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DESIGN AND OPERATION OF TU-BERLIN WIND TUNNEL WITH ADAPTABLE WALLS

U. Ganzer, Y. Igeta and J. Ziemann
Technische Universität Berlin
Germany

Abstract

At the Technical University of Berlin two test sections with adaptive walls have been developed: A 2-D section with flexible top and bottom wall is mainly used for aerofoil tests but also for some 3-D model tests. An octagon test section with eight flexible walls was specifically designed for the test of three-dimensional models.

The test sections are described with particular reference to their constructional details and their automatic control system. Representative test results are exhibited and the problems which have occurred in course of the first years of operation are discussed.

The implications of using the octagon test section at supersonic flow conditions are outlined in some detail.

1. Introduction

More then 10 years ago, in 1973, Ferri and Baronti /1/ als well as Sears /2/ suggested a simple procedure to control the boundary condition in a wind tunnel test section to avoid wall interferences. This gave the background for what is widely known by now as the adaptive wall technique.

In this technique differential adjustment of the wall boundary condition is exerted such that the streamlines near the wind tunnel wall are allowed to take its interference free shape. The means of achieving this are either flexible walls with its shape controlled by a number of jacks, or ventilated walls with devices to control the flow through the wall locally /3/. The control is based on a comparison of two flow variables measured on a control surface near the wall, usually the pressure and the flow direction. It has to be emphasized, that no information is needed about the wind tunnel model itself.

Wind tunnels with adaptive walls have been used very successfully for aerofoil tests. In fact they have been found to give the most reliable test results in this case /4/. For the test of three-dimensional models, however, the experience with adaptive wall tunnels is still very limited. At subsonic speeds the technique was only proved in principle two years ago while at supersonic speeds no experimental results are available up till now /5,6/.

In the following some experimental results will be presented to demonstrate the state of the art and to illustrate the problems involved in this technique. In addition, the implications of testing at supersonic speeds will be discussed.

It may be mentioned here that at the TU-Berlin the concept of f l e x i b l e s o l i d walls was given preference. This concept is seen to have two important advantages:

- The measurement of the two flow variables on the control surface near the wall is very simple.
- The drag due to wall ventilation is avoided. This results into a substantial reduction in driving power requirement for the wind tunnel, estimated to 30 % at transonic speeds.

Thus, it will be decisive to demonstrate, that the use of flexible adaptive walls is not restricted to subsonic flow condition.

2. Constructional features of the 2-D test section

A schematic layout of the test section for two-dimensional model tests is shown in Figure 1, which represents the main features of this design. It has a cross section of 15 x 15 cm and a total length of 69 cm. The flexible walls extend over a region of 55 cm, i.e. 5.5 model chords. They are made of fibre glass, 1 mm thick.

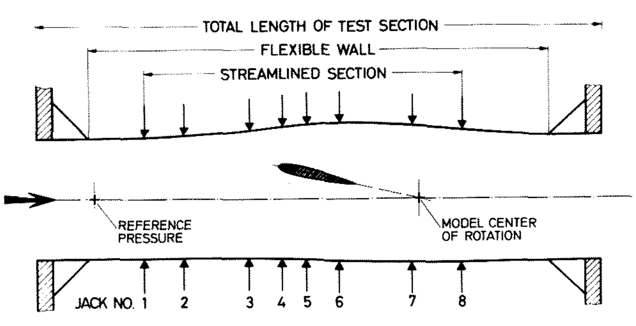


Figure 1: 2-D test section layout

In the regions of large jack spacing the walls are stiffened by additional fibre glass plasters. Each wall can be adjusted by means of 8 jacks with a maximum displacement of ± 25 mm. The jacks are driven by DC electro-motors. Eight potentiometric displacement feeler touch a piece of metal, glued on the flexible wall at each jack position. With this arrangement the reading of wall position is within an accuracy of 0.07 mm.

A double-hinge system of the jacks allows local inclination of the wall as well as some longitudinal displacement.

In Figure 1 the wall configuration is sketched for a conventional NACA 0012 aerofoil at M_∞ = 0.5 flow condition much beyond stall. Maximum wall displacement for this flow case is about 19 mm including compensation of wall boundary layer displacement thickness.

The aerofoil model is mounted exentrically in a disk which is turned for changing angle of attack, so that the aerofoil translates vertically with increase of incidence. That helps to centralize the model between the walls in the presence of increasing up and downwash.

The grouping of the jacks such that for the various flow conditions adequate control of the wall shape can be achieved, was to quite an extend done by intuitive engineering judgement. There are four jacks per wall near the model with a spacing less than 0.4 model chord, while upstream and downstream the jack spacing increases up to 0.9 chord. The number of jacks and its grouping was based on calculated streamline shapes for a NACA 0012 aerofoil. These calculations provided information about the order of magnitude of the local streamline inclination as well as the streamline curvature and the position of maximum displacement.

In view of the planned development of a 3-D test section it was the explicit aim to use only a minimum number of jacks. Since the flexible walls are positioned only at a finite number of jacking points, there is no direct control of the wall shape between the jacks. However, these wall portions will have shapes corresponding to bending lines which are expected to be a good approximation of streamline shapes. In order to minimize wall pressure loading provisions are made to vent the plenum on the other side of the flexible walls.

3. Operational experience with the 2-D test section

3.1 Airfoil pressure distributions

Experiments had started with the conventional NACA 0012 aerofoil and later were concentrated on the supercritical CAST 7 with moderate rear loading and moderate adverse pressure gradient on the upper surface. This aerofoil exhibits high sensitivity to changes in Mach number and angle of incidence near design condition, so that its pressure distribution is considered to be particularily suited a measure for wall adaptation assessment /7/.

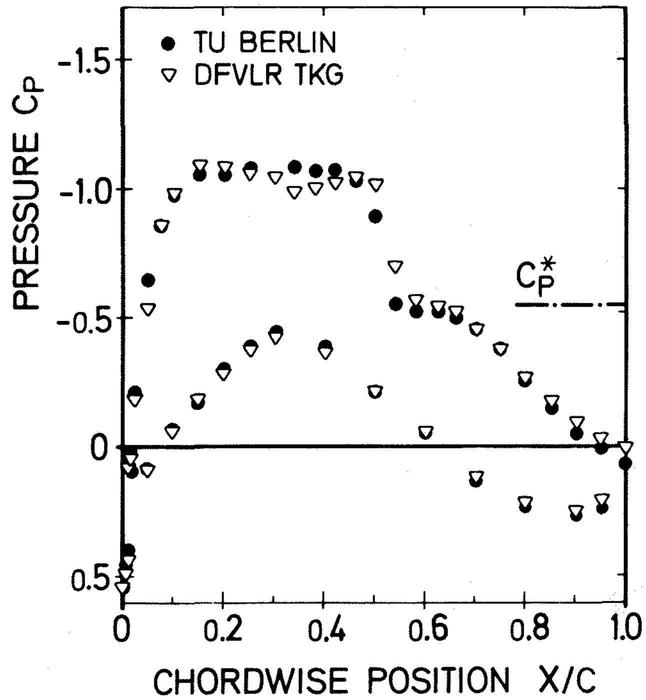


Figure 2: Pressure distribution
CAST 7 aerofoil

Figure 2 shows a representative result for the CAST 7 aerofoil at a condition near the design point. Comparison is made between test result obtained in the TU-Berlin tunnel with adapted walls and that obtained in the DFVLR transonic 1 x 1 m tunnel using the same model. The tunnel height to model chord ratio was 1.5 in the TU Berlin tunnel against 10 in the TKG, so that in the latter case interference from upper and lower wall was assumed to be negligible. In order to have similar side wall interferences the model in the TKG was mounted between 1 x 1 m side plates.

Although the two pressure distributions are very similar and the remaining discrepancies may be considered acceptable, possibilities of improving the accuracy of TU-Berlin data have to be checked carefully. Such possibilities will be discussed in the following section.

3.2 Analysis of wall data

The analysis of wall data reveals some uncertainties in the adaptation procedure which are caused primarily by the finite length of the test section. Furthermore it was found that small inaccuracies in wall pressure measurements due to local wall waviness may have remarkable effects.

Linear small pertubation theory is used for the wall shape calculation which yields a simple integral relation between pressure and wall curvature. This calculation method exhibits one unsolved problem: The calculation v at the upper and lower test section wall as well as the calculation of the new wall shape was carried out to date taking into account the measured disturbance velocity u just over the region of the compliant walls, i.e. 5.5 aerofoil chords. Possible disturbances further upstream resp. downstream were considered to be of negligible effect. But actually the aerofoil flow affects the wall pressure distribution still at the ends of the test section and thus the arbitrary truncation of the integration interval (x/c = 2.25 to 3.25) can lead to erroneous results:

For several test runs the calculated v-disturbances due to the experimental u-velocity distribution, taken along the length of the compliant walls, were decaying only at a distance of about 10 aerofoil chords. Thus it is disadvantageous to get surface pressure measurements for only a relatively small region. An enlargement of the integration interval to a length much longer than the test section is therefore desirable but feasible only when the measured velolity field is complemented by an appropriate extrapolation method. It is envisaged to adopt an extrapolation procedure in the near future similar to that developed by ONERA for the T2 wind tunnel /8/.

A second problem due to the finite length of the test section is related to the fact that the wall contour can be adjusted to the calculated shape only over a finite part of the flexible wall. This is because the wall at both ends is fixed into a horizontal direction. Thus the wall may be considered fully streamlined only between first and last jack, see Figure 1.

As already pointed out a lack in accuracy of wall pressure measurement may lead to an additional error in wall shape. The sensitivity of the calculation method due to some scatter in measured wall pressures is indicated in Figure 3. Two wall contours are shown which were calculated by using the original measured pressure distribution (solid line) on the one hand and a corresponding locally smoothed pressure distribution (dotted line) on the other hand. Note here the local distorsions caused by fixing at the test section wall at its downstream end.

An improvement of the accuracy of test results beyond the present state seems to be attainable by imposing slight modifications to the current design:

First of all it is felt that the length of the adjustable section should be extended in order to reduce the truncation effects. An extension of altogether two airfoil chords is planned, involving additional bearing. The row of pressure taps will be extended accordingly.

With the extension of the test section the joint between nozzle and test section will be moved upstream away from the model. This is important since this joint was found to cause a remarkable disturbance. The reason for this disturbance is that the nozzle ends with a four sided divergent

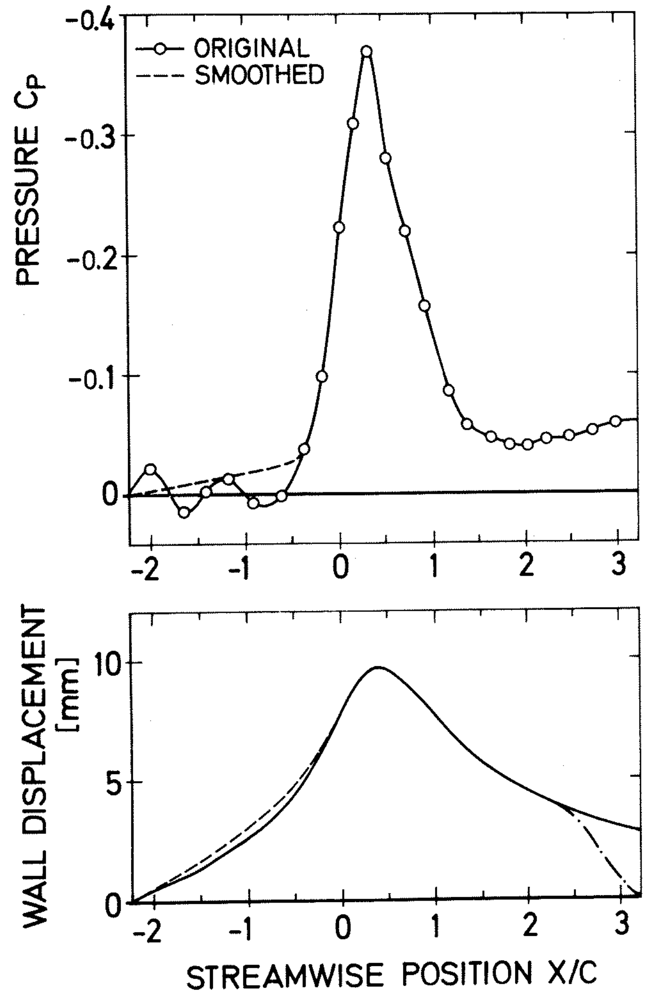


Figure 3: Effect of scatter in wall pressure measurements

wall to account for the wall boundary layer growth while in the test section only top and bottom wall provide the required divergence. The sudden change in wall divergence leads to a disturbance showing up in the wall pressure distribution in the neighbourhood of the joint.

In addition to the test section extension it is planned to replace the fix joint for the wall at the test section end by an additional jack. This will allow to adapt the cross section at the exit to the flow condition.

Improved streamline contours could possibly achieved with flexible walls having varying thickness in accordance with the jack spacing. The feasibility of manufacturing such walls as well as their applicability is just being studied. This includes measurements of actual shapes of the currently fitted walls with high precision instruments.

A general remark has to be added here: With all the effort made to reduce wall interferences stemming from top and bottom wall one has to be aware of the fact that improvement of aerofoil test technique will also require a treatment of the sidewall effects /9/. This problem has been excluded here from the discussion.

4. On the use of the 2-D test section for 3-D model tests

First tests have been made with a body of revolution in the test section with two flexible walls. A one-step procedure of Wedemeyer was employed to adapt the walls in the presence of that three-dimensional model /10/. A fairly long model sting had to be used mounted on a quadrant downstream of the test section. The lack of stiffness of this sting was made up for by steel cords spanned across the test section. They embraced the sting at 0.5 model length downstream of the model. This device did avoid oscillations of the model.

The body of revolution tested is a calibration model designed by ONERA. Interference-free data for the surface pressure distribution are available from /11/. Surface pressure distributions were measured with the walls plane and adapted. The results are plotted as surface Mach number distribution in Figure 4.

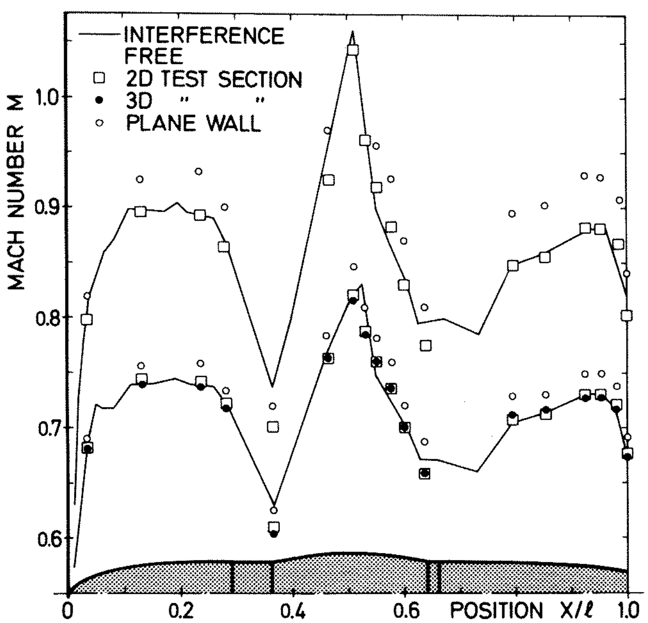


Figure 4: Mach number distribution for ONERA C-model $M_\infty = 0.70$ and $M_\infty = 0.84$

For subcritical as well as for transonic flow condition the measured values come close to the interference free data. At the higher main stream Mach number $M_\infty = 0.84$ the test section was choked with the plane wall configuration. It has to be explained that the term 'plane wall' is used here for the slightly divergent wall which produces a constant Mach number in the empty test section. In addition, for the transonic test case, the last jack on top and bottom wall had to be adjusted to eliminate blockage due to the support system. The additional displacement was 1.75 mm.

The success in achieving fairly interference free flow condition for a 3-D model in a test section with only two flexible walls indicates that there is a fair chance to use this comparatively simple test section design for general tests of 3-D lifting configurations at transonic speeds. Theoretical considerations by Smith /12/ and some measurements reported by Harney /13/ do support this view.

5. The automatic control system for the adaptive walls

Manual operation of 16 jacks, as carried out during the early tests with the 2-D test section, was already found to be very laborious. The development of the 3-D test section with something like 80 jacks made it mandatory to use an automatic control system, with the control circuit is made up by potentiometric displacement transducers, a multiplexer, a microprocessor, the motor drive unit and the DC motors, Figure 5.

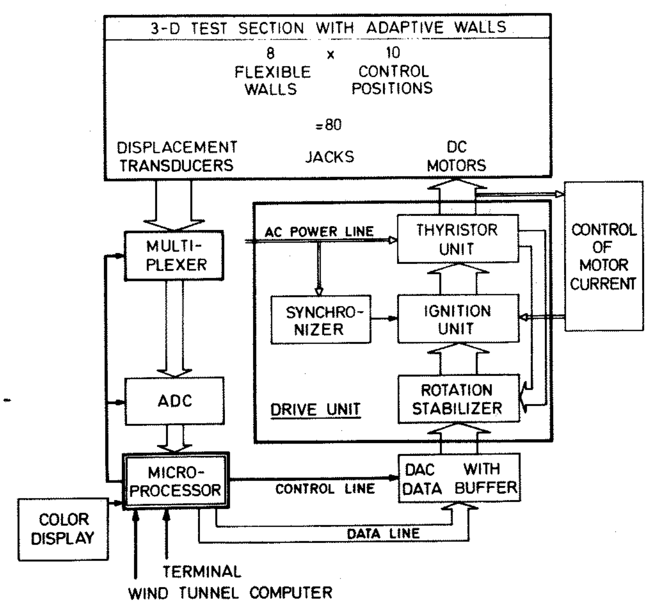


Figure 5: Flow diagram of control system for adaptive walls

The displacement transducers provide analog values of the actual jack positions, the transducers are scanned by the multiplexer and the readings are feeded into the microprocessor via an analog/digital converter.

The microprocessor as the key element of the control circuit compares actual and nominal jack position of up to 80 jacks every 100 ms and, if indicated, initiates a displacement of the jacks. There are two different algorithms to calculate a voltage to be assigned to a DC motor for driving a jack:

For required jack displacements greater than 0.3 mm, all jacks will be assigned a voltage proportional to their displacement. The jack which has to produce the largest displacement will be moved with maximum speed. In this way all motors will start at the same time driving the jacks and also stop at the same time. Thus the loads on the flexible walls are minimized.

The second algorithm is of quasi-steady PD (proportional, differential) time responds type. It becomes active, when the required wall displacement is in the order of 0.3 mm or less, and provides a speed control, that increases the voltage for those motors, which have to move against loads.

Within the control interval of 100 ms there is time left for the processor to carry out a drawing of the last measured wall positions on a colour graphic monitor. Such quasi continous display of the actual wall shapes is one of a variety of safety measures in order to prevent damages to the test section walls due to overload. Besides that the user is free to store the wall shapes of all iterations for comparison.

As an other safety precaution the processor examines regularily the absolute displacements as well as the differential displacements of neighbouring jacks and stops the motors if a prescribed safety limit is reached. Moreover the control soft-ware data are regularily checked in order to avoid memory failures.

An additional, independent safety measure is the regular control of the motor current. The motor current is a measure of the local wall load due to the wall displacement. A limit is set to avoid overload. The power supply from the drive unit is interrupted for all motors when the limit is reached.

In general, the experience with the automatic control system has shown, that it provides high accuracy of wall positioning (± 0.015 mm) and is very reliable when the safety procedures described above are incorporated. Since the control is in the first place build up by software outfit of the dedicated microprocessor the system is very flexible and very economical.

6. Constructional features of the 3-D test section

From the very first beginning the research on adaptive walls at the TU Berlin was aimed at an application to the test of three-dimensional models. The development of the 2-D test section and the aerofoil test's were ment to provide the necessary experience with this technique.

While for a 2-D test section the choice of flexible walls instead of ventilated walls with local control of ventilation is very natural, the decision is not so straightforward for a 3-D test section: A two-dimensional wall shape can easily be arranged with flexible walls but a three-dimensional wall shape can only be approximated, since the mechanical complexity has to be limited.

However, clear preference was given to the use of flexible solid walls also for the 3-D test section design on the grounds of the following arguments:

- Solid flexible walls offer the unique possibility of a simple measurement of the two flow components (u and v-disturbance) necessary for the wall adaptation procedure. This is done by measuring the static pressure along the wall and the wall shape.
- The reliable measurement of these two flow quantities is essential also for applying modern methods for the assessment of residual wall interference.
- Ventilated walls of transonic test sections are causing appriciable losses in the flow of a wind tunnel. A coarse estimate may be taken from Figure 6 which is based on data published in /14/. It is indicated that ventilated walls account for something like 30 % of the total power requirement of a wind tunnel.

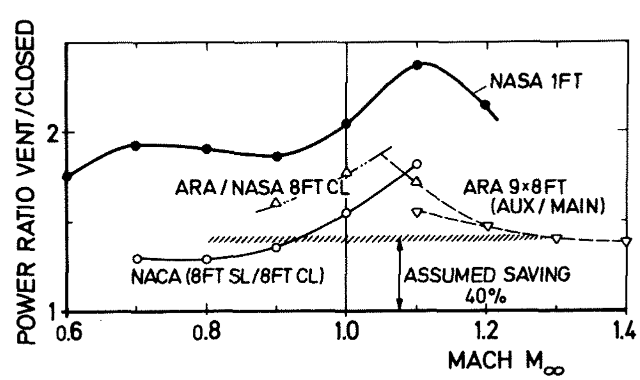


Figure 6: Power requirement of ventilated wind tunnels in comparison to closed tunnels

The particular feature of the TU-Berlin test section design is its octagonal cross-sectional shape, with the test section bounded by eight flexible walls. Each flexible wall can be deformed two-dimensionally by 10 jacks. It is obvious that no complete 3-D wall shaping is obtainable with this construction but the choice of eight walls was considered a good compromise between the desired three-dimensional wall shaping and limited mechanical complexity, leaving some room for a span-wise variation of wall contour.

A number of overlaping slim strips of spring steel are used to seal the slots between two adjacent walls. One end of each strip is spot welded on one wall, the other end slides on the inside of the adjacent wall. These spring steel lamellas provide a flexible fairing between the two-dimensionally deformed walls, thus giving an interpolated wall contour. The test section actually used for the research at the TU-Berlin is of 18 cm width and 15 cm height. More details of the construction of this test section are given in /7/.

The steel construction of the test section is in principle suitable for cryogenic operation. This is of importance, when one consideres the application of this technique to cryogenic test facilities such as the European Transonic Wind tunnel (ETW). In fact the suggestion has been made to apply the octagon design to the ETW. A very coarse first outline of a construction is presented in the following sketches. These figures give an idea how an adapting wall test section would look like at a larger scale.

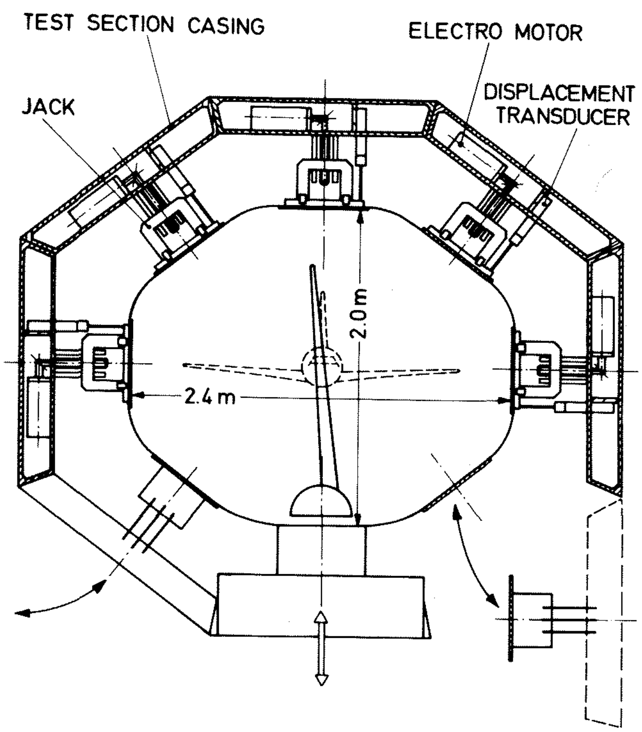


Figure 7: Details of octagon test section (suggested for ETW)

In Figure 7 an octagon test section with adaptive walls 2.4 m wide and 2.0 m high is sketched. The electric instruments like elctro motors and potentiometric displacement transducers are isolated. They are installed in the solid test section casing, which also carries all mechanics.

The arrangement of the octagon test section in a plenum chamber as currently planned for the ETW is shown in Figure 8.

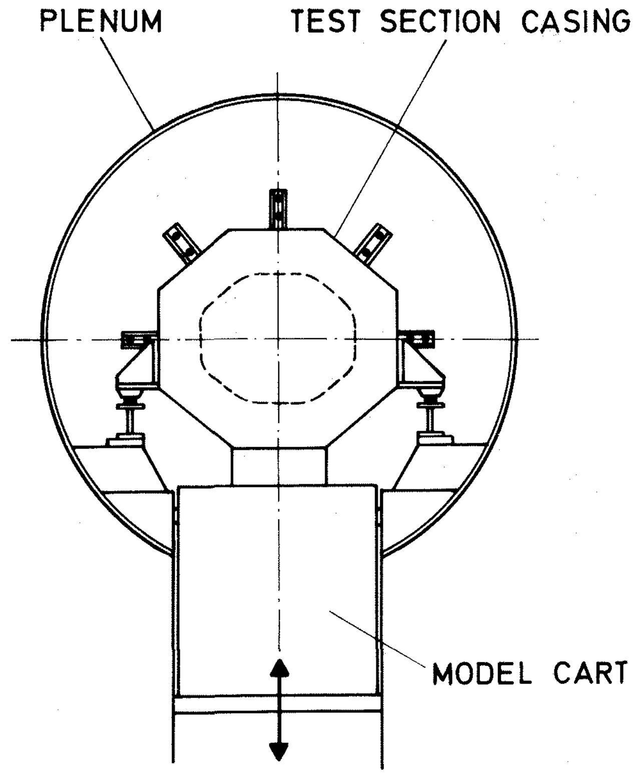


Figure 8: Octagon test section in ETW plenum chamber

For model change the two lower diagonal test section walls have to be rotated around the pivot, see Figure 7. Then the model cart together with the model support system and the lower flexible wall can be lowered to move the cold model into the temperature conditioning room, Figure 9. There are two different model carts, one for a sting supported full model and a flexible bottom wall and one for a half model investigation.

The cross-section area of the octagon test section is by 17 % smaller than the corresponding rectangular cross section (4.0 m^2 against 4.8 m^2). Accordingly, the power requirement of the wind tunnel is reduced, which comes on top of the savings due to a solid wall boundary. In addition to that, the adaptive wall test section has the advantage that it may allow to increase the test Reynolds number as a result of using larger models in nearly interference free flow and the calculation of residual interference based on the wall pressure measurement.

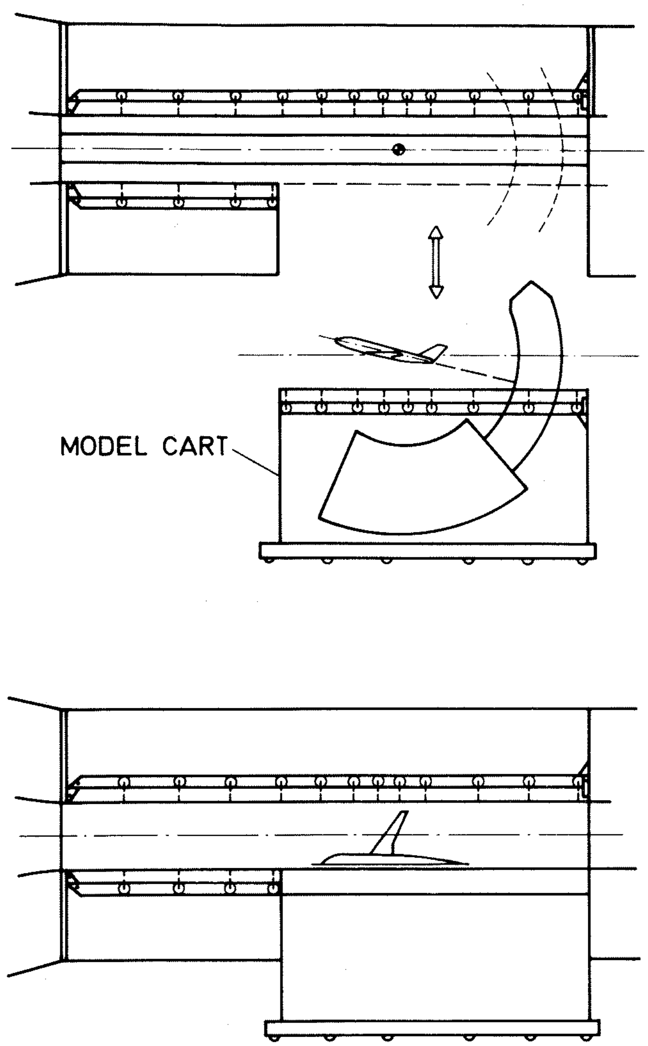


Figure 9: Model change (full & half model)

7. Operational experience with the 3-D test section

Two different models have been chosen for exploratory tests in the octagonal test section. The first one is a body of revolution called ONERA C-model for pressure measurements. Interference free data are available from /11/. First tests with the C-model demonstrated that in principle the octagon test section can avoid transonic blockage and reduce wall interference /5/, see also Figure 4.

The second model is a wing-body combination called ZKP-F4 used for force measurements. It is an Airbus-like configuration with supercritical wing sections and a span of 120 mm. A small internal three-component balance had to be designed for this model. The balance with a diameter of 8 mm was built by the DFVLR in Göttingen. To obtain interference free data, a series of force measurements were made using the same model and the balance in the 1 x 1 m transonic wind tunnel of the DFVLR /15/.

A problem occured during the measurements because of the unsufficient stiffness of the fairly long support sting for the balance and the model. At certain flow conditions mainly at high incidence and Mach number, the model began to oscillate intensively. It was possible to reduce the oscillation substantially with the help of an oscillation damper build into the model head. The remaining oscillation was small, but may still be seen as causing some uncertainty about the reliability of the test results /16/.

The scatter of the test data from the 1 x 1 m tunnel, as shown in Figure 10, must be related to at least two more problems:

In course of the measurements the model surface was slightly damaged due to some pollution in the airflow. Thus the surface roughness was changed. (During the tests in the TU-Berlin tunnel the model surface remained unharmed).

The initial calibration set for the 3-component balance was not as accurate as necessary. A new calibration set was designed and build for the test series in the TU-Berlin tunnel giving improved accuracy. This included the use of carrier frequency bridge instead of a direct current amplifier.

In general it may be stated, that the results obtained in the TU-Berlin octagon test section with adaptive walls compare fairly well with those obtained in the 1 x 1 m DFVLR transonic tunnel.

Although the TU-Berlin data didn't show any scatter when repeating the test's some doubts were raised about the reliability of the results observed in the wall pressure measurements in the upstream and downstream part of the test section. In order to study this problem wall position measurements were made using a traversing system with a high precision displacement transducer (accuracy ± 0.001 mm). An example of the measured 'plane' wall contour and an adapted wall configuration is shown in Figure 11.

The measurements reveal that in the model area the wall position is correct well within the accuracy of the potentiometric displacement feelers installed in the test section (0.03 mm). However, in the upstream and downstream part considerable deviations occur which may be due to the tolerances of the wall material (± 0.13 mm) or due to some bending of the wall between the jacks.

Although the example shown is the worst obtained for all walls, it indicates, that it may be advisable to use some smoothing or correction to the wall data. It may be necessary to incorporate an extrapolation procedure which extends the wall data measured near the model into upstream and downstream direction, as planned for the 2-D wall adaptation. Anyhow, it has to be

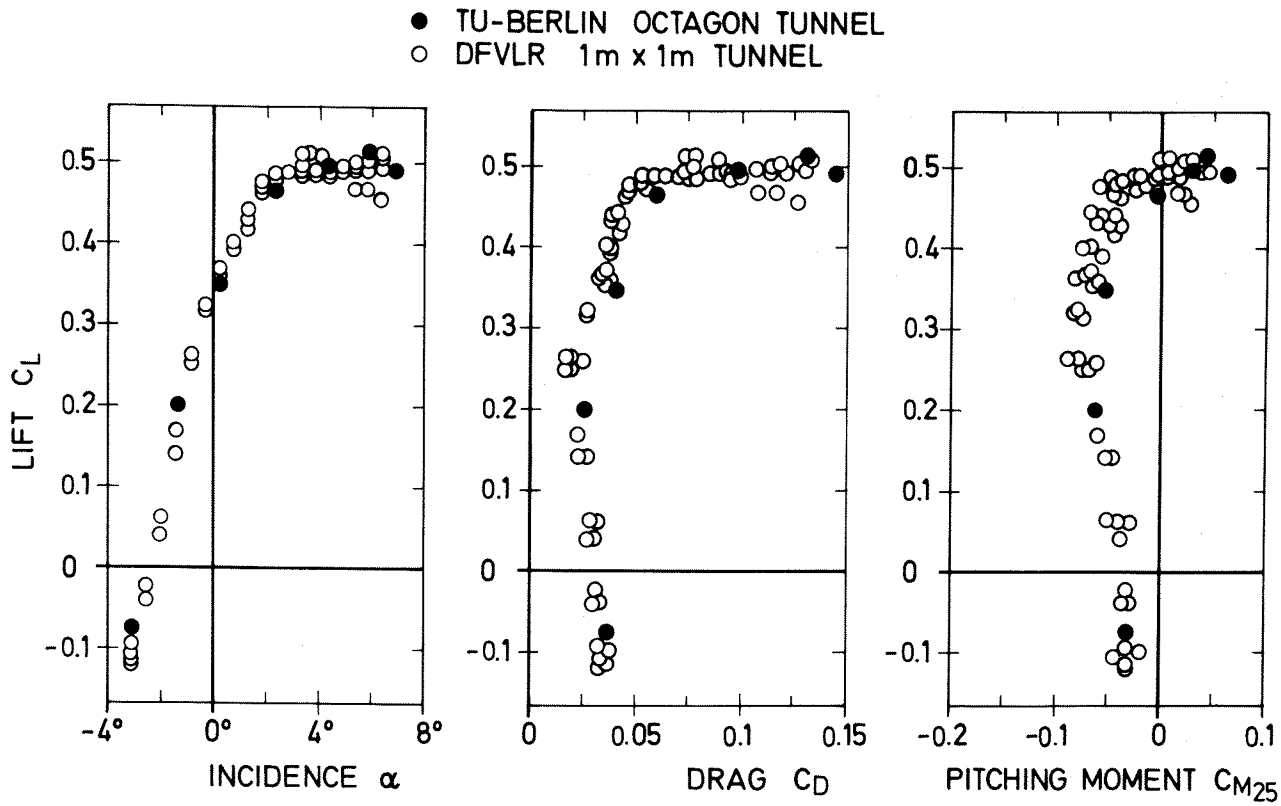


Figure 10: F4 - model, lift, drag and pitching moment, $M_\infty = 0.7$

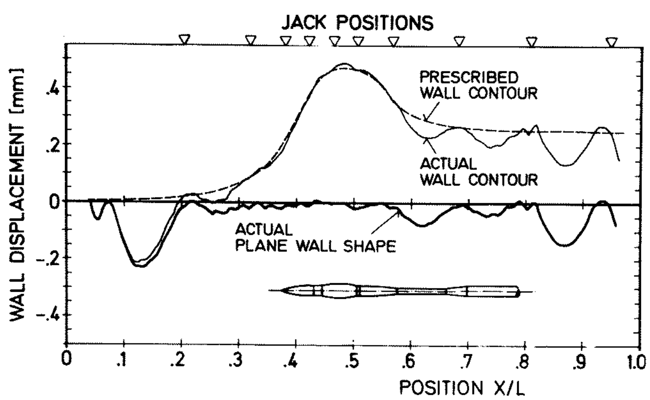


Figure 11: Measured wall contours

pointed out that this is a particular problem of the very small test section where accuracy requirements are much more difficult to handle than in a larger test section.

8. Adaptive wind tunnel walls for supersonic flow

Up till now, not much attention was devoted to the problems of using adaptive walls at supersonic speeds. Only a small scale study is reported by Dowell /17/. He suggest's to construct a compliant flexible wall in such a way that the aerodynamic loads produced by the model would cause the walls to deform naturally - i.e. without the help of computer controlled jacks - to minimize interference. The flow field produced by a flat plate 'aerofoil' at supersonic speed was used to study the required wall characteristics. The flexible wall was modelled as a hinged plate with a torsional spring at the hinge line. In principle this concept is only conceivable for two-dimensional supersonic flow. But even in this case the dependance of aeroelastic coefficients on (local) Mach number may make the concept impratical.

In the following the implications of using the classical flexible wall concept at supersonic speeds is discussed. Aeroelastic effects are excluded here and the wall shape is assumed to be entirely determined by the position of the jacks.

At subsonic as well as at supersonic flow conditions wall interference is always related to the fact that the flow near the wall is forced into a direction other the one it would take in an unrestricted flow field. While at subsonic speeds any local deviation from the interference-free flow direction will have a global effect, at supersonic speeds the effect is first of all a local one with communication only downstream along Mach lines or shock waves.

An explanation of the physical phenomenon at supersonic speeds is usually given by referring to an oblique shock wave impinging on a solid wall, Figure 12.

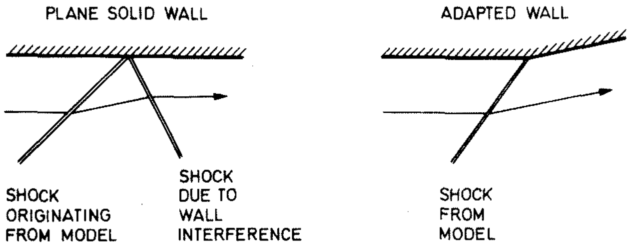


Figure 12: Oblique shock wave impinge on a solid wall

In the case of a plane wall the flow has to change direction again via an additional shock to fullfill the boundary condition at the wall. It looks like the first shock being reflected. In fact, the term 'shock reflection' is commonly used for this phenomenon, but it is misleading, since the first shock ends at the wall. The second shock is only a consequence of the required adjustment of flow direction. If the wall contour would be adapted properly to the flow direction downstream of the shock, there would be no justification for the existance of a second shock.

In the more general case of three-dimensional flow such adaptation of the wall contour is complicated, since three-dimensional models produce shock waves (and other waves) of approximately conical shape. Its intersection with a plane test section wall is a hyperbolic curve, Figure 13.

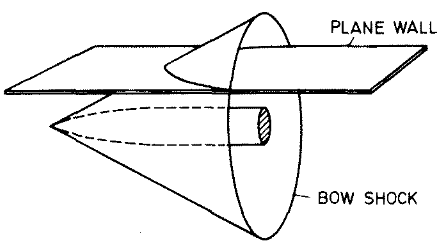


Figure 13: Intersection between conical bow shock and plane wall

Thus, wall adjustment would have to be arranged along such hyperbolic lines in order to achieve interference free flow condition. This is practically not feasible, although the situation in transonic is somewhat relieved by the fact that the intersection curves develope towards a straight line as the supersonic Mach number is reduced towards Mach 1.

On the other hand, the problem could in fact largely be avoided, when the test section could have a circular cross-sectional shape instead of the usual rectangular one, since the intersection of a cone and a cylinder is a straight line. An octagonal cross-section shape, as used for the TU-Berlin adaptive wall test section yields indeed substantial advantage in this respect, Figure 14.

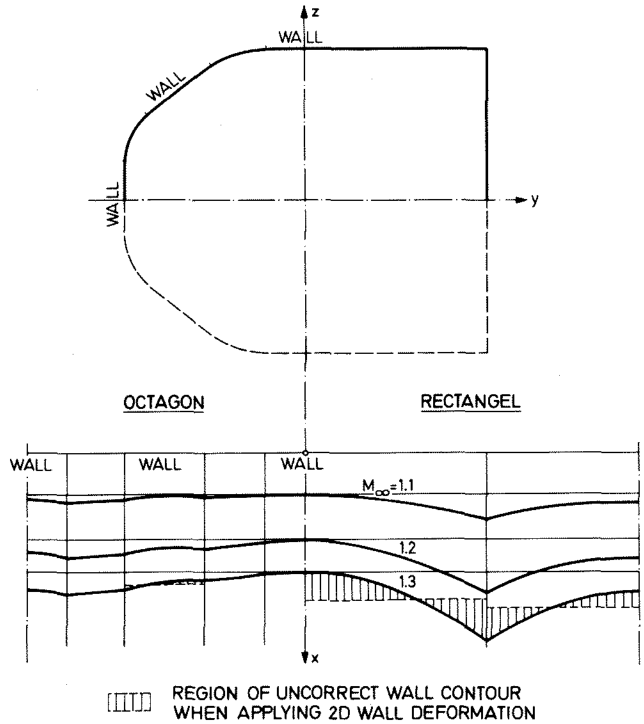


Figure 14: Intersection-lines of model bow shock and wall for different main stream Mach numbers

For an application of the adaptive wall concept to supersonic flow condition some fundamental questions arise: How large are the changes in flow direction in typical flow fields, are the corresponding wall adjustments feasible and how can one determine the required proper wall shape. These questions will be dealt with in the following.

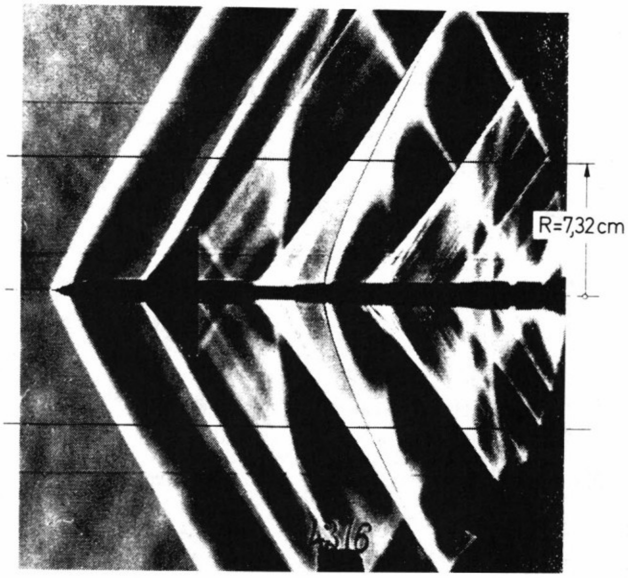
8.1 Examples of measured flow conditions around a wing-body combination at $M_\infty = 1.2$

The Swedish Research Establishment FFA has made a comprehensive study of the flow field around a wing body combination at low supersonic speeds /18/. The contour of the model may be seen in the Schlieren picture, Figure 15.

A 5-hole probe has been used to determine flow direction and static pressure along lines above, lateral of and below the model (φ = 0°, 90° and 180°). The measurements made in a distance of R = 7.32 cm away from the model axis are used here for further analysis. This corresponds to the geometry of the TU-Berlin octagon test section. The model has a length of L = 11,3 cm and gives a blockage ratio of 0.7 %.

Test results are available for an incidence range of α = 0° - 28°. Two typical examples are given in Figure 16.

They show, that due to the bow shock sudden changes in flow direction occur in the order of $\angle\Theta$ = 2°. Larger changes are



Figures 15: Schlieren picture of FFA test model at M_∞ = 1,20 and α = 0°

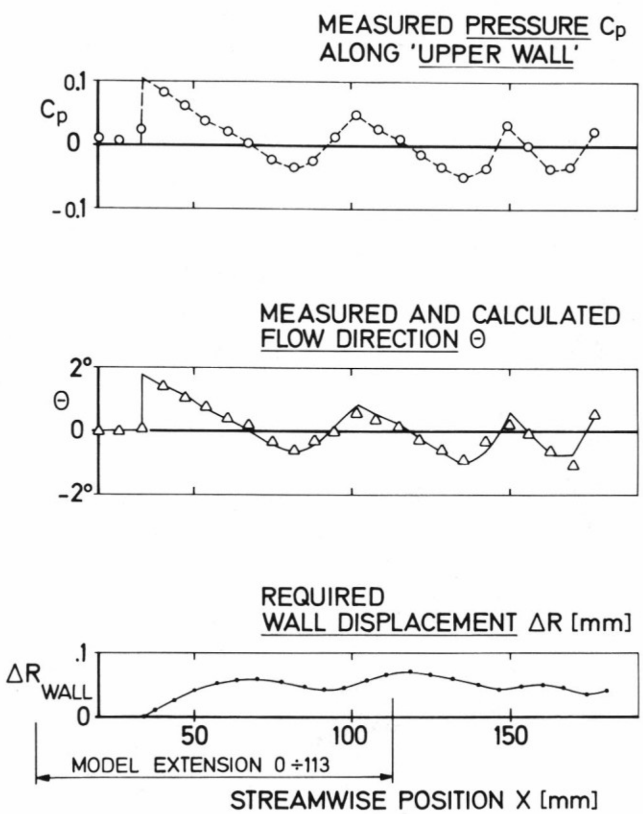


Figure 16 a: Pressure, flow direction and streamline contours in the flow field of a wing-fuselage model at M_∞ = 1,20, α = 0°

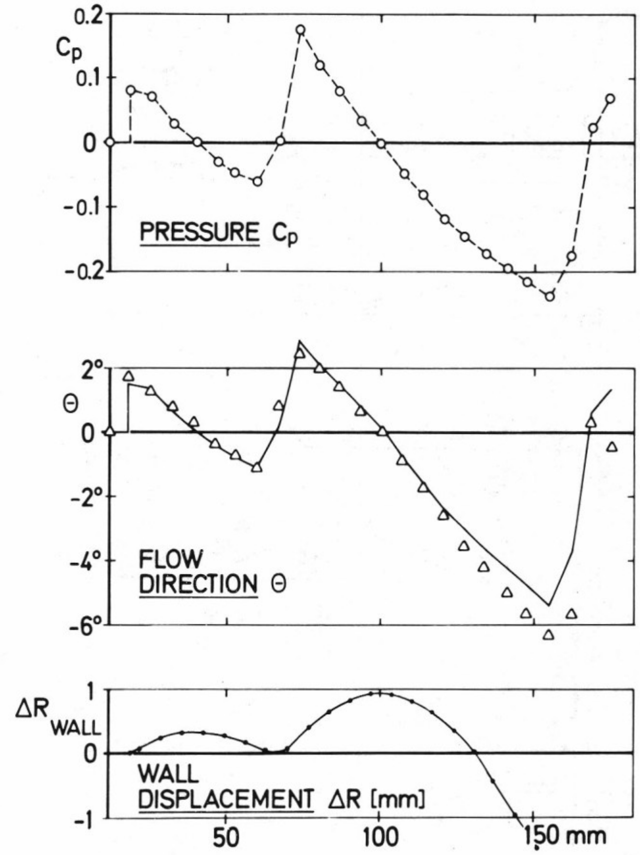


Figure 16 b: Pressure, flow direction and streamline contours in the flow field of a wing-fuselage model at M_∞ = 1,20, α = 27.9°

produced by the wing at high angles of incidence. The information about the flow direction has been used to estimate the wall contour required to avoid wall interference effects. The wall displacement in the middle of two measurement points x_n and x_{n+1} was calculated from

$$R(x_n + \Delta x/2) = \Delta x \left[\frac{1}{2}\tan\theta_1 + \sum_{i=2}^{n} \tan\theta_i\right]$$

with the distance between two measurement points being Δx = 6.8 mm. The results for various angles of incidence are compiled in Figure 17. The estimated wall contours require a wall displacement of typically ΔR = 0.5 - 2.5 mm in a test facility like the TU-Berlin octagon test section. For a test section of ETW size (2.4 x 2.0 m²) the displacements would be of the order of ΔR = 7 ÷ 35 mm.

Note that wall adjustment is in fact only necessary in the region from which Mach lines could still reach the model or influence the model base flow. This is approximately not further than a streamwise position of x = 80 mm.

Thus, in general it may be concluded that the required wall shaping seems feasible although a somewhat larger number of jacks in the vicinity of the model is required as compared to the subsonic case.

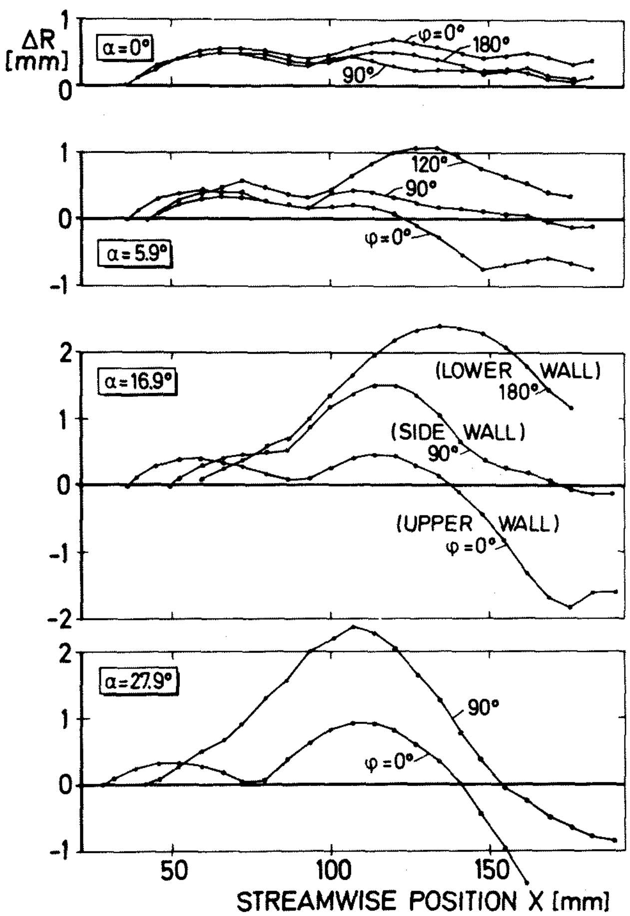


Figure 17: Wall displacement required for interference free flow

8.2 Determination of flow direction from measured pressure distribution

Similar to the procedure at subsonic speeds, wall adaptation at supersonic flow conditions requires a theoretical relation between two independent flow variables such as pressure c_p and flow direction Θ. The existance of shock waves in the flow field suggests the use of the oblique shock relation. For a certain pressure jump by Δc_p the corresponding change in flow direction $\Delta\Theta_s$ is calculated from

$$\Delta\Theta_s = \arctan\left\{2 \cot\sigma \, (M^2 \sin^2 - 1) / \left[M^2 (\varkappa + \cos 2\sigma) + 2\right]\right\}$$

with the shock angle

$$\sigma = \arcsin \sqrt{\left(\frac{\varkappa+1}{2} M^2 \Delta c_p + 1\right)} \, / \, M_\infty$$

For small changes in pressure the wave relation may be used

$$\Delta\Theta = \frac{\sqrt{M^2-1}}{M^2} \, \frac{M_\infty^2}{2 + \varkappa M_\infty^2 c_p} \, \Delta c_p$$

with the local Mach number given by

$$M^2 = \left[\left(\frac{2}{\varkappa-1} + M_\infty^2\right) / \left(1+\frac{\varkappa}{2} M^2 c_p\right)^{\frac{\varkappa-1}{\varkappa}} - \frac{\varkappa}{\varkappa-1}\right]$$

The wave relation has been evaluated for three different main stream Mach numbers M_∞ in Figure 18. In this Figure also the relation between local pressure coefficient and local Mach number is given.

It is important to note that the relation between pressure and flow direction is non-linear and differs for different main stream Mach numbers. That means a certain change in pressure requires different changes in flow direction depending on the local Mach number as well as on the main stream Mach number. Thus, it becomes obvious that ventilated walls of constant porosity, as used for conventional transonic test section, cannot provide the correct boundary condition.

A similar diagramme could be presented by exploring the oblique shock relation for a given change in pressure turns out to be very similar, as may be seen from the following table:

	M_∞=1.1		1.2		1.3	
Δc_p	$\Delta\Theta$	$\Delta\Theta_s$	$\Delta\Theta$	$\Delta\Theta_s$	$\Delta\Theta$	$\Delta\Theta_s$
0.10	1.071	1.079	1.719	1.720	2.204	2.203
0.16	1.334	1.444	2.556	2.564	3.353	3.354
0.20	-	-	3.014	3.036	4.043	4.045

Thus, for practical application it seems acceptable to employ the wave relation throughout the entire flow field.

The test results form FFA have been used to check the applicability of the wave relation again. Part of Figure 18 is shown enlarged in Figure 19 and compared with the experimental data points. Except for two points which were taken very close to a shock, where the probe measurement becomes very uncertain, all data are-within the accuracy claimed-well presented by the wave relation.

The wave equation has been utilized to calculate the flow direction from the measured pressure distribution. The calculated flow direction is compared with the measured one in Figure 16. The finite number of pressure measurement points and in view of the uncertainties inherent in the measurements, the following procedure was applied:

Upstream of the bow shock the pressure coefficient was taken as cp = 0. If there was a pressure reading indicating only a moderate pressure jump, the shock position was assumed at the corresponding pressure tap. The usually negative pressure gradient between the two following readings was extrapolated upstream to obtain the pressure jump across the shock. Otherwise the first high pressure reading is taken as the determining value for Δc_p.

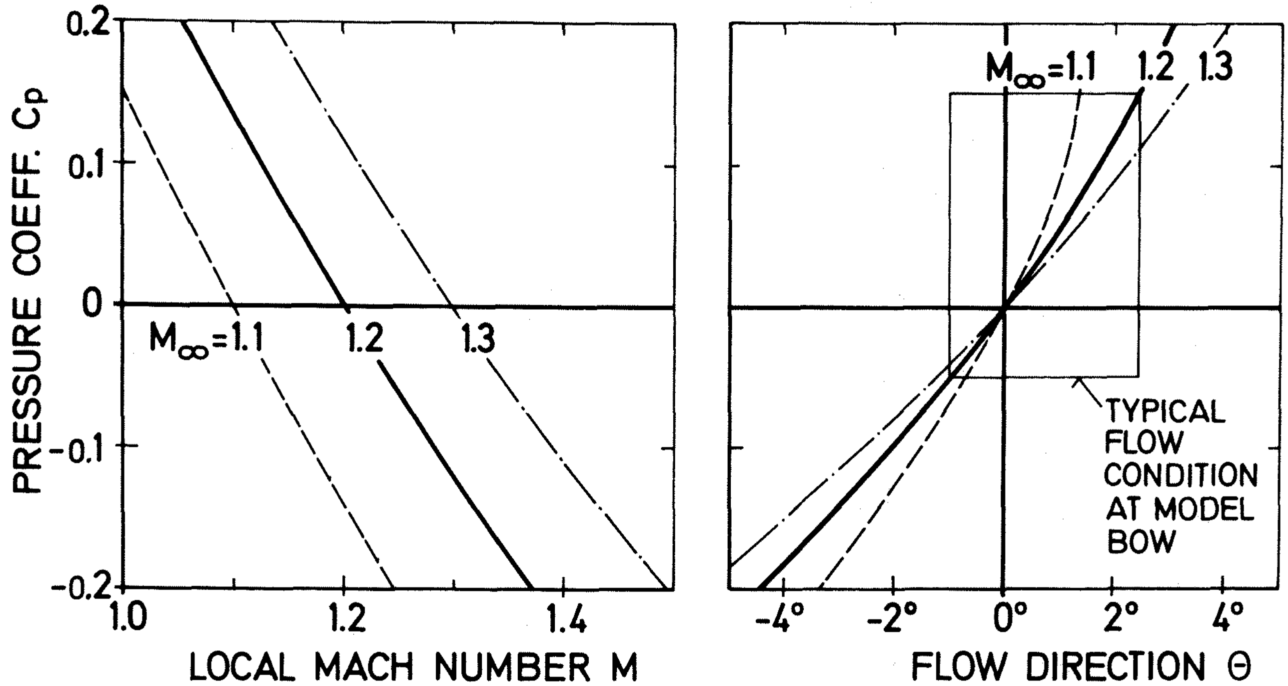


Figure 18: Relation between local pressure, Mach number and flow direction (supersonic wave theory)

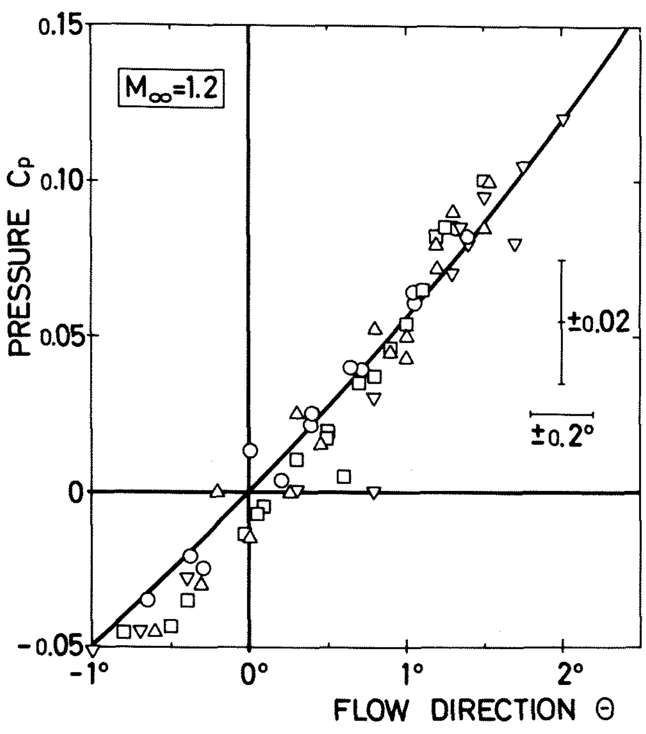


Figure 19: Relation between pressure and flow direction, FFA measurements and wave theory

In fact, for the bow shock the oblique shock relation was employed while downstream of that shock the wave relation was applied. There, the change in flow direction from one measuring station to the next was calculated using a mean value of pressure coefficient c_p and local Mach number M between the two stations. The local flow direction at a point x_n is then simply

$$\theta_n = \Delta\theta_s + \sum_{i=1}^{n-1} \Delta\theta_i$$

The comparison between calculated and measured flow direction, as shown in Figure 16, is representative of what has been obtained for altogether eleven test cases. It demonstrates that the suggested procedure is adequate for determining the flow direction from measured pressure distribution.

8.3 Procedure for adapting the walls at supersonic speeds

In principle, to adapt the walls at supersonic speeds, the very same procedure as exerted for subsonic flow, can be employed. Like for the subsonic case, a unique relationship between two independent flow variables (pressure and flow direction) exists. The pressure distribution along the test section wall can be measured and wall contour data are taken to calculate in a fictitious external flow field the theoretical pressures for comparison. If there is a difference between measured and calculated pressure distribution, the mean value between the two will be used to calculate the wall shape which would produce this mean pressure distribution in the exterior flow.

The great advantage of the supersonic flow condition, by the way, is the fact, that no integration is required, but only the local conditions need to be considered. Thus, the wall adaptation may be done by proceeding stepwise downstream.

8.4 On the use of solid adaptive walls to form a laval nozzle

The availability of a fairly long test section with adaptive walls suggests the use of the jack system in the upstream part of the test section to form the supersonic part of a laval nozzle.

Since the octagon test section comes close to an axially symmetric shape, an approximate method, as suggested by Foelsch /19/, may be used to estimate the required wall displacements.

Wall contours have been calculated for the TU-Berlin octagon test section using this method for the Mach numbers M_∞ = 1.1, 1.2 and 1.3. The result is shown in Figure 20. The amount of wall displacement required for an ETW-sized test section is obtained by multiplying with a factor of 15. For such a test section max. displacement would be in the order of 40 mm.

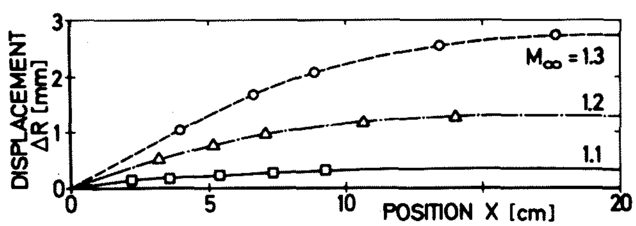


Figure 20: Contours of axially symmetric laval nozzles (TU-Berlin test section, throat radius 8.5 cm)

9. Conclusion

The adaptive wall technique can be considered as well established for aerofoil testing and ready for general practical application. Notwithstanding, for the TU-Berlin test section some improvement of test accuracy beyond the present state are seen to be attainable by imposing slight modifications to the current design and refinements in the adaptation procedure.

The success in achieving fairly interference free flow condition for a 3-D model in a test section with only two flexible walls indicates that there is a fair chance to use this comparativly simple test section design for general tests of 3-D lifting configurations at transonic speeds.

The results obtained in a test section with eight flexible walls using a body of revolution and a wing-body combination demonstrate the general feasibility of the adaptive wall technique for 3-D model testing at high subsonic speeds. Sketches are presented of a large scale test section design to indicate how the octagon concept could be employed for a cryogenic production test facility such as the ETW.

The applicability of the adaptive wall concept to supersonic flow condition is shown to be feasible. Simple supersonic wave theory may be used to derive the wall shape nesessary to eliminate wall interferences from the measured wall pressure distribution.

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