

ON THE AERODYNAMICS OF HYPERSONIC CRUISE VEHICLES AT OFF-DESIGN CONDITIONS

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Abstract

A survey is presented concerning aerodynamic research work carried out at the Technical University of Berlin:

The experimental work dealt with the lee-side flow of delta wings with sharp leading edges at supersonic speeds and various angles of attack and with the flow field of two waverider-configurations (Nonweiler and Jones) at subsonic and low supersonic speeds.

The theoretical work was concerned with shock induced separations on the leeside of delta wings, with the flow along the lower surface of a Nonweiler wing with attached shock and with the flow around bodies of general shape under conditions justifying the slender-body approximation.

I. Introduction

About two decades ago - at the end of the 50's - considerable effort started at various west-european research institutions to explore the possibilities of hypersonic transport. (1) Fundamental analysis of some performance aspects indicated that hypersonic transport at Mach numbers greater than 5 might be feasible and compatible with subsonic conveyance. (2,3) The hypothetical hypersonic aircraft was thought as being a waverider, i.e. a body producing lift by a shock wave, and having ramjet propulsion - possibly with supersonic combustion of liquid hydrogen. It has been argued that the performance of such an aircraft would benefit from the fact that the propulsion efficiency increases with Mach number while aerodynamic efficiency shows only a modest decrease. So it was expected that the combined aerodynamic and propulsion efficiency - which determines the range of an aircraft - would be of the same order as that for subsonic airplanes. (See Fig 1).

Very high cruising speed thus seemed to be possible without a range penalty. This was a strong argument in favour of hypersonic transport.

Looking at this argument today one has to admit that the advances in subsonic aircraft design have led to very economic configurations with range factors nearly twice as high as one could expect for waverider aircraft. Therefore the prospects of hypersonic transport are considerably less bright today than they appeared to be twenty years ago. In any case, there seem to remain two arguments to justify research activities on hypersonic configurations. The first argument is that space shuttle vehicles of a second generation might require more sophisticated aerodynamic design. A second argument comes from some space research programmes, which revealed the desirability of low flying satellites (at altitudes of 120 km or even lower). In such cases one might

consider an aerodynamically flying vehicle as an alternative to the pure satellite.

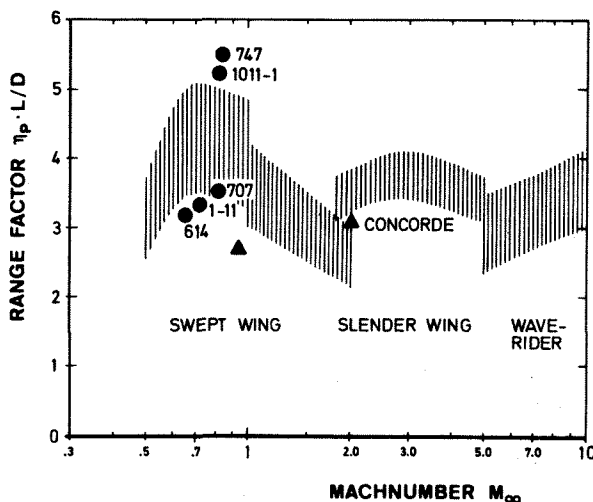


Figure 1. Combined propulsive & aerod. efficiencies of various types of aircraft

The research programmes carried out at the Technical University of Berlin were aimed at providing some fundamental information about the aerodynamics of hypersonic cruise configurations. All configurations considered were delta wings with sharp leading edges. Their flow fields had been studied at supersonic main-stream velocities and also to some extent at subsonic conditions.

II. Experimental Work

The experimental studies dealt with, on the one hand, detailed flow surveys on very simple delta wings, and on the other hand, general aerodynamic properties of somewhat more sophisticated shapes waveriders.

Leeside flow:

The flow field on the leeside of a delta-wing at supersonic main stream conditions has been the subject of a detailed experimental investigation. (4) The majority of windtunnel tests were made using the model sketched in Figure 2.

This model has a flat upper surface and a triangular cross-section. Leading edge sweep angle is $\Lambda = 73^\circ$. For correlating the experimental results, components of Mach number and angle of attack have been used as they appear in a plane normal to the leading edge. It was expected that the flow field for similar delta wings of different sweep angle could be correlated by the use of these normal components.

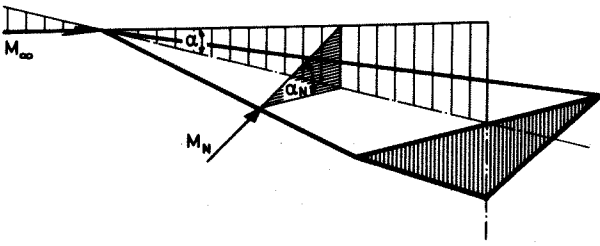


Figure 2. Delta wing with triangular cross-section and flat upper surface

A summary of the results for the leeside flow structure is presented in Figure 3.

In this diagram the Mach number $M_N = 1$ is expected to be of particular importance as it divides flow regions with subsonic and supersonic leading edges. Stanbrook and Squire found that a boundary exists very near $M_N = 1$, depending slightly on angle of attack, dividing two distinct flow regions, one showing leading edge separation rolling up into spiral vortices and another, attached flow at the leading edge followed by a shock induced separation. (5) The flow structures anticipated by Stanbrook and Squire correspond to those sketched in the diagram to the left and far to the right of the Stanbrook-Squire boundary.

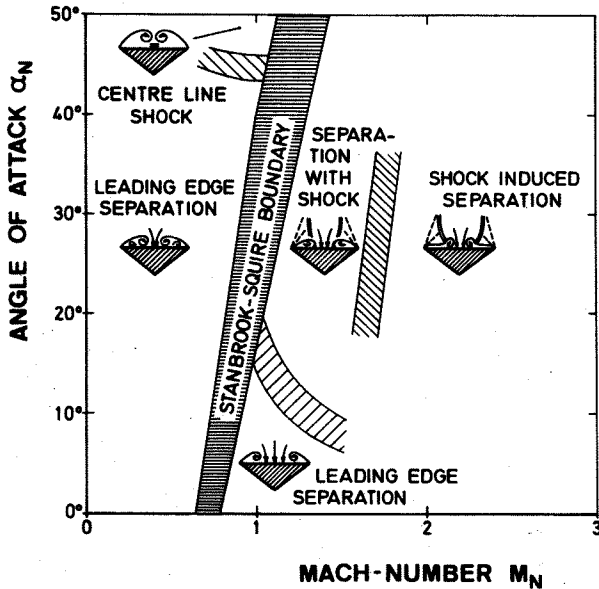


Figure 3. Leeside flow types for delta wings Stanbrook-Squire and Szodrach boundaries

The aim of our research programme was first to study the change from one type of flow to the other when crossing the Stanbrook-Squire boundary. It has been found that the change of flow type was a continuous process and the most convenient way to determine the boundary was by observing the disappearance of the secondary vortex in the leading edge separation. Our experiments also revealed however, that shock induced separation occurs only at much higher Mach number and that another type of flow exists just on

the right hand side of the Stanbrook-Squire boundary, this flow is characterised by a separation bubble starting at the leading edge and a shock on the top of the bubble. This shock does not cause the separation, but is a more or less independent part of the outer flow field terminating a local supersonic flow region.

The classification into just two types of flow provided by the Stanbrook-Squire boundary has also undergone revision as a result of detailed studies at various angles of attack. In the case of subsonic leading edges, an increase in angle of attack causes the pair of primary leading edge vortices to approach the plane of symmetry. The secondary vortices disappear as the intensity of the primary vortices is reduced due to interference. At high angles of attack a shock is found in the plane of symmetry nearly parallel to the upper surface.

In the case of supersonic leading edges experimental observation leads to the conclusion that the shock on top of the separation disappears at very small angles of attack. This confirms the assumption that the separation bubble is not due to shock-boundary layer interaction and the original labeling of this type of flow as "shock induced separation" is incorrect.

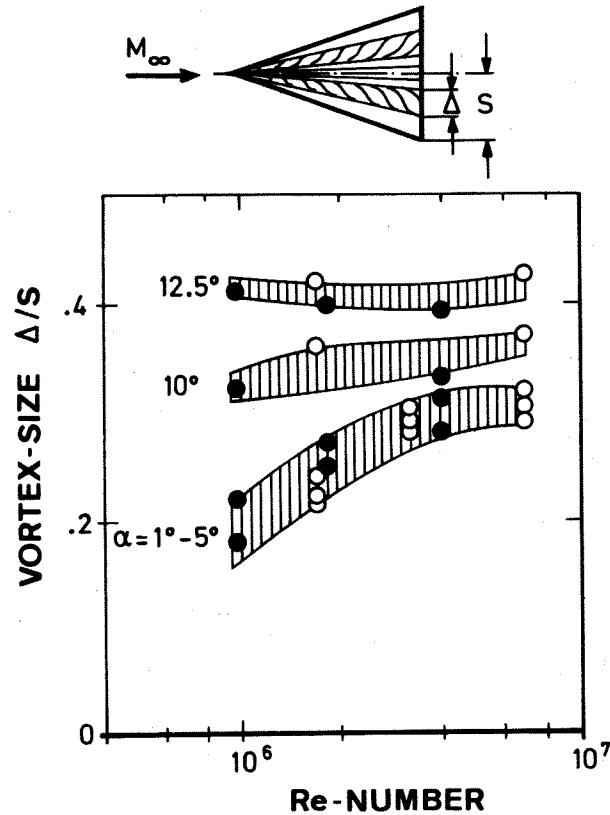


Figure 4. Variation of leading edge vortex size with Reynolds number $M_\infty = 2.5$

Increasing the angle of attack causes the pair of separation bubbles to extend toward the plane of symmetry and finally to merge. Above the separation, shock waves exist and within the separation bubble vortex-like flow can be seen. No fundamental change of the flow type has been observed when angle of attack is increased further.

The conclusion that might be drawn from this work is that an adequate description of the leeside flow field of thick delta wings at supersonic speeds requires more than two flow models as suggested by the Stanbrook-Squire boundary.

The results presented thus far can be taken as a general guideline for determining the types of flow that might be expected for delta wings at various supersonic mainstream conditions. However some restrictions that might limit the general validity of the results must be discussed. In particular Reynolds number effects, nonconical flow, and variation of the upper and lower side wing-shape must be considered.

Reynolds number effects have been studied by using two windtunnel models of different length and two wind-tunnels of different stagnation pressure. (6) The mainstream conditions were chosen such that a leading edge vortex and separation bubble with embedded shock did occur. In other words the intention was that two types of flow were to be investigated, one to the left of the Stanbrook-Squire boundary, and the other to its right.

For the leading edge vortex flow, Reynolds number affects the size (and position) of the vortices when incidence is small. (See Fig 4). The growth in vortex size is connected with increased suction on the upper surface of the wing. At higher incidences Reynolds number influence is weak. In any case the type of flow remains basically unchanged.

The influence of Reynolds number on the flow with separation bubble and embedded shock is different. Although Figure 5 shows very similar effects on the extension of the separation, the suction peak on the upper surface of the wing has been found to increase with Reynolds number for all angles of attack. Schlieren pictures revealed that the separation flattens when Reynolds number is increased, finally resulting in a change in flow type: full expansion, around the leading edge occurs followed by a shock which induces separation further inboard.

As far as the assumption of conical flow is concerned, it can be said that for most of the tested cases this assumption was found to be justified. However deviation from conical flow conditions have been observed for very small and very high angles of attack near the apex and near the trailing edge.

The cross-sectional shape of the lower side of the delta wing has found to be of influence on the lee-side-flow, particularly when the Mach number is comparatively low. (4) The influence depends on the distance between stagnation line on the lower surface and the leading edge. (See Fig 6). At low Mach numbers, when leading edge separation exists, only one stagnation line is found in the plane of symmetry. With increasing Mach number an additional pair of stagnation lines moves toward the leading edges. These stagnation lines finally reach the leading

edges, when at high enough Mach number the bow shock on the lower side of the wing attaches to the leading edges and the leeside flow becomes entirely independent of lower side shape.

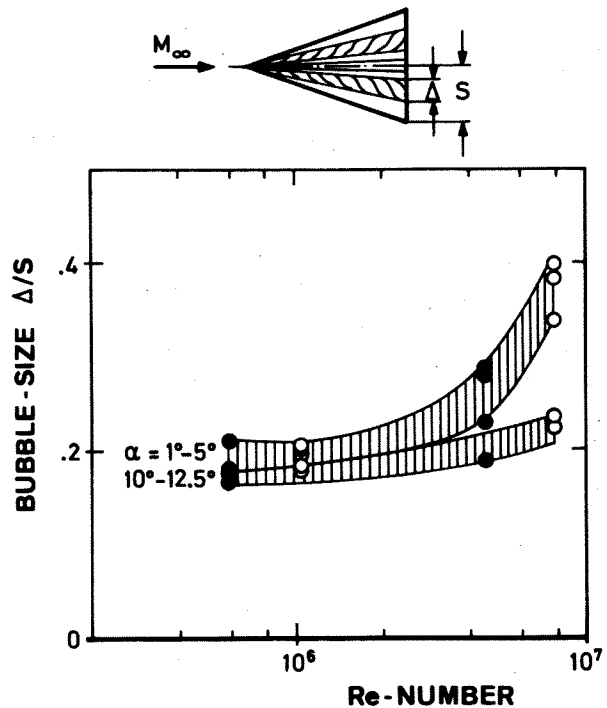


Figure 5. Variation of separation bubble size with Reynolds number $M_\infty = 3,5$

The influence of cross-sectional shape of the upper side of the wing has been studied in detail for one particular configuration: The delta wing shown in Figure 2 has been used with the delta-cross-section shape as the upper side. The experimental results have been compared with those for the flat upper surface. (7) The main difference compared to the flow types obtained for the flat leeside wing surface is that for the delta-shaped upper surface no separation bubble with shock is found on the right hand side of the Stanbrook-Squire boundary, but instead shock induced separation as sketched in Figure 7. In addition at high Mach numbers the leeside flow was found to be strongly non-conical.

Nonweiler Waverider:

The experimental work on a Nonweiler Waverider configuration was not so much aimed at providing details about the flow structure, but at gaining some information about the aerodynamic efficiency of such a waverider under off-design conditions in subsonic and low supersonic flow and at providing pressure distributions for comparison with theoretical results.

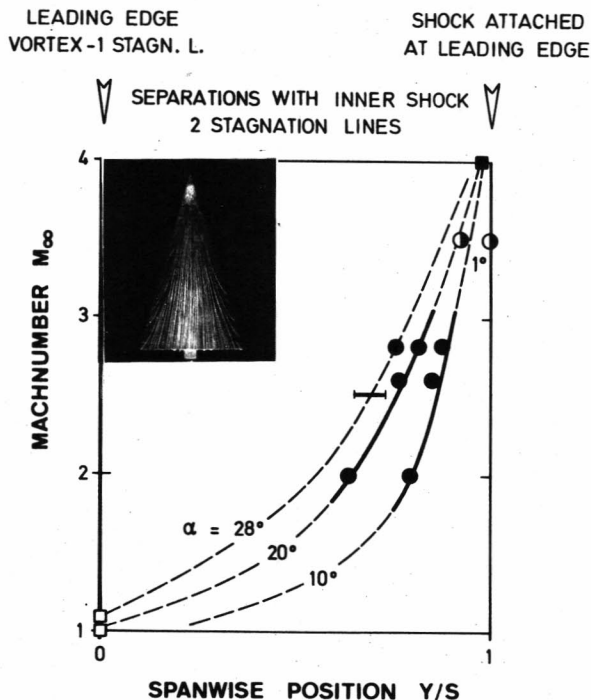


Figure 6. Position of stagnation lines on the (flat) lower surface of a delta wing

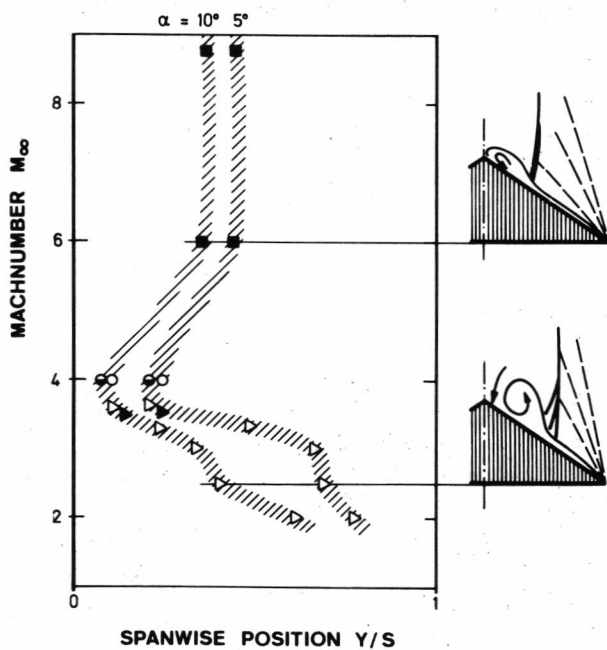


Figure 7. Movement of primary vortex separation line with Mach number

The Nonweiler wing is the simplest type of waverider, having straight leading edges and ridge lines, and between these, plane surfaces. (See Fig 8.) At its design condition this waverider produces a plane shock wave attached to the leading edges and between this shock and the lower surface the flow is parallel to the lower ridge line. The flow condition corresponds to that of a flow around a two-dimensional wedge of angle θ .

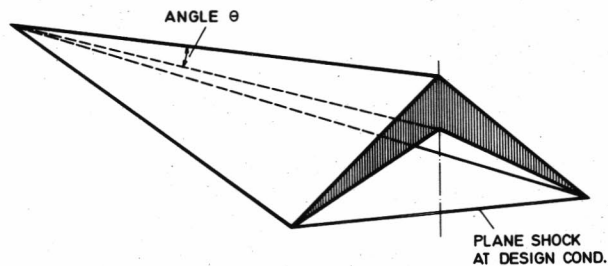


Figure 8. Nonweiler waverider

If the Mach number is reduced below the design value (or angle of attack is increased) the shock wave bulges and finally separates from the wing. Thus flow around the leading edges occurs and separations are formed, rolling up into spiral vortex sheets. The vortices account for low pressure on the upper surface of the wing producing a considerable amount of additional lift and thus improving the lift-to-drag ratio. (See Fig 9). At subsonic speeds the low pressure due to the leading edge vortices provides roughly half of the total lift in this particular case.

The values for lift and drag have been calculated from measured pressure distributions. The pressure measurements along the surface of the Nonweiler wing revealed remarkable non-conical effects for subsonic and low supersonic free stream conditions. In particular strong upstream influences of base flow are shown to exist especially for the lower surface of the wing. Thus model mounting in the windtunnel becomes critical as it effects the surface pressure via the base flow. (8)

Some of the results obtained by surface pressure measurement will be presented later as a comparison with theoretical results.

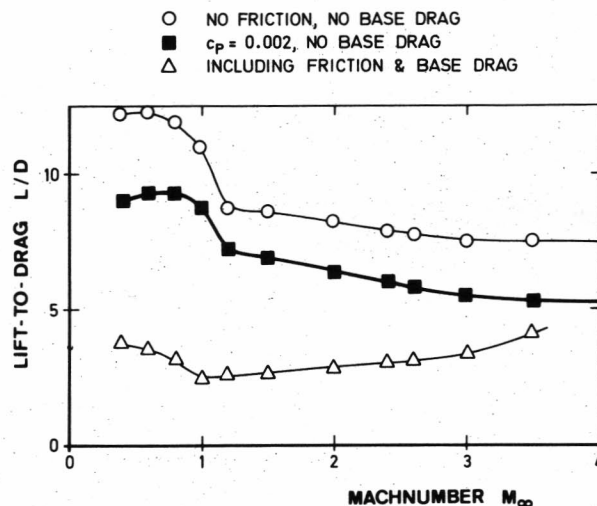


Figure 9. Lift-to-drag-ratio of a Nonweiler wing

$$M_{\text{design}} = 7 \quad s/l = 0,3 \quad \tau = 0,08 \quad \alpha = 0^\circ$$

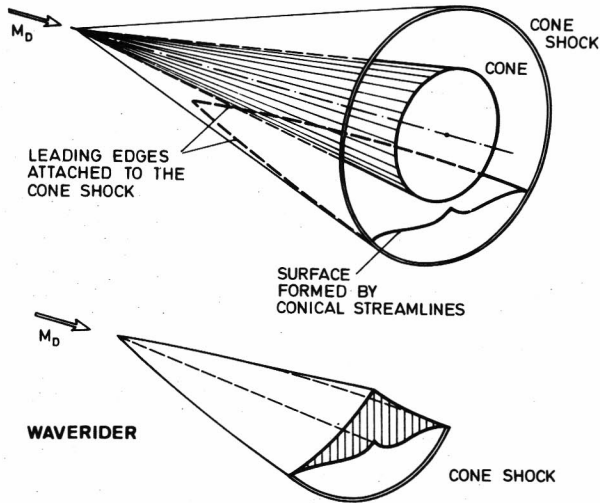


Figure 10. Design of a Jones Waverider

Jones Waverider:

While the Nonweiler waverider was designed to produce at the lower surface a flow equivalent to a wedge flow, i.e. a parallel flow following a plane shock, the Jones waverider is based on a cone flow. (See Fig 10). At the design condition, the shock wave is conical and attached to the leading edges, and the pressure along the curved lower surface is not constant. For practical application of the waverider concept to a hypersonic cruise configuration the Jones wing is much more realistic, because of reduced dihedral and more favourable volume distribution as compared to the Nonweiler wing. At Messerschmidt Boelkow Blohm (MBB) the applicability of the Jones wing to a hypersonic transport configuration has been demonstrated. (See Fig 11). (9)

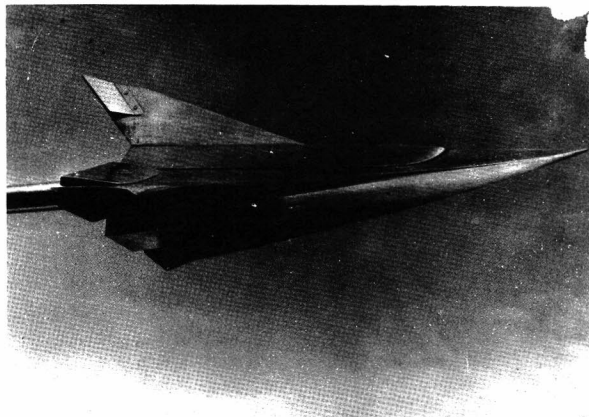
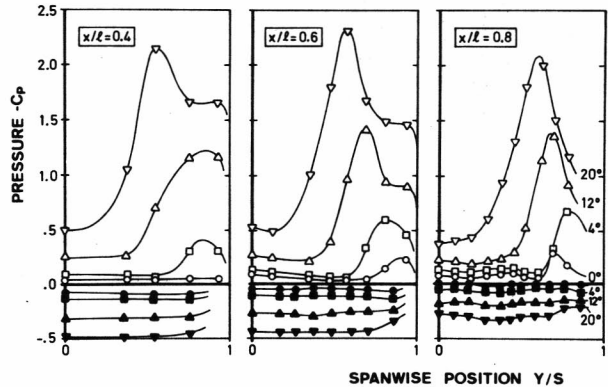


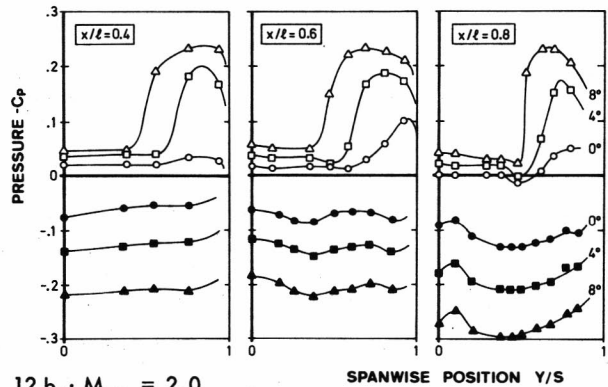
Figure 11. Model of a hypersonic transport MBB-Design $M_D = 7$

The flow field of the Jones waverider at off-design condition is more complex, because it is basically non-conical. In particular, when the upper surface is designed to produce an expansion equivalent to a Prandtl-Meyer flow, this surface will also be curved and generally no similarity will exist between cross-section shapes at various chord-

wise positions. Accordingly the flow structure varies with chordwise position as can be seen from some measured pressure distributions presented in Figure 12. Such surface pressure measurements have been made for various subsonic and supersonic free stream conditions in order to provide data for comparison with theoretical results. Such comparison will be presented later.



12 a.: $M_\infty = .5$



12 b.: $M_\infty = 2.0$

Figure 12. Jones Waverider
Pressure distributions for three cross-sections x/l
Various angles of attack

III. Theoretical Work

Shock induced separation on the leeside

An attempt has been made to calculate the flow field on the leeside of the delta wing in the particular case of shock induced separation. (4) The flow model used for the calculation has been derived from the experimental results at high Mach numbers. It is sketched in Figure 13. The main stream is seen to undergo an expansion after passing the bow shock and the leading edge. This expansion causes a flow-deflection toward the plane of symmetry. It is assumed that the restoration to a flow direction parallel to the plane of symmetry takes place across an inner shock and that this shock leads to a thickening of the boundary layer. In the calculation empirical relations have been used for determining the shape and position of the bow shock as well as for the boundary layer thickness distribution. Then an iterative procedure leads to the determination of the inner shock position and shape. The calculated

pressure distributions are in fair agreement with experimental results for moderate angles of attack. (See Fig 14). At high incidence the agreement is poor. The comparison also indicates that the assumption made for the boundary layer near the leading edge and in the plane of symmetry seems to be somewhat inaccurate. This is confirmed by Figure 15, which shows the results of a flow field calculation compared with measured Pitot-isobars. On the other hand this figure reveals that the calculation provides a fairly accurate description of the inner shock and of the flow structure just upstream and downstream of the shock.

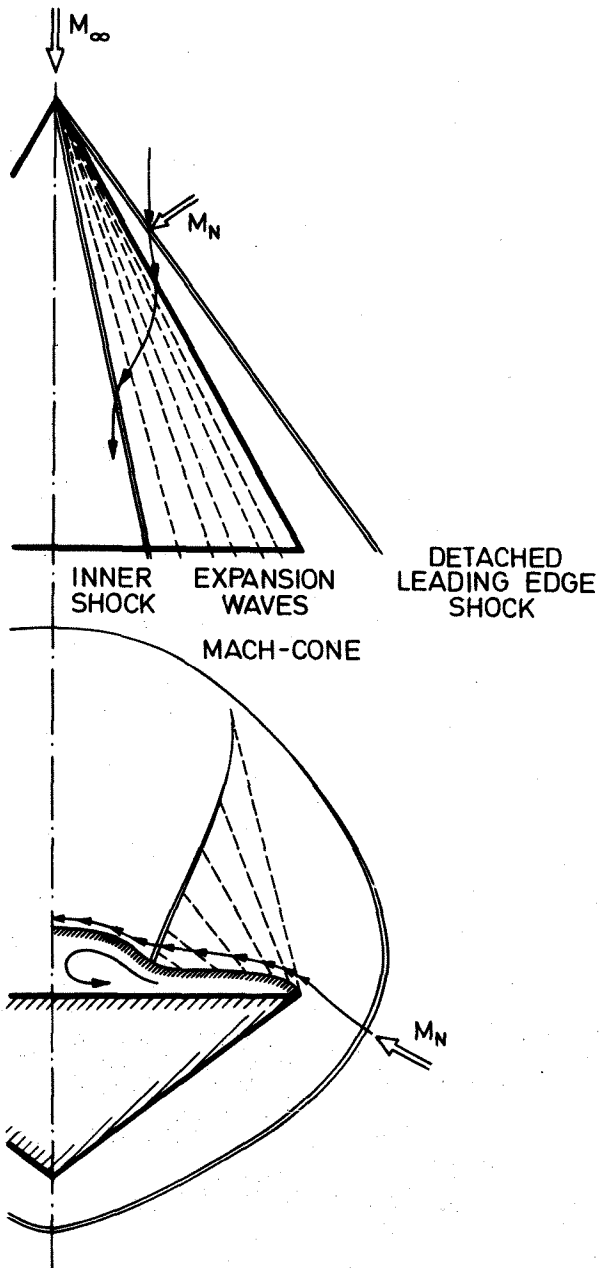


Figure 13. Flow model for calculation of shock induced separation

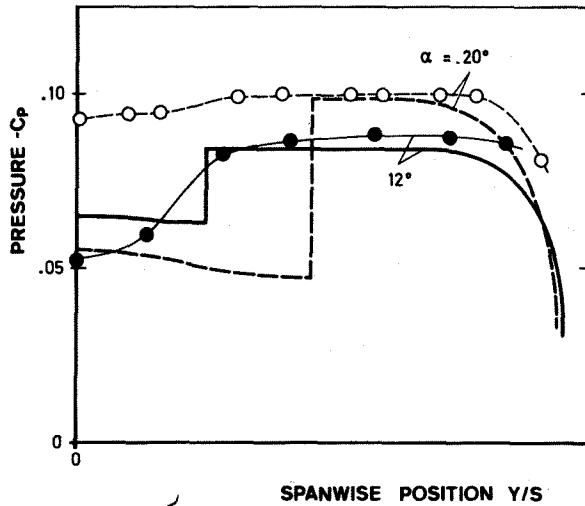


Figure 14. Pressure distribution for shock induced separation; Theory & experiment; $M_\infty = 3,6$

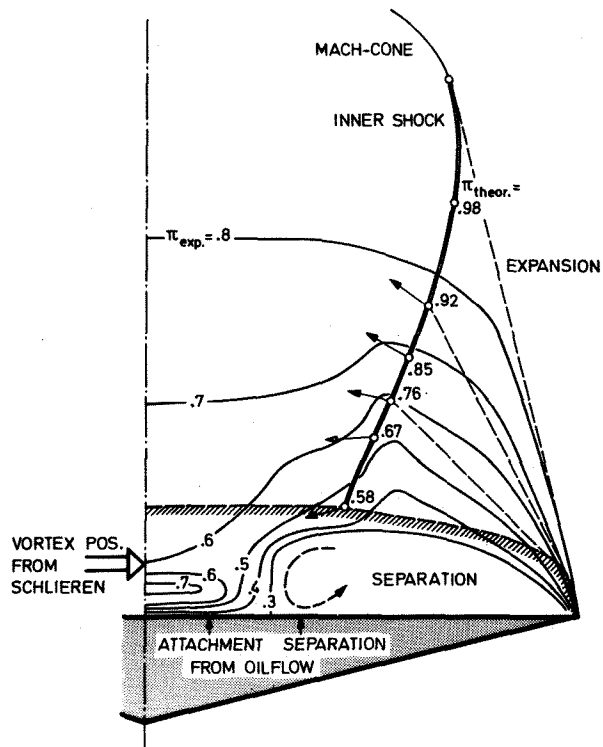


Figure 15. Calculated shock and measured Pitot-isobars for shock induced separation (Exper. Isobars: Monnerie & Werlé, delta wing: $\Lambda = 75^\circ$ $M_\infty = 4,0$ $\alpha = 10^\circ$)

Nonweiler wing with attached shock on the lower side

If for the Nonweiler wing the free stream Mach number is just slightly below the design value (or angle of attack is just above) the shock wave bulges with a plane portion attached to the leading edges. (See Fig 16). Near the leading edges the flow is still parallel with constant pressure. Within the Mach cone the flow direction varies while the flow expands towards the lower ridge line. Such a flow is amenable to an exact but inviscid calculation. (10)

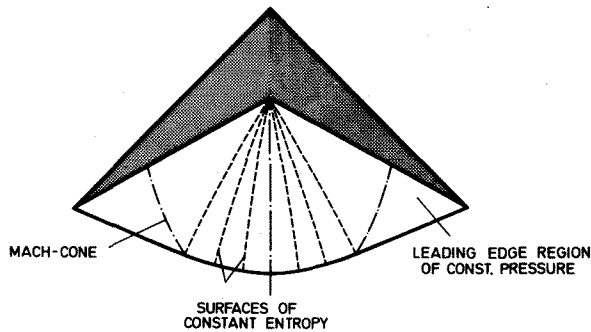


Figure 16. Flow-structure on the bottom side of a Nonweiler wing at $M_\infty \approx M_{Design}$

The solution for the leading edge region is straightforward by the use of oblique shock relations. For the central region within the Mach cone a finite-difference solution was necessary. Starting with a zeroth-order solution chosen in such a way that the boundary conditions are satisfied, an iterative process leads to the exact solution. As an example a calculated pressure distribution is shown in Figure 17. Although the Nonweiler waverider is of fairly restricted practical importance, such results are of interest for testing approximate solutions.

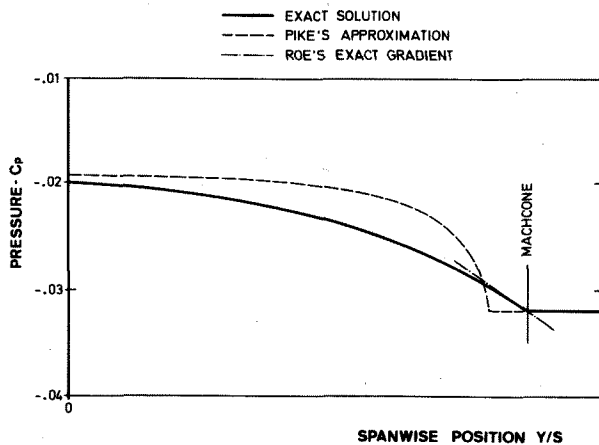


Figure 17. Calculated pressure on the lower surface of a Nonweiler wing ($M_{DES} = 7,0$ $s/l = .3$ $\tau = .08$ $M_\infty = 4,5$ $\alpha = 2,5^\circ$)

Slender bodies of general shape

For subsonic and moderate supersonic speeds an attempt has been made to calculate the flow field for bodies of general shape, such as the Jones waverider. The assumption was made that the body had to be slender, so that Slender-Body-Theory could be used, but otherwise the body could have sharp leading edges and finite thickness. Following a suggestion of Hummel (11) two-dimensional singularities have been distributed over the surface contour of the body in the cross-sections perpendicular to the main stream direction. (See Fig 18). In this way the method is applicable to slender bodies of any cross-sectional and planform shape. For bodies with sharp leading edges, how-

ever, leading edge separation occurs even for small angle of attack. Such separations are not accounted for in the theory so that one has to expect that calculated pressure distributions will show considerable discrepancies when compared with experimental results on the upper surface of the wing near the leading edge.

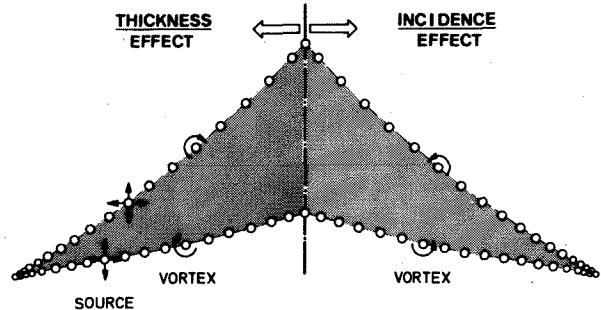


Figure 18. Singularity distribution on the surface of a slender body in a cross-section $x = \text{const.}$

Although the computer program was developed for general shapes, it was first tested for a Nonweiler wing. Figure 19 shows that theory compares fairly well with experimental results except in regions of leading edge separation. For supersonic free stream conditions the theory predicts nearly no change in pressure distribution for various cross-sections while for the subsonic case a remarkable decrease in pressure level with increasing downstream position is calculated. This compares fairly well with experimental results (See Fig 20). The figure also reveals the influence of base flow which causes a decreased pressure level particularly for the lower surface. Therefore pressure measurements in this case can only be considered free of base flow influences when made at least 30 % of chord upstream of the base.

Theoretical results obtained for the Jones waverider are shown in Figure 21. They again compare fairly well with experimental results not only for subsonic free stream conditions but also for the supersonic Mach number. The weak point remains the leading edge separation region. As the comparison is made for the cross-section at 60 % chord position, the experimental results are assumed to be free of base-pressure influence.

The chordwise variation of pressure distribution for the Jones waverider is shown in Figure 22. While for the Nonweiler wing the pressure level is nearly constant at supersonic free stream conditions, for the Jones waverider theory predicts variation of pressure level on the lower surface due to the basically non-conical shape of the body.

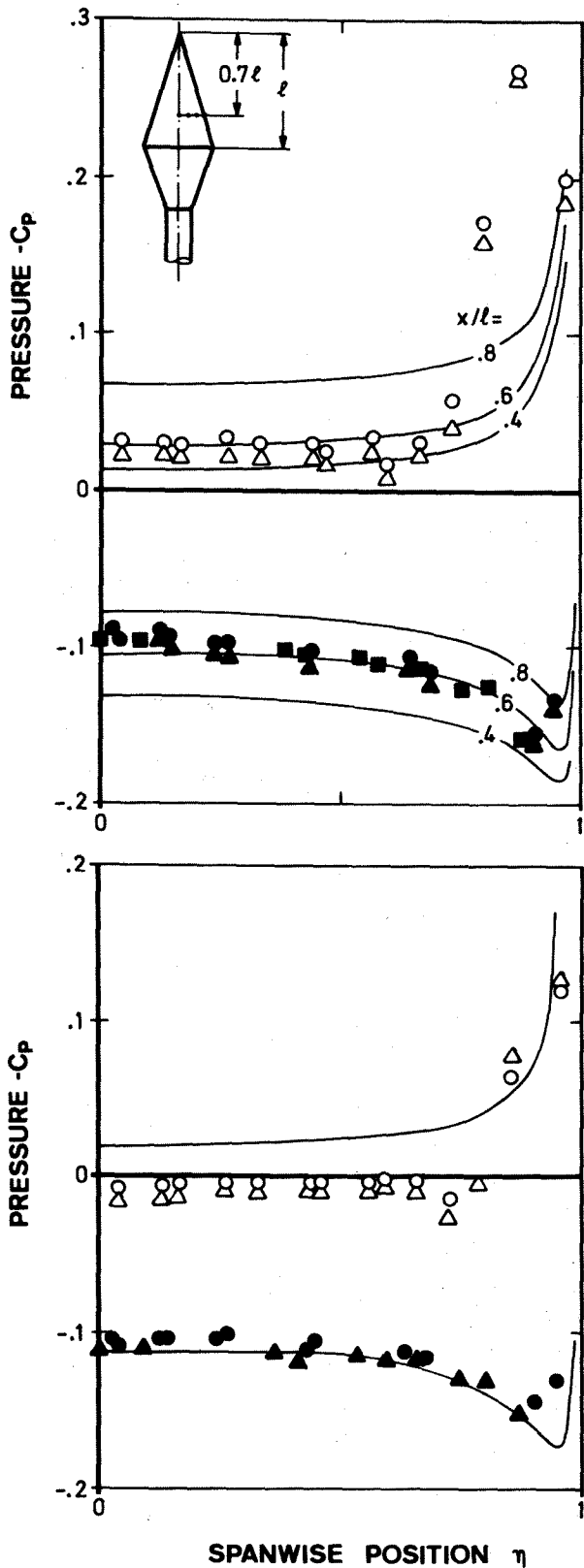


Figure 19. Pressure distribution for a Nonweiler wing
Theory & experiment
above: $M_\infty = .7$ $\alpha = 0^\circ$
below: $M_\infty = 2.2$ $\alpha = 0^\circ$

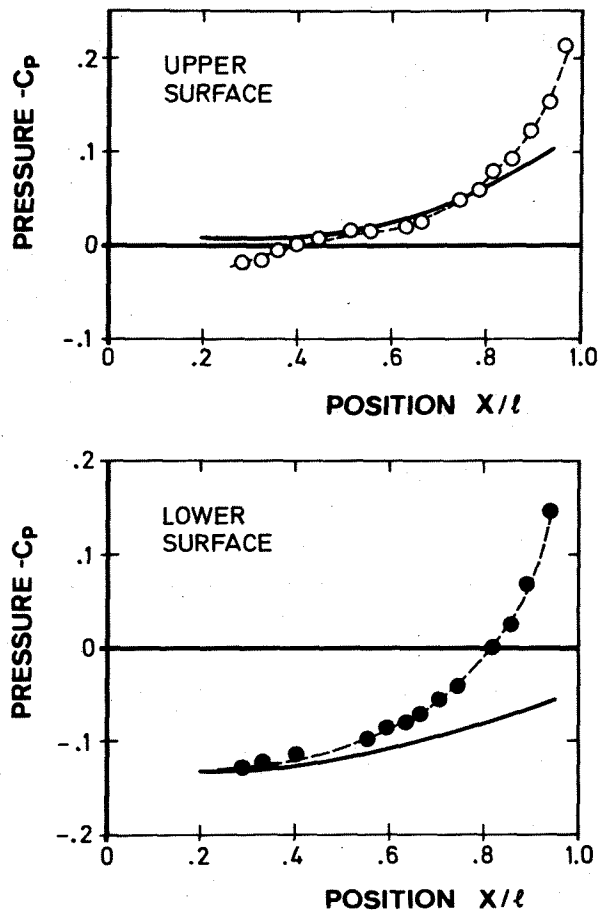


Figure 20. Pressure distribution along the ridge line of a Nonweiler wing
Theory & experiment $M_\infty = 0.8$

Conclusions

The results of our experimental and theoretical studies provide some information about selected aerodynamic problems of hypersonic cruise configurations with the emphasis on two topics:

For the leeside flow of delta wings at supersonic speeds various flow types have been studied fairly comprehensively. However, all results refer to wings with straight leading edges. It is intended to extend the research programme by investigating the interference of two types of flow produced by an adequate change of leading edge sweep between apex and trailing edge.

For our theoretical work the Slender-Body-Theory has been found to be an appropriate starting point for describing the flow around waveriders of finite thickness and arbitrary non-conical crosssections, using singularities distributed on the surface. However, for wings having sharp leading edges, flow-separations occur. It is intended to extend the theory in order to include the effects of these leading edge separations.

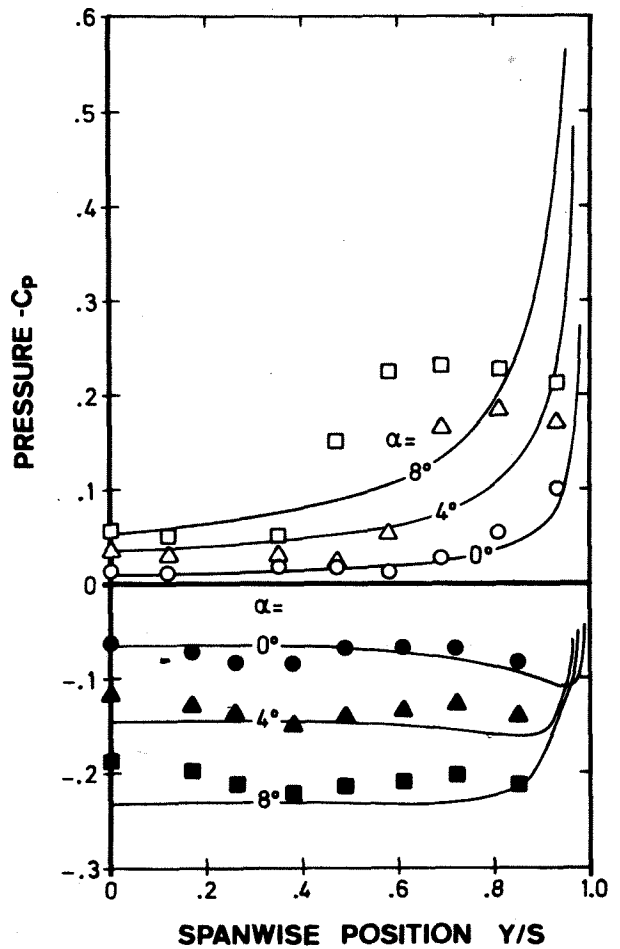
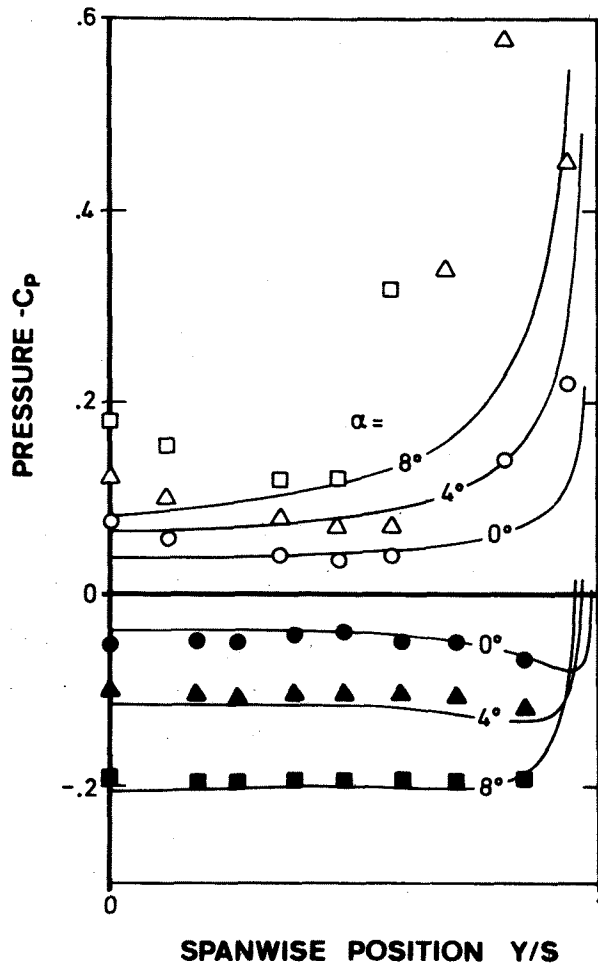


Figure 21. Jones waverider; Theory & experiment; Effects of incidence

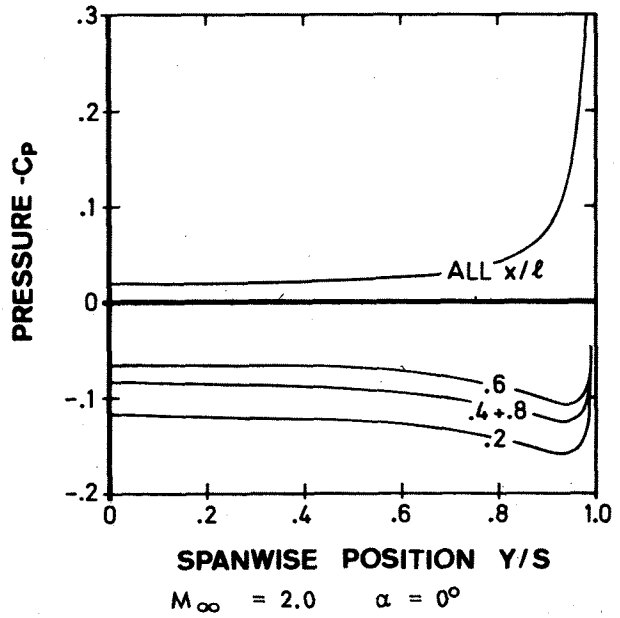
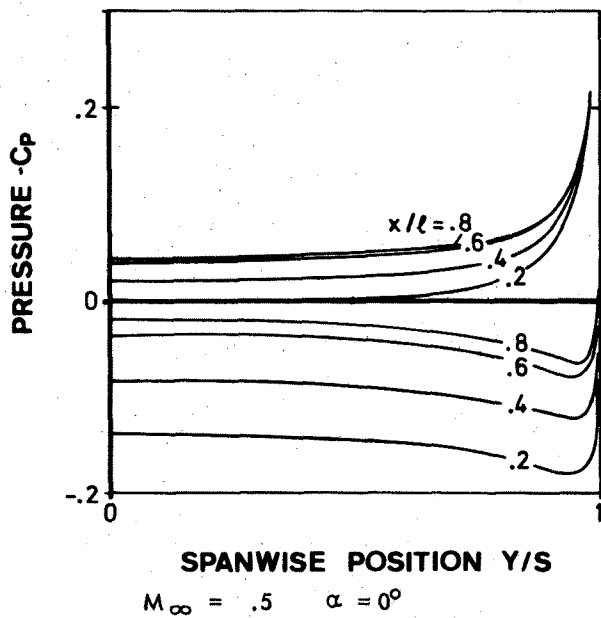


Figure 22. Jones waverider theory; Pressure distribution for various chordwise pos. x/l

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