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Part 4 - Wing Flow

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LOW SPEED FLIGHT TESTS ON A TAILLESS DELTA WING AIRCRAFT (AVRO 707B)

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SUMMARY

Flow visualization tests on a tailless delta wing research aircraft with thickness-chord ratio 0.10 (Avro 707B) showed that the growth of the separated area occurred by progressive inboard movement of a separation boundary which lay roughly chordwise across the wing. Most of the measured changes in the aerodynamic characteristics of the aircraft at high incidence were readily explained by the growth of this separation area.

Some tests were also made to investigate the properties of wing fences and a notch in reducing the longitudinal instability which was caused by the flow separation. Although unsuccessful in preventing the instability the tests gave some insight into the flow changes produced by these devices.

*Replaces R.A.E. Report Aero 2639 - A.R.C. 22256.

CONTENTS

1 INTRODUCTION

The introduction of large angles of sweepback has led to many problems at the lift coefficients used during take off and landing. Many of the stability and control difficulties stem directly from changes in the aerodynamic characteristics of the aircraft produced by quite small areas of separated flow which are present on these wings, even at moderate lift coefficients. Some of these problems cannot be studied in the wind tunnel, because of their dynamical nature, and the available tunnel information must be constantly checked by flight measurement to provide information on scale effects and to study the importance of minor excrescences, which are often not represented on the wind tunnel model.

This is the fourth part of a report which describes flight tests made at the R.A.E. on the Avro 707B, a tailless delta winged research aircraft which was built to investigate the low speed characteristics of this planform. Measurements of lift and drag, longitudinal stability and lateral stability have been reported in the previous parts (Refs. 1, 2 and 3). In this part investigations of the way in which wing flow separation occurred at high lift coefficient are described.

In an attempt to modify the breakdown in flow over the outer parts of the wing and thus prevent the longitudinal instability which it caused at high lift coefficient, the effect of two types of wing fence and of a notch in the wing leading edge have been investigated. Although none of these devices was successful in removing the instability, the tests gave some insight into the changes in wing flow which they produced.

Two test methods which are not in general use were employed. Measurements of the pressure changes produced by the wing modifications were made possible by using pressure plotting tubes attached to the wing surface (see section 3.2), and some flow visualization experiments using a smoke filament discharged above the wing leading edge are described in section 3.1.

2 DESCRIPTION OF THE AIRCRAFT

2.1 General description

A three view general arrangement drawing of the Avro 707B is given in Fig. 1, general views of the aircraft in Figs. 2 and 3, and the principal dimensions in Table 1.

The Avro 707B was a single seat, tailless delta winged research aircraft powered by one Rolls Royce Derwent VIII turbo jet engine mounted in the rear of the fuselage. The intake was in the dorsal position ahead of the fin. The mean sweepback at the wing quarter chord line was 44.5^{o}, the thickness chord ratio 10% and the taper ratio 0.04. The delta shape of the wing planform had been slightly modified by sweeping back the trailing edge of the ailerons to provide increased aileron chord. Control was provided by separate ailerons and elevators mounted along the wing trailing edge and by a conventional fin and rudder. A detailed description of the wing planform and of the variation in wing section along the span appears in Ref. 1.

2.2 Wing modifications

The Notches. Experimental notches were formed in the wing leading edge at the wing ballast mounting positions, since these were covered by conveniently detachable leading edge panels. Their position is shown in Fig. 11b. The notch was cut through the leading edge panel, and the notch section shaped from a block of mahogany which filled the ballast compartment. The shape of the notch is shown in Fig. 4; its depth along the wing chord line was 5 in (0.08c) and the spanwise width 2,75 in. The spanwise position, 0.78 $\frac{b}{2}$, was roughly mid way between the two fence positions.

The wing fences. The fences were tested at two spanwise positions, 0.745 $\frac{b}{2}$ and 0.84 $\frac{b}{2}$. Both fences extended from the leading edge of the ailerons on the upper surface to 10% of the local chord on the under surface but the fences tested at the outboard station were 5.1 in (0.10c) in depth whilst those at the inboard station were 3.65 in (0.05c).

For a few of the tests the outboard fences were extended back to the trailing edge of the ailerons on the upper surface but this arrangement was abandoned because of the severe aileron buffeting which occurred.

The fences were attached to the wing by screwing them to small angle brackets which were riveted to the wing surface. The brackets for both fences remained on the wing for all the tests with fences.

Drawings of the fences and brackets are shown in Fig. 5 and photographs of both fences in position on the wing in Figs. 6 and 7.

2.3 Test conditions

The tests were made at a mean height of 10,000 feet and covered the speed range from about 160 knots to 90 knots eas with the engine idling. The Reynolds Number based on the wing standard mean chord varied from 14×10^6 to 9×10^6. Most of the tests were at the mid c.g. position $h = 0.328\bar{c}$, but a few tests were made at the forward position $h = 0.305\bar{c}$. The investigation covered both the cruising and the landing configurations but most of the results are presented for the cruising configuration only.

3 TEST METHODS

3.1 Flow visualization

Wing tufting. The development of areas of separated flow on the wing at high incidence was studied by taking film records of tuft behaviour during low speed flight. Nylon tufts were used in three positions: on the surface (Fig. 8), on two-tuft masts (Fig 9a), and on four-tuft masts (Fig. 9c). In this way it was hoped to study the depth of the separated region and the shape of its boundaries.

For the tests with fences, surface tufts were used inboard of the fence and four-tuft masts outboard; the two top tufts on the masts were generally visible over the fence.

Two cameras were used to photograph the tufts. A 16 mm G.S.A.P. camera with ½ in reversed telephoto lens was mounted in the dorsal fin; its field of view covered the outer third of the wing leading edge and almost all of the trailing edge. A 35 mm Bell and Howell A-4 camera mounted in the fuselage ahead of the intake was used to study the flow nearer to the wing root. These fields of view are shown in Fig 9b.

Smoke filament studies. The tuft investigation showed that the depth of the separated region was too great to be revealed by tufts positioned on masts of reasonable size. Attempts to trace the outline of the separated region by long streamer tufts attached to the leading edge were also unsuccessful owing to the weight of the tuft.

A simple form of smoke generator was constructed by carrying a smoke signal cartridge at a convenient position under the wing and arranging for it to discharge over the wing leading edge through a suitable nozzle. The cartridge used was electrically fired and produced a dense smoke filament for about 40 seconds. Several different nozzle positions were tried before the final arrangement shown marked a in Fig. 10 was chosen. It was found that with nozzles close to the wing leading edge, arrangements c and d of Fig. 10, the smoke diffused rapidly over both upper and lower wing surfaces; moving the nozzle position forward along the wing chord line, as in b of Fig. 10, produced a more suitable filament but this again became diffused as it flowed close to the wing surface. The most satisfactory position, shown at a in Fig. 10, was with the nozzle about 6 inches above the leading edge.

The smoke filament was photographed with the 16 mm G.S.A.P. camera positioned in the dorsal fin, using colour film to provide greater contrast with the red signal smoke.

It was found that the smoke pipe rapidly became blocked with a hard deposit which was difficult to remove. The length of the pipe should therefore be kept as short and its diameter as large as possible. In these tests 1 in diameter pipe was used.

3.2 Pressure plotting

A technique for measuring the chordwise pressure distribution on an aircraft wing by laying pressure plotting tubes on the surface is described in Ref. 5. In the present tests the changes in pressure produced by the fences and the notch have been investigated by laying surface tubes at fifteen spanwise positions.

The plastic tubes were embedded in Araldite fairings which extended from 2% chord on the wing upper surface to the rear of the wing just ahead of the control gaps; they were then carried spanwise along the rear of the wing to the wing body junction and thence forward into the observer bay. The positions of the tubes and the cross section of the fairings are shown in Fig. 20 whilst some of the tubes in position on the wing may be seen in the photographs, Fig. 6 and 7.

The internal diameter of the tube was 0.08 inches and each tube was 27 feet long. A hole, 0.04 inch diameter, was drilled through the top of the fairing and into the plastic tube at the selected chordwise position. Each of the tubes was connected to a standard aircraft altimeter which was photographed on a camera auto observer, their readings later being converted to pressures by the standard calibration. The 'free stream' static and dynamic pressures were calculated from the readings of the aircraft A.S.I. and altimeter systems, corrected for position error.

Leaks occurred in several of the tubes due to cracking and distortion of the Araldite fairing produced by relative movement of the wing skin panels and it proved impossible to seal these leaks without relaying the tubes. In later tests with surface tubes at speeds up to 200 knots eas it was found possible to dope the tubes to the surface with a covering of linen tape and this led to a simpler and more flexible installation.

3.2.1 The effects of cross flow on the pressure measurements

The tests with the surface tubes were proposed primarily to investigate the pressures within the various separated regions in order to assess their relative effect upon the wing pitching moment. The velocities within these regions were expected to be quite small so that the pressure changes produced by local cross flow over the tube fairings could safely be ignored. Measurements were also made however, in the attached flow region where, because of the much larger velocities, the likely pressure changes due to cross flow should be more carefully examined. The changes in pressure coefficient for various angles of cross flow, estimated on the basis of simple inviscid theory are shown below. For these estimates the actual cross section of the fairing, shown in Fig. 20a, was represented by a semi-circle.

Angle of cross flow	ΔC_P
5^o	-0.023
10^o	-0.090
15^o	-0.201
20^o	-0.351
30^o	-0.750

The estimates show that the errors in pressure measurement may be quite large for moderate angles of cross flow. Tests made by Port and Morrall[5] showed that measurable pressure errors were produced by a fairing 3½ in wide and ¼ in deep mounted on the wing of a Meteor aircraft between fuselage and nacelle. In this case the wing chord was 132 in and the fairing was yawed at 25^o to the stream. It appears therefore that considerable care must be taken in designing the fairings for surface tubes if cross flow is likely to occur.

In the tests described in this report the cross flow at the leading edge indicated by surface tufts, was generally small but the quantitative measurements in the attached flow region must nevertheless be regarded with caution.

3.3 Measurement of pitching moment

The pitching moment coefficient was calculated from measurements of the elevator angles needed to trim the aircraft, using the relationship for the equilibrium of pitching moments:-

$$C_{M_o} + \frac{\partial C_M}{\partial \eta} \cdot \eta = 0$$

The elevator effectiveness, $\frac{\partial C_M}{\partial \eta}$, was calculated from measurements made at the two centre of gravity positions, for if $\Delta\eta$ is the change in elevator angle to trim at constant lift coefficient for the two c.g. positions, $h_1\bar{c}$ and $h_2\bar{c}$:-

$$\frac{\partial C_M}{\partial \eta} = \frac{(h_1 - h_2)C_L}{\Delta\eta}$$

4 RESULTS AND DISCUSSION

4.1 Flow separation on the basic wing

Film records of wing tuft behaviour when reducing speed slowly and in stabilised flight at high incidence have been analysed so that the continuous development in separated area with lift coefficient has been established. The variation in separated area at 0.1 increments in lift coefficient between C_L = 0.40 and C_L = 0.80 is shown in Figs. 11a to 15a whilst the areas at C_L = 0.84 and C_L = 0.91 are shown in Fig. 16.

Separation occurred initially at the wing tips, the regions of attached and separated flow being divided by a well defined boundary which lay roughly chordwise across the wing. With increasing incidence this boundary moved progressively inboard whilst retaining its chordwise orientation. The inboard movement has been illustrated in Fig. 17 by showing the variation in the position at which the boundary crosses the wing trailing edge with incidence. The tests at the highest incidences suggested that the inboard movement was retarded by the proximity of the body.

The tests with tufts at various heights above the wing surface showed similar positions for the boundary, indicating that the separation developed immediately to a considerable depth. The behaviour of individual tufts as the incidence was gradually increased also gave some information about local changes in the flow. The approach of the separation boundary was marked by slight quivering of the raised tufts whilst the surface tufts, which had previously been lying parallel to the wing chord line, began to show 10 or 20 degrees of outflow at the trailing edge. The onset of separation was quite sudden, all the tufts in a chordwise row simultaneously becoming violently disturbed, but this gradually subsided as the boundary moved even further inboard, until tufts which were well inside the separated region showed comparatively little disturbance but were pointing predominantly forward.

To investigate the depth and shape of the separated region a smoke filament was discharged above the wing leading edge from the smoke generator described in section 3.1. Photographs of the smoke filament discharged above regions of attached and separated flow are shown in Figs. 19a and b and the paths are compared in Fig. 19c. The tests showed that the separated region was of considerable depth and that, as far as could be ascertained from the path of the filament, there was no tendency for the flow to reattach towards the trailing edge. Examination of continuous records taken as the separation boundary passed the position of the nozzle emphasised the discontinuity at the boundary, the filament path changing abruptly from that of Fig. 19a to that shown in Fig. 19b.

The growth of separated flow by the progressive inboard movement of a chordwise boundary has been found on several aircraft with planform sweepback and thickness chord ratio similar to the test aircraft. The types of flow which may exist on these wings and the nature of such separation boundaries have been discussed in Ref. 7. It was shown that a 'part span vortex sheet' would be formed to divide the main flow from the comparatively slowly moving flow within the separated region, this sheet having characteristics similar to the boundary found in these tests.

4.1.1 Flow separation in the landing configuration

The measurements of the area of separated flow on the basic wing were repeated for the aircraft with the undercarriage down and the airbrakes extended. The positions of the airbrakes, which hinge forward on the upper and lower surface, are shown in Fig. 1 and they may also be seen in the photograph, Fig. 3. The airbrakes produced considerable turbulence in the flow over the rear of the wing but no separation was observed. The measurements of lift[1] show that there was a reduction in lift for a given incidence between the cruising and landing configurations. It was found that the spanwise position of the separation boundary for the two configurations agreed well when compared at the same incidence rather than the same lift coefficient.

4.1.2 Comparison with wind tunnel tests

Measurements of the area of separated flow were also made during wind tunnel tests[4] on a 1/8th scale model using small surface tufts. The Reynolds Number of the tunnel tests, based on the wing standard mean chord, was 1.45×10^6 compared with the flight range from 9×10^6 to 14×10^6. The boundary between the attached and separated regions did not appear to be as well defined in the tunnel tests as that found in flight but this may have been because still photographs were used in analysing the tunnel results.

The variation of the boundary position with incidence is shown in Fig. 17 for comparison with the flight measurements. The earlier flow separation in the tunnel, which may be attributed primarily to the lower tunnel Reynolds Number, has been reflected in several of the tunnel-flight comparisons contained in earlier parts of this report. In particular the earlier reduction in the slope of the lift v. incidence curve and the onset of the separation drag rise are shown in Fig. 18, taken from Ref. 1 whilst its effect on the tunnel measurements of aileron power and rolling moment due to sideslip are discussed in Ref. 3. To enable the overall effects of flow separation to be readily assessed the flight measurements of the aerodynamic characteristics which were presented in previous parts of this report are reproduced in Fig. 33. For a detailed discussion of each of these characteristics reference should be made to the appropriate report.

4.2 Changes in the flow produced by the notch and fences

The areas of separated flow which occurred on the wing when the notch or fences were fitted are shown in Figs. 11 to 15 ~~for~~ 0.1 increments in lift coefficient between $C_L = 0.40$ and $C_L = 0.80$.

Several features were common to the results for all the wing modifications tested. The fence or notch appeared to have little effect upon the inboard movement of the separation boundary along the trailing edge but in each case a small wedge shaped area of attached flow was maintained at the leading edge just outboard of each modification. In addition to the main separation boundary moving in from the wing tips, an isolated area of separated flow occurred just inboard of the fence or notch. Initially this area was confined to a small bubble at about one third of the chord, but with increasing incidence the region extended rearwards until it became detached at the trailing edge. The attached area at the leading edge was maintained, even when the tip and inboard separations had eventually joined to form a new boundary lying further inboard, and in the case of the fences, small areas of attached flow could still be observed at the highest lift coefficients tested.

The lift coefficient at which the inboard separation first appeared varied with the spanwise position of each modification. That at the outboard fence appeared at $C_L \triangleq 0.35$, that at the inboard fence at $C_L \triangleq 0.45$ and that at the notch slightly earlier. Subsequent stages in the development of the separation maintained this difference in lift coefficient for the three positions.

The inboard fence configuration was exceptional in showing considerable amounts of cross flow in the neighbourhood of the inboard separation; about 30° of inflow at the leading edge and 30° outflow at the trailing edge was indicated by the surface tufts. These tufts also showed that the surface flow inside each of the inboard separations was predominantly forward.

The handling qualities of the aircraft with each of the modifications tested were much impaired by severe aileron buffeting and overbalance associated with the inboard flow separation. A few tests were made with the outboard fence extended back to the trailing edge of the ailerons on the upper surface. Whilst this modification appeared to have no effect on the area of separated flow, there was a considerable increase in the aileron buffeting and the tests were therefore discontinued.

4.2.1 Changes in static pressure produced by the notch and fences

The changes in wing flow discussed in the previous section were also examined by measuring the changes in static pressure which occurred in the neighbourhood of the wing modifications. Since the aircraft was not provided with surface pressure plotting holes the system of external tubes, described in section 3.2, was employed. The positions of the tubes, shown in Fig. 20, included stations in the junctions on either side of the fences and on either lip of the notch.

The variation of the pressure coefficient, C_p, with total wing lift coefficient for eight of the spanwise stations on the basic wing is shown in Fig. 21. The experimental points have generally been omitted for clarity but those given for station 6 are representative of the experimental scatter. The chordwise measuring position, 7%c, was chosen to be sufficiently far behind the leading edge to avoid large changes in the pressure observations due to slight chordwise variation of the position of the peak suction. The chordwise pressure gradient in the neighbourhood of the measuring position was examined by repeating the tests for two of the spanwise stations, one close to the tip and one further inboard, with the hole at 9% chord. The results in Fig. 22 show that the increased gradient at the outboard station was larger than that which might be expected from the increase in loading towards the tips. This suggests that the chordwise position of the peak suction was further aft at the tip sections; a result which is contrary to previous experience.

The measurements on the basic wing, Fig. 21, showed that at those stations which were unaffected by the direct influence of wing tip or wing body junction, the suction pressure coefficient increased almost linearly with total wing lift coefficient up to the value at which local flow separation occurred. This pressure coefficient at separation was fairly constant for all the spanwise stations so that lower separation incidence of the tip sections, shown by the flow visualization tests, is readily explained by their relatively higher loading. As the incidence was increased beyond that required for separation the pressure coefficient fell, rapidly at first but then more slowly until a steady value was reached with the measuring station well inside the separated area. This final suction pressure coefficient varied slightly from station to station, being larger towards the root sections.

The changes in pressure coefficient produced by the wing modifications are shown in Fig. 23a to Fig. 23j. In each of these figures the variation of pressure coefficient with lift coefficient at one station on the wing with a notch or fence is compared with similar measurements on the basic wing. The positions of the measuring stations and of the wing modifications are shown in Fig. 20.

The pressure changes measured in the neighbourhood of the two fences were generally similar. Outboard of the fence the suction pressures were considerably reduced, this effect being especially marked at the station closest to the fence. At these positions the sudden drop in pressure coefficient, associated on the basic wing with the onset of separation, was also absent. These tests thus confirmed that small areas of attached flow were maintained on the wing just outboard of the fence and showed that this effect was probably achieved by the reduction in the maximum suction to values below those which were found to correspond to separation on the basic wing.

Inboard of the fences the measured pressure changes were at variance, those made close to the inboard fence showing an increase in suction pressure, Fig. 23h, whilst those at the outboard fence showed a decrease, Fig. 23d. In both cases the inboard flow separation noted in the flow visualization tests was clearly indicated. It seems more likely that this inboard separation would be associated with increased suction pressures (see Ref. 7), but the measurements at the inboard fence must be considered less reliable because of the pressure errors incurred by the larger angles of crossflow (section 4.2).

The pressure changes produced by the notch were similar in form to those produced by the fences but of smaller magnitude. The reduction in suction pressure just outboard of the notch and the apparent absence of separation may be seen in Fig. 23e, whilst a slight increase in suction and the earlier separation inboard of the notch is shown in Fig 23f.

The pressure coefficients measured in the wing fence junction on either side of the fences and on the lips of the notch are compared in Fig. 24a, b and c. The pressures measured in the junctions on the outer side of the fences are very similar to those measured in the wing body junction on the basic wing Fig. 22.

4.3 The effect of the notch and fences on longitudinal stability

The notch and fences were fitted primarily in an attempt to control the longitudinal instability which was caused by the wing tip flow separation at high lift coefficient. Although the tests reported in the previous sections have shown that some modification of the wing flow was achieved, it was apparent from the stability tests that these changes were much too small to affect the behaviour of the aircraft appreciably. The tests are nevertheless valuable in showing the way in which these devices may be expected to work under less severe conditions.

Measurements of the elevator angles needed to trim the aircraft in steady power-off flight were made at the mid and forward centre of gravity positions for the basic aircraft and for the aircraft with both fence configurations. The measurements with the notch were made only at the mid c.g. The test measurements, corrected to the neutral position of the spring and trim tabs are shown in Figs. 25, 26, 27 and 28. The difficulty of trimming the aircraft accurately at high lift coefficients, due to the reduced longitudinal and lateral stability, is indicated by the increased scatter of the experimental points.

The measurements of elevator angles to trim have been reduced to pitching moment curves using a value of the elevator power calculated from the test results at two c.g. positions (see section 3.3). No significant difference could be found for the elevator power for the basic aircraft and for the aircraft with fences so that this value was also used in calculating the pitching moment of the aircraft with the notch. The pitching moment curves measured about the mid c.g. position for the four configurations are compared in Fig. 29. The changes in stability produced by the wing modifications were small; with the outboard fence the instability occurred at a slightly lower lift coefficient than for the basic wing, whilst with the inboard fence the onset of instability was slightly delayed but the final instability was in this case more severe. The changes in pitching moment due to the notch were too small to be considered significant.

Directly comparable wind tunnel measurements of the effect of wing fences or a notch on the stability of this aircraft have not been made but tunnel tests on the Avro 707A aircraft fitted with a fence which corresponded closely in size and position to the outboard fences of the current flight tests are reported in Ref. 6. The planform and stability of these two aircraft were sufficiently similar to allow a reasonable comparison, but whilst the tunnel measurements showed complete suppression of the instability up to a considerably higher lift coefficient, Fig. 31, the flight tests showed little change, Fig. 30. Flight measurements on the almost identical Avro 707A aircraft also confirmed that the predicted changes in stability were not realised.

4.4 Summary of aerodynamic measurements

To enable the overall effects of flow separation to be readily assessed the flight measurements of the aerodynamic characteristics which were presented in previous parts of this report are reproduced in Fig. 33. For a detailed discussion of these characteristics reference should be made to the appropriate report.

5 CONCLUSIONS

The tests have shown that the flow separation on this aircraft occurs by progressive inboard movement of a chordwise separation boundary. Most of the measured changes in longitudinal and lateral stability at high lift coefficient are consistent with the changes in aerodynamic loading produced by this flow separation.

Although the fences and notch tested were unsuccessful in removing the longitudinal instability at high lift coefficient, the tests showed that the reduced suction pressures outboard of the modifications were associated with delayed separation in this region; this favourable effect was offset however by earlier separation inboard of the modification.

The use of pressure plotting tubes attached externally to the wing surface proved to be a convenient method of measuring local surface pressures but care should be used in interpreting their readings if large angles of cross flow are present.

The smoke filament technique forms a valuable extension of present flow visualization methods especially when the flow some distance away from the aerofoil surface is to be investigated.

LIST OF SYMBOLS

Symbol	Definition	Unit
b	Wing span	feet
c	Local wing chord	feet
$\bar{c}$	Standard mean chord (S.M.C.)	feet
C_{L_T}	Total trimmed lift coefficient $= \frac{L}{\frac{1}{2}\rho V^2 S}$	-
C_P	Pressure coefficient $= \frac{p-p_o}{q}$	-
C_M	Pitching moment coefficient $= \frac{M}{\frac{1}{2}\rho V^2 S\bar{c}}$	-
$h\bar{c}$	Position of centre of gravity on the S.M.C.	-
L_T	Total trimmed lift of aircraft	pounds
p	Static pressure at wing surface	pounds/feet^2
p_o	Free stream static pressure	pounds/feet^2
q	Free stream dynamic pressure $= \frac{1}{2}\rho V^2$	pounds/feet^2
S	Wing area	feet^2
s	Wing semispan $= \frac{b}{2}$	feet
V	True air speed	feet/second
x	Distance along wing chord line from leading edge	feet
y	Spanwise distance from aircraft centre line	feet
α	Wing incidence	degrees
η	Elevator angle	degrees
ρ	Air density	slugs/feet^2

Note that $\bar{c}$, not $\bar{\bar{c}}$ is used in this report.

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TABLE 1

Principal dimensions

Wing

Area (Apex definitions) S.	366.5 sq ft
Span. b	33 ft
Standard mean chord $\bar{c}$	11.11 ft
Aerodynamic mean chord $\bar{\bar{c}}$	14.35 ft
Aspect ratio. A	2.97
Sweepback of ¼ chord line (mean)	44.5 deg
Sweepback of leading edge	52.43 deg
Dihedral	-0.85 deg
Chord - on centre line	21.67 ft
at fuselage side	18.85 ft
tip	0.87 ft
Section	modified NACA 0010
Thickness-chord ratio	10%

For detailed description of wing planform and section see Ref. 1.

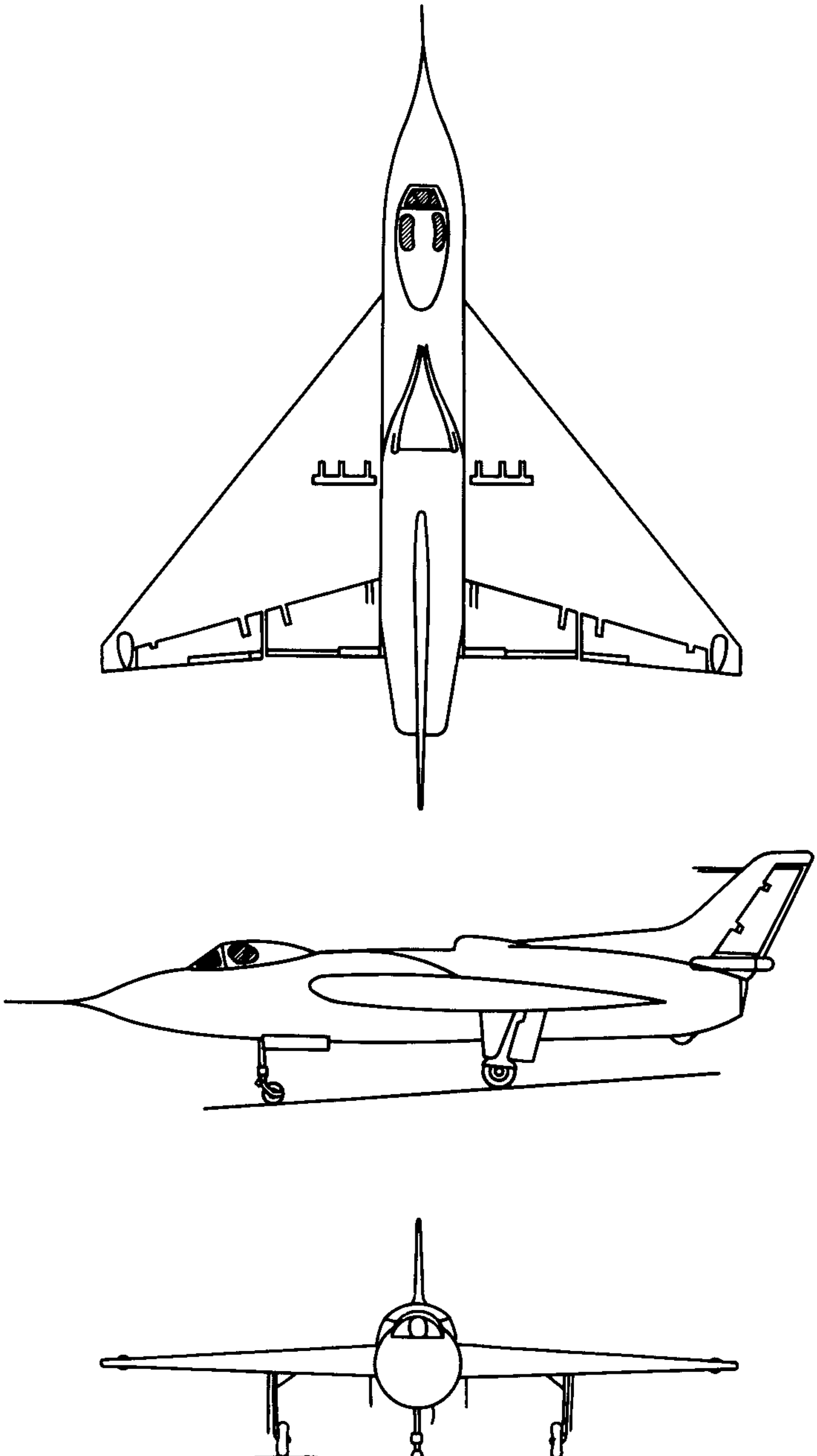


Fig.1 Avro 707 B general arrangement

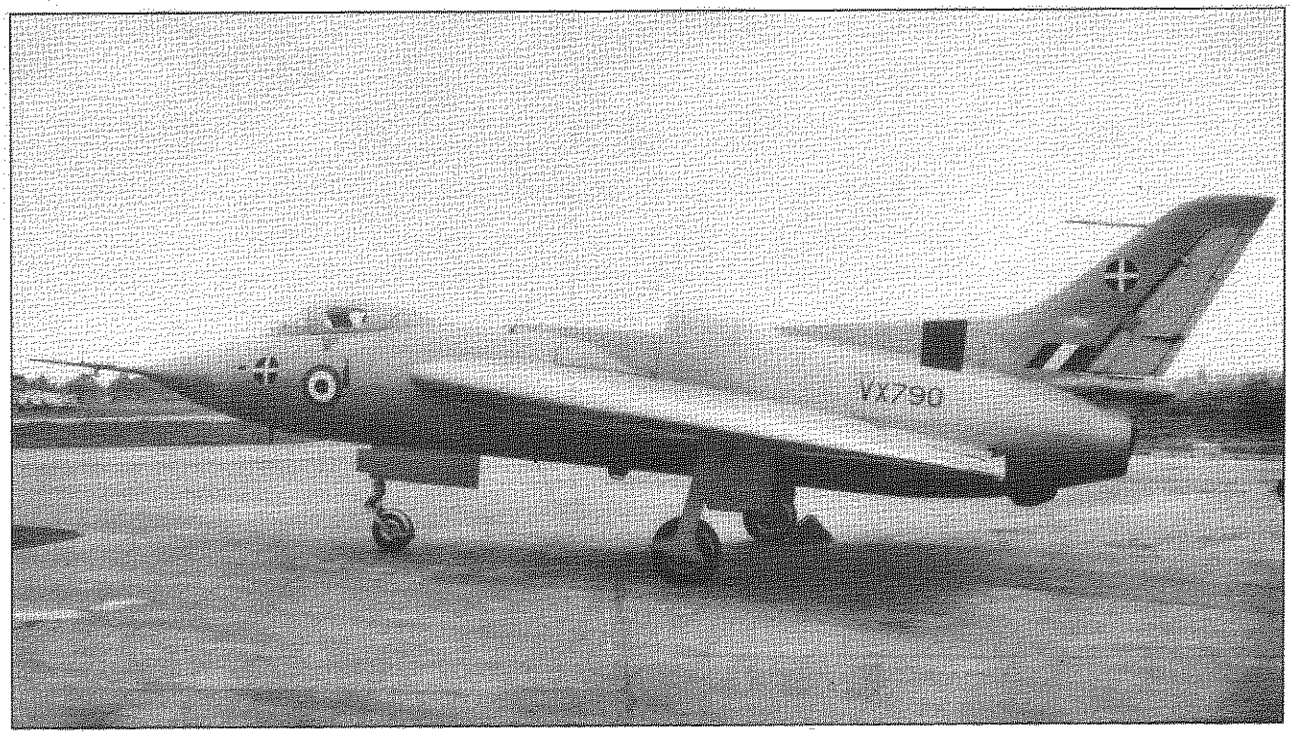


FIG.2

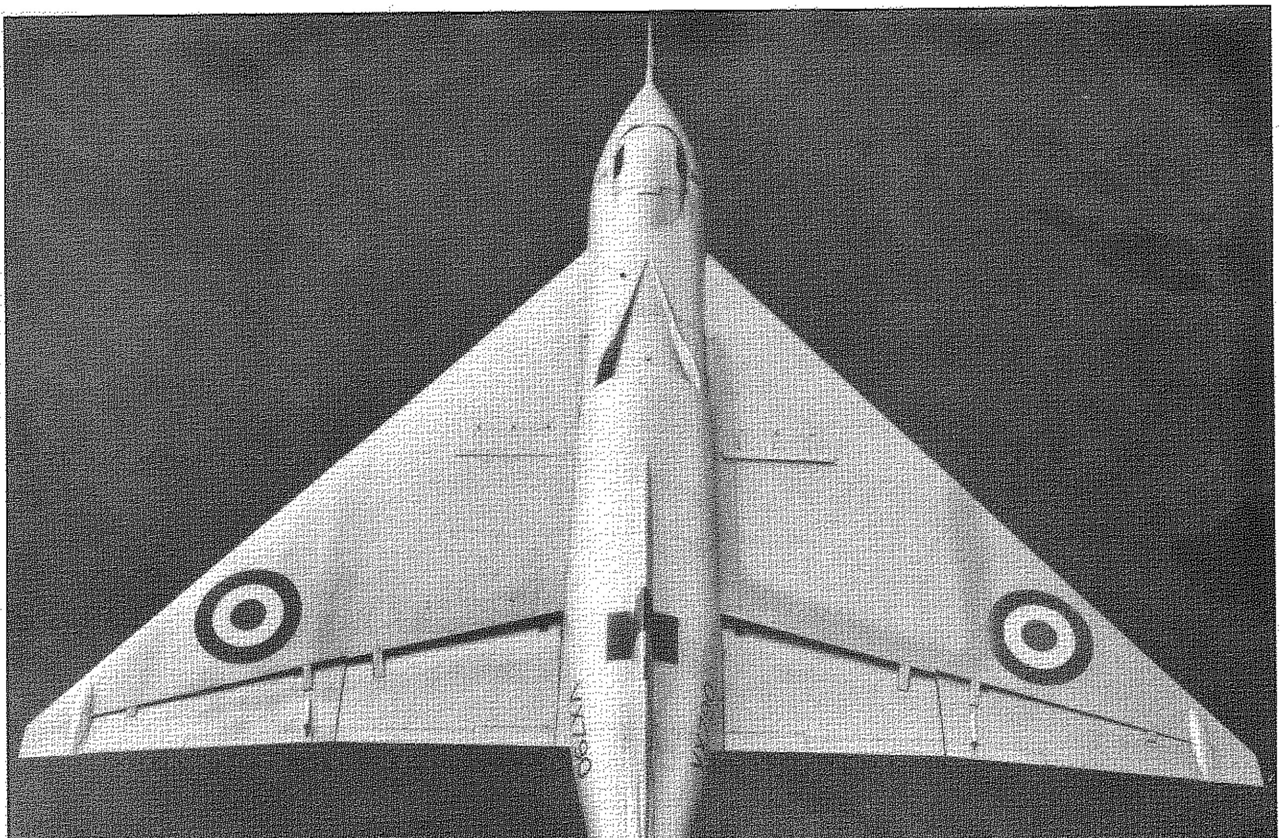


FIG.3

FIG.2 & 3. GENERAL VIEWS OF THE AIRCRAFT

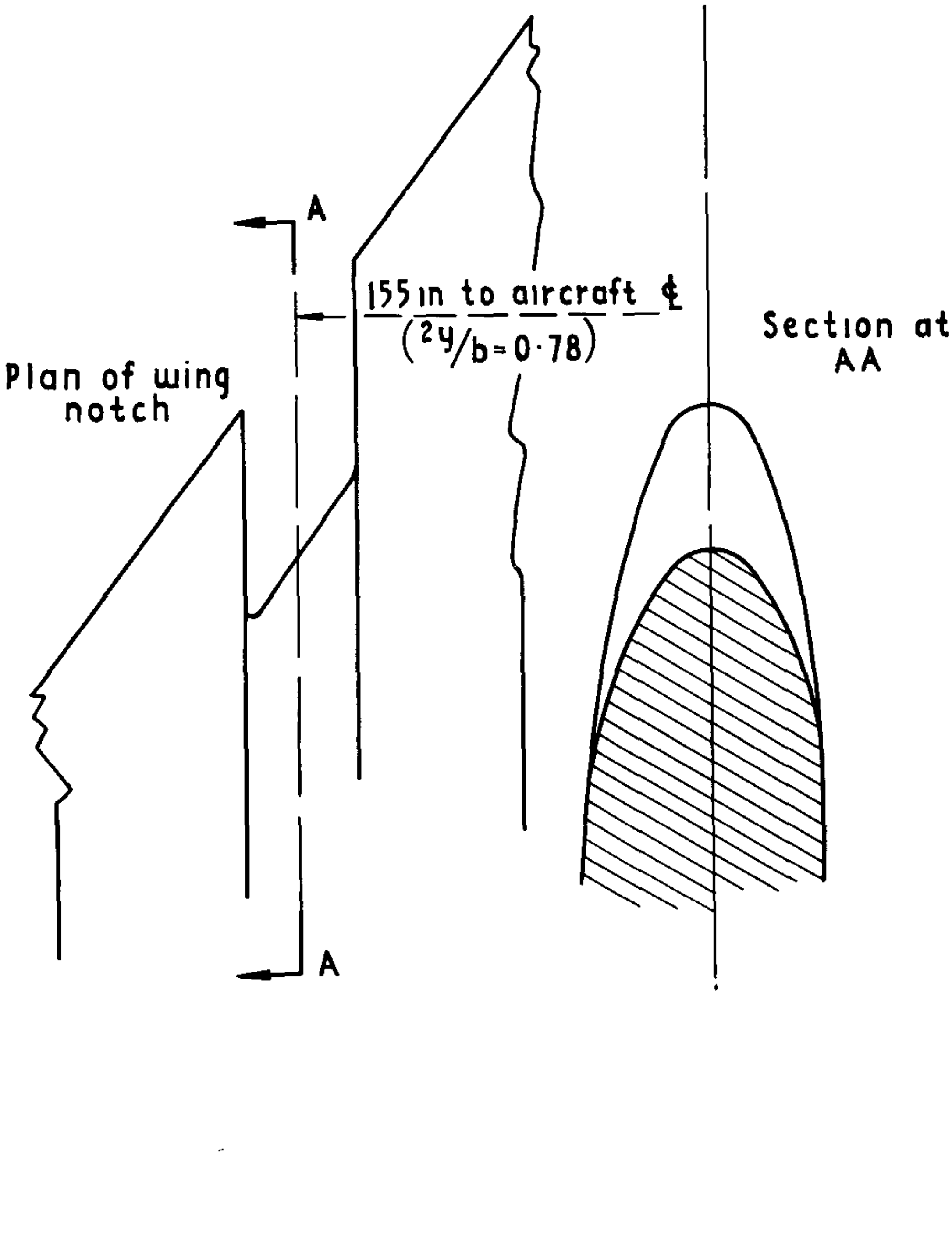


Fig. 4 Detail of wing notch
(scale ¼ full size)

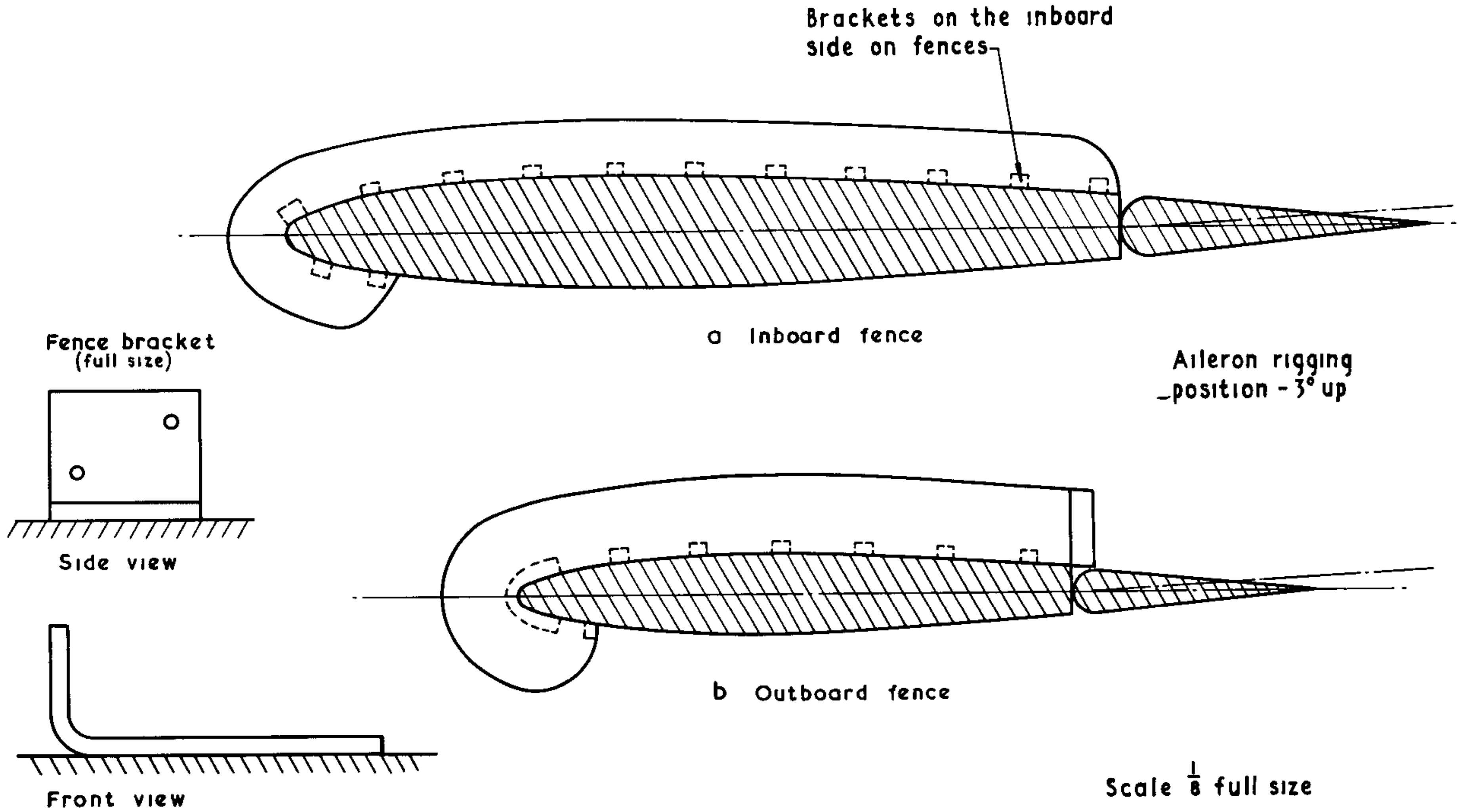


Fig.5 a & b Wing fences (spanwise positions shown in fig)

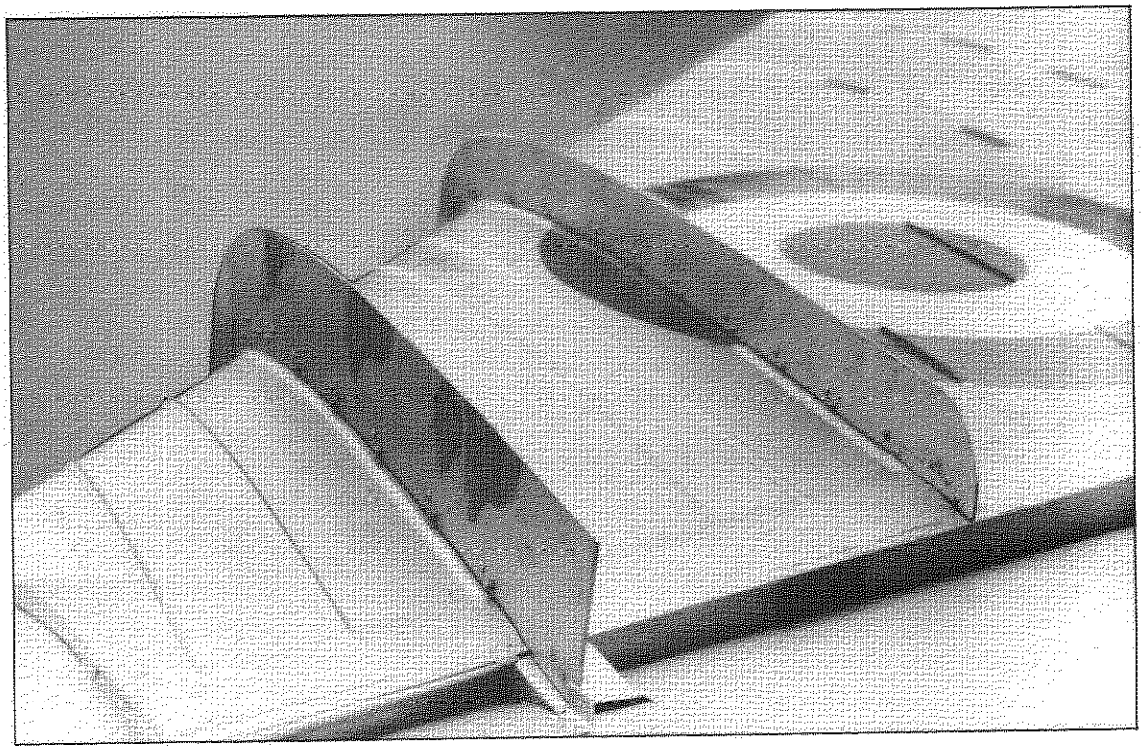


FIG.6

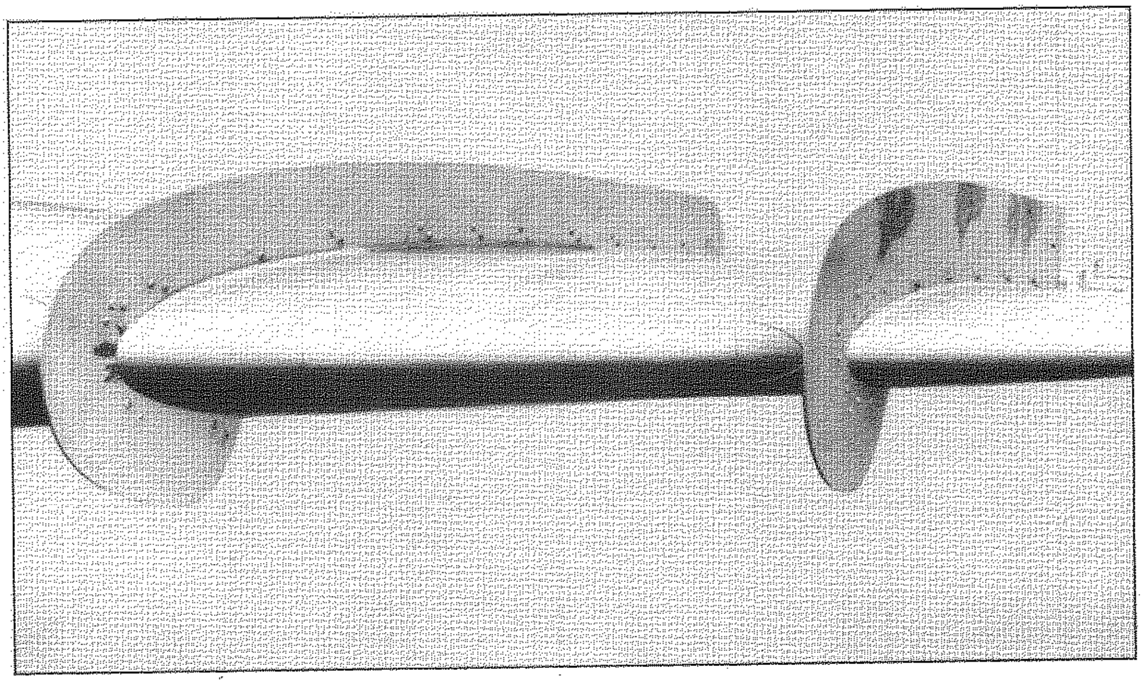


FIG.7

FIG.6 & 7. WING FENCES & SURFACE PRESSURE PLOTTING TUBES

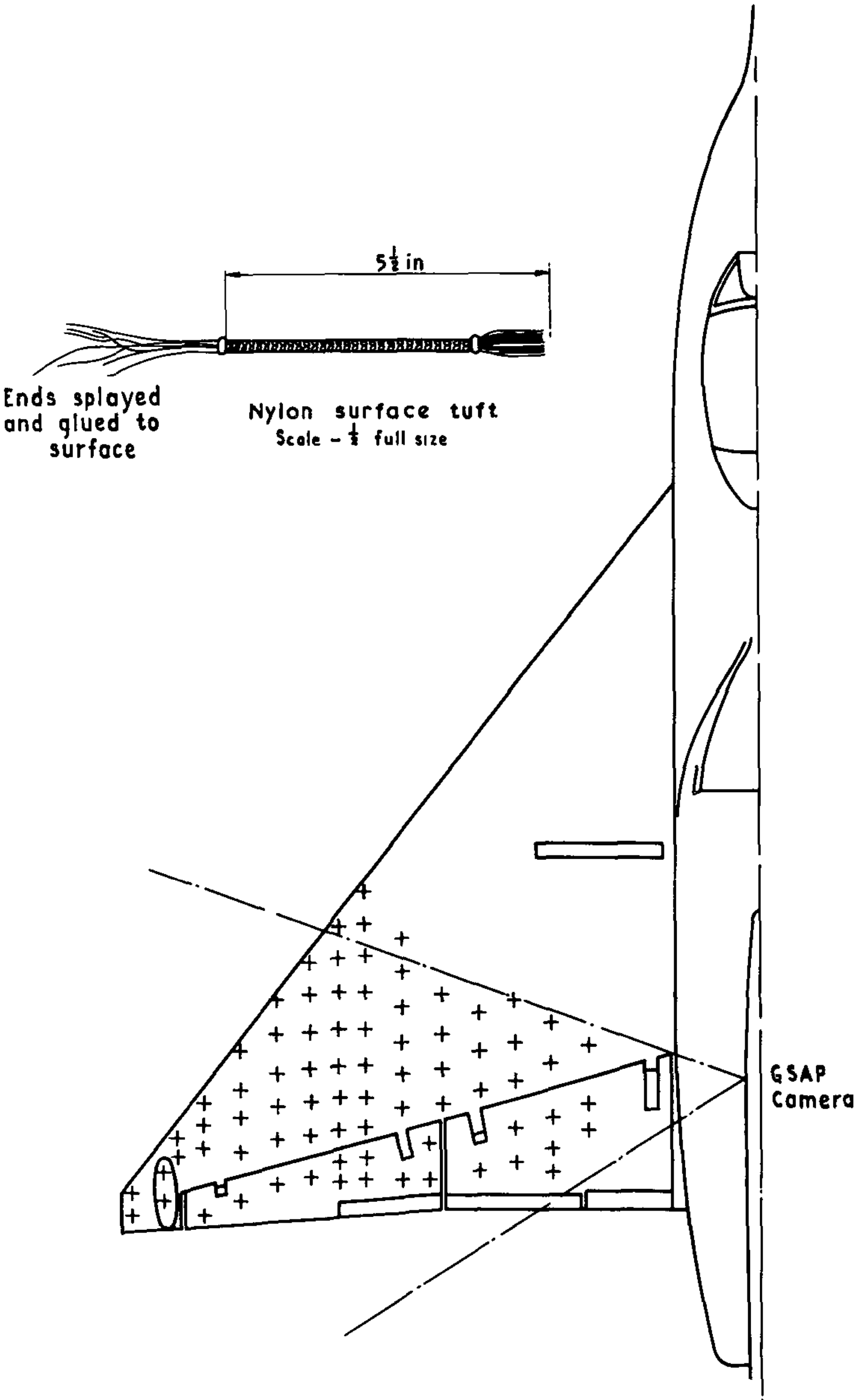


Fig.8 Position of surface tufts

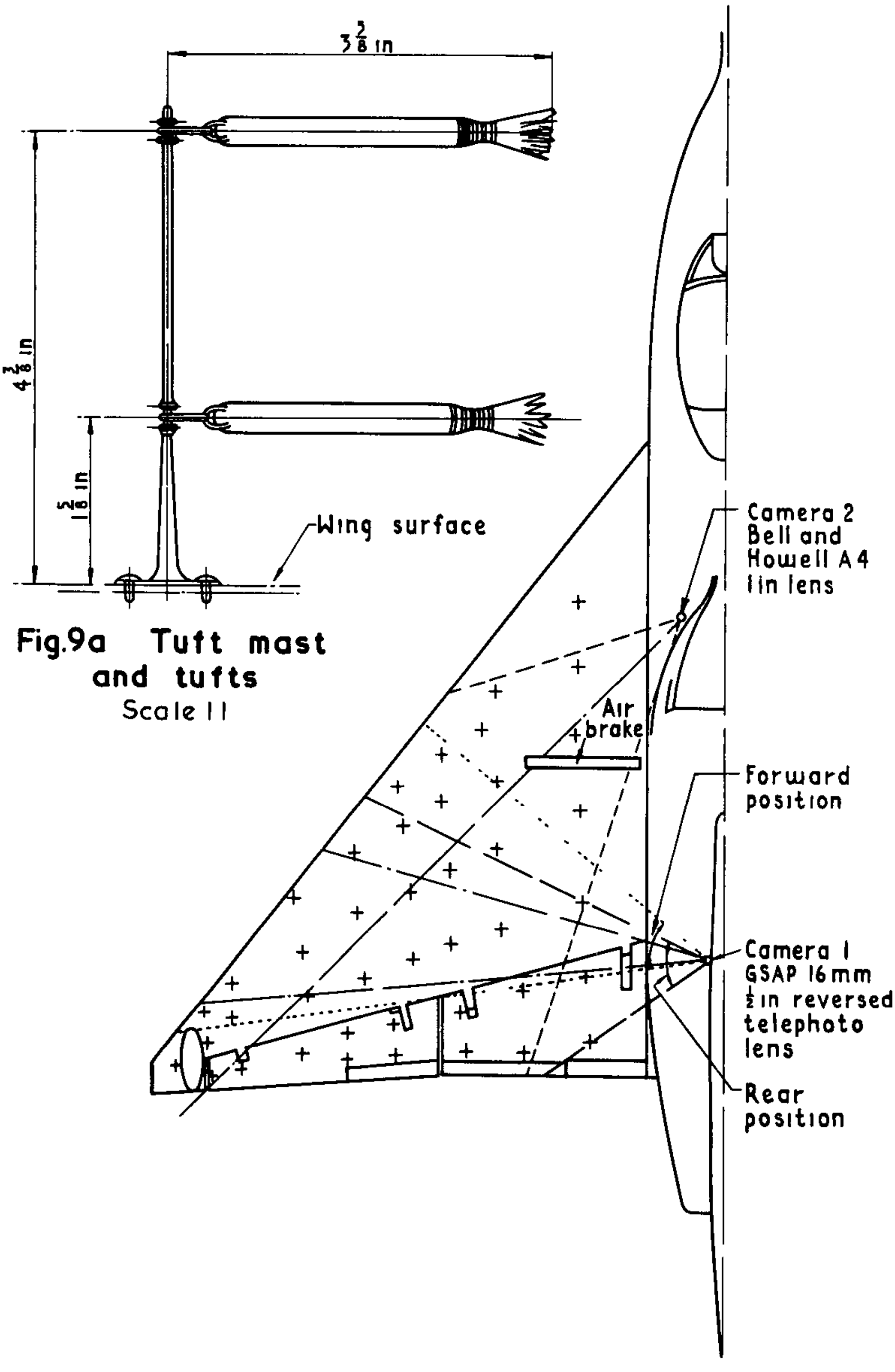


Fig.9a Tuft mast and tufts
Scale 1:1

Fig 9b Tuft mast positions and camera fields of view

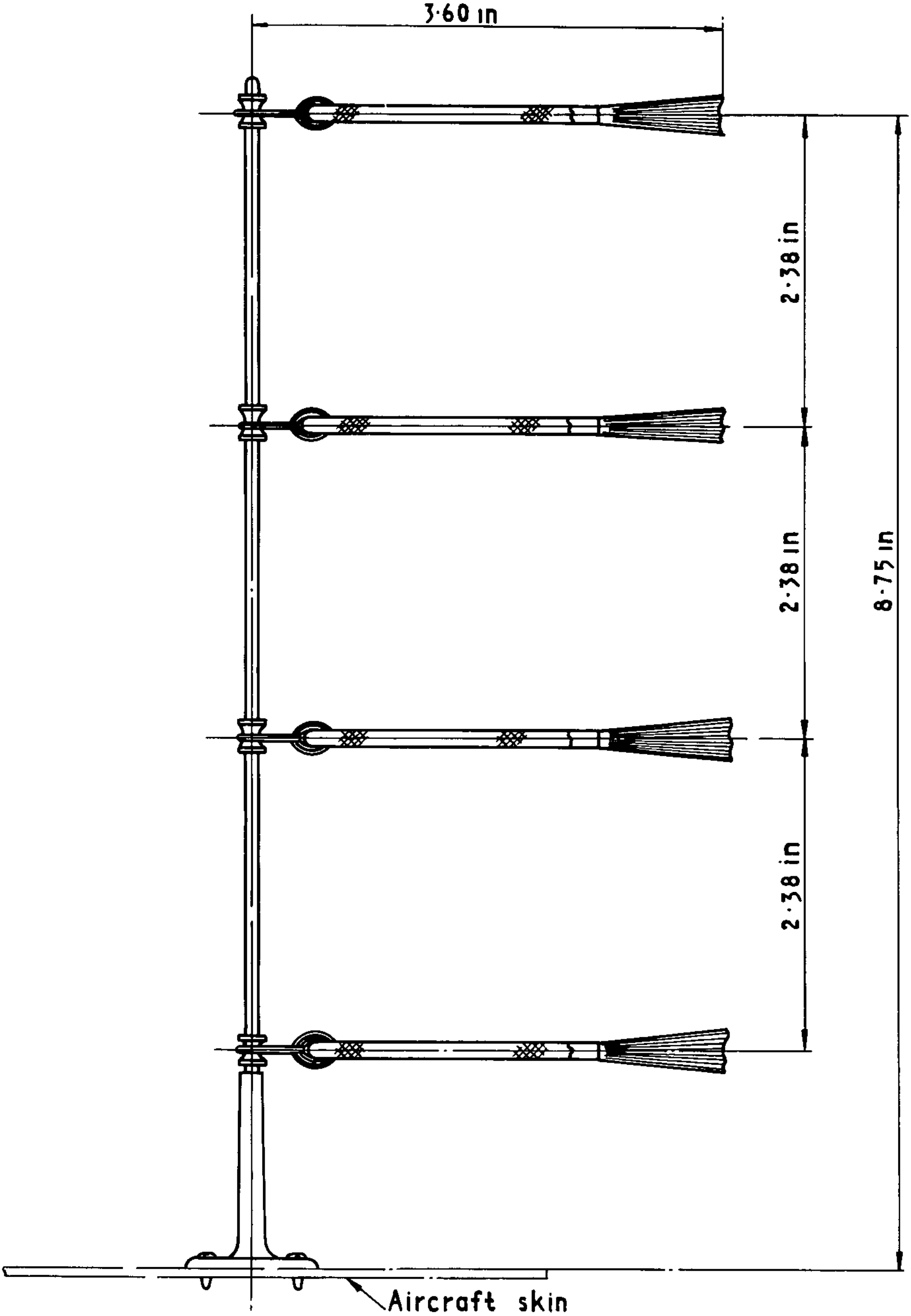


Fig.9 c Four tuft mast

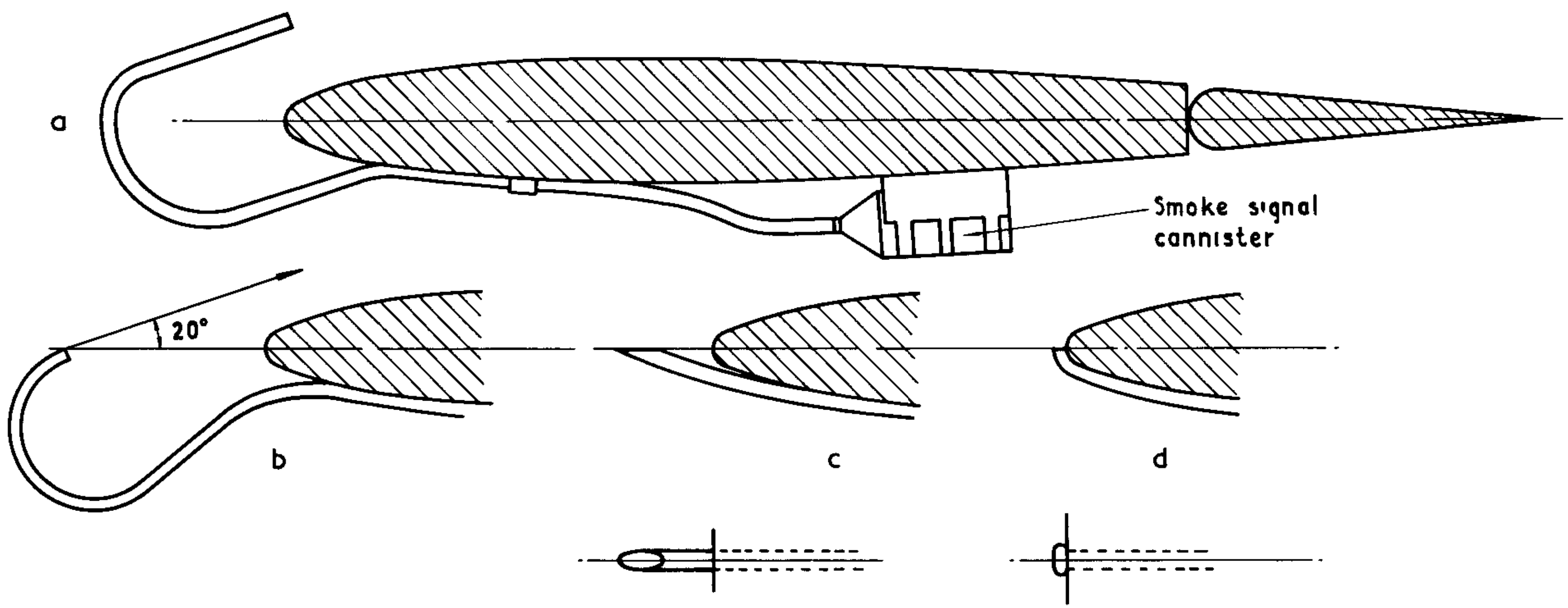


Fig.10 a–d Smoke generator and nozzles

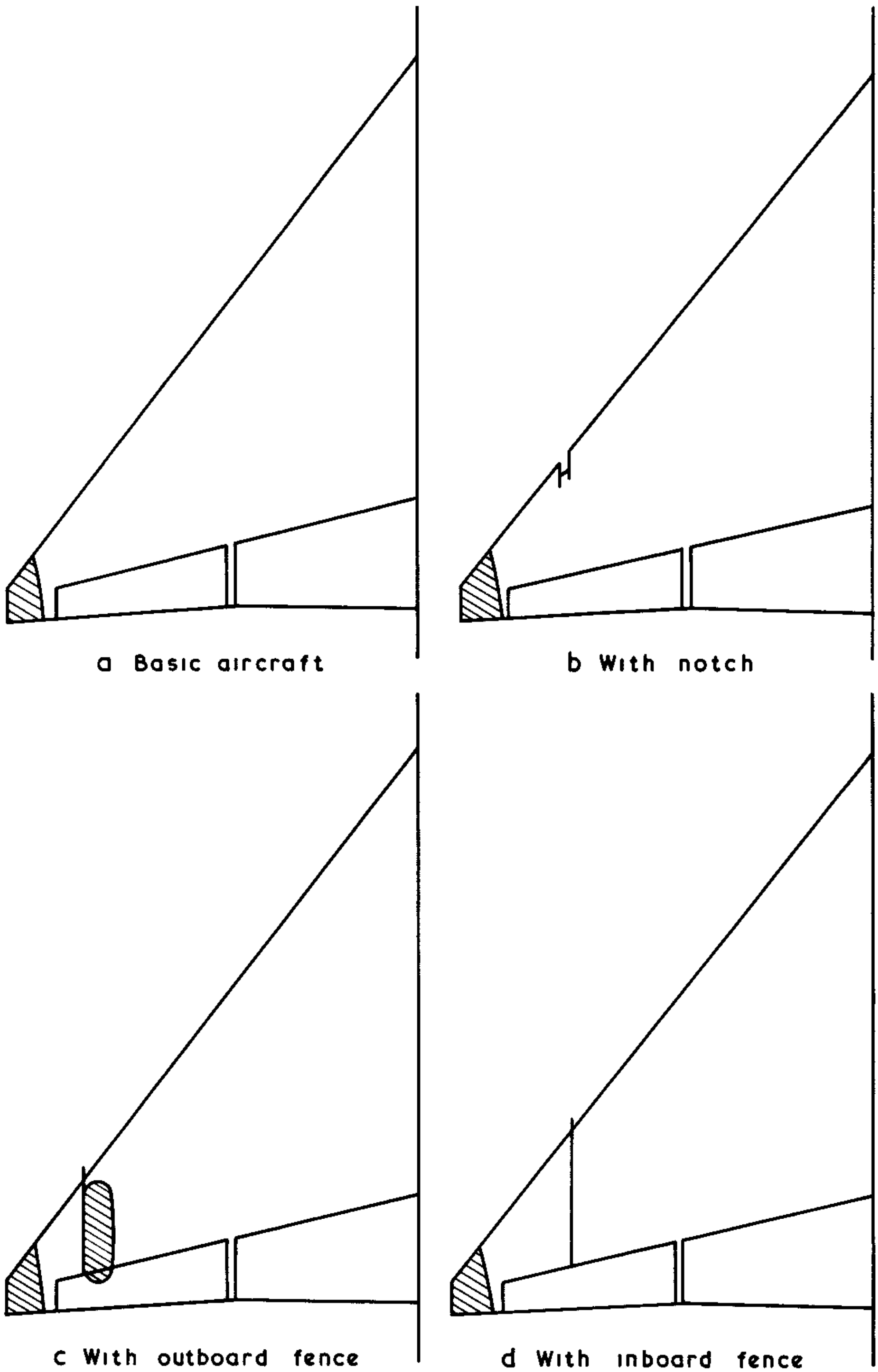


Fig.11 a-d Development of flow separation.
Trimmed $C_L = 0.40$

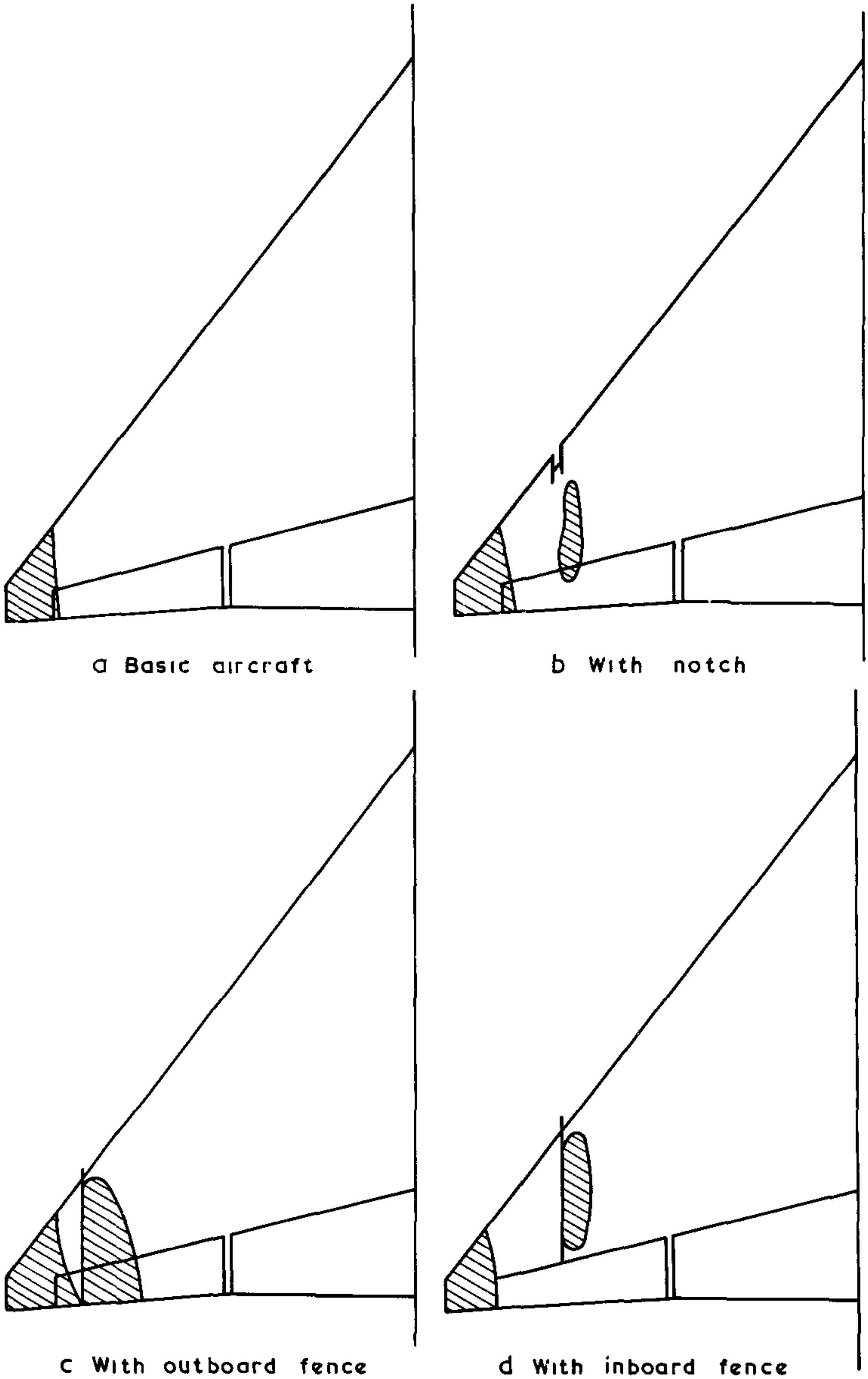


Fig.12a-d Development of flow separation.
Trimmed $C_L = 0{\cdot}50$

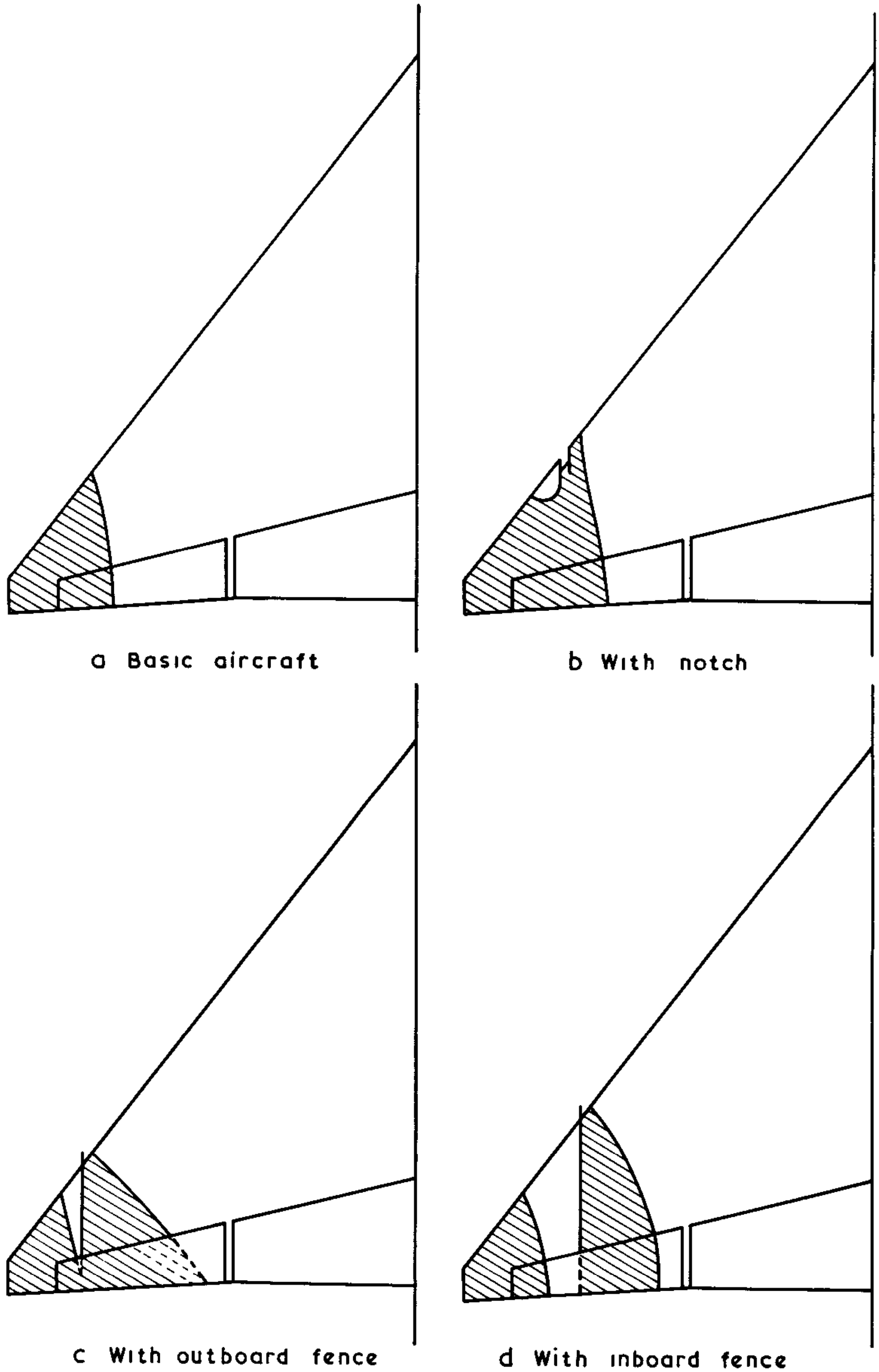
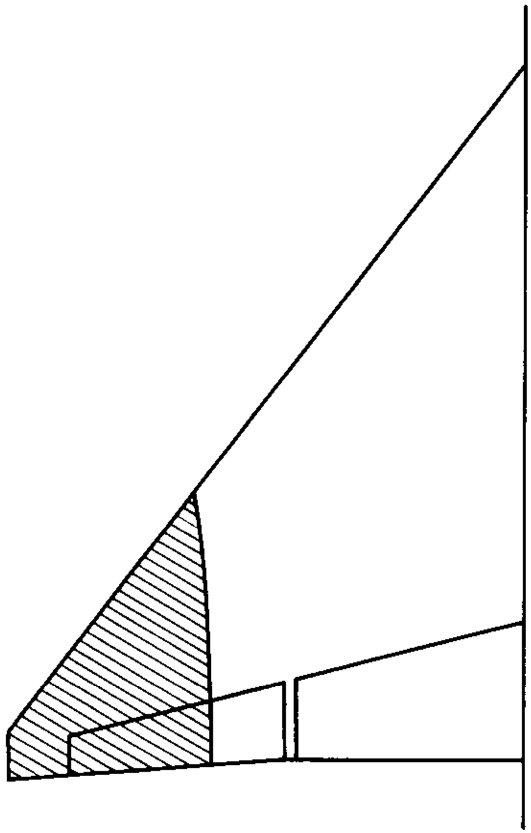
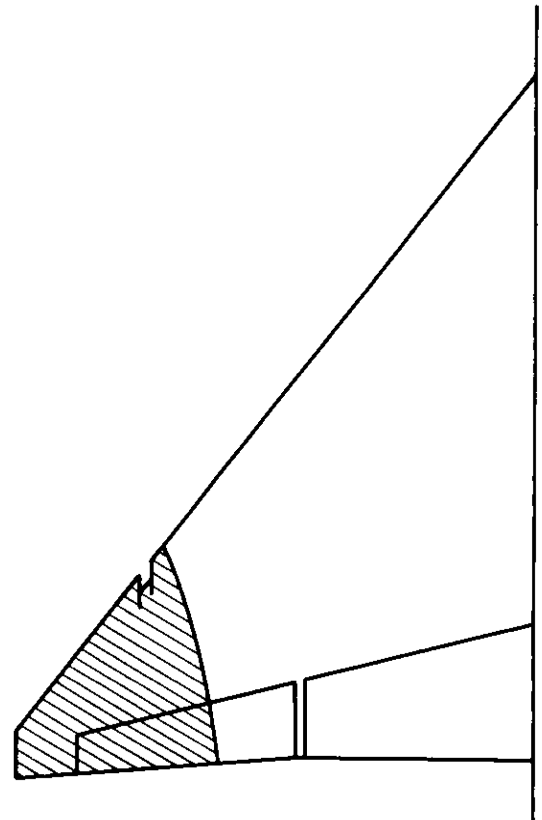


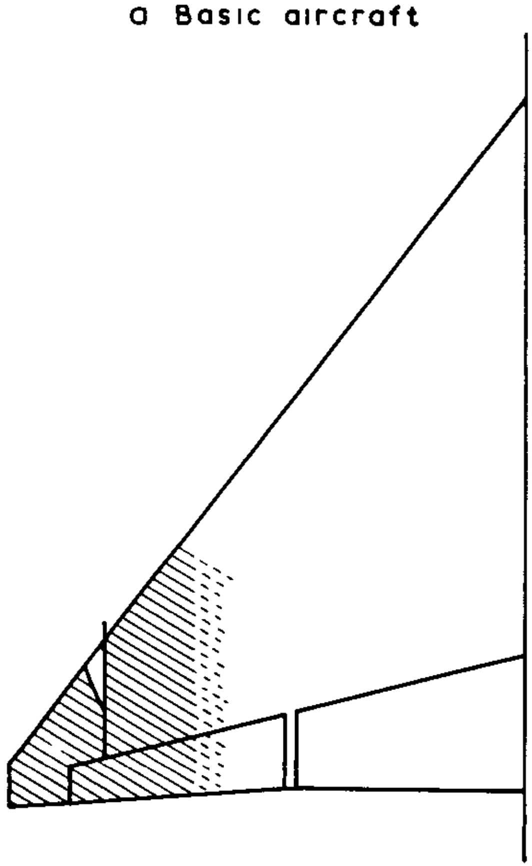
Fig.13 a-d Development of flow separation.
Trimmed $C_L = 0{\cdot}60$



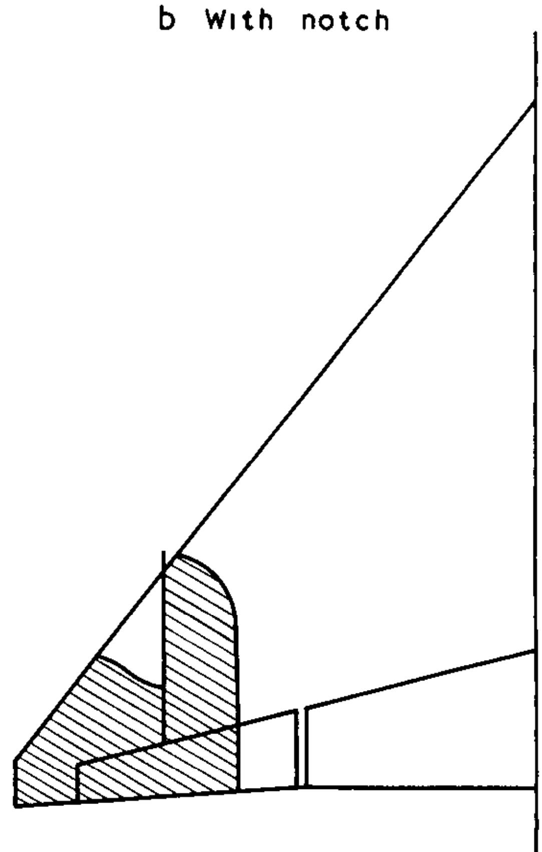
a Basic aircraft



b With notch



c With outboard fence



d With inboard fence

Fig.14 a-d Development of flow separation.
Trimmed $C_L = 0{\cdot}70$

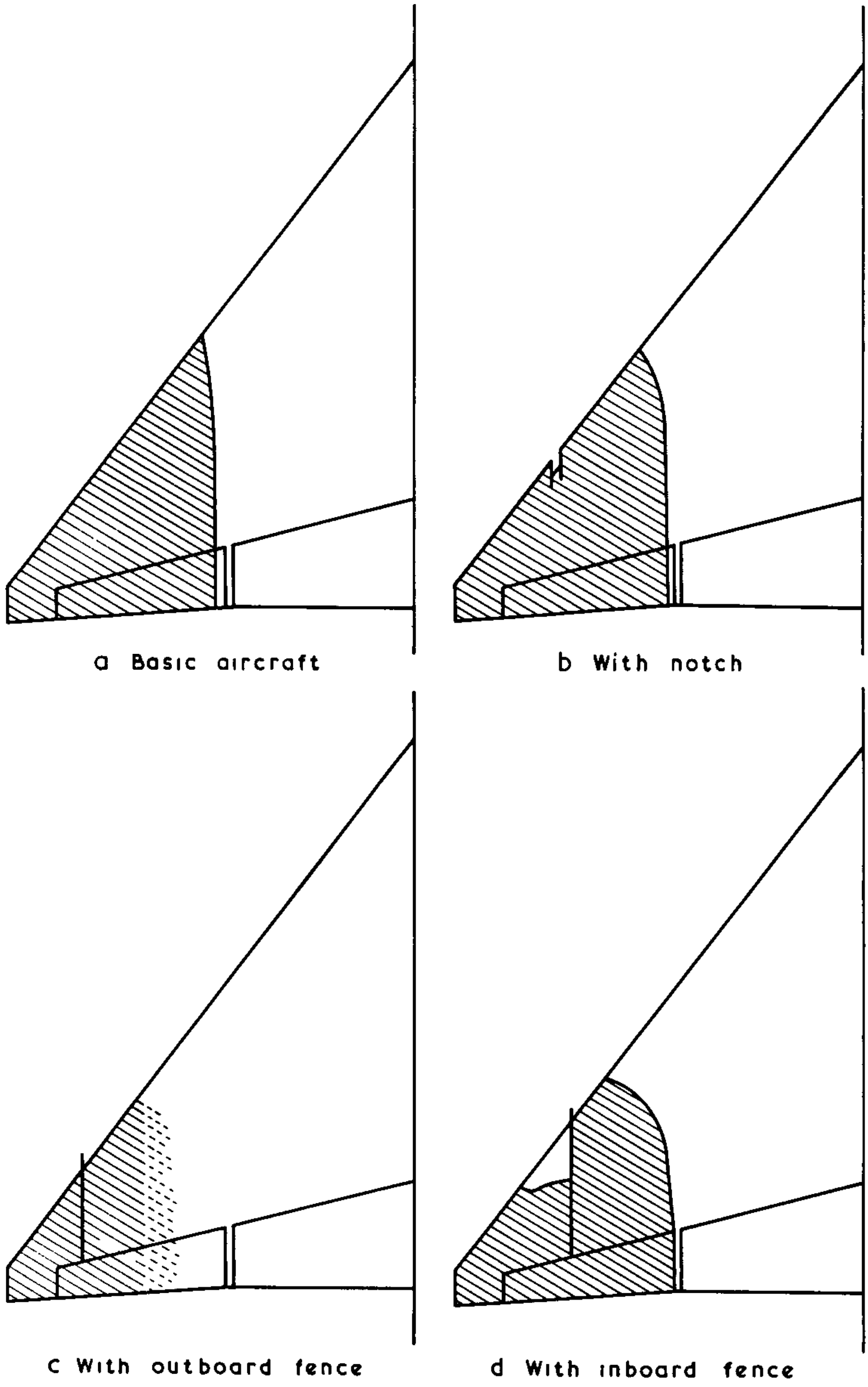


Fig.15 a–d Development of flow separation. Trimmed C_L=0·80

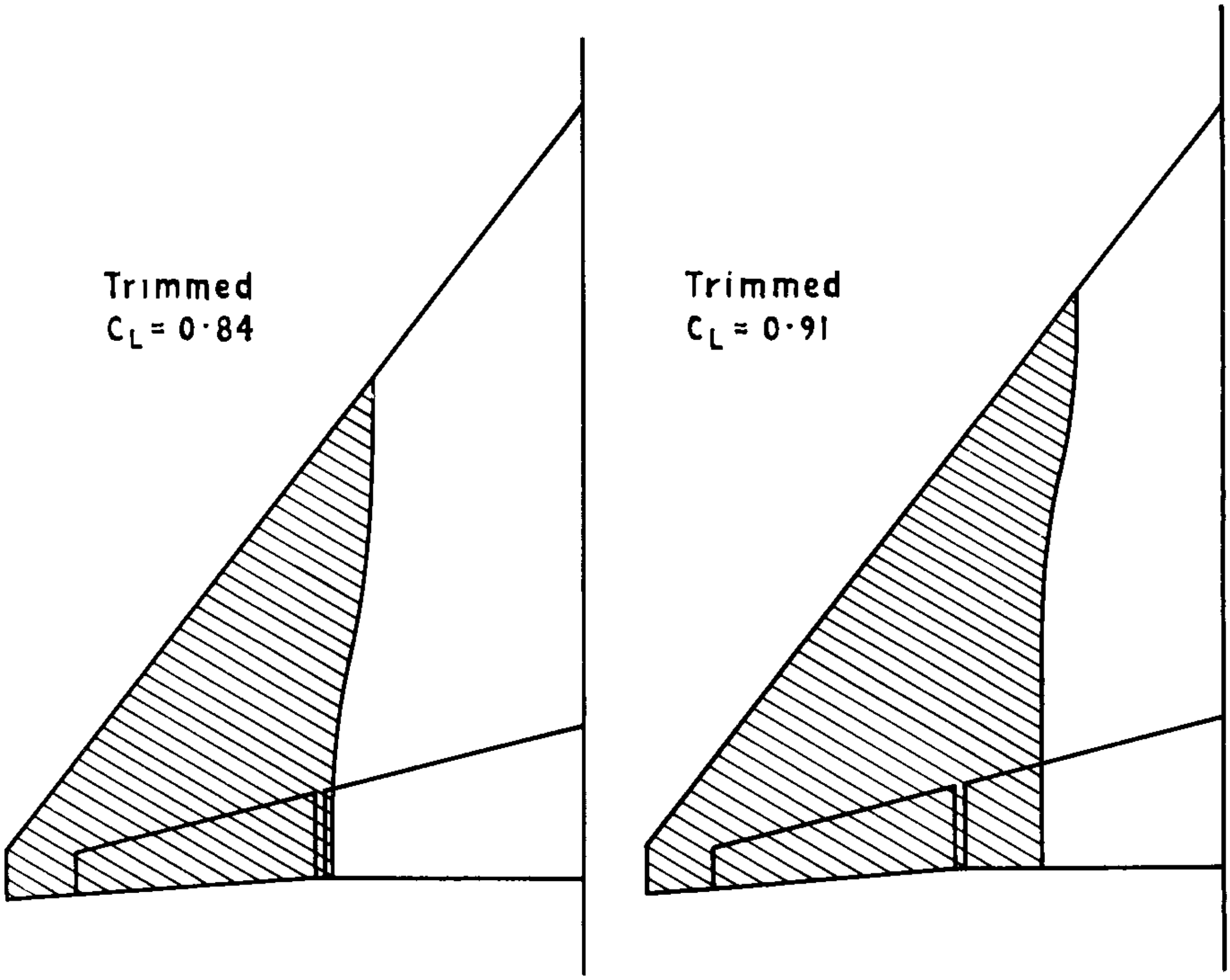


Fig 16 Areas of separated flow on the basic wing at high lift coefficients

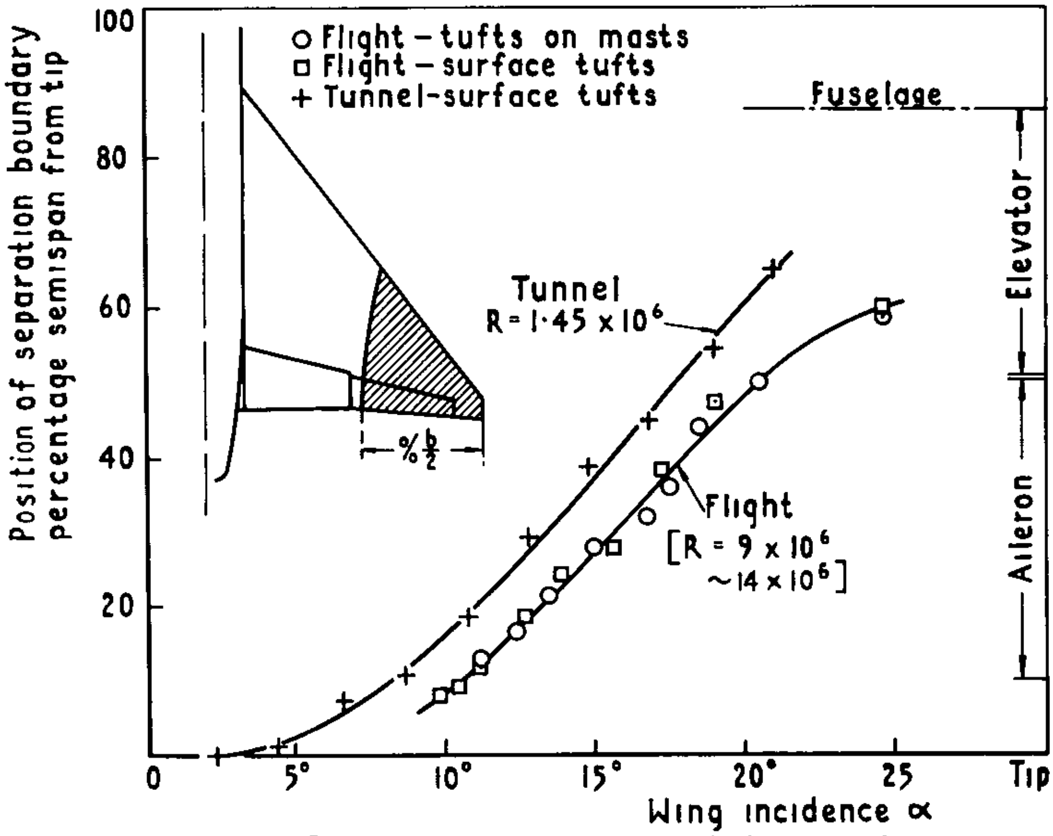


Fig.17 Comparison of flight and tunnel measurements of the position of the separation boundary at the trailing edge

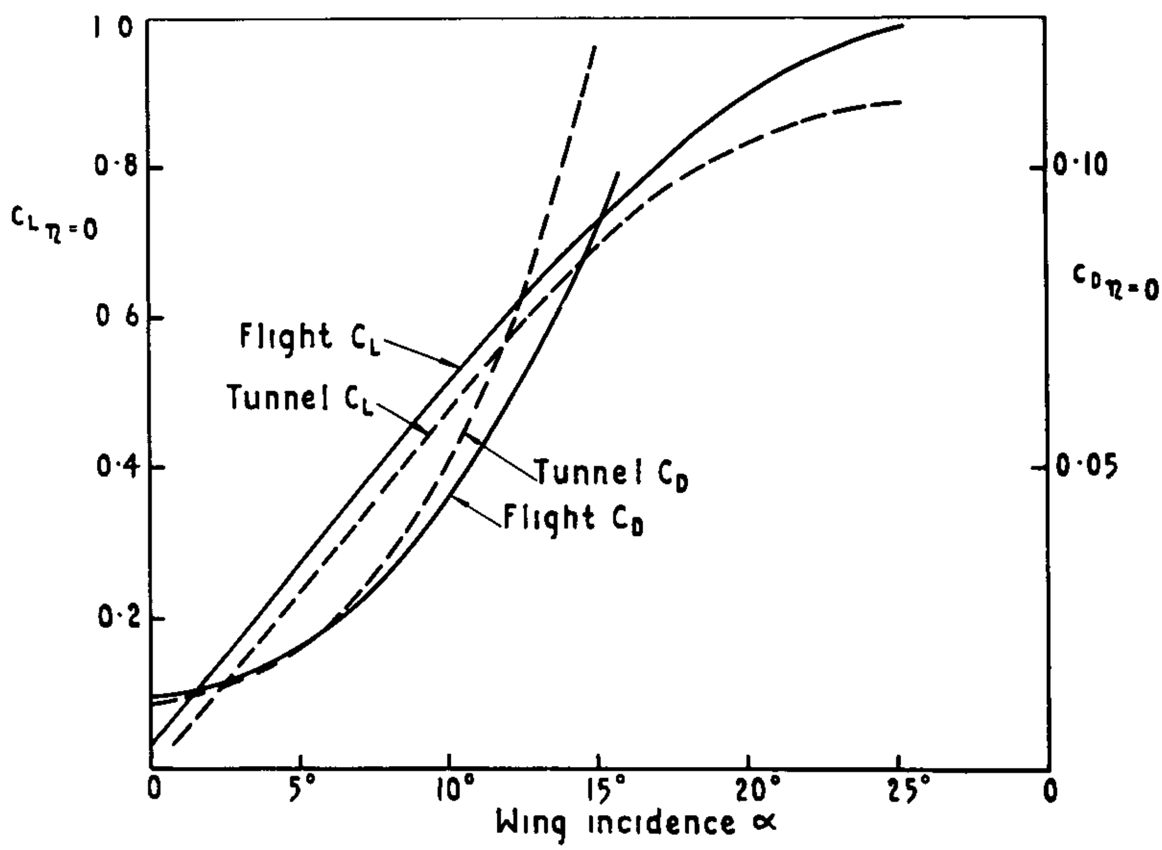


Fig.18 Comparison of flight and tunnel measurements of lift and drag

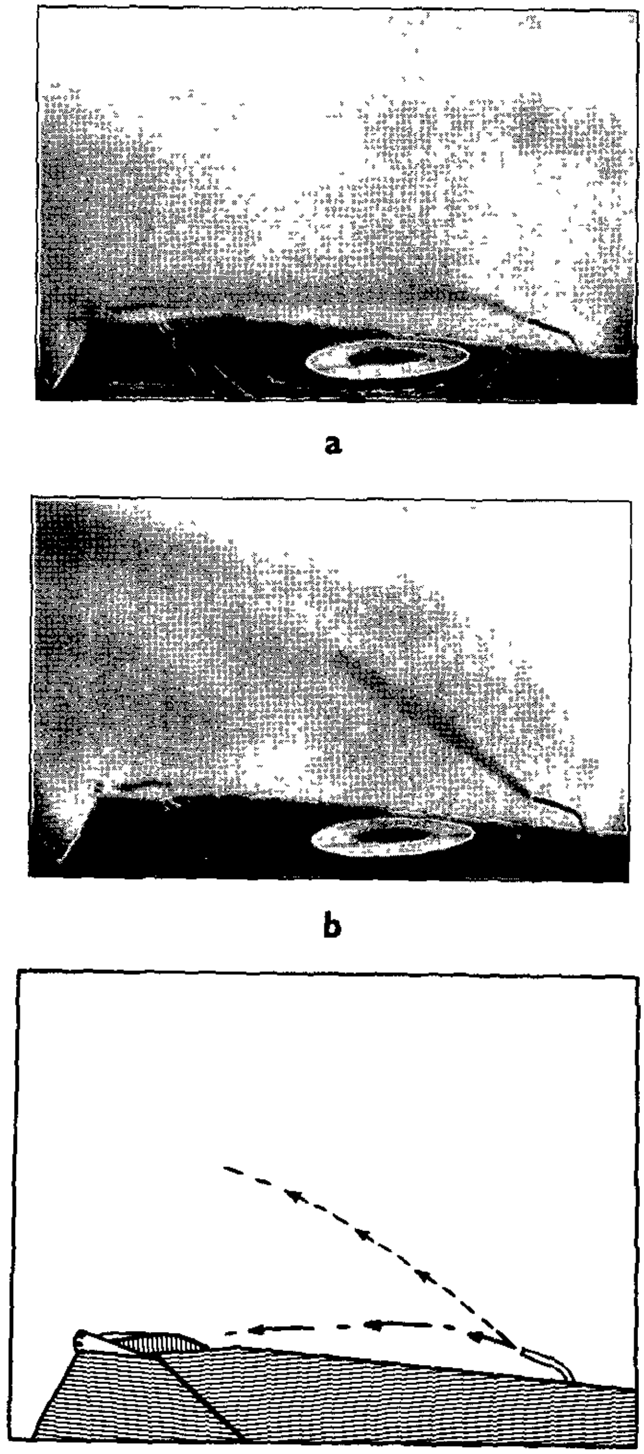
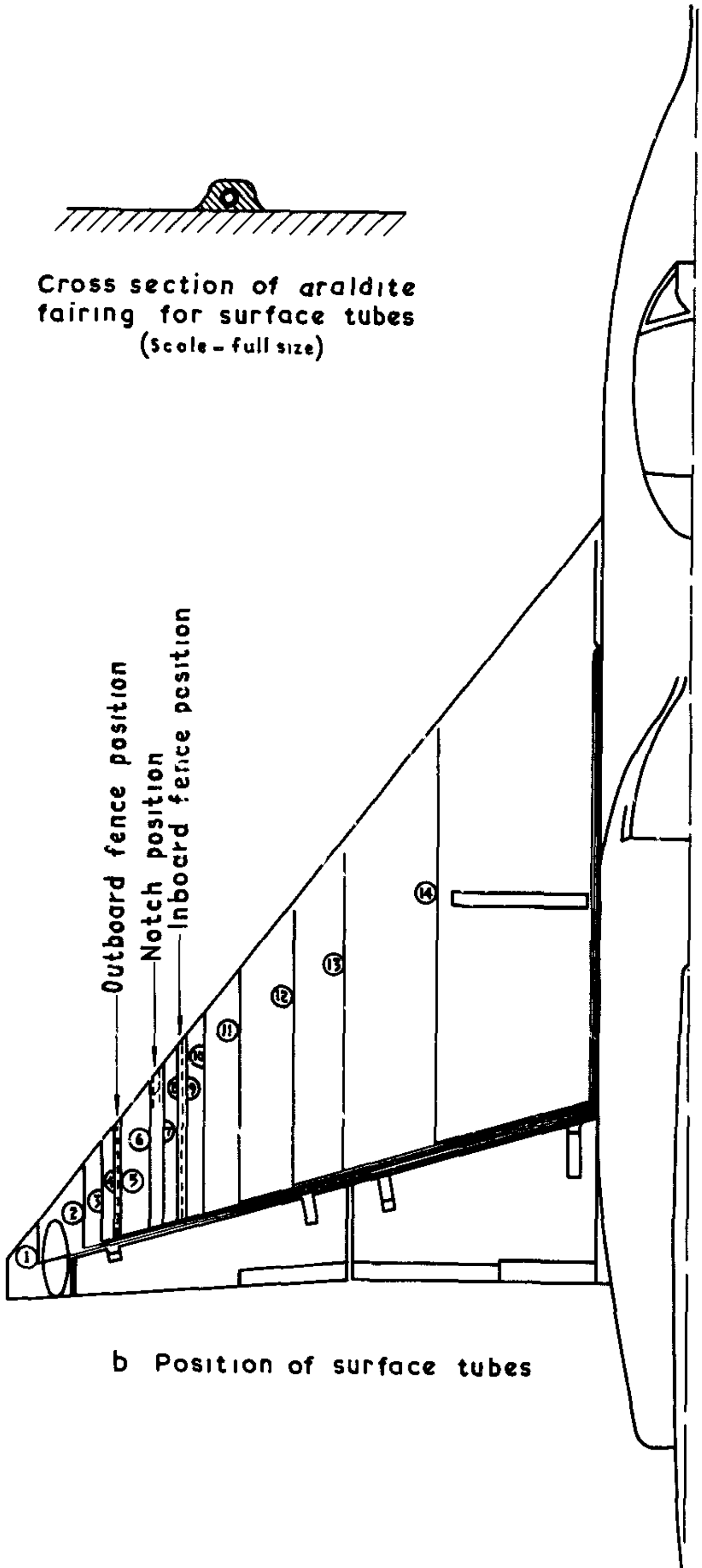


FIG.19. COMPARISON OF SMOKE PICTURES FOR ATTACHED AND SEPARATED FLOW



a Cross section of araldite fairing for surface tubes (Scale – full size)

b Position of surface tubes

Fig. 20 a & b Layout of surface pressure plotting tubes

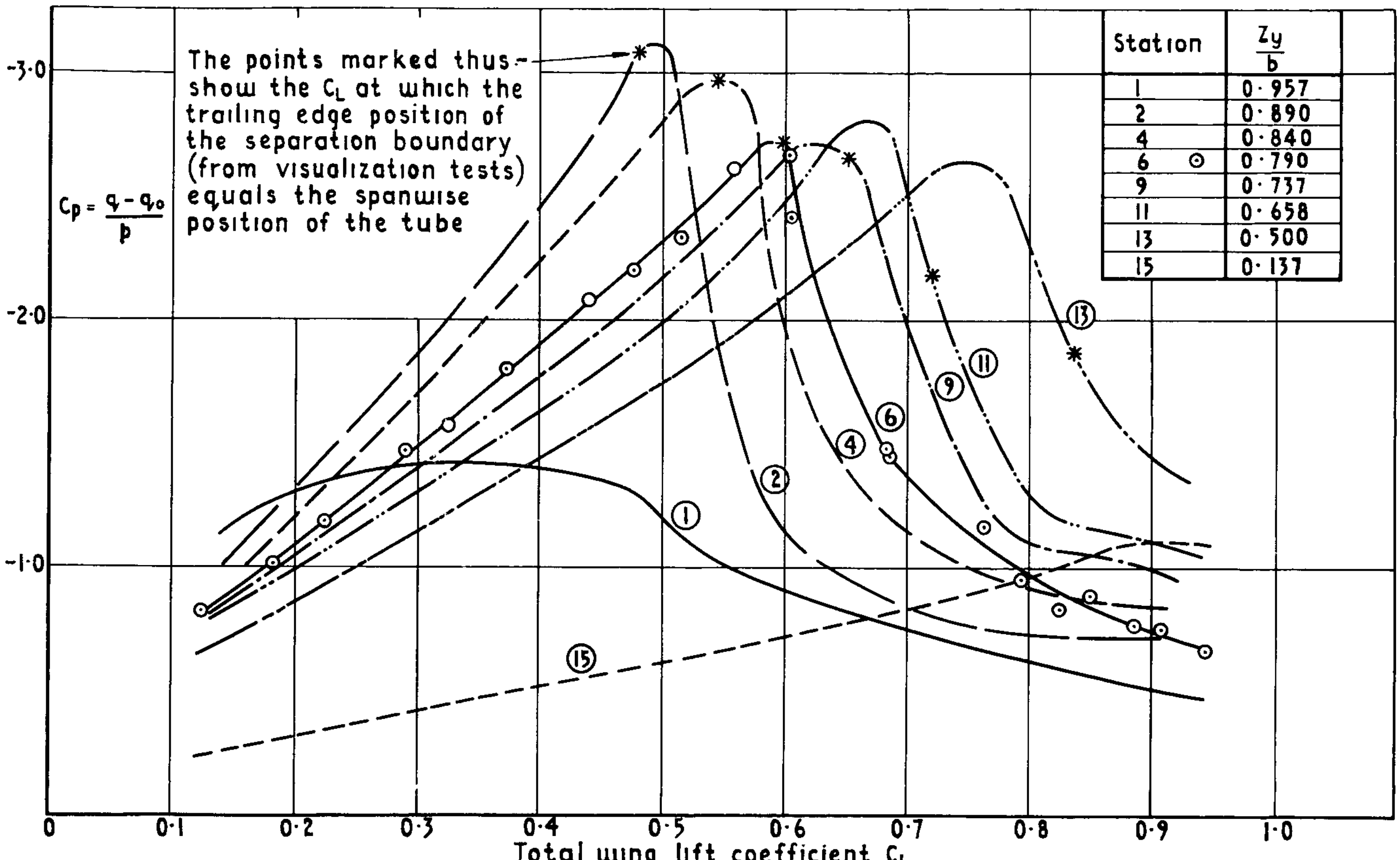


Fig.21 Variation of C_p with C_L at different spanwise positions. 7% chord basic wing

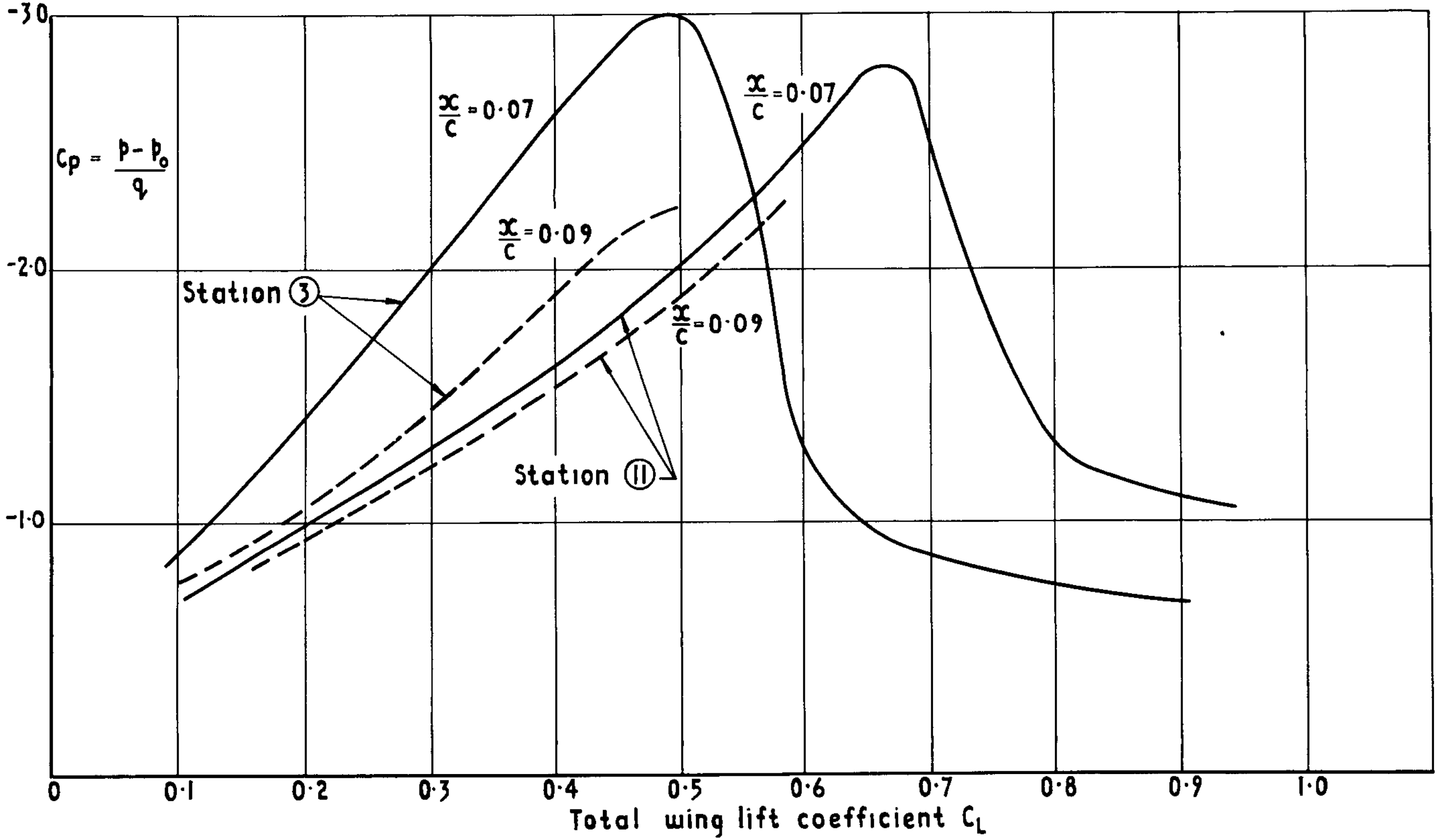


Fig.22 Comparison of the effect of changing the chordwise position of the measuring hole at stations (3) and (11)

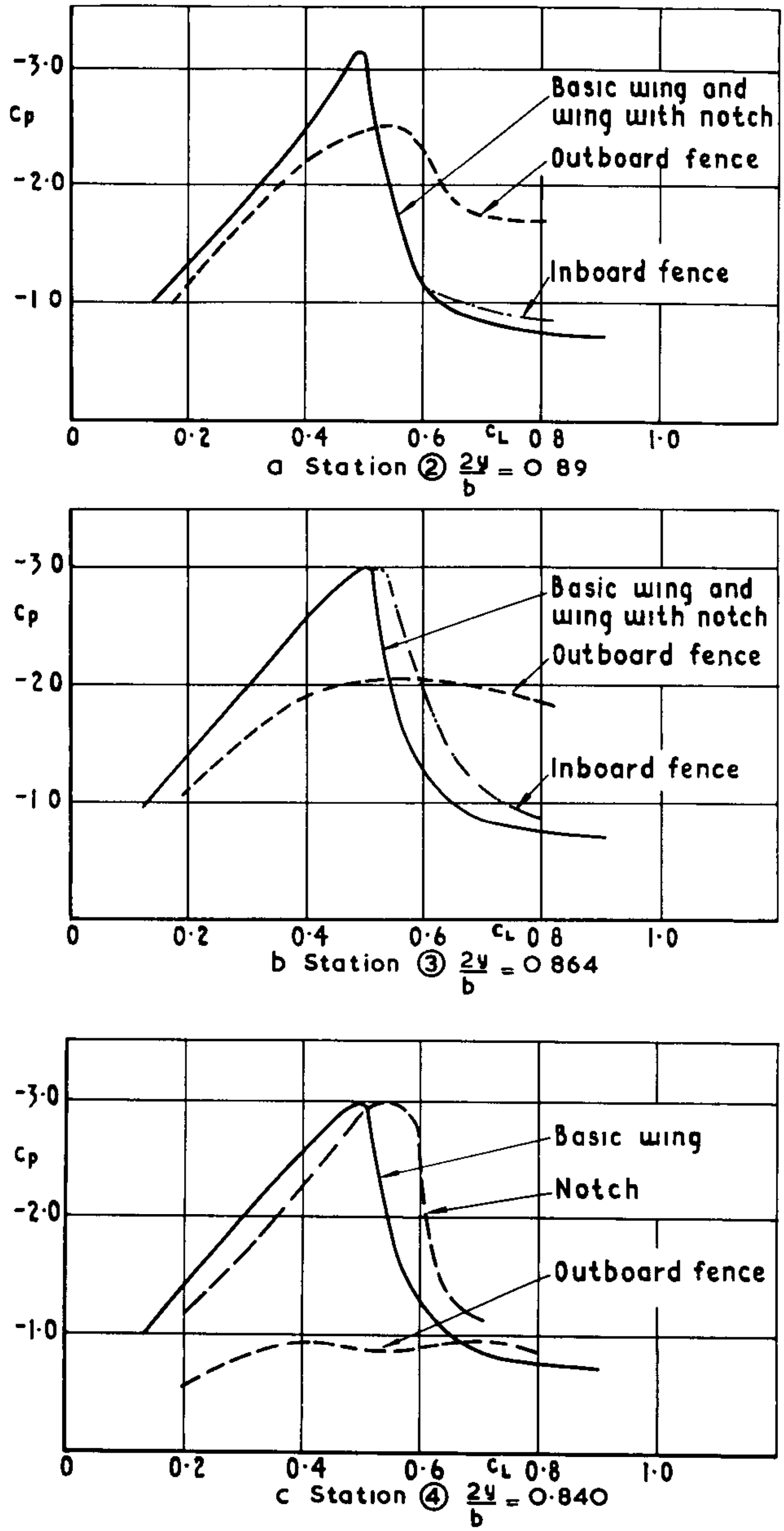


Fig. 23 a-c The effect of wing fences and a notch on the pressure at various spanwise stations. 7% chord

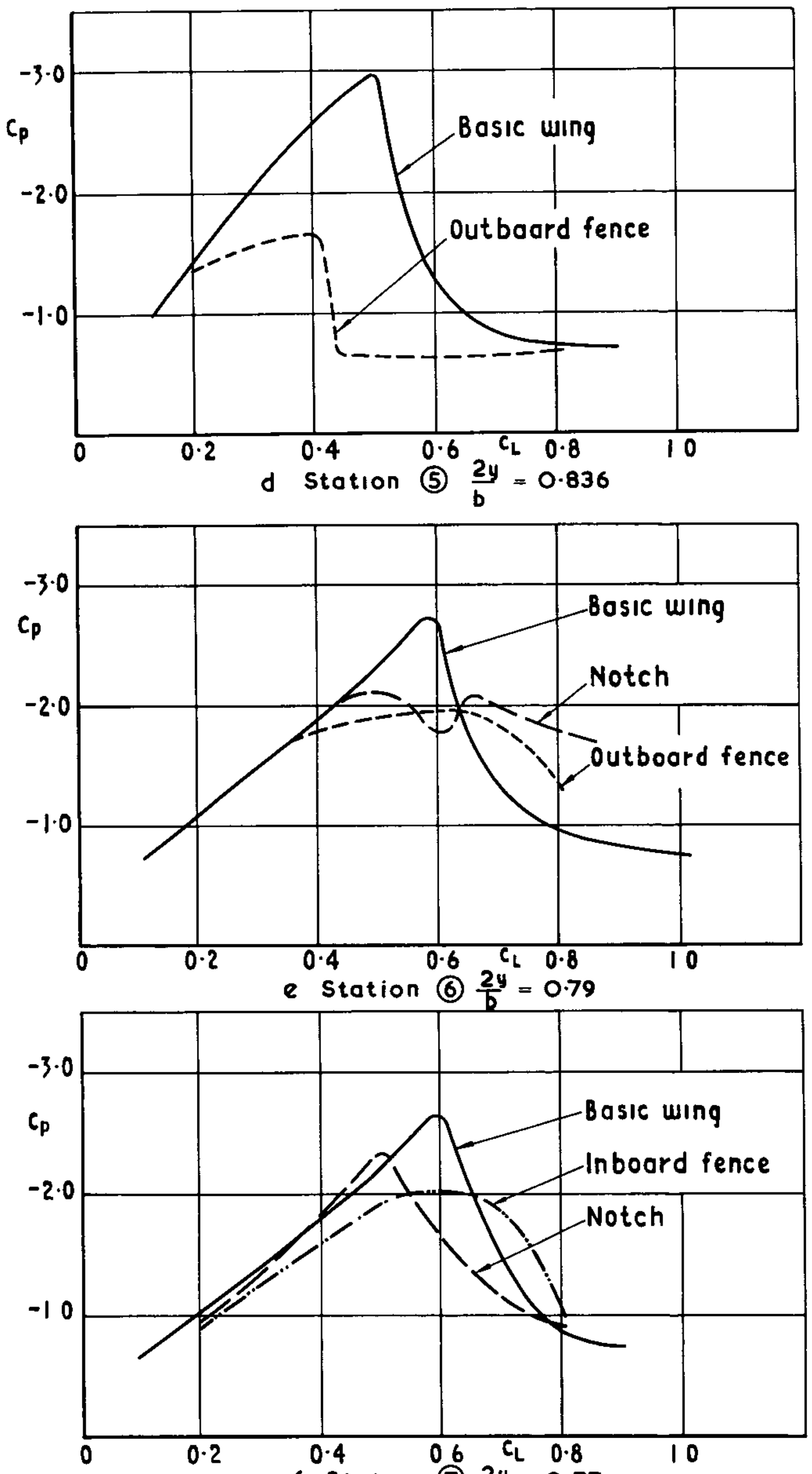


Fig. 23 d-f The effect of wing fences and a notch on the pressure at various spanwise station. 7% chord

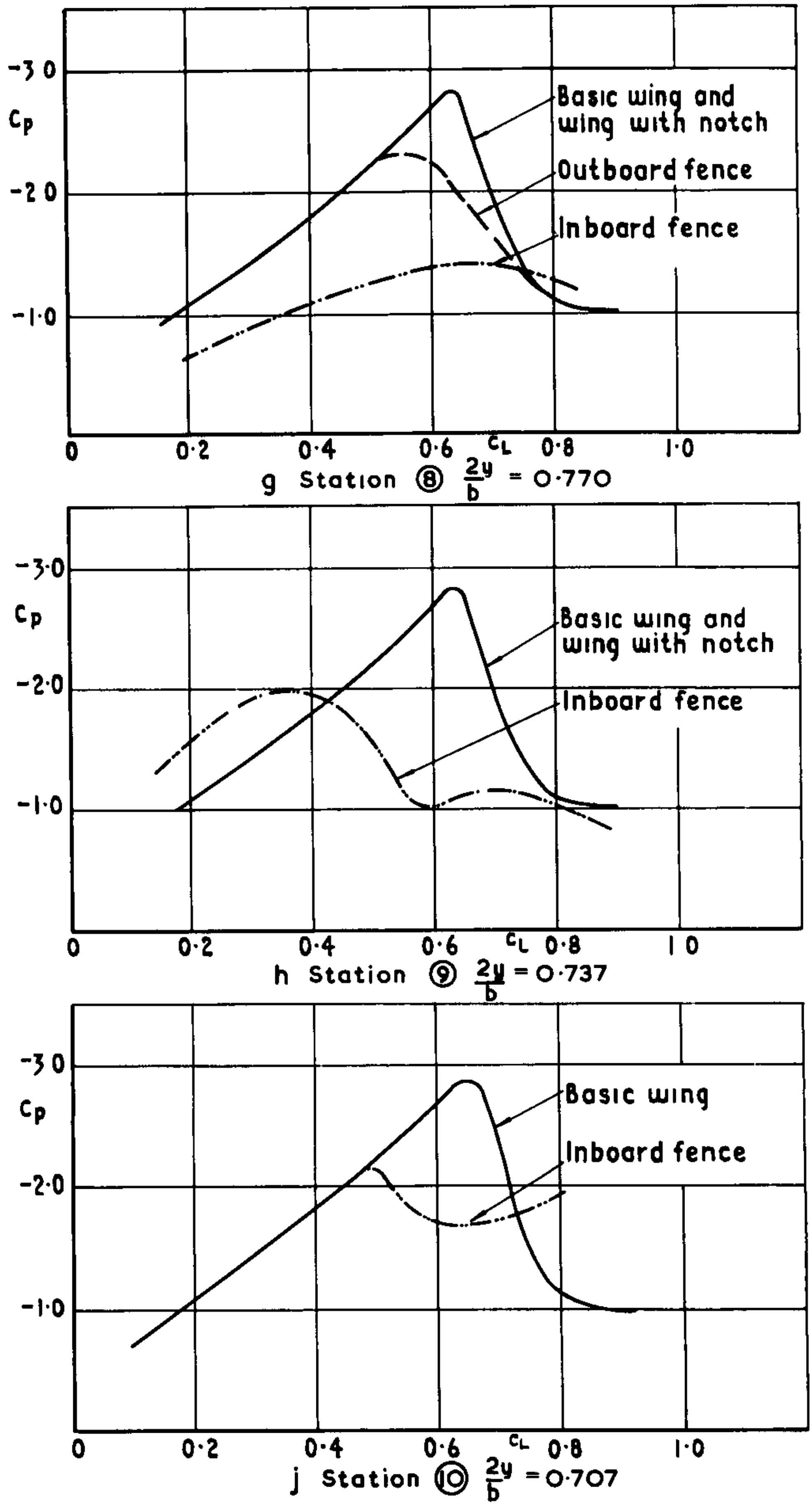


Fig. 23 g-j The effect of wing fences and a notch on the pressure at various spanwise station. 7% chord

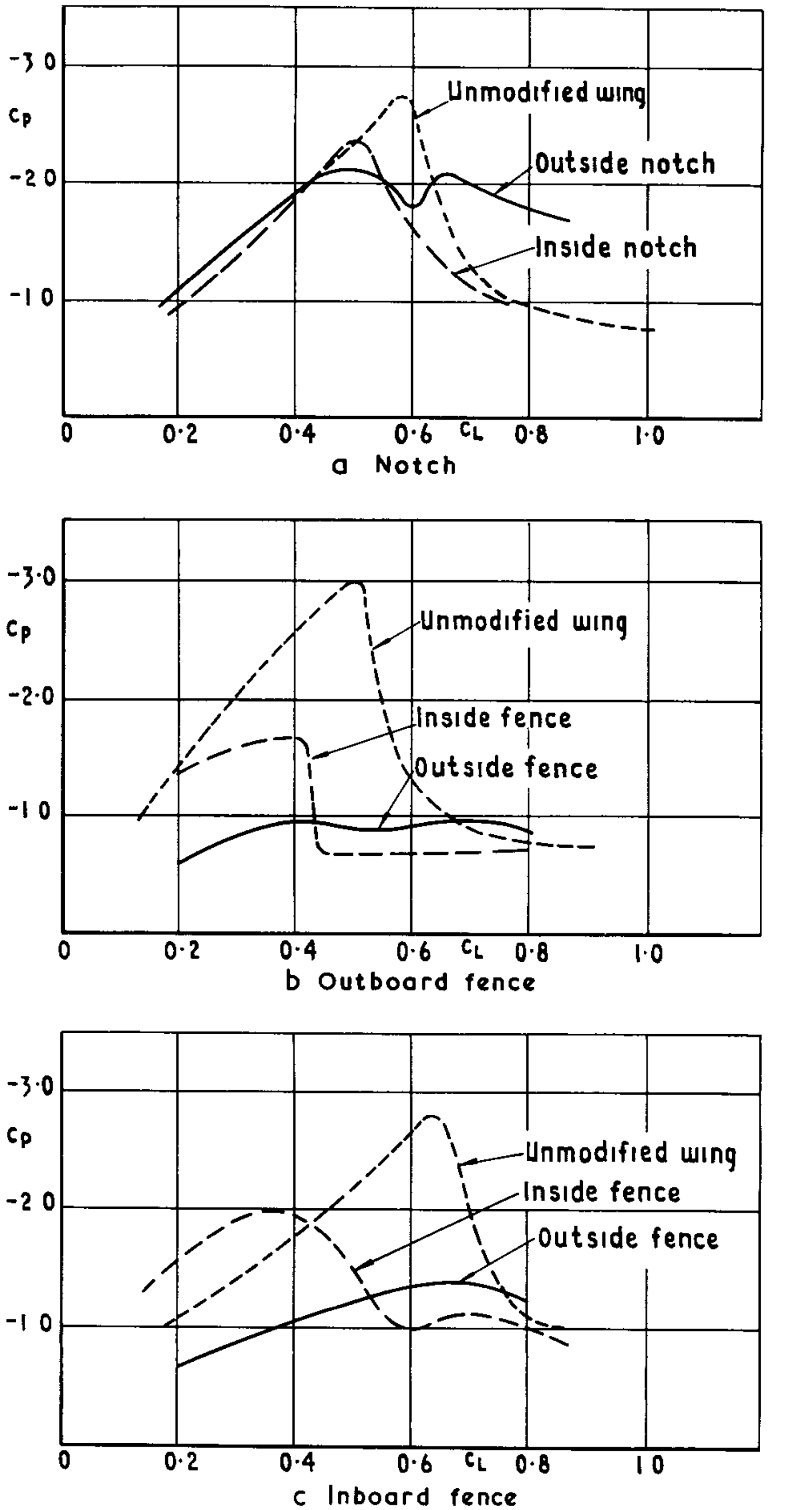


Fig.24 a–c Comparison of pressures on either side of notch and fences

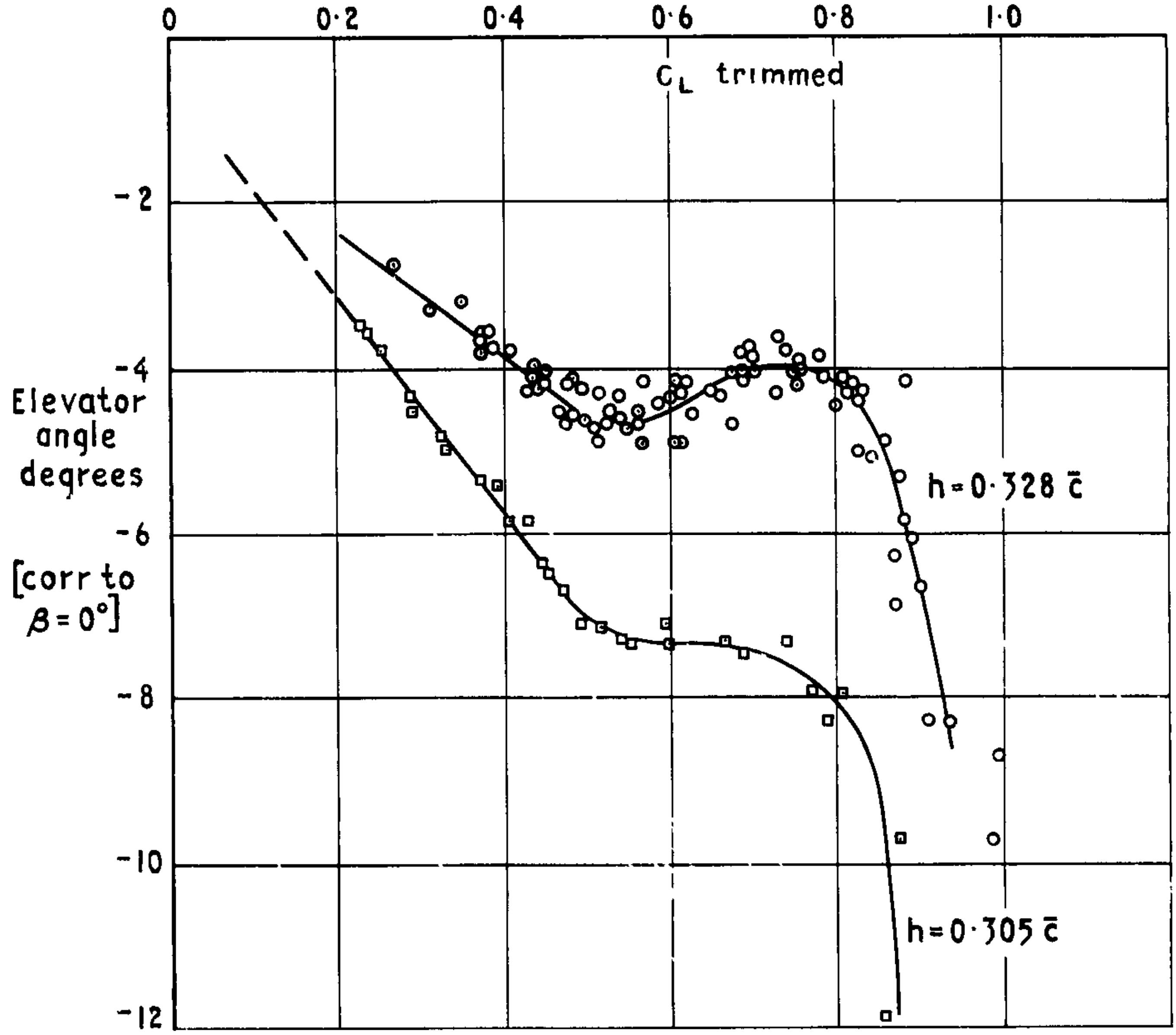


Fig.25 Elevator angles to trim. Basic aircraft

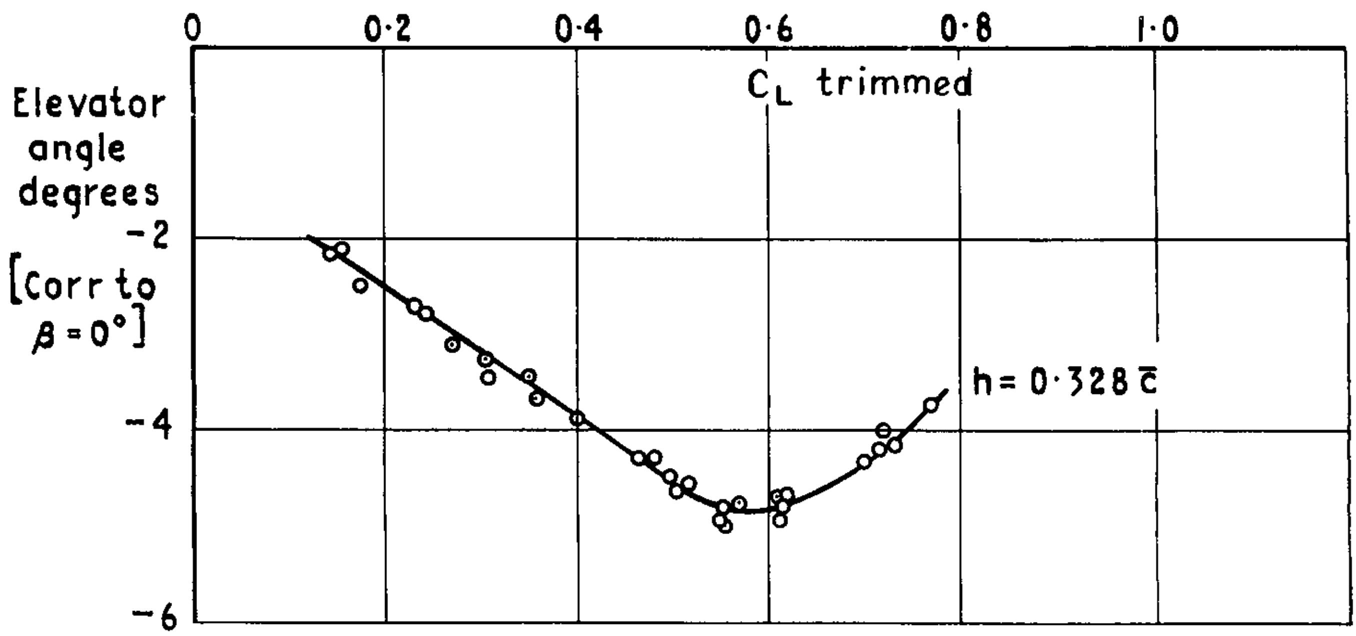


Fig 26 Elevator angles to trim. Aircraft with notch

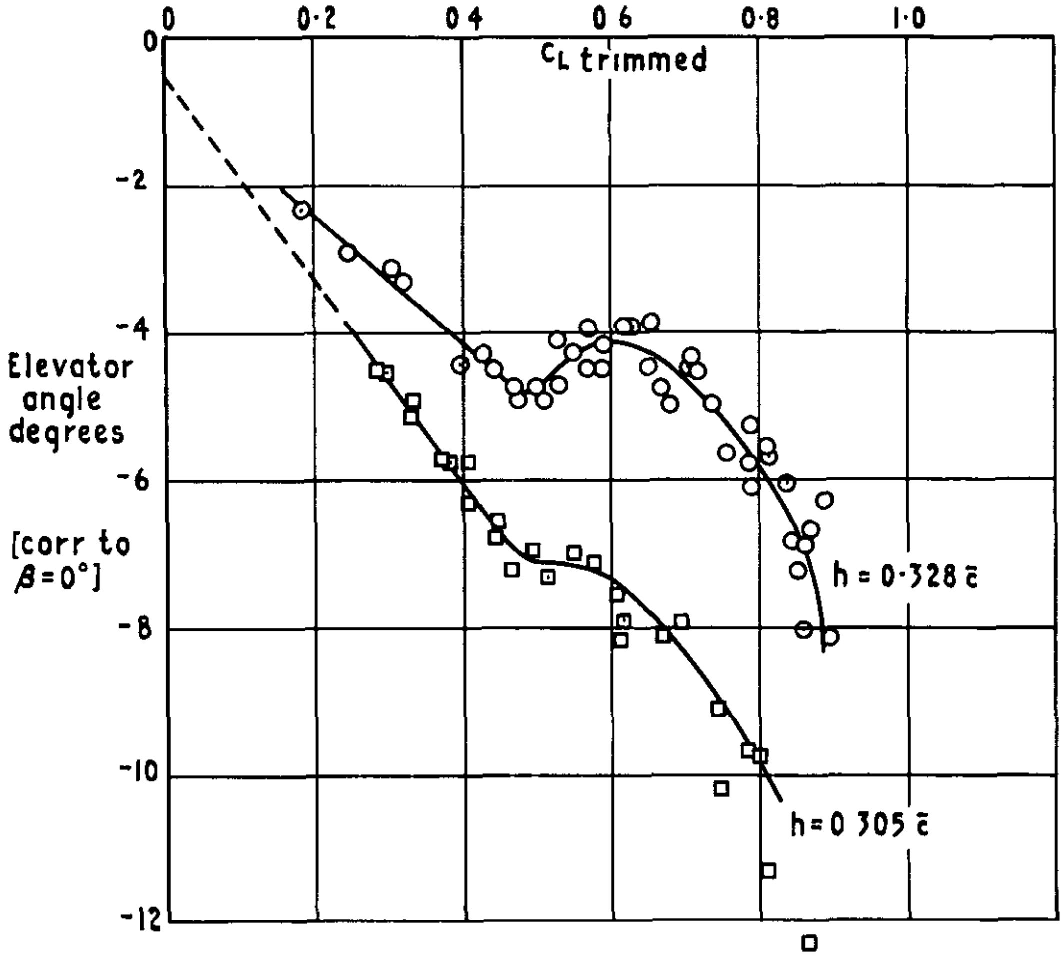


Fig.27 Elevator angles to trim. Aircraft with fences at 84% semispan

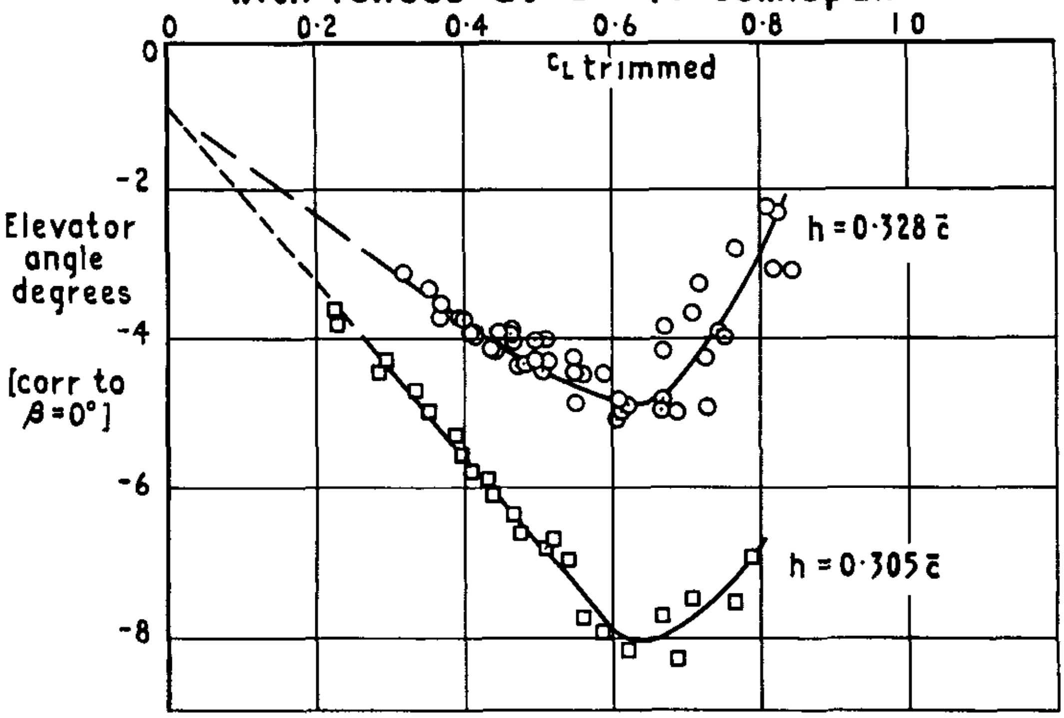


Fig.28 Elevator angles to trim. Aircraft with fences at 74·5% semispan

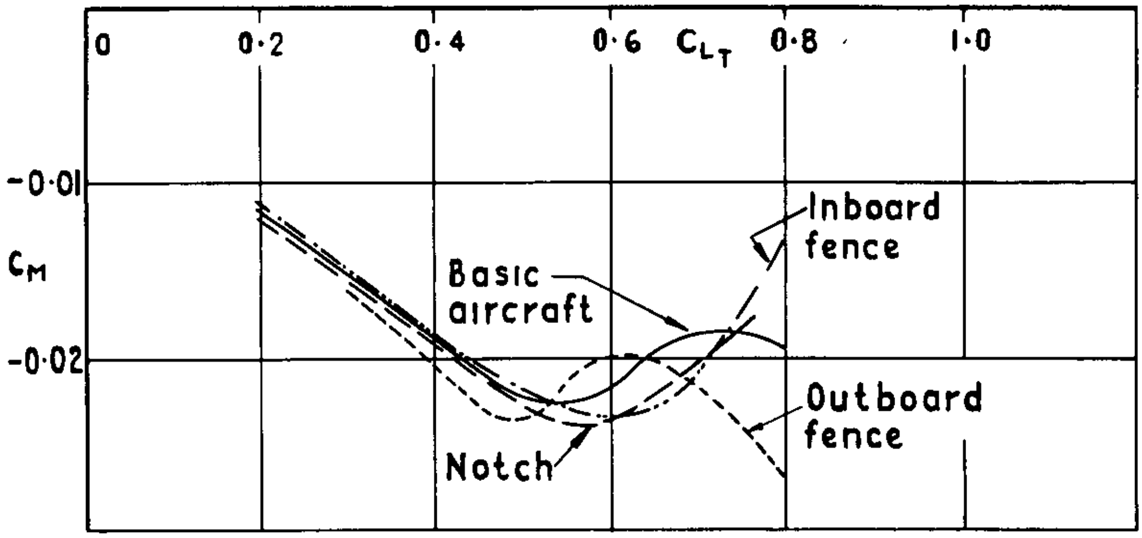


Fig.29 Comparison of pitching moment curves for aircraft with notch & fences h=0·328 $\bar{c}$

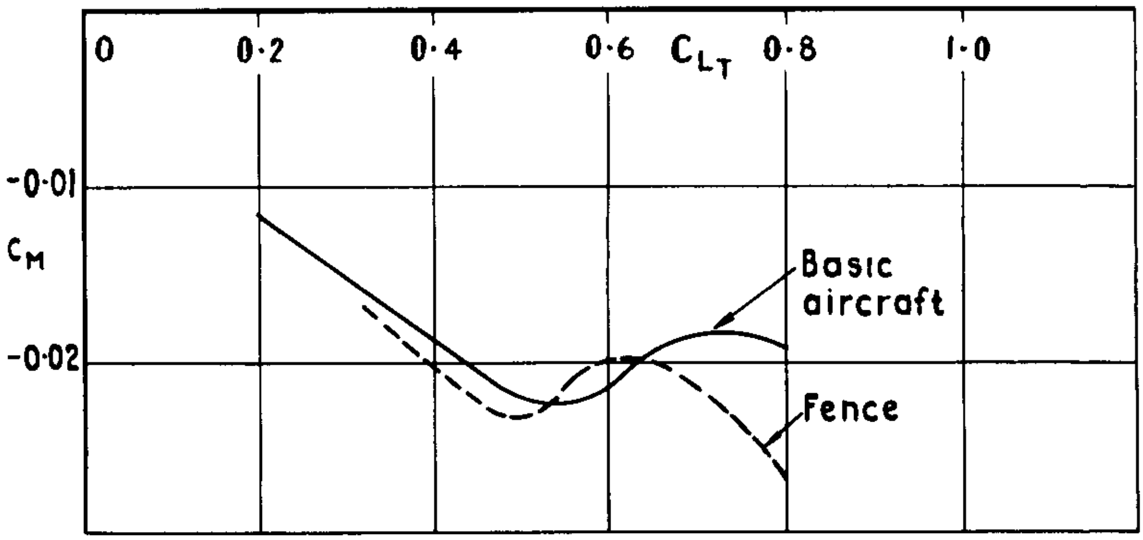


Fig 30 Flight measurements of the effect of a fence at 84% semispan on pitching moment

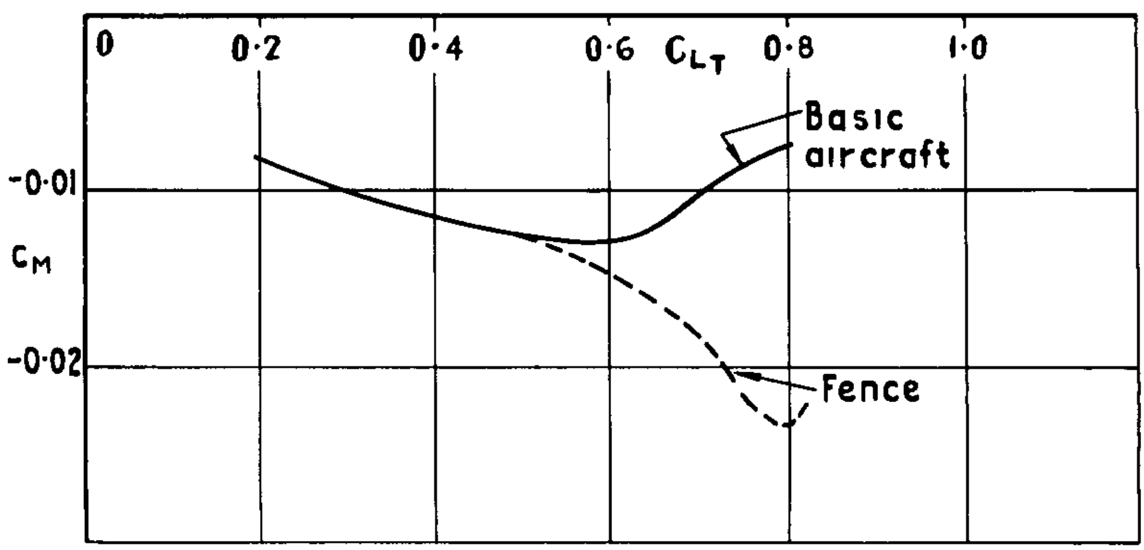


Fig 31 Tunnel measurements of the effect of a fence at 84% semispan on pitching moment (Avro 707A – from Ref 6)

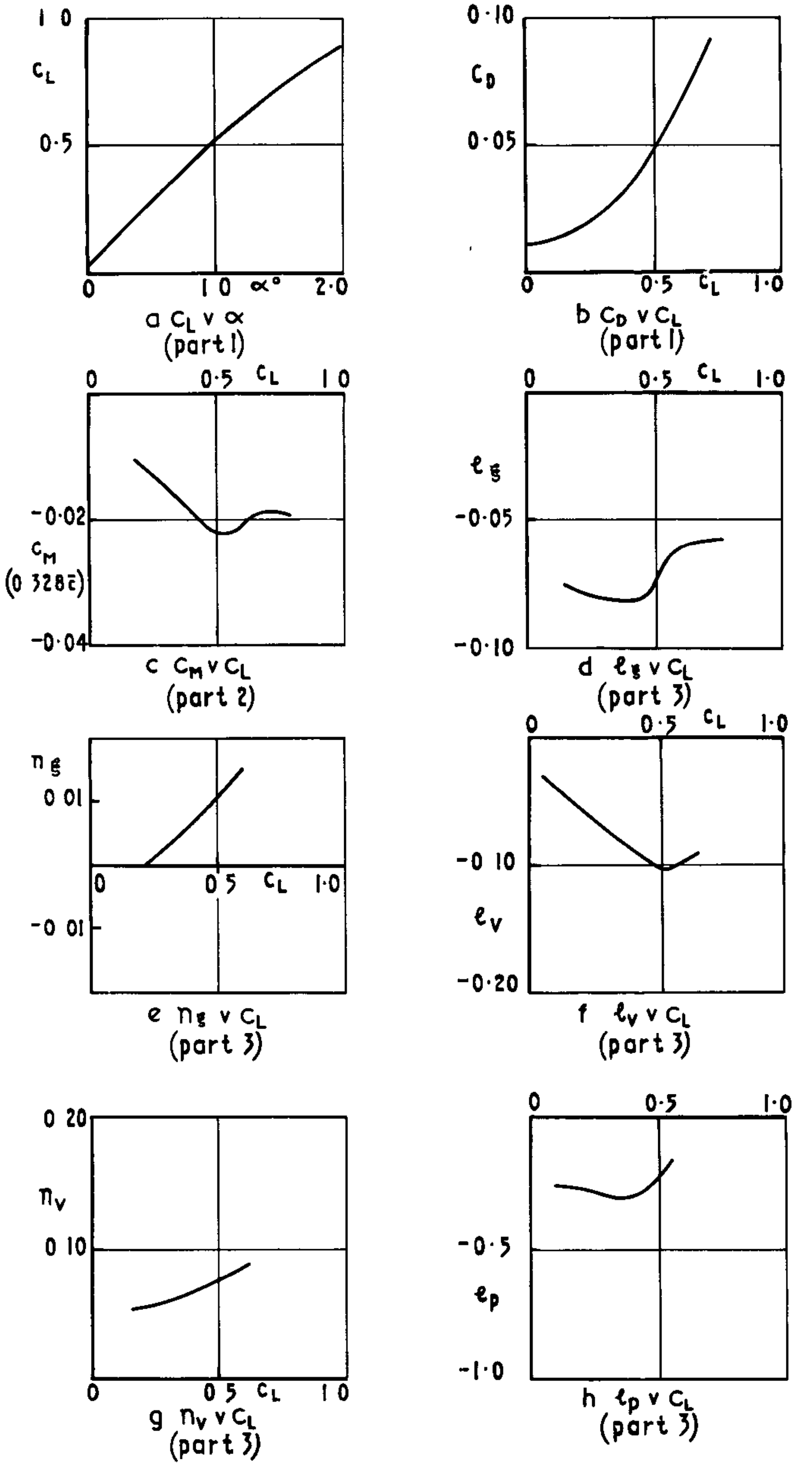


Fig.32a-h Summary of aerodynamic characteristics

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Morrall, J. C.

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1.2.2.2.3.1
1.1.7.4.2

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