In-Flight Pressure Distribution Measurements; instrumentation, data handling and comparison with windtunnel data

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Summary

As part of the certification of the Fokker 100 aircraft, in-flight wing pressure distributions were measured.

The objective of the analysis presented in the present paper is to make a comparison of flight test data with windtunnel data.

To this end an accurate calibration was made of the in-flight angle of attack, providing the appropriate reference conditions used for the comparison with windtunnel data.

Corrections were applied for differences in wing shape, due to differences in flight test and windtunnel conditions as well as due to production tolerances. This way the validity of the test methods and available extrapolation techniques was investigated. The results of this investigation confirm that for the type of wing section used in the Fokker 100, full-scale characteristics of the wing can be predicted from windtunnel model experiments.

Nomenclature

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
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<tbody>
<tr>
<td>a</td>
<td>speed of sound</td>
</tr>
<tr>
<td>b</td>
<td>span of main wing</td>
</tr>
<tr>
<td>c</td>
<td>local chord</td>
</tr>
<tr>
<td>c_1</td>
<td>local lift coefficient</td>
</tr>
<tr>
<td>c_1o</td>
<td>lift curve slope</td>
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<tr>
<td>c_m</td>
<td>local pitch moment coefficient about x_w=0.25c</td>
</tr>
<tr>
<td>c_M</td>
<td>Mach number derivative</td>
</tr>
<tr>
<td>c_p</td>
<td>(p-p_0)/q_∞</td>
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<tr>
<td>c_qv</td>
<td>pitch rate derivative</td>
</tr>
<tr>
<td>c_r</td>
<td>calibration correction for an offset in sideslip</td>
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<tr>
<td>c_rB</td>
<td>calibration correction for sideslip influence</td>
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<tr>
<td>c_yB</td>
<td>sideslip derivative of the dimensionless sideslip</td>
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<tr>
<td>c_α</td>
<td>basic coefficient</td>
</tr>
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<td>c_α</td>
<td>angle of attack derivative</td>
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<tr>
<td>g</td>
<td>acceleration due to gravity</td>
</tr>
<tr>
<td>h</td>
<td>altitude</td>
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<tr>
<td>m</td>
<td>mass of the aircraft</td>
</tr>
<tr>
<td>M</td>
<td>Mach number</td>
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<tr>
<td>n</td>
<td>normal load factor</td>
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<td>p</td>
<td>pressure</td>
</tr>
<tr>
<td>p_0</td>
<td>undisturbed static pressure</td>
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<tr>
<td>q_∞</td>
<td>dynamic pressure</td>
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<tr>
<td>p_∞</td>
<td>dp_∞/dt</td>
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<tr>
<td>q_∞</td>
<td>pitch rate</td>
</tr>
<tr>
<td>q_∞</td>
<td>Reynolds number based on mean aerodynamic chord</td>
</tr>
<tr>
<td>Re_c</td>
<td>area of the main wing</td>
</tr>
<tr>
<td>t</td>
<td>time</td>
</tr>
<tr>
<td>u</td>
<td>velocity component along x_body axes</td>
</tr>
<tr>
<td>v</td>
<td>velocity component along y_body axes</td>
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<td>V</td>
<td>true airspeed</td>
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<td>V_E</td>
<td>see figure 5</td>
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<td>see figure 5</td>
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<td>V_X</td>
<td>see figure 5</td>
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<tr>
<td>V_Y</td>
<td>see figure 5</td>
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<tr>
<td>w</td>
<td>velocity component along z_body axes</td>
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<tr>
<td>WT</td>
<td>windtunnel</td>
</tr>
<tr>
<td>x</td>
<td>x_body axes along fuselage centreline, positive pointing forward</td>
</tr>
<tr>
<td>x_w</td>
<td>chordwise position (x_w=0 is the leading edge)</td>
</tr>
<tr>
<td>y</td>
<td>y_body axes perpendicular to x and z axes</td>
</tr>
<tr>
<td>y</td>
<td>see figure 5</td>
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<tr>
<td>z</td>
<td>z_body axes in plane of symmetry perpendicular to x axes</td>
</tr>
<tr>
<td>a</td>
<td>angle of attack as determined from h calibrated angle of attack (relative to fuselage centreline)</td>
</tr>
<tr>
<td>a_1</td>
<td>angle of attack readings of the left fuselage vane</td>
</tr>
<tr>
<td>a_2</td>
<td>angle of attack readings of the right fuselage vane</td>
</tr>
<tr>
<td>g</td>
<td>angle of sideslip</td>
</tr>
<tr>
<td>Δα</td>
<td>α offset</td>
</tr>
<tr>
<td>Δβ</td>
<td>β offset</td>
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<tr>
<td>Δβ_o</td>
<td>bias due to sideslip</td>
</tr>
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<td>Δθ</td>
<td>wing deformation</td>
</tr>
<tr>
<td>ρ</td>
<td>density of air</td>
</tr>
<tr>
<td>V/b</td>
<td>pitch angle</td>
</tr>
<tr>
<td>δ</td>
<td>bank angle</td>
</tr>
<tr>
<td>θ</td>
<td>heading</td>
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</table>

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1. Introduction

As part of the certification of the Fokker 100 aircraft, it was decided that in-flight pressure distribution measurements were required on the main wing. The measurements were performed by Fokker in order to verify the wing loading and to compare flight test data with windtunnel data for design process validation purposes. Pressure distributions were measured at a number of chordwise sections of the main wing, using pressure belts. These belts consist of plastic tubes which were attached to the wing surface of the test aircraft. The pressure tubes at these measurement sections were connected to several scanivalves. The measurement data delivered by the scanivalves as well as data defining the flight condition were recorded on tape. Data processing was performed by the National Aerospace Laboratory (NLR). The main goal of the data reduction was to compute the aerodynamic coefficients $c_p$ as accurately as possible applying various corrections and interpolations.

In the past in-flight pressure distribution measurements were performed by many investigators (Refs. 1 through 7); usually correlation with windtunnel data was one of their main purposes. An interesting point, which has not been highlighted in literature in much detail up to now, is that verification of the results with windtunnel data is only possible if the relation between flight test condition and windtunnel condition is known at all cases of interest. Here the conditions cover the subsonic flight regime for level flight and the transonic flight region for a range of load factors. For the analysis presented here, the angle of attack was, located at the fuselage, was calibrated accurately in order to provide the true in-flight angle of attack to be used for comparison with windtunnel data. Also corrections are required for the difference in wing shape between model and aircraft due to deformation and production deviations.

2. Instrumentation

The in-flight wing pressure measurements were obtained from five rows of orifices on the upper and lower surfaces of the main wing (three on the right wing, two on the left wing) at the span stations shown in figure 1. The chordwise locations of the 30 flight wing pressure orifices in each row are presented in table 1. The chordwise extent of the pressure belt at the sections 1 and 2 was limited to 80 percent of the local chord on upper and lower surface because of the presence of the aileron. At the flap, present in the sections 3, 4 and 5, it was possible to take pressure measurements as long as the flaps were retracted (take-off with retracted flaps; the pressure tubes were disconnected by a deflection of the flap). The pressure belts made of plastic multibuks were taped to the wing surface (see Fig. 2) with the pressure orifices located in each of the multiple tubes. The 150 pressure holes were connected to four scanivalves (the cycle period of the scanivalve is 4.8 sec, number of ports is 48), each provided with one Druck differential transducer. For each scanivalve the pressure difference of two holes was also measured continuously by an accurate Rosemount differential transducer (see Fig. 3). From this simultaneous pressure measurement a scale factor has been obtained for the less accurate Druck transducer.

![Figure 1 - Measurement sections on the Fokker 100 main wing. (flight: test sections at right wing 2, 3, 4; test sections at left wing 1, 5; windtunnel model: test sections I through VIII.)](image)

<table>
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<th>3</th>
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</tbody>
</table>
3. Test programme

The pressure distribution measurements were taken in stabilized flight at various flight levels, speeds, normal load factors (by varying the bank angle) and flap settings. During flight the scan valves were stepping continuously to produce about ten cycles of readings in each run. Only 21 runs (clean configuration, different conditions) from a much larger amount of measurement runs were used in the actual analysis.

These runs can be split up into two groups: (1) subsonic flight regime (at one flight level with load factor 1 and varying speeds); (2) transonic flight regime (at one flight level with varying load factors between 1 and 2 and varying Mach numbers; see table 3).

| TABLE 3 Measurement runs used for the validation analysis |
|-------------|-----------|------------|----------------|----------------|
| run number | flight level | velocity/ Mach number | approximate normal load factor n | nominal bank angle deg |
| 33 | 24000 | M=0.7 | 1.0 | 0 |
| 35 | 24000 | M=0.7 | 1.5 | -68 |
| 39 | 24000 | M=0.7 | 2.0 | -60 |
| 41 | 24000 | M=0.72 | 1.0 | 0 |
| 47 | 24000 | M=0.72 | 1.5 | 48 |
| 49 | 24000 | M=0.72 | 2.0 | 60 |
| 51 | 24000 | M=0.75 | 1.0 | 0 |
| 53 | 24000 | M=0.75 | 1.5 | -48 |
| 55 | 24000 | M=0.75 | 2.0 | -60 |
| 57 | 24000 | M=0.78 | 1.0 | 0 |
| 59 | 24000 | M=0.78 | 1.5 | 48 |
| 61 | 26000 | M=0.78 | 2.0 | 60 |
| 81 | 10000 | 200KCAS | 1.0 | 0 |
| 83 | 10000 | 190KCAS | 1.0 | 0 |
| 85 | 10000 | 180KCAS | 1.0 | 0 |
| 87 | 10000 | 170KCAS | 1.0 | 0 |
| 89 | 10000 | 160KCAS | 1.0 | 0 |
| 91 | 10000 | 150KCAS | 1.0 | 0 |
| 93 | 10000 | 139KCAS | 1.0 | 0 |
| 95 | 10000 | 128KCAS | 1.0 | 0 |
| 97 | 10000 | 118KCAS | 1.0 | 0 |

After calibration the accuracies of the measured parameters were as listed in Table 2.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Name</th>
<th>Unit</th>
<th>Range</th>
<th>Accur.</th>
<th>Samp.fr.</th>
<th>Symbol</th>
<th>Remarks</th>
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<tr>
<td>angle of attack left</td>
<td>AAV1</td>
<td>deg</td>
<td>-20/60</td>
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<td>angle of attack right</td>
<td>AAV2</td>
<td>deg</td>
<td>-20/60</td>
<td>0.01</td>
<td>16</td>
<td></td>
<td></td>
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<tr>
<td>angle of sideslip</td>
<td>ASV</td>
<td>deg</td>
<td>-40/40</td>
<td>0.02</td>
<td>0.02</td>
<td>16</td>
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<td>kPa</td>
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<td>0.022</td>
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<td>deg</td>
<td>-180/180</td>
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<td>Y acceleration</td>
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<td>0.005</td>
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<td>0</td>
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</tbody>
</table>

1498
4. Data reduction

The flight test data were read from the flight test tape by the data reduction station DVSV, calibrated and transferred to the NLR Cyber 180-855 mainframe computer. Processing of the in-flight pressure data was performed by the program PRESS to compute the aerodynamic coefficients $c_p$.

This program makes use of the NPP (NLR Post Processing) database software package (see Fig. 4).

![Data processing scheme diagram]

FIGURE 4 - Data processing scheme

Although much attention was paid to keep the flow condition constant during the flight tests, deviations from the initial condition of the recordings are unavoidable. This fact was accounted for during the data processing. Pressure distributions in the form of 'snap shots' were computed for each scanivalve cycle. The main functions of the data reduction program PRESS are to provide:
- linear time interpolations of the measured time series,
- corrections for time delays in all the pressure tubes (time delays are less than 1 sec),
- application of position error corrections to the reference static pressure hole and to the total pressure probe,
- computations of the momentaneous aerodynamic coefficients $c_p$,
- quick-look monitoring of momentaneous pressure distributions of each scanivalve cycle.

5. Calibration of the angle of attack vanes at the fuselage

Accurate calibration of the in-flight angle of attack (especially at the transonic flight regime and for normal load factors up to 2) is of major importance for the validation and interpretation of the pressure distribution measurement results. It was decided to use the calibrated angle of attack vane readings instead of flight path reconstruction (estimation) for validation purposes. The reason for this is that the airload is a function of the aerodynamic angle of attack while the flight path may deviate due to the wind field being of a rather complicated behaviour especially at conditions during flight turn manoeuvres (normal load factor larger than 1). The determination of the angle of attack calibration equation as described in the present paper requires the combination of runs with varying bank angle (left and right turning manoeuvres as well as level flight) and varying Mach numbers. The redundant information of these combined runs was used to estimate the calibration equation accurately by regression analysis. This calibration procedure will be described in three steps:
- computation of a theoretical angle of attack from the pressure altitude variation and the attitude of the aircraft during flight turning manoeuvres,
- estimation of the angle of attack calibration equation by means of regression analysis,
- application of the calibration equation to the measured data.

5.1 Computation of a theoretical angle of attack from the pressure altitude variation

This computation can be split up in four steps:
- computation of the pressure altitude variation,
- determination of the sideslip angle,
- extraction of the angle of attack from the pressure altitude variation,
- determination of the offset in the angle of attack due to sideslip uncertainties.

- Computation of the pressure altitude variation.
The pressure altitude variation can be computed from:

$$ h = -\frac{\dot{p}_m}{g \rho} $$

(1)

- Determination of the sideslip angle
The computation of the angle of attack from the pressure altitude variation during a turning manoeuvre requires the availability of sideslip data. The sideslip angle was computed (estimated) using the body acceleration $\dot{g}$ from:

$$ g = \frac{m \dot{A}_y}{\frac{1}{2} \rho V^2 S c_{\gamma b}} $$

(2)
where $C_{yB}$ is the side force derivative coefficient with respect to the sideslip determined during earlier flight tests. Note that the measurement runs were flown close or equal to zero degrees slip angle.

- Extraction of the angle of attack from the pressure altitude variation. The analysis starts from the well-known transformation equations:

$$u = V_N \cos \theta \cos \psi + V_E \cos \theta \sin \psi + h \sin \theta \tag{3}$$
$$v = V_N (\sin \theta \sin \psi \cos \theta - \cos \psi \sin \theta) + V_E (\sin \theta \sin \psi \sin \theta \cos \psi + \cos \theta \cos \psi \sin \theta) - h \sin \theta \cos \theta \tag{4}$$
$$w = V_N (\cos \theta \cos \psi \sin \psi + \sin \psi \sin \theta) + V_E (\cos \theta \cos \psi \sin \psi - \sin \psi \sin \theta) \tag{5}$$

where $u,v,w$ are the velocity components along the body axes and $V_N$ (north), $V_E$ (east), $h$ (vertical) are the velocity components along an earth fixed coordinate system (fig. 5). Here the velocity vector $(u,v,w)$ as well as the velocity vector $(V_N, V_E, -h)$ are defined as the momentaneous velocity of the aircraft relative to the surrounding air.

![Figure 5](image)

**FIGURE 5 - Applied coordinate system and sign conventions**

For further analysis the next transformation is essential (see fig. 5):

$$V_X = V_E \sin \psi + V_N \cos \psi \tag{6}$$
$$V_Y = V_E \cos \psi - V_N \sin \psi \tag{7}$$

where $V_X$ is directed along the projection of the body axes onto the horizontal plane. Substitution of (6) and (7) into (3), (4) and (5) yields:

$$u = V_X \cos \theta \cos \psi$$

$$v = V_X \sin \theta \sin \psi \cos \theta + V_E \sin \theta \sin \psi \sin \theta \cos \psi + h \sin \theta \cos \theta \tag{9}$$
$$w = V_X \cos \theta \sin \psi - V_E \sin \theta \sin \theta \hcos \theta \cos \psi \tag{10}$$

Substitution of $v = V \sin \theta$ into (9) results in:

$$V_Y = \frac{\sin \theta \cos \psi - V_X \sin \theta \sin \theta + V \sin \theta}{\cos \theta} \tag{11}$$

From

$$v^2 = V_X^2 + V_Y^2 + h^2 \tag{12}$$

it follows that:

$$V_X = \sqrt{v^2 - V_Y^2 - h^2} \tag{13}$$

Equations (11) and (13) form a couple of nonlinear equations which can be solved iteratively starting from equation (11) with $V_X$ (initial) $= V$. The angle of attack can be computed by substitution of $w = V \cos \theta \sin \theta$ into (10):

$$\alpha = \sin^{-1} \frac{V \cos \theta \sin \theta - V_Y \sin \theta - h \cos \theta \cos \theta}{V \cos \theta} \tag{14}$$

- Influence of the sideslip uncertainties on the angle of attack.

A first order simplification of equation (11) yields:

$$\Delta V_Y = \frac{V \Delta \theta}{\cos \theta} \tag{15}$$

and equation (14) can be simplified to:

$$\Delta \alpha = -\frac{\Delta V_Y \sin \theta}{V} \tag{16}$$

Combining (15) and (16) results in:

$$\Delta \alpha = -\Delta \theta \tan \theta \tag{17}$$

Thanks to the fact that left and right turning manoeuvres are combined before regression the $\Delta \theta$ offset can be estimated very well as will be shown later ($\Delta \theta$ is estimated to be 0.1 deg).

5.2 Estimation of the angle of attack calibration equation by means of regression

Two steps are distinguished:
- to set up the regression equation,
- analysis of the regression equation by deleting non-significant terms.
- The following regression equation was used in the analysis:

\[ \alpha = c_o + \frac{1}{2} c_\alpha (a_1 + a_2) + c_M + 14c_v \alpha q / V + c_R \tan \theta \]

+ \( c_{RB} \alpha \tan \theta \)

(18)

where the following constants have to be estimated:

- \( c_o \) = \( \alpha \_O \) (offset)
- \( c_\alpha \) = \( \frac{da}{d\alpha} \) (vane readings) (calibration factor)
- \( c_M \) = \( \frac{da}{dM} \) (Mach number effect)
- \( c_v \) = \( \frac{da}{dV} \) (pitch rate effect due to an armlength of 14m)
- \( c_R \) = \( \frac{d\Delta \alpha}{d\alpha} \) (\( \Delta \alpha \) offset)
- \( c_{RB} \) = \( \frac{d\Delta \alpha}{d\alpha} \) (additional calibration on \( \Delta \alpha \))

Note that the left hand side parameter \( \alpha \) (eq. 18) is hardly influenced by the windvector (small random deviations only) because of the fact that (1) its value is computed from the pressure altitude variation and (2) the direction of flight (and as a consequence the direction of the wind relative to the aircraft) was varying over the manoeuvres to be combined.

- Analysis of the regression equation

Equation (18) has been analysed for two separate cases:
- calibration equation for subsonic conditions (M<0.4),
- calibration equation for transonic conditions (0.7<M<0.8).

- Calibration equation for subsonic conditions

The regression analysis is based on nine combined runs (table 3) taken at subsonic speeds and level flight. At these conditions the influence of sideslip on the angle of attack calibration is supposed to be negligible and can be omitted in the regression equation (18).

From regression analysis (see table 4) it follows that the standard error of the estimated calibration factor \( c_o \) is \( \sigma = 0.7 \). In addition it became clear that the influence of the Mach number \( c_M \) and the influence of the pitch rate \( c_v \) may not be neglected.

The resulting calibrated angle of attack \( \alpha_c \) and the originally computed angle of attack \( \alpha \) are both plotted in figure 6.

- Calibration equation for transonic conditions

The regression analysis is based on 12 combined runs (table 3) taken at transonic speeds and one flight level with varying load factors between 1 and 2 and varying speeds: W=7, 72, 75, 78.

A first analysis did show that the term \( c_{RB} \) is of a non-significant value and can be omitted. From a physical point of view this means that the adopted sideslip angle \( \beta(\alpha) \) does not need further calibration to improve the accuracy of the estimated calibration equation significantly.

From regression analysis (see table 4) it follows that the standard error of the estimated calibration factor \( c_o \) is \( \sigma = 0.3 \).

The resulting calibrated angle of attack \( \alpha_c \) and the originally computed angle of attack \( \alpha \) are both plotted in figure 7 at Mach number 0.72 and increasing normal loads: \( n=1.0,1.5,2.0 \).

![Figure 6 - The computed angle of attack \( \alpha \) and the calibrated angle of attack \( \alpha_c \)](image)

**TABLE 4** Calibration equation as result of regression analysis

<table>
<thead>
<tr>
<th>Mach number range:</th>
<th>( c_o )</th>
<th>( c_\alpha )</th>
<th>( c_M )</th>
<th>( c_v )</th>
<th>( c_R )</th>
<th>( c_{RB} )</th>
<th>total correlation</th>
</tr>
</thead>
<tbody>
<tr>
<td>( M&lt;0.4 )</td>
<td>0.0603</td>
<td>0.5671</td>
<td>0.0109</td>
<td>1.0000</td>
<td>0.0000</td>
<td>0.0000</td>
<td>0.99994</td>
</tr>
<tr>
<td>( 0.7&lt;M&lt;0.8 )</td>
<td>0.0000</td>
<td>0.0199</td>
<td>0.0737</td>
<td>0.0000</td>
<td>0.0000</td>
<td>0.0000</td>
<td>0.98676</td>
</tr>
</tbody>
</table>

The coefficients marked by * were fixed (not estimated).
The angle of attack $\alpha$ computed from the pressure altitude variation together with the calibrated angle of attack $\alpha_c$ is shown in the figures 6 and 7.

For further analysis of the pressure distribution measurements the calibrated angle of attack $\alpha_c$ will be used instead of $\alpha$ because $\alpha$ is corrupted with much more noise than $\alpha_c$. Note that $\alpha_c$ is the momentaneous angle of attack of the aircraft measured relative to the surrounding air, and is therefore the basic parameter on which the pressure distribution (airload) depends.

The final accuracies are listed in table 5.

### TABLE 5 Estimated accuracies of the results

<table>
<thead>
<tr>
<th>parameter</th>
<th>maximum unaccuracy</th>
<th>remarks</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\alpha_c$</td>
<td>0.05 deg</td>
<td>due to the accuracy of the IRS</td>
</tr>
<tr>
<td>$c_1$</td>
<td>2%</td>
<td>observed spread in the final results (the pressure belts do not affect $c_1$ significantly since the effect at the lower side cancel out against the upper side during the integration)</td>
</tr>
<tr>
<td>$c_{10}$</td>
<td>2%</td>
<td>almost identical to the accuracy of $c_1$</td>
</tr>
<tr>
<td>$M$</td>
<td>0.001</td>
<td>as a result of $p_1$, $p_8$ and PEC unaccuracies</td>
</tr>
<tr>
<td>$\Delta \theta$</td>
<td>0.1 deg</td>
<td>due to uncertainty in the stiffness data</td>
</tr>
</tbody>
</table>

6. Comparison of flight data with wind tunnel data

The flight test data were compared to HST (High Speed Tunnel at NLR) wind tunnel test data taken using a 1:20 scale model (the measurement sections are shown in Fig. 1). Since in-flight test conditions and measurement sections are generally different from those in the windtunnel, interpolations of windtunnel data to flight conditions or vice versa had to be made. The in-flight lift coefficients are derived by integration of the chordwise pressure distributions during each scanivalve cycle.

6.1 Comparison of local lift coefficients

Figure 8 presents a direct comparison between the midspan lift curves as obtained from flight tests and wind tunnel tests at transonic Mach numbers between $M=0.742$ and $M=0.749$. Because the Mach number varied during the flight tests, the two adjacent Mach numbers of the wind tunnel test results are shown.

![Figure 8 - Comparison of midspan lift curves](image)

No corrections applied for differences in wing shape.

A fair comparison can only be made when the difference in shape of the wing in the windtunnel with respect to the full-scale aircraft is taken into account for each condition.
The following factors concerning the differences in wing shape were taken into account:
1. Deviation of the angle of attack along the span of the main wing due to production tolerances. Deviations of the contour of the airfoil sections due to production tolerances have been neglected; they are small in comparison to the dimensions of the pressure belt.
2. Deformation due to flight conditions ($\alpha_\infty$, Mach, $\alpha_c$).
2.a Windtunnel model deformation.
2.b Aircraft deformation.
3. Deformation due to normal load condition (note: this deformation equals zero for the n=1g condition).
4. Deformation due to the presence of wing fuel.

Factors 2, 3 and 4 were determined using estimated stiffness data. These stiffness data of the aircraft were improved with results obtained from ground vibration tests.

Lifting line theory was used to process these differences in wing shape into a correction to the spanwise lift distribution. The two-dimensional section lift curve slope at a number of sections (extracted from the spanwise lift distribution based on windtunnel data) are input for the required lifting line computations to determine the correction on the spanwise lift distribution.

Figures 9 and 10 show comparisons of lift coefficients versus the angle of attack for the midspan section at a low speed and at a transonic condition.

Figure 10 - The lift coefficient at transonic condition (midspan) as a function of the angle of attack

The transition from laminar to turbulent boundary layer was fixed at 5-7% chord in the windtunnel tests. The Reynolds number (based on the mean aerodynamic chord) varies from about 3 million in the windtunnel to about 14 to 24 million for the low speed flight conditions and to about 32 million for the transonic flight conditions.

Figure 11 shows lift curves obtained from flight test data (corrected to windtunnel model shape as well as uncorrected) and from windtunnel data at three spanwise locations. The average Mach number during these flight tests was found to be $M=0.746$.

1503
The windtunnel data presented in this figure were obtained after linearization and interpolation between the eight available measurement sections and the two adjacent Mach numbers (Ma=.74 and Ma=.76). In figure 11
the observed change in slope of the section lift curves varies from about 13% at Ma=.206 to about 6% at
Ma=.873.

Using the same change of Reynolds number (based on m.a.c.) from 3 to 32 million the DATCOM (Ref. 8)
yields an increase of the slope of the section lift curves of about 6%. This value is valid for low Mach
number conditions. Windtunnel tests (HST at NLR) on a Fokker 100 halfmodel reveal that the Reynolds number
hardly affects the angle of attack for zero (tail-off) lift, Clto=0.0. Furthermore, these windtunnel tests
indicated an increase of the tail-off lift curve slope of about 5% as a result of a change in Reynolds number
from 3 to 8 million for the same Mach number. It is therefore considered that the observed differences in
lift curve slope as shown in figure 11 between flight (corrected to windtunnel shape) and windtunnel are
mainly caused by the Reynolds number effect.

6.2 Comparison of pressure distributions

Figure 12 shows a comparison of pressure distributions as obtained from windtunnel tests and flight
tests for a transonic flight condition (no corrections were applied for differences in wing shape). The
reduced aerodynamic loading in the tip region due to the deformation of the wing is most pronounced in the
suction peak at the lower surface. For this case the local difference in wing angle of incidence between
the windtunnel and flight tests is estimated to be Δδ=0.9 deg at Ma=.873.

\[ C_p = \frac{\text{Pressure difference}}{\text{Dynamic pressure}} \]

Figure 12 - Comparison of wing pressure distributions at three spanwise locations. No corrections
applied for differences in wing shape.
The (calibrated) angles of attack and the Mach numbers are approximately equal.

Figure 13 shows a comparison of flight test data with computed results of the computer program XPL022
(Ref. 9). The XPL022 computations were performed for the deformed wing while the midspan section lift
was matched. Excellent agreement was obtained between the computed results and the flight test data.

Comparison of windtunnel data with flight test data show similar trends as shown in figure 12 (e.g.
differences in the shock position, in the trailing-edge pressure and in the aftloading being typical Reynolds
number effects). It is concluded that the observed differences in pressure distributions between
flight and windtunnel data are mainly due to Reynolds number effects, since corrections for wing deformation
have been taken into account.

Figure 13 - Computed and measured wing pressures (wind tunnel and flight) The c₄ at midspan was
matched, copied from Ref. 9.

7. Conclusions

Accurate calibration of the flight angle of attack, which turned out to be dependent on the Mach number,
is of major importance for validation and interpretation of in-flight pressure distribution measurement
results. Regression analysis based on the pressure altitude variation enables the calibration of
the angle of attack wanes to be determined with an accuracy of 0.05 deg. After correction for differences
in wing shape reasonable agreement with windtunnel data was found. The remaining discrepancies between
lift curve slopes can largely be attributed to the Reynolds number effect. For the prediction of accurate aero-
dynamic wing loads both Reynolds number effect and wing deformation must be accounted for. The accuracy of
the local lift, derived from the in-flight measurement of the pressure distribution, is certainly better than 2% in
all cases.
Acknowledgements.

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<table>
<thead>
<tr>
<th>Reference</th>
<th>Authors</th>
<th>Title</th>
<th>Source</th>
</tr>
</thead>
</table>