Reprint

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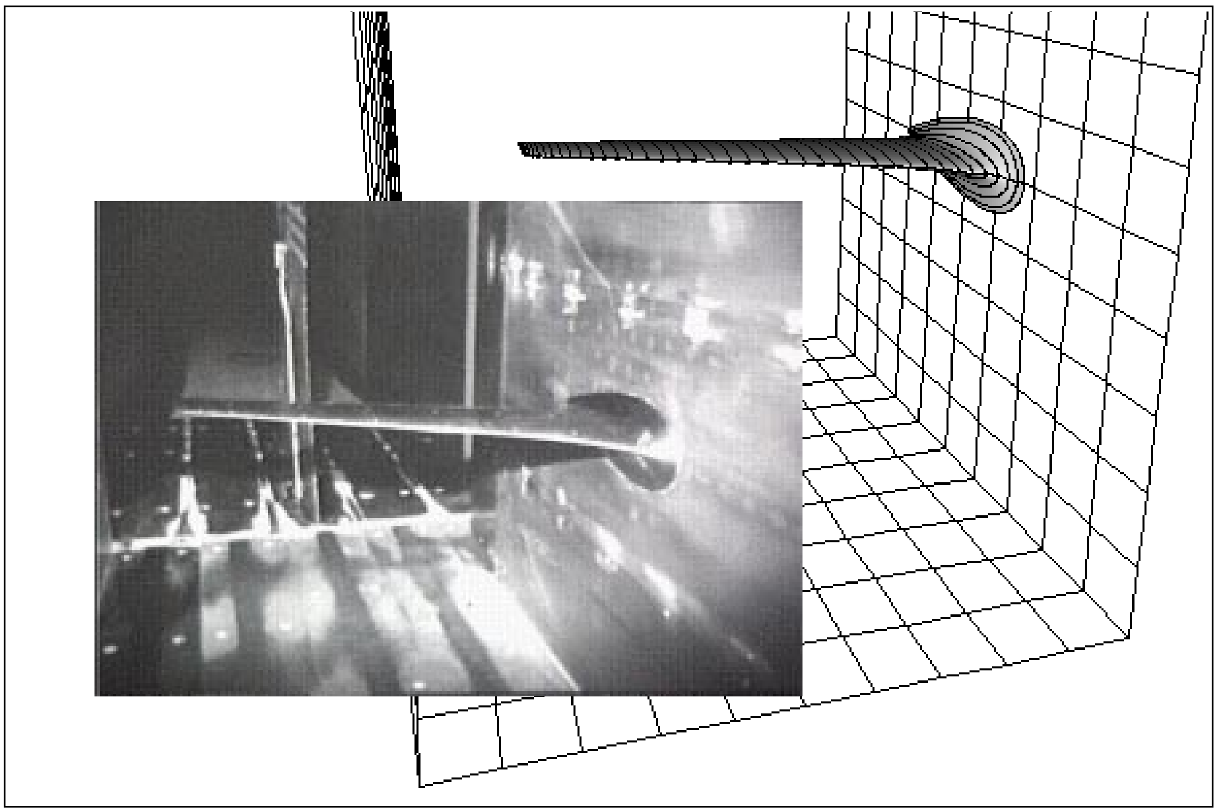
DLR - F5:

Test Wing for CFD and Applied Aerodynamics



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DLR - F5:
Test Wing for CFD and Applied Aerodynamics

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INTRODUCTION

A swept wing with symmetrical sections was originally created to serve two purposes [1]:

First, the surface generator used for data definition was under development for aerodynamic design and optimization. The wing created was intended therefore to be a selected case of a whole family of configurations obtained by variation of the input parameters. Aerodynamic design and optimization strategies call for such variations.

Second, CFD code development needs both accurate test case geometries as well as experimental results from wind tunnels. The latter usually suffer from corrections which still might suit practical purposes of measuring aerodynamic coefficients but fall short of the requirement to define the flow conditions to the same accuracy as geometrical boundaries are known.

Using the generator software, a compromise was chosen by including the closed wind tunnel wall geometry as a channel boundary surrounding an aerodynamic component. In order to also avoid model support problems, a wing half model mounted on and include a splitter plate was used as "configuration". Geometry of the flow boundaries was completely defined through the simple rectangular channel geometry completed by chosen inlet and exit planes. Flow data were required at these planes to formulate a boundary value problem for CFD.

In a workshop to compare CFD results with the first test case experiment (1986), partners had obtained and used a computer code to generate the wing and the windtunnel boundary conditions, along with the absolutely necessary parameters to formulate fluid dynamic boundary conditions for the Navier - Stokes equations [2]. This software is a simplified version of the geometry generator for aerospace configurations and CFD grid generation [3] which has since been further developed as an industrial tool for design aerodynamics. The experiment and the refined half - model technique was published [4, 5] and the results of the workshop, comparing numerical results, have been summarized [6]. Based on these results we may conclude that for CFD this test case turnes out to be a complicated one basically because of the observed viscous flow phenomena on the wing. On the other hand, the definition of the complete boundary value problem makes the case rather unique and, with the help of generator software and experimental data, easy to implement for CFD validation. The workshop software updated by experimental results for surface pressure distributions [7] is made available to potential users in one package.

1 GENERAL DESCRIPTION

1.1 Model name and flow conditions — DLR - F5 Test Wing

1.2 Model type and flow conditions — 20 deg swept wing (AR = 9.5) in transonic flow.
Supercritical wing flow at Mach ~ 0.82

1.3 Design requirements — Definition of a complete boundary value problem for CFD.
Analytical description of wing geometry.

1.4 Dominant flow physics — Swept wing flow with large fillet avoiding vortex at wing root leading edge.
Laminar / transitional / turbulent flow
Initially laminar separation bubble at shock - boundary layer interaction.

2 DETAILS OF MODEL

2.1 General geometric arrangement — see Fig. 1

2.2 Configurations tested — Wing in fixed wind tunnel boundary box

2.3 Wing and airfoil data — (see Fig. 2)

2.3.1 Planform — leading edge sweep: 20^o
trailing edge sweep: 12^o
Aspect ratio: 9.5
no twist
wing root: large fillet smoothing corner
wing tip: rounded
m. a. c.: 170 mm

2.3.2 Wing sections — defined from analytical blending of an
NACA 0036 airfoil at root with a
13% thick, shock-free designed (M_{inf} = 0.78)
laminar flow - type, symmetrical airfoil.
l. e. radius / chord = 0.005,
t. e. thickness / chord = 0.005
over most of the wing.

2.4 Body data — No body, wing mounted to splitter plate (see Fig.3)

2.5 Other components — Vertical splitter plate extends through top and bottom wind tunnel wall, to allow for rotation of 5^o.

2.6 Engine / pylon / nacelle data	None
2.7 Geometric definition of all components	Wing, splitter plate and 3 tunnel walls are solid parts of geometric boundary. Inlet and exit plane also define boundary. Resulting surfaces defined by software, evaluating analytical relations of geometry generator code E88 [3]. Model production with new CAM software based on same code, exceptional section accuracy, max. dihedral deflection $\Delta z = 0.1$mm
2.8 Model support details	see Fig. 2, 3.

3 GENERAL TUNNEL INFORMATION

3.1 Tunnel designation	Transonic Windtunnel Göttingen (TWG), see Ref. [8]
3.2 Organization running the tunnel	DLR - German Aerospace Research Establishment Central Windtunnels Division Göttingen Research Center, D-37073 Göttingen, Germany
3.3 Tunnel Characteristics	Continuous, closed circuit
3.4 Test Section	Cross section 1 x 1m, length 4m; box length 0.938m where measurements taken; Slotted top and bottom walls; in the present tests these walls were closed; closed side walls.
3.5 Freestream conditions	Transonic: $0.5 < M_{inf} < 1.2$
3.5.1 Reference flow conditions	Stagnation pressure: Var. between 0.4 and 1.6 bar, measured in settling chamber. Stagnation temperature: ~310 K, measured in settling chamber and used to determine static temperature. In the present tests all boundary conditions including static and total pressures in the inflow plane were measured.
3.5.2 Tunnel calibration	Static pipe and side wall static pressures, Ref. [9]. Last calibration1986

3.6 Flow quality

3.6.1 Flow uniformity

$M_{inf} \sim 0.8$: $\Delta M = 0.001$
$\Delta\alpha = \Delta\beta < 0.05^{o}$, measurements in the inflow plane by wedge probe (detailed data simulation given by generator code).

3.6.2 Temperature variations

Negligible

3.6.3 Flow unsteadiness

Overall turbulence level: < 0.35%
Overall noise level: $(nF(n))^{1/2} < 0.001$

4 INSTRUMENTATION

4.1 Model position

Angle of attack calibration with Incremental Sensor, accuracy 0.01^{o}

4.2 Model pressure measurements

Scanivalve pressure measurements on wing, splitter plate and three tunnel walls.

4.2.1 Total number and position

Wing:
230 pressure orifices along 10 sections
(Figures 2, 6).
Each section: 20 on upper, 3 on lower surface.
Due to symmetry of the sections, the effective number of orifices was 20 on each surface, realized by a corresponding variation in angle of attack. The 3 orifices on the lower surface were used to check data compatibility.

Splitter plate:
87 pressure orifices.

Tunnel walls:
67 pressure orifices.

4.2.2 Range and accuracy of pressure transducers

PSI 780 B Pressure Measurement System with 32 ESP Sensor Modules
Range: 15 PSI;
Static error band: typical 0.07% FS;
calibrated on-line

4.3 Force and moment measurements	None
4.4 Boundary layer and flow field measurements	Boundary layer measurements at the location of the inlet and exit planes on the three tunnel walls and the splitter plate; flow field measurements in the inlet and exit planes (Figures 4, 5)
4.4.1 Measurement technique applied	Traversable calibrated probes (Fig. 4): pressure (static and pitot), temperature and horiz. and vert. flow angularity Pitot pressure rakes (Fig. 5) for boundary layer and wake profiles.
4.4.2 Flow regions investigated	Inlet flow plane: Total and static pressure, flow angles, temperature, boundary layer profiles; Exit flow plane: As inlet, plus wake profiles.
4.4.3 Influence of probe support	All pressure measurements on model were taken for each probe position to control influence on model flow quality. Final surface pressure data taken without probes
4.5 Surface flow visualization	
4.5.1 Measurement technique applied	Acenaphtene method (sublimation technique) for visualizing lamina/turbulent surface flow boundaries (Fig. 7)
4.5.2 On which surfaces is visualization technique applied?	On wing upper and lower surface
4.5.3 In what form are data available	An analytical ‘transition ramp function’ is modelled from the visualization and added to the data generator code.
4.6 Flow field visualization	None
4.7 Tunnel wall measurements	Surface pressure measurements, see 4.2

4.8 Other measurements	Monitoring unsteady flow onset by tip mounted accelerometer and observing root bending moment measured by strain gages.

5 TEST MATRIX AND CONDITIONS

5.1 Detailed test matrix	Mach(inlet plane) = 0.82 Angles of attack: 0 and 2 degrees (geometric) Re =1.5 Mill. (based on mean chord 150 mm)
5.2 Model / tunnel relations	Configuration consists of whole tunnel boundary box with wing; - maximum blockage: 0.0091 - model span / tunnel width: 0.77 - adiabatic wall conditions have been reached.
5.3 Transition details	Free transition; see 4.5.1

6 DATA

6.1 Availability of data

6.1.1 Organization owning data	DLR
6.1.2 Responsibility for data	H. Sobieczky tel. (49) (551) 709 2287 fax (49) (551) 709 2101 internet e-mail: Helmut.Sobieczky@dlr.de
6.1.3 Are data freely available	yes.
6.2 Suitability of data for CFD validation	Complete formulation of viscous flow boundary value problem, with geometry, analytical inlet and exit flow models. Data already used for Navier-Stokes workshop computations, Ref.[6].
6.3 Type and form in which data are available	Diskette with analytical geometry generator FORTRAN code and selected surface pressure data [7].
6.4 Corrections applied to data	No corrections to measurements. Elastic model deformation for lifting flow case ($\alpha = 2^o$) negligible; determined by bench test.

7 DATA ACCURACY AND REPEATABILITY ASSESSMENT

7.1 Accuracy estimates

7.1.1 Freestream conditions

Inlet Mach number (0.82) and flow angularity measured with calibrated probe:
$\Delta M \sim \pm 0.001$
$\Delta \alpha \sim \pm 0.02^{o}$
Pressure coefficients: $\Delta c_p \sim 0.002$

7.1.2 Acenaphtene visualization

Photographic evaluation and analytical approximation by suitable generator functions (Fig. 8).

7.2 Repeat measurements

Within a few days some of the measurements were repeated several times. Differences of $\Delta c_p \leq 0.002$ were observed.

7.3 Other tests made

A new experiment with DLR - F5 has been made in 1990 with other flow parameters. These data are not yet available.

8 REFERENCES

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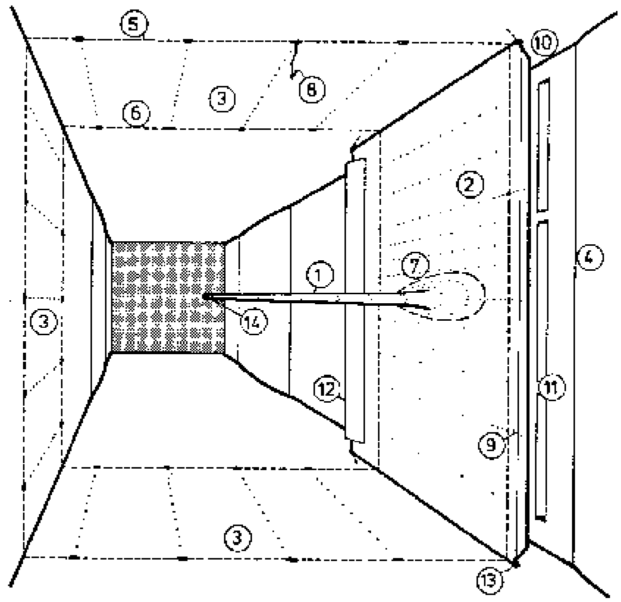


Figure 1: Wind tunnel arrangement: 1 Wing, 2 Splitter plate, 3 Tunnel walls, 4 Fourth wall contoured, 5 Inlet flow control plane, 6 Exit flow control plane, 7 Pressure orifices (total: 416) on wing, plate and walls, 8 Traversable probe or rake in inlet and exit planes, 9 Splitter plate leading edge with transition strip, 10 Bypass channel, 11 Suction devices, 12 Diffusor flap, 13 Rotation device for plate plus wing, 14 Acceleration and strain gages.

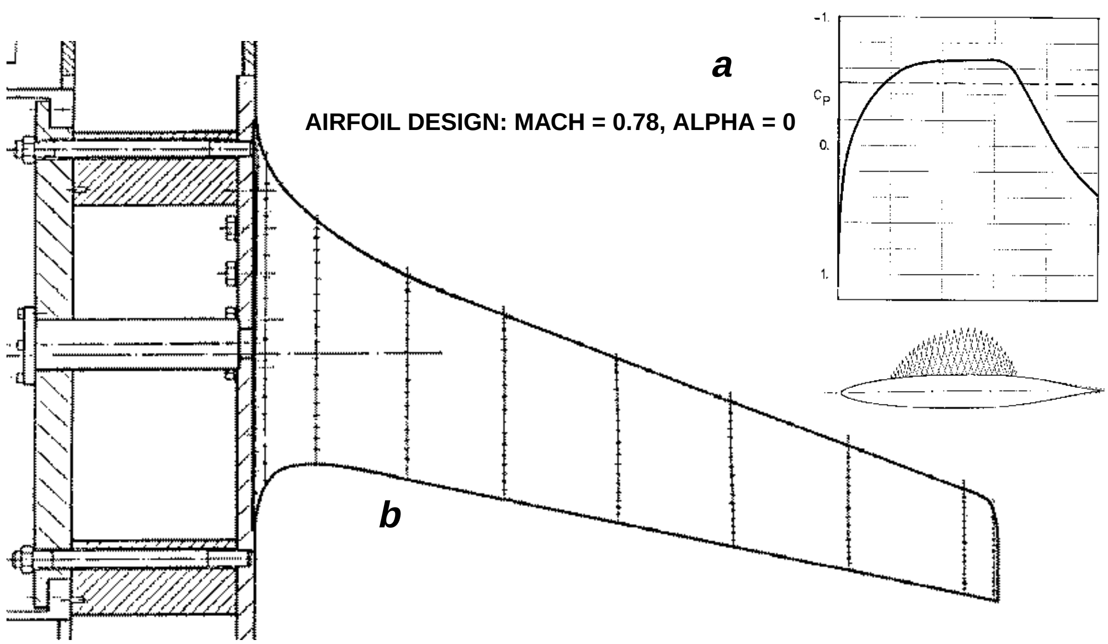
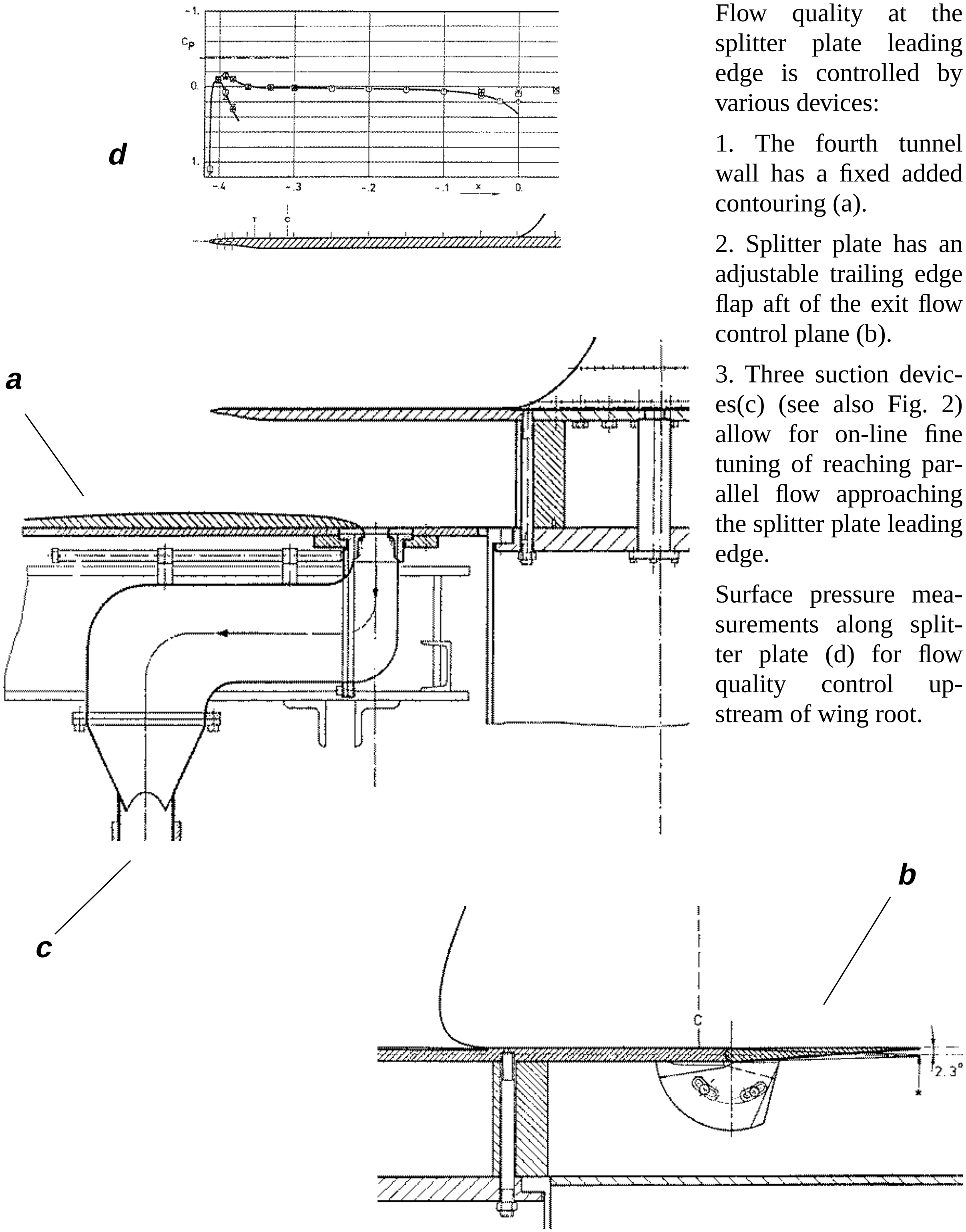


Figure 2: (a) Theoretically shock-free designed symmetrical airfoil as basic wing section,
(b) Wing planform view, mounting to splitter plate and tunnel wall. Pressure orifices on wing.



Flow quality at the splitter plate leading edge is controlled by various devices:

1. The fourth tunnel wall has a fixed added contouring (a).

2. Splitter plate has an adjustable trailing edge flap aft of the exit flow control plane (b).

3. Three suction devic-es(c) (see also Fig. 2) allow for on-line fine tuning of reaching par-allel flow approaching the splitter plate leading edge.

Surface pressure mea-surements along split-ter plate (d) for flow quality control up-stream of wing root.

Figure 3: Details of Half-model technology with fine-tuned splitter plate flow quality.

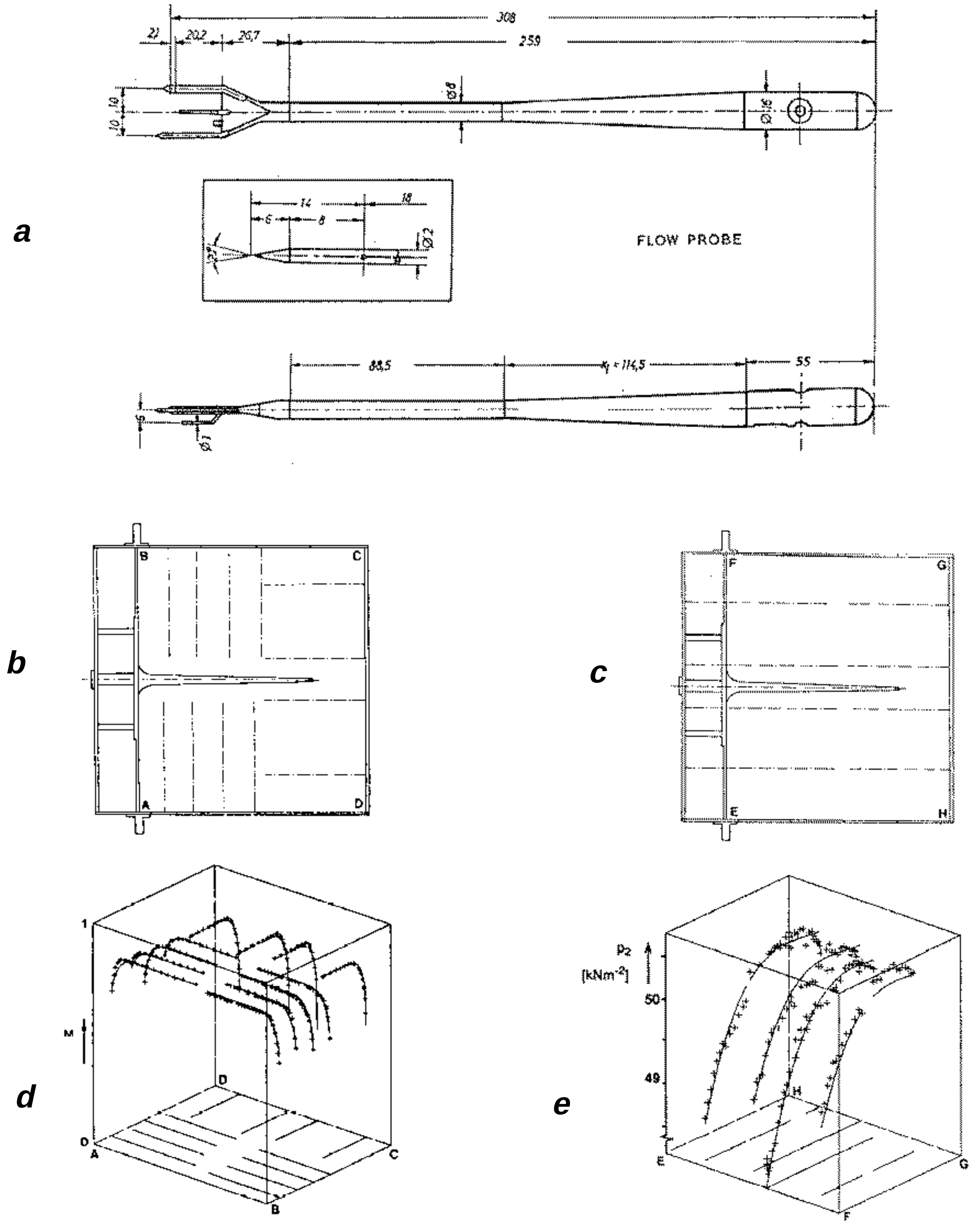


Figure 4: Traversable Probe (a) for static and pitot pressure, horizontal and vertical flow angularity, temperature. Traversing paths of probe in inlet plane (b) and exit plane (c).

Inlet flow Mach number distribution (d), exit flow static pressure distribution (e); analytic approximation by boundary conditions generator code.

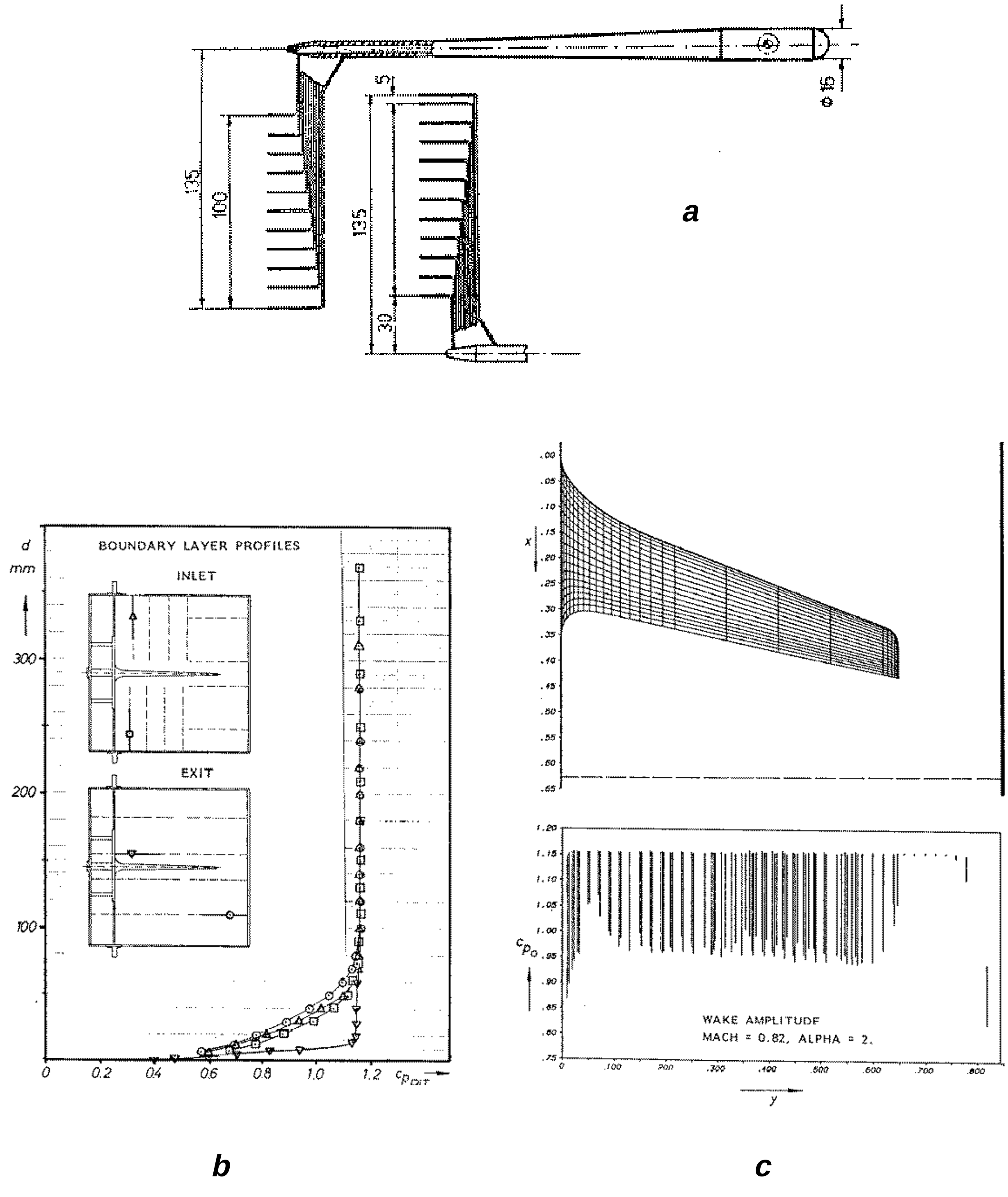


Figure 5: Wake rakes (a), Measured boundary layer profiles in inlet and exit plane (b).
Measured wake amplitude distribution behind wing in exit plane (c)

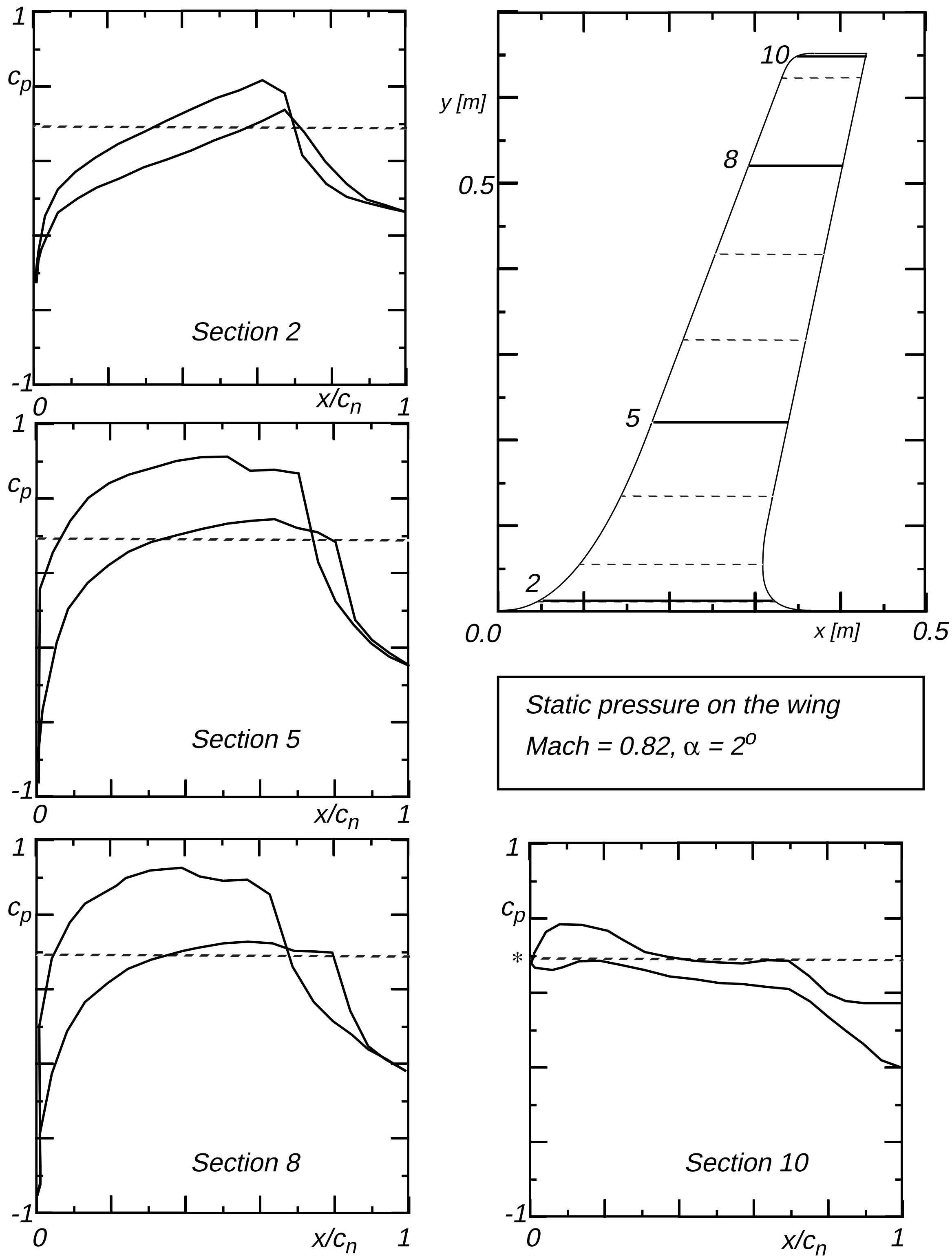


Fig. 6: Selected pressure distributions for Mach = 0.82, $\alpha = 2^o$

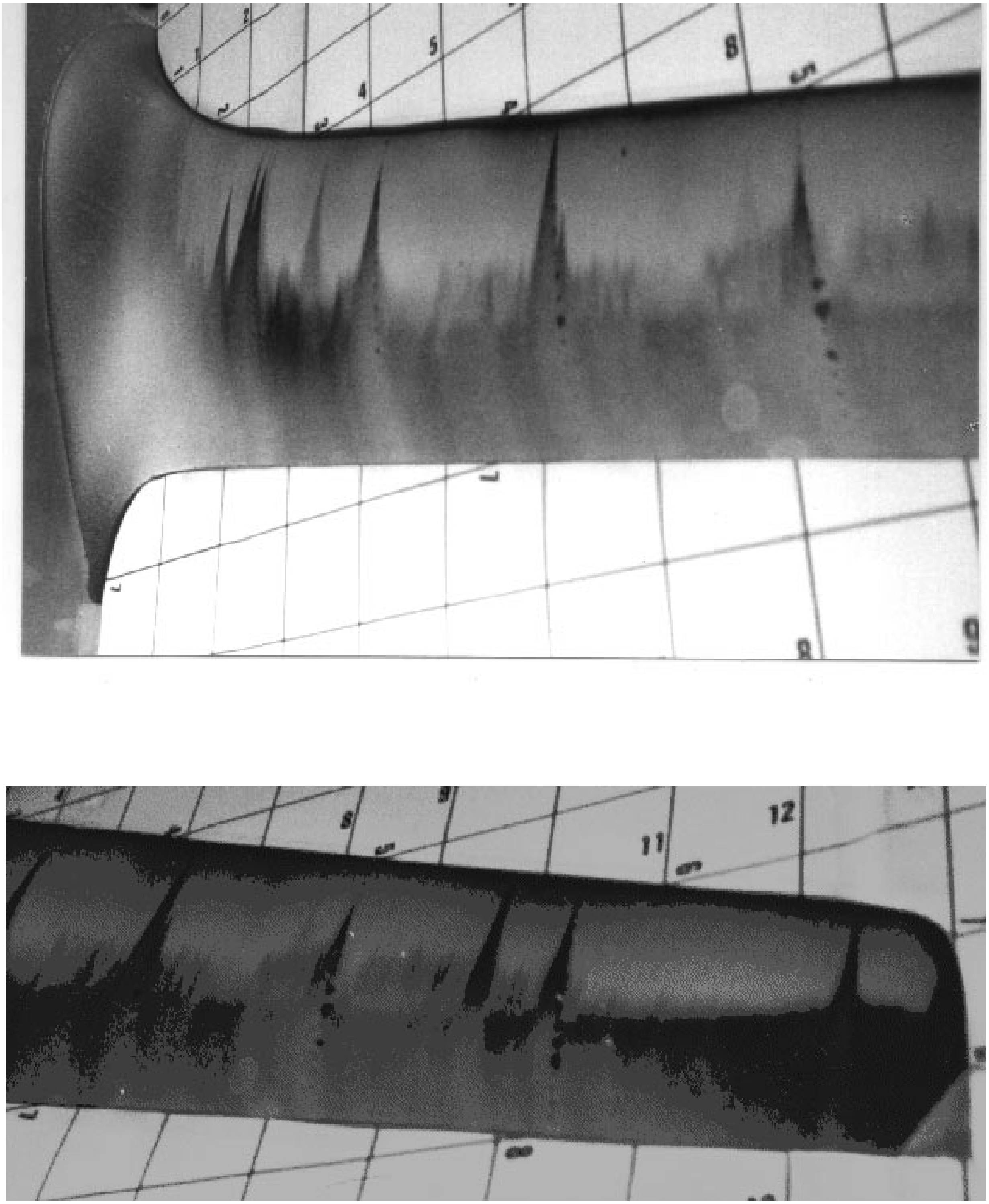


Figure 7: Acenaphtene visualization (Sublimation technique) of laminar / turbulent boundary layer on the wing surface.

(Mach = 0.82, $\alpha = 2^{o}$, upper surface)

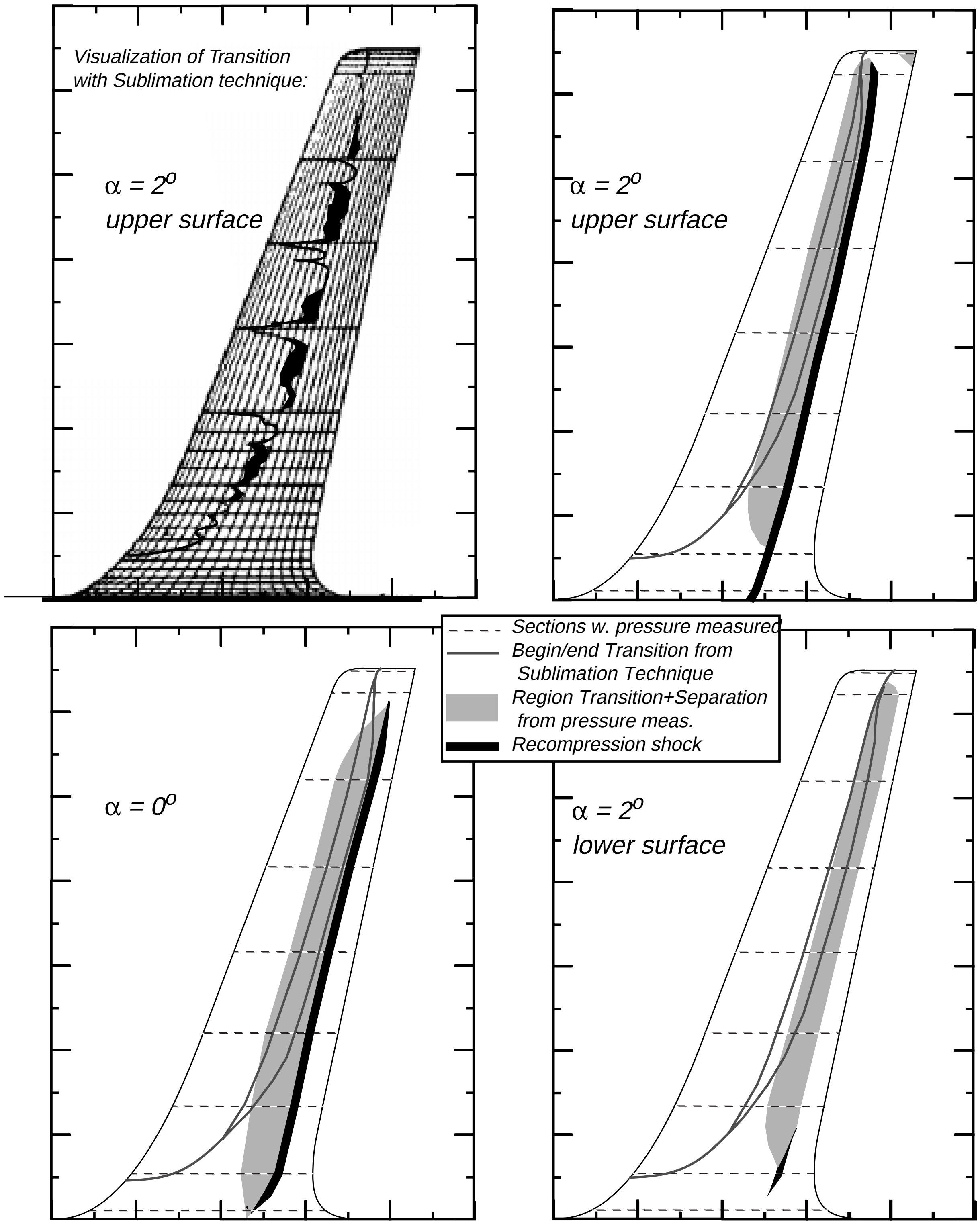


Fig. 8: Evaluation from pressure measurements and sublimation technique: Regions and curves indicating Transition line or laminar separation begin and shock location.