



EXTRAPOLATING WIND-TUNNEL DATA TO FULL-SCALE REYNOLDS NUMBER Part 3: Example (i) Choice of simulation criteria and transition-strip locations for the F4 Wing/Body combination at the design condition

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Dr S. Prince — The City University, London

Dr S. Shaw — Airbus UK, Filton

Mr P.W.C. Wong — Aircraft Research Association Limited, Bedford

The construction and subsequent development of the Data Item were undertaken by

Dr D. Philpott — Senior Transonic Aerodynamics Engineer.

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1. INTRODUCTION

The general principles, by which aft-fixing of transition can be used in wind-tunnel testing to simulate flow characteristics* at a Reynolds number higher than the test value, are described in Reference 1. As part of that process it is essential to obtain a good physical understanding of important flow features, such as shock/boundary-layer interaction, likely flow separations *etc.*, at the test Reynolds number, full-scale Reynolds number and any other intermediate Reynolds numbers that are of interest. Prior to the test programme, this understanding is obtained by the use of CFD methods and past experience of similar configurations. It is also important to have a clear understanding of the test objective(s).

Following Reference 1, a simulation criterion is then selected that is closely linked to a dominant flow feature and the objective of the test. For example, where conditions at the shock are of particular significance, the transition band positions may be selected to match the shock position at the Reynolds number it is desired to simulate in the test. Alternatively, a boundary-layer parameter (*e.g.* momentum thickness) immediately upstream of the shock may be chosen. If upper-surface separation near the trailing edge is of particular significance, then the momentum thickness at the trailing edge is an appropriate parameter to choose. Once a suitable criterion has been chosen, forced transition locations (x_{tru} and x_{trl}) used in low-Reynolds-number tests can be directly mapped to a higher Reynolds number (Re_{eff}) simulated by the use of aft fixing. It should be noted that it is not possible to satisfy all possible criteria simultaneously and the degree to which the selected criterion is satisfied will be a compromise. For example, if shock position is the chosen criterion, care must be taken to ensure that accurate matching of this parameter does not result in a serious mismatch in the boundary-layer condition immediately upstream of the shock.

Because of these difficulties it is hard to provide a general “rule-of-thumb” procedure and the purpose of this document is to illustrate the procedure through the particular example of the F4 wing/body combination² at its design condition. It is shown how suitable simulation criteria may be selected and how upper-surface and lower-surface transition locations, suitable for a test at low Reynolds number, may be chosen. Although the framework outlined in Reference 1 is generally followed, other flow parameters are examined where it is thought these are useful in developing a sound understanding of the flow.

In this example the test objective is taken to be the estimation of drag at the design condition. A real test programme will cover a range of C_L , rather than being focussed on a single design point but the detailed analysis at the design point serves to illustrate the procedure that should be applied over the range to be tested. However, as outlined in Reference 1, at other values of C_L , the investigation may lead to the adoption of different simulation criteria and CFD extrapolation procedures.

* It is implicit that the simulation represents a flow where transition on both upper and lower surfaces occurs at or close to the leading edge, as is usually the case at full scale.

2. NOTATION

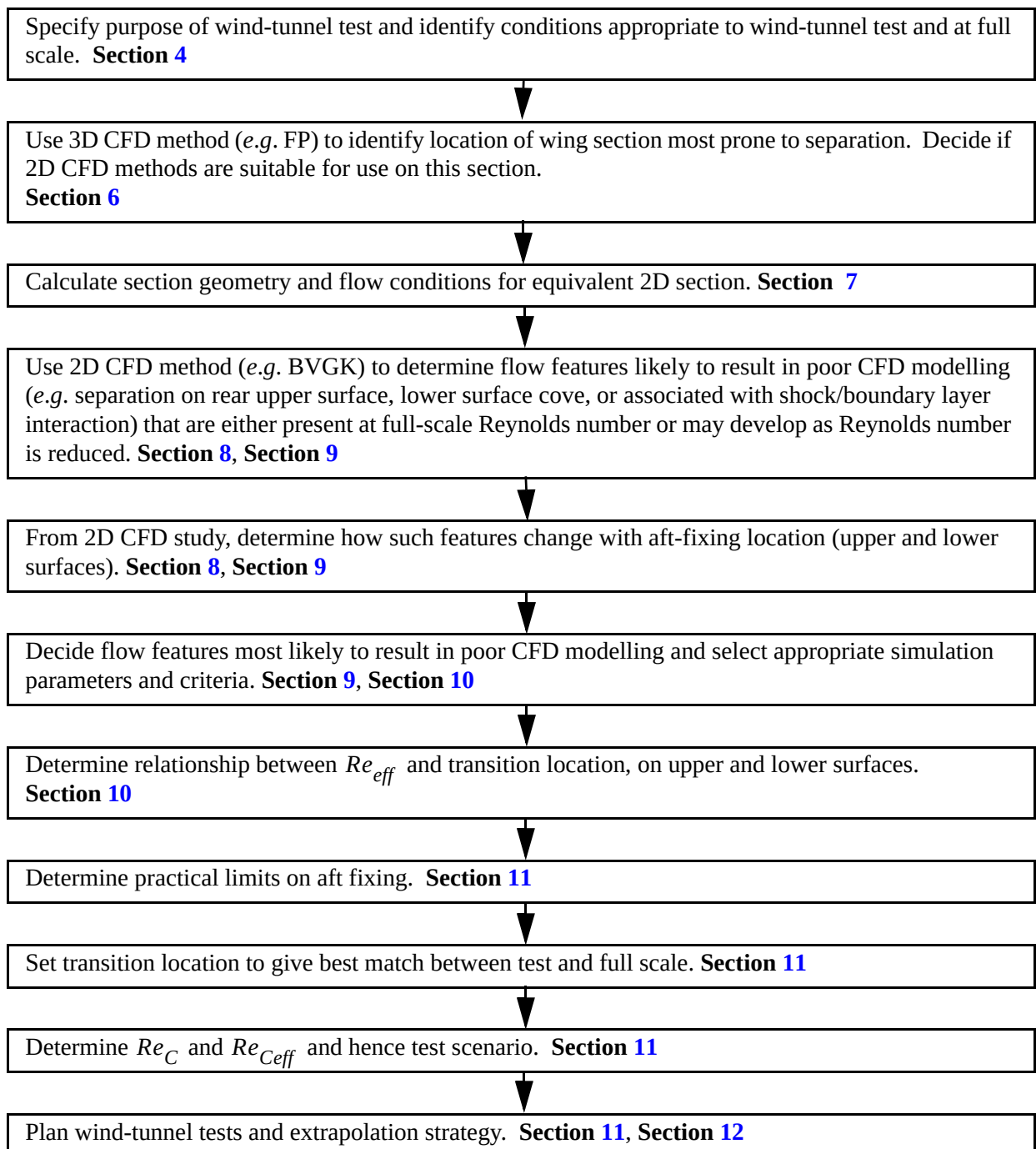
C_d	sectional drag coefficient for section normal to mid-chord sweep line
C_{dw}	wave drag coefficient (Section 9.1)
C_L	overall lift coefficient
C_l	sectional lift coefficient for section normal to mid-chord sweep line
C_{lLF}	sectional lift coefficient for line-of flight section
C_p	pressure coefficient
c	chord of local section in line-of-flight direction
c_0	chord of wing at centre line obtained by extrapolating leading and trailing edges
c_{2D}	chord of local section normal to mid-chord line
$\bar{c}$	mean chord
M_{LF}	M_∞ plus Mach number increment due to presence of body (Section 7.2)
M_{sh}	Mach number upstream of shock wave on aerofoil
M_{2D}	Mach number used for 2D calculation (Section 7.2)
M_∞	freestream Mach number
Re_{Beff}	maximum effective Reynolds number in aft-fixed test
Re_C	Reynolds number above which indirect Reynolds-number effects may be considered to be small
Re_{Ceff}	Re_C estimated using data from aft-fixed tests or CFD with transition specified at locations aft of full-scale location
Re_{eff}	effective Reynolds number of 2D section simulated by aft fixing section
Re_{LF}	Reynolds number based on local line-of flight approach velocity and chord
Re_{2D}	Reynolds number used for 2D calculation (Section 7.2.1)
Re_∞	Reynolds number based on freestream velocity and $\bar{c}$
s	wing semi-span
s^*	wing semi-span measured in wing reference plane (Sketch 4.1)
V_{LF}	local line-of-flight approach velocity

V_{2D}	local approach velocity normal to mid-chord line
V_{∞}	freestream velocity
x	coordinate measured along body axis (Sketch 4.1)
x^*	coordinate measured relative to the wing reference plane (Sketch 4.1)
x_{sep}	location of start of separation on upper surface
x_{sh}	location of shock wave (Section 9)
x_{trl}	transition location on lower surface
x_{tru}	transition location on upper surface
y	spanwise distance measured from centre of body (Sketch 4.1)
y^*	coordinate measured relative to the wing reference plane (Sketch 4.1)
z	vertical coordinate of (Sketch 4.1)
z^*	coordinate measured relative to the wing reference plane (Sketch 4.1)
z_{LF}	vertical coordinate of aerofoil section in line-of-flight direction (Section 7.1)
z_{2D}	vertical coordinate of 2D aerofoil section (Section 7.1)
α	incidence
Δ	modulus of change in variable over specified range
Δ_{max}	maximum value of Δ over specified range
$\Delta\delta^*$	difference in displacement thickness between upper and lower surfaces at trailing edge $\Delta\delta^* = (\delta_{teu}^* - \delta_{tel}^*)$
δ_{tel}^*	boundary layer displacement thickness on lower surface at trailing edge
δ_{teu}^*	boundary layer displacement thickness on upper surface at trailing edge
η	y/s
θ	momentum thickness of boundary layer
θ_{sh}	momentum thickness just upstream of shock wave
θ_{teu}	momentum thickness at trailing edge (upper-surface)
Λ	local sweep angle at given chordwise station
Λ_{le}	leading edge sweep angle
$\Lambda_{0.5}$	mid-chord sweep angle
ν	kinematic viscosity

3. PURPOSE OF EXAMPLE AND OUTLINE OF PROCEDURE

This example relates to a typical transonic transport aircraft configuration and a detailed analysis is presented for a single test condition at the nominal wing “design point”. Although the chosen configuration (the F4 wing/body combination) has been extensively tested in a number of wind tunnels, experimental evidence is deliberately ignored in this example in order to imitate the situation where tests are being planned for a new configuration. An analysis of available test results, following the methods outlined in Reference 1, will be addressed in a later document.

The basic strategy used is outlined in the following flow diagram and references to the appropriate sections in the text are included.



4. THE F4 WING/BODY AND TEST DETAILS

The F4 wing/body combination is chosen for this analysis because it has been widely investigated both experimentally², and using CFD³. The model geometry was also used as a test case for the European Transonic Wind-tunnel⁴. The configuration represents a typical high-aspect-ratio transonic transport aircraft and illustrates some problems that might be encountered in a typical test of a new design. Details of the model geometry are shown in Sketch 4.1.

4.1 Test Objectives

In this example the purpose of the test is taken to be the measurement of the aerodynamic characteristics of the wing/body combination under normal operating conditions close to cruise. The main requirement is, therefore, for reliable extrapolation of wind-tunnel measurements of parameters, such as lift and drag coefficient *etc.* to full-scale conditions.

4.2 Full-Scale and Wind-Tunnel Conditions

The design conditions for the F4 wing/body³ are:

$$M_{\infty} = 0.785$$

$$C_L = 0.5.$$

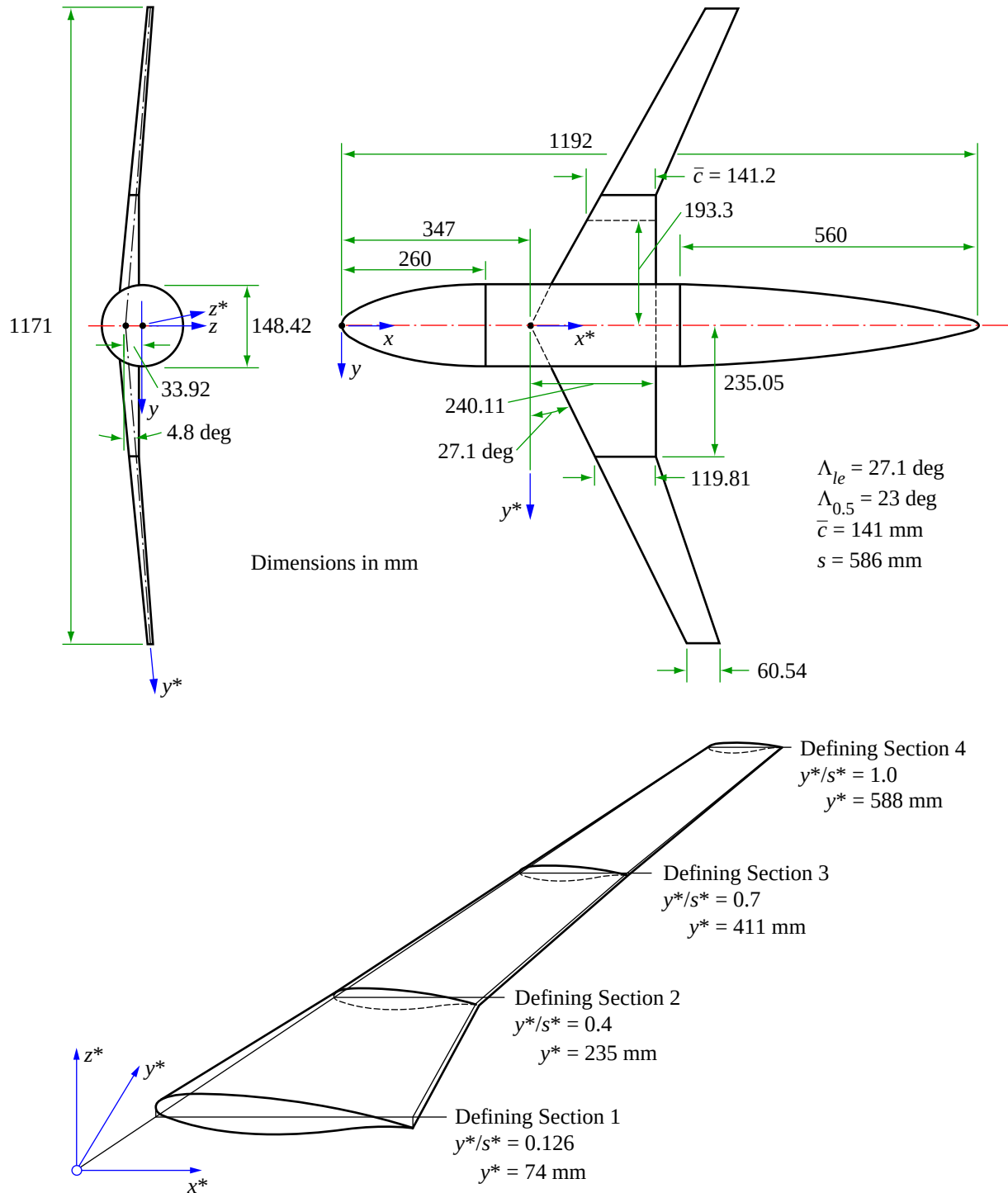
The Reynolds numbers chosen for analysis are:

$$\text{wind-tunnel Reynolds number } Re_{\infty} = 3.0 \times 10^6$$

$$\text{full-scale Reynolds number } Re_{\infty} = 33.0 \times 10^6.$$

The wind-tunnel Reynolds number considered here corresponds to a majority of available experimental data and the full-scale Reynolds number is typical for the class of aircraft represented by the model and also coincides with the high Reynolds number test cases described in Reference 4.

With the chosen test objective it is unlikely that there will be extensive areas of rear separation on the wing at full scale, or separation induced by a severe shock/boundary-layer interaction. However, such separations may occur under test conditions at low Reynolds number and the purpose of this example is to estimate the extent to which aft fixing can be used in the wind-tunnel test to remove such unrepresentative features.

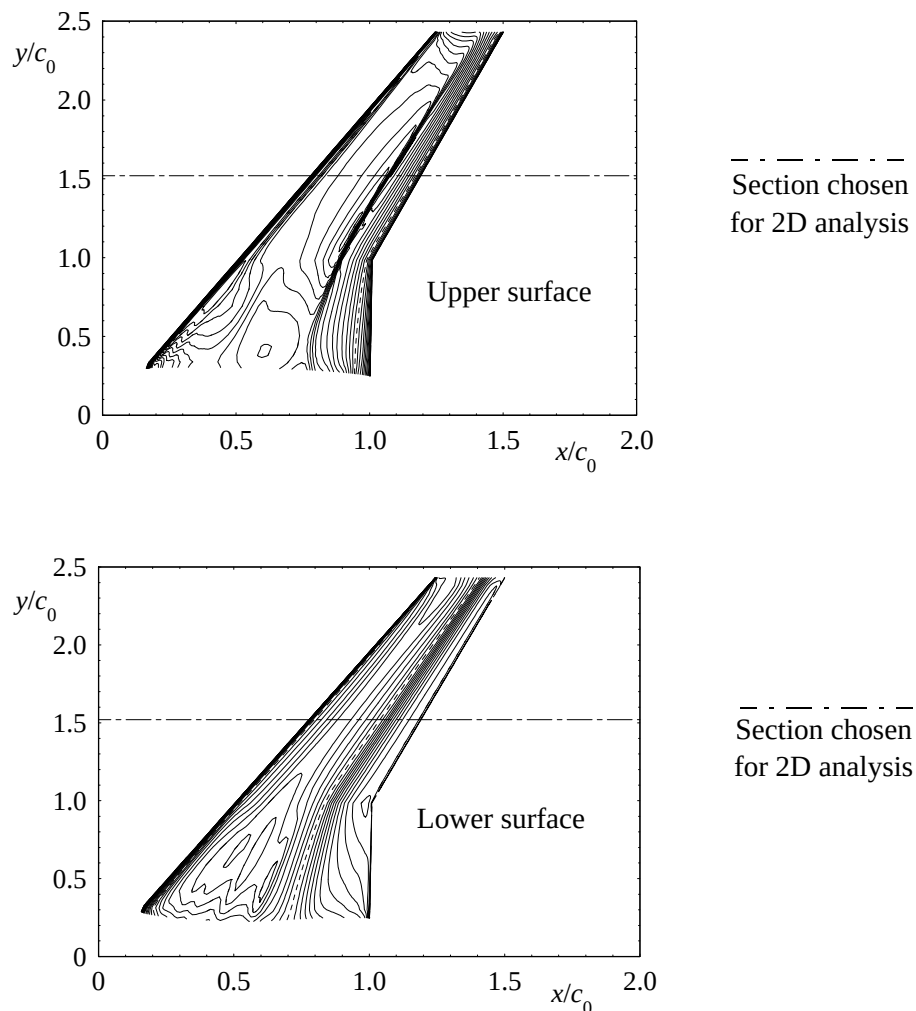


Sketch 4.1 DLR F4 Model geometry (from Reference 3)

5. CFD METHODS

The procedures outlined in Reference 1 demand repetitive CFD calculation, therefore it is appropriate to use a set of quick and economical CFD methods such as those adopted here. Initially an inviscid wing/body program (FP^{5,6}) is used to assess the main flow characteristics. From these the spanwise location at which Reynolds-number sensitivity is likely to be most severe is identified. The aerofoil program BVGK^{7,8} is then used to assess the flow over the equivalent two-dimensional section at this spanwise location and to produce a mapping of transition location position onto equivalent Reynolds number¹. BVGK is particularly suitable for this purpose as it is quick to run and allows the user to specify transition locations independently on the upper and lower surfaces of the aerofoil.

6. INITIAL INVESTIGATION USING FP

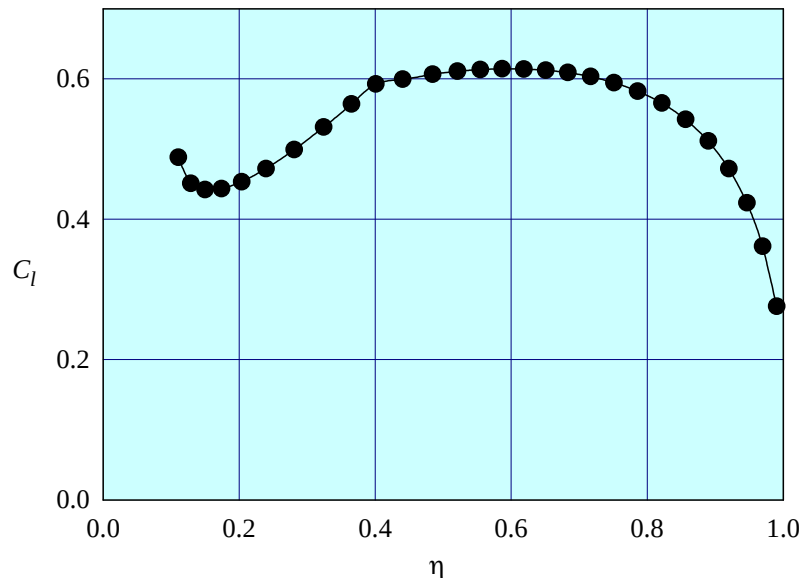


Sketch 6.1 Surface isobars (infinite fuselage, $M_\infty = 0.785$, $C_L = 0.5$)

In order to assess the likely general flow characteristics at high Reynolds number, inviscid calculations at the design condition ($M_\infty = 0.785$, $C_L = 0.5$) are made using FPIBE⁶, an improved version of the FP program. In these calculations the fuselage is represented by an infinite cylinder of appropriate radius. Sketch 6.1 shows isobars on the upper and lower wing surfaces. A shock wave is present on much of the upper surface, at about two-thirds chord and is strongest at $y/c_0 \approx 1.52$ ($\eta \approx 0.635$). There is an adverse

pressure gradient downstream of this shock wave, leading to the possibility of separation at the trailing edge at lower Reynolds numbers. An adverse pressure gradient, which intensifies towards the wing tip, is also present downstream of the start of the lower-surface cove. Experimental results should be carefully assessed for evidence of shock waves or separation in this region.

Sketch 6.2 shows the variation of the lift coefficient on line-of-flight sections across the span for the design conditions ($M_\infty = 0.785$, $C_L = 0.5$), calculated using FPIBE.



Sketch 6.2 Variation of sectional lift coefficient across span

7. SELECTION OF LINE-OF-FLIGHT SECTION FOR BVGK ANALYSIS

To reduce the computational burden, a single section is chosen for more detailed analysis using BVGK. Sketch 6.2 shows that the sectional lift coefficient is almost constant for $0.5 < \eta < 0.7$. A line-of-flight section at $\eta = 0.635$ ($y/c_0 = 1.52$) was chosen for analysis*. This section is in a region close to the maximum strength of the shock wave and where the flow appears to be quasi two-dimensional (Sketch 6.1).

7.1 Derivation of Two-Dimensional Section for BVGK Analysis

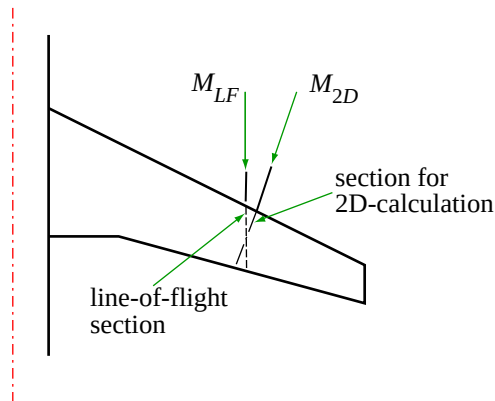
The line-of-flight section at $\eta = 0.635$ is determined by interpolation between the sections defined at $\eta = 0.4$ and $\eta = 0.7$ (Sketch 4.1). This section is then transformed to an equivalent section normal to the 50% chordline (Sketch 7.1) using the relationship given in Reference 9:

$$\left(\frac{z}{c_{2D}} \right) = \left(\frac{z}{c_{LF}} \right) \frac{\cos(\Lambda)}{\cos^2(\Lambda_{0.5})}.$$

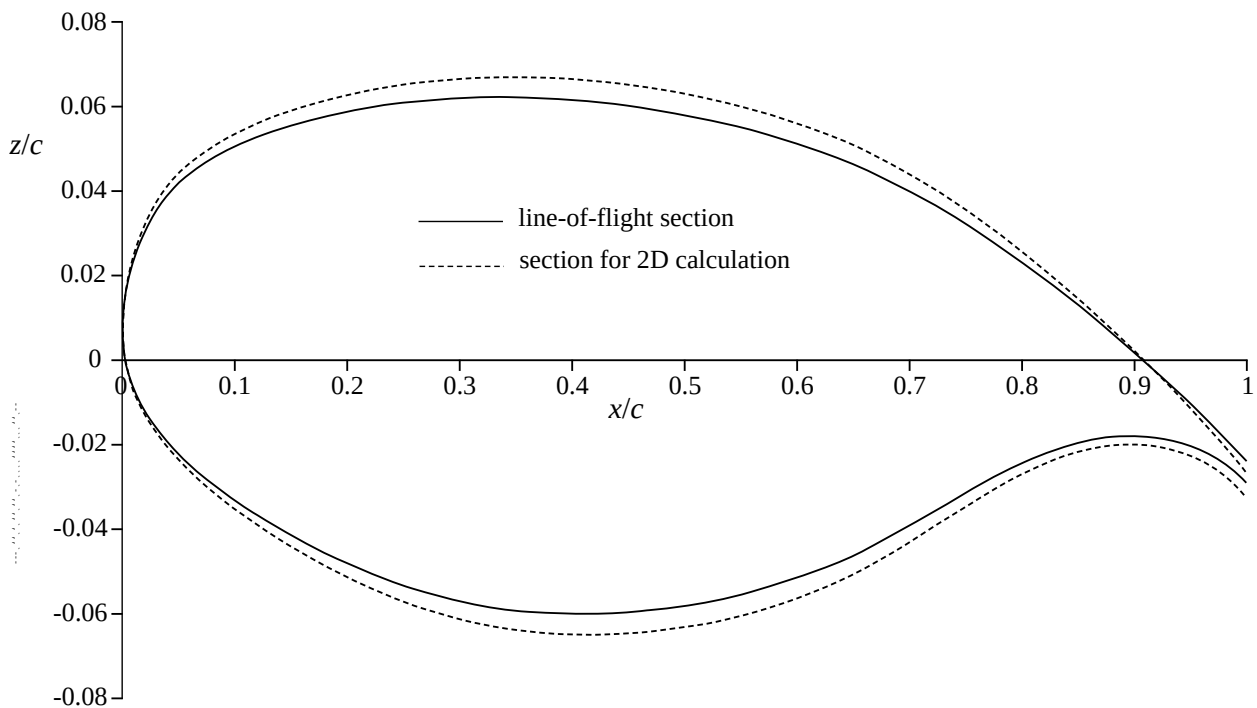
A comparison between the line-of-flight and transformed sections is given in Sketch 7.2†.

* This section was chosen to coincide with the location of available experimental pressure measurements on the F4 wing/body combination.

† This transformation method is used in preference to that suggested in Reference 6 to allow comparison with results given in Reference 4. In the general case experimental data would not be available and the section to be analysed might be chosen on the basis of the initial calculations of the strength of the upper surface shock or of the pressure gradient approaching the trailing edge.



Sketch 7.1 Relationship between line-of flight section for 2D calculation and section for 2D calculation



Sketch 7.2 Line-of-flight section and transformed section for two-dimensional calculation

7.2 Flow Conditions Applicable to Transformed Section

The design $M_\infty = 0.785$. The local Mach number at the chosen line-of-flight section ($\eta = 0.635$) is increased slightly by the influence of the fuselage, which can be estimated by considering the flow field around the fuselage in isolation. Data from Reference 5 imply that this Mach number increment is approximately 0.008. The approach Mach number at the section, M_{LF} is thus 0.793. The mid-chord sweep angle is 23 degrees and so the effective Mach number for the transformed section is:

$$M_{2D} = M_{LF} \cos(\Lambda_{0.5}) = 0.793 \cos(23 \text{ deg}) = 0.730.$$

7.2.1 Reynolds number for BVGK calculations

Re_{2D} should be chosen so that the boundary-layer characteristics, calculated on the two-dimensional section using BVGK, match those on the wing at the chosen chordwise station as closely as possible. This