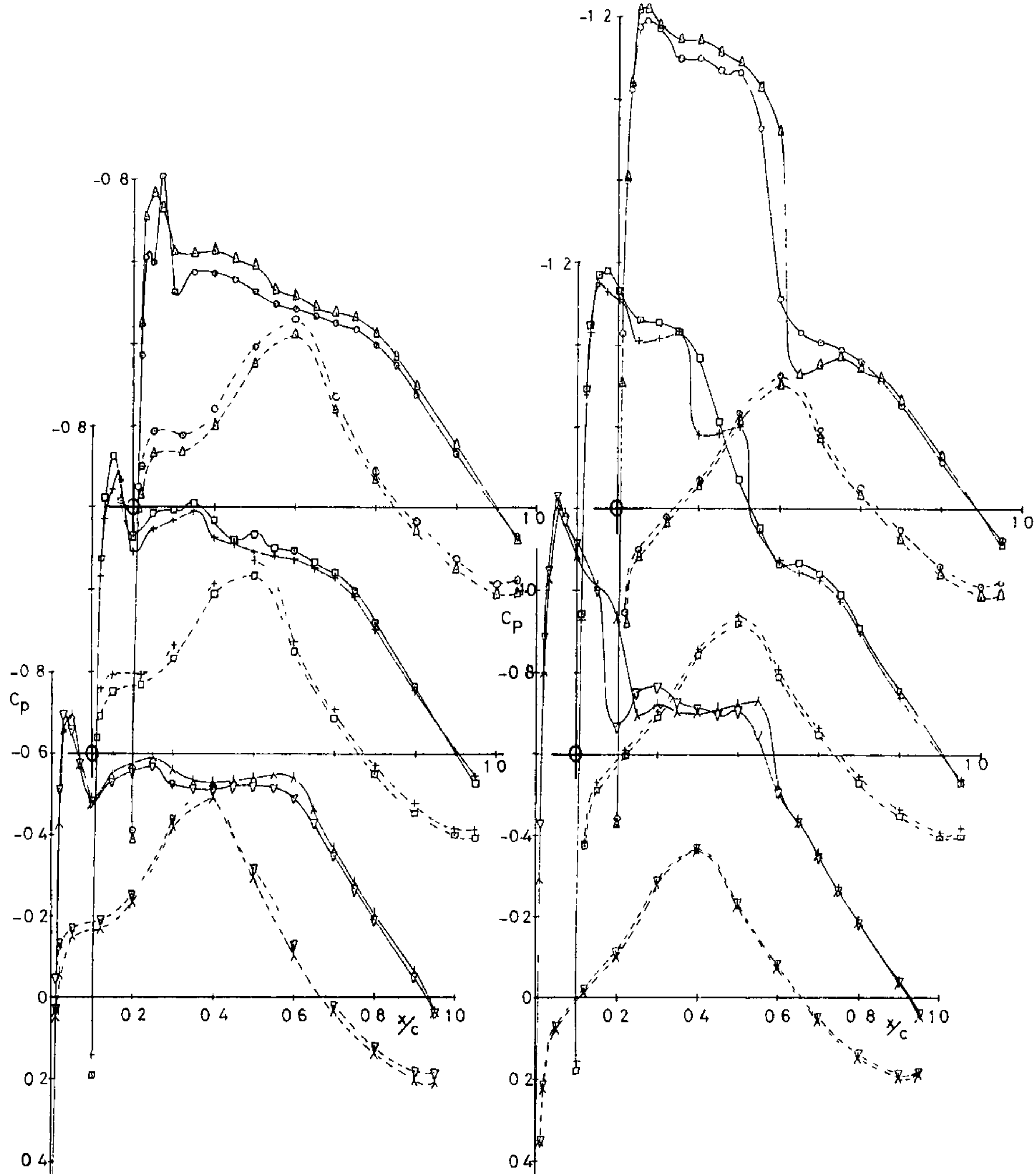


FIG 8b PRESSURE DISTRIBUTIONS AT M = 0 78



TOP SURFACE ROUGHNESS	C_L	$\alpha \backslash \eta$	0 438	0 60	0 80
FORWARD	0 211	-0 52	∇	+	o
NONE	0 253	-0 36	⅄	□	Δ
UPPER SURFACE ———					
LOWER SURFACE - - - -					

TOP SURFACE ROUGHNESS	C_L	$\alpha \backslash \eta$	0 438	0 60	0 80
FORWARD	0 437	1 40	∇	+	o
NONE	0 461	1 52	⅄	□	Δ
UPPER SURFACE ———					
LOWER SURFACE - - - -					

FIG 9a PRESSURE DISTRIBUTIONS AT M=0 80

FIG 9b

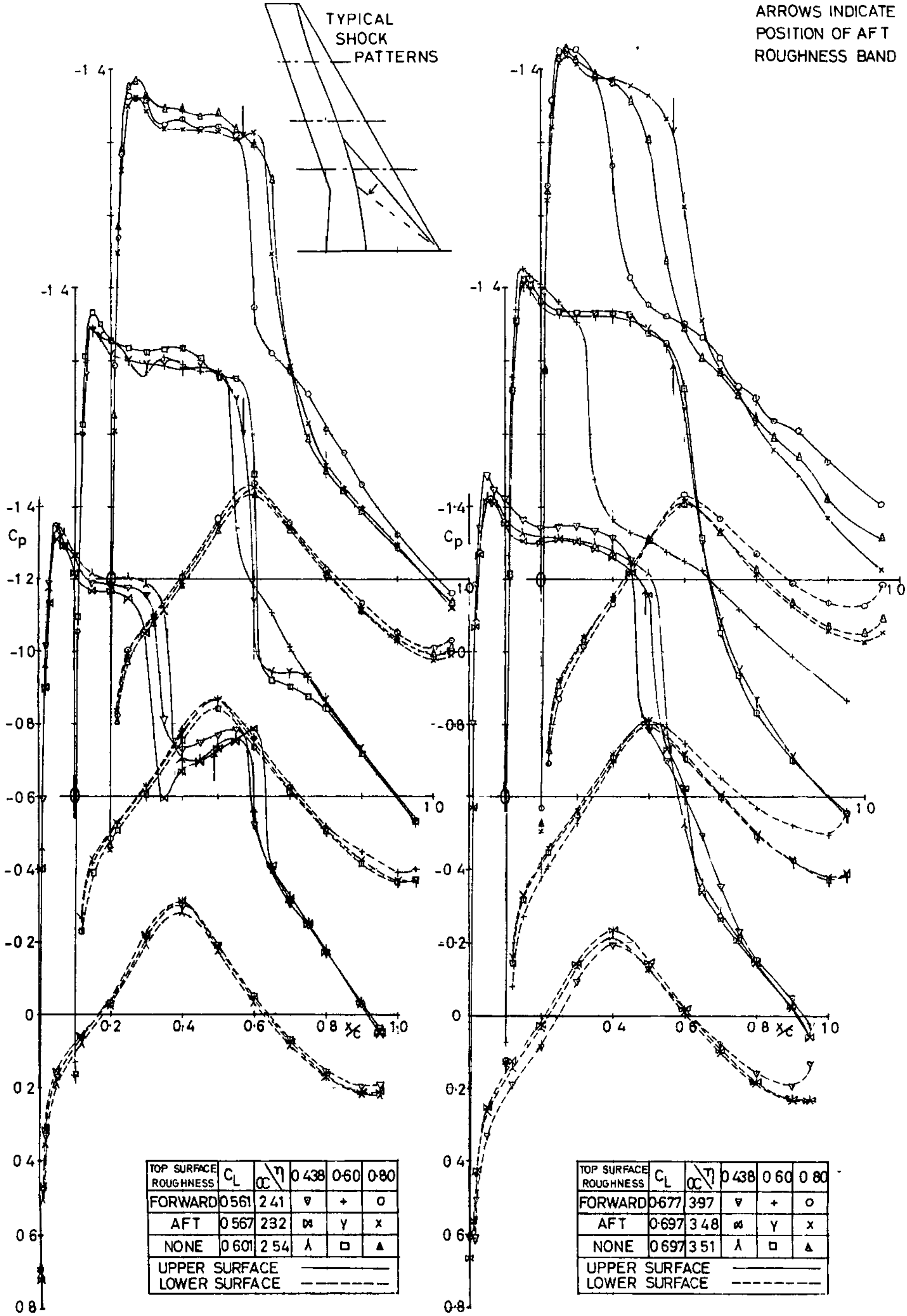


FIG 9b PRESSURE DISTRIBUTIONS AT M=0.80

FIG 10a

ARROWS INDICATE
POSITION OF AFT
ROUGHNESS BAND

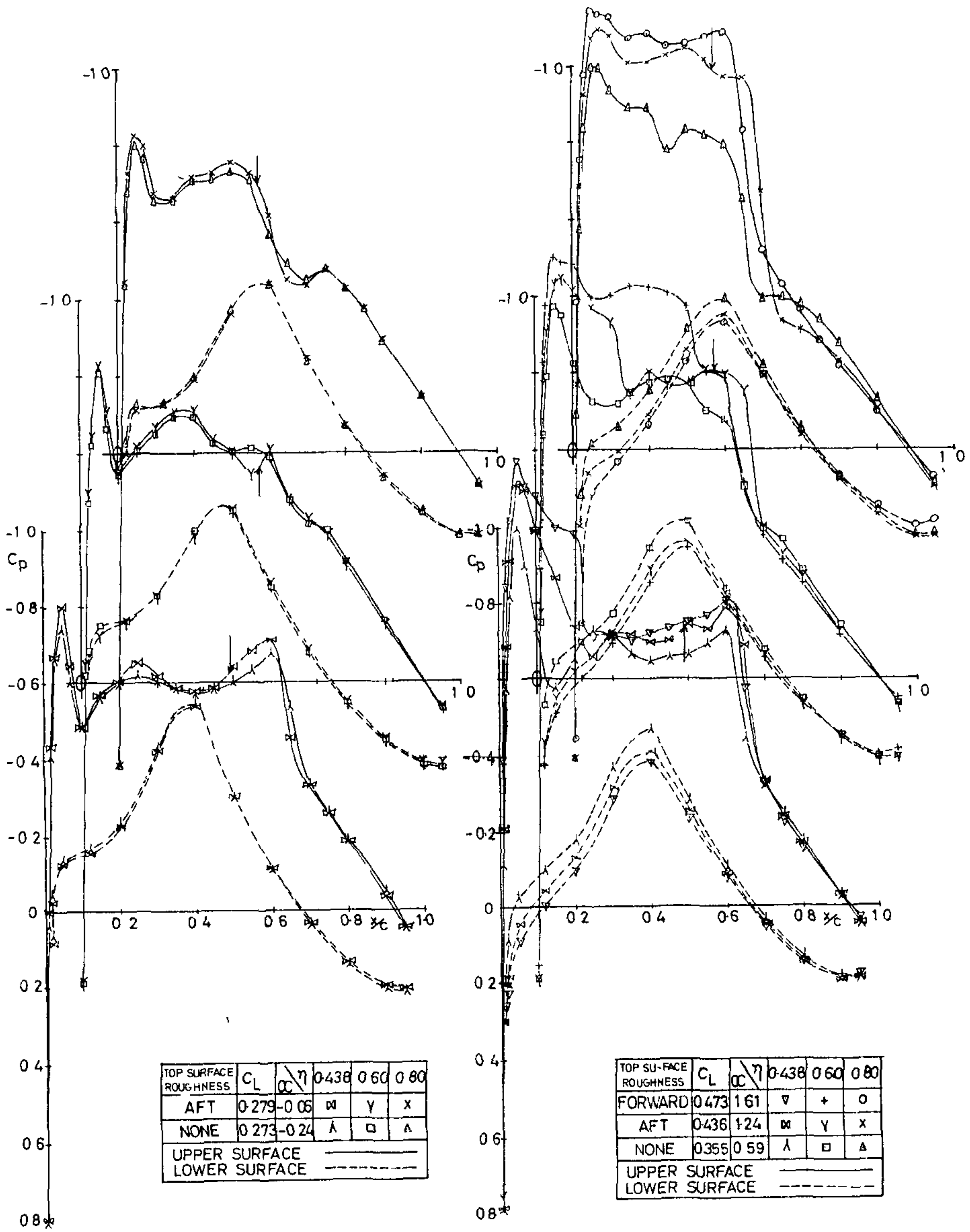
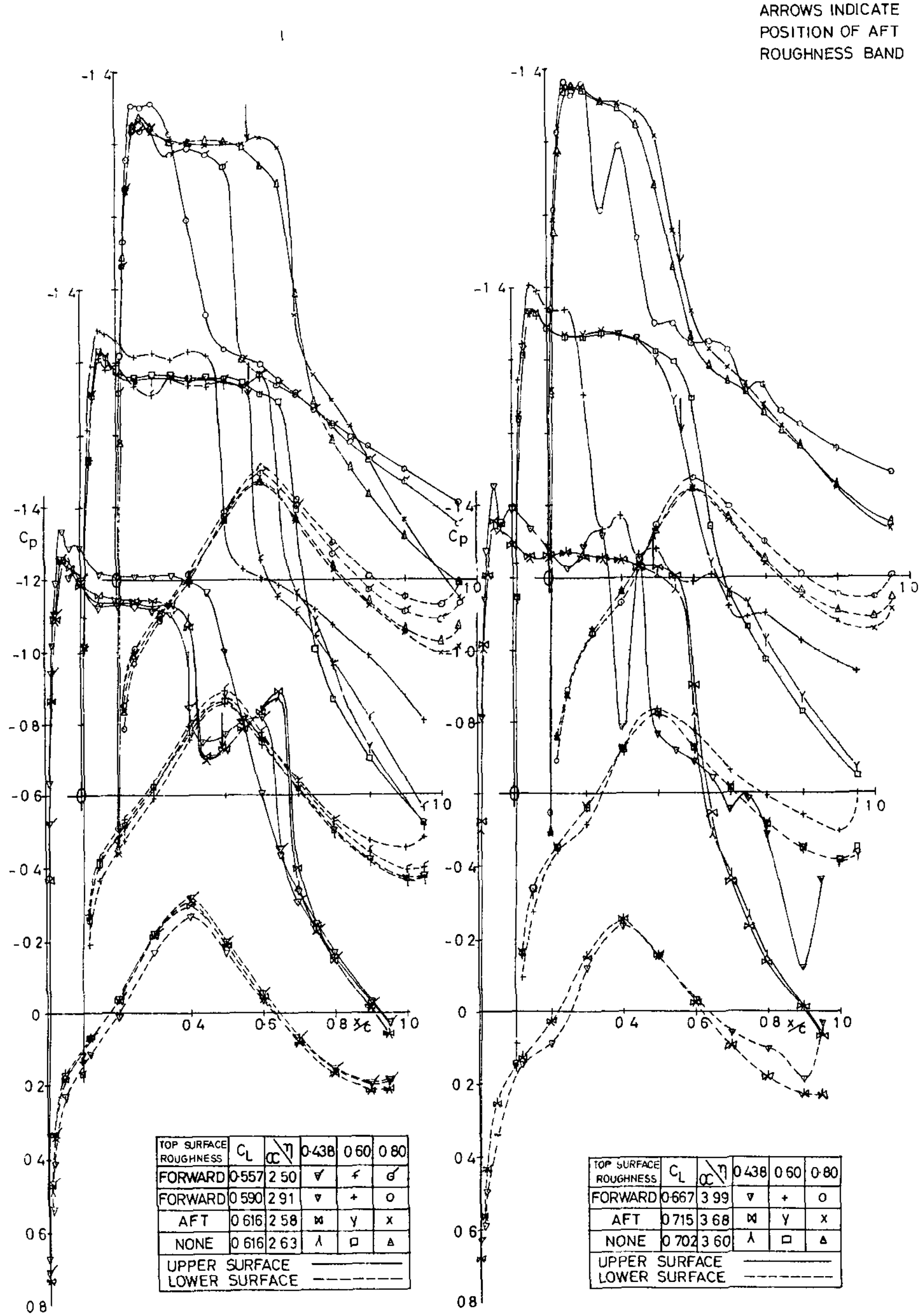


FIG 10a PRESSURE DISTRIBUTIONS AT M = 0·82

FIG 10b



TOP SURFACE ROUGHNESS	C_L	α \ η	0·438	0 60	0 80
FORWARD	0·557	2 50	∀	+	O
FORWARD	0 590	2 91	∇	+	O
AFT	0 616	2 58	⋈	Y	X
NONE	0 616	2 63	λ	□	Δ

TOP SURFACE ROUGHNESS	C_L	α \ η	0 438	0 60	0·80
FORWARD	0·667	3 99	∇	+	O
AFT	0 715	3 68	⋈	Y	X
NONE	0 702	3 60	λ	□	Δ

UPPER SURFACE ————
LOWER SURFACE - - - - -

FIG 10b PRESSURE DISTRIBUTIONS AT M = 0 82

FIG 11a

ARROWS INDICATE
POSITION OF AFT
ROUGHNESS BAND

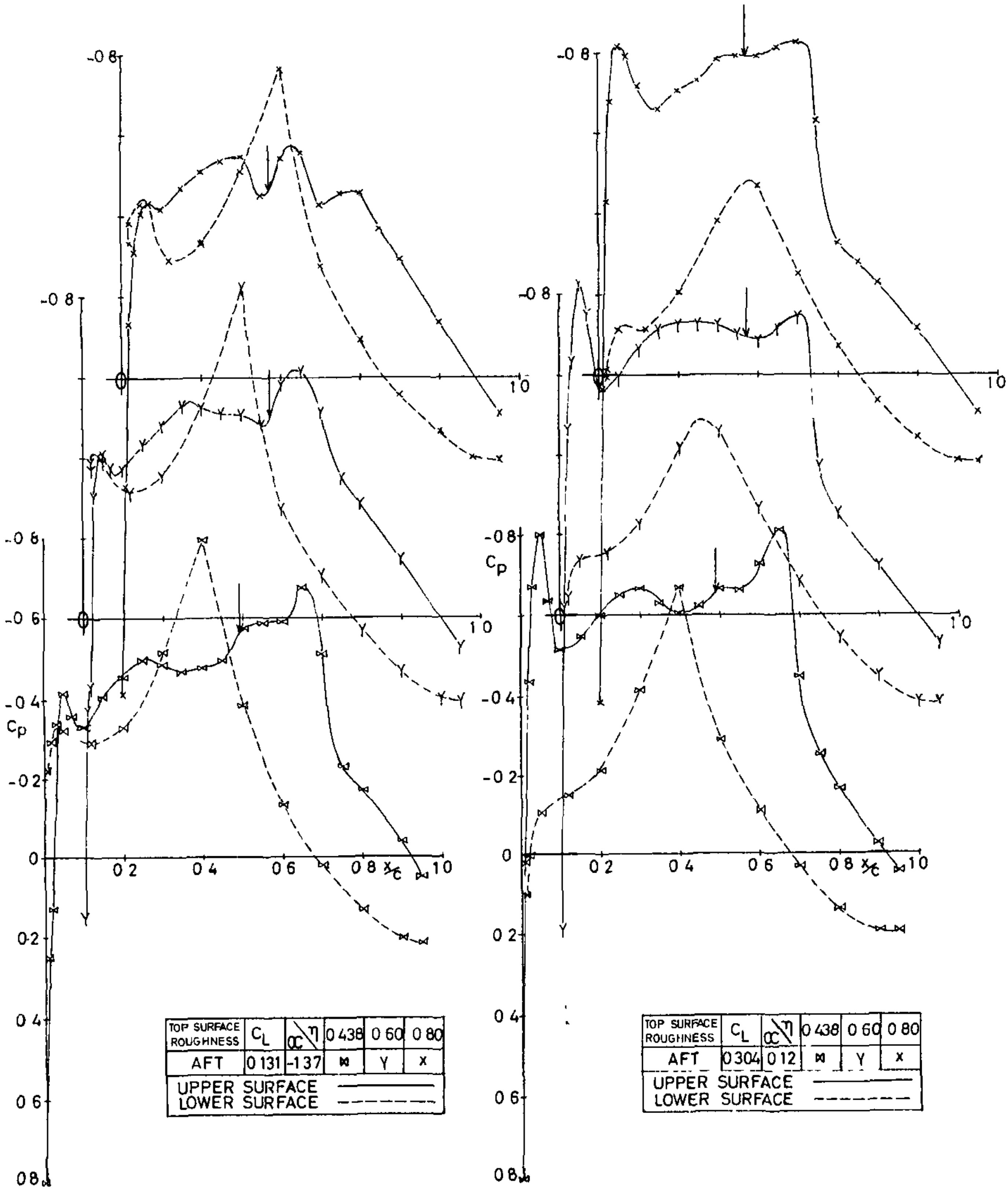
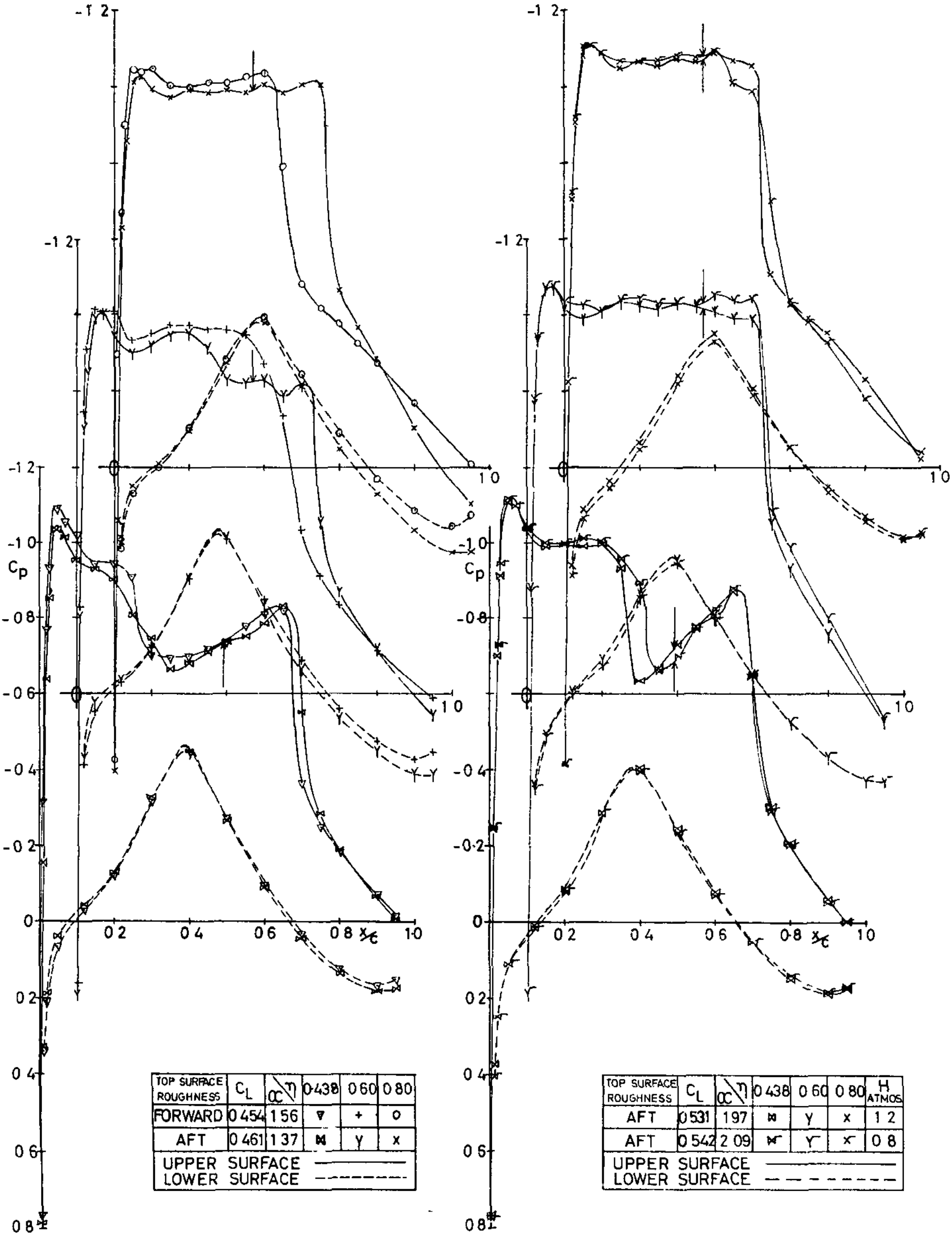


FIG 11a PRESSURE DISTRIBUTIONS AT M = 0 84

FIG 11b

ARROWS INDICATE
POSITION OF AFT
ROUGHNESS BAND



TOP SURFACE ROUGHNESS	C_L	$\alpha \setminus \eta$	0 438	0 60	0 80
FORWARD	0 454	1 56	▽	+	o
AFT	0 461	1 37	⋈	Y	x

UPPER SURFACE ————
LOWER SURFACE - - - - -

TOP SURFACE ROUGHNESS	C_L	$\alpha \setminus \eta$	0 438	0 60	0 80	H ATMOS
AFT	0 531	1 97	⋈	Y	x	1 2
AFT	0 542	2 09	⋈¯	Y¯	x¯	0 8

UPPER SURFACE ————
LOWER SURFACE - - - - -

FIG 11b PRESSURE DISTRIBUTION AT M = 0 84

ARROWS INICATE
POSITION OF AFT
ROUGHNESS BAND

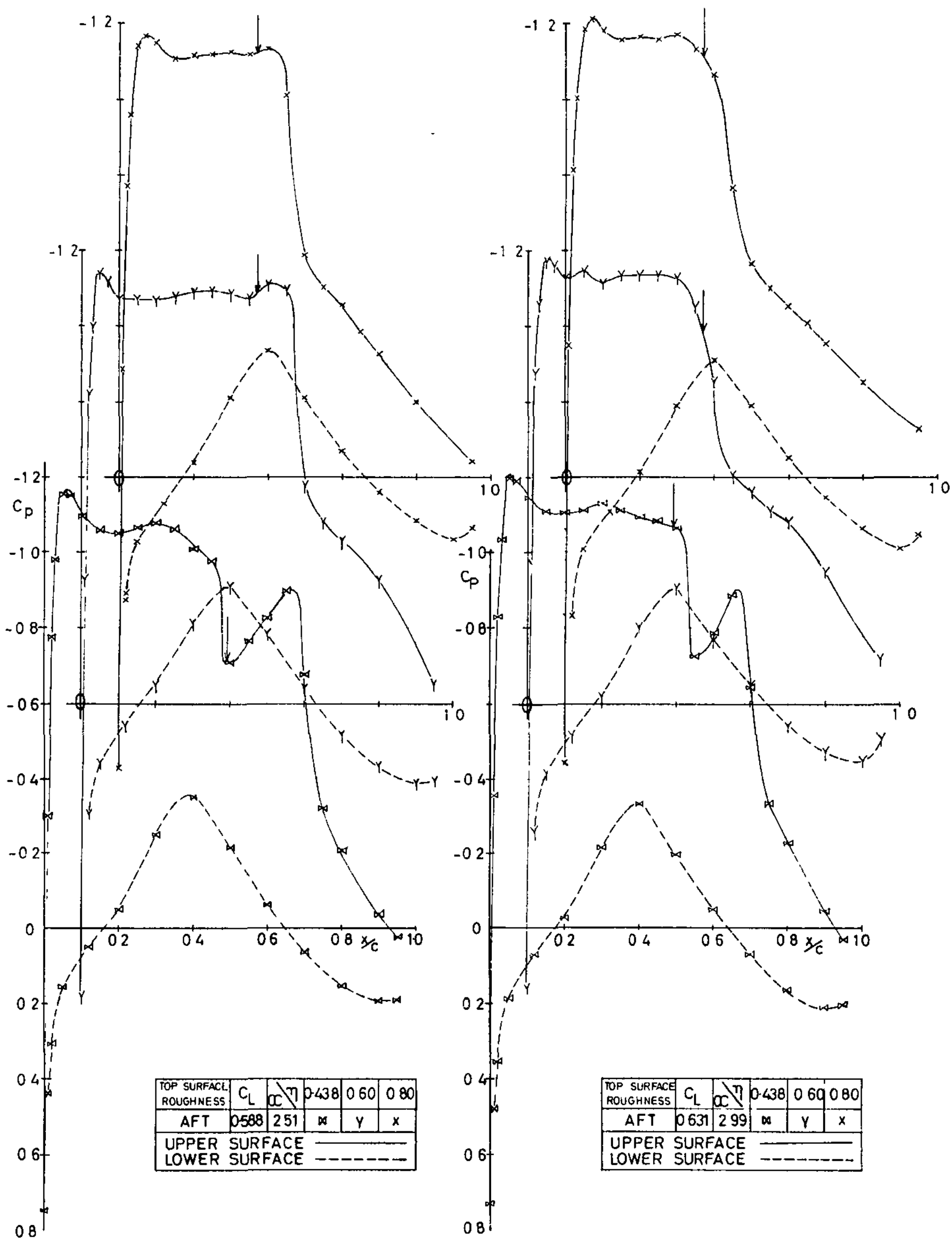


FIG 11c PRESSURE DISTRIBUTION AT M = 0·84

ARROWS INDICATE
POSITION OF AFT
ROUGHNESS BAND

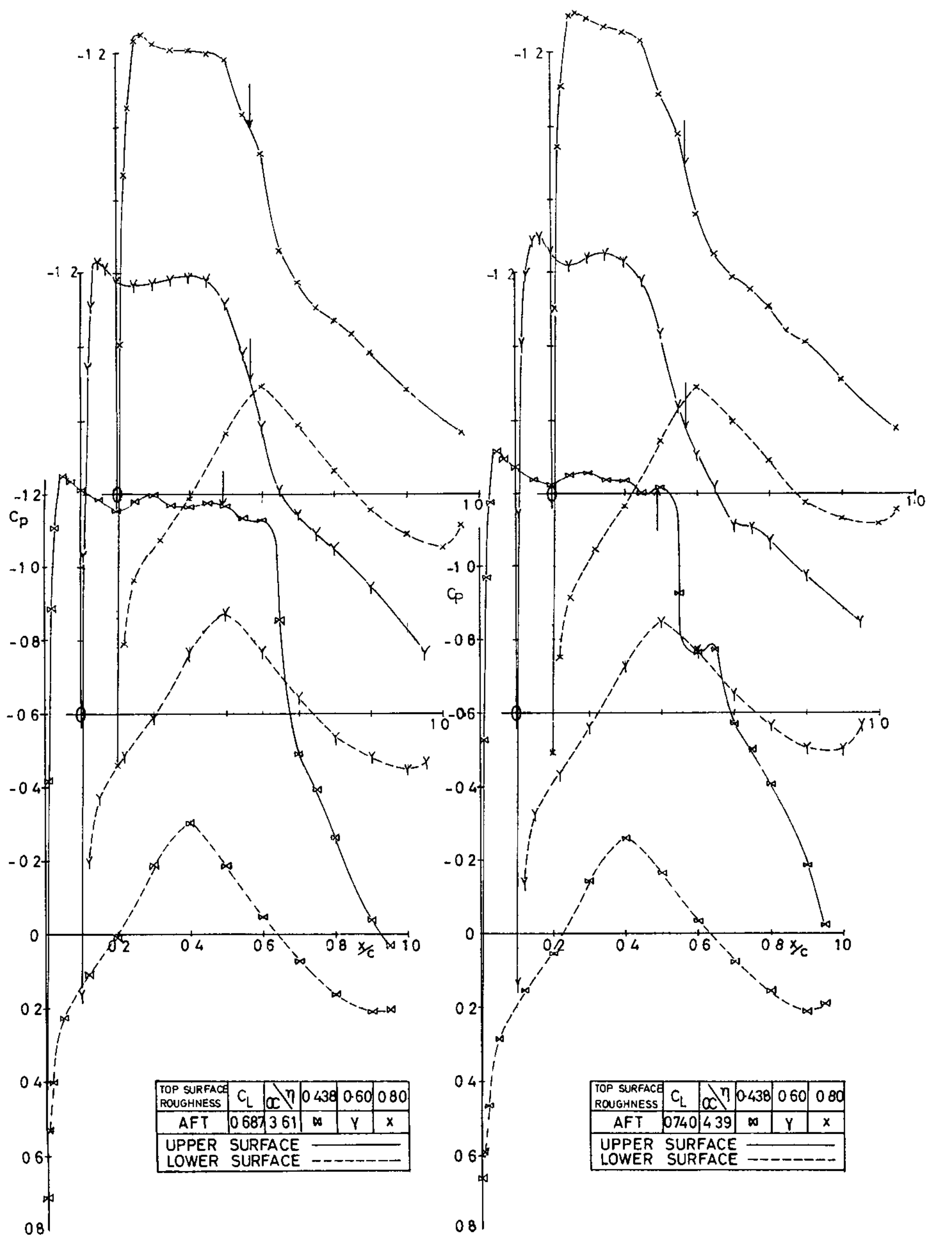


FIG 11d PRESSURE DISTRIBUTIONS M = 0 84

FIG 12a

ARROWS INDICATE
POSITION OF AFT
ROUGHNESS BAND

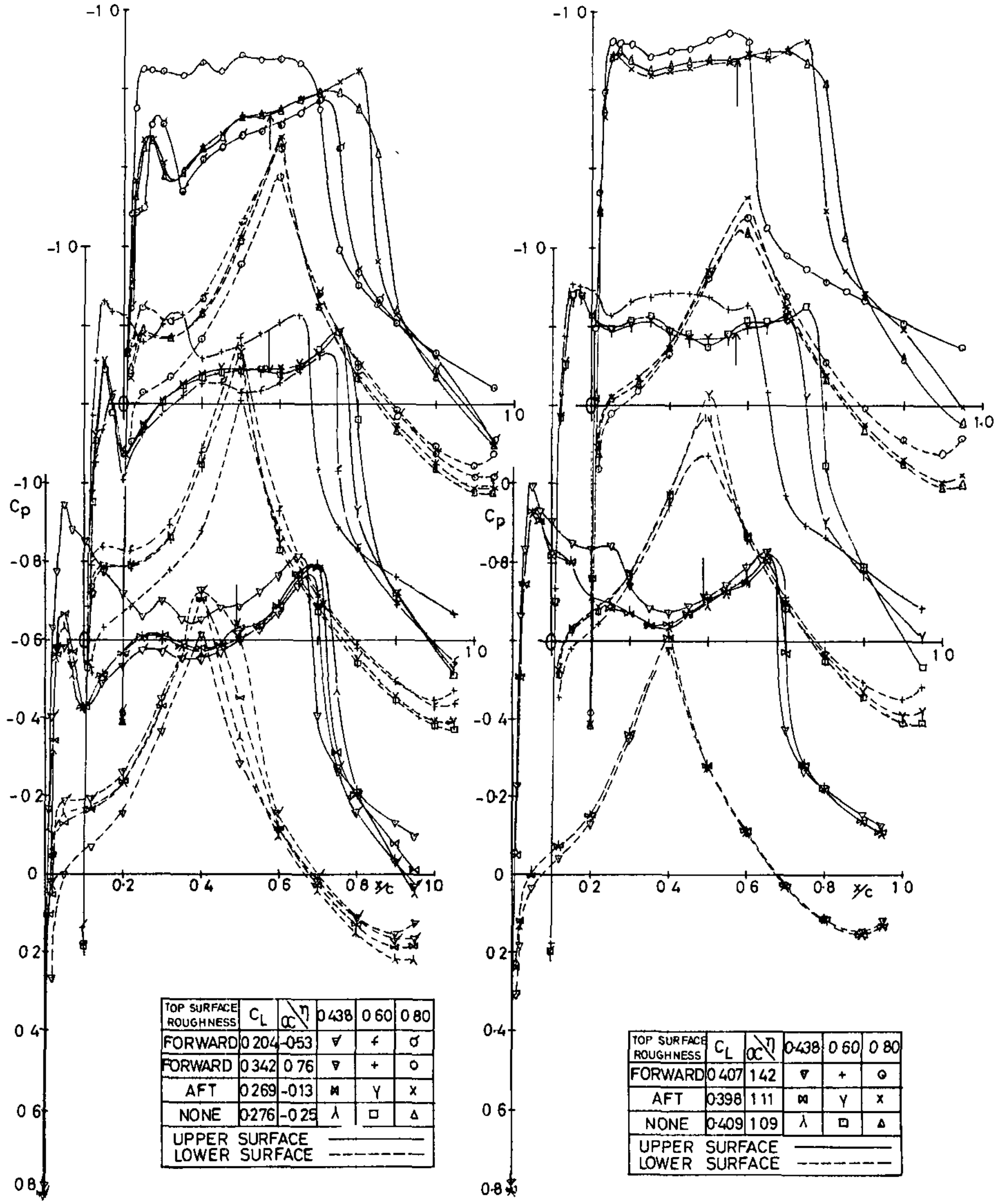
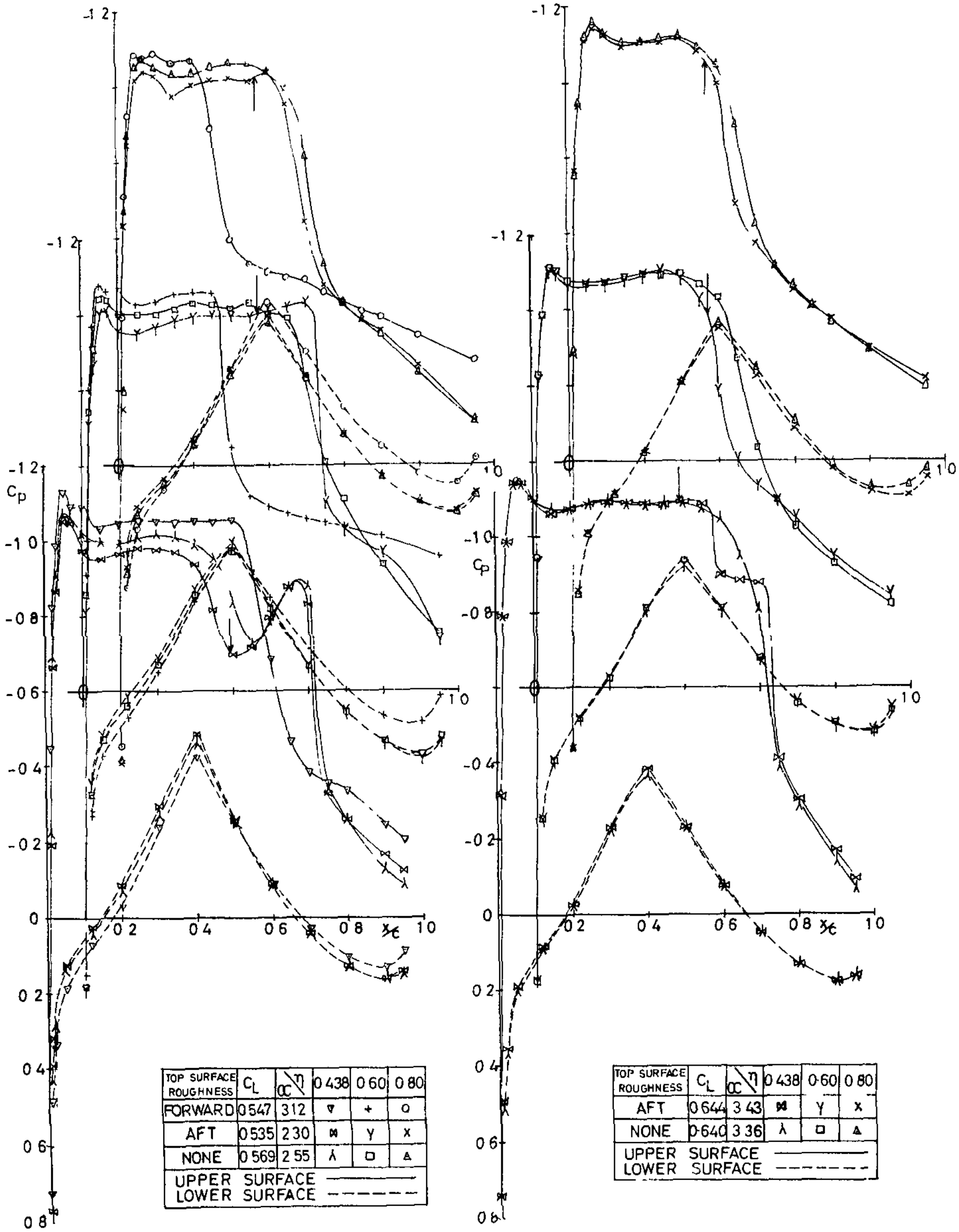


FIG 12a PRESSURE DISTRIBUTION M = 0·86

FIG 12b

ARROWS INDICATE
POSITION OF AFT
ROUGHNESS BAND



TOP SURFACE ROUGHNESS	C_L	α° \ η	0 438	0 60	0 80
FORWARD	0 547	3 12	▽	+	○
AFT	0 535	2 30	⋈	Y	X
NONE	0 569	2 55	λ	□	Δ

TOP SURFACE ROUGHNESS	C_L	α° \ η	0 438	0 60	0 80
AFT	0 644	3 43	⋈	Y	X
NONE	0 640	3 36	λ	□	Δ

FIG 12b PRESSURE DISTRIBUTION AT M = 0 86

ARROWS INDICATE
POSITION OF AFT
ROUGHNESS BAND

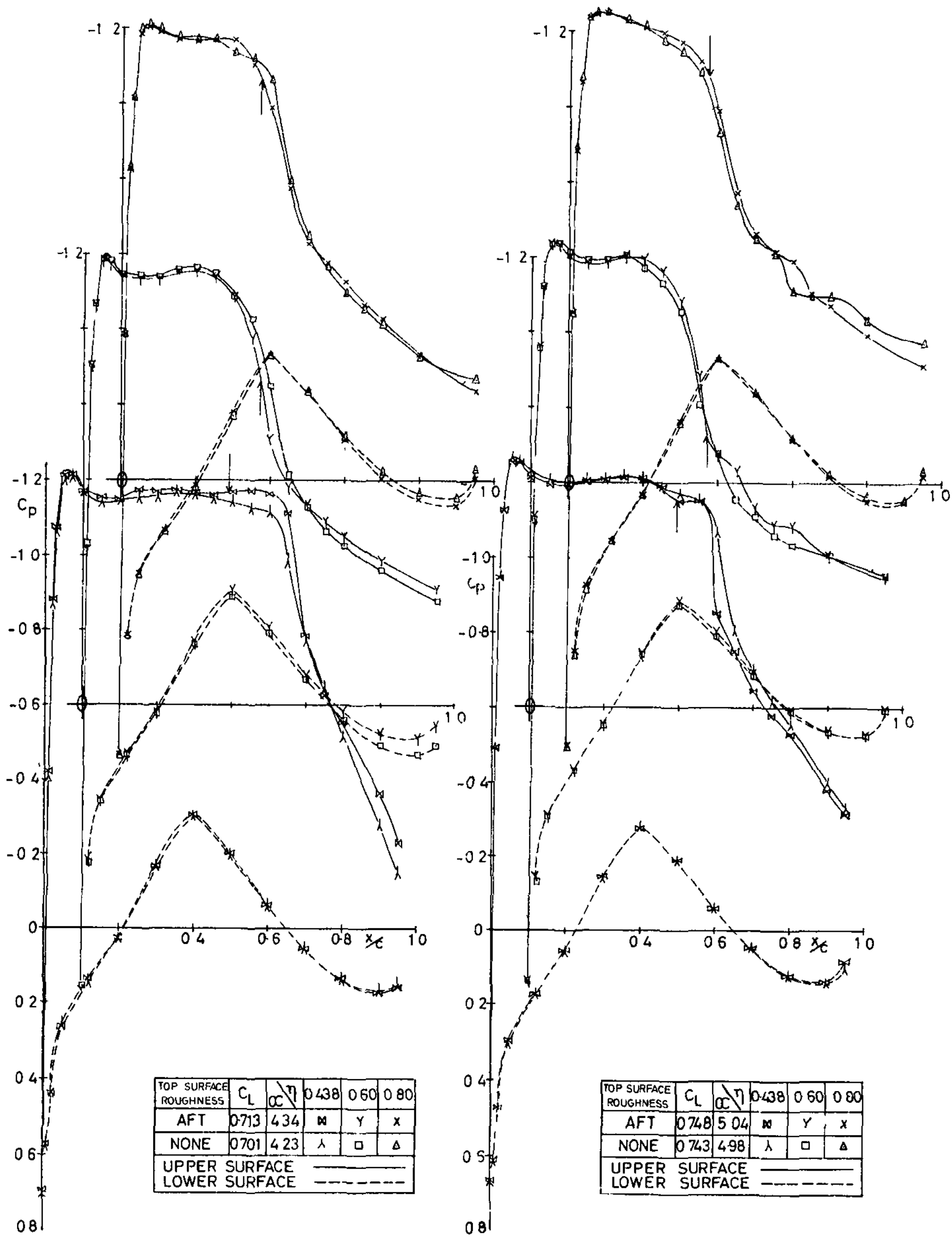


FIG 12c PRESSURE DISTRIBUTIONS AT M = 0 86

FIG 13a

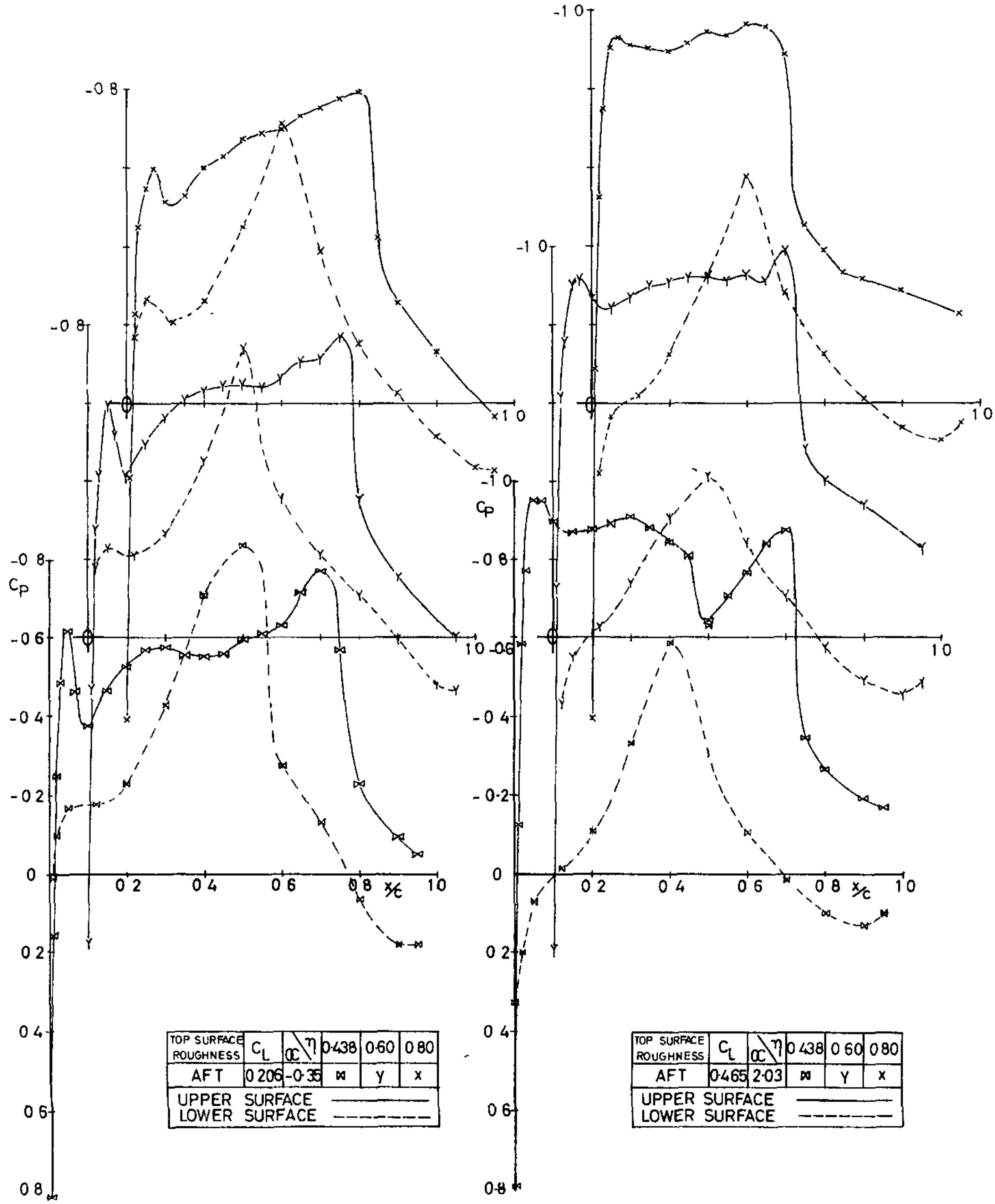


FIG 13a PRESSURE DISTRIBUTIONS AT M = 0·88

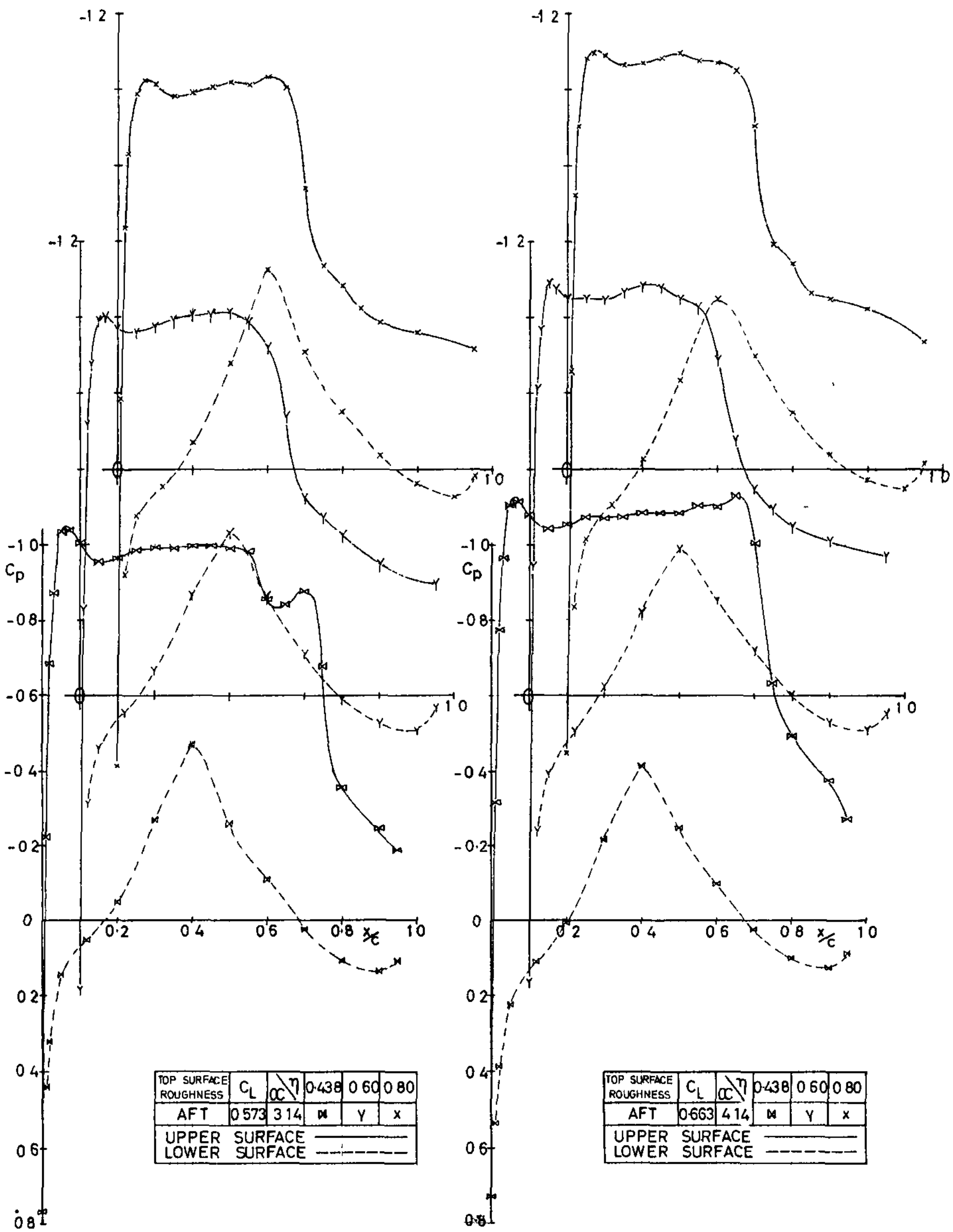


FIG 13b PRESSURE DISTRIBUTION AT M = 0 88

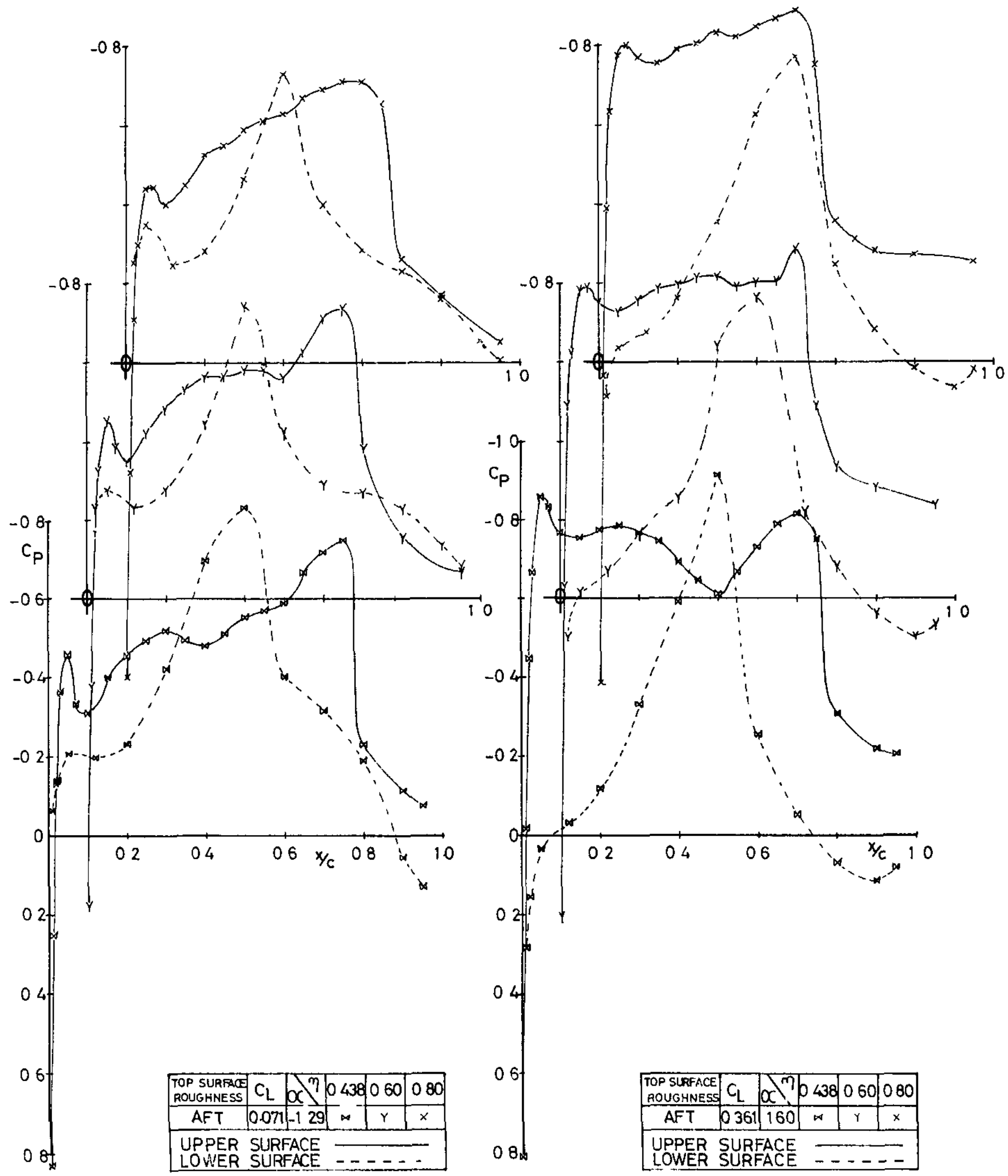


FIG 14 PRESSURE DISTRIBUTIONS AT M = 0 90

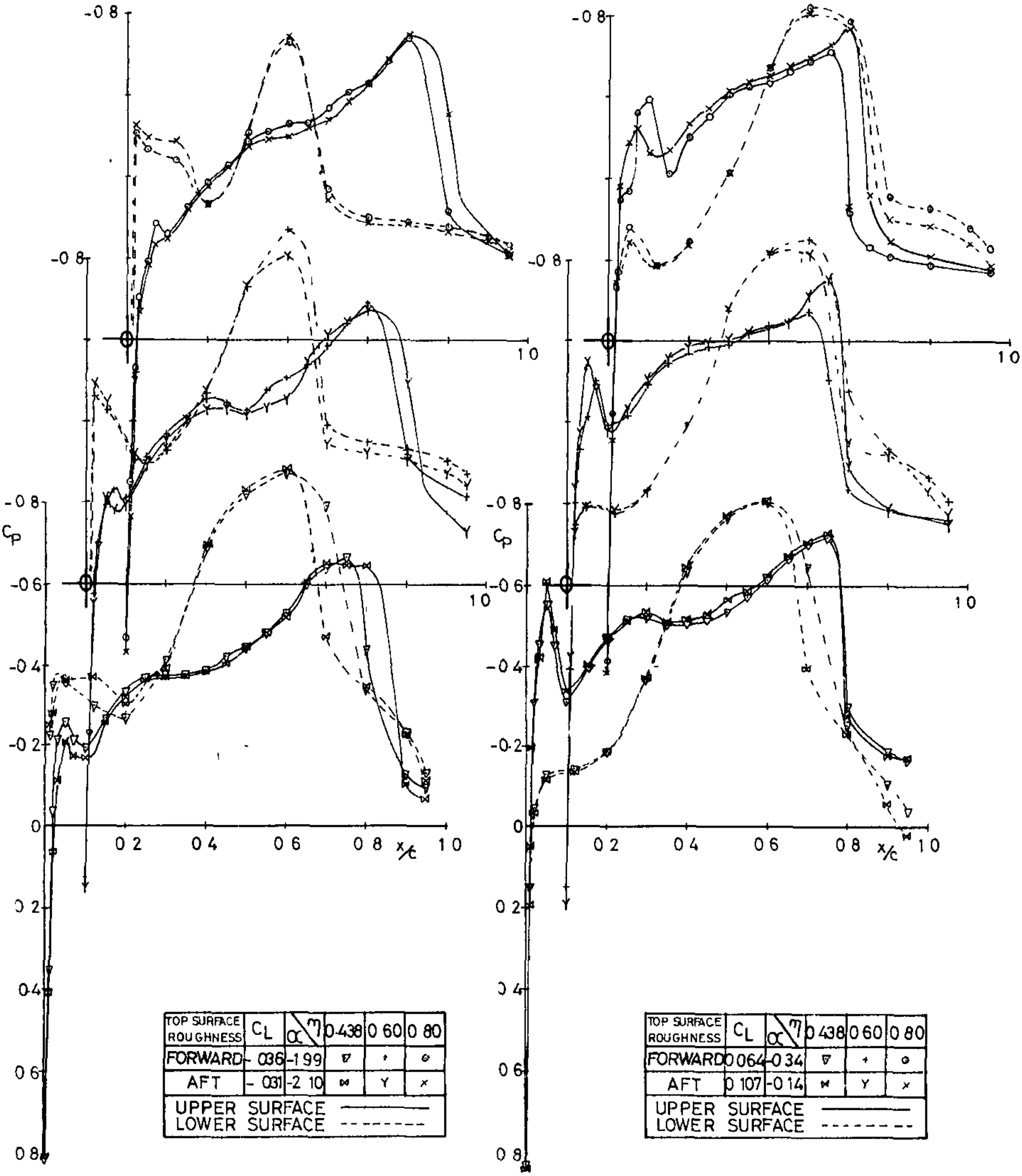


FIG 15a PRESSURE DISTRIBUTIONS AT M=0.82

FIG 15 b

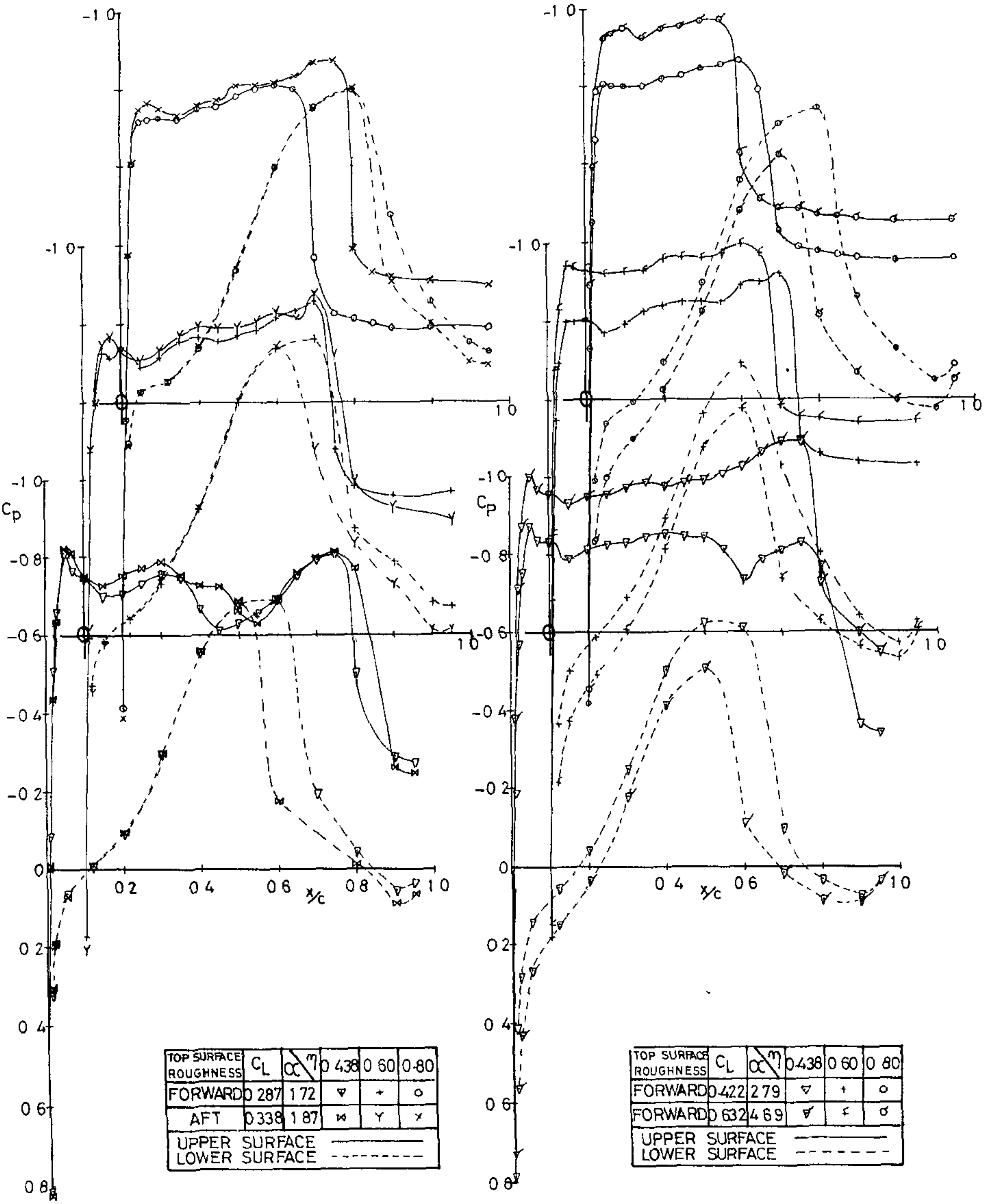


FIG 15 b PRESSURE DISTRIBUTIONS AT M = 0 92

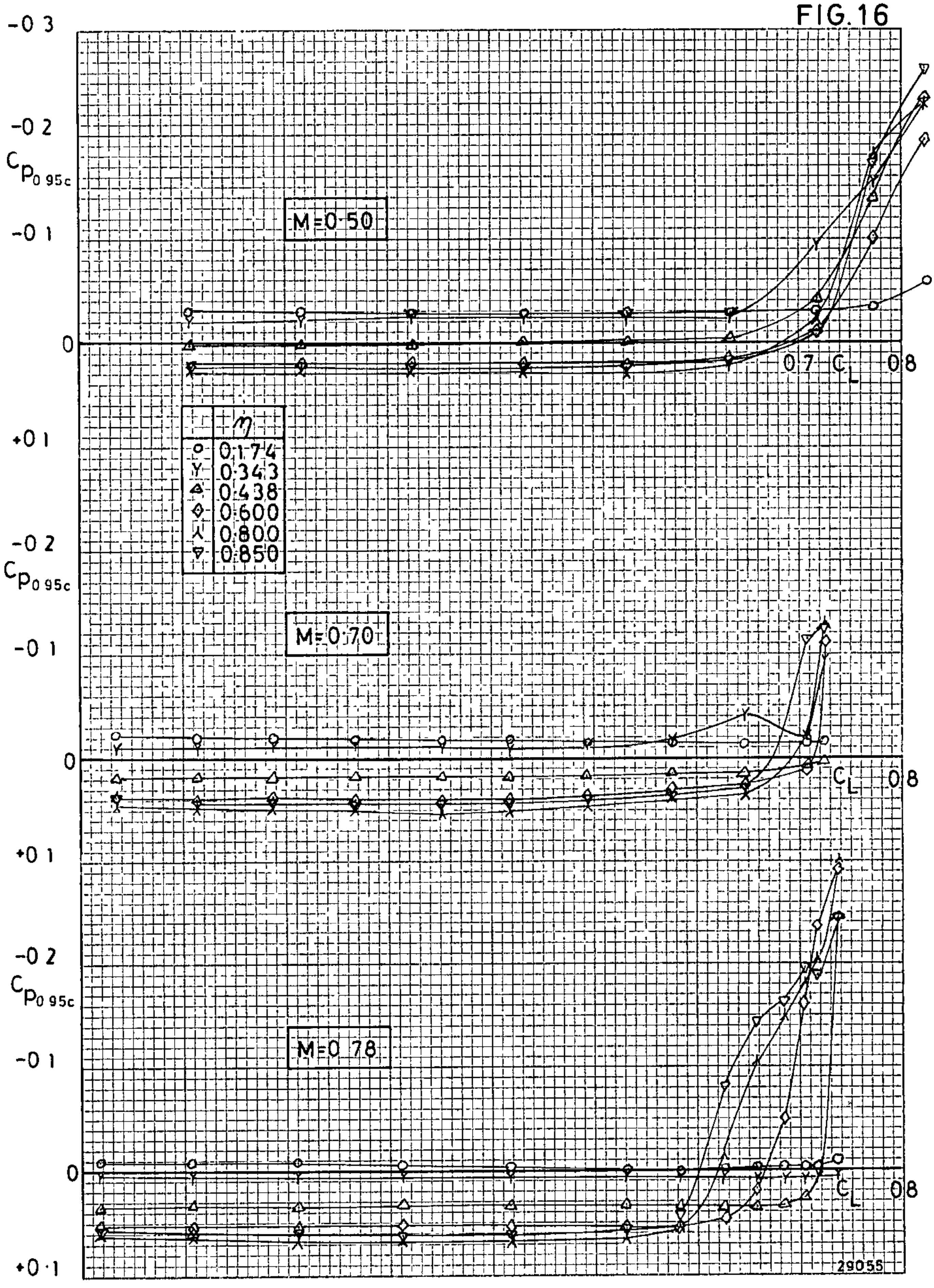


FIG 16 TRAILING EDGE PRESSURE M=0·50, 0·70, 0·78

FORWARD TOP SURFACE ROUGHNESS BAND

FIG. 17a

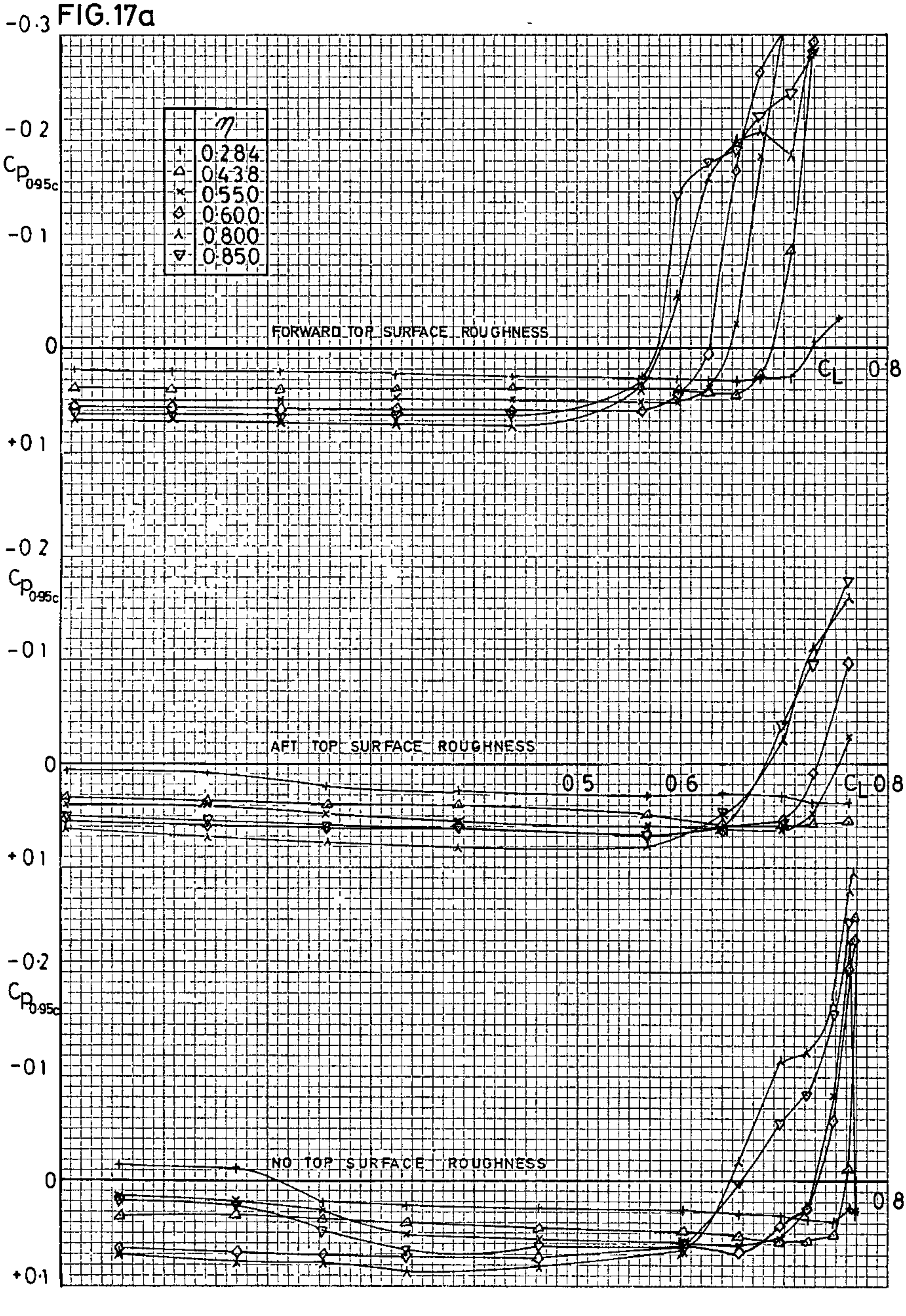


FIG 17a TRAILING EDGE PRESSURES M= 0·80
VARIOUS TRANSITION FIXES

FIG.17b

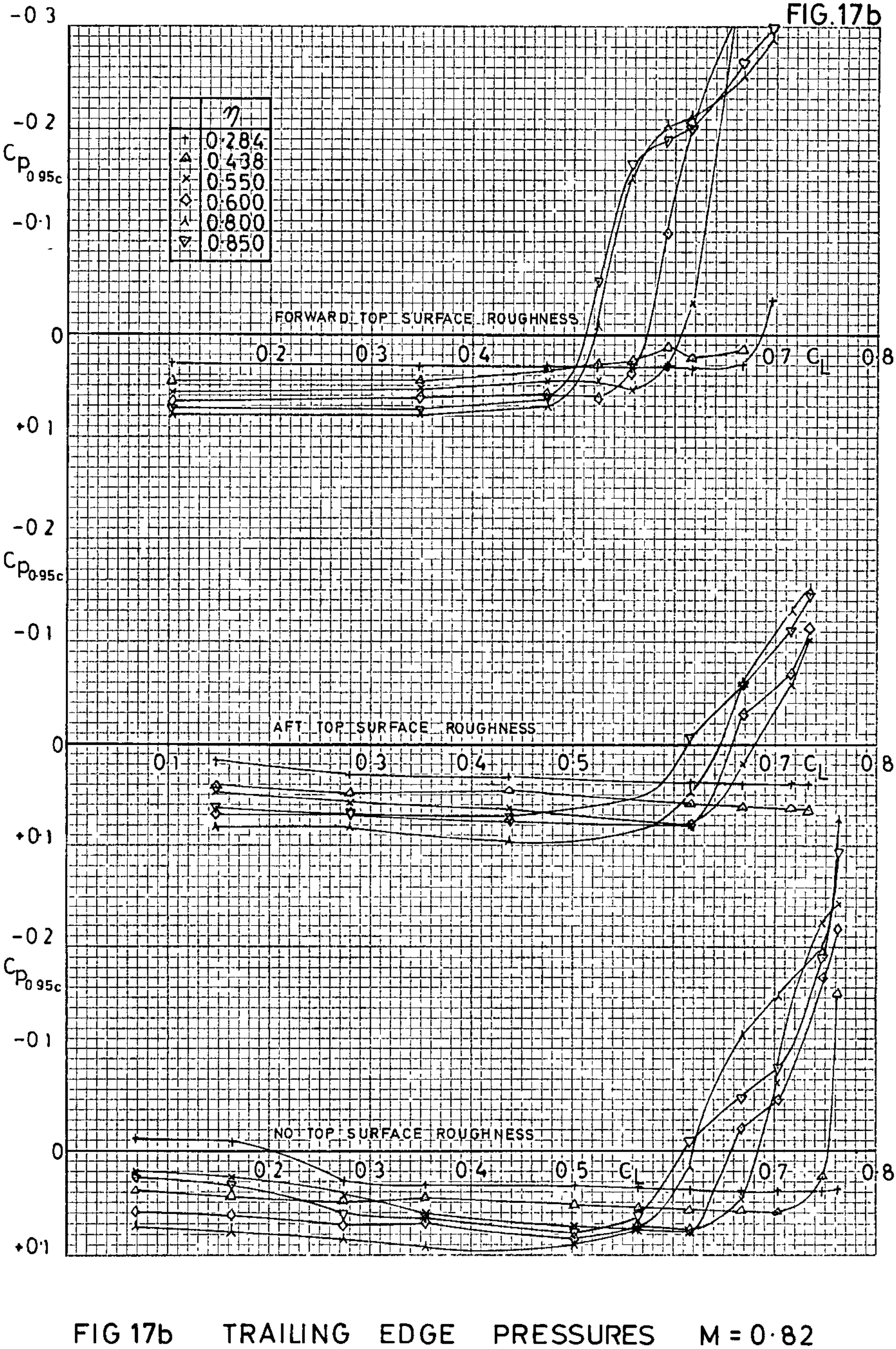


FIG 17b TRAILING EDGE PRESSURES $M = 0\cdot82$

VARIOUS TRANSITION FIXES

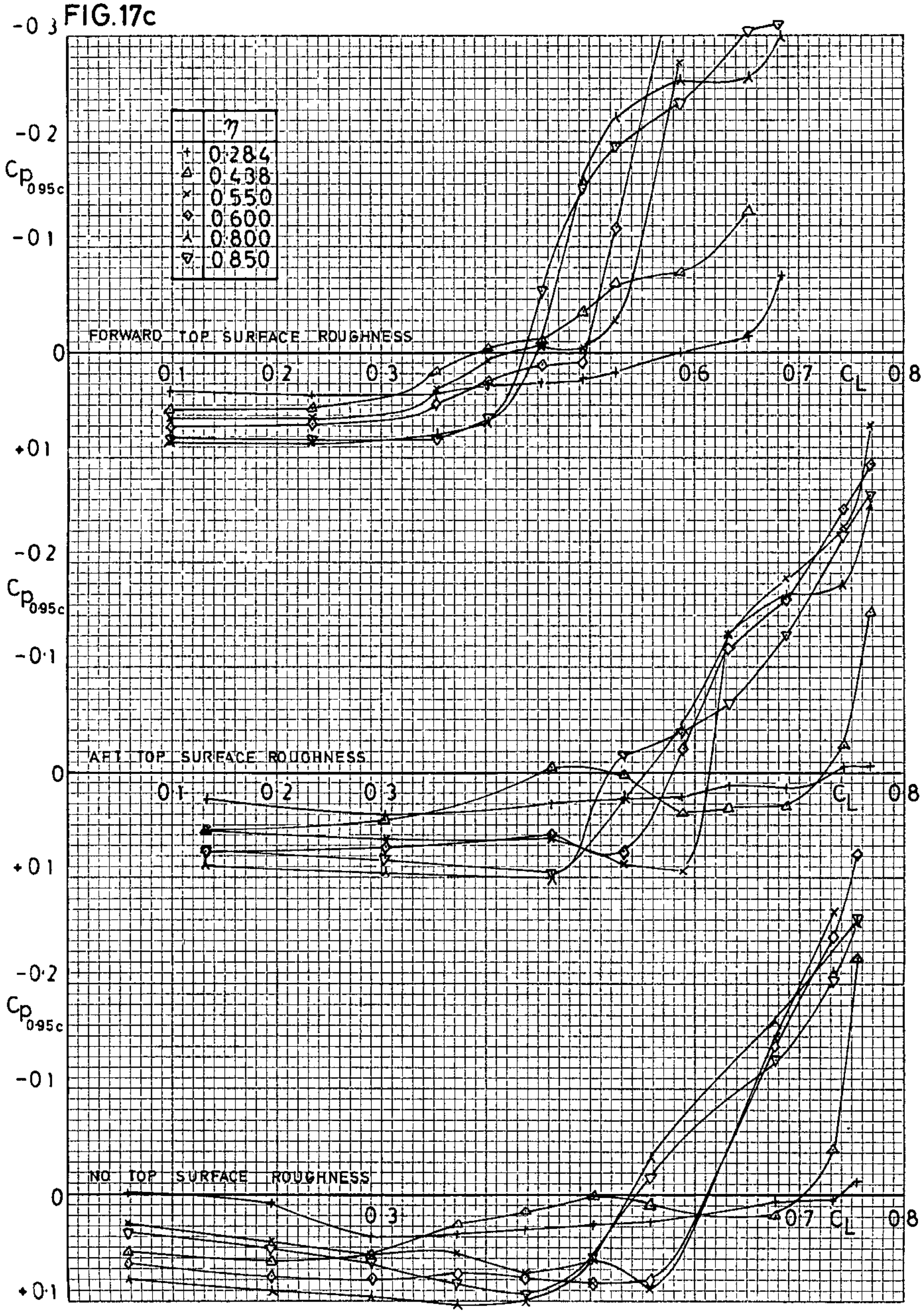


FIG. 17c TRAILING EDGE PRESSURES M = 0·84
VARIOUS TRANSITION FIXES

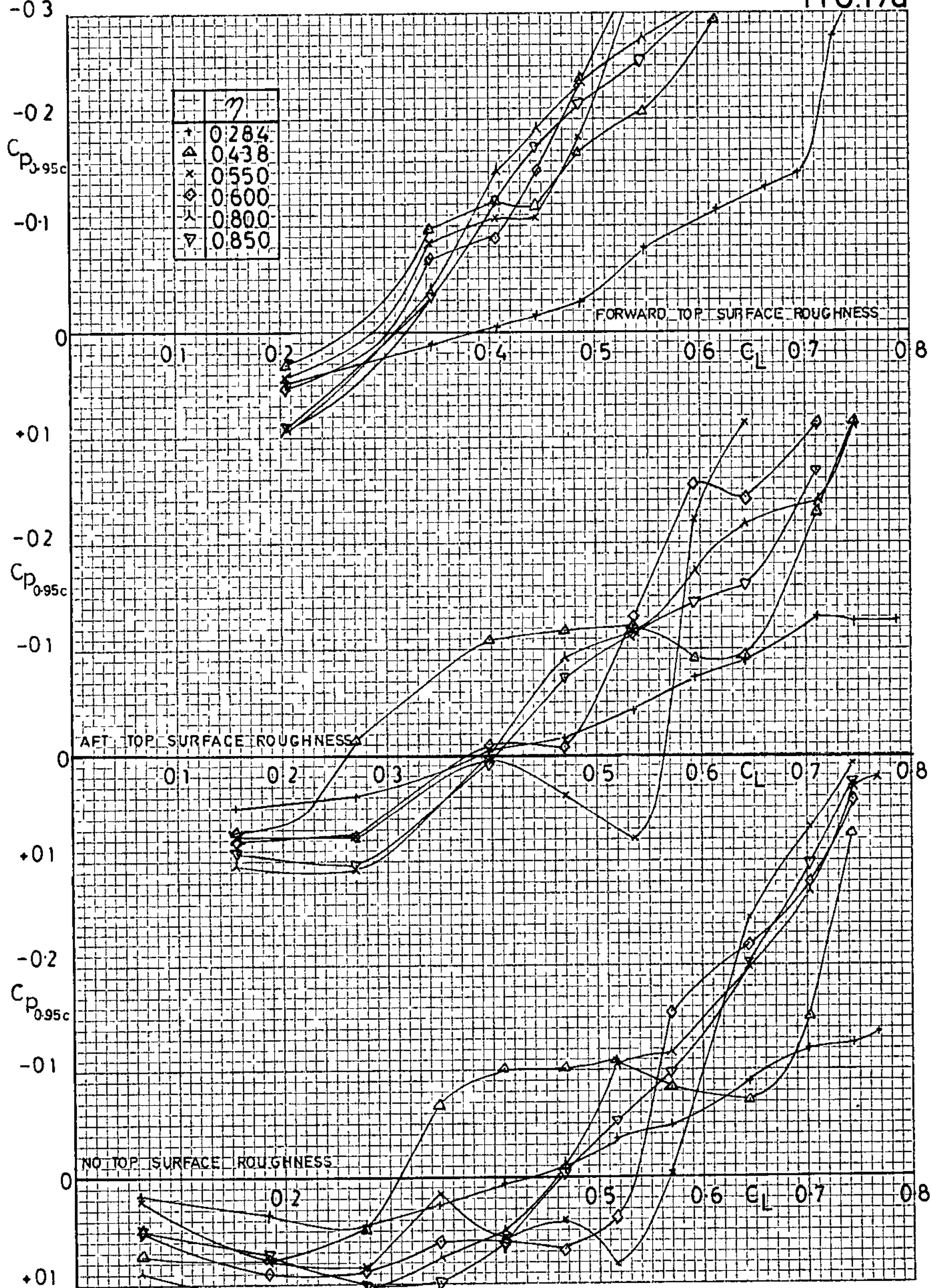


FIG.17d TRAILING EDGE PRESSURES M = 0·86
VARIOUS TRANSITION FIXES

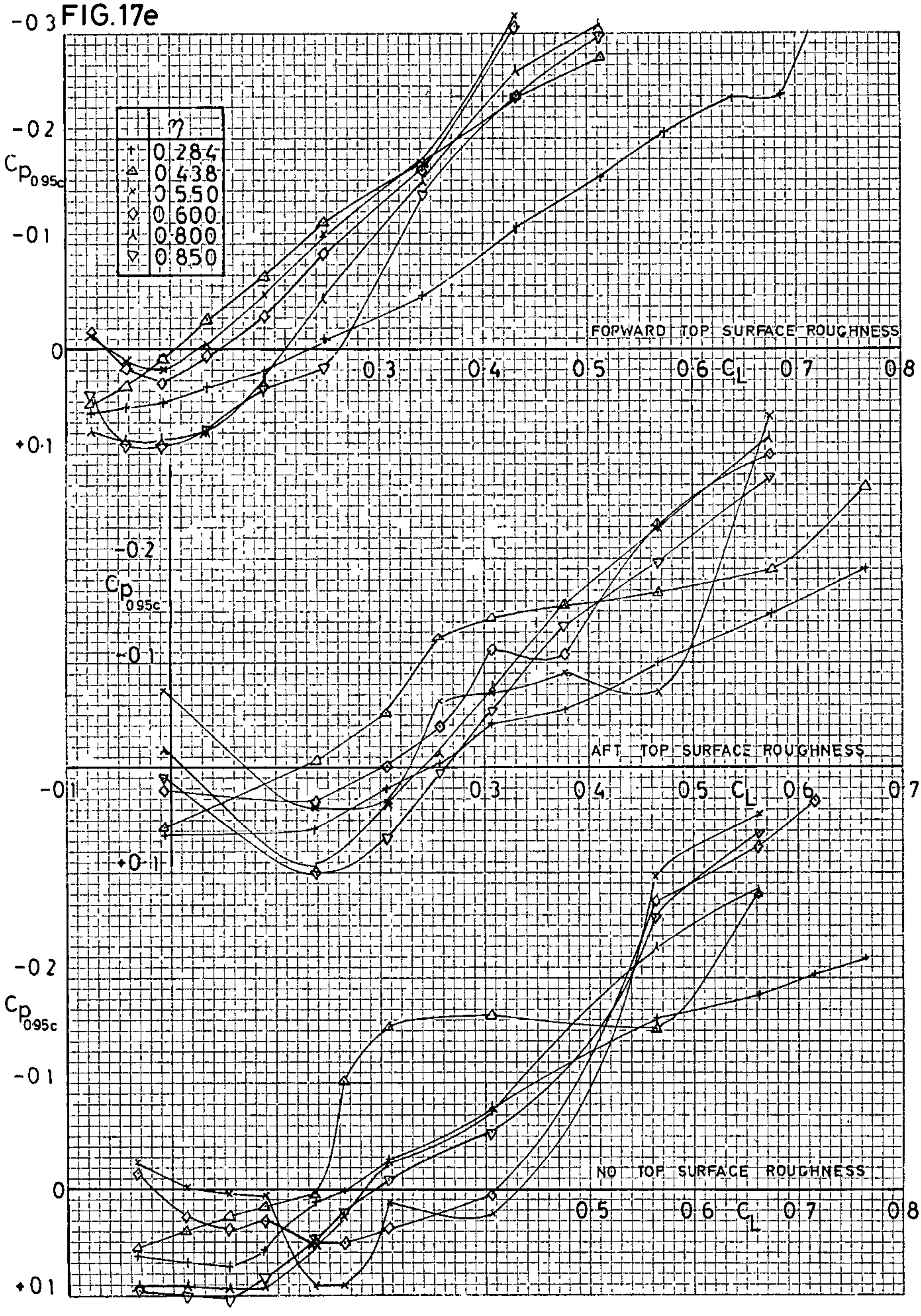


FIG 17e TRAILING EDGE PRESSURES M = 0·88

VARIOUS TRANSITION FIXES

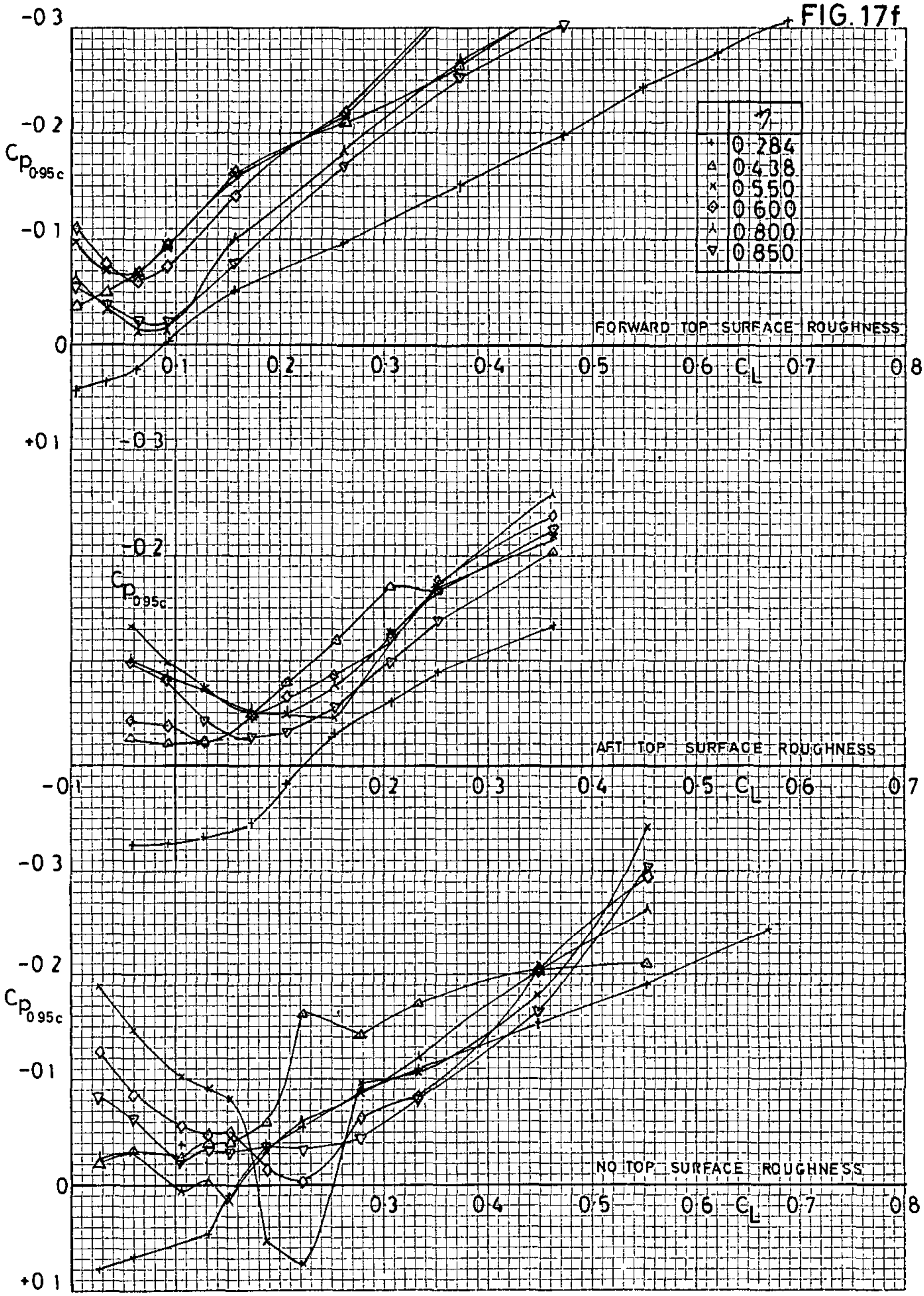


FIG. 17f TRAILING EDGE PRESSURE M = 0·90

VARIOUS TRANSITION FIXES

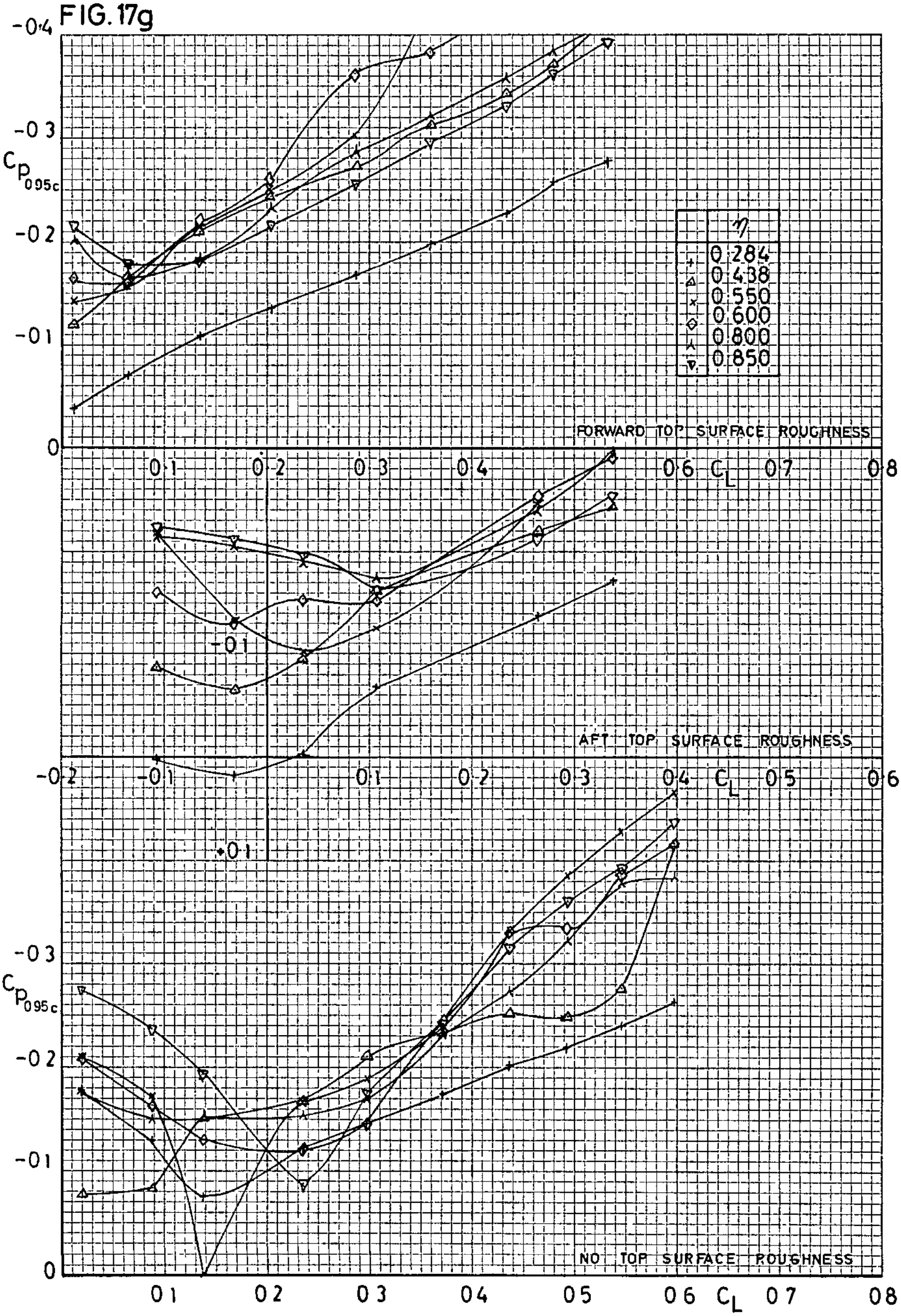


FIG.17g TRAILING EDGE PRESSURES M = 0·92
VARIOUS TRANSITION FIXES

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July, 1970

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AN INVESTIGATION OF TRANSITION FIXING TECHNIQUE FOR A 10.5% THICK, 28° SWEPTBACK WING AT HIGH SUBSONIC SPEEDS AND R $\simeq$ 3×10^6

Gives forces and pressures on a wing of modern design with upper surface transition free or fixed by alternative roughness bands. Development of separated flow and shock positions were very sensitive to transition conditions. Applicability to full scale is discussed.

Illustrates the care needed in fixing transition with an incipient rear separation (class B flow) and the value of pressure plotting. Gives examples of results.

(i)/

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DETACHABLE ABSTRACT CARDS

(i) invalidated by a laminar boundary layer-shock interaction,

(ii) invalidated by interaction between a rear separation and the boundary layer-shock interaction,

(iii) optimistic because the roughness band was too close to the shock.

For reliable prediction of full scale behaviour for similar wings, tests will be needed with several different roughness bands.

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