C.P. No. 742

C.P. No. 742

MINISTRY OF AVIATION

AERONAUTICAL RESEARCH COUNCIL

CURRENT PAPERS

Use of a Wind Tunnel to Determine the Performance of Slender Wings Suitable for a Supersonic Transport Aircraft

by

J. Y. G. Evans

LONDON: HER MAJESTY'S STATIONERY OFFICE

1964

PRICE 4s 6d NET

U.D.C. No. 629.137.1 : 533.6.011.5 : 533.693.3 : 533.6.071.011.5

C.P. No. 742
March, 1963

USE OF A WIND TUNNEL TO DETERMINE THE PERFORMANCE OF SLENDER WINGS SUITABLE FOR A SUPERSONIC TRANSPORT AIRCRAFT

by

J. Y. G. Evans

SUMMARY

Drag coefficients obtained from surface pressure measurements are compared with those derived from balance measurements, both with free and with artificially fixed boundary layer transition, in order to judge whether such wind tunnel tests can be used to predict the performance of slender wings of the types being studied for supersonic transport aircraft.

While a satisfactory level of accuracy can be achieved in most cases at zero lift, a better understanding of the boundary layer behaviour over such wings at incidence is required before results on a lifting model can be extrapolated to full scale with full confidence.

Replaces R.A.E. Tech. Note No. Aero 2884 - A.R.C. 24818.

LIST OF CONTENTS

		Page
1	INTRODUCTION	3
2	EXPERIMENTAL DETAILS	4
	2.1 Description of the models	4
	2.2 Transition detection	4
	2.3 Transition fixing	4
	2.4 Range of tests and estimated accuracy	7
3	ESTIMATION OF SKIN FRICTION DRAG	8
	3.1 Fully turbulent boundary layer	8
	3.2 Free transition	9
4	DRAG MEASUREMENTS	10
5	CONCLUSIONS	11
6	ACKNOWLEDGEMENTS	11
LIST OF REFERENCES		12
ILLUSTRATIONS - Figs.1-7		-
DETACHABLE ABSTRACT CARDS		-

LIST OF ILLUSTRATIONS

	Fig.
Model A	1
Model B	2
Model C	3
Sublimation patterns - free transition	4
Model A. Drag variation with Reynolds number	5
Model B. Drag variation with Reynolds number	6
Model C. Drag variation with Reynolds number	7

1 INTRODUCTION

Drag can be determined either by integrating surface pressures measured in a wind tunnel and adding a skin friction term estimated for the flight conditions or by measuring the total drag in the wind tunnel and subtracting an estimated difference in skin friction drag between tunnel and flight Reynolds numbers. The first method is usually more informative and the pressure distributions can be compared directly with theoretical estimates, but often a very large number of pressure points are required to give adequate accuracy. Consequently, it is a more common practice to measure to overall forces on the model and to attempt to estimate the viscous drag at the tunnel Reynolds number as well as for the flight case. For slender wing designs of the types being studied for supersonic transport aircraft at Mach numbers between 2 and 3, the skin friction drag in the wind tunnel may form a large portion of the total drag (sometimes as much as 70% of the drag at zero lift), and in this Note drag values from balance measurements are compared with those obtained from surface pressure measurements in order to judge how successfully such wind tunnel data can be used to predict the full scale performance of these designs.

In order to estimate the viscous drag at the tunnel Reynolds number, it is necessary first to locate the regions of laminar and turbulent boundary layer. If transition is allowed to occur naturally on the model, this can be done by observing the rate of sublimation of a surface film of a suitable chemical, but as the model must be resprayed for each condition of test the technique is used more easily in a small intermittent tunnel than in a pressurised continuous-operation tunnel. Partly for this reason and partly because there is a danger that regions of laminar flow which would not be present full scale may modify the external flow, e.g. by affecting flow separations or shock-wave/boundary layer interaction, an alternative technique has been developed where transition is artificially induced near the leading edge by a narrow band of distributed roughness applied to the surface. The roughness, which consists of carborundum grit or Ballotini (spherical glass beads) should be just large enough to cause transition close behind the band. Use of a roughness band obviates the need for repeated investigation of the transition front, but raises other problems for, in addition to the increase in skin friction drag with forward movement of the transition region, some allowance must be made for the drag of the roughness including any changes in surface shear due to distortion of the boundary layer profile just downstream of the roughness.

Balance measurements both with free and with fixed transition and also pressure distributions have been obtained on a number of slender wings in the Royal Aircraft Establishment 8 ft × 8 ft wind tunnel, and results for three of these wings are compared over a range of Reynolds numbers at a Mach number of 2.0. The wings chosen are (1) an uncambered slender ogee of smooth planform and thickness distribution (type C of Ref.1), (2) another uncambered slender ogee with a less smooth planform and thickness distribution[2] and (3) a thin-wing/body arrangement which is one version of the British Aircraft Corporation/Sud Aviation 'Concorde'.

2 EXPERIMENTAL DETAILS

2.1 Description of the models

General views and some relevant dimensions of the three models are given in Figs.1, 2 and 3. For the present investigation, the models are identified as A, B and C, where A and B are the two ogee wing designs and C is the wing/body layout.

The ogee wings are slender (Mach number normal to the leading edge does not exceed 0.65 at a free stream M = 2.0), the leading edges are sharp and the thickness distributions are chosen to give low wave drag and to cause little or no adverse pressure gradients on the surface. The planform of model A is close to a delta except for the streamwise tips, but that of model B is more 'waisted' and the forepart is practically a body with a small chine which develops into a wing further aft. Model C has a thin wing (less than 3% thickness-chord) which is mounted low on a cylindrical body; the planform is less slender (Mach number normal to the leading edge is close to unity at free stream M = 2.0), the wing is slightly cambered and has rounded leading edges of small radius.

Thus the models A, B and C have flow characteristics of increasing complexity. On model A, surface contour changes are very gradual and, except in the immediate vicinity of the leading edge, the velocity perturbations do not exceed 3%. Model B is intermediate in the sense that it goes some way towards the central body and thin wing complexity of C, but avoids any abrupt changes in slope or any steep pressure gradients.

All three models were mounted in turn on the same 2.25in.diameter internal strain-gauge balance and 2.1 in. diameter rear sting. On model C, the sting emerges from the truncated rear end of the body, but models A and B are distorted by a cylindrical sting shroud which extends to the wing trailing edge. Surface pressures have been measured with this shroud present and consequently the wave drags quoted are lower than those for the clean wings.

2.2 Transition detection

The location of the natural transition front on each surface of the wings without roughness bands was obtained by a sublimation technique[3], using acenapthene as an indicator. The tunnel temperature is kept low in order to delay the sublimation process until the required speed, density and model attitude are reached. Usually the pattern is formed about half to one hour after reaching test conditions and a photographic record is made while the tunnel is running.

2.3 Transition fixing

The band of roughness used to promote early transition was placed 0.10 in. clear of the wing leading edge, measured in the direction normal to the edge, and various widths of band have been tried from 0.5 in. down to 0.1 in. Ideally the grade of roughness used should vary with the conditions of test, the optimum size increasing with increasing Mach number and decreasing with increasing Reynolds number. Current practice in the R.A.E. 8 ft × 8 ft tunnel is to use Grade 60 carborundum grit throughout the range of supersonic tests on slender wings as this size has been found by experience to be just sufficient to move the transition front forward at the higher end of the Mach number range ($M = 2.8$, Rey. No./ft approx. 2.2×10^6). The grains are sprinkled on to a thin layer of Araldite adhesive (about 0.001 in. thick), the band position and width being controlled by temporarily masking the neighbouring surface. The average density of particles is about 200 per square in., i.e. less than 2% of the adhesive is covered by the grit.

Carborundum grains have an irregular shape and although the nominal grain size of 60 grade is 0.0098 in., careful sieving of a typical commercial sample showed that 27% was caught by a No.52 mesh screen (aperture 0.0116 in.) 68% passed through this screen, but was caught by a No.72 mesh screen (aperture 0.0083 in.) while the remaining 5% passed through both screens. Sieved particles caught between the two screens are used, but obviously particles exceeding 0.0116 in. in one dimension can be included and these may stand on end in the roughness band. Consequently it is difficult to establish an effective roughness height with any precision. Inspection of a sample length of band revealed that:-

3% of the particles were over 0.014 in. high,

9% of the particles were between 0.012 in. and 0.014 in. high,

15% of the particles were between 0.010 in. and 0.012 in. high, and

73% of the particles were less than 0.010 in. high.

Thus not only do some particles stand on end to give roughness heights up to 0.015 in. or so, but the bulk of the grains lie flat on the surface and contribute little to the transition process. Carborundum grit is not ideal for this purpose and a roughness element of a more uniform size and shape is desirable. Ballotini (spherical glass beads) have been tried with some success, but a better solution may be to use a prefabricated roughness consisting of a single row of suitably shaped (triangular planform?) elements at appropriate spacing, which are either etched or moulded on a very thin backing strip of metal or plastic.

Several attempts have been made to establish generalised data for the minimum height of distributed roughness necessary to cause premature transition. At low speeds and for roughness heights which are appreciably smaller than the local boundary layer height, experimental data correlates well when expressed in terms of a roughness Reynolds number, R_k, formed from the roughness height, k, and the flow conditions in the undisturbed boundary layer at the position of the top of the roughness element. A similar approach has been tried at supersonic speeds, for example in Ref.4 Braslow and Knox recommend a value of R_k 'slightly larger than 600' for transition movement in the absence of pressure gradients or heat transfer at Mach numbers up to 5. In a later paper[5], Braslow notes that larger values of the critical Reynolds number may be needed when the roughness height is equal to or greater than the boundary layer thickness.

At subsonic speeds, the transition front moves forward to the roughness band following a slight increase in Reynolds number above the critical value at which turbulent spots first appear in the wake of the roughness. At higher speeds, the forward movement may be less rapid and a very large increase in R_k may be needed to bring the transition right up to the band. Van Driest and Blumer[6] have shown clearly that for a single row of roughness elements on a 10° cone at $M = 2.7$, the transition front moves forward rapidly with increase of R_k until it is at a distance behind the elements equivalent to a Reynolds number of 0.5×10^6 but very little change in this value occurs with a further increase of two or three times in unit Reynolds number. At $M = 2.0$, the corresponding distance has a Reynolds number which is thought to be about 0.3×10^6. Potter and Whitfield[7] have attempted to use the very limited evidence available to form generalised curves for predicting the forward movement of the transition front and conclude that, at $M = 3$ say, a value of R_k of about 10,000 is required to bring transition right up to the roughness. It is doubtful whether such large values of roughness have any practical significance because the associated roughness drag and the distortion of the boundary layer profile would be excessive. A smaller size of roughness appears to be preferable even if the transition front then occurs at a short distance behind the roughness band.

Roughness heights calculated from the data given by the authors mentioned are shown in the following table together with an estimate of the local boundary layer height, δ. The distance from the leading edge to the effective roughness measured in the streamwise direction is taken to be 0.5 in.

TABLE 1

Estimated boundary layer and minimum roughness height

		M = 2.8	M = 2.0		
		R/ft = 2.2×10^6	R/ft = 1×10^6	1.5×10^6	2×10^6
Boundary layer height	δ in.	0.013	0.016	0.013	0.011
Braslow and Knox $R_k > 600$	k in.	> 0.009	> 0.011	> 0.0085	> 0.007
Van Driest and Blumer (Transition just dowstream of roughness)	k in.	0.0105	0.013	0.010	0.008
Potter and Whitfield (Transition at roughness)	k in.	0.052	0.035	0.023	0.017

The calculations cover a range of Reynolds numbers/ft from 1 to 2×10^6 because balance measurements on the ogee wings with 60 grade carborundum suggest that most of the transition movement occurs in the neighbourhood of 1.5×10^6. At this condition and also at the upper end of the speed range, the effective height of the carborundum is close to the boundary layer height; it is slightly larger than the height given by the $R_k = 600$ criterion or by the Van Driest and Blumer formulae, but it is considerably less than that suggested by Potter and

Whitfield. In making these comparisons it should be remembered that the high leading edge sweep of these models, the short distance of the band from the leading edge and local pressure gradients (especially on model C which has a rounded leading edge) can have a considerable effect on the transition process.

2.4 Range of tests and estimated accuracy

Surface pressures have been measured on models A and B only. Tests were at a Reynolds number of 2×10^6/ft and the wave drag at $M = 2.0$ is taken from mean curves through data obtained over a range of Mach numbers from 1.4 to 2.8. Each model had approximately 100 pressure holes placed so as to give the best information for subsequent integration. Examination of possible experimental inaccuracies suggests that any errors in the drag coefficients should not exceed ±0.0001.

Balance measurements at $M = 2.0$, at zero incidence have been made on wing A at Reynolds numbers of 2.1×10^6/ft and 4.3×10^6/ft allowing free transition, at Reynolds numbers from 2 to 4.6×10^6/ft with an 0.1 in. wide roughness band and at 2×10^6/ft with an 0.25 in. wide band. Corresponding measurements have been made on wing B at a Reynolds number of 2.5×10^6/ft allowing free transition, at Reynolds numbers between 2 and 4.5×10^6/ft with an 0.1 in. wide roughness band and at 2 and 3×10^6/ft with a 0.5 in. wide band. The 0.1 in. wide band on wing B extended only over a short distance near the apex ($0 \leq x/c_0 \leq 0.15$) and over the rear ($0.45 \leq x/c_0 \leq 1$) of the leading edge because the free transition test showed that the sweep and thickness distribution was sufficient to make the boundary layer self-contaminating over the forepart of this wing.

Model C has been tested at the same Mach number but at both $C_L = 0$ and 0.1. Measurements have been made at a Reynolds number of 4.6×10^6/ft allowing free transition, at Reynolds numbers between 2 and 4.6×10^6/ft with an 0.1 in. wide roughness band and at 2.15×10^6/ft with an 0.25 in. wide band. Free transition on this model was restricted to the wing surfaces and transition was fixed on the body by a narrow band of roughness near the apex.

In all cases, a mean result is quoted for each model tested both right-way-up and inverted. Possible errors in the drag coefficients are estimated to be less than ±0.0002 at the lower end of the Reynolds number range and less than ±0.0001 at the highest Reynolds numbers.

For later reference, the ratio of the area of the roughness band to the reference area of the wing for each of the configurations tested is given in the following table:-

TABLE 2 - Ratio of roughness band area to reference wing area

Model	Roughness band area/wing area		
	0.5 in. band	0.25 in. band	0.1 in. band
A	-	0.072	0.029
B	0.18	-	0.025*
C	-	0.065	0.026

*minimum length band ($0 \leq x/c_o \leq 0.15$ and $0.45 \leq x/c_o \leq 1$).

3 ESTIMATION OF SKIN FRICTION DRAG

3.1 Fully turbulent boundary layer

The skin friction drag for a fully turbulent boundary layer has been determined by calculating that for a flat plate of the same planform, using strip integration, and increasing the flat plate value in the ratio of total wetted areas of model to flat plate. (In one case, model B, a local thickness factor was applied to each strip before integration but the resulting drag was close to the value obtained by using an overall thickness factor.) The method cannot be applied with any confidence to complex three-dimensional shapes having high local curvatures and steep pressure gradients; for instance unpublished investigations in the 8 ft × 8 ft tunnel on a delta wing with adverse pressure gradients due to camber have indicated local values of surface shear which are less than half of the corresponding flat plate values and while such low shear regions are usually partly offset by high shear elsewhere, it is not difficult to believe that the overall friction drag estimate could be in error by 10% or more. However, models A and B are smooth and at zero incidence are free from large pressure gradients and the 'flat plate' estimate should not be much in error. Model C has a greater measure of uncertainty and it is unfortunate that the pressure integration is not available to provide a comparison in this case.

A major difficulty arises on all the slender wings when incidence is increased. Separated flow from the highly swept leading edges (and for the body on wing/body arrangements) produces strong coiled vortex sheets above the wing with consequent high scrubbing of the local surface. The distribution of surface shear is modified and the total skin friction may differ considerably from that at zero incidence. There is little, if any, experimental evidence to indicate the order of difference possible. Present practice is to apply the whole of the drag difference due to incidence measured in the tunnel to the full scale prediction, i.e. to make no allowance for the possibility that induced drag factors obtained in this way may contain an element of skin friction drag which is sensitive to Reynolds number.

Apart from difficulties arising from the three-dimensional character of the flow, there is still some doubt about the drag of flat plates. The preferred formulae is that given by Prandtl and Schlichting[8] for low speeds with the intermediate enthalpy method of Eckert and Monaghan[9] to allow for compressibility,

$$\bar{C}_f = 0.455 \frac{T_e}{T^*} (\log_{10} Re^*_x)^{-2.58}$$

where $\bar{C}_f$ is the mean skin friction coefficient, T_e is the absolute temperature at the edge of the boundary layer and T^* and Re^*_x are evaluated at the temperature corresponding to intermediate enthalpy. At the tunnel Reynolds numbers, this formula gives lower values than would be estimated from the Royal Aeronautical Society Data Sheet curves[10] which are derived from Young and Kirkby[11]. Corresponding estimates at $M = 2.0$ for both tunnel and flight Reynolds numbers are tabled below:-

TABLE 3

Estimated turbulent skin friction drag

Reynolds number	Model A		Model B		Model C	
	P.-S.	R.Ae.S.	P.-S.	R.Ae.S.	P.-S.	R.Ae.S.
2×10^6/ft	0.00526	0.00549	0.00535	0.00556	0.00769	0.00801
3×10^6/ft	0.00491	0.00509	0.00498	0.00515	0.00716	0.00744
4×10^6/ft	0.00467	0.00484	0.00473	0.00487	0.00683	0.00706
5×10^6/ft	0.00450	0.00464	0.00452	0.00464	0.00657	0.00677
Full scale					0.00409	0.00395

The differences at full scale Reynolds numbers are smaller but of opposite sign and thus use of the Prandtl-Schlichting formula gives a prediction of the full scale performance obtained from, say, the 2×10^6/ft tunnel results which is about 0.0004 in drag coefficient less favourable than that obtained from the R.Ae.S. data sheets.

3.2 Free transition

The skin friction component in the tests with free transition will not be far below the value for a fully turbulent boundary layer because the regions of laminar flow indicated by the sublimation technique are not very extensive on any of the models; see Fig.4. But the magnitude of the difference cannot be estimated precisely without some knowledge of the surface shear in the transition region and particularly of the relationship between the shear and the sublimation pattern. Winter, Scott-Wilson and Davies[12] have compared azobenzine pictures with surface-pitot measurements on a 10^o cone at $M = 3.25$ and conclude that the boundary indicated by the sublimation pattern occurs at the upstream end of a gradual rise in mean shear from the laminar to the turbulent level. It could be argued that sublimation would occur most rapidly in the region of highest shear, i.e. at the start of the fully turbulent boundary layer, which may be a distance equivalent to a Reynolds number of about 10^6 further downstream, but a natural transition front fluctuates rapidly and the mechanism of sublimation in this region is not understood. Consequently the skin friction differences necessary to correct the transition free results to the fully turbulent case have been calculated in three alternative ways:-

(1) by assuming that the local skin friction jumps directly from the laminar boundary layer value to the turbulent value, having the same momentum thickness, at the sublimation front,

(2) by assuming that the local friction rises linearly with distance downstream from the laminar value at the sublimation front to the turbulent value, having the same momentum thickness, at a distance downstream equivalent to a Reynolds number of 10^6.

(3) by assuming that (1) applies in regions where turbulence is spread by lateral contamination and (2) applies in regions where transition occurs naturally, as far as it is possible to distinguish these areas by close examination of the sublimation pictures.

TABLE 4

Corrections to transition free measurements

Assumption (see above)	Model A		Model B	Model C	
	2.1×10^6/ft $C_L = 0$	4.3×10^6/ft $C_L = 0$	2.5×10^6/ft $C_L = 0$	4.35×10^6/ft $C_L = 0$	4.6×10^6/ft $C_L = 0.1$
(1)	0.00065	0.00021	0.00046	0.00036	0.00042
(2)	0.00092	0.00034	0.00069	0.00052	0.00065
(3)	0.00088	0.00032	0.00063		

4 DRAG MEASUREMENTS

Drag values obtained by the different techniques are compared in Figs. 5, 6 and 7 for models A, B and C respectively. Total drag coefficients are shown, plotted against Reynolds number. The balance measurements for the models with roughness are plotted directly while those for the models without roughness are adjusted to correspond to the fully turbulent boundary layer case by adding the estimated skin friction differences from Table 4; a thick vertical line is used to show the range of values resulting from the alternative assumptions about surface shear in the neighbourhood of the sublimation front.

Corresponding total drag coefficients from the pressure plotting tests on Models A and B are obtained by adding the skin friction drag values of Table 3 (Prandtl-Schlichting formula) to the integrated pressure coefficients. These total drag curves are shown as broken lines in Figs. 5 and 6.

Examination of Figs. 5 and 6 reveals that all three methods give consistent results provided that a drag due to the roughness band is assumed which increases in relation to the area of the band. The drag values obtained by pressure plotting plus calculated skin friction and also those obtained from free transition tests differ from a 'best fit' curve, which is drawn as a full-line curve in each figure, by less than the estimated experimental error and the same can be said of the tests with roughness if the correction applied to the experimental points to allow for roughness drag is assumed to be approximately of the form

$$K \left(\frac{\text{roughness band area}}{\text{reference wing area}}\right) .$$

The ratio of areas is taken from Table 2 and the factor K is about 0.004 at the lower end of the Reynolds number range. As the boundary layer becomes thinner with increasing Reynolds number, the velocity and viscosity over the roughness elements change and K becomes larger. This trend is illustrated in Fig. 6 where the roughness drag of the 0.5 in. wide band increases from 0.0005 at 2×10^6/ft to 0.0008 at 3×10^6/ft. However, for the 0.1 in. wide band, which is the size most likely to be used in practice, the roughness drag is only about 0.0001 and it is not necessary to determine K to any great accuracy.

The results on model C, Fig.7, do not follow the turbulent skin friction law at Reynolds numbers below 3×10^6/ft and it seems probable that the transition front is still moving forward at the lower end of the test range. This difference between the behaviour of the first two models and model C may be a consequence of the less slender wing or possibly of the rounded leading edge or camber. Above 3×10^6/ft, the behaviour is similar to that of the other models and at $C_L = 0$ there is reasonable agreement between free and fixed transition values.

Measurements at incidence have been compared on Model C only (at $C_L = 0.1$) and in this case there is a difference between the fixed and free transition values of about 0.0003 which is slightly too large to be explained by experimental inaccuracy. The induced drag factor from the tests with free transition on the wings is about 8% higher than that obtained from the tests with roughness. This lack of agreement may arise either from a change in the external flow characteristics about the wing, in which case the transition fixed tests should be more representative of the full scale flow, or from changes in shear which are not predicted accurately by the theory used for estimation of friction drag. The lift and pitching moment coefficients are in good agreement which would indicate that any flow changes are not extensive, but quite small, local changes near the leading edges could alter the drag coefficient by the amount in question. A more detailed investigation, including measurements of pressure and local surface shear, is required to resolve this discrepancy and also to help towards a more refined method of calculating the skin friction on such wings at incidence.

5 CONCLUSIONS

The experimental evidence at $M = 2.0$ suggests that for slender wings of smooth shape at zero incidence, turbulent skin friction drag can be estimated with sufficient accuracy at tunnel Reynolds numbers and balance measurements with either free transition or artificially provoked transition can give wave drag coefficients within ± 0.0002 or better. For measurements with fixed transition, it is essential that the roughness used is kept to a minimum and, to this end, a more uniform shape and size of roughness element than carborundum grit is desirable.

At incidence, $C_L = 0.1$, the free and fixed transition measurements of drag are not in such close agreement and further investigation is needed to determine whether this implies any difference in flow outside the boundary layer. In general, a much better understanding of the boundary layer behaviour on slender wings at incidence is required before tunnel measurements can be extrapolated with full confidence to full-scale conditions.

6 ACKNOWLEDGEMENTS

The author is indebted to Mr. K.G. Smith for the calculations of skin friction drag, to Mr. A.O. Ormerod for the experimental results on models A and C and to Mr. C.R. Taylor for the experimental results on model B.

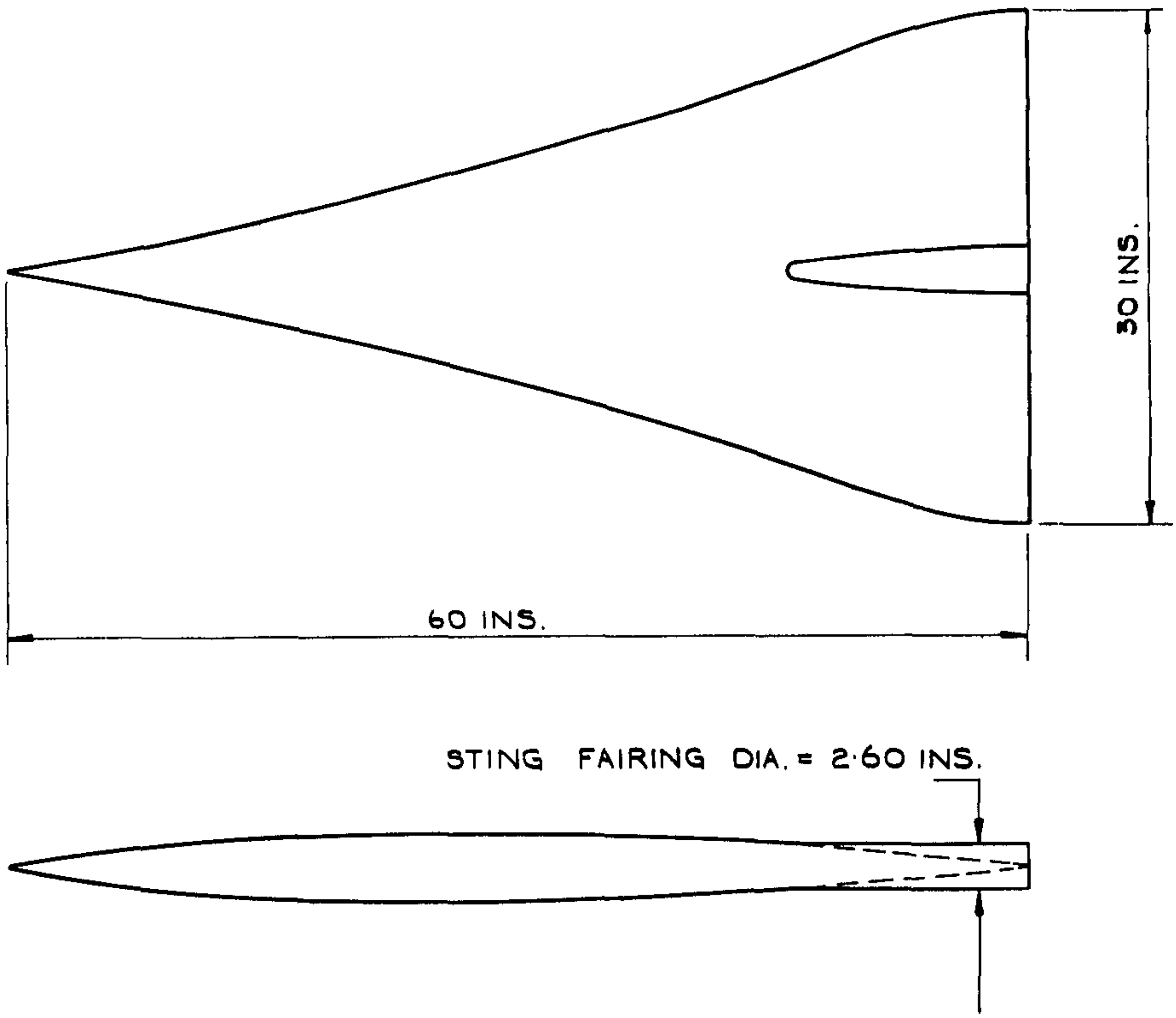
LIST OF REFERENCES

Ref. No.	Author	Title, etc.
1	Küchemann, D. Spence, A.	A brief outline of some of the present aerodynamic work on supersonic transport aircraft. Unpublished M.O.A. Report.
2	Taylor, C.R.	Measurements, at Mach numbers up to 2.8, of the longitudinal characteristics of one plane and three cambered slender "ogee" wings. A.R.C. R.&M. 3328. December, 1961.
3	Main-Smith, J.D.	Chemical solids as diffusible coating films for visual indication of boundary layer transition in air and water. A.R.C. R.&M. 2755. February 1950.
4	Braslow, A.L. Knox, E.C.	Simplified method for determination of critical height of distributed roughness particles for boundary layer transition at Mach numbers from 0 to 5. NACA T.N. 4363, September 1958.
5	Braslow, A.L.	Review of the effect of distributed surface roughness on boundary layer transition. AGARD Report 254, April 1960.
6	Van Driest, E.R. Blumer, C.B.	Effect of roughness on transition in supersonic flow. AGARD Report 255, April 1960.
7	Potter, J.L. Whitfield, J.D.	Effects of slight nose bluntness and roughness on boundary layer transition in supersonic flows. Jnl. Fl. Mech. Part 4, April 1962, 501-35.
8	Schlichting, H.	Boundary layer theory. McGraw-Hill, 4th Ed., 1960.
9	Monaghan, R.J.	Formulae and approximations for aerodynamic heating rates in high speed flight. A.R.C. C.P.360. October, 1955.
10		Data Sheets - Royal Aeronautical Society Aerodynamics, Vol.2. Wings S.02.04.12, issued July 1956.
11	Young, A.D. Kirkby, S.	The profile drag of biconvex and double wedge wing sections at supersonic speeds. Proceedings of the NPL Symposium on boundary layer effects in aerodynamics. HMSO, 1955.

LIST OF REFERENCES (CONTD)

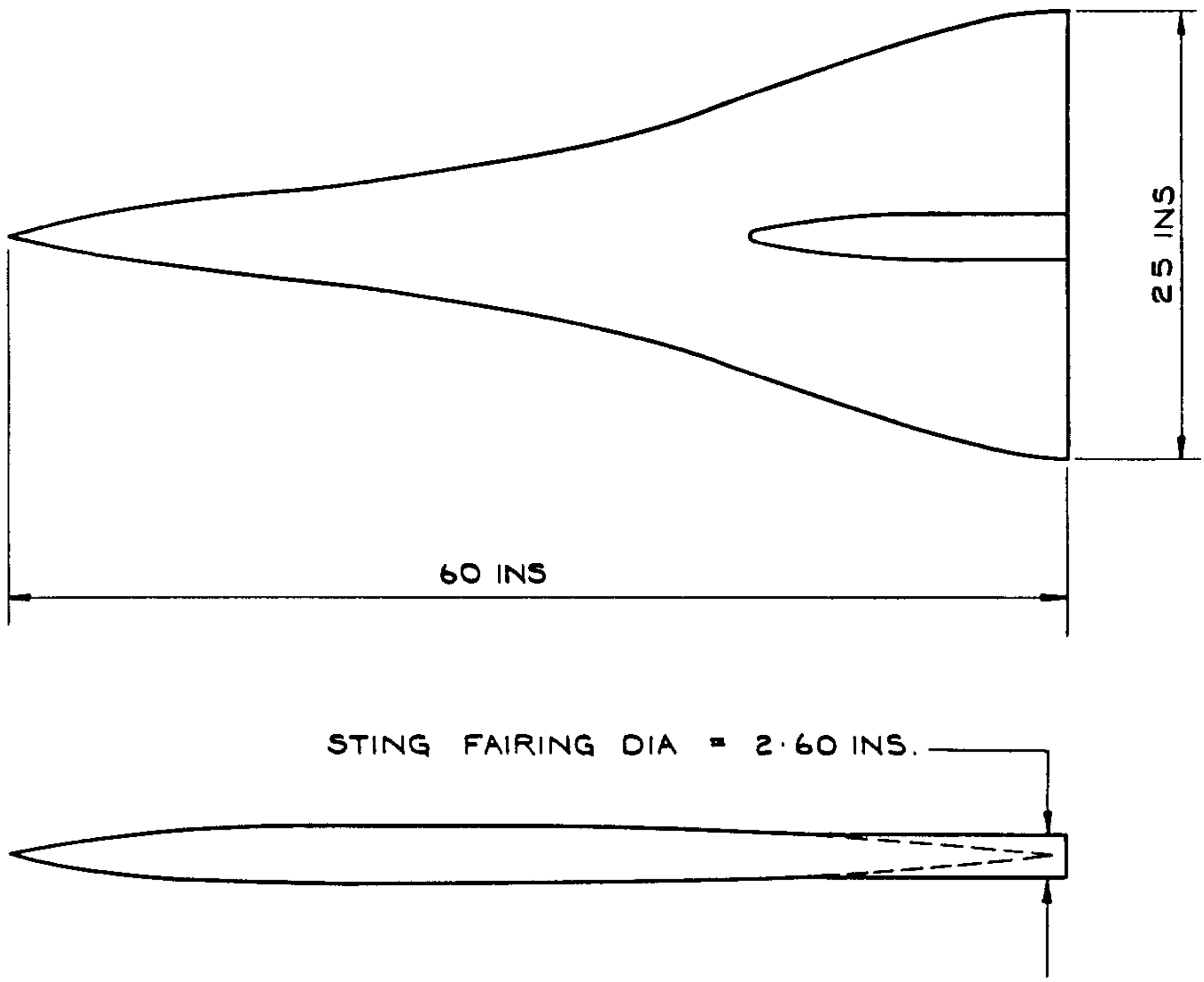
Ref. No.	Author	Title, etc.
12	Winter, K.G. Scott-Wilson, J.B. Davies, F.V.	Methods of determination and of fixing boundary layer transition on wind tunnel models at supersonic speeds. A.R.C. C.P.212. September, 1954.

Printed in England for Her Majesty's Stationery Office by the Royal Aircraft Establishment, Farnborough. W.T.60.K4.



PLAN AREA = REFERENCE AREA	=	855 SQ INS
ASPECT RATIO	=	1·053
$P = \dfrac{\text{PLAN AREA}}{\text{LENGTH} \times \text{SPAN}}$	=	0·475
$\dfrac{\text{VOLUME (EXCLUDING STING FAIRING)}}{\text{LENGTH}^3}$	=	0·00421
$\dfrac{\text{WETTED AREA (INCLUDING STING FAIRING)}}{\text{REFERENCE AREA}}$	=	2·10

FIG 1 MODEL A.



PLAN AREA = REFERENCE AREA = 674 SQ INS.

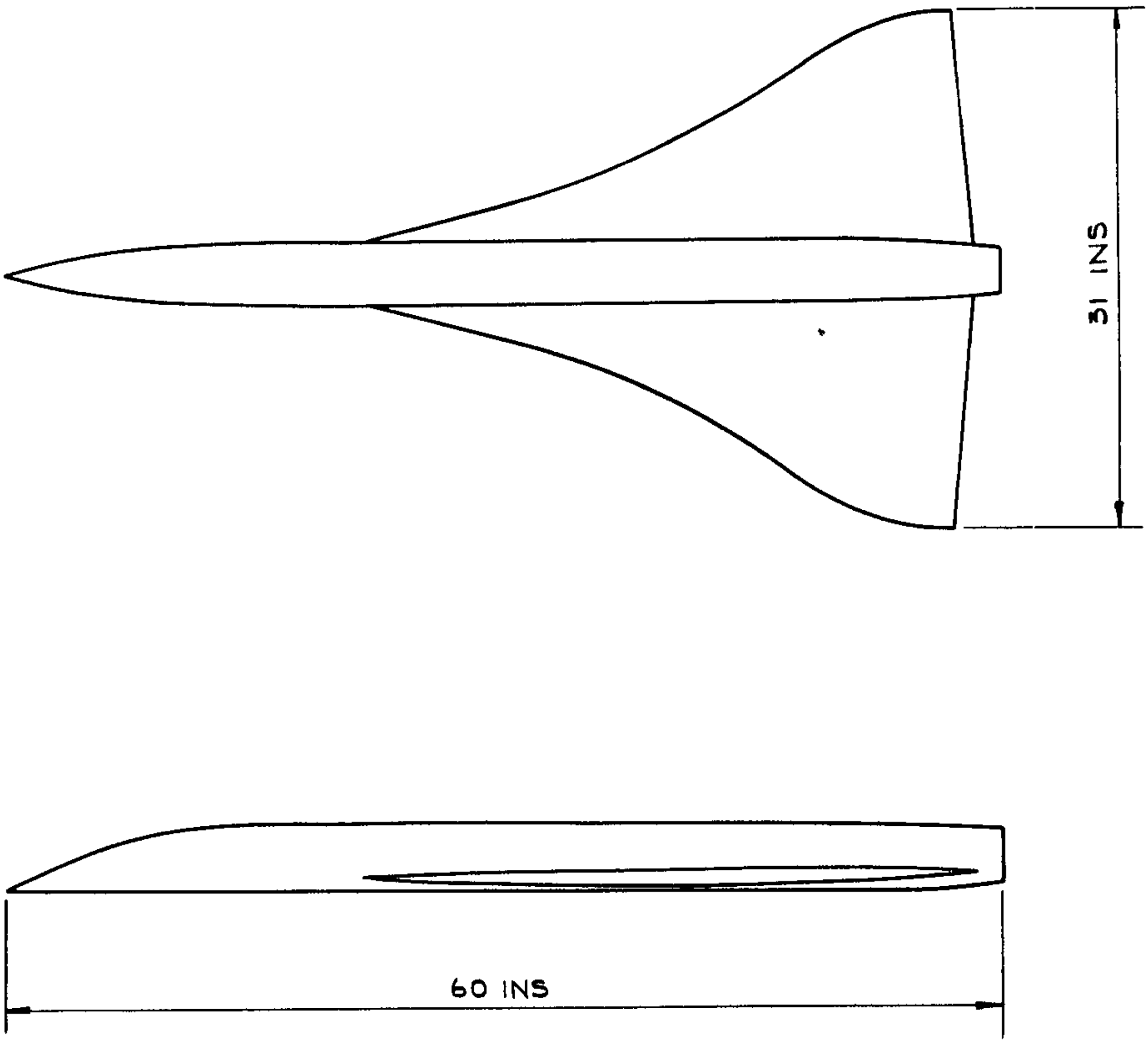
ASPECT RATIO = 0·925

$$P = \frac{\text{PLAN AREA}}{\text{LENGTH} \times \text{SPAN}} = 0{\cdot}45$$

$$\frac{\text{VOLUME (EXCLUDING STING FAIRING)}}{\text{LENGTH}^3} = 0{\cdot}00333$$

$$\frac{\text{WETTED AREA (INCLUDING STING FAIRING)}}{\text{REFERENCE AREA}} = 2{\cdot}10$$

FIG. 2. MODEL B.



PLAN AREA OF EXPOSED WING = 455 SQ INS

GROSS AREA OF WING = 616 SQ INS
(LEADING & TRAILING EDGES CONTINUED TO ℄)

ASPECT RATIO OF GROSS WING = 1·58

REFERENCE AREA = 534 SQ INS.

$\dfrac{\text{WETTED AREA OF EXPOSED WING}}{\text{REFERENCE AREA}}$ = 1·73

$\dfrac{\text{WETTED AREA OF BODY}}{\text{REFERENCE AREA}}$ = 1·34

FIG. 3 MODEL C.

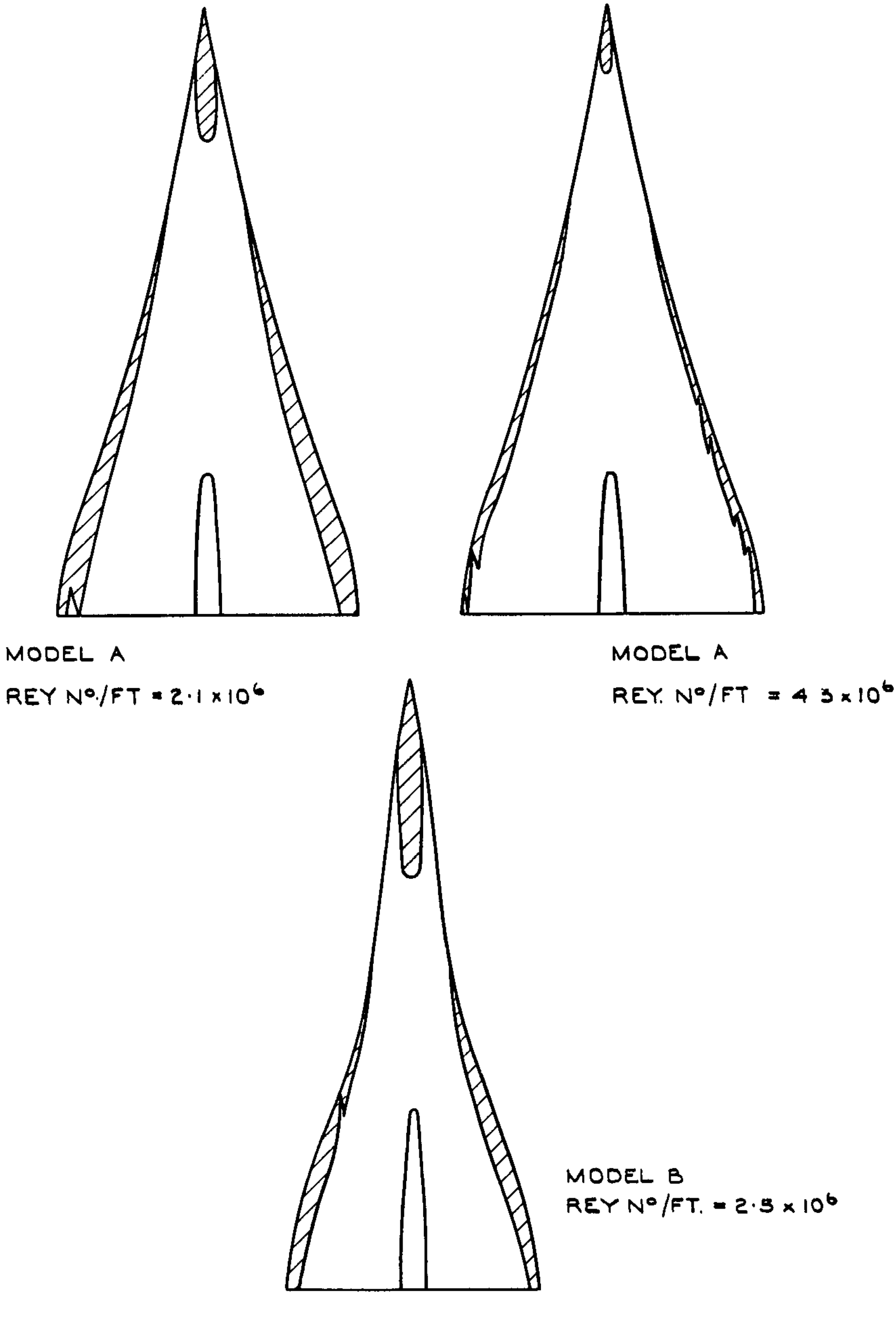


FIG. 4. SUBLIMATION PATTERNS - FREE TRANSITION.

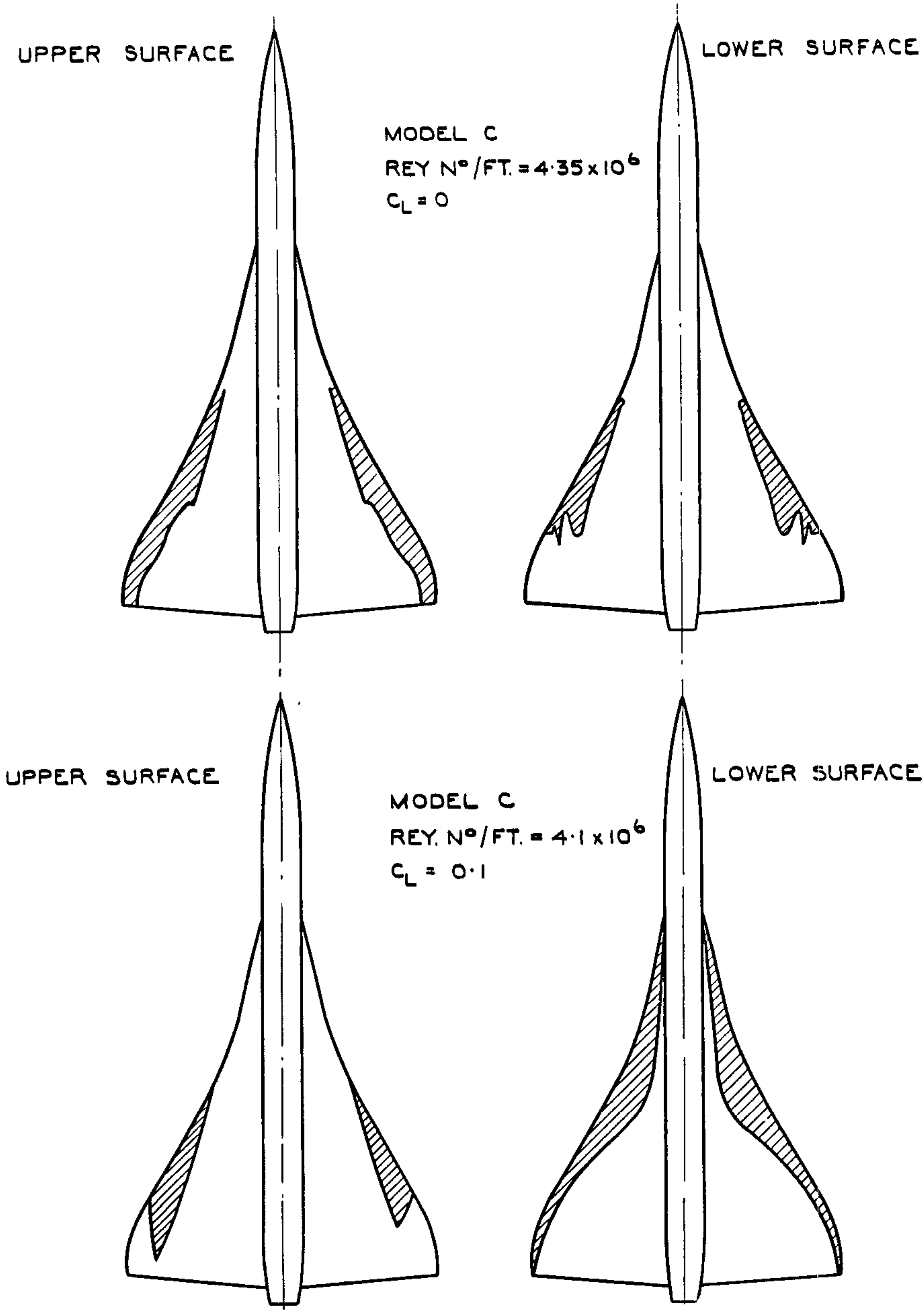


FIG. 4. (CONT'D.) SUBLIMATION PATTERNS – FREE TRANSITION.

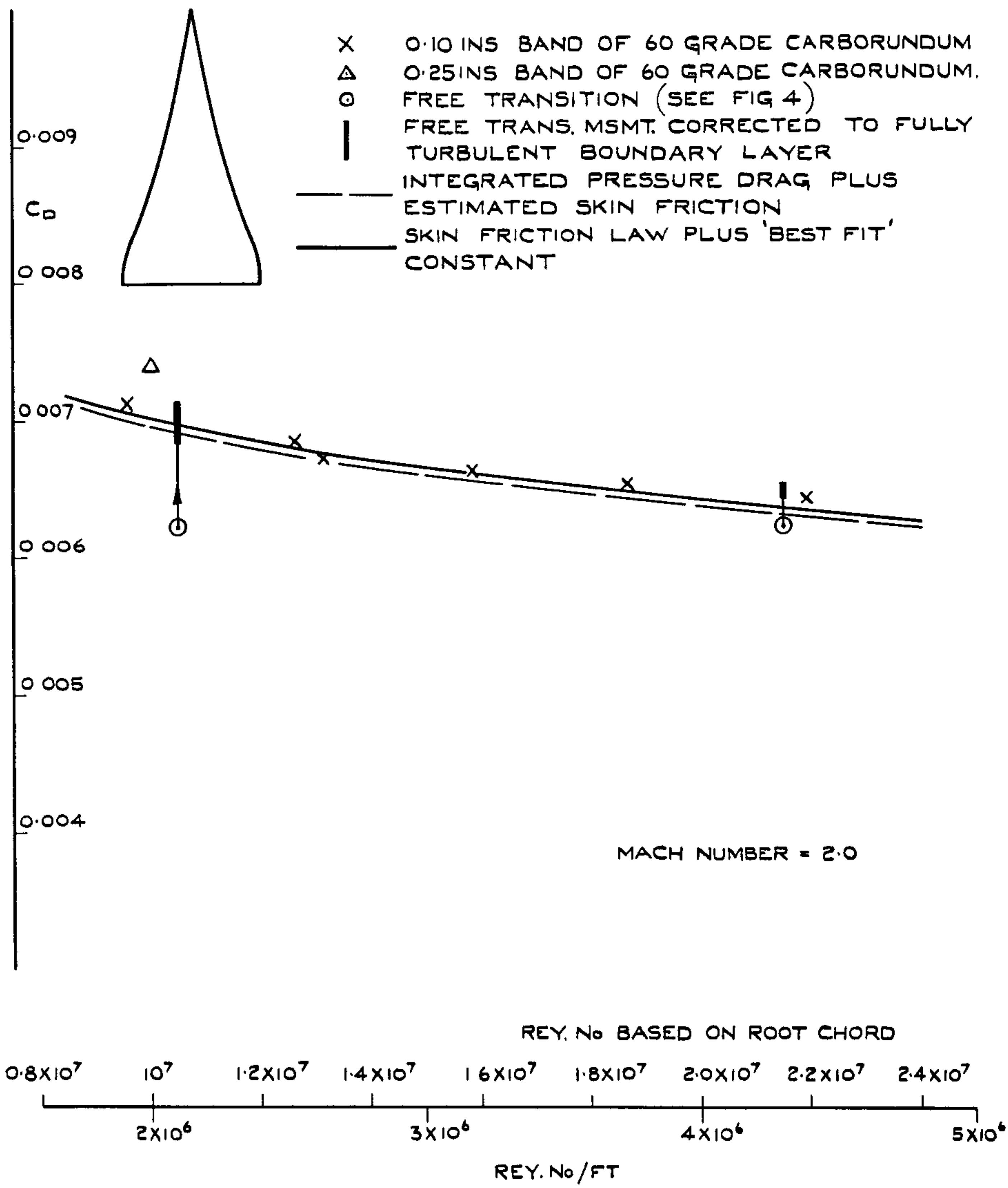


FIG. 5. MODEL A, DRAG VARIATION WITH REYNOLDS NUMBER.

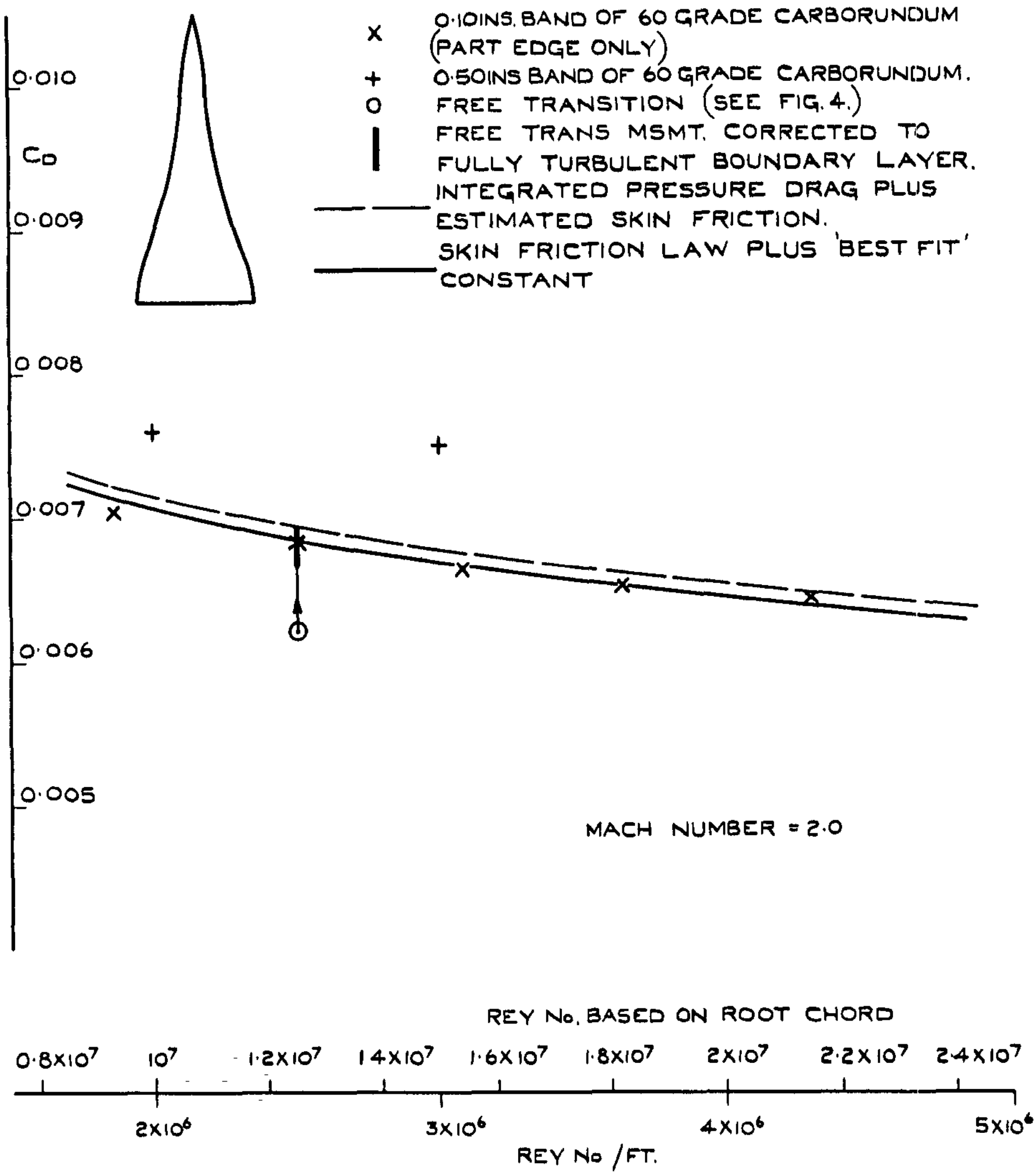


FIG. 6. MODEL B, DRAG VARIATION WITH REYNOLDS NUMBER.

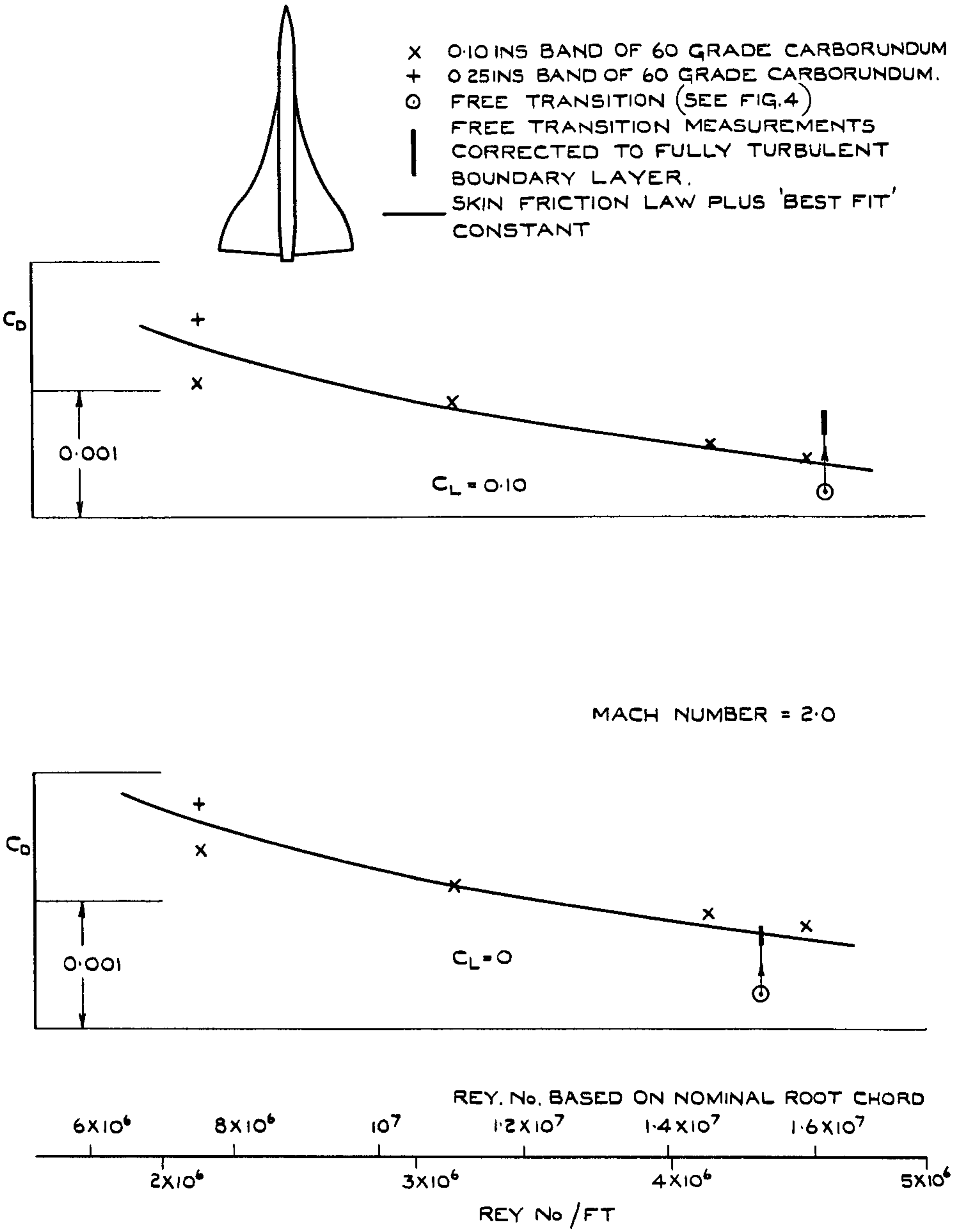


FIG 7 MODEL C, DRAG VARIATION WITH REYNOLDS NUMBER.

A.R.C. C.P. No.742

629.137.1:
533.6.011.5:
533.693.3:
533.6.071.011.5

USE OF A WIND TUNNEL TO DETERMINE THE PERFORMANCE OF SLENDER WINGS SUITABLE FOR A SUPERSONIC TRANSPORT AIRCRAFT. Evans, J. Y. G. March, 1963.

Drag coefficients obtained from surface pressure measurements are compared with those derived from balance measurements, both with free and with artificially fixed boundary layer transition, in order to judge whether such wind tunnel tests can be used to predict the performance of slender wings of the types being studied for supersonic transport aircraft.

While a satisfactory level of accuracy can be achieved in most cases at zero lift, a better understanding of the boundary layer behaviour over such wings at incidence is required before results on a lifting model can be extrapolated to full scale with full confidence.

A.R.C. C.P. No.742

629.137.1:
533.6.011.5:
533.693.3:
533.6.071.011.5

USE OF A WIND TUNNEL TO DETERMINE THE PERFORMANCE OF SLENDER WINGS SUITABLE FOR A SUPERSONIC TRANSPORT AIRCRAFT. Evans, J. Y. G. March, 1963.

Drag coefficients obtained from surface pressure measurements are compared with those derived from balance measurements, both with free and with artificially fixed boundary layer transition, in order to judge whether such wind tunnel tests can be used to predict the performance of slender wings of the types being studied for supersonic transport aircraft.

While a satisfactory level of accuracy can be achieved in most cases at zero lift, a better understanding of the boundary layer behaviour over such wings at incidence is required before results on a lifting model can be extrapolated to full scale with full confidence.

A.R.C. C.P. No.742

629.137.1:
533.6.011.5:
533.693.3:
533.6.071.011.5

USE OF A WIND TUNNEL TO DETERMINE THE PERFORMANCE OF SLENDER WINGS SUITABLE FOR A SUPERSONIC TRANSPORT AIRCRAFT. Evans, J. Y. G. March, 1963.

Drag coefficients obtained from surface pressure measurements are compared with those derived from balance measurements, both with free and with artificially fixed boundary layer transition, in order to judge whether such wind tunnel tests can be used to predict the performance of slender wings of the types being studied for supersonic transport aircraft.

While a satisfactory level of accuracy can be achieved in most cases at zero lift, a better understanding of the boundary layer behaviour over such wings at incidence is required before results on a lifting model can be extrapolated to full scale with full confidence.

C.P. No. 742

Published by
HER MAJESTY'S STATIONERY OFFICE
To be purchased from
York House, Kingsway, London w.c.2
423 Oxford Street, London w.1
13A Castle Street, Edinburgh 2
109 St. Mary Street, Cardiff
39 King Street, Manchester 2
50 Fairfax Street, Bristol 1
35 Smallbrook, Ringway, Birmingham 5
80 Chichester Street, Belfast 1
or through any bookseller

S.O. CODE No. 23-9015-42

C.P. No. 742