INVESTIGATION OF THE UNSTEADY AIRLOADS ON A TRANSPORT AIRCRAFT TYPE AIRFOIL WITH TWO INTERCHANGEABLE OSCILLATING TRAILING EDGE FLAPS, AT TRANSONIC SPEED AND HIGH REYNOLDSNUMBERS*

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Abstract

Motivated by the intention to explore the potential of active control technology for transport aircraft wings, investigations in unsteady aerodynamics were performed as this seemed to be the most critical element in the new technology. Wind tunnel tests using a large two-dimensional model with two interchangeable fast moving trailing edge control surfaces of different relative chord have been carried out. They served as means to evaluate prediction methods for unsteady viscous subsonic and transonic flows about airfoils and to gain insight into those flow phenomena relevant to the design of active control functions.

I. Introduction

Reducing flight operation costs is the dominant motivation directing research and development activities for future transport aircraft. A considerable contribution to cost savings will be achieved by reducing the structural weight of an airplane. A large portion of the overall structural weight is concentrated in the airplane's wing, because this is a severely loaded and thus heavy component. The structural weight of the wing is mainly determined by gust and maneuver loads as well as flutter margin and fatigue requirements. These phenomena are governed by the unsteady aerodynamic loads impacting the wing. Thus unsteady aerodynamic forces and moments affect transport mission costs through their impact on structural weight. The active control of aerodynamic loads employing counteracting control surfaces to reduce wing loads is a promising way to improve the economy of flight operations. Critical elements in such an active control system are the aerodynamic control surfaces. They still have to fulfil properly their primary functions of flight path control, whilst additionally serving as elements of a control loop designed for effective load alleviation. This poses very demanding requirements on the aerodynamic capabilities of the control surfaces. They must provide rapid changes of aerodynamic forces without inadmissable lag. It follows that an insight into the unsteady aerodynamics of control surfaces is a prerequisite for the development of active control technology.

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To explore the potential of active control technology a research and technology programme has been developed, which includes appropriate theoretical and experimental investigations in unsteady aerodynamics. One of these programmes which was performed jointly by the French and German aerospace research institutions ONERA and DFVLR, and MBB provided the basis for the following investigations.

II. Test equipment concept and description

Experimental investigations in wind tunnels generally offer the most promising

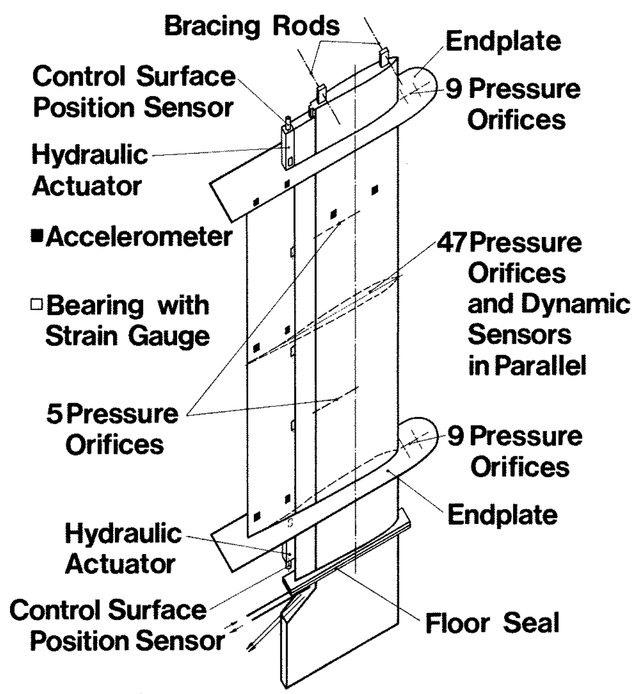

Fig. 1: Model geometry and sensor locations

methods to gain insight into the complex phenomena of flows about wings with oscillating control surfaces. Of great consequence for the transfer of wind tunnel results to full scale conditions is the mismatch existing between the Reynoldsnumber range of wind tunnels available and Reynoldsnumbers associated with large transport aircraft. It is impossible to simulate viscous flow about a wing on a subscale model because growth rates of boundary layer properties are different at different Reynoldsnumbers.

Fast movements of control surfaces will add considerably to the complexity of the flow. Pressure disturbances generated by the flap motion propagate in the flow field at the speed of sound which is of the same magnitude as the local velocity. Thus unsteady aerodynamics presents its own variety of complex flow conditions but there are strong interactions with the structure of the steady flow field.

Systematic investigations of the transonic flow around oscillating airfoils were performed by NLR. H. Tijdeman [1] has presented a comprehensive survey and very remarkable findings. The Reynoldsnumbers in these investigations were not representative for transport aircraft wings. Experimental studies on oscillating airfoils at transonic speed by S.S. Davis and G.N. Malcolm [2] cover considerably higher Reynoldsnumbers but were restricted to airfoils in pitching motion. Though offering very useful insight,the resources available seemed inadequate for their use as a data base for the design of active control systems for wings with fast moving control surfaces. Thus more specific investigations were initiated including extensive wind tunnel tests.

As wind tunnels offering variation of the Reynoldsnumber were not available, a very large two-dimensional model was preferred to a halfwing model. This permitted Reynoldsnumbers not too far from full scale conditions and well above about 10 million. Below this limit, changes in Reynoldsnumber cause considerable shifts in the location of laminar-turbulent transition and consequently changes in shock position and trailing edge pressure. Avoiding this critical régime is especially important for many advanced airfoil designs. They incorporate flat Machnumber distributions on the upper surface and high rear-loading. This makes them very susceptive to scaling effects.

For the design of the model, the experience of J. Cahill, S.C. Treon and W.R. Hofstetter [3] with a similar model but for steady flow proved very useful. The model was manufactured from two solid aluminium halfshells providing maximum stiffness and high eigenfrequencies. The chord measured 1 m (40 '') and the span 2 m (80'') between endplates, Fig. 1. The profile geometry was representative of modern airfoils with 12.5 percent maximum thickness, a rather flat upper surface and considerable aft cambering. Two exchangeable trailing edge flaps were available. A control surface of 30 percent chord ratio represents the mean chord ratio of a typical outboard aileron. A control surface of 12.5 percent chord ratio represents the mean chord ratio of the last element of a trailing edge high lift system. In the center section 47 pressure orifices and 47 dynamic pressure transducers in identical chordwise positions were installed. Pressure orifices in two further parallel sections and on the endplates were available to check spanwise pressure distributions.

The facility used for the low speed tests was the 3 m x 3 m (10 ft x 10 ft) low speed tunnel of the DFVLR in Goettingen, which provides a free stream Machnumber $M_\infty = 0.2$ and Reynoldsnumbers with the chord as a reference length of about $Re_c = 3.7$ millions. For these tests very large endplates (2.5 m x 2 m, about 8 1/4 ft x 6 1/2 ft) were installed. Fig. 2 shows the mounting of the model in the tunnel. Using DFVLR equipment, mean and unsteady pressure distributions were measured from both sources - pressure orifices and transducers. Test parameters included reduced frequencies (with chord c as a reference length) from zero to more than one and combinations of angles of attack and flap deflection causing trailing edge separation.

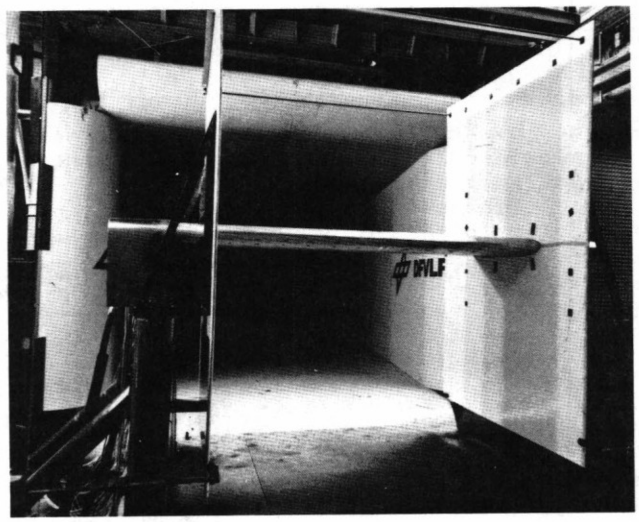

Fig. 2: Model in the DFVLR Goettingen low speed tunnel

The high speed tests were performed in the atmospheric ONERA S1 transonic tunnel in a circular test section with a diameter of 8 m (about 26 1/2 ft). Fig. 3 presents the model mounting in the test section. Maximum Machnumbers were $M_\infty = 0.85$; the corresponding Reynoldsnumbers about 12.5 million. In these tests ONERA data acquisition equipment was used, measuring steady pressure distributions through the pressure orifices and unsteady pressure distributions by transducers. The endplates in these tests were not very large (0.55 m x 1.45 m (22'' x 58'') rectangular, except for a forward semi-circular part). The endplates increase the effective aspect ratio from 2 to about 3. However they do not provide real two-dimensional flow.

Even on a wing segment mounted between tunnel walls of a closed test section the flow will be disturbed by a variety of three-dimensional effects like wall boundary layers, intersecting boundary layers, corner vortices , corner shocks and blockage effects. The advantage of this arrangement is, that all aforementioned three-dimensional disturbances are of comparatively small order. The disadvantage of small endplates is a flow which is not purely two-dimensional . But unlike boundary layer and corner shock phenomena, which are difficult to treat theoretically, the effect of endplates can be described by potential theory methods.

Thus the assumption has to be made that in the middle section of the model, pressure distributions can be generated which are practically identical with the pressure distributions in undisturbed two-dimensional flow, generally at slightly different free stream Machnumbers and angles of attack. This is already a proven procedure applied e.g. for comparisons of wind tunnel test results with computations based on ideal two-dimensional conditions.

No boundary layer trips were used in these tests.

III. Subsonic Pressure Distributions

The investigations in the subsonic flow régime aimed at gaining insight into the prevailing flow phenomena, examining viscosity effects on unsteady aerodynamic forces for high angles of attack and flap deflection and checking the validity of the numerical prediction method available.

For the numerical calculations the panel method of W. Geissler (4) for incompressible flow was used. This is a three-dimensional singularity prediction method for an arbitrary wing planform and thickness distribution applying non-linear boundary conditions and the non-linearized Bernoulli equation. This method simultaneously predicts the steady and unsteady pressure distributions for a given wing with control surfaces at incidence performing harmonic oscillations.

For the present calculations a rectangular wing with an aspect ratio of 6 was prescribed to simulate three dimensional effects. Furthermore the boundary layer thickness of the center section has been calculated and added to the airfoil thickness. A comparison of pressure distributions derived from computations and wind tunnel tests at low incidence are exhibited in Fig. 4. The chordwise distribution of the steady pressure coefficient c_p is typical for supercritical airfoils with extensive aft-cambering and blunt leading edges at low speed. Adding the boundary layer displacement thickness to the airfoil geometry provides improved agreement between theoretical and experimental results, except for the region of aft-cambering. There the contour pressure is considerably influenced by the development of the wake, which is not taken into account in the theory.

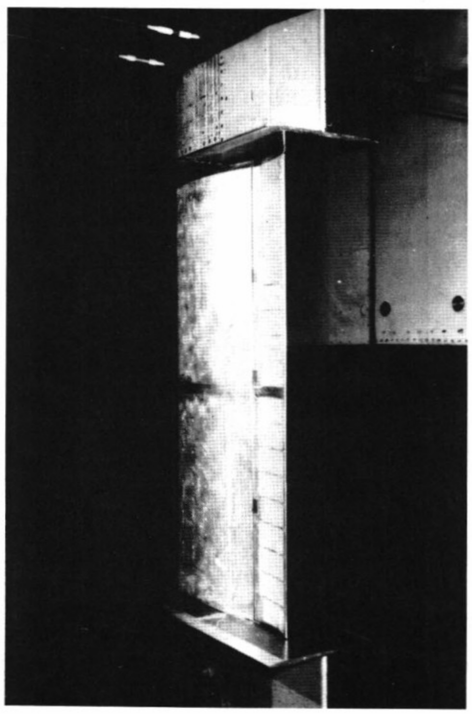

Fig. 3: Model in the ONERA S1 transonic tunnel

The unsteady distribution is presented as the real (c_p') and imaginary part (c_p'') of the unsteady pressure coefficient $\bar{c}_p=c_p'+ic_p''$. The present theory properly predicts details of the unsteady chordwise pressure distribution in the region of the suction peak and the rear-loading which are neglected by linear theories. Differences between theory and experiment can be mainly attributed to viscosity effects which are only partially simulated by the present theory. Adding the boundary layer displacement thickness of the steady flow to the airfoil thickness provides improved agreement especially forward of the flap.

Fig. 5 presents comparisons for high angles of attack and flap deflection and with a separation region on the upper surface of the flap. Discrepancies between theoretical and experimental pressure distributions indicate the influence of the boundary layer. If, at any particular point on the contour, viscosity has a noticeable influence on the steady pressure distribution it also affects the local unsteady pressure distribution with the same order of magnitude. The areas surrounded by c_p' (x) and c_p'' (x) represent the in-phase and out-of-phase portions of the unsteady lift coefficient respectively. Viscosity effects on the upper surface forward of the flap considerably reduce the in-phase portion but increase the out-of-phase portion. Thus the modulus of the unsteady lift coefficient is roughly the same for viscous and inviscous flow, but there is a considerable lead in the phase caused by viscosity. Taking this into account. the steady flow displacement thickness noticeably reduces the discrepancies between theoretical and experimental pressure distributions for steady flow but is considerably less effective in the unsteady case.

Separation, as appearing here near the trailing edge, locally fully restructures the unsteady pressure distribution. Apparently control surfaces are still effective in generating unsteady airloads when trailing edge separation is present, but separation has a strong influence on the phase angle.

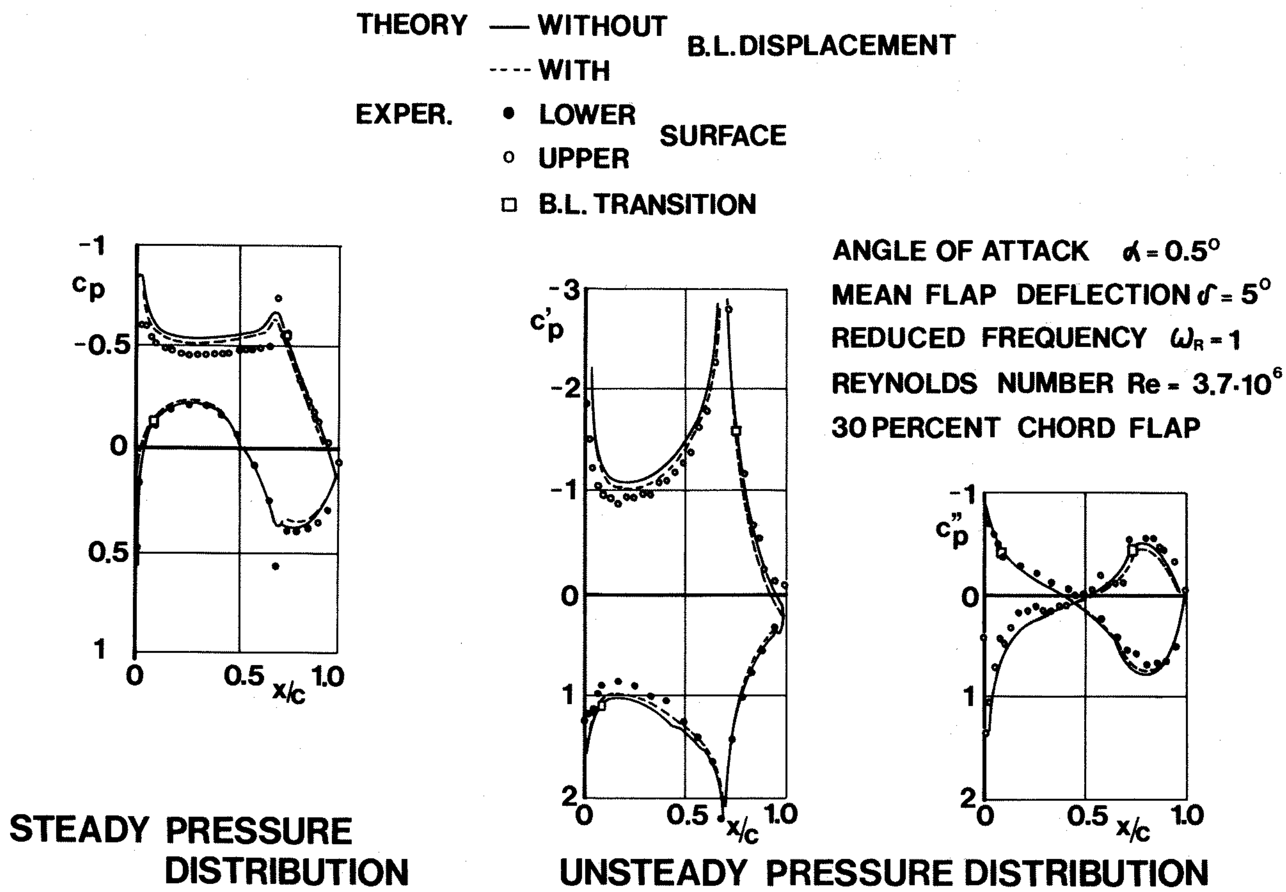

Fig. 4: Comparison of theoretical (panel method) and experimental pressure distributions for incompressible flow and a low angle of attack

Subsonic unsteady aerodynamics of supercritical airfoils which are characterized by high suction peaks at high angles of attack and extensive rear loading is very susceptive to viscous effects. These predominantly cause considerable phase shifts of the unsteady airloads. Trailing edge separation on the control surface does not necessarily decrease the capability of the control surface to generate unsteady aerodynamic loads. The panel method presented here properly predicts features of supercritical airfoils not covered by linear theories, for steady and unsteady pressure distributions not dominantly determined by viscous effects.

IV. Supercritical Pressure Distributions

Predominant practical application of active control technology is expected for the transonic flight régime. Thus the desire to understand and predict unsteady transonic forces initiated the following investigations.

Two numerical methods for the prediction of harmonic contour pressure distributions in transonic flow were examined in connection with these experiments. Both are based on the assumption of transonic small perturbations, though the ONERA finite difference method developed by M. Couston and J.J. Angelini [5] is limited to low frequencies while the DFVLR integral equation method developed by R. Voss [6,7] has no such restrictions. It treats the steady flow independently of the unsteady flow. Thus for unsteady flow a potential equation is derived which is dependent on steady flow variables and linear in relation to the time axis. Applying Green's theorem transforms the differential equation into an equation of line and surface integrals. From this a system of linear equations is derived which after prescribing the boundary conditions is solved to determine the unsteady pressure distribution on the airfoil. The advantage of this method compared with finite difference or element methods is a considerable reduction in computation time.

Outstanding characteristics of the ONERA finite difference method are a careful consideration of the boundary conditions, the mathematical modelling of the wake and the equation for the pressure coefficient. For the numerical solution an alternative direction implicit scheme is used. This potential method has been coupled with an integral method for unsteady boundary layers [8]. Thus steady and unsteady pressure distributions on airfoils with considaration of viscous effects are determined.

Fig. 6 presents a pressure distribution with an extended supersonic region termina-

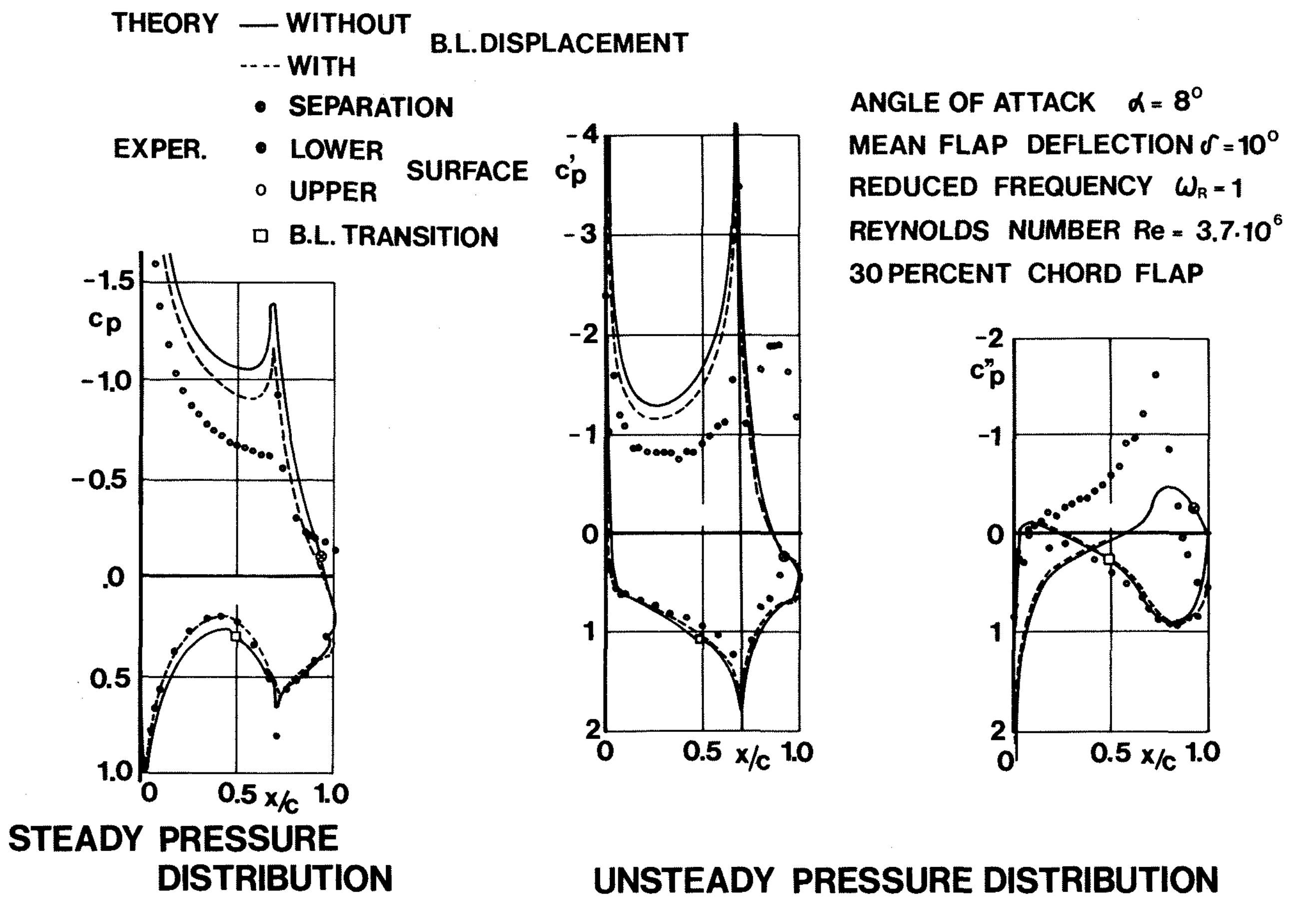

Fig. 5: Comparison of theoretical (panel method) and experimental pressure distributions for incompressible flow and a high angle of attack

ted by a shock which is typical for a supercritical airfoil at higher than design free stream Machnumber. The upper surface pressure distribution is flat up to the shock. Due to this constant chordwise pressure and steep gradients further downstream small changes of any flow parameter cause considerable changes in shock location and strength. This experience was one of the dominant motivators for investigating the influence of control surfaces on such pressure distributions.

The input parameters for the computation by the integral equation method were selected to gain an acceptable match between theoretical and experimental steady pressure distributions. Discrepancies in the rearloading region are unavoidable because viscous effects are dominant there but are not considered by this theory. For the unsteady flow the differences between upper and lower surface pressure distributions per radian of oscillation amplitude Δc_p are presented.

Investigations in subsonic flow revealed that if viscosity has a noticeable effect on the steady pressure distribution it likewise affects the unsteady pressure distribution. In this case in contrast to the discrepancies in the rear-loading region in steady flow there is a good agreement between experiment and theory for the unsteady pressure distribution.

The shock and the supersonic region of the steady flow field dramatically change the structure of disturbance propagation in comparison with subsonic flow represented here by the results from a linear theory. The integral equation method in contrast predicts the influence of compressibility effects on the unsteady pressure distribution fairly well. Acceptable agreement between theory and experiment in the shock region is partly due to selecting a high reduced frequency. Unsteady and especially non-linear contributions to shock location and strength considerably decrease with increasing reduced frequency.

Treating shock boundary layer interactions is not implemented in this theory. The subsonic sublayer of the boundary layer permits disturbances to propagate forward of the shock, giving higher experimental unsteady pressure amplitudes just forward of the shock than the theory can predict.

The integral equation method proved to

be an appropriate tool to predict unsteady pressure distributions on airfoils in transonic flow offering as its main advantage computation times only 10 percent of those of commonly used methods.

For a comparison of steady and unsteady pressure distributions derived from these experiments and the theory including boundary layer effects a case for a supercritical flow but nearly zero lift was selected ($\alpha=-1^{\circ}$, 30 percent chord flap, upward flap deflection $\delta=-1.75^{\circ}$). This was done because three dimensional effects tend to decrease with approaching zero lift.

By systematical variation of the freestream Machnumber a theoretical steady pressure distribution was found which fairly matches the selected experimental pressure distribution, Fig. 7. The flow on the upper surface is subsonic. Considering the boundary layer moves the theoretical shock position on the lower surface slightly downstream and - what is more important for a comparison of unsteady pressure distributions - closer to the experimental shock position. The unsteady pressure distribution is presented by the absolute value of the pressure coefficient $|c_p|$ and the phase angle φ. Considering boundary layer effects in the theory offers better agreement with experiments though the same is not generally true for the steady pressure distribution.

The shock on the lower surface is not very strong but has a predominant influence on the unsteady pressure distribution. Implementing the unsteady boundary layer in the computations offers better agreement in shock location. This is a prerequisite for obtaining similarity in the theoretical and experimental pressure distributions particularly forward of the shock.

The two computational methods presented in comparison with the experiments demonstrate their potential to predict unsteady transonic pressure distributions on airfoils. The advantage of the integral equation method is its low computing time while the finite difference method in combination with the unsteady boundary-layer code offers a more extensive mathematical modelling of the flow phenomena involved.

	EXPER.	THEORY
M_∞ =	0.78	0.75
α =	3.8	1.15
δ =	-2°	-2°
Re =	$11.78\cdot10^{6}$	NONVISCOUS
ω_R =	0.64	0.64

30 PERCENT CHORD FLAP

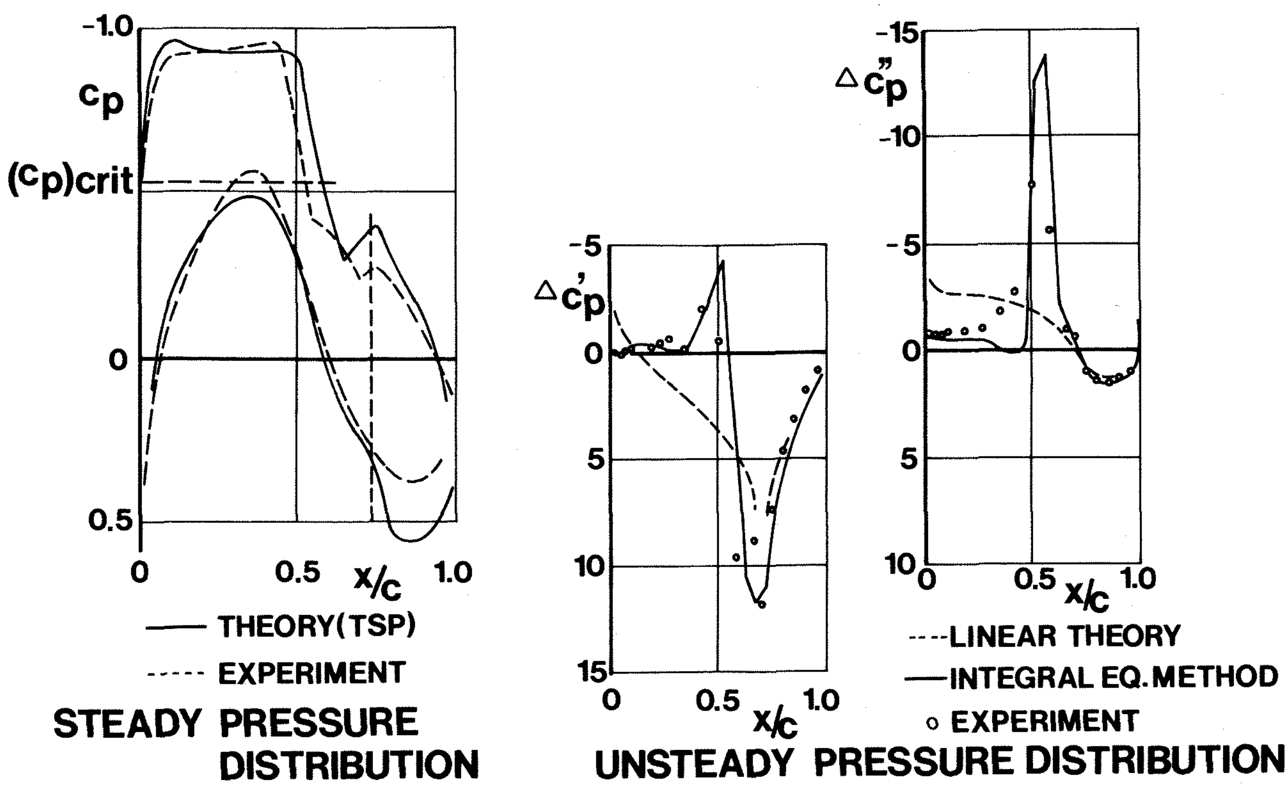

Fig. 6: Comparison of theoretical (integral equation method) and experimental pressure distributions for transonic flow

V. Analysis of unsteady aerodynamic coefficients

For the investigation of unsteady aerodynamics as an element of an active control system designed to limit the loads on a wing it is appropriate to evaluate the aerodynamic lift and the pitching moment. The influence of the parameters systematically investigated in the present tests is generally very similar for the section lift and the pitching moment. This allows discussion of the remarkable features representatively e.g. by means of the unsteady lift coefficient k_c.

A valuable basis for the interpretation of unsteady flow is already given by quasi-steady results, because apart from the phase angle, the general flow structure appears to be very similar to those observed in fully unsteady flow. The quasi-steady pressure distribution can be interpreted as the unsteady pressure distribution for zero frequency i.e. infinitely slow oscillations. As there exists only the real part of the unsteady lift coefficient k_c there is a very natural separation of the influence of different test parameters.

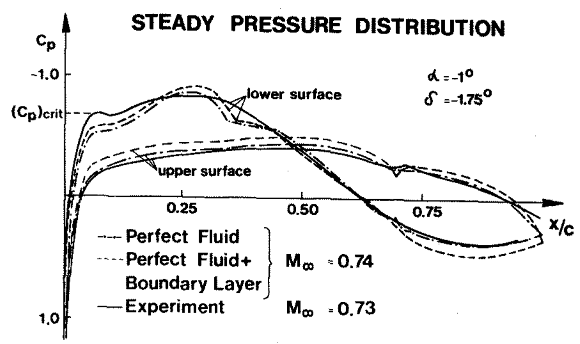

Fig. 8 presents the quasi-steady value of the unsteady lift coefficient k_c for the 30 percent and the 12.5 percent chord flap as a function of the Machnumber M_∞. For subsonic flow the influence of the Machnumber can properly be described by linear compressibility rules already commonly in use for steady compressible flow. There is no noticeable change in unsteady lift with angle of attack and flap deflection. But the unsteady lift coefficient becomes considerably more sensitive to all these parameters after exceeding the critical Machnumber due to their influence on the development of the supersonic region. For supercritical flow the gradient of $k_c=f(M_\infty)$ initially is steeper than predicted by linear compressibility rules. This feature is also typical for the steady lift of supercritical airfoils. If the flow is supercritical, there is a noticeable influence from a change in angle of attack or mean flap deflection, which is far more pronounced for the larger control surface than for the smaller one. The quasi-steady lift coefficient still incrases slightly beyond the design point (M_∞= 0.73, α= 2°) of this airfoil. This indicates that the design pressure distribution is able to sustain disturbances caused by small control surface deflections without experiencing a breakdown.

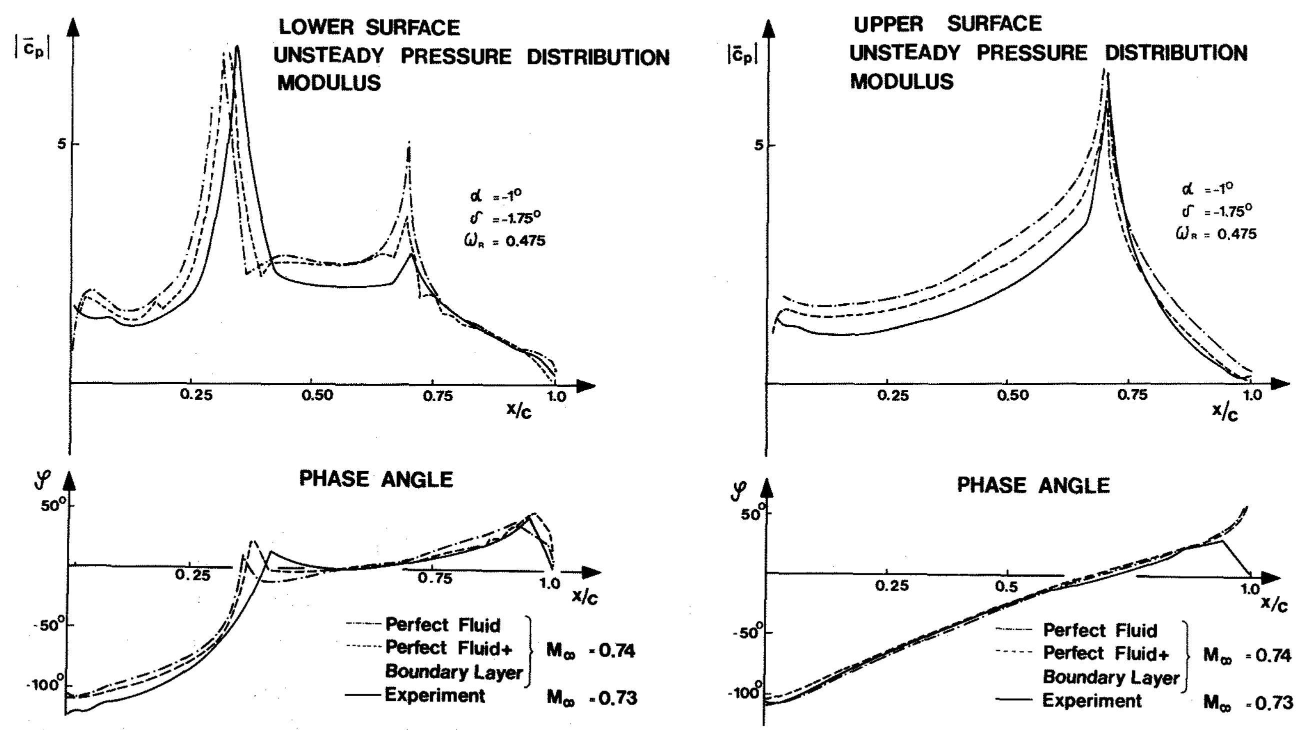

Fig. 7: Comparison of theoretical (TSP and unsteady boundary layer method) and experimental pressure distributions for transonic flow

From experience with steady transonic flow it is well known, that growing shock strength with increasing Machnumber initiates shock induced separation causing an abrupt decline in steady lift. The same phenomenon affects the quasi-steady lift coefficient, but due to the sensitivity of the separation prone or separated flow and the different airfoil shapes involved in steady and quasi-steady flow, the freestream Machnumbers associated with the maxima of the lift coefficients and the gradients are different for steady and quasi-steady flow. The sharp decline in quasi-steady lift shown here for higher Machnumbers is closely connected to the aft-cambering which is a characteristic of most advanced airfoils. At higher Machnumbers a shock develops also on the lower surface forward of the aft-cambering. If this shock is strong enough to cause separation the rear loading - which normally contributes considerably to the lift - fully breaks down.

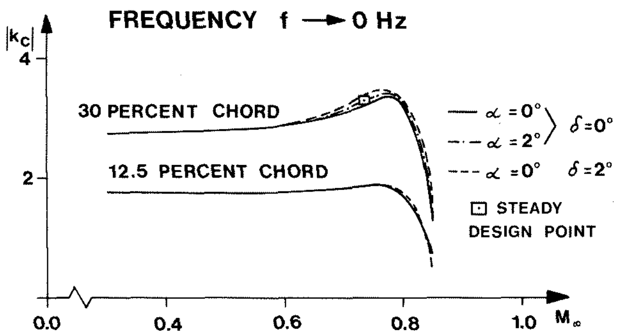

Fig. 8: Quasi-steady lift coefficient as a function of the freestream Machnumber.

If the comparatively flexible wing of a transport aircraft has been excited e.g. by gusts, it tends to react predominantly by performing oscillations in its lower eigenfrequencies. Thus knowledge about unsteady aerodynamic loads generated by harmonic oscillations of wing sections are of practical interest for the design of active control systems. The influence of the reduced frequency ω_R (using the chord c as a reference length) on the unsteady lift coefficient represented by its absolute value $|k_c|$ and its phase angle φ is shown in Fig. 9 for different freestream Machnumbers. Test results for the 30 percent chord control surface have been selected here, but all findings are valid likewise for the 12.5 percent chord flap.

The lower limit i.e. the reduced frequency $\omega_R \rightarrow 0$ is the quasi-steady case which has already been thoroughly discussed by means of Fig. 8.

For low subsonic freestream Machnumbers ($M_\infty = 0.3$) the unsteady lift coefficient k_c slightly reduces in magnitude with increasing reduced frequency, while the corresponding phase angle increases. For the design Machnumber ($M_\infty = 0.73$), which roughly corresponds to the cruise Machnumber, a remarkably different behaviour is evident. The unsteady lift initially remains constant with increasing reduced frequency, though the corresponding phase angle shows

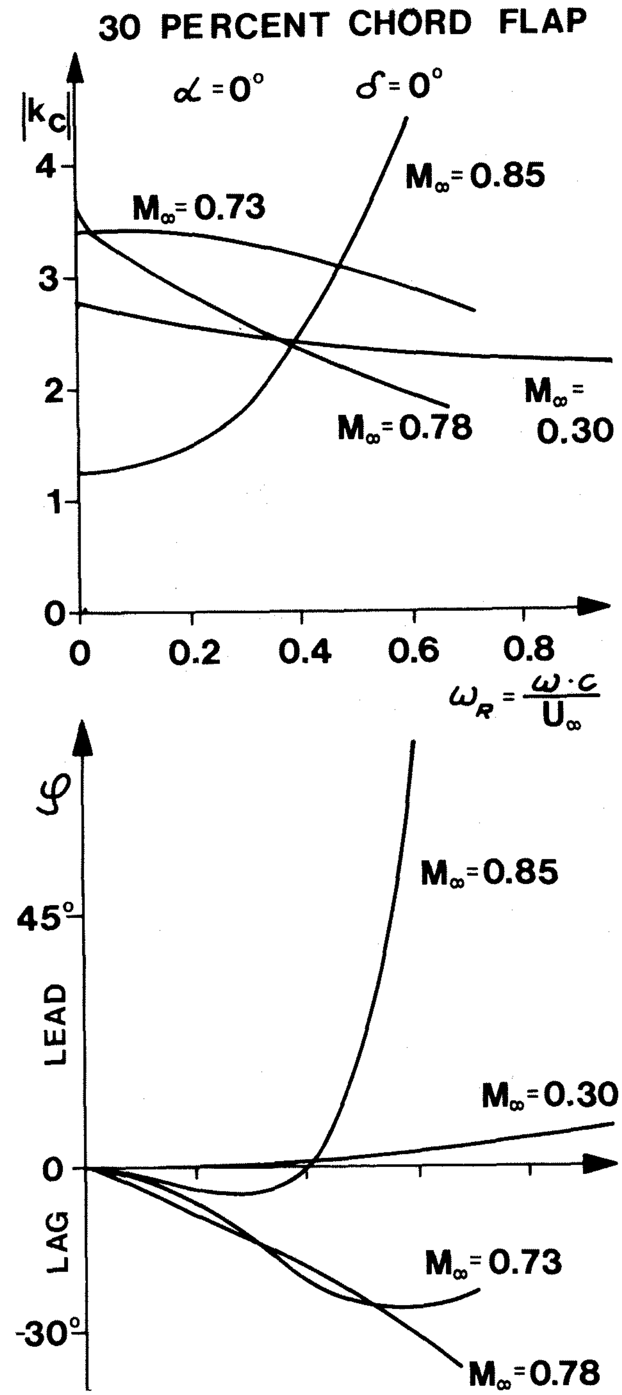

Fig. 9: Unsteady lift coefficient as a function of the reduced frequency.

an increasing lag. Although not considered in the design of the airfoil its pressure distribution demonstrates a high sensitivity to the control surface oscillations and apparently it incorporates a potential for good control surface effectiveness.

Concerning the allevation of gust loads by active control it is worthwile remembering that according to FAR 25.335 the design speed for which maximum gust intensity in the stress analysis has to be considered need not exceed the cruise speed.

For freestream Machnumbers ($M_\infty = 0.78$) exceeding the design Machnumber there is still a minor increase in unsteady lift with the Machnumber but a decline with growing reduced frequency.

For the highest freestream Machnumber ($M_\infty = 0.85$) there exists a very complex flow with shocks of oscillating strength and location on the upper and the lower surface which for some phases of a period cause separation, and in others allow full reattachment. It is very remarkable that in these conditions an increase in reduced frequency produces higher unsteady lift but also very large changes in the phase angle. Thus control surfaces prove to be effective also for cases where the steady flow clearly indicates trailing edge flow separation. But it has still to be investigated if these properties can effectively be utilized in an active control system.

VI. Conclusions

The development of supercritical airfoils offered the opportunity for considerable advancements in the design of more efficient transport aircraft wings. Further contributions are expected from the active control technology using the control surfaces for load alleviation. As the pressure distributions generated on supercritical airfoils generally are very sensitive to changes of the flow conditions, there existed considerable uncertainty, whether a combination of both technologies would allow to utilize also both contributions to higher efficiency. This initiated wind tunnel tests in subsonic and transonic flow using a large chord 2-dimensional model with a supercritical airfoil geometry and two interchangeable trailing edge flaps of different relative chord. Steady and unsteady midsection pressure distributions were measured for both configurations and different angles of attack, flap deflection, Machnumbers, amplitudes and reduced frequencies.

Results from these tests on the one hand were used to check appropriate theoretical prediction methods. For subsonic flow the panel method developed by W. Geissler demonstrates good agreement with experiments for flows not dominated by viscous effects. In the transonic flow régime the integral equation method of R. Voss offers the advantage of very low computation time. Nevertheless it predicts unsteady pressure distributions with acceptable accuracy. The ONERA transonic small perturbation method coupled with an unsteady boundary layer method offers more detailed insight into the influence of the boundary layer on the unsteady pressure distribution and better agreement with test results.

Extensive experimental and theoretical research work has been devoted to the development and analysis of supercritical airfoils. The experience thus gained for steady transonic viscous flow also serves as a sound basis for the interpretation of unsteady flow about such airfoils. But there are also some features unique to unsteady flow:

In subsonic flow trailing edge separation causes considerable reduction in steady lift. Nevertheless in such cases there is a far less influence on the magnitude of the unsteady lift generated by an oscillating control surface. Only the corresponding phase angle changes dramatically.

In transonic flow the development of the supersonic region and the shock strength and location demonstrate a predominant influence on steady as well as unsteady pressure distributions. The properties of the supersonic regions and the shocks change considerably with all parameters investigated. The pressure distributions of this airfoil even for a wide range of off-design conditions are able to sustain disturbances caused by small control surface deflection without breakdown. This is an indication of good control surface effectiveness, which is a prerequisite for the application in active control systems.

Shock induced separation causes substantial decreases in steady lift. The unsteady lift is far less affected and even shows a strong increase with increasing reduced frequency.

VII. References

(1) TIJDEMAN, H.: Investigations of the transonic flow around oscillating airfoils. NLR TR 77090 U(1977)

(2) DAVIS, S.S.; MALCOLM, G.N.:Unsteady aerodynamics of conventional and supercritical airfoils. AIAA paper 80-0734 (1980)

(3) CAHILL, J.; TREON, S.L.; HOFSTETTER, W.R.: Feasibility of testing a large chord, swept-panel model to determine wing shock location at flight Reynoldsnumber AGARD CP-83 (1971)

(4) GEISSLER, W.: Nonlinear unsteady potential flow calculations for three-dimensional oscillating wings. AIAA Journal Vol. 16, No. 11, November 1978 pp 1168-1174 (1978).

(5) COUSTON, M.; ANGELINI, J.J.: Numerical solutions of nonsteady two-dimensional transonic flows. Journal of Fluids Engineering, Vol 101 Sept. 1979 pp 341-347

(6) GEISSLER, W.; VOSS, R.: Investigations of the unsteady airloads on airfoils with oscillating control in sub- and transonic flows. Paper presented at the International Symposium on Aeroelasticity, Oct. 5-7, 1981, Nuremberg, FRG

(7) VOSS, R.: Time linearized calculation of two dimensional unsteady transonic potential flow at small disturbance. European Space Agency ESA TT-728 (1982).

(8) COUSTON, M.; ANGELINI, J.J.; LE BALLEUR, P.; GIRODROUX-LAVIGNE, P.: Prise en compte d'effects de couche limite instationaire dans un calcul bidimensionel transsonique 51st AGARD SMP Meeting Aix-en-Provence Sept.14-19 (1980).