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Flight Measurements of the Drag of an Aircraft Fitted with a Rear Fuselage Fairing Designed to Reduce the Transonic Drag

by

D. R. Andrews, F. W. Dee & D. Waters

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Flight Measurements of the Drag of an Aircraft fitted with a Rear Fuselage Fairing designed to reduce the Transonic Drag

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SUMMARY

Tests have been made to determine what reduction in drag could be achieved by fitting a fairing to the rear fuselage of a Hawker Hunter aircraft. This fairing was designed to control the shock wave movements in the wing body junction and so reduce the transonic drag rise. Area distributions at M = 1.08 were also improved to some extent.

Comparison of the results before and after modification showed that the fairing had delayed the onset of the transonic drag rise by about 0.015 in Mach number at $C_L = 0.1$ and that the magnitude of the drag rise itself had been reduced by about 0.007 in C_D at M = 1.00. Beyond M = 1.00 this gain decreased progressively, disappearing entirely at M = 1.17. It is noteworthy that these gains were achieved for no increase in the subsonic drag and for no change in the handling characteristics of the aircraft.

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1 Introduction

Numerous tests on wind tunnel and free flight models have shown that substantial reductions can be made in the transonic drag rise of conventional swept-winged aircraft by suitable modifications to the body shape. Physical explanations for this and design methods for achieving such drag reductions have been offered by the area rule concepts propounded by Whitcomb[1] and by Jones[2] and in the ideas concerning shockwave control and elimination put forward by Küchemann[3,4] and Bagley[5,6]. The largest gains are to be obtained by designing aircraft from the outset to incorporate any necessary waisting of the fuselage, but even on existing aircraft it should be possible to reduce the drag appreciably by the addition of suitable fairings.

The sonic area rule may be deduced as a limit of linearised supersonic theory, and in this form it has the important merit of enabling the drag rises of two aircraft to be compared quantitatively[7]. Such a quantitative comparison was made between the Hunter and a hypothetical aircraft with an appropriate optimum area distribution for low drag rise[8]. This comparison emphasised the possibility of achieving significant reductions in the drag rise of the Hunter by adding volume to certain regions of the fuselage.

However, because of practical restrictions on the regions where modifications could be made to the Hunter fuselage, it was found that the fuselage fairings required to make the aircraft conform fully with this optimum area distribution were very extreme in shape, and it seemed likely that these fairings would lead to a much lower critical Mach number and possibly to flow separations.

As the extent to which a partial application of area rule could be made was uncertain, it was decided to consider instead the use of the Küchemann design method for determining the shape of a rear fuselage fairing. This method not only offered an increase in the critical Mach number and a reduction in the drag rise, but also enabled the pressure distribution on the fuselage to be controlled to some extent. The fairing produced by this method was found to be of an acceptable size and also to represent a partial improvement in the mean area distribution at a low supersonic Mach number. It was accordingly decided to fit such a fairing to a Hunter aircraft to see what reduction in drag could be obtained. Drawings and photographs of the modified aircraft are shown in Figures 1, 2, and 3.

This report gives details of the design of this fairing and of the results obtained from the flight tests to measure the drag of the modified aircraft. Comparison with the drag measurements already made on the unmodified aircraft[9] shows the extent of the improvement made.

2 Design of the Fairings

2.1 The Küchemann Design Method

The design of suitable fairings for the Hunter according to the Küchemann theory has been described by Bagley[5,6]. It is shown there, and in Küchemann's paper[4], that the critical Mach number of the aircraft can be increased, and the rate of the drag rise at supercritical speeds reduced, by appropriate shaping of the wing-fuselage junction region. These improvements are obtained by modifying the pressure distribution on the wing, particularly in and near the region of the wing-body junction. In the present instance, the opportunities for improvement were limited by practical restrictions on the positions in which fairings could be applied; positions ahead of the intakes and on top of the fuselage being considered undesirable for various reasons. These practical restrictions, in effect, limited the modification to the sides of the rear fuselage.

Figure 4 shows the calculated velocity distribution in the wing-body junction at M = 0.92, together with the velocities on an infinite sheared wing with the same section and sweepback as the outer wing panels of the aircraft. This velocity distribution was considered to be a good approximation to that obtaining near the mid-semispan station on the wing. Also shown in Figure 4 is a "desirable" velocity distribution for the wing-body junction. This distribution satisfies the general principles laid down in Reference 4 for the reduction of transonic drag rise by controlling shock positions, and also offers some increase in isobar sweepback over the wing root region and hence an increase in the critical Mach number for the aircraft. The size and shape of the rear fuselage fairing needed to achieve this junction pressure distribution is shown in the lower part of Figure 4,* and the velocity distribution on this fairing is also shown.** It is immediately obvious that the negative velocity increments produced in the wing root by such a fairing are necessarily accompanied by positive velocity increments on the bulge itself. Although these increased velocities might lead to shocks and even to boundary layer separations on the rear fuselage it was thought that in this case the fairing was not so extreme as to suffer from these adverse effects. These positive velocity increments on the fairing are a necessary consequence of the fact that in the present case the modifications had to be additional to the basic Hunter outline.

Using this calculated shape for the centre line, the bulge was faired elliptically to zero at the top and bottom of the fuselage as shown in Figure 2. A G.A. of the modified aircraft is shown in Figure 1 and general views of it in Figure 3.

2.2 Comparison with Area Rule

Figure 5 shows the area distribution for the basic Hunter aircraft at M = 1.00 together with the modification to this area distribution brought about by the addition of the Küchemann design of rear fuselage fairing. It is seen that although the hollow in the distribution has been filled in to some extent by the fairing, the sharp gradients have been little changed. This suggests that a fairing of about the same volume, designed to eliminate these steep gradients, may perhaps produce greater drag reductions than the Küchemann design. However, it is recognised[10] that it is preferable to design for a Mach number slightly above unity using the supersonic area rule of Jones[2]. It was therefore interesting to note that the fuselage fairing designed by the Küchemann method resulted in a partial improvement in the mean area distribution for a Mach number of 1.08 (Figure 6).

3 Details of Aircraft and Instrumentation

The modification described above was made to the Hawker Hunter Mk.1 (Serial No. W.T.571) which had been tested previously[9,11]. Details of the aircraft as tested are given in Table I.

The instrumentation and equipment used was identical to that described in Reference 9 except that the "Barnes Type" accelerometers were replaced by S.F.I.M. transmitting accelerometers of ranges 0 + 2g, - 1 + 5g, and ± 1g. The damping was adjusted to 0.7 critical and maintained constant by sealing the accelerometers in airtight boxes.

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* It may be noted that this fuselage fairing was strictly designed for use in conjunction with a fairing over the forward sections of the stub wings, although this was in fact never fitted.

** It should be pointed out that the theory assumes that the wing thickness in the junction is not large compared with the body diameter there; in this instance that condition is not satisfied. Empirical evidence from low speed wind tunnel tests suggests that for a wing thickness/body diameter ratio similar to that at mid chord of the present design only about 50% of the calculated velocity increment due to the body will be obtained. This factor has been applied in the present case.

4 Method of Test and Analysis of Results

4.1 Drag Measurements

The aircraft drag was determined by several different methods depending upon the combination of C_L and M under investigation. Techniques employing stabilised levels, dives and "wind-up" turns were all adopted at different times.

It is shown in Reference 12 that the drag, D, is given approximately by

$$\frac{D - F_N}{W} = R + Q\psi \qquad (1)$$

where F_N = nett thrust

W = aircraft weight

R = reading of longitudinal accelerometer in g units

Q = reading of normal accelerometer in g units

ψ = angle between flight path and longitudinal accelerometer axis (radians)

In stabilised levels $D = F_N$, but under other conditions we must also measure R, Q, and ψ.

The nett thrust F_N was measured using the jet pipe pitot method[12].

4.11 Stabilised Levels

To facilitate fairing and cross plotting, the stabilised levels were made at specified Mach numbers and C_Ls. For each Mach number and C_L the pilot was given a chart showing indicated airspeed and altitude plotted against fuel contents so that he could select the speed and altitude appropriate to the fuel state. Great care was taken to stabilise speed and altitude accurately before taking a reading.

The drag of the aircraft was determined from the stabilised levels by equating the horizontal component of nett thrust to drag, in the usual manner.

Measurements of incidence were made during these stabilised levels, the incidence being obtained directly from the readings of a Hussenot pendulum level[9].

4.12 Dives and Pull-outs, and Wind-up Turns

The drag in dives and pull-outs and in wind-up turns was determined from equation (1) by measuring the accelerations R and Q, the quantity ψ being obtained from estimated values of lift curve slope and zero lift angle (para. 5.1).

Results were obtained at a constant C_L of 0.1 for Mach numbers between 0.9 and 1.05 by giving the pilot charts similar to those used for the stabilised levels, but with additional lines appropriate to the various dive angles also plotted. The drag at higher C_Ls and Mach numbers was determined from contiruous records of dives and pull-outs, and from wind-up turns at constant Mach numbers.

4.2 Corrections to Flight Data

The static and pitot pressure errors of the unmodified aircraft have been measured thoroughly and the results given in Reference 11. These results were also used for the modified aircraft, as brief check measurements showed that the rear fairing had not altered the static pressure error of the nose boom system.

Corrections for pressure lag in the static and pitot lines of the airspeed system and in the jet pipe pitot system were applied to all flight measurements in dives and glides. The lag in these systems was determined from ground tests in the usual manner.

The impact air temperature bulb was calibrated for Mach number effects by the standard technique. Tests at 10,000 ft and 35,000 ft gave recovery factors of 0.96 and 0.91 respectively.

5 Results and Discussion

5.1 Incidence and Lift Curve Slope

Figure 7 shows the overall aircraft lift curve slope and zero lift angle as obtained from measurements during stabilised levels. Comparison with the results obtained previously[9] shows that the modification to the aircraft has had a negligible effect on these quantities, any small differences that do exist probably being within the experimental accuracy.

As it was again not found possible to measure incidence directly at transonic and supersonic speeds, some assumptions were necessary in this respect in order that drag could be deduced using the accelerometer technique. In the absence of any more reliable information, the same values of zero lift angle and lift curve slope as had been used previously[9] were used. For convenience these values are shown in Figure 7. The errors in drag introduced by any errors in these assumed values of incidence have been discussed in Appendix I of Reference 9.

5.2 Drag

The drag results obtained for the modified aircraft during stabilised levels and accelerated manoeuvres are presented in Figure 8, and a comparison with the results obtained before modification[9] is shown in Figure 9.

The addition of the rear fuselage fairing is seen to have postponed the drag rise at $C_L = 0.1$ by about 0.015 in Mach number, and to have reduced the drag at $M = 1.00$ by about 0.007. As Mach number is increased above this value so the gain progressively reduces and disappears entirely by $M = 1.17$. Beyond this Mach number the results suggest that the fairing may be increasing the drag of the basic aircraft. It is noteworthy that these gains have been achieved for no increase in the subsonic drag throughout the range of C_L covered in these tests.

Although the induced drag supersonically seemed of the same order as for the unmodified aircraft, the larger experimental scatter that seemed a feature of the present tests precluded any specific values of induced drag factor being obtained. Hysteresis errors in the longitudinal accelerometer are probably the cause of this larger scatter.

6 General Handling Characteristics of the Modified Aircraft

The modification to the rear fuselage was found to have a negligible effect on the handling characteristics of the aircraft throughout the whole speed and Mach number range.

A brief formation check with another Hunter indicated that the rear fairing had improved the maximum level speed performance by about 0.01 to 0.015 in Mach number. Comparison between full throttle levels made before and after modification tended to confirm this improvement. Such an improvement would be expected from the delay in the onset of the drag rise already noted in para. 5.2

7 Conclusions

A Hawker Hunter aircraft was fitted with a rear fuselage fairing designed to control shock wave movements in the wing body junction and so reduce the transonic drag rise. It also represented a partial fulfillment of the requirements of area rule at M = 1.08.

Comparison of the results before and after modification showed that

(a) the fairing had delayed the onset of the transonic drag rise by about 0.015 in Mach number at $C_L = 0.01$.

(b) the magnitude of the drag rise at M = 1.0 had been reduced by about 0.007 in C_D. Beyond M = 1.0 this gain decreased progressively, disappearing entirely at M = 1.17.

(c) the reduced drag at transonic and supersonic speeds was obtained for no increase in the subsonic drag and for no change in the handling characteristics of the aircraft apart from a slightly improved level flight performance.

8 Acknowledgments

The authors are grateful to Mr. W.T. Lord and Mr. J.A. Bagley for their help in both the design of the fairings and the preparation of this report.

NOTATION

x = distance from nose of aircraft or from L.E. of wing root chord

$S(x)$ = mean of areas cut by planes tangential to the Mach cone having its apex at x

ℓ = fuselage length (45) ft

c = wing root chord

Δy = depth of fairing in plane of wing

V = local velocity

V_o = free stream velocity

C_L = $W.Q./\frac{1}{2}\rho V^2 S$

C_{D_T} = total drag coefficient based on wing area

M = true Mach number

$\left(\frac{\partial C_L}{\partial \alpha}\right)_M$ = overall aircraft lift curve slope at constant Mach number

NOTATION (Contd)

α = aircraft incidence measured from wing datum

α_o = aircraft zero lift angle measured from wing datum

F_N = nett thrust

W = aircraft weight

R = reading of longitudinal accelerometer in g units

Q = reading of normal accelerometer in g units

ψ = angle between flight path and longitudinal accelerometer axis

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No.	Author	Title, etc.
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2	R.T. Jones	Theory of wing-body drag at supersonic speeds. RM A53 H18(a) (TIB 3890). September 1953.
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4	D. Küchemann and D.E. Hartley	The design of swept wings and wing-body combinations to have low drag at transonic speeds. ARC 17871. April 1955.
5	J.A. Bagley	Some proposals to modify the shape of the Hawker Hunter aircraft. Unpublished RAE work. July 1954.
6	J.A. Bagley	Methods of designing wing-body combinations to have low drag at transonic speeds. RAE Report in preparation.
7	W.T. Lord	Examples of wing-body combinations of very small theoretical drag rise at transonic speeds (illustrating a new approach to the design of fuselages for high speed aircraft). ARC 16695a. December 1953.
8	W.T. Lord and E. Eminton	Slender bodies of minimum wave drag. Journal of Aeronautical Sciences Vol.21 No.8. August 1954. pp. 569-570.

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No.	Author	Title, etc.
9	D.R. Andrews and J.E. Nethaway	Flight measurements of the drag of a swept wing aircraft (Hunter Mk.1) at Mach numbers up to 1.2, together with some measurements of lift curve slope. ARC 17947. June 1955.
10	G.H. Holdaway	An experimental investigation of reduction in transonic drag rise at zero lift by the addition of volume to the fuselage of a wing-body tail configuration and a comparison with theory. NACA R M A54F22 (TIB 4312). August 1954.
11	D.R. Andrews and J.E. Nethaway	Flight measurements of the pressure errors of a nose and a wing boom airspeed system on a swept-wing aircraft (Hunter F.1) at Mach numbers up to 1.2. ARC 17629. January 1955.
12	D.J. Higton, R.H. Plascott and D.A. Clarke	The measurement of the overall drag of an aircraft at high Mach numbers. R & M 2748. January 1949.

TABLE I

Details of Hunter W.T.571 (After Modification)

Aircraft	
Wing Area	340 sq ft
Wing Span	33 ft 8 ins
Aspect Ratio	3.33
Taper Ratio	0.41
Sweepback at $\frac{1}{4}$ chord	40°
Standard mean chord $\bar{c}$	10.1 ft
Wing setting to fuselage datum	$1\frac{1}{2}^{\circ}$
Max. thickness/chord ratio (symmetrical)	8.5%
Position of max. t/c	37.5% chord
A.U.W. (Take-off)	15808 lb
Fuel contents	334 gals
C.G. position at take-off	0.313 $\bar{c}$
" " with rear tanks empty (274 gals. left)	0.285 $\bar{c}$
" " with rear and centre tanks empty (202 gals. left)	0.283 $\bar{c}$
" " no fuel	0.332 $\bar{c}$
Engine	
Rolls Royce AVON 113	

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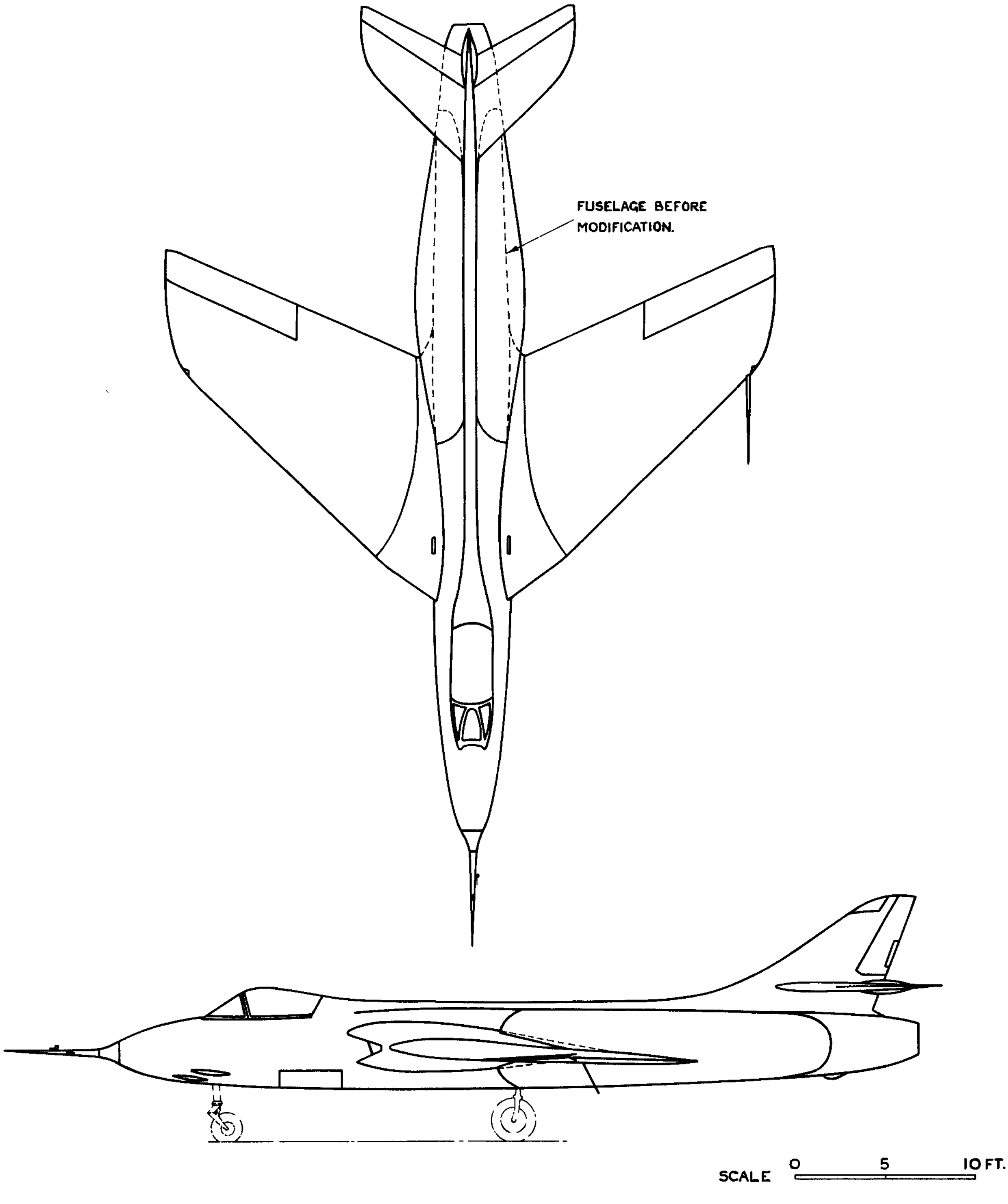
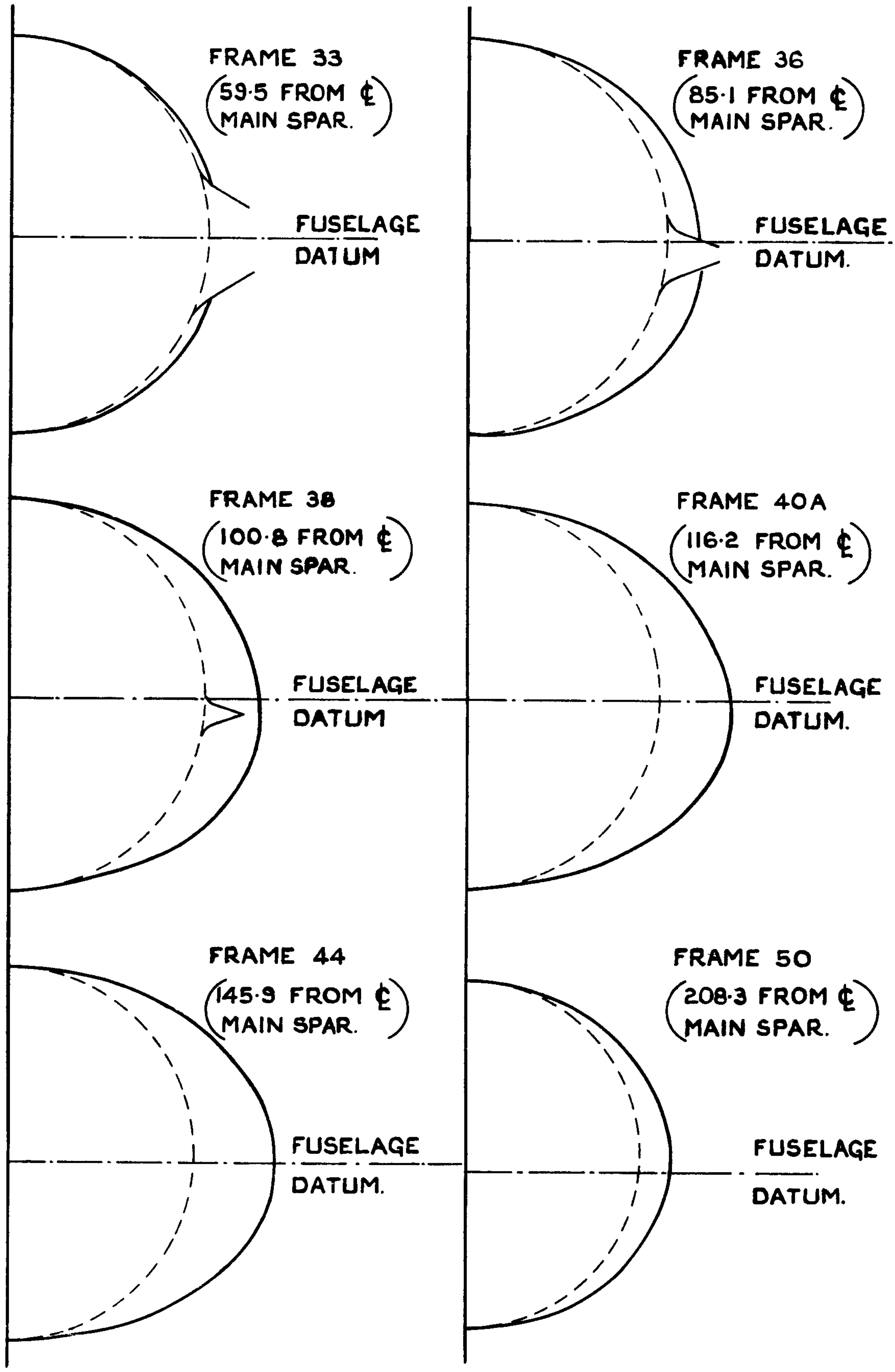


FIG. 1. G.A. OF HAWKER HUNTER F. MK.I FITTED WITH REAR FUSELAGE FAIRING.



FAIRING STARTS	39·05"	FROM ₵ MAIN SPAR.
FAIRING ENDS	256·45"	" " " "
END OF FUSELAGE	313·54"	" " " "
NOSE OF FUSELAGE –	223·72"	" " " "

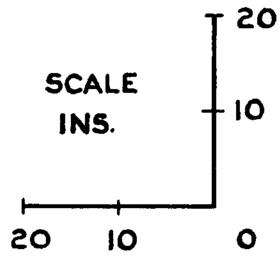


FIG. 2. TYPICAL SECTIONS THROUGH REAR FUSELAGE FAIRING.

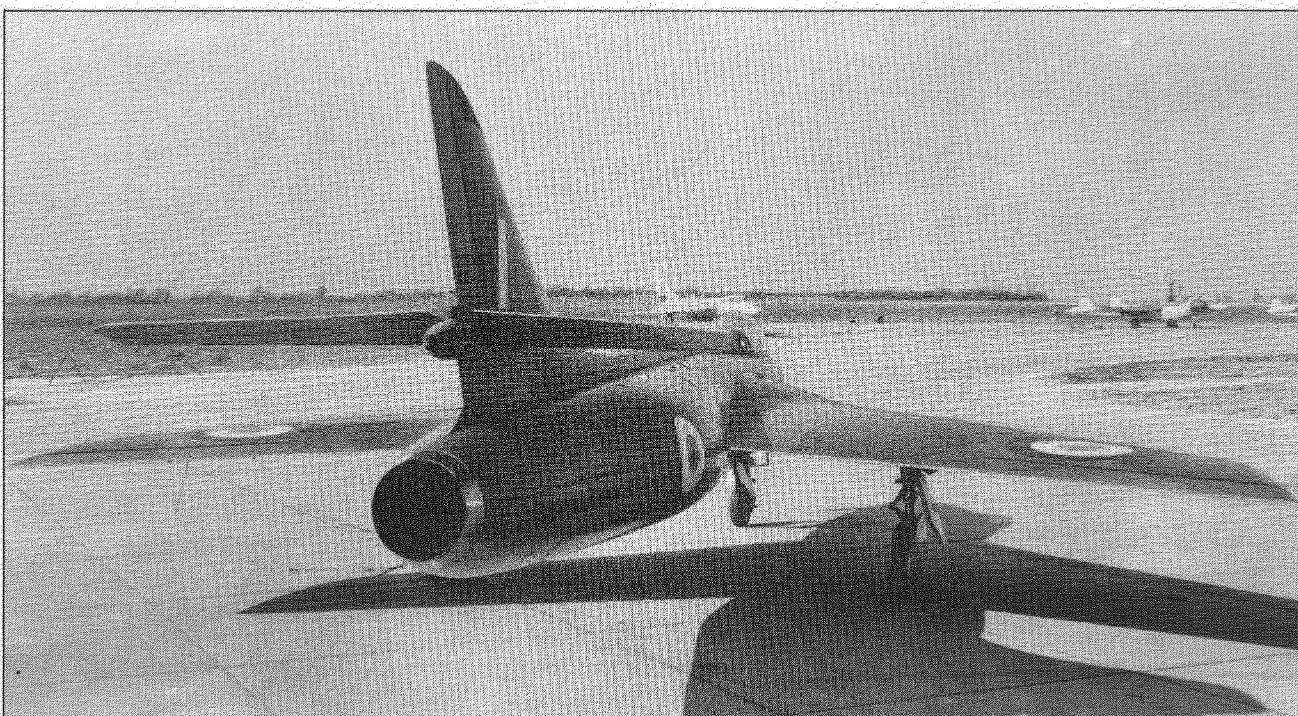


FIG.3. GENERAL VIEWS OF MODIFIED AIRCRAFT

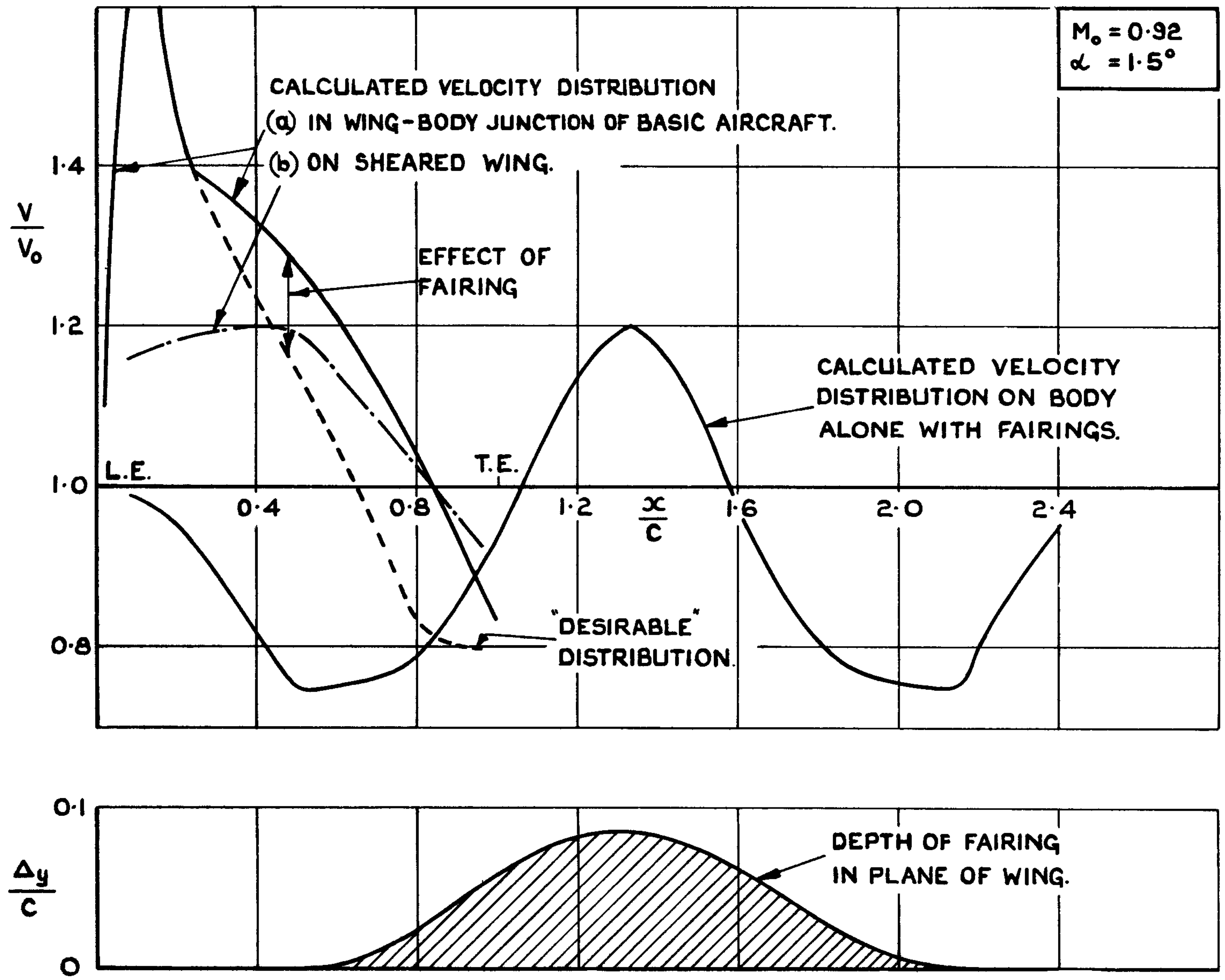


FIG. 4. SHAPE AND VELOCITY DISTRIBUTION OF REAR FAIRING. (REFS. 5 & 6.)

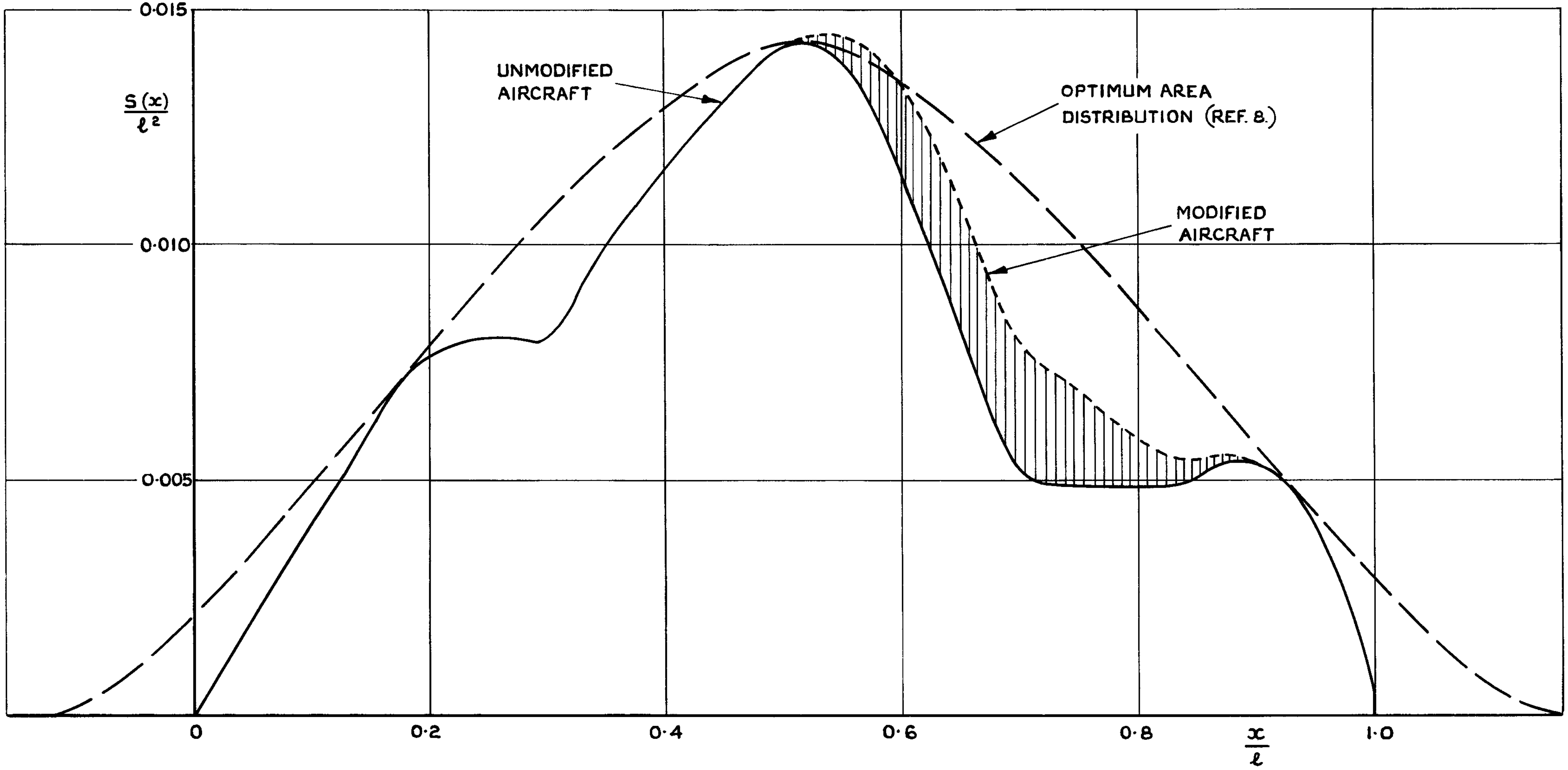


FIG. 5. AREA DISTRIBUTIONS AT M = 1·00.

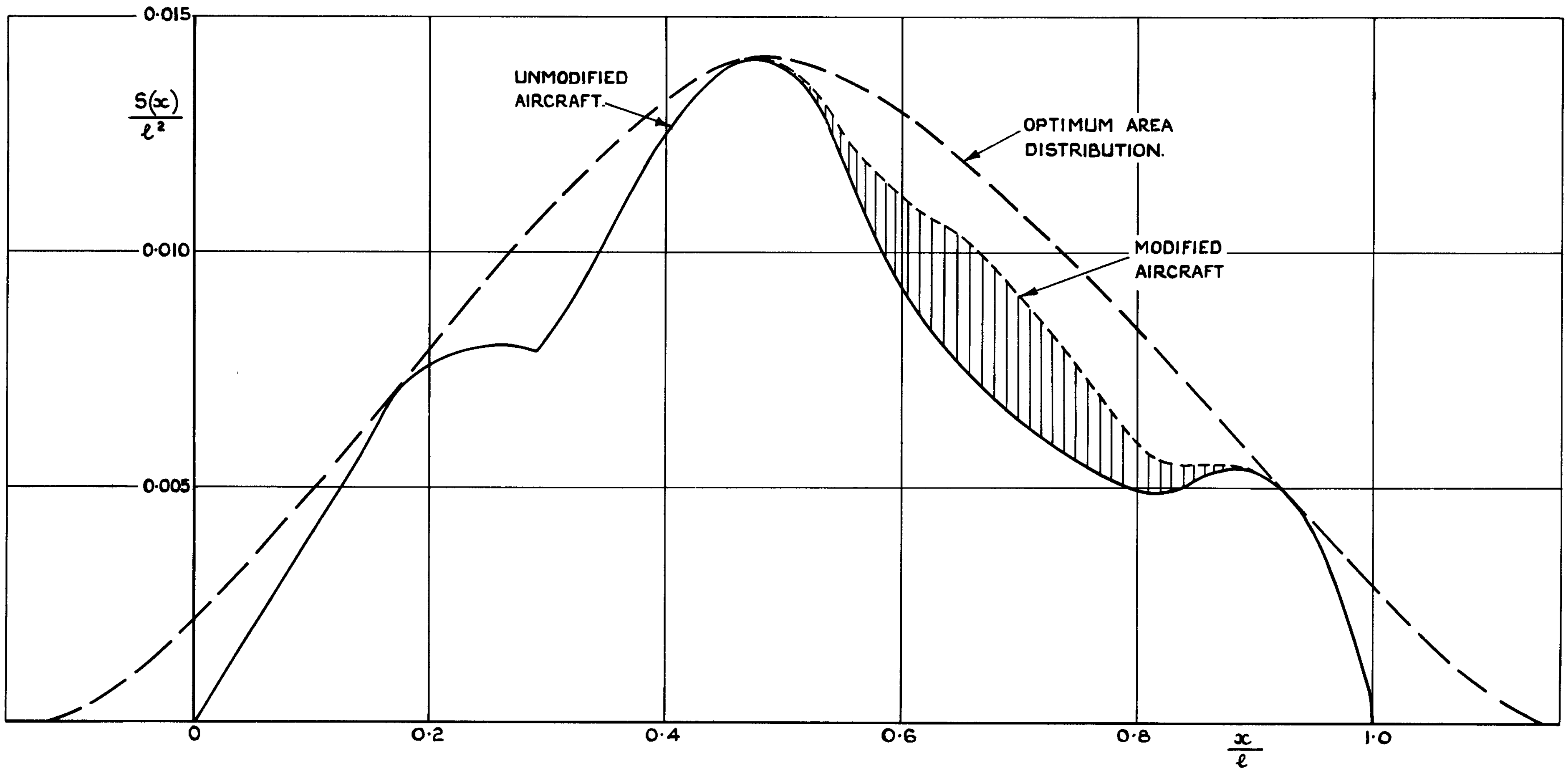
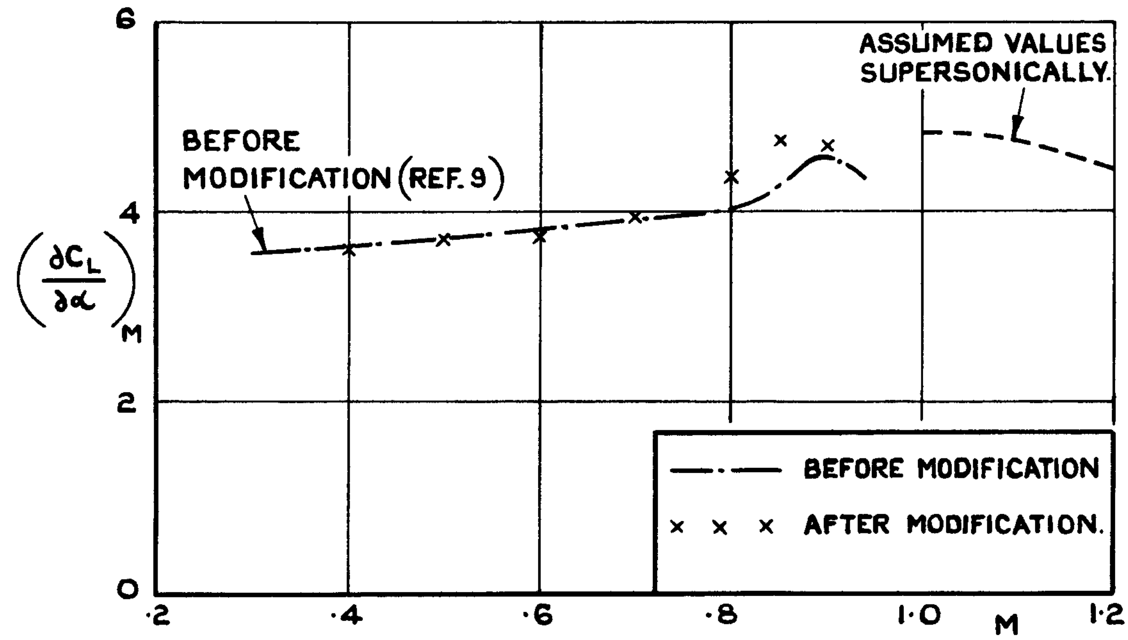
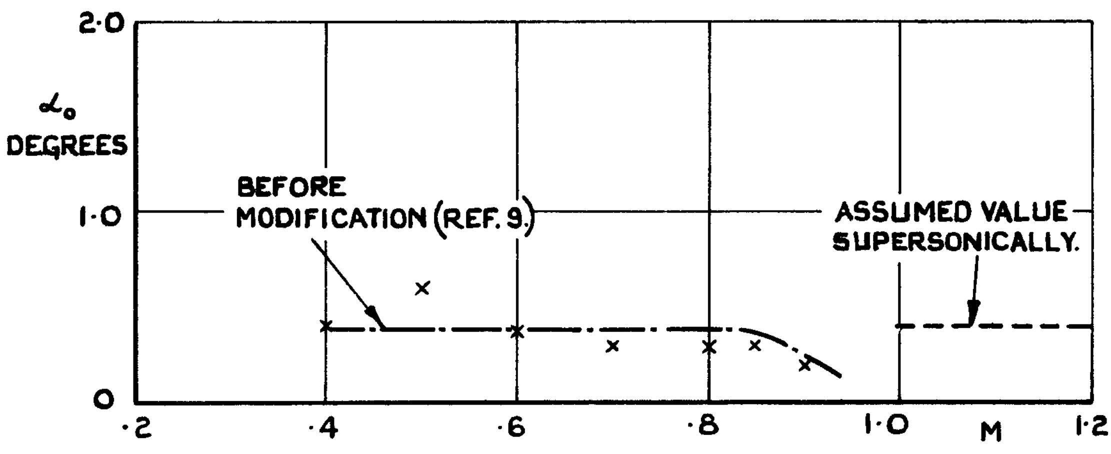


FIG. 6. AREA DISTRIBUTIONS AT M = 1·08.



(a) LIFT CURVE SLOPE.



(b) ZERO LIFT ANGLE.

FIG. 7(a & b) LIFT CURVE SLOPE AND ZERO LIFT ANGLE FOR AIRCRAFT TRIMMED AT 0·285 $\bar{c}$ MEAN C.G.

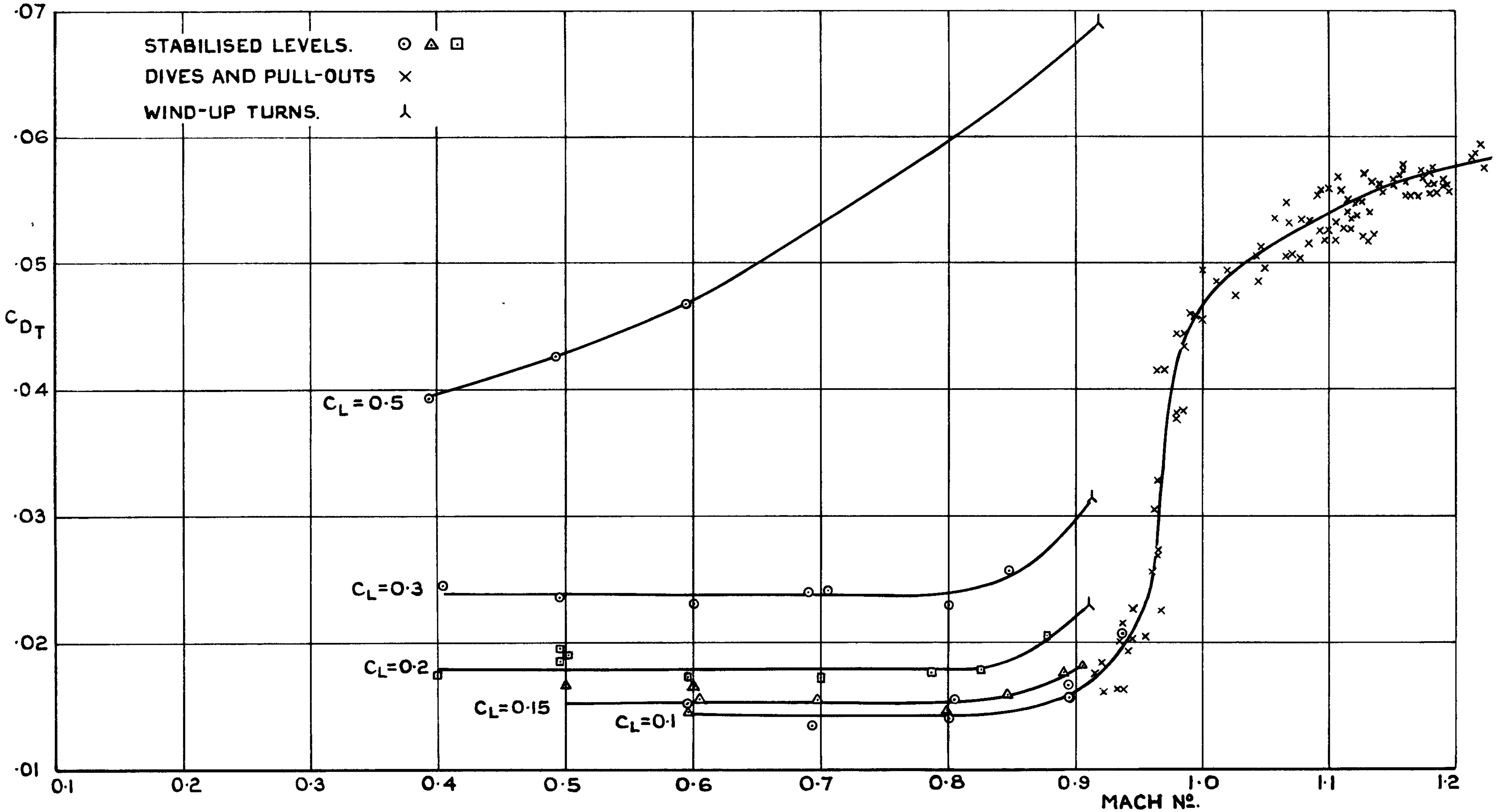


FIG. 8. MEASURED DRAG OF AIRCRAFT WITH REAR FUSELAGE FAIRING.

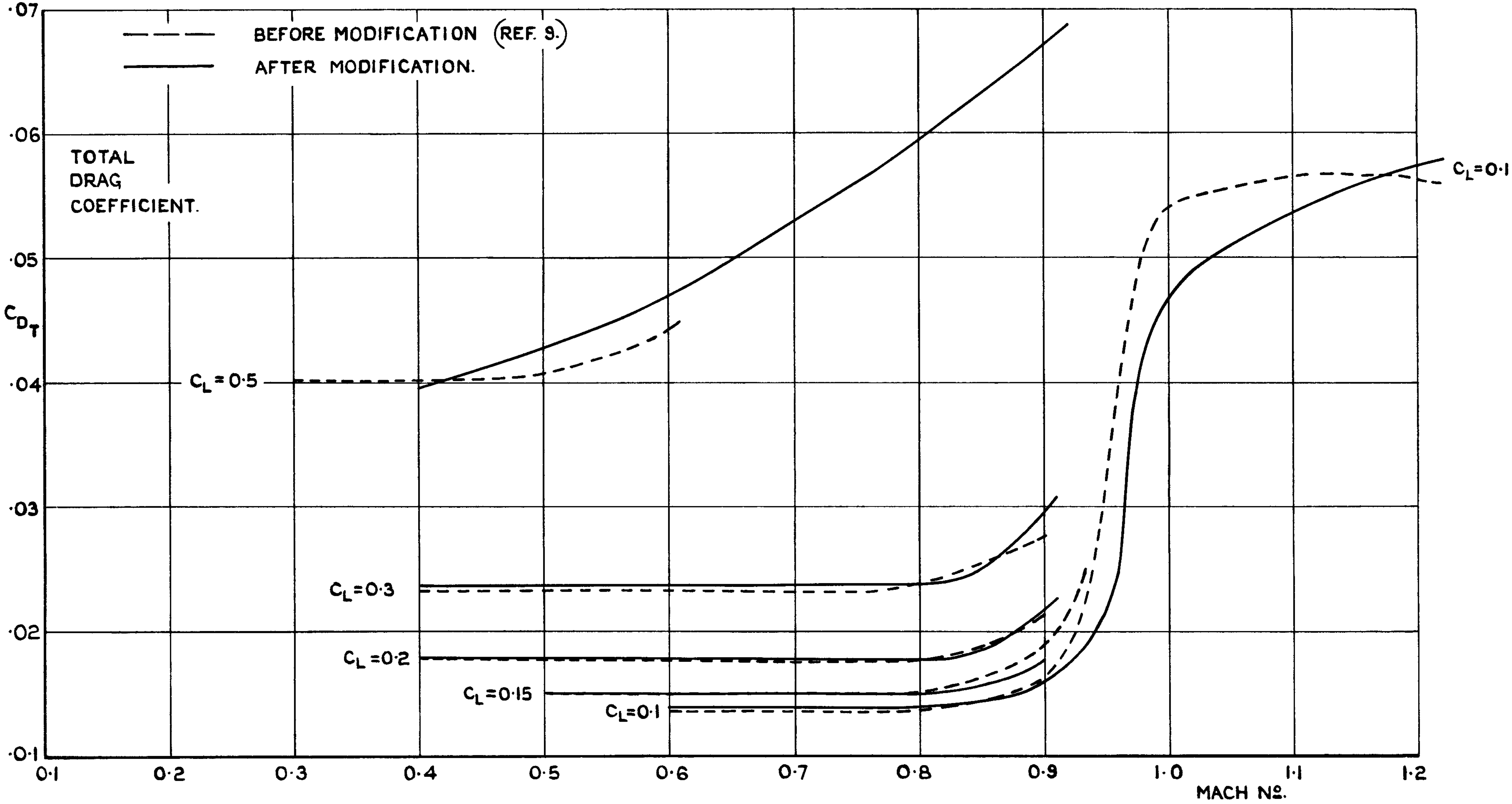


FIG. 9. COMPARISON OF DRAG BEFORE AND AFTER MODIFICATION.

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