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The pressure distribution at zero incidence over selected families of blunt axisymmetric forebodies

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THE PRESSURE DISTRIBUTION AT ZERO INCIDENCE OVER SELECTED FAMILIES OF BLUNT AXISYMMETRIC FOREBODIES

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THE PRESSURE DISTRIBUTION AT ZERO INCIDENCE OVER SELECTED FAMILIES OF BLUNT AXISYMMETRIC FOREBODIES

1. NOTATION AND UNITS

		<i>SI</i>	<i>British</i>
C_p	pressure coefficient (see Equation (4.1))		
D	maximum diameter of forebody	m	ft
K_n	factor for effect of nose radius on pressure distribution (see Equation (4.2))		
K_s	factor for effect of shoulder radius on pressure distribution (see Equation (4.3))		
l_f	forebody length	m	ft
M	local surface Mach number		
M_1	free-stream Mach number		
p	local static pressure on body	N/m ²	lbf/ft ²
p_H	stagnation pressure	N/m ²	lbf/ft ²
p_{H0}	stagnation pressure at body nose ($x = 0$)	N/m ²	lbf/ft ²
p_{H1}	free-stream stagnation pressure, corresponding to M_1	N/m ²	lbf/ft ²
p_{H2}	stagnation pressure behind a normal shock wave with Mach number M_1 upstream	N/m ²	lbf/ft ²
p_1	free-stream static pressure, corresponding to M_1	N/m ²	lbf/ft ²
R	maximum radius of forebody	m	ft
R_D	Reynolds number based on forebody maximum diameter		
r_n	forebody nose radius	m	ft
r_s	forebody shoulder radius	m	ft
T	local surface temperature (see Section 6)	K	K
T_1	free-stream temperature	K	K
x	axial co-ordinate with origin at nose	m	ft
y	radial co-ordinate with origin at nose	m	ft
δ	cone semi-apex angle	deg	deg
ρ_n	nose blunting ratio, r_n/R		
ρ_s	shoulder blunting ratio, r_s/R		

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Subscripts

ρ_n	pertaining to any value of ρ_n
$\rho_n =$	pertaining to specified value of ρ_n
ρ_s	pertaining to any value of ρ_s
$\rho_s =$	pertaining to specified value of ρ_s

2. INTRODUCTION

This Item gives pressure distribution data for the flow of air* over four families of blunt forebody at zero incidence; ellipsoidal, spherically-curved nose with radiused shoulder, spherically-blunted cone and spherically-blunted tangent ogive. The families chosen (see Section 3) cover most of those likely to be used in missile and re-entry body applications. Included among the families are those considered in Item No. 68021 (Reference 21) and Item No. 80021 (Reference 24) and as such the present Item acts as a companion to those two.

The ranges of geometry and Mach number covered for each family are as wide as possible dependent on the availability of data. The pressure distribution data are presented (Section 4) in a direct form as a ratio of the local static pressure to the stagnation pressure at the nose of the forebody, while methods of estimating the stagnation pressure itself are given in Section 5. The abscissal parameter was selected as that being most appropriate to the family under consideration. In this respect data for forebody fineness ratios (l_f/D) of 0.5 or less are presented against a dimensionless radial co-ordinate, y/R , which satisfactorily accommodates the case of the flat face, while for forebody fineness ratios of 0.5 or more the data are presented against an appropriate dimensionless axial co-ordinate.

Section 6 provides a means of estimating, for inviscid flow, the surface Mach number distribution and the temperature distribution with no heat transfer from the corresponding pressure distribution.

Section 7 considers the accuracy and applicability of the data, which will typically be used in loading applications.

Section 8 lists the sources of data, both theoretical and experimental, used to derive the data given while Table 10.1 provides a breakdown of the sources of data used for each pressure distribution Figure.

Section 9 gives a worked example concerning a typical application of the Item.

* Any formulae or theoretical methods used to generate data in this Item requiring a value of the ratio of specific heat capacity at constant pressure to the specific heat capacity at constant volume used 1.4 for air.

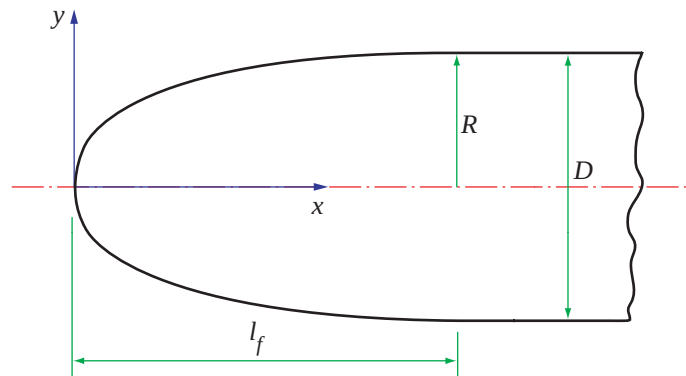
Appendix A summarises the geometrical relationships defining the forebody shapes used in this Item. Also included are relationships for surface length, which is sometimes used in experimental work, for example in defining the pressure hole locations on test bodies.

The data for the hemispherical head are used in Appendix B to describe the flow development over a typical blunt body as the free-stream Mach number traverses the full range from low subsonic to high supersonic.

3. FAMILIES OF FOREBODY SHAPES CONSIDERED

The families of forebody shapes and the ranges of parameters to be considered are as follows.

(i) Ellipsoidal



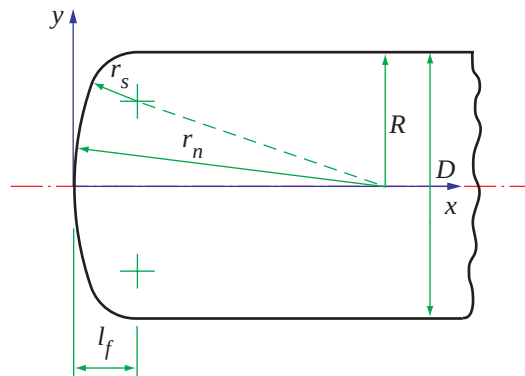
Sketch 3.1

$l_f/D = 0, 0.25, 0.333, 0.5, 1.0, 1.5, 2$ and 4 .

(Note: $l_f/D = 0$ is a flat face, $l_f/D = 0.5$ is a hemispherical head.)

$y/R = 0$ to 1 , $x/l_f = 0$ to 2 .

(ii) Spherically-curved nose with radiused shoulder



Sketch 3.2

Sharp shoulder: $\rho_n (= r_n/R) = 1 \text{ to } \infty$ for $\rho_s (= r_s/R) = 0$
 $y/R = 0 \text{ to } 1.$

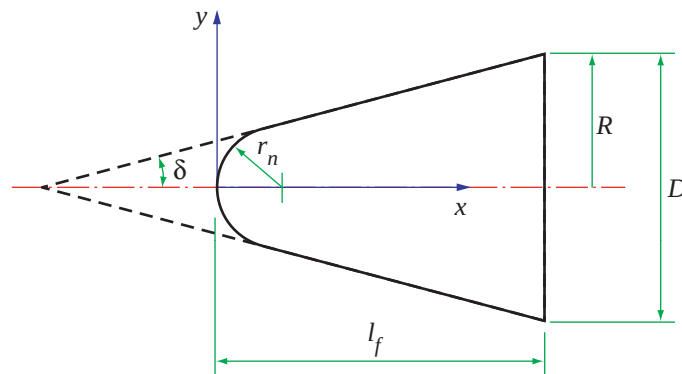
(Note: $\rho_n = \infty$ is a flat face, $\rho_n = 1$ is a hemispherical head.)

Radiused shoulder: $\rho_s = 0 \text{ to } 1$ for $\rho_n = 2 \text{ and } \infty$

$y/R = 0 \text{ to } 1.$

(Note: $\rho_s = 0$, $\rho_n = \infty$ is a flat face, $\rho_s = 1$ is a hemispherical head.)

(iii) Spherically-blunted cone



Sketch 3.3

$\delta \leq 40^\circ$: $\delta = 0^\circ, 5^\circ, 10^\circ, 15^\circ, 20^\circ, 30^\circ, \text{ and } 40^\circ$
 $x/r_n = 0.1 \text{ to } 100.$

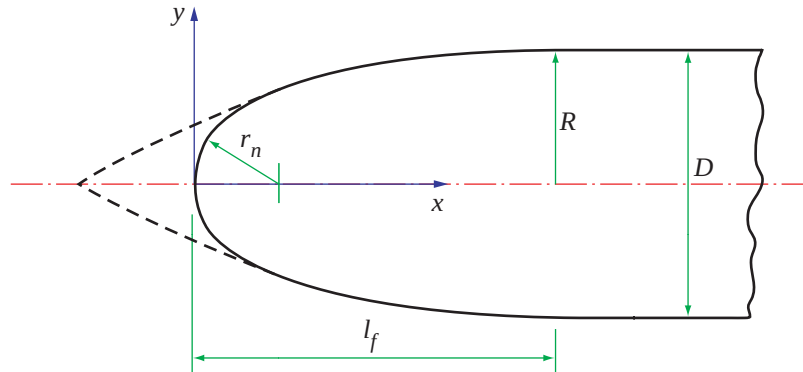
$\delta = 45^\circ$ is considered for one Mach number only (see Section 4.3).

$\delta > 40^\circ$: $\rho_n < 0.5$, $\delta = 60^\circ, 70^\circ, 80^\circ \text{ and } 90^\circ$
 $y/R = 0 \text{ to } 1.$

(Note: $\delta = 90^\circ$ is a flat face.)

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(iv) Spherically-blunted tangent ogive



Sketch 3.4

$$l_f/D = 1, 1.5 \text{ and } 2$$

$$\rho_n = 0, 0.6 \text{ and } 0.8$$

$$x/l_f = 0 \text{ to } 2.$$

Some applications of the data presented herein may require a knowledge of certain geometrical relationships for the body under consideration. Such relationships are given in Appendix A for each of the families considered in this Item.

4. PRESENTATION OF DATA

The pressure distribution data are presented in the Figures in terms of a ratio of the local static pressure to the stagnation pressure at the forebody nose, p/p_{H0} . This parameter was chosen as being more closely allied to the actual pressure distribution than the more usual pressure coefficient, C_p . In addition, the use of p/p_{H0} provides a clearer separation of the curves appropriate to various free-stream Mach numbers, M_1 , than does the use of C_p . Should C_p be required, however, the following equation gives the relationship with p/p_{H0} ,

$$C_p = \frac{[(p/p_{H0})/(p_1/p_{H0})] - 1}{0.7M_1^2}. \quad (4.1)$$

Section 5 gives methods of estimating p_1/p_{H0} in order to determine p from p/p_{H0} or to convert given p/p_{H0} data to C_p data via Equation (4.1) if required.

The pressure distribution data presented in the Figures were largely derived either from theoretical methods involving inviscid finite-difference techniques or the method of characteristics, or from experimental data. In view of the sometimes *ad hoc* combination of theoretical and experimental data used to give a complete coverage of data for some of the forebodies it was felt useful to provide a table giving a detailed breakdown of the sources of data for each pressure distribution Figure. This is given in Table 10.1.

Salient points concerning the Figures presented for each of the forebody families considered are given in Sections 4.1 to 4.4 following.

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4.1 Ellipsoidal

Figures 1 to 7 present the data for the ellipsoidal family of forebodies. Each Figure gives data for forebodies of particular fineness ratio with curves applicable to as wide a range of free-stream Mach number as possible.

Figure 1 gives data for the flat face ($l_f/D = 0$) and forebodies of fineness ratio 0.25 and 0.333 for supersonic speeds.

The data for $l_f/D = 0$ and 0.25 were derived from experiment, which in the latter case covered a very restricted range of Mach number. The data for $l_f/D = 0.333$ were obtained from theoretical solutions using the method of characteristics (Derivation 9) and these, together with the data for the flat face and hemispherical head (see later), were used to derive the interpolated curves (shown broken) for $l_f/D = 0.25$.

Figures 2 and 3 provide data for the hemispherical head, essentially covering the full Mach number range. The data were obtained (Derivation 19) from finite-difference methods for Mach numbers up to 1 (Derivation 17) and for $M_1 = 2$ (Derivation 18) and the method of characteristics (Derivation 10) for Mach numbers of 3 and above. The curve for a Mach number of 1.5 is an interpolation based on a cross-plot against Mach number, using the experimental data of Derivations 4 and 5 as a guide. The presentation in Figure 3 clearly demonstrates the presence of shock waves on the body at transonic Mach numbers (see Appendix B for a detailed description of the flow development).

Figures 4 to 6 present similar data for ellipsoidal forebodies of fineness ratio 1, 1.5 and 2 respectively. The data are based on theoretical solutions (Derivation 19) for Mach numbers up to 1 and experimental data (Derivations 4 and 5) for Mach numbers between 1 and 2. Figure 4 also provides data for Mach numbers from 3 to ∞ based on theoretical solutions using the method of characteristics (Derivation 9). Finally, Figure 7 gives data for a forebody of fineness ratio 4 for Mach numbers in the range 1 to 2, based solely on experimental data from Derivations 4 and 5.

It can be seen from Figures 3 to 6 that for Mach numbers other than transonic the surface pressure has nearly reverted to the free-stream value by two forebody lengths downstream from the nose ($x = 0$).

The Figures relating to the ellipsoidal family clearly illustrate a trend that is characteristic of blunt bodies in general, namely, that the effect of free-stream Mach number variation becomes less pronounced supersonically and more pronounced subsonically as the bluntness increases (*i.e.* as l_f/D decreases).

4.2 Spherically-curved Nose with Radiused Shoulder

Figures 8a to 10d present data relevant to the case of families of forebodies with a spherically-curved nose and a radiused shoulder. These Figures were derived from the comprehensive experimental data given in Derivations 11 and 14 over the Mach number range 2.5 to 4.6. The range of forebody families considered in these Derivations had a common member at one extreme, the hemispherical head. Also, for the case of the forebody family with a sharp shoulder a common member at the other extreme was the flat-faced cyclinder. Now for these two extreme members the effect of Mach number over the full supersonic range is known (Figures 1a and 2) and it was found possible to make use of these data to devise an empirical means of applying the results of Derivations 11 and 14 to a much wider range of Mach number.

At a given radial station, y/R , the difference between the pressure for a given family member and that for the hemispherical head, when expressed as a fraction of the difference between the pressures on the extreme members of the family, was found to be substantially independent of Mach number (but see Section 7). This process was applied in order to establish the effect of nose blunting ratio (ρ_n) on the forebody family with sharp shoulders, *i.e.* $\rho_s = 0$ (Figures 8a to 8d) and the effect of shoulder blunting ratio on bodies with given nose blunting ratio, $\rho_n = 2$ and ∞ (Figures 9a to 9d and 10a to 10d respectively).

Figure 8a shows the factor K_n , found to be independent of y/R and assumed independent of Mach number, derived by this process, where

$$K_n = \frac{(p/p_{H0})_{\rho_n} - (p/p_{H0})_{\rho_n = 1}}{(p/p_{H0})_{\rho_n = \infty} - (p/p_{H0})_{\rho_n = 1}}. \quad (4.2)$$

Thus, with a knowledge of $(p/p_{H0})_{\rho_n = 1}$ and $(p/p_{H0})_{\rho_n = \infty}$ from Figures 2 and 1a respectively, the pressure distribution for a forebody with any intermediate value of ρ_n can be estimated using Figure 8a. This has been done for the range of ρ_n from 1 to ∞ for $M_1 = 2.5, 5$ and ∞ and the results are given in Figures 8b, 8c and 8d respectively.

Similarly, Figures 9a and 10a present the factor K_s , assumed independent of Mach number, for forebody families with $\rho_n = 2$ and ∞ respectively, where

$$K_s = \frac{(p/p_{H0})_{\rho_s} - (p/p_{H0})_{\rho_s = 1}}{(p/p_{H0})_{\rho_s = 0} - (p/p_{H0})_{\rho_s = 1}}. \quad (4.3)$$

Thus, with a knowledge of $(p/p_{H0})_{\rho_s = 1}$ (i.e. the hemispherical head) from Figure 2 and $(p/p_{H0})_{\rho_s = 0}$ (i.e. $(p/p_{H0})_{\rho_n}$ derived by means of Figure 8a for the appropriate value of ρ_n), the pressure distribution for a forebody with any value of ρ_s can be estimated using Figure 9a for $\rho_n = 2$ or Figure 10a for $\rho_n = \infty$. This has been done for the complete shoulder radius range of ρ_s from 0 to 1 for $M_1 = 2.5, 5$ and ∞ and the results are shown in Figures 9b, 9c and 9d and Figures 10b, 10c and 10d for $\rho_n = 2$ and ∞ respectively.

4.3 Spherically-blunted Cone

The data for spherically-blunted cones are given in Figures 11a to 11e and 12. Figures 11a to 11e for cone semi-apex angles less than 40° for Mach numbers from 3 to ∞ were obtained from data calculated by the method of characteristics (Derivation 10) and a finite-difference method (Derivation 15). The broken lines on the Figures indicate regions of extrapolation and interpolation. Figure 11a for $M_1 = 3$ also contains data for $\delta = 45^\circ$ obtained from experiment (Derivation 16). In Figures 11a to 11e p/p_{H0} is plotted against distance downstream from the nose made dimensionless with respect to the radius of the spherical blunting.

From these Figures it can be seen that for a given cone semi-apex angle the pressure distribution is divided into three distinct regions. The first is that covered by the nose blunting, the second is a region starting at the point of tangency of the nose blunting with the conical portion of the forebody in which the effects of the blunting are gradually reduced with increasing distance downstream, and the third is a region of constant pressure the same value as that which would have been obtained on the same cone with a sharp nose.

For cones with large semi-apex angles, in which even without nose blunting the bow shock would be detached and the surface flow wholly subsonic, it is more appropriate to present the pressure distribution data in terms of the radial coordinate y/R . Figure 12 presents data for such cones ($\delta \geq 60^\circ$) obtained from experimental data given in Derivations 12 and 13* for $\rho_n = 0.25$ and Derivation 16 for $\rho_n = 0.39$, and $\delta = 70^\circ$ only. The curves can be assumed to apply to sharp cones, or blunted cones with $\rho_n < 0.5$, say. The experimental data show a small systematic Mach number dependency, particularly for $\delta = 60^\circ$, the effect decreasing as δ increases to the flat face case ($\delta = 90^\circ$). In view of this the curves of Figure 12 are applicable to limited ranges of Mach number, see Section 7 and Table 10.1.

* There are of course other data supporting the curve for $\delta = 90^\circ$ (the flat face) see Table 10.1 with reference to Figure 12.

4.4 Spherically-blunted Tangent Ogive

Figures 13a, 13b and 13c present data for spherically-blunted tangent ogives of fineness ratio 1 and blunting ratios, ρ_n , of 0, 0.6 and 0.8, respectively, for Mach numbers up to 1. Figures 14a to 14c and 15a to 15c deal similarly with fineness ratios of 1.5 and 2, respectively. The extreme blunting ratio case $\rho_n = 1$ (i.e. the hemispherical head) is presented in Figure 3. No data have been given for blunting ratios between 0 and 0.6 because the effects on pressure distribution in this range are not as significant as for higher blunting ratios. The particular values of 0.6 and 0.8 have been selected for inclusion since the forebody normal pressure drag usually attains a minimum somewhere between the two for high subsonic Mach numbers and fineness ratios in the range 1 to 2 (see Reference 24).

The mechanism by which the minimum in normal pressure drag is achieved can be explained in general terms by reference to the development of the pressure distribution for a given fineness ratio and Mach number as the blunting ratio is increased. The pressure distribution from the nose ($x = 0$) to a certain point on the forebody gives rise to a drag component while the pressure distribution between this point and the junction with the cylinder gives rise to a thrust component. In general, as the blunting ratio is increased from about 0.6 to 0.8 or 0.9 the drag and thrust components increase. However, the increase in the thrust component initially outweighs the increase in the drag component, with the net result that the overall normal pressure drag for the forebody attains a minimum somewhere in the blunting ratio range 0.6 to 0.8.

The data presented in Figures 13a to 15c were obtained (Derivation 19) from finite-difference calculations using the method of Derivation 17.

5. ESTIMATION OF p_{H0}

$M_1 \leq 1$:

For $M_1 \leq 1$ the isentropic flow relation for air

$$\frac{p_1}{p_{H1}} = \left(\frac{5}{5 + M_1^2} \right)^{3.5} \quad (5.1)$$

may be used to determine $p_{H0}(= p_{H1})$.

$M_1 > 1$:

For $M_1 > 1$ a bow shock wave is present. It is detached from the body for a blunt-nosed body, or a sharp-nosed body with nose semi-apex angle greater than a certain limiting value which is dependent on M_1 (see Item No. Aero S.00.03.16, Reference 20, for cones). In this situation a good approximation to the stagnation pressure at the nose, p_{H0} , is given by the normal shock relation for air

$$\frac{p_1}{p_{H2}} = \left(\frac{7M_1^2 - 1}{6} \right)^{2.5} \left(\frac{5}{6M_1^2} \right)^{3.5}, \quad (5.2)$$

in which $p_{H2} = p_{H0}$ for the present purpose.

Figures 16a and 16b presents p_1/p_{H1} and p_1/p_{H2} against M_1 .

6. ESTIMATION OF LOCAL SURFACE MACH NUMBER AND TEMPERATURE

For isentropic inviscid flow in air the relationship between local surface values of Mach number and static pressure is given by Equation (5.1) when applied to local values and rearranged, *i.e.*,

$$M = \left\{ 5 \left[\left(\frac{p}{p_H} \right)^{-2/7} - 1 \right] \right\}^{1/2} \quad (6.1)$$

in which p_H is here the stagnation pressure appropriate to the point on the body under consideration. For the purposes of this Item it is satisfactory to assume that p_H in Equation (6.1) is equal to the stagnation pressure at the nose, p_{H0} , and the pressure distribution data given in the Figures herein may be used directly in the equation. This is reasonable even for those cases where a shock exists on the body (in Figures 3 to 6 and 13a to 15a). The pressure distribution data for these cases were obtained by a method (Derivation 17) which assumed no change in stagnation pressure across the shock, see Section 7.

In isentropic inviscid flow in air with no heat transfer to or from the surface, the local surface temperature is given by

$$\frac{T}{T_1} = \left(\frac{p}{p_h} \right)^{2/7} (1 + 0.2M_1^2) \quad (6.2)$$

in which, as in Equation (6.1), p_H is the stagnation pressure appropriate to the point on the body under consideration, assumed equal to p_{H0} for the purposes of this Item.

For real, viscous, flows over bodies with heat transfer the local Mach number and temperature obtained from Equations (6.1) and (6.2) will be good approximations to the values at the edge of the boundary layer and could be used as input to Item No. 69012 (Derivation 22) to determine the equilibrium surface temperature distribution.

7. APPLICABILITY AND ACCURACY

The data for the pressure distributions over the four families of blunt axisymmetric forebodies at zero incidence given in this Item have been derived by three basic means. These were, firstly, a straightforward presentation of experimental data, secondly, correlation of experimental data and, thirdly, a straightforward presentation of the results of calculations using inviscid theoretical methods based on either finite-difference techniques or the method of characteristics. In order to provide a complete coverage for some of the forebodies the data used were sometimes a combination of experimental and theoretical data. Table 10.1 gives a detailed breakdown of the sources of data for each pressure distribution Figure.

The range of applicability of the Figures in terms of Mach number is self-evident in most cases by reference to the Figure. There are two exceptions to this: the Figures involving the forebody family having a spherically-curved nose with a radiused shoulder (Figures 8a to 10a) and the Figure involving large-angle cones (Figure 12).

The data used (Derivations 11 and 14) to construct Figures 8a, 9a and 10a were limited to the Mach number range from 2.5 to 4.6 and over that range the factors K_n and K_s can be considered to be substantially independent of Mach number. In view of the fact that Mach number effects on pressure distribution reduce as Mach number increases, the use of Figures 8a, 9a and 10a is expected to be satisfactory for any Mach number above about 2. For Mach numbers less than about 2 these Figures should be used with caution.

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The data used (Derivations 12, 13 and 16) to construct Figure 12 (except for $\delta = 90^\circ$) were limited to the Mach number range 2.3 to 4.6. The effect of Mach number reduces as the cone semi-apex angle (δ) increases. The result is that as δ increases the range of applicability of the curves increases, as shown in Table 10.1, *i.e.*

δ (degree)	60	70	80	90
Range of M_1	3 to 5	≥ 2	≥ 2	≥ 1

Figure 12 should be used with caution for Mach numbers outside these ranges. The curves can be assumed to apply to sharp cones or blunted cones with $\rho_n < 0.5$, say.

The effect of Reynolds number on the pressure distribution over any forebody is considered to be insignificant provided that R_D is greater than about 10^5 .

The scatter of the experimental data for any given forebody at a particular free-stream Mach number together with comparisons between the results of the theoretical methods and various experimental data have indicated an overall accuracy of about ± 0.02 in the curves for p/p_{H0} , except near any shock on the body where poorer accuracy should be expected.

With regard to the estimation of local surface Mach number using Equation (6.1) for those cases where a shock exists on the body, it was noted in Section 6 that the method of Derivation 17 assumed no change of stagnation pressure across the shock. This is commented on in Reference 23 and the conclusion drawn is that, even if the change in stagnation pressure is allowed for, the error in local Mach number downstream of the shock will be less than about 3 per cent provided that the local Mach number just upstream of the shock is less than about 1.3. This is a problem area in real, viscous, flows anyway since, with shock upstream Mach numbers of about 1.3 to 1.4*, there is a likelihood of shock-induced separation. In using Equation (6.1) for those cases with a bow shock it is implicit that the stagnation pressure downstream of the shock is constant along a streamline and is not noticeably affected by viscosity.

8. DERIVATION AND REFERENCES

8.1 Derivation

The Derivation lists selected sources that have assisted in the preparation of this Item.

1. MOECKEL, W.E. Experimental investigation of supersonic flow with detached shock waves for Mach numbers between 1.8 and 2.9. NACA RM E50D05 (TIB 2416), 1950.
2. CHAUVIN, L.T. Pressure distribution and pressure drag for a hemispherical nose at Mach numbers 2.05, 2.54 and 3.04. NACA RM L52K06 (TIB 3547), 1952.
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* The actual value is likely to be critically dependent on the state of the boundary layer.

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8.2 References

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9. EXAMPLE

In order to determine the structural strength of a missile nose fairing the pressure, Mach number and temperature distributions are required. The fairing is ellipsoidal with a fineness ratio of 1. The missile is assumed to be travelling at a Mach number (M_1) of 0.9 at zero incidence in an ambient atmospheric pressure (p_1) of $101.3 \times 10^3 \text{ N/m}^2$ and temperature (T_1) of 288 K.

From Figure 4, for $M_1 = 0.9$, data for the variation of p/p_{H0} with x/l_f up to $x/l_f = 1$ can be extracted and tabulated as follows:

TABLE 9.1

x/l_f	0	0.05	0.1	0.2	0.3	0.4	0.5	0.6	0.7	0.8	0.9	1.0
p/p_{H0}	1	0.855	0.765	0.643	0.570	0.518	0.476	0.441	0.412	0.387	0.366	0.355

From Equation (5.1) at a Mach number of 0.9

$$\frac{p_1}{p_{H1}} = \left[\frac{5}{5 + (0.9)^2} \right]^{3.5} = 0.591.$$

Thus $p_{H1} = p_{H0} = 101.3/0.591 = 171.4 \times 10^3 \text{ N/m}^2$, which is used to determine the local static pressure (p) distribution from the values of p/p_{H0} in Table 9.1.

The distribution of pressure coefficient (C_p) is similarly obtained using Equation (4.1).

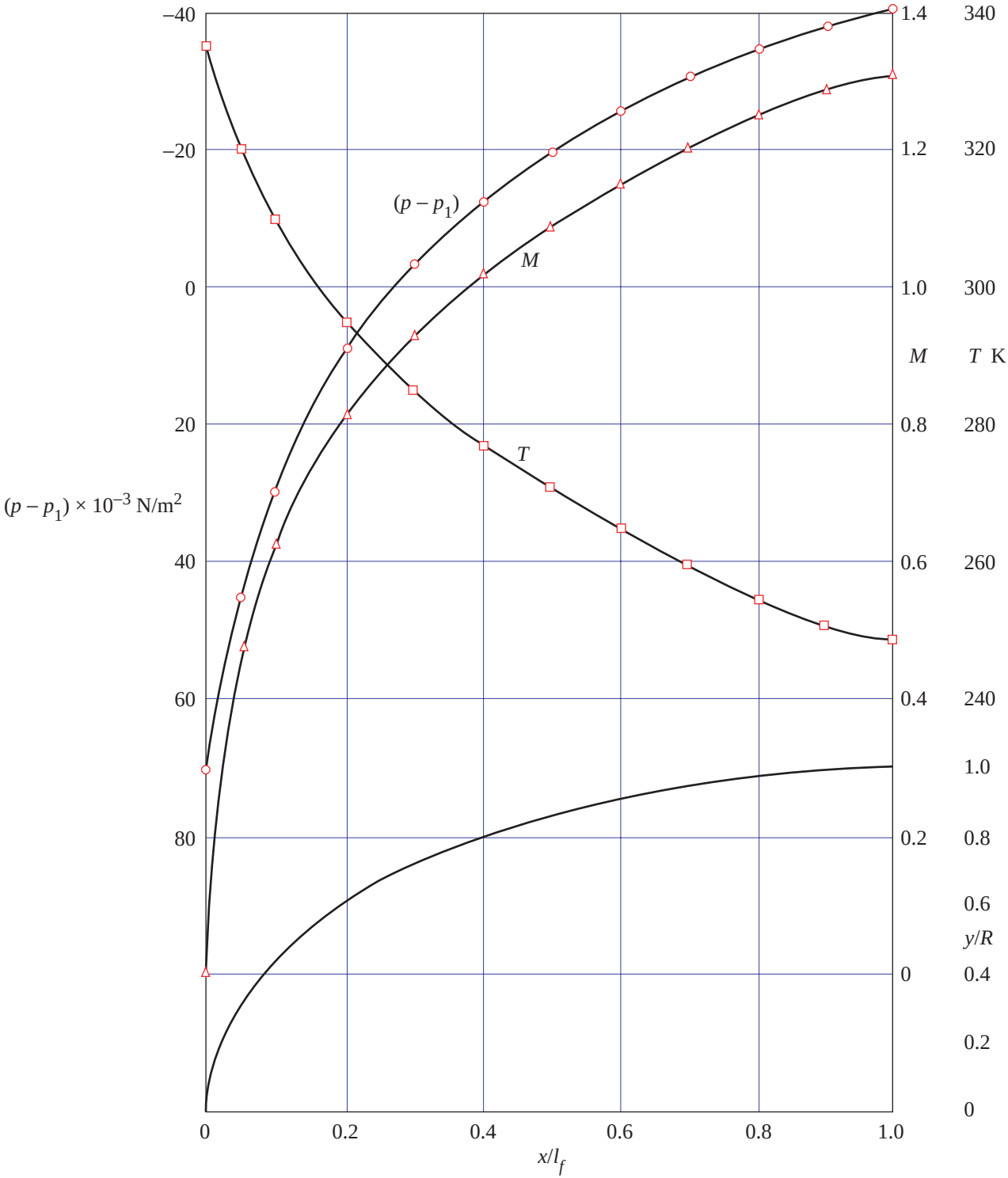
The local Mach number (M) and temperature (T) distributions are calculated using Equations (6.1) and (6.2) with $p/p_H = p/p_{H0}$ given in Table 9.1.

The values of pressure, pressure difference ($p - p_1$), pressure coefficient, Mach number and temperature resulting from the calculations are given in Table 9.2 and with the exceptions of p and C_p are illustrated graphically in Sketch 9.1.

TABLE 9.2

x/l_f	0	0.05	0.1	0.2	0.3	0.4	0.5	0.6	0.7	0.8	0.9	1.0
$p \times 10^{-3} \text{ (N/m}^2\text{)}$	171	147	131	110	97.7	88.8	81.6	75.6	70.6	66.3	62.6	60.8
$(p - p_1) \times 10^{-3} \text{ (N/m}^2\text{)}$	70.1	45.2	29.8	8.9	-3.6	-12.5	-19.7	-25.7	-30.7	-35.0	-38.6	-40.5
C_p	1.22	0.788	0.519	0.155	-0.063	-0.218	-0.343	-0.448	-0.534	-0.609	-0.674	-0.704
M	0	0.478	0.631	0.820	0.933	1.02	1.09	1.15	1.20	1.25	1.29	1.31
$T \text{ (K)}$	335	320	310	295	285	277	271	265	260	255	251	249

In order to establish the equilibrium surface temperature distribution in a real viscous flow with heat transfer Item No. 69012 (Derivation 22) should be used.



Sketch 9.1

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10. TABLE GIVING DATA SOURCES

TABLE 10.1 Table of Source Data for Figures 1 to 15c

Forebody Geometry	Fig No.	M_1	Derivation	Comments
Ellipsoidal				
$l_f/D = 0$ (flat-face)	1a	≥ 1	3 to 5, 13, 14	Expt., $R_D = 0.23$ to 2×10^6
$l_f/D = 0.25$	1b	1 to 2	4, 5	Expt., $R_D = 0.23 \times 10^6$
$l_f/D = 0.333$	1c	4, 6, ∞	9	Theory, method of charac. (inviscid)
$l_f/D = 0.5$ (hemispherical head)	2	0.4 to 1.0	19	Theory, finite-diff. (inviscid)
		1.5	4, 5	Expt., $R_D = 0.23 \times 10^6$
		2.0	18	Theory, finite-diff. (inviscid); $y/R \leq 0.85$
		2.5	1, 2, 7, 8	Expt., $R_D = 0.44$ to 6.6×10^6 ; $y/R > 0.85$
		3, 6, ∞	2, 6, 14	Expt., $R_D = 0.71$ to 6.6×10^6
	3	0.4 to 1.0	10	Theory, method of charac. (inviscid)
		1.5	19	Theory, finite-diff. (inviscid)
		2.0	4, 5	Expt., $R_D = 0.23 \times 10^6$
		2.5	18	Theory, finite-diff. (inviscid); $x/l_f \leq 0.5$
		3, 6, ∞	1, 2, 7, 8	Expt., $R_D = 0.44$ to 6.6×10^6 ; $x/l_f > 0.5$
$l_f/D = 1$	4	0.4 to 1	2, 6, 14	Expt., $R_D = 0.71$ to 4.6×10^6 ; $x/l_f \leq 1$
		1 to 2	14	Expt., $R_D = 1.9 \times 10^6$; $x/l_f > 1$
		3, 4, 6, ∞	10	Theory, method of charac. (inviscid)
$l_f/D = 1.5$	5	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 2$	6	0.4 to 1	19	Theory, finite-diff. (inviscid)
		1 to 2	4, 5	Expt., $R_D = 0.23 \times 10^6$
$l_f/D = 4$	7	1 to 2	4, 5	Expt., $R_D = 0.23 \times 10^6$
Spherically-curved nose with radiused shoulder				* See Section 4.2 and 7
$\rho_s = 0$, ρ_n varying (sharp shoulder)	8a to 8d	*	11, 14*	Expt., $R_D = 1.9 \times 10^6$
$\rho_n = 2$, ρ_s varying	9a to 9d	*	11, 14*	Expt., $R_D = 1.9 \times 10^6$
$\rho_n = \infty$, ρ_s varying	10a to 10d	*	11, 14*	Expt., $R_D = 1.9 \times 10^6$

TABLE 10.1 Table of Source Data for Figures 1 to 15c (Continued)

Forebody Geometry	Fig No.	M_1	Derivation	Comments
Spherically-blunted cone				
$\delta = 0, 5^\circ, 10^\circ, 20^\circ, 30^\circ, 40^\circ$ $\delta = 45^\circ$	11a	3	10 16	Theory, method of charac. (inviscid) Expt., $R_D = 1.1 \times 10^6$
$\delta = 0, 5^\circ, 10^\circ, 20^\circ, 30^\circ, 40^\circ$	11b	4	10	Theory, method of charac. (inviscid)
$\delta = 0, 5^\circ, 10^\circ, 20^\circ, 30^\circ, 40^\circ$	11c	6	10	Theory, method of charac. (inviscid)
$\delta = 0, 30^\circ, 40^\circ$ $\delta = 5^\circ, 10^\circ, 15^\circ, 20^\circ$	11d	10	10 15	Theory, method of charac. (inviscid) Theory, finite-diff. (inviscid)
$\delta = 0, 30^\circ, 40^\circ$ $\delta = 5^\circ, 10^\circ, 15^\circ, 20^\circ$	11e	∞	10 15	Theory, method of charac. (inviscid) Theory, finite-diff. (inviscid)
$\delta = 60^\circ, \rho_n < 0.5$ $\delta = 70^\circ, \rho_n < 0.5$ $\delta = 80^\circ, \rho_n < 0.5$ $\delta = 90^\circ$	12	3 to 5 ≥ 2 ≥ 2 ≥ 1	12 13, 16 13 13	Expt., $R_D = 2 \times 10^6$ Expt., $R_D = 1.1, 2 \times 10^6$ Expt., $R_D = 2 \times 10^6$ Expt., $R_D = 2 \times 10^6$ (See also Derivation 12, 13, 16)
Spherically-blunted tangent ogive				
$l_f/D = 1, \rho_n = 0$	13a	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 1, \rho_n = 0.6$	13b	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 1, \rho_n = 0.8$	13c	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 1.5, \rho_n = 0$	14a	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 1.5, \rho_n = 0.6$	14b	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 1.5, \rho_n = 0.8$	14c	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 2, \rho_n = 0$	15a	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 2, \rho_n = 0.6$	15b	0.4 to 1	19	Theory, finite-diff. (inviscid)
$l_f/D = 2, \rho_n = 0.8$	15c	0.4 to 1	19	Theory, finite-diff. (inviscid)