

THE CORRECTION OF FLIGHT-TEST ANEMOMETRIC DATA

1. NOTATION, UNITS, CONSTANTS AND CONVERSION FACTORS

Most numerical expressions in this Item are expressed in non-dimensional form so that correct results are obtained with any consistent set of units. Where dimensional results are given, the following systems of units are used

- (i) coherent SI units with speeds in m/s and pressure in N/m²,
- (ii) non-coherent SI units with speeds* in kn and pressures in mb,
- (iii) non-coherent British units with speeds* in kn and pressures in lbf/ft².

Numerical constants and conversion factors are given in Sections 1.2 and 1.3.

1.1 Notation and Units

		<i>SI</i>	<i>British</i>
a	speed of sound*	m/s, kn	kn
A_{sp}	total area of static port(s)	m ²	ft ²
C_{p_L}	local pressure coefficient at total temperature-probe station, $(p_{s_L} - p_a)/0.7p_a M^2$		
E	electrical rating of temperature probe	mW	mW
f	denotes functional relationship of type $f[\dots]$		
f_x	reading of body-mounted accelerometer aligned with fuselage longitudinal datum† and expressed in units of g_0 , $(-\ddot{x}_{in} - g \sin \Theta)/g_0$		
f_γ	temperature correction factor for effects of caloric imperfection $(= (\gamma - 1)/0.4)$, see Section 3.3.4		
g	acceleration due to gravity	m/s ²	ft/s ²
H	geopotential height	m, ft	ft
H_p	pressure height	m, ft	ft
h_e	energy height	m, ft	ft
k	volumetric proportion of pressure system within pressure shell of aircraft		

For footnotes see end of Notation.

* The knot (kn) is one international nautical mile per hour, see Section 1.3.

l	characteristic length used in definition of Reynolds number; in this Item it is the length appropriate to boundary layer growth at static sensor ports	m	ft
l_x	effective length of a pressure system as derived for inertial corrections, <i>i.e.</i> distance from transducer to pressure sensor port(s) projected on to fuselage longitudinal datum (forward is positive)	m	ft
M	Mach number, V/a ($= V/(a_0 \theta^{1/2}) = V_e/(a_0 \delta^{1/2})$)		
M'	Mach number obtained after correcting for any non-steady flight effects, see Section 6.2.4		
M_L	local Mach number at edge of airframe boundary layer at total temperature probe station in the absence of the probe		
p_a	ambient static pressure [‡]	N/m ² , mb	lbf/ft ²
p_p	fully corrected value of pitot pressure [‡] in pitot system for steady-level-flight conditions	N/m ² , mb	lbf/ft ²
p_p'	sensed pitot pressure [‡] in steady-level-flight conditions (alternatively, value obtained after correction for any effects of non-steady flight)	N/m ² , mb	lbf/ft ²
p_{p_i}	value of pitot pressure [‡] registered (indicated) by transducer, (note for steady level flight $p_{p_i} = p_p'$) see Section 4.1	N/m ² , mb	lbf/ft ²
p_s	sensed static pressure [‡] in static system in steady-level-flight conditions	N/m ² , mb	lbf/ft ²
p_{s_i}	value of static pressure [‡] registered (indicated) by transducer (note for steady level flight, $p_{s_i} = p_s$)	N/m ² , mb	lbf/ft ²
p_{s_L}	static pressure [‡] on airframe surface at total temperature probe station	N/m ² , mb	lbf/ft ²
p_t	total (or isentropic) pressure [‡] , $p_a(1 + 0.2M^2)^{3.5}$	N/m ² , mb	lbf/ft ²
q	kinetic pressure, $0.7p_aM^2$	N/m ² , mb	lbf/ft ²
q_c'	impact pressure for steady-level-flight conditions, $(p_p' - p_s)$	N/m ² , mb	lbf/ft ²
r	overall recovery factor for total temperature probe for $M \leq 1$, $(= [(T_r/T_a) - 1]/0.2M^2)$		
r'	effective recovery factor for total temperature probe for $M > 1$		

For footnotes see end of Notation.

R	gas constant for air	N m/kg K	lbf ft/slug K
Re	Reynolds number based on freestream conditions and length, l ($= \rho V l / \mu$ for coherent units) see Section 8.6		
SEP	specific excess power, dh_e/dt	m/s	ft/s
T_a	ambient temperature	K	K
$T_{a_{std}}$	ambient temperature at height H_p in International Standard Atmosphere	K	K
T_p	pitot pressure system mean temperature	K	K
T_r	temperature registered by ideal total temperature probe (recovery temperature) or value after correction for probe errors, see Section 8.2	K	K
T_{r_i}	temperature actually registered by a total temperature probe	K	K
T_s	static pressure system mean temperature	K	K
T_t	total air temperature, $T_a(1 + 0.2M^2)$	K	K
t	time	s	s
V	true airspeed*	m/s, kn	kn
V_c	calibrated airspeed*	m/s, kn	kn
V_e	equivalent airspeed*, $V\sigma^{1/2}$	m/s, kn	kn
V_i	indicated airspeed*	m/s, kn	kn
V_w	horizontal component of wind velocity* (relative to Earth)	m/s, kn	kn
V_{ol}	air volume within a pressure system	m ³	ft ³
x	co-ordinate in x -direction, <i>i.e.</i> fuselage longitudinal datum or body axis (Sketch 1.1 shows positive directions)	m	ft
α	angle of attack (angle between projection of true airspeed vector on to aircraft plane of symmetry and angle of attack datum)	deg	deg
α_{fus}	local angle of attack of fuselage, allowing for effects of downwash and local changes in fuselage datum setting	deg	deg

For footnotes see end of Notation.

α_p	local angle of attack of a pitot or pitot-static probe	deg	deg
β	angle of sideslip (angle between true airspeed vector and its projection on to aircraft plane of symmetry)	deg	deg
β_{fus}	local angle of sideslip of fuselage	deg	deg
γ	ratio of specific heat capacities of air		
ΔC_{p_a}	coefficient form of static-pressure-error correction (position-error correction), ($p_a - p_s$)/0.7 $p_a M^2$ (= Δp_a /0.7 $p_a M^2$)		
Δp_a	static-pressure-error correction, ($p_a - p_s$)	N/m ² , mb	lbf/ft ²
ΔT_a	ambient temperature increment relative to ISA, ($T_a - T_{a_{std}}$)	K	K
δ	relative ambient pressure, p_a/p_{a_0}		
δp_p	contribution to pressure error in pitot-pressure system	N/m ² , mb	lbf/ft ²
δp_s	contribution to pressure error in static-pressure system	N/m ² , mb	lbf/ft ²
δT_r	contribution to error in temperature registered by total temperature probe	K	K
δ_η	deflection of aircraft pitch control surface	deg	deg
ε_p	angle of downwash at pitot-pressure probe	deg	deg
λ	lag time constant	s	s
Θ	inclination angle, <i>i.e.</i> pitch attitude of fuselage datum to horizontal plane	deg	deg
θ	relative ambient temperature, T_a/T_{a_0}		
θ_{fus}	setting of fuselage datum relative to angle of attack datum	deg	deg
θ_p	pitot-pressure-probe setting angle measured relative to angle of attack datum in aircraft plane of symmetry	deg	deg
μ	coefficient of dynamic viscosity	N s/m ²	lbf s/ft ²
ρ	ambient air density	kg/m ³	slug/ft ³
$\Sigma(\delta p)$	summation of pressure system errors	N/m ² , mb	lbf/ft ²
σ	relative ambient density, ρ/ρ_0		

For footnotes see end of Notation.

ϕ	geographic latitude, positive in Northern Hemisphere, negative in Southern Hemisphere	deg	deg
χ_W	wind azimuth angle (North corresponds to $\chi_W = 0$ and χ_W differs by 180° from wind direction used in meteorology, for example, $\chi_W = 90^\circ$ for wind from the West)	deg	deg
τ	acoustic lag time constant for a pressure system	s	s
ψ	azimuth angle (<i>i.e.</i> aircraft heading; $\psi = 0$ corresponds to North)	deg	deg
ω_p	resultant angle of incident flow at pitot-pressure probe	deg	deg

Superscript

–	denotes the average value of a parameter taken over an interval of time
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Subscripts

a	denotes ambient conditions
ad	denotes adiabatic flow process
$crit$	denotes a critical value
$hole$	denotes a geometric change to static pressure sensor ports
i	initial value of a parameter, usually a registered value, that is subject to corrections to arrive at a true value
in	denotes either: inertial error in pressure system, or: value relative to inertial frame of reference
int, s	denotes interference flow error in static-pressure-system pressure
lag	denotes lag error
$local$	denotes value at pitot probe station, outside airframe boundary layer
ns	denotes non-steady flight conditions
p	denotes pitot-pressure system
pe	denotes pressure error due to position of pitot probe
rad	denotes radiation error

For footnotes see end of Notation.

s	denotes static-pressure system
sf	denotes steady level flight conditions
sh	denotes self-heating error
std	denotes value of parameter in ISA conditions
T	denotes total temperature probe
ws	denotes weather system correction to specific excess power
ω	denotes error due to local incident flow angle
0	denotes conditions at sea level in International Standard Atmosphere, ISA
1	denotes stream conditions ahead of normal shock
2	denotes stream conditions behind normal shock

tion.

* The knot (kn) is one international nautical mile per hour, see Section 1.3.

† See Item No. 79018, Section 5 for accelerometer equations and associated sign conventions.

‡ Note that these pressures are absolute.

1.2 Constants Used in the Method and Associated Conversion Factors

In the following Table the SI values are those defined for sea-level conditions in the International Standard Atmosphere (see Reference 10). Values in British units are obtained using the *exact* values of the conversion factors given in Section 1.3.

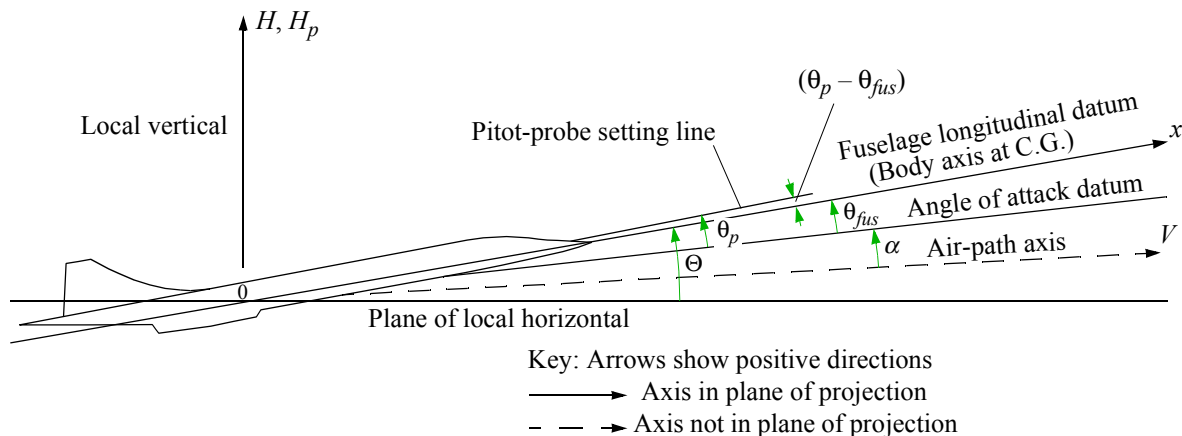
Quantity	SI units	British units	To convert from SI to British units multiply by	
			<i>exact</i>	<i>approx.</i>
a_0	340.294 m/s	661.479 kn	$\frac{3600}{1852}$	1.943 84
	661.479 kn	1 116.45 ft/s	$\frac{1852}{3600 \times 0.3048}$	1.687 81
g_0	9.806 65 m/s ²	32.174 0 ft/s ²	$\frac{1}{0.3048}$	3.280 84
p_{a_0}	101 325 N/m ² (1013.25 mb)	2116.22 lbf/ft ²	$\frac{(0.3048)^2}{9.806 65 \times 0.453 592 37}$	0.020 885 4
R	287.052 87 N m/kg K	3089.811 4 ft lbf/slug K	$\frac{1}{(0.3048)^2}$	10.763 9
T_{a_0}	288.15 K	288.15 K	1	1
ρ_0	1.225 kg/m ³	$2.376 89 \times 10^{-3}$ slug/ft ³	$\frac{(0.3048)^4}{0.453 592 37 \times 9.806 65}$	0.001 940 32
γ	1.4 [*]	1.4 [*]	1	1

* Effects of changes in the value of γ are considered in Sections 3.3.4, 5.2.2 and 8.2 and in Appendix A.

1.3 Exact Conversion Factors Used or Implied

$$\begin{aligned}
 1 \text{ ft} &= 0.3048 \text{ m}, & 1 \text{ n mile} &= 1852 \text{ m}, \\
 1 \text{ bar} &= 10^5 \text{ N/m}^2, & {}^\dagger 1 \text{ lbm} &= 0.453 529 37 \text{ kg}, \\
 (T_a \text{ or } T_r \text{ or } T_t \text{ } ^\circ\text{C}) &= (T_a \text{ or } T_r \text{ or } T_t \text{ K}) - 273.15.
 \end{aligned}$$

[†] The British unit of mass used in the Performance Series is the slug.
 1 slug = 32.1740 lbm.



Sketch 1.1 Angles and axis definitions in aircraft plane of symmetry (unbanked flight)

2. INTRODUCTION

Anemometric data are the numerical values of those quantities which define completely the ambient conditions and airspeeds of flight for an aircraft. Practical considerations prevent such data being obtained directly and all that is possible is to register by transducers (or other instruments) the values of certain sensed parameters. The primary sensed parameters consist of pitot and static pressures, total air temperature (recovery temperature) from a total temperature probe and aircraft angles of attack and sideslip. Air data systems which are used for sensing and registering such flight data are described in Data Item No. 86031, "Introduction to air data system parameters, errors and calibration laws", (Reference 16). That Item introduces both the present Item and References 17 to 20. In all these Data Items the value of a "registered" parameter is termed "indicated", the convention implying that corrections have to be applied to obtain true values. True values of anemometric data are of prime importance for all aspects of the performance and airworthiness certification of aircraft.

This Item presents a comprehensive and systematic approach to the correction of indicated anemometric data although, because of its general applicability, some of the correction processes are not required for particular flight cases. The Item is directed particularly to measurements made during flight tests on prototype or test aircraft and the methods should be generally applicable for speeds* up to $M \approx 4$ and for heights* up to 120 000 ft provided that, for each flight case analysed, the flight path is approximately rectilinear. This last restriction is desirable rather than essential since it avoids the need for more complex correction processes not considered in the Item. A further restriction is the assumption that conditions of turbulence, icing-risk or high humidity are avoided by judicious flight planning.

In the case of supersonic aircraft, the methods assume a slender forebody and, in particular, the use of static probes (for example, as on a boom-mounted pitot-static nose probe). Supersonic aircraft with blunt forebodies need special treatment for data analysis – see Appendix A.

* The Mach number limit is set by consideration of the changes needed to the methods used in its derivation, due to caloric and thermal imperfection in the airflow (see Section 3.3.4) at higher values. The quoted combination of M and H_p corresponds to a value of

$$0.7p_a M^2 = \frac{\text{wing loading}}{\text{lift coefficient}} = 250 \text{ lbf/ft}^2 \text{ (} 12 \times 10^3 \text{ N/m}^2 \text{)}.$$

It is unlikely that lower values of this parameter would be appropriate to rectilinear flight at supersonic speeds.

The layout of the Item is as follows.

Section 3 outlines the air data systems' geometric data and the sensor calibration data required for use in the correction processes.

Section 4 states the requirements for the flight data in terms of their form and quantity.

Section 5 deals with the preliminary processing of the flight data for steady (Section 5.1.1) and non-steady (Section 5.1.2) flight conditions. For the latter, the corrections for lag effects in pressure and temperature systems are considered in Section 5.2.4. Further "lag-like" effects are considered, in Section 5.2.4 for cases where the static system is vented at small ports so producing an "interference pressure error", and in Section 5.2.5 for cases where the longitudinal acceleration is extreme.

Section 6 presents the corrections required to obtain true pressures in the air data systems, the static-pressure-error correction and hence ambient static pressure. (The derivation of ambient pressure provides an example of the need to apply corrections in the proper sequence.)

Section 7 defines relationships for those anemometric data that depend only upon pitot and ambient pressures, *i.e.* H_p , M , V_c , V_e , q , p_t .

Section 8 gives the steps required to obtain ambient temperature, T_a , and to derive other temperature-related quantities of interest, *i.e.* θ , ΔT_a , σ , ρ , μ , V , Re .

Section 9 gives a method for obtaining specific excess power, SEP .

Flow Charts A to D show the correct sequences in which the many individual calculation steps of this Item should be applied.

Flow Chart A lays out the "core" of the calculation processes and for many cases of steady flight is a complete description of the method.

Flow Chart B gives the steps required for the analysis of non-steady conditions and also when Reynolds number is needed for static-pressure-error corrections.

Flow Chart C gives the additional calculations needed when the static system is vented by a few small ports (see Section 5.2.4).

Flow Chart D gives the additional calculation steps required for conditions of extreme longitudinal acceleration when inertial effects may become significant in the recorded pressures in the pitot or static pressure systems.

Appendix A deals with the treatment of temperature-dependent quantities for stagnation temperatures such that $400 < T_{t2} \leq 1\,000$ K.

This Item is complementary to sources such as References 1, 5, 8 and 14 which deal mainly with the methods of carrying out flight tests, including the calibration of sensors, and to sources such as Reference 7 and 12, which are concerned with instrumentation.

3. AIR DATA SYSTEMS' GEOMETRIC AND CALIBRATION INFORMATION REQUIRED FOR USE IN THE CORRECTION PROCESSES

Before the flight data can be corrected, a knowledge of the errors in the pressure and temperature systems is necessary together with appropriate geometric properties of the aircraft and its anemometric systems.

This section lists the aircraft and systems information required. Not all of the items listed are essential in every case so indications are given concerning their application.

3.1 Calibrated and Computed Data for Pressure Systems in Steady-Flight Conditions

The calibrations which produce these data are assumed to be made on systems which comply with the standards of Reference 8, so that errors in pressure recording due to leaks may be neglected.

3.1.1 Definitions of angles and axes

The axes and setting lines shown in Sketch 1.1 are defined as follows.

- (i) The air-path axis is directed along the aircraft true airspeed vector, V , and so lies in the aircraft plane of symmetry only when sideslip angle, β , is zero.
- (ii) The angle of attack datum is defined in the aircraft plane of symmetry. It is taken here as the aircraft zero-lift line for the given configuration (flap setting, centre of gravity position, *etc*) and Mach number and, for some aircraft, may vary with height.
- (iii) The fuselage longitudinal datum is defined in the aircraft plane of symmetry and in this Item is identical with the aircraft "x" body axis. It is the datum to which pitch attitude angle, θ , is measured and is set at angle θ_{fus} to the angle of attack datum.
- (iv) The pitot-probe setting line is defined in the aircraft plane of symmetry and lies at angle θ_p to the angle of attack datum and hence at angle $(\theta_p - \theta_{fus})$ to the fuselage longitudinal datum.

Note that pitot probe setting angle, θ_p , may be affected by airframe aeroelastic properties and may therefore change under the influence of gravitational, aerodynamic and inertial forces.

3.1.2 Pitot-probe pressure error (see Data Item, Reference 17)

Pitot-pressure probes are relatively accurate as sensors of stagnation pressure* of a moving fluid. An error, $(\delta p_p)_\omega$, occurs, however, when the probe incidence angle, ω_p , relative to the local flow exceeds a critical value, $\omega_{p_{crit}}$. Also, at supersonic speeds, for a probe within the flow field of an aircraft, an error, $(\delta p_p)_{pe}$, may arise from the position of the probe. Errors from these two sources may be determined by flight calibration using, as a standard, a shielded pitot or a nose probe, respectively. Methods for estimating these errors are given in Reference 17, so that the total error, $\delta p_p = (\delta p_p)_\omega + (\delta p_p)_{pe}$, may be obtained.

Some general statements from the Reference are as follows:

- (i) Values of the error $(\delta p_p)_\omega$ and the angle $\omega_{p_{crit}}$ are very sensitive to probe geometry and are best obtained from probe manufacturers' data. An appreciation of these quantities may be obtained, however, from References 8 and 12 where $\omega_{p_{crit}}$ for normal configurations is seen to lie between 10 deg and 20 deg.

* At supersonic speeds this includes the loss of pressure due to a normal shock at entry to the probe.

- (ii) Reference 17 shows that a typical value of the error, when $(\omega_p - \omega_{p_{crit}}) = 10$ degrees, is

$$(\delta p_p)_\omega = -0.060 \frac{\gamma}{2} p_a M^2. \quad (3.1)$$

- (iii) For a mast-mounted pitot stationed on the maximum width of a circular section fuselage the angle of incident flow* is affected by the cross-flow velocity field of the fuselage and, for $\beta = 0$,

$$\omega_p = \alpha_p \approx 2 \left[\alpha \left(1 - \frac{d\varepsilon_p}{d\alpha} \right) + \theta_{fus} \right] + (\theta_p - \theta_{fus}). \quad (3.2)$$

- (iv) For a boom-mounted nose probe the angle of incident flow* is given (for β small) by

$$\omega_p = \cos^{-1} \left\{ \cos \left[\alpha \left(1 - \frac{d\varepsilon_p}{d\alpha} \right) + \theta_p \right] \cos \beta \right\}. \quad (3.3)$$

- (v) When a pitot is positioned so that it is subject to airframe interference, there is for supersonic speeds a “position-error” which must be taken into account. This arises because the shock loss at the mouth of the pitot (given by the “normal-shock” relationships) is a function of the local Mach number, M_{local} , of the flow at the probe location. For given flight conditions the value of M_{local} (and hence both the magnitude and sign of $(\delta p_p)_{pe}$) depends on airframe geometry and so will differ for different locations on the forebody or any further-aft part of the airframe.

For any particular installation

$$\frac{(\delta p_p)_{pe}}{p_{p_{local}}} = f[M', \alpha_{fus}, \beta_{fus}]. \quad (3.4)$$

This relationship is best obtained by flight tests, using a nose probe as a standard. Alternatively, wind-tunnel tests at large scale should be satisfactory. In the absence of such calibrations the error may be estimated for a strut-mounted pitot on a forebody (see Reference 17).

The equivalent steady level flight Mach number, M' , (see Section 6.2.4) is used in Equations (3.1) and (3.4) and in any variants of these general expressions.

3.1.3 Static-pressure-error correction, Δp_a

The ports of the static-pressure system cannot, in general, sense true ambient pressure due to their location in the aircraft flow field. The correction for this pressure error, sometimes known as position-error correction, is determined by flight calibrations carried out in steady conditions. In some cases, wind-tunnels may be useful in a complementary role. For non-steady conditions corrections have to be applied first to adjust the data to equivalent-steady-state values. Section 6 gives the necessary calculation procedure in the correct sequence.

Data Item No. 86006 (Reference 17) discusses the methods of using these calibrations in flight-test analysis work. The static-pressure-error correction, Δp_a , is usually reduced to one of the following non-dimensional

* In Equations (3.2) and (3.3), $d\varepsilon_p/d\alpha$ is negative for any probe stationed ahead of the wing, and is zero for $M > 1$.

functional relationships, applicable to steady-flight conditions and to given high-lift-device settings, *etc.*,

$$\frac{\Delta p_a}{q_c'} = f\left[\alpha, \frac{q_c'}{p_s}\right], \quad (3.5)$$

or, if Reynolds number is also a variable*,

$$\frac{\Delta p_a}{q_c'} = f\left[\alpha, \frac{q_c'}{p_s}, Re\right]. \quad (3.6)$$

For certain types of aircraft, such as those of delta planform, the pitch-control deflection may have an influence, in which case,

$$\frac{\Delta p_a}{q_c'} = f\left[\alpha, \frac{q_c'}{p_s}, \delta_\eta\right]. \quad (3.7)$$

In Expressions (3.5) to (3.7) q_c'/p_s may be replaced by the equivalent steady-flight Mach number, M' , (see Section 6.2.4). In the analysis of flight conditions that are steady, $M' = \bar{M}_i$, but for non-steady conditions there are corrections to be applied to $\bar{p}_{p_i}$ and $\bar{p}_{s_i}$ to arrive at the value of $q_c' (= (p_p' - p_s))$ and p_s (see Sections 6.1.3 and 6.2.3).

For those installations which employ a static probe or a pitot-static probe it may be better to replace α in Equations (3.5) to (3.7) by ω_p . (For a nose-probe installation at supersonic speeds, as implied by Section 3.1.2, $\omega_p = \alpha_p = (\alpha + \theta_p)$ for the usual condition of $\beta = 0$.)

The condition that Reynolds number is also a parameter is usually restricted to pitot-static probes at supersonic speeds where boundary-layer transition has moved forward of the static ports and the boundary layer has become turbulent. In the turbulent flow conditions that are present on such probes, the static-pressure-error correction is sensitive not only to Reynolds number but markedly also to α_p . The Example of Section 11.1 deals with the calibration of such a nose probe.

It may be noted that ΔC_{p_a} is the coefficient form of Δp_a and these quantities are related by

$$\Delta C_{p_a} = \frac{\Delta p_a}{0.7 p_s (M')^2}. \quad (3.8)$$

One of the complexities in the analysis of non-linear flight-path conditions referred to in the Introduction is the change incurred in the functional relationships presented in Equations (3.5) to (3.7). The relationships assume a unique correspondence between the flow condition of the static ports and the angle of attack of the aircraft. This relationship is altered by flight-path curvature and by airframe bending due to inertial forces. However, for a nose-boom pitot-static installation, downwash at supersonic speeds is zero so that the problem is reduced to considering only the effects of flight-path curvature and airframe-plus-boom bending on probe angle of attack and thus may be tractable.

* The static-pressure-error correction for some static probes shows a variation with Reynolds number for supersonic flight conditions. In Appendix A it is suggested that, for values of total temperature above 400 K ($M > 2$, approx.), Reynolds number might be replaced as a variable by T_r (see Equation (A3.1)).

3.2 Calibrated and Computed Data for Pressure Systems in Non-Steady Flight Conditions

These data apply when pitot or static-system pressures are time dependent.

3.2.1 Pressure-lag time constants

If the performance in climb or descent is high it may be necessary to carry out lag calibrations both in the hangar and in flight, see Item No. 85011 (Reference 18). The hangar calibrations provide the following time constants of pressure lag,

- (i) λ_p, λ_s , time constants due to friction in pitot and static systems where, for example, for a given pressure-system geometry and for small values of rate of change of pressure with time (see References 8 and 12),

$$\lambda_p \approx f[\bar{p}_{p_i}, \mu_p] \approx f[\bar{p}_{p_i}, T_p] \quad \text{and} \quad (3.9a)$$

$$\lambda_s \approx f[\bar{p}_{s_i}, \mu_s] \approx f[\bar{p}_{s_i}, T_s]. \quad (3.9b)$$

In many cases the time constant may be related to its value in ISA sea level conditions by the following simple relationship (see Section 8.4 for μ ; T_{p_0} taken as 288.15 K),

$$\frac{\lambda_p}{\lambda_{p_0}} = 0.081 \cdot 48 \left[\frac{T_p^{1.5}}{T_p + 110.4} \right] \frac{p_{p_0}}{p_{p_i}}. \quad (3.10)$$

The expression for λ_s/λ_{s_0} is similar.

- (ii) τ_p, τ_s , time constants due to acoustic delay in pitot and static systems where, for a given system geometry,

$$\tau_p, \tau_s = f[T_p, T_s] \quad \text{or, in general,} \quad (3.11)$$

$$\tau_p/\tau_{p_0} = (T_{p_0}/T_p)^{0.5}, \quad (3.12)$$

$$\tau_s/\tau_{s_0} = (T_{s_0}/T_s)^{0.5}. \quad (3.13)$$

The temperatures T_p and T_s in the pressure systems depend on the following variables (see Section 5.2.3),

$$T_p, T_s = f[k_p, k_s, T_{a_i}, M_i]. \quad (3.14)$$

The geometric data required are as follows,

k_p , volumetric proportion of pitot system within pressure shell of aircraft,

k_s , volumetric proportion of static system within pressure shell of aircraft.

Where it is desirable and possible to carry out flight calibrations of lag, an overall lag constant is obtained made up of $(\lambda_p + \tau_p)$ for the pitot-pressure system and $(\lambda_s + \tau_s + \lambda_{int,s})$ for the static-pressure system.

The quantity $\lambda_{int,s}$ is a lag-like error which is significant for some static systems and is apparent whenever

static pressure is time dependent. If flight calibrations of pressure lag are not available, the error $\lambda_{int,s}$ may have to be estimated. The subject is dealt with in Reference 18, where it is shown that the error results from the interactions between the external airstream and flow entering or leaving the static-system ports (see also References 2 to 4). It is shown in Reference 18 that the estimation of $\lambda_{int,s}$ should be required only if the quantity $[\bar{M}_i(Vol)_s/A_{sp}]$ is likely to exceed 100 m, approximately, or if $(\lambda_{int,s}/\lambda_{s_0}) \geq 0.1$ (see Section 5.2.4, item (ii)) in which case the following geometric data must be known:

A_{sp} , total area of the static-pressure-system ports,

$(Vol)_s$, volume of the static-pressure system.

It should be noted that if there are flight calibrations of pressure lag, any inadequacy of Mach number coverage may be made good by using the method of Reference 18 to determine trends.

3.2.2 Inertial considerations

For a limited class of aircraft where the performance is very high (typically thrust/weight exceeds unity) the longitudinal accelerations for some flight conditions give rise to significant pressure changes in pressure systems due to inertial effects. These errors are derived in Reference 18 and take the form,

$$\frac{(\delta p_p)_{in} T_p}{p_{p_i} l_{x_p}} = \frac{(\delta p_s)_{in} T_s}{p_{s_i} l_{x_s}} = f[f_x, \Theta], \quad (3.15)$$

with $f(f_x, \Theta)$ being approximated using speed and temperature data where accelerometer and attitude information are not available, see Section 5.2.5.

The geometric data required are as follows:

l_{x_p} , effective length of pitot-pressure system,

l_{x_s} , effective length of static-pressure system,

k_p , volumetric proportion of pitot system within pressure shell of aircraft,

k_s , volumetric proportion of static system within pressure shell of aircraft,

θ_{fus} , setting of fuselage datum relative to angle of attack datum.

3.3 Calibrated and Computed Data for a Total Temperature Probe

It is assumed in this Item that the basic data for the probe are those provided by the manufacturer and that installation effects, which are a function of the Mach number, M_L , local to the probe, may be estimated.

However, if reliable flight calibrations are available, they should be used. Reference should be made to Data Item Reference 19 which is concerned with the methods of handling the basic data for these probes.

3.3.1 Recovery factor, r

The recovery factor, r , is defined by the relationship

$$T_a = T_r / (1 + 0.2rM^2) \quad (3.16)$$

for conditions where the probe experiences free stream Mach number. In general this is not the case and corrections, dealt with in Sections 5.2.1 and 8.1, are applied (in Sections 5.2.2 and 8.2). For supersonic flow conditions further allowances have to be made because the probe is operating in the flow downstream of a normal shock.

3.3.2 Radiation error, $(\delta T_r)_{rad}$

This is negligible at subsonic and moderate supersonic speeds for probes fitted with radiation shields, but as the total temperature increases with speed or as the air density reduces with increasing altitude the error may become appreciable (see manufacturer's data). The error is a function of p_p/p_a and p_a .

3.3.3 Self-heating error, $(\delta T_r)_{sh}$

The self heating considered here is that heat generated in the element of a resistance thermometer by the current within the measuring circuit. The error is a function of ambient pressure and Mach number. It is also proportional to the power passing through the element. The power rating of the circuit in milliwatts is dependent on the installation and the instrumentation. The assumption is made in the introduction that by judicious flight planning the de-icing heat error of probes fitted with ice protection is not relevant. Such heat errors can be large and the use of anti-icing heat is to be avoided during flight tests where possible.

3.3.4 Correction for caloric imperfection*, f_γ

At Mach numbers in excess of $M = 2$ the total air temperature, T_{t_2} , is greater than 400 K and caloric imperfection* in the airflow sensed by the total temperature probe results in values of $\gamma < 1.4$. It is convenient to provide the correct value of γ in deriving ambient temperature, T_a , in Equations (5.13) and (8.12) by using the factor $f_\gamma = (\gamma - 1)/0.4$. This factor departs significantly from $f_\gamma = 1.0$ only for values of $T_{t_2} > 400$ K and otherwise may be taken as unity. As discussed and illustrated in Appendix A, this factor is a function mainly of T_{t_2} but is weakly dependent on T_a , with the implication that an iterative process may be required when the value guessed for T_a in the above equations is not near to that derived.

Typical values of f_γ and M (for $T_a = 220$ K), and of the errors in ambient temperature due to the use of $\gamma = 1.4$ are as follows.

T_{t_2}	f_γ	M	$\Delta T_a/T_a$ per cent	ΔT_a
600 K	0.987	2.96	-0.84	-1.85
800 K	0.969	3.69	-2.39	-5.03
1000 K	0.949	4.32	-4.01	-8.81

3.3.5 Local pressure coefficient, C_{p_L}

A value of local surface pressure coefficient is required for the position on the airframe of the total temperature probe when the local Mach number, M_L , is to be computed. The coefficient, C_{p_L} , may be obtained from wind-tunnel tests or be estimated. In general it is a function of Mach number, angle of attack and sideslip.

* This subject is dealt with in Appendix A. Although unimportant for stagnation temperatures up to 400 K, it is significant for higher values. Such effects are not important for pitot-pressure sensors until speeds beyond those considered in this Item are reached. Caloric imperfection arises when the value of γ is no longer independent of temperature and pressure, although the air may be considered as thermally perfect, i.e. $p = \rho RT$.

3.3.6 Temperature-lag time constant, λ_T

When T_p is time dependent, the variation with time is almost always linear, *i.e.* a ramp change. The probe response may be allowed for by computing the error $[\lambda_T(dT_p/dt)]$, where λ_T is the probe time constant which may be a flight-calibration value or, more probably, be based on manufacturers' data.

Where the data from the manufacturer are given for a step change in temperature, the value of λ_T for a ramp change has to be computed. This computation is straightforward for a probe for which the response equation consists of a single term, but in the more usual case there are two terms – the “fast-response” of the element itself which is affected by the “slow-response” of the element housing and the surrounding radiation shields. Means of handling the required computations are given in Data Item, Reference 19.

3.4 Flow-Direction Probe Data

These may be vanes or other devices mounted on a pitot-static nose boom or they may be positioned on the forward fuselage. Angle of attack and sideslip vanes mounted on a nose boom are usually arranged so that there is no appreciable coupling between angles of attack and sideslip. In such cases:

$$\alpha = f\left[\text{vane deflection}, \frac{d\epsilon_p}{d\alpha}, \theta_p, M\right], \quad (3.17)$$

$$\beta = f[\text{sideslip vane deflection}, \text{sidewash ratio}, M]. \quad (3.18)$$

Note that θ_p may vary significantly with the normal acceleration experienced by the aircraft. In manoeuvring flight, additional corrections have to be made for any non-linearity of the flight path (see Section 4.4.1 of Reference 11 and Section 3.1.3 of the present Item). Where vanes are mounted on the fuselage there are additional terms in the calibrations such as β entering into the functional relationship for α . Problems of calibration are not considered further in this Data Item and the values of angle of attack and sideslip are taken as corrected values.

4. FLIGHT DATA REQUIRED FOR THE CORRECTION PROCESS

The basic flight information is usually sampled digitally by the flight-test instrumentation system at time intervals suited to the flight condition. This information is then pre-processed using ground or laboratory transducer calibrations to give the data in the engineering units used in this method. In some cases visual or photographic records of instrumentation such as altimeter and airspeed indicator may be used and in these instances the laboratory calibrations that are applied are the instrument-error corrections.

In a test in steady level flight, only one value of each of the sampled quantities might seem to be necessary. In practice, for high accuracy, samples should be taken every 10–20 seconds for 1 or 2 minutes so that any effects of small motions of the aircraft and/or random errors can be removed by averaging. In manoeuvring flight samples are taken at shorter intervals. For example, if ranges of Mach number and/or angle of attack are being explored, at least five samples should be obtained within each of a number of time zones in which the values of Mach number or angle of attack are close to those that are significant for analysis.

4.1 Sampled Quantities

- (i) $(p_{p_i} - p_{s_i})$, pitot-minus-static pressures registered (indicated) by a differential pressure transducer
 or p_{p_i} , absolute pressure in pitot system registered (indicated) by a pressure transducer (note that for steady-level-flight cases, where no corrections for non-steady effects are required, $p_{p_i} = p_p'$).
- (ii) p_{s_i} , absolute pressure in static system registered (indicated) by a pressure transducer (note, in steady level flight, $p_{s_i} = p_s$).
- (iii) T_{r_i} , temperature registered by a total temperature probe (recovery temperature).
- (iv) α, β , angles of attack and sideslip from flow-direction probe, where fitted, and where the estimation of local Mach number at total temperature probe is required, see Section 5.2.1. In some cases, incidence angles may be estimated with a sufficient accuracy to avoid the need for flow-direction probes. For many applications sideslip angle may be taken as zero.
- (v) δ_η , deflection of control surface for pitch. This may be required for position-error corrections, depending on the aircraft geometry and the sophistication of the position-error calibrations.
- (vi) f_x , accelerometer reading in direction of fuselage longitudinal datum, if available. This is desirable where inertial effects are likely to be significant, see Section 5.2.5.
- (vii) Θ , inclination angle, *i.e.* pitch attitude of fuselage datum to horizontal. This is needed where an accelerometer is available (see (vi) above) and required in the analysis. It is desirable to obtain pitch attitude directly from a stabilised platform but if this is not available Θ may be approximated (see Section 5.2.5).

4.2 Use of Flight Instruments to Obtain Pressures

Where visually read or photographically recorded flight instruments only are available, they may be used as follows.

- (i) $(p_{p_i} - p_{s_i})$ may be obtained from an airspeed indicator, giving V_i , as

$$(p_{p_i} - p_{s_i}) = p_{a_0} \left[\left\{ \frac{1}{5} \left(\frac{V_i}{a_0} \right)^2 + 1 \right\}^{3.5} - 1 \right] \text{ for } V_i/a_0 \leq 1.0 \quad (4.1)$$

or

$$(p_{p_i} - p_{s_i}) = p_{a_0} \left[\frac{166.921 \ 58 (V_i/a_0)^2}{\{7 - (a_0/V_i)^2\}^{2.5}} - 1 \right] \text{ for } V_i/a_0 > 1.0. \quad (4.2)$$