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# Free Flight Measurements at Supersonic Speeds of the Drag of a Particular Delta Wing having a Simple Pressure Distribution

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SUMMARY

Squire has drawn attention to a family of delta wings, having aerofoil-like sections with a rounded leading edge, whose supersonic wave drag at zero incidence is theoretically determinable, so long as the leading edge is subsonic. Drag measurements have been made, at a Reynolds number of $7 \times 10^6$, on one member of this family using the ground-launched rocket-boosted model technique. The experimental wave drag is in good agreement with tunnel measurements on another wing of the family, both methods giving values less than theory.

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## 1 Introduction

As yet little theoretical work exists on the drag of round-nosed aerofoils at supersonic speeds. Using linearised theory, Squire[1] has determined the pressure distribution, when the leading edge is subsonic, for a flat elliptic cone with the equation

$$\frac{z}{2t_o} = \left(\frac{x^2 - y^2 \tan^2\Lambda}{c^2}\right)^{\frac{1}{2}}$$

and for a flat 'elliptic hyper-cone' with the equation

$$\frac{z}{2t_o} = \frac{|x|}{c}\left(\frac{x^2 - y^2 \tan^2\Lambda}{c^2}\right)^{\frac{1}{2}}$$

where $x$ = upstream distance from the apex,

$y$ = distance to starboard,

$z$ = distance downwards,

$c$ = root chord,

$\Lambda$ = sweepback angle,

and $t_o$ = constant determining the thickness.

Squire has combined the results for these two surfaces and obtained the pressure distribution for a wing-like surface with the equation

$$\frac{z}{2t_o} = \left(1 - \frac{|x|}{c}\right)\left(\frac{x^2 - y^2 \tan^2\Lambda}{c^2}\right)^{\frac{1}{2}}$$

This is a delta wing with round-nosed sections except at the root, where the section is biconvex. It has a root thickness/chord ratio of $t_o/c$. In a corrigendum to the original report, Squire applies a correction to this linearised theory solution to allow for the high pressures at the rounded leading edge. This correction is due to R. T. Jones[2].

To obtain an experimental check on the theory the drag has been measured, in free-flight, of a 45° sweptback delta wing of this family having a root thickness/chord ratio of 0.06. The theoretical geometric characteristics of the design wing and the actual characteristics of the wing tested are shown in Fig.1.

The result and a comparison with theory are given in this report. Comparison is also made with tunnel tests on another member of the family having 60° sweepback and a root thickness/chord ratio of 0.10, and with a wing having R.A.E.101 section and a uniform thickness/chord ratio of 0.06 throughout.

The flight test of this report was made in July 1950.

## 2 Test Vehicle and Test

The test vehicle (Figs. 2 and 3) was based on a 5 inch L.A.P. solid fuel rocket. The rocket was encased in a bakelised paper tube, to which was attached a 4-calibre ogival head of moulded perspex with a wooden nose. The two wings, which had a root chord of 20 inches, were made of wood with tufnol trailing edges. Stability in yaw was achieved by fitting two stabilising fins rearward of the wings and at right-angles to them. These fins were flat plates of duralumin with chamfered leading and trailing edges.

The vehicle was launched at the Larkhill range by Trials Wing, Guided Weapons Dept. The flight path was observed by kiné-theodolites which gave the trajectory. The velocity along the line of sight was given by the radio reflection doppler method. These and atmospheric data permitted the evaluation of the total drag of the test vehicle[3] over the Mach number range $0.95 < M < 1.6$. The Reynolds number of the test varied from $5 \times 10^6$ to $10 \times 10^6$ as $M$ increased from 0.95 to 1.6 (See Fig.4).

## 3 Results and Comparison with Theory and other Tests

The total drag coefficient of the test vehicle is plotted in Fig.4. The drag coefficient of the body and stabilising fins, as obtained from separate tests[3], is also shown. From these the drag coefficient of the wing is derived by subtraction and is shown in Fig.5, where it is based on exposed wing area.

Fig.6 gives a comparison with theory and also with some tests, as yet unpublished, made in the R.A.E. 9" supersonic tunnel. In this figure the results are plotted in the form $\dfrac{C_D\sqrt{M^2-1}}{{t_o/c}^2}$ against $\cot\Lambda \cot\mu$,

where $C_D$ is the wave drag coefficient based on exposed wing area, $t_o/c$ the root thickness/chord ratio, $\Lambda$ the sweepback angle and $\mu$ the Mach angle. This method of plotting enables direct comparison to be made with the tunnel tests; these tests were made on a wing of different sweepback and thickness. The theoretical curve is that given by Squire[1]. The wave drag coefficient used in the free-flight curve was derived from the experimental drag coefficient by subtracting 0.005, to allow for the probable value of skin friction drag coefficient. The tunnel wave drag coefficients were obtained from pressure distribution measurements on a half wing. A summary of the relevant test data is given below.

| | Wing | Mach No. | Reynolds No. |
|---|---|---|---|
| Free Flight | $\Lambda = 45°$, $t_o/c = 0.06$ | 0.9 - 1.6 | $7 \times 10^6$ mean |
| Tunnel | $\Lambda = 60°$, $t_o/c = 0.10$ | 1.6 | $0.85 \times 10^6$ |
| | | 1.6 | $2 \times 10^6$ |
| | | 1.8 | $2 \times 10^6$ |

It is seen that the free-flight results are in good agreement with the tunnel tests, and the experimental curve has the same general shape as the theoretical, although the linearised theory overestimates the drag. In the range $0.3 < \cot\Lambda \cot\mu < 0.7$ say, where a closer agreement with theory was expected, the theory overestimates the wave drag coefficient by 35% or more. No explanation can at present be given for this lack of quantitative agreement between theory and experiment. Linearised theory breaks down as sonic velocity is approached and also as the Mach number at which the leading edge becomes sonic is approached, so the lack of agreement in these regions was not unexpected. It is worth pointing out that at the Mach number at which the leading edge becomes sonic there is no tendency for a peak in the drag coefficient against Mach number curve (see Fig.5). This is in agreement with previous experience with wings having leading edge sweepback.

In order to complete the picture a comparison is given in Fig.7 between the drag of the present wing and of a wing of identical plan-form, having a standard round-nosed section (R.A.E.101) and a uniform thickness/chord ratio (0.06 throughout). The thickness/chord ratio of the present wing varies across the span and if it is assumed that the drag at any spanwise station is proportional to the product of the square of the local thickness/chord ratio and the local chord, it can be shown, that this wing has a mean weighted thickness/chord ratio* of 0.068. In Fig.7 the results are plotted in the form $C_D/\tau^2$ against Mach number, where $C_D$ is the wave drag coefficient based on exposed wing area and $\tau$ is the mean thickness/chord (0.06 for the R.A.E.101 section wing and 0.068 for the Square section wing). The maximum thickness line of the present wing is shown in Fig.1, the maximum thickness being at 0.5 chord at the root, 0.366 chord at mid span and 0.333 chord at the tip. The R.A.E.101 section has its maximum thickness at 0.31 chord. It is seen that on this basis of comparison the supersonic drags are in good agreement.

## 4 Conclusions

1 The flight test on the present wing is in good agreement with tunnel tests on another wing of the family.

2 Both methods give wave drags less than theory.

3 If allowance is made for the spanwise variation in thickness/chord ratio of the present wing the experimental total drag at supersonic speeds ($1.0 < M < 1.4$) is in good agreement with the drag of a wing of identical planform having R.A.E.101 section and a uniform thickness/chord ratio.

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$$* \ \tau = \left[ \int_0^s (t/c)^2 \, c \, dy \Big/ \int_0^s c \, dy \right]^{\frac{1}{2}}$$

## REFERENCES


| No. | Author | Title, etc. |
|---|---|---|
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| 3 | Lawrence, Swan and Warren | Development of a Transonic Research Technique using Ground-launched Rocket-boosted Models. Pt.II Drag Measurements. A.R.C. 14,167 - March, 1951. |
| 4 | Poulter | Comparative Drag Tests on Three Related 45° Delta Wings using Ground-Launched Rocket-boosted Models. RAE Tech. Memo. No. Aero 117, Aug. 1950. |


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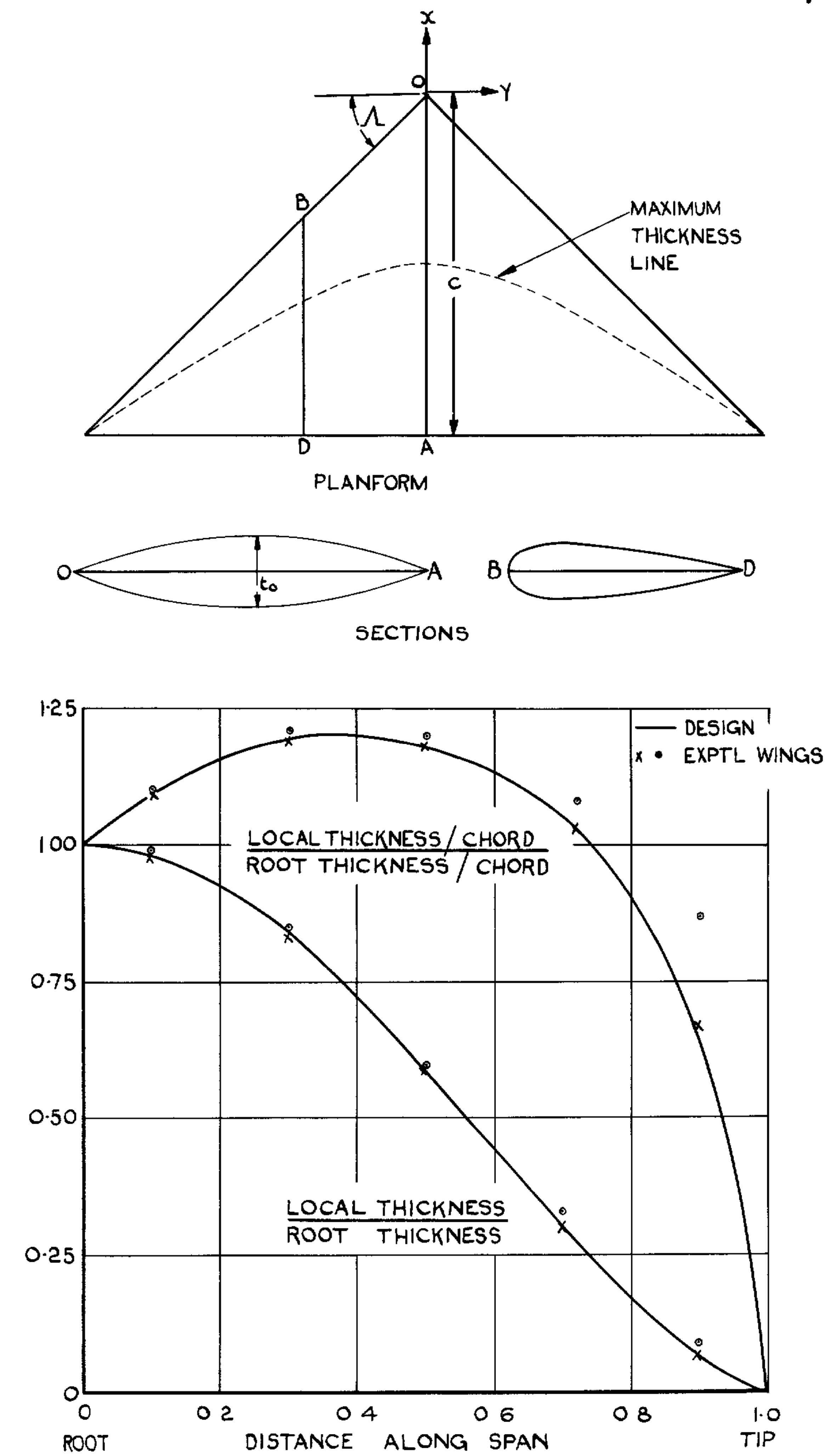
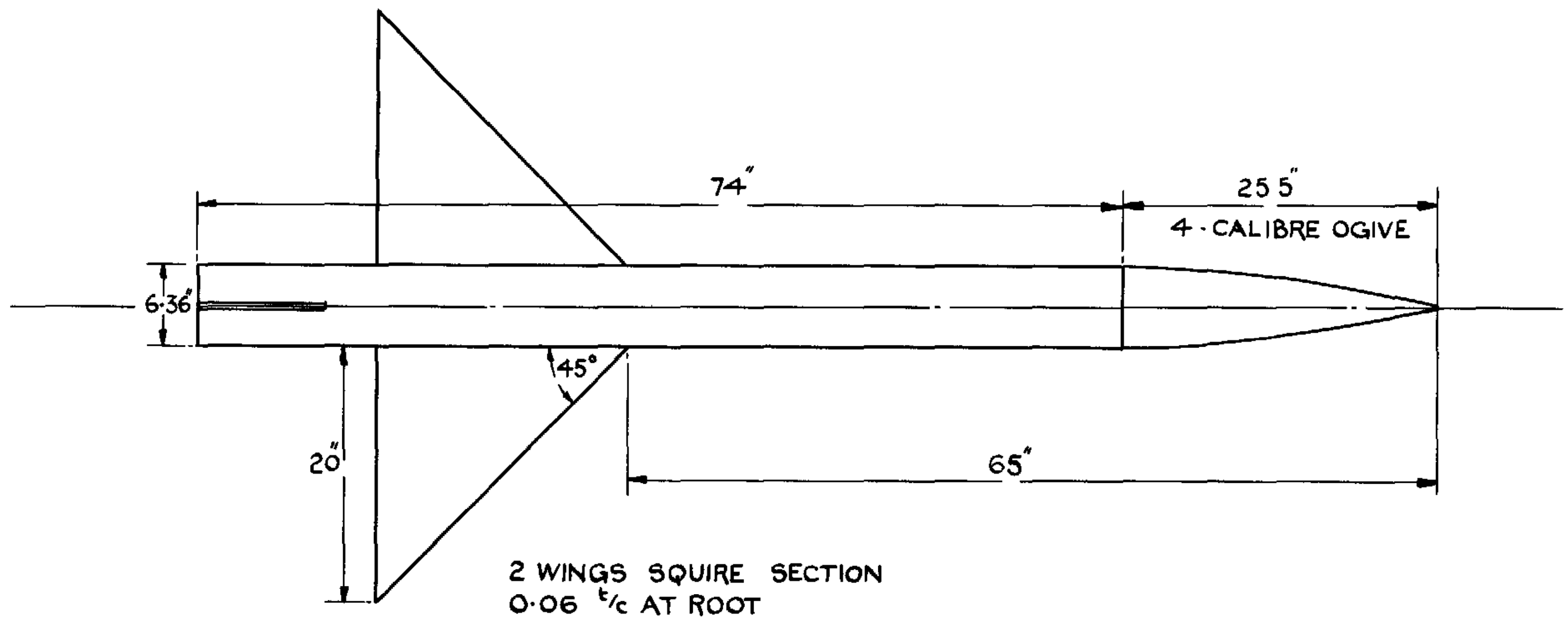



FIG. 1 GEOMETRIC CHARACTERISTICS OF THE WING.



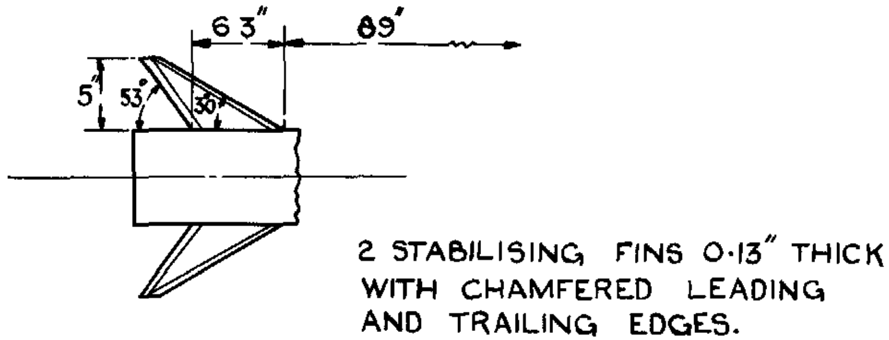




FIG. 2 TEST VEHICLE DIMENSIONS.

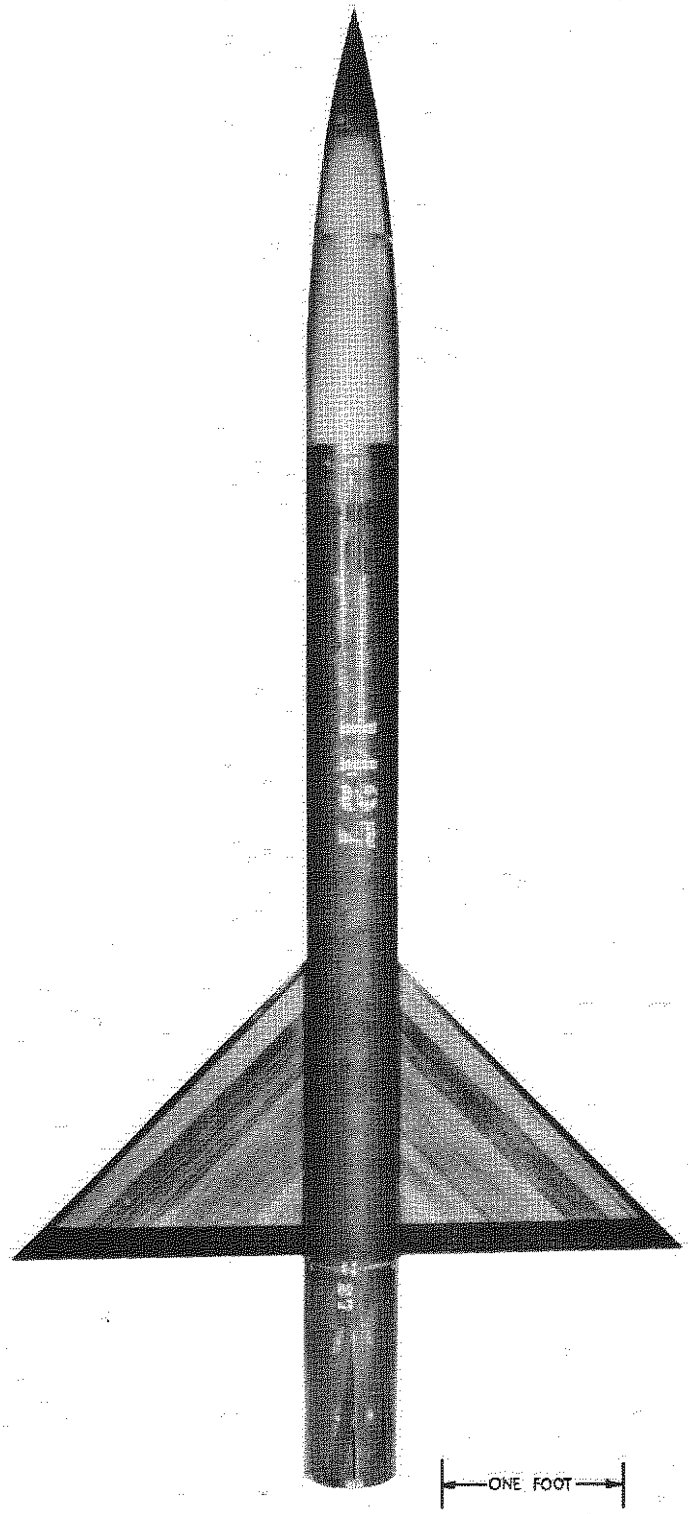



FIG.3. TEST VEHICLE

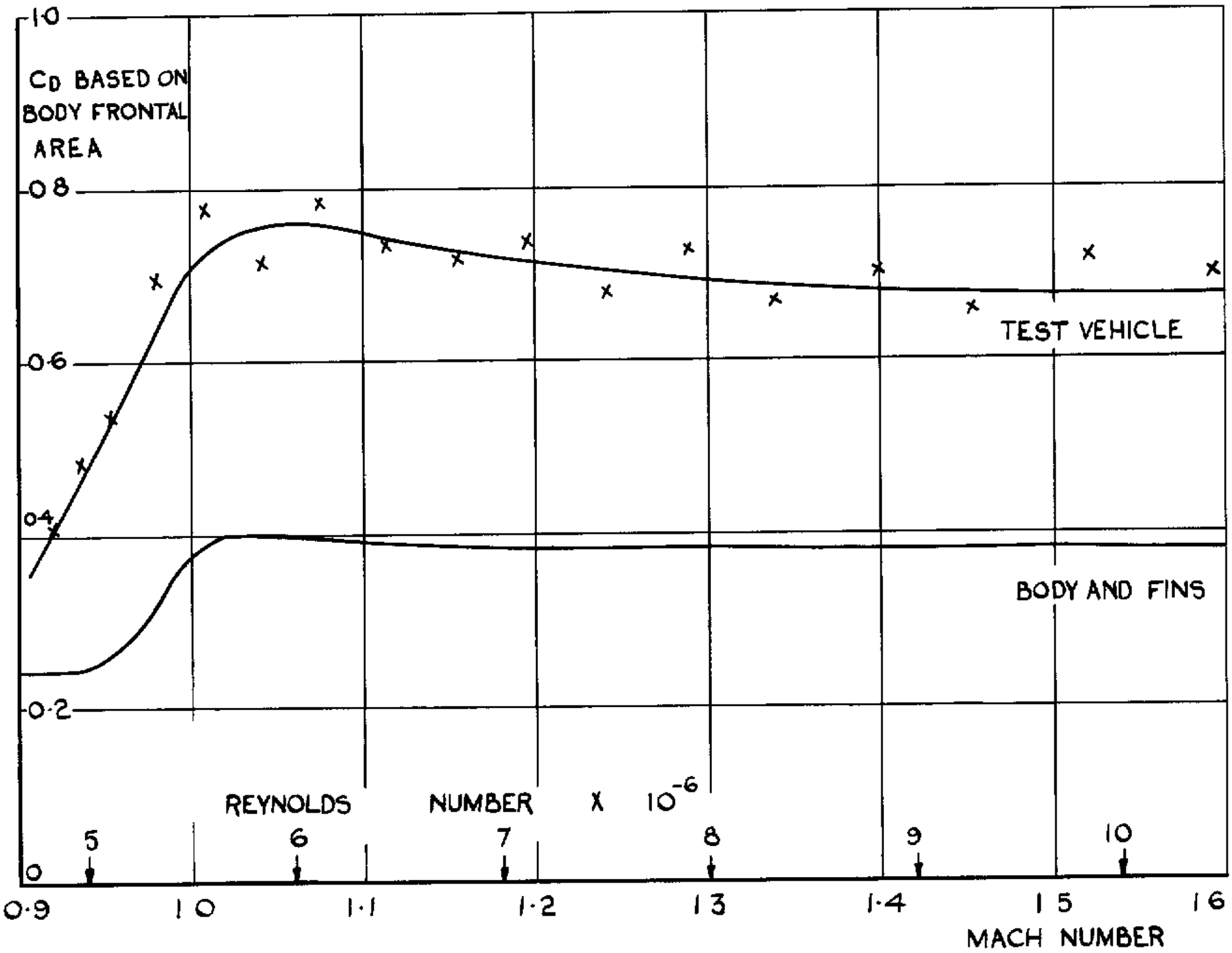



FIG.4. DRAG COEFFICIENT OF THE TEST VEHICLE.

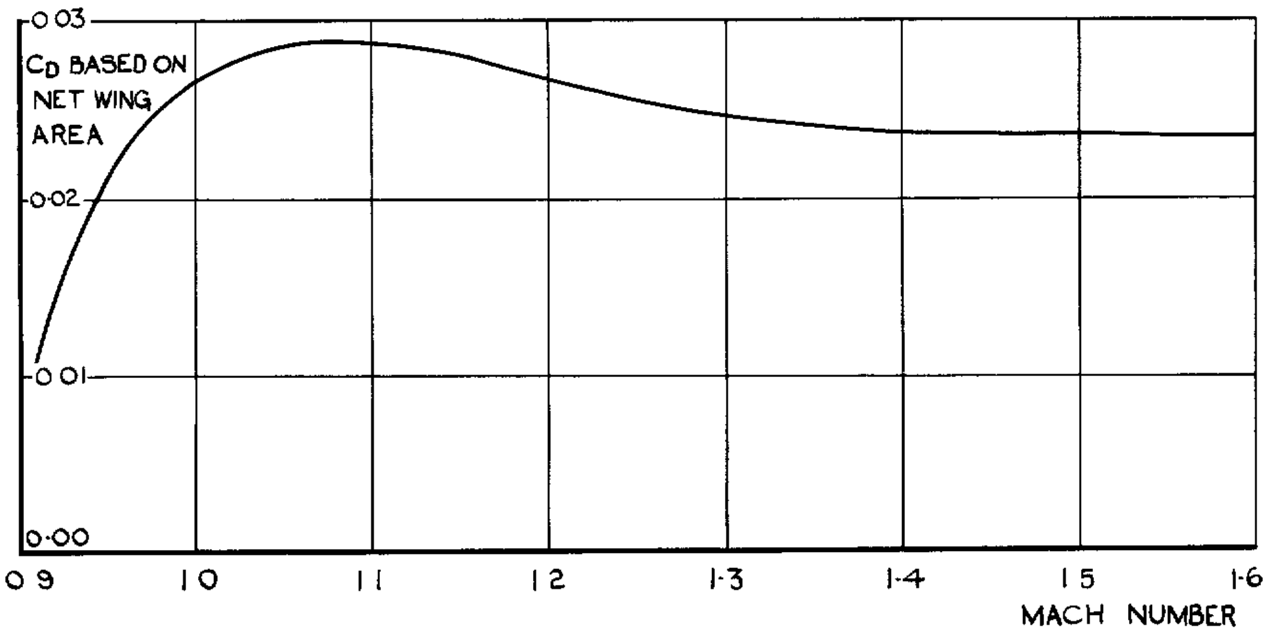
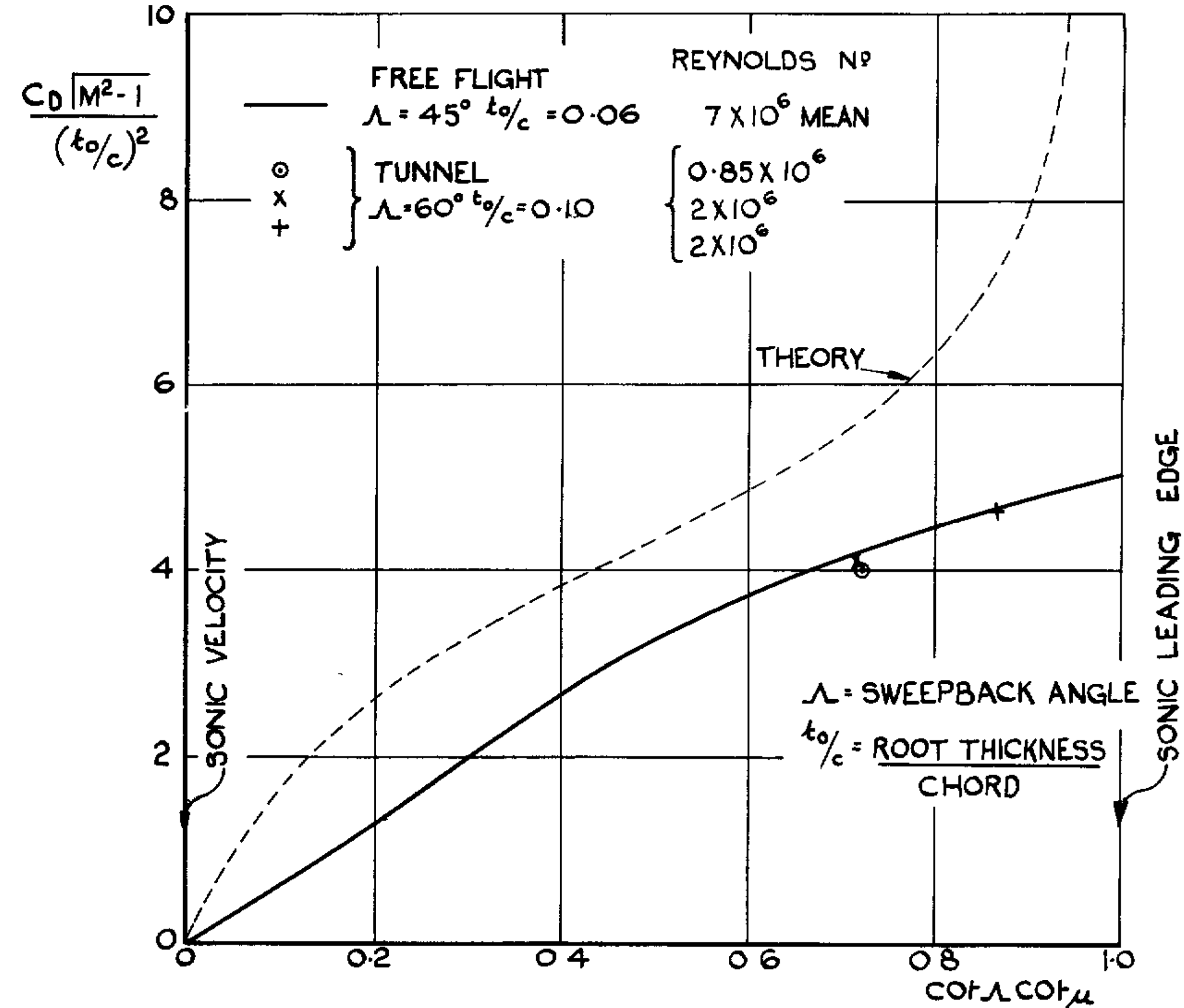
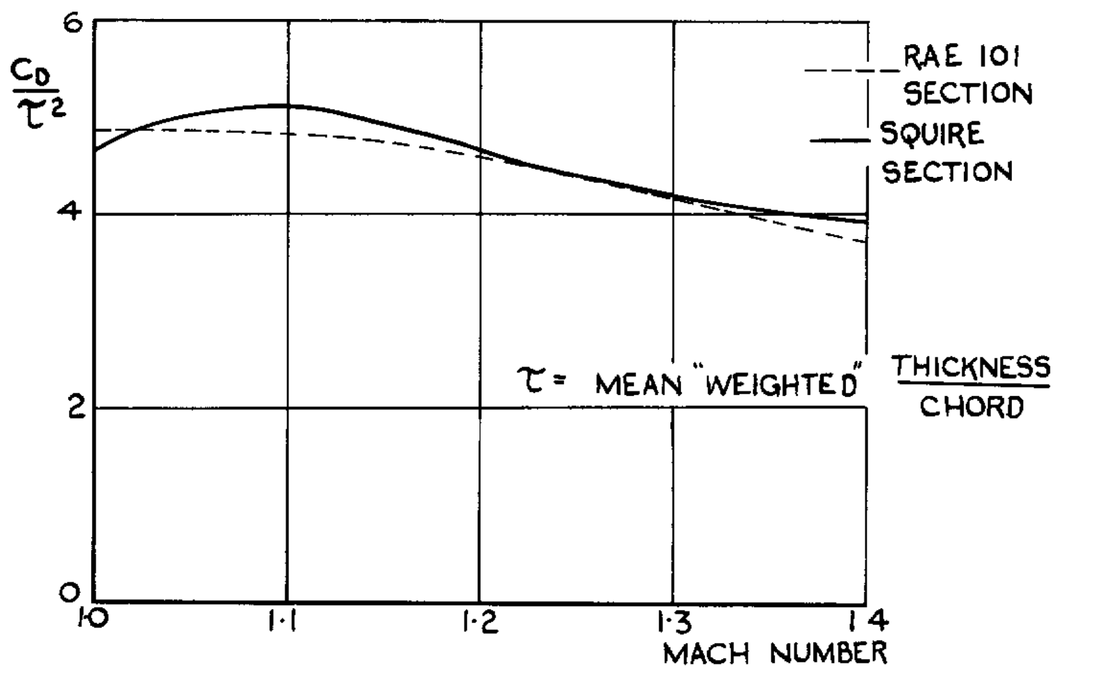



FIG. 5. DERIVED WING DRAG COEFFICIENT.
(THIS INCLUDES SKIN FRICTION.)




# FIG. 6 COMPARISON OF THEORETICAL AND EXPERIMENTAL WAVE DRAGS.

NB 1. 0·005 HAS BEEN SUBTRACTED FROM THE FREE-FLIGHT $C_D$ TO ALLOW FOR SKIN FRICTION.
2. THE TUNNEL RESULTS WERE OBTAINED FROM PRESSURE DISTRIBUTION MEASUREMENTS.




# FIG. 7 COMPARISON WITH A SIMILAR WING.

HAVING AN IDENTICAL PLANFORM AND RAE 101 SECTION WITH UNIFORM THICKNESS / CHORD = 0·06.

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