TR 70235

DECEMBER

1970

ROYAL AIRCRAFT ESTABLISHMENT

TECHNICAL REPORT 70235

WING-BODY SHAPES FROM KNOWN SUPERSONIC FLOW FIELDS

by

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FARNBOROUGH HANTS

UDC 533.695.12 : 533.6.04 : 533.6.011.5 : 533.6.011.72 :
533.6.048.2 : 533.6.013.12/13

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SUMMARY

A method of designing wing-body aircraft shapes with a known inviscid pressure distribution at a particular Mach number and incidence is proposed. An example is presented with a known pressure distribution at M = 3, and a calculated lift to drag ratio of 3.6. Further development of various features of the configuration is discussed.

Departmental Reference: Aero 3149

CONTENTS

1 INTRODUCTION

It has been demonstrated how, at high supersonic speeds[1,2,3] integrated all-wing lifting surfaces with known pressure distributions may be obtained from known flow fields. Here it is shown how wing-body shapes may be obtained from a 'matched' pair of such flow fields. The wing-body shapes have the same property as the integrated wings, in that, at a particular incidence and Mach number, the inviscid pressure distribution over the configuration is known exactly. An example is presented demonstrating the advantageous use of wing-body interference at M = 3. This example also provides exact pressures over a devised wing-body shape against which the pressure predictions of approximate theories may be checked.

2 THE DESIGN OF WING-BODY SHAPES

In the method used for designing integrated lifting configurations described in Refs.1 to 3, a single stream surface from a known flow field is used to define the lower surface of the wing and the upper surface is defined by a streamwise region and one or more expansion surfaces. The method of designing wing-body shapes uses the same basic principles, but pairs of stream surfaces from 'matched' flow fields are used to define the complete wing-body shape.

The matched flow fields are required to have the same shock wave and body shape upstream of a particular characteristic surface in the flow. In Fig.1a an example of a pair of matched axisymmetric flow fields external to two annular bodies (labelled A and B) is shown. Upstream of the characteristic surface indicated, the two body shapes are the same. Downstream of this characteristic surface body A enlarges to a maximum diameter at the base whilst body B contracts, so that its base diameter is small. The resulting pressure distributions along the bodies are shown in Fig.1b. It can be seen that the maximum difference in the body pressure occurs just downstream of the characteristic, where the body radii differ by only a small amount. The expansion of the flow near body A and the recompression near body B, results in the pressures again becoming equal at about x = 0.75, where the body radii are very different. Aft of this point the pressure on body B is greater than that on body A.

Consider any line in the shock wave which connects the nose and base planes, as for example the 'heavy' line shown in Fig.2. Then, this line, together with its reflection in a vertical plane through the axis of symmetry, divides the shock wave between the nose and the base planes into two. Following streamlines downstream from these lines in the flow fields of bodies A and B gives stream surfaces which are coincident where the flow fields are the same, but differ downstream of the characteristic surface shown by the broken lines in the figure. In order that the surfaces will diverge from one another in the manner shown in Fig.2, it is important that the lines in the shock wave 'spiral' slightly with respect to the axis of symmetry. That is, if (r, θ) are cylindrical coordinates based on the axis of symmetry, then $d\theta/dr$ is required to be non-zero along the chosen lines in the shock wave.

Each of the flow fields A and B is divided into two by its respective stream surfaces and body surface. By matching the upper flow region of body B to the lower flow region of body A and replacing the stream surfaces downstream of the characteristic with solid surfaces, a wing-body configuration results. The wing leading edge lies on the characteristic surface, and the body shape upstream of the wing is the same as that common to the two flow fields. Further the body shape above the wing resembles a segment of body B and that below the wing a segment of body A. The inviscid pressures over the configuration at the free stream Mach number can be obtained from the known pressures in the flow field.

Fig.3 shows general views and sections of a wing-body shape derived from the flow fields shown in Fig.1. The coordinates and pressure coefficients are given in Tables 1 and 2. The body in side view has a profile which is the same as those of bodies A and B shown in Fig.1. The leading edge of the wing lies on the characteristic surface, and is thus 'sonic'. The configuration has an engine intake in the nose, as is shown in the general view and the section at $x = 0$. The sections at $x = 0.2$, 0.4 etc. depict the development of the shape downstream to a bluff base at $x = 1$. On the left of Fig.3b are shown the pressure distributions of the respective sections. The area contained within these curves represents the lift from the section. Thus the section at $x = 0.4$ contributes a relatively large amount of lift, whilst at the base the lift from the wing outboard is largely offset by the negative lift inboard. The region of negative lift near the base can be removed by modifying the planform, perhaps by removing the shaded region shown in Fig.3, to give a more usual swept wing planform. The trailing edge can be modified without affecting the inviscid pressures upstream provided that it remains 'supersonic' everywhere. Further, the rear of the body can be extended downstream and a tailplane added without affecting the known pressures upstream, although the pressures over the added portion of the body will not be determinable by the present method.

There are other reasons why the unmodified shape is not satisfactory in the shaded region. Firstly, the adverse pressure gradient on the upper surface in this region may cause the boundary layer to separate, making the inviscid pressures unrepresentative of those which would occur in a viscous flow. This adverse pressure gradient can be reduced by removing volume from the upper surface near the base. Secondly, it would be advantageous to reduce the base area to that required by the engine nozzle, so reducing the possibility of base drag. This requirement is compatible with the previous one, and together they provide an indication as to the design of desirable rear body shapes.

The lift and pressure drag coefficients given for M = 3 are based on the shape as shown, with the total plan area as reference area. The base drag is not included in pressure drag coefficient C_{DP}, neither is any drag connected with the intake; even so the L/D_P of 3.6 is encouraging. The volume coefficient (τ = Volume/(Plan area)$^{3/2}$) of the unmodified shape is 0.08.

3 FURTHER DEVELOPMENTS

The shape shown in Fig.3 is based on the flow about an axisymmetric annular body at M = 3. The method can be used to obtain shapes at other Mach numbers, and to extend the range of shapes at a particular Mach number, by calculating the appropriate flow fields. The intake in the nose of the configuration in Fig.3 can be removed by using flow fields about pointed bodies. The engine housings then likely to be required in the wing, can be added whilst preserving the known pressure distribution, by following streamlines in the flow downstream from the intake in a manner similar to that described in Ref.2.

The method is not confined to axisymmetric flow fields. For example matched pairs of flows over an inclined body would give a shape with both the body and the wing at incidence. The wing leading edge can be made thicker by using a flow field with a second shock wave as indicated in Fig.4.

Even with these developments the method will not have the required freedom to reproduce a specified shape. However it should be possible to produce wing-body shapes which are typical of realistic aircraft, and may be used both for comparison with other shapes and as known non-trivial configurations against which approximate theories may be tested.

4 CONCLUSIONS

A method of designing wing-body aircraft shapes using stream surfaces from known flow fields is presented. The potential of the method is demonstrated by the design of a conventional type of wing-body shape. Further developments are indicated for designing

(i) more complex shapes (e.g. shapes with engine nacelles in the wing)

(ii) a range of shapes (e.g. to investigate the variation of lift and pressure drag with body shape), and

(iii) shapes to test approximate theories (e.g. wave drag determination).

Table 1

BODY RADII AND PRESSURE COEFFICIENT

Coordinates of Axis $y = 0$, $z = -0.0237$

x	Body A		Body B	
	r	C_P	r	C_P
0	0.0237	0.231	0.0237	0.231
0.1	0.0475	0.139	0.0475	0.391
0.2	0.0672	0.118	0.0672	0.118
0.3	0.0908	0.105	0.0875	−0.052
0.4	0.1070	0.059	0.0765	−0.082
0.5	0.1180	0.029	0.0622	−0.072
0.6	0.128	0.009	0.0495	−0.056
0.7	0.134	−0.006	0.0362	−0.033
0.8	0.137	−0.016	0.0230	−0.004
0.9	0.138	−0.024	0.0137	+0.037
1	0.138	−0.029	0.0091	+0.074

B.l. calc $Re = 10^7$

Table 2

WING COORDINATES AND PRESSURE COEFFICIENTS

	Lower surface			Upper surface		
	y	z	C_P	y	z	C_P
x = 0.4	0.0642	0.0619	0.059	0.0459	0.0375	-0.082
	0.08	0.0680	0.0688	0.08	0.0547	-0.0599
	0.12	0.0734	0.0997	0.12	0.0730	0.0814
	0.13	0.0770	0.0980	0.13	0.0770	0.0980
x = 0.6	0.0768	0.0751	0.0090	0.0297	0.0159	-0.0560
				0.04	0.0226	-0.0574
	0.08	0.0791	0.0131	0.08	0.0501	-0.0637
	0.12	0.0819	0.0289	0.12	0.0735	-0.0530
	0.16	0.0941	0.0490	0.16	0.0914	-0.0125
	0.20	0.1048	0.0701	0.20	0.1043	0.0404
	0.24	0.1137	0.0850	0.24	0.1137	0.0850
x = 0.8	0.0822	0.0859	-0.016	0.0138	-0.0053	-0.0040
				0.04	0.0181	-0.0204
				0.08	0.0466	-0.0399
	0.12	0.0873	-0.0047	0.12	0.0700	-0.0504
	0.16	0.0990	0.0095	0.16	0.0893	-0.0476
	0.20	0.1100	0.0240	0.20	0.1049	-0.0287
	0.24	0.1193	0.0386	0.24	0.1170	-0.0016
	0.28	0.1273	0.0533	0.28	0.1265	0.0278
	0.32	0.1342	0.0705	0.32	0.1340	0.0624
	0.34	0.1370	0.0840	0.34	0.1370	0.0840
x = 1.0	0.0828	0.0867	-0.029	0.0055	-0.0164	0.0740
				0.04	0.0166	0.0197
				0.08	0.0446	-0.0087
	0.12	0.0891	-0.0222	0.12	0.0672	-0.0284
	0.16	0.1011	-0.0120	0.16	0.0862	-0.0409
	0.20	0.1126	-0.0013	0.20	0.1024	-0.0428
	0.24	0.1225	0.0095	0.24	0.1160	-0.0331
	0.28	0.1309	0.0206	0.28	0.1271	-0.0166
	0.32	0.1385	0.0321	0.32	0.1364	0.0027
	0.36	0.1450	0.0441	0.36	0.1443	0.0235
	0.40	0.1513	0.0566	0.40	0.1511	0.0458
	0.439	0.1571	0.0693	0.439	0.1571	0.0693

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No.	Author(s)	Title, etc.
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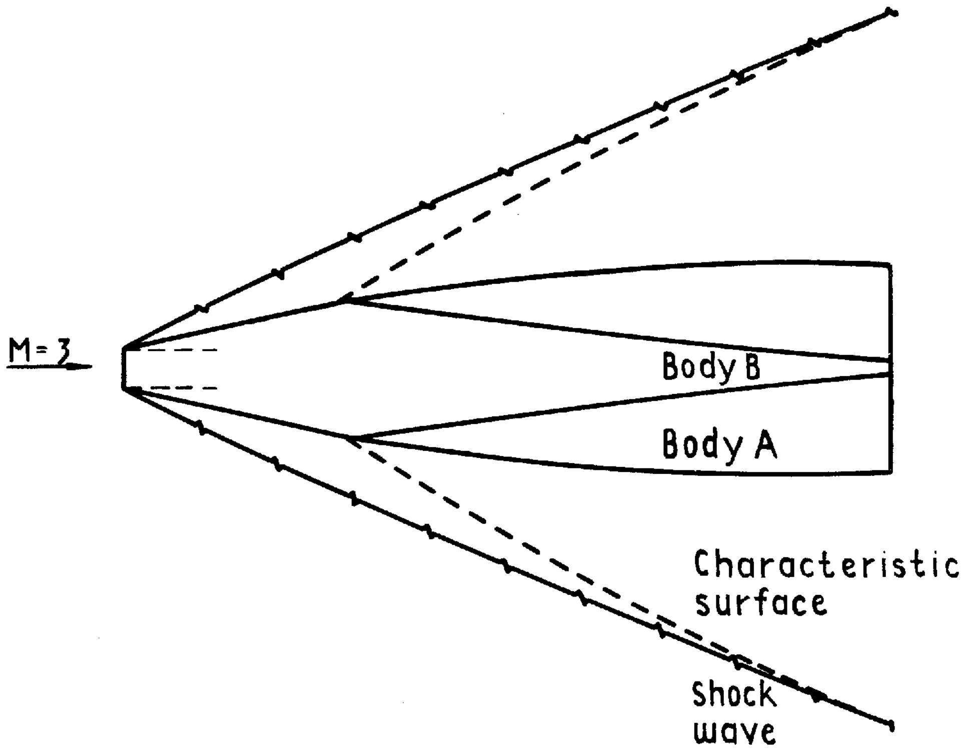
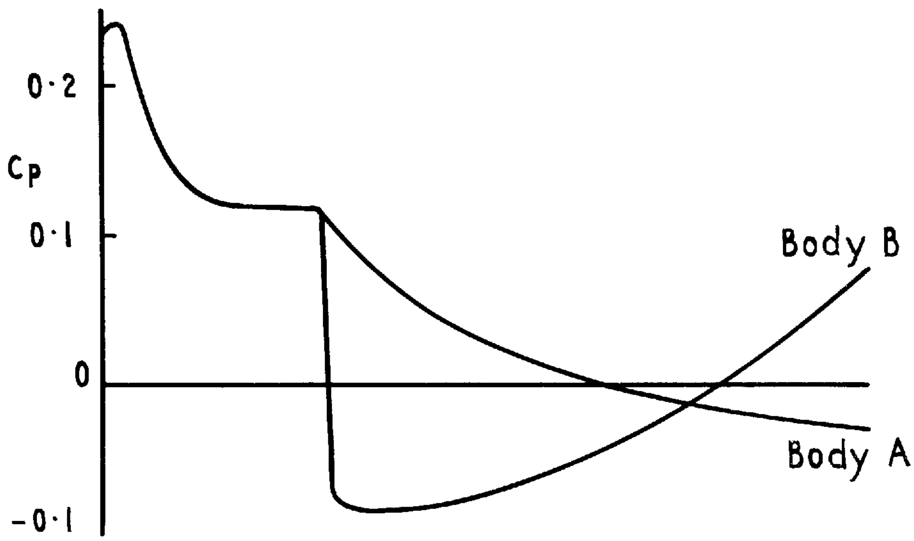
2	J.G. Jones K.C. Moore J. Pike P.L. Roe	A method for designing lifting configurations for high supersonic speeds using axisymmetric flow fields. Ingenieur-Archiv, 37 Band, 1 Heft, (1968) p.56-72
3	J. Pike	On conical waveriders. RAE Technical Report 70090 (1970)



a Shock wave and body shape



b Pressure distribution along body

Fig 1 a & b A matched pair of flow fields

Fig. 2

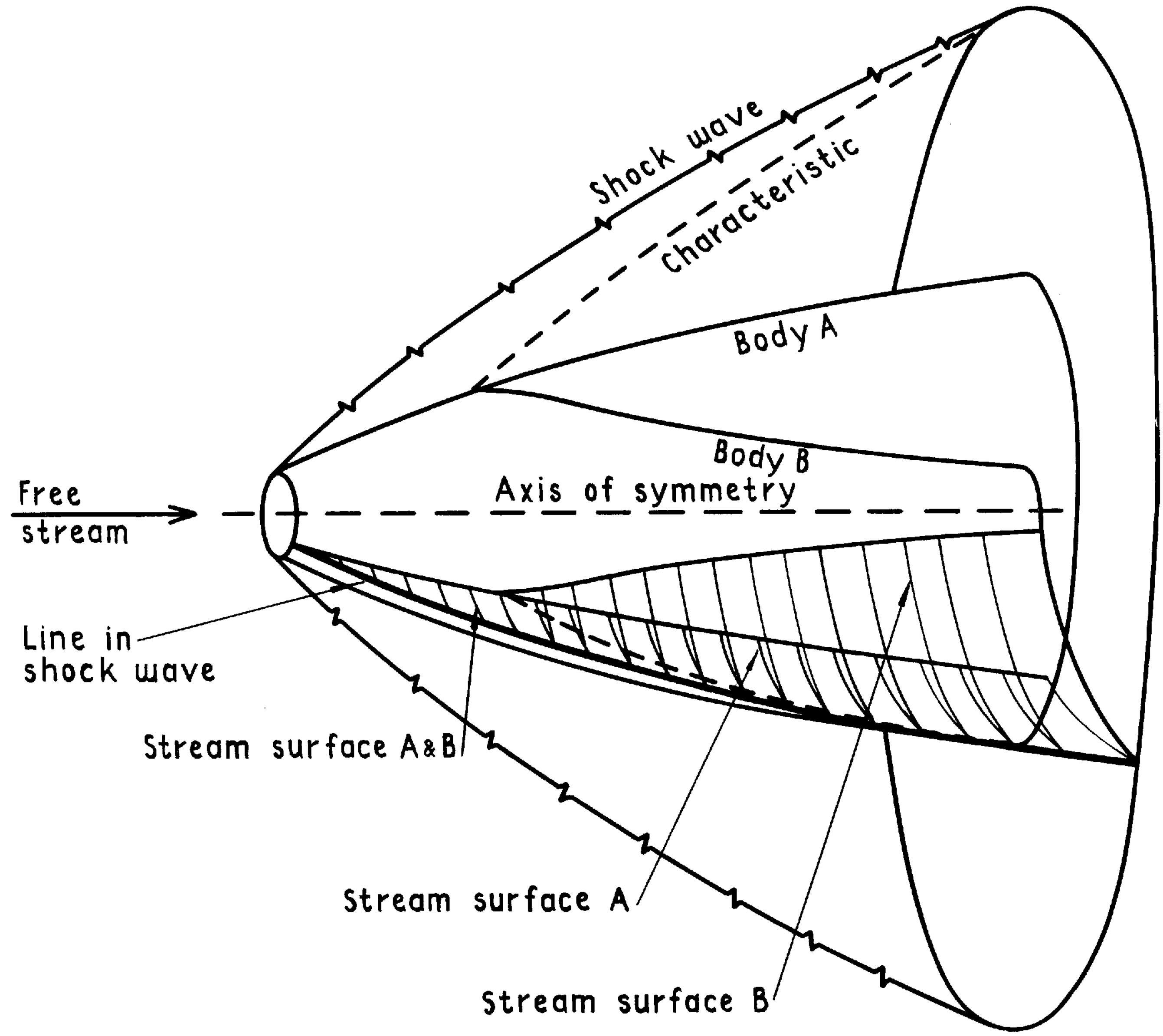
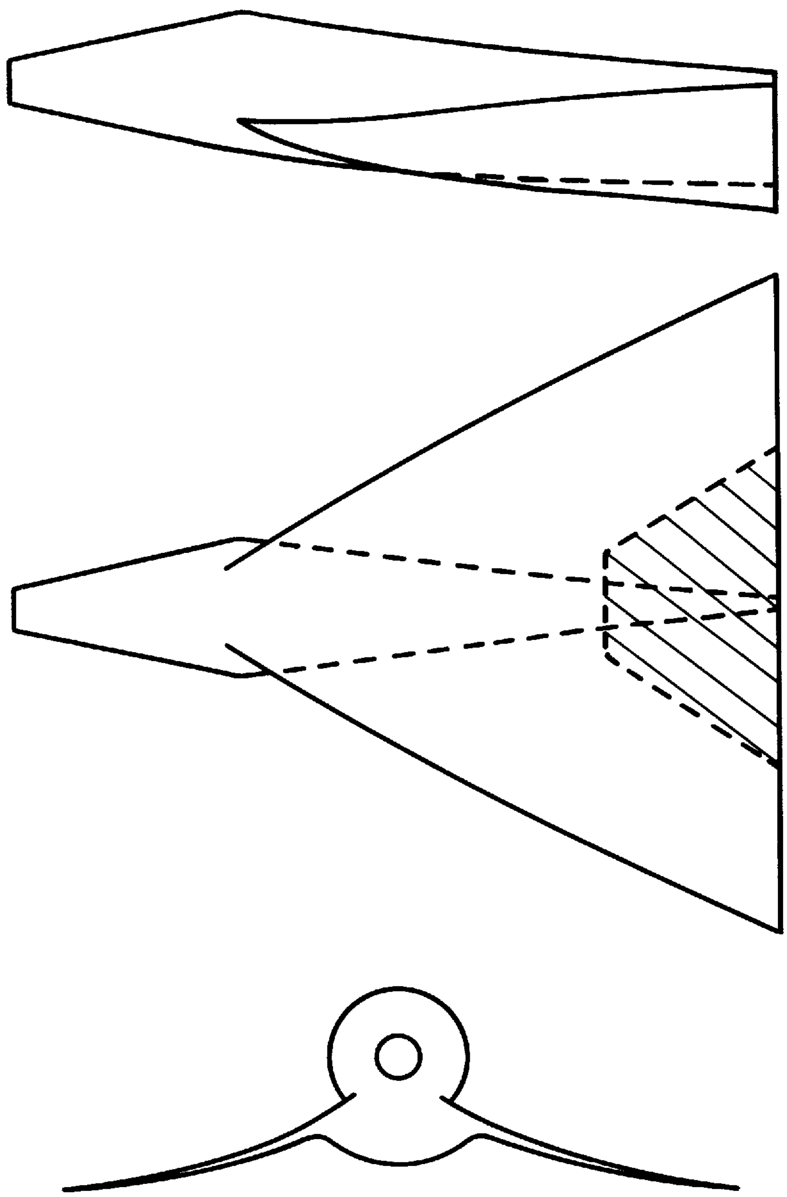


Fig. 2 Matched stream surfaces in the flow fields

Fig. 3a

At $M=3$ $C_L = 0{\cdot}0406$ $C_{D_p} = 0{\cdot}0113$ $L/D_p = 3{\cdot}6$

$\tau = 0{\cdot}08$



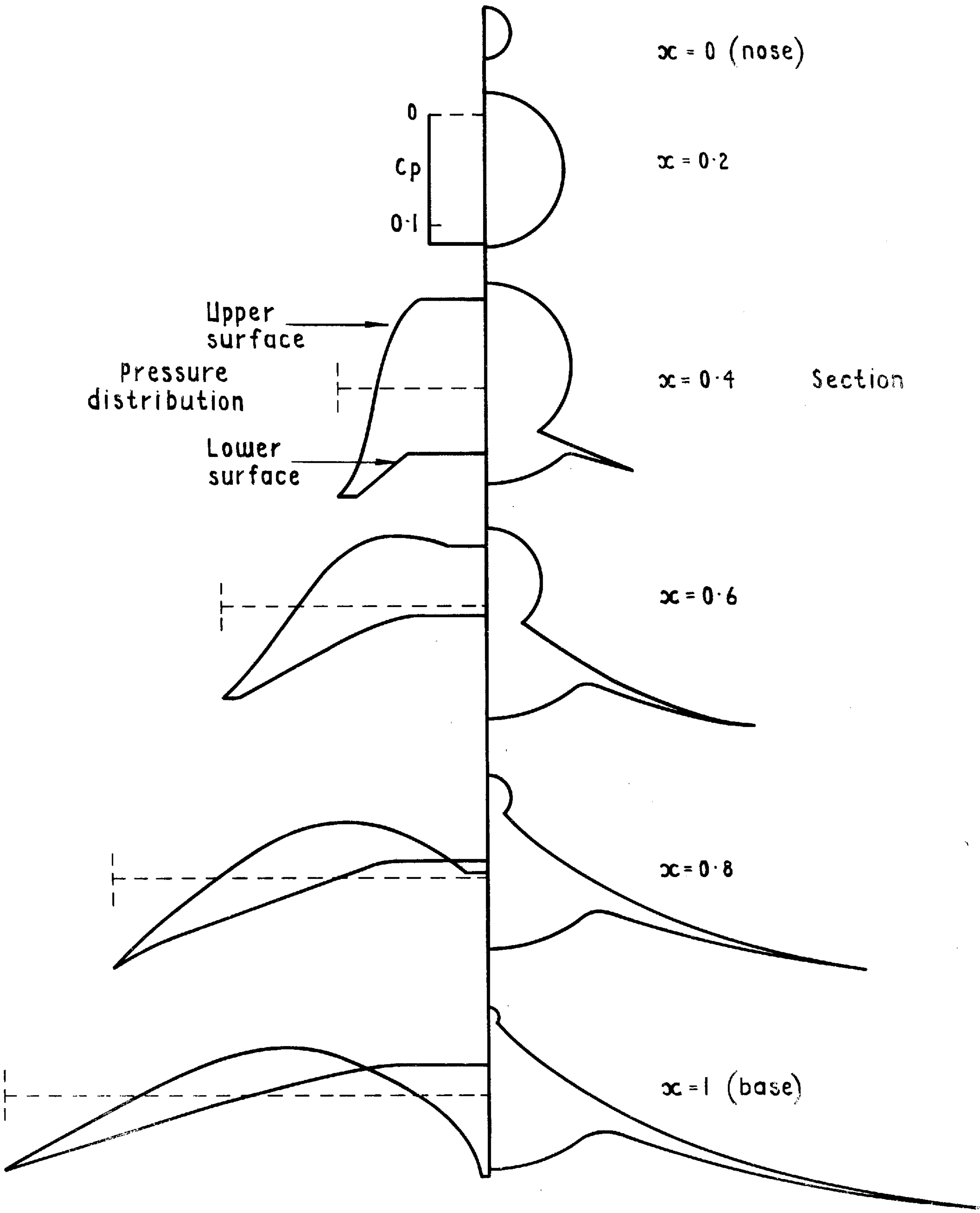
Side view

Plan view

Front view

a General views

Fig. 3 A wing-body aircraft shape



b Spanwise pressure distributions and sections

Fig. 3 contd A wing-body aircraft shape

Fig. 4

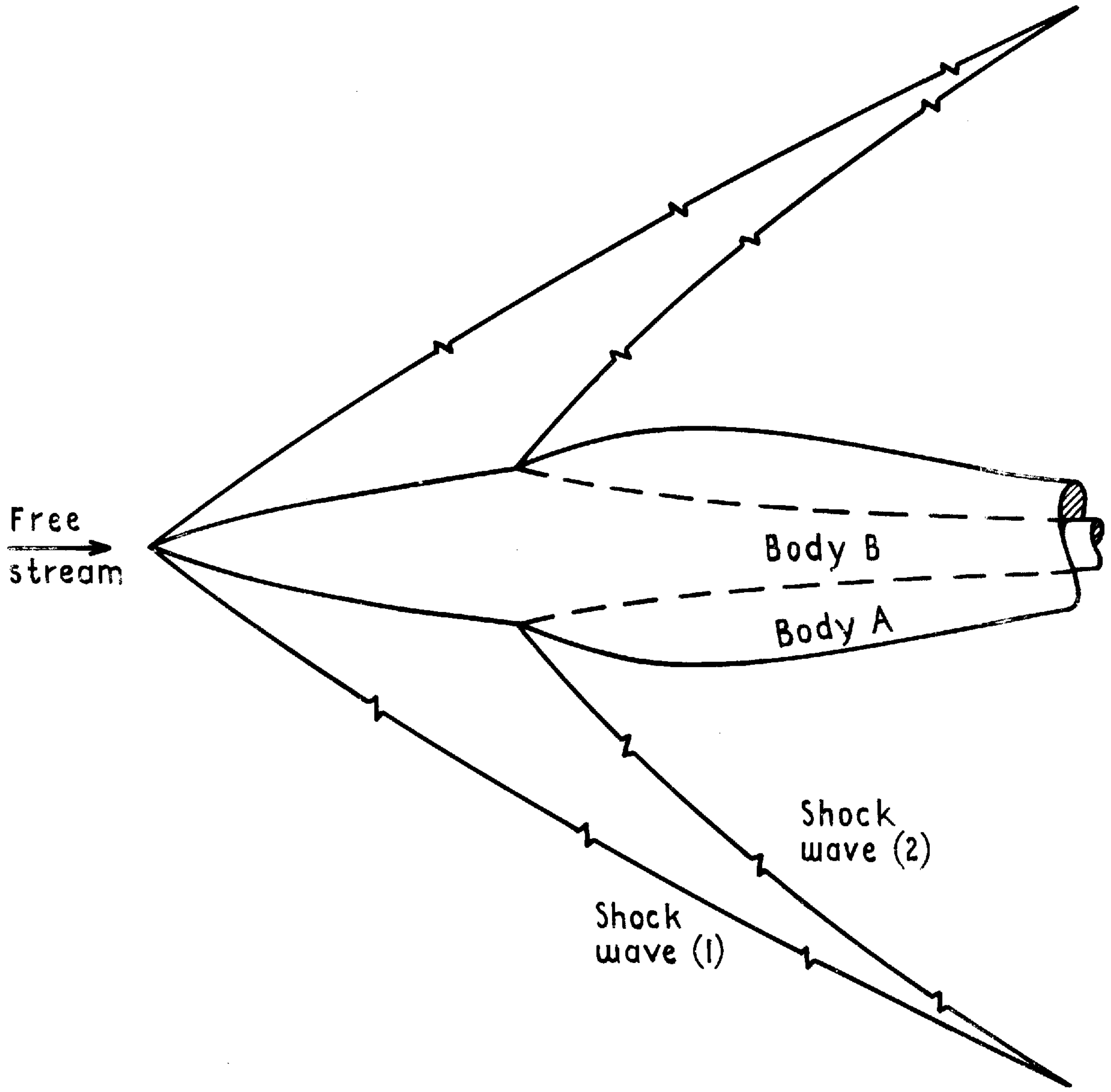


Fig 4 A matched pair of flow fields with two shock waves for body A