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

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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 criti-
cal 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 predic-
tion 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 trans-
port aircraft. A considerable contribution
cost savings will be achieved by reduc-
ing 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 em-
ploying counteracting control surfaces to
reduce wing loads is a promising way to im-
prove the economy of flight operations.
Critical elements in such an active con-
trol system are the aerodynamic control
surfaces. They still have to fulfill proper-
ly 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 inadmissible lag. It follows
that an insight into the unsteady aerody-
namics of control surfaces is a prere-
quisite for the development of active
control technology.

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To explore the potential of active con-
trol technology a research and technology
programme has been developed, which in-
cludes appropriate theoretical and experi-
mental investigations in unsteady aerody-
namics. 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
methods to gain insight into the complex
phenomena of flows about wings with oscilla-
ting control surfaces. Of great consequence
for the transfer of wind tunnel results to
full scale conditions is the mismatch exi-
siting between the Reynolds number range of
wind tunnels available and Reynolds numbers
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 Reynolds numbers.

Fig. 1: Model geometry and sensor locations
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. Tijmsman (1) has presented a comprehensive survey and very remarkable findings. The Reynolds numbers 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 Reynolds numbers 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 Reynolds number were not available, a very large two-dimensional model was preferred to a halfwing model. This permitted Reynolds numbers not too far from full scale conditions and well above about 10 million. Below this limit, changes in Reynolds number 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 susceptible 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 alleron. 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 Mach number \( M_a = 0.2 \) and Reynolds numbers 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 \( \frac{1}{4} \) ft x 6 \( \frac{1}{2} \) 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) front 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](image)

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 \( M_a = 0.85 \); 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-dimension-

al disturbances are of comparatively small
order. The disadvantage of small endplates
is a flow which is not purely two-dimension-
al. But unlike boundary layer and corner
shock phenomena, which are difficult to
treat theoretically, the effect of end-
plates can be described by potential theory
methods.

Thus the assumption has to be made that
in the middle section of the model, pres-
sure distributions can be generated which
are practically identical with the pres-
sure distributions in undisturbed two-
dimensional flow, generally at slightly
different free stream Machnumbers and an-
gles 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
regime aimed at gaining insight into the
prevailing flow phenomena, examining vis-
tcosity 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 calcula-
tions the panel method of
W. Geissler (4) for incom-
pressible flow was used.
This is a three-dimensional
singularity prediction me-
thod for an arbitrary wing
planform and thickness dis-
tribution applying non-
linear boundary conditions
and the non-linearized
Bernoulli equation. This
method simultaneously pre-
dicts the steady and un-
steady pressure distribu-
tions for a given wing with
trol surfaces at inci-
dence performing harmonic
oscillations.

For the present calcula-
tions a rectangular wing
with an aspect ratio of 6
was prescribed to simu-
late three dimensional ef-
fector. Furthermore the
boundary layer thickness of the center section has
been calculated and added to the airfoil
thickness. A comparison of pressure distribu-
tions 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 typ-
ical for supercritical airfoils with exten-
sive aft-cambering and blunt leading edges at
low speed. Adding the boundary layer
displacement thickness to the airfoil geo-
metry provides improved agreement between
theoretical and experimental results, ex-
cept for the region of aft-cambering. There
the contour pressure is considerably influ-
enced by the development of the wake, which
is not taken into account in the theory.

The unsteady distribution is presented
as the real \( c_p^u \) and imaginary part \( c_p^v \) of
the unsteady pressure coefficient \( c_p = c_p + ic_p^u \).
The present theory properly predicts details
of the unsteady cordwise pressure distri-
bution 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 attri-
buted 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 es-
specially forward of the flap.

Fig. 5 presents comparisons for high
angles of attack and flap deflection and
with a separation region on the upper sur-
face of the flap. Discrepancies between
theoretical and experimental pressure dis-
tributions indicate the influence of the
boundary layer. If, at any particular point
on the contour, viscosity has a noticeable
influence on the steady pressure distribu-
tion it also affects the local un-
steady pressure distribu-
tion with the same order of
magnitude. The areas sur-
ronded by \( c_p (x) \) and \( c_p^u (x) \) represent the in-phase and
out-of-phase portions of the
unsteady lift coefficient
respectively. Viscosity ef-
facts on the upper surface
forward of the flap consid-
erably reduce the in-phase
portion but increase the
out-of-phase portion. The
modulus of the unsteady
lift coefficient is roughly
the same for viscous and
inviscid flow, but there
is a considerable lead in
the phase caused by vis-
cosity. Taking this into account
the steady flow displacement
thickness noticeably redu-
ces the discrepancies be-
tween theoretical and experi-
mental pressure distribu-
tions for steady flow but
is considerably less effec-
tive in the unsteady case.

Separation, as appearing here near the
trailing edge, locally fully restructures
the unsteady pressure distribution. Ap-
parently control surfaces are still effective
in generating unsteady airloads when tra-
oling edge separation is present, but sepa-
ration has a strong influence on the phase
angle.
Subsonic unsteady aerodynamics of supercritical airfoils which are characterized by high suction peaks at high angles of attack and extensive rear loading is very susceptible 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 regime. 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.U. Angelini (5) is limited to low frequencies while the DPVLR 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 consideration 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.
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 systematic variation of the free-stream Mach number 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 $\phi$. 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.

<table>
<thead>
<tr>
<th>EXPER.</th>
<th>THEORY</th>
</tr>
</thead>
<tbody>
<tr>
<td>$M_a = 0.78$</td>
<td>0.75</td>
</tr>
<tr>
<td>$\alpha = 3.8$</td>
<td>1.15</td>
</tr>
<tr>
<td>$\delta = 2^\circ$</td>
<td>$-2^\circ$</td>
</tr>
<tr>
<td>Re $= 11.78 \times 10^6$</td>
<td>NONVISCOUS</td>
</tr>
<tr>
<td>$\omega_r = 0.64$</td>
<td>0.64</td>
</tr>
<tr>
<td>30 PERCENT CHORD FLAP</td>
<td></td>
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</tbody>
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![Diagram](image)

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 represented 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 Mach number $M_\infty$. For subsonic flow the influence of the Mach number 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 Mach number due to their influence on the development of the supersonic region. For supersonic flow the gradient of $k_C=\text{f}(M_\infty)$ initially is steeper than predicted by linear compressibility rules. This feature is also typical for the steady lift of supersonic 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 increases slightly beyond the design point ($M_\infty=0.73$, $\alpha=20^\circ$) 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 Mach number 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 Mach numbers 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 Mach numbers is closely connected to the aft-cambering which is a characteristic of most advanced airfoils. At higher Mach numbers 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 Mach number.

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 $\omega R$ (using the chord $c$ as a reference length) on the unsteady lift coefficient represented by its absolute value $|k_c|$ and its phase angle $\phi$ is shown in Fig. 9 for different freestream Mach numbers. 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 Mach numbers ($M_a = 0.3$) the unsteady lift coefficient $k_c$ slightly reduces in magnitude with increasing reduced frequency, while the corres-
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 worthwhile 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 \( M_a = 0.78 \) 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 \( M_a = 0.85 \) 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 regime 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)
