MEASUREMENTS ON AIRFOILS WITH FLAPS AT MEDIUM REYNOLDS-NUMBERS

by

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Flaps are used for handling an airplane in the air. By deflecting the flap the camber of the airfoil is varied resulting in an increase or decrease of lift. The increase of drag encountered with this lift-effect should be as small as possible. Flaps are normally used as ailerons on wings and at the horizontal and vertical tailplanes.

For modern airplanes with high performance each part of the plane should be of highest performance itself. Aerodynamic performance and handling performance together will give a successful airplane.

To give the constructor some idea about the effectiveness and performance of various flaps, measurements are carried out on some cambered and symmetrical airfoils in the "Laminarwindkanal des Instituts fur Aerodynamik und Gasdynamik, Universitat Stuttgart" (Ref. 1). As a result of the measurements on symmetrical airfoils a consideration on the influence of the airfoil polar and the aspect ratio on the drag of horizontal tailplanes will be given too.

This paper has appeared, in German, in Aero Revue and in OSTIV Publication XI.

1. Cambered Airfoils with Flaps

To get an aileron, the trailing edge of the airfoil is cut off in the outer part of the wing and used as a flap. As no measured polars of the airfoil with flap are commonly available some examples will be given which show the influence of various parameters such as type of the airfoil used, chord of the flap, position of its hinge, thickness of the airfoil, deflection angle of flap, Reynolds-number, and gap between airfoil and flap.

By the deflection of the flap at a fixed angle of attack the camber of the airfoil and thereby its velocity distribution is changed. At the side of the airfoil opposite to the deflection of the flap there arises a peak in the velocity distribution at the knee of the flap, followed by a pressure rise. At the same time the velocity is raised on the nose of the airfoil, too. Depending on the angle of attack of the airfoil these two velocity peaks can cause transition to move forward eventually causing laminar or turbulent separation. This will especially happen at low Reynolds-numbers and at flaps with large chords and large deflection angles.

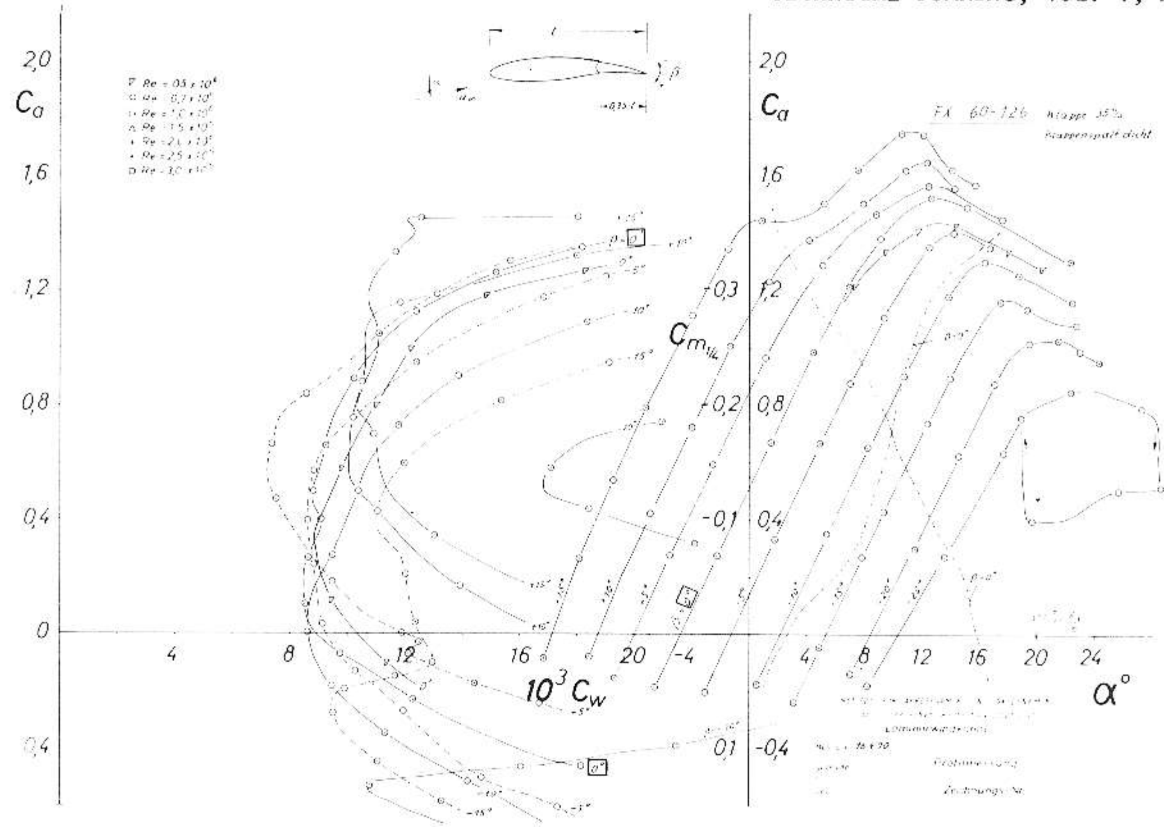

Figure 1

The effect of the flap and of its deflection is different for the type of airfoil used:

At the "normal" airfoil the maximum of the velocity is rather near the nose on both sides (10 to 20 per cent of chord). The pressure rise is flat so that the boundary layer can be laminar up to say 50% of chord, resulting in a rather big drag. With increasing angle of attack the transition point steadily moves forward on the upper side and backward on the lower side. By cutting the trailing edge as forward as 65% chord and using it as a flap, the knee of the flap will lie in a region of a rather thick boundary layer which will tend to separate at growing flap deflection owing to the suction peak at the knee of the flap described above. The suction peak at the nose induced by flap deflection will be of minor effect since the maximum velocity (or minimum pressure) is already lying in this region.

As an example measurements on the airfoil FX 60-126 were carried out for flaps with 20, 25, 30, 35% chord of the airfoil chord. This airfoil is used on many sailplanes in the region of the wing tip.

Figure 1 shows polar diagrams of this airfoil with a flap chord of 35% at Re = 0.7 and 0.5×10^6. All flaps tested on this airfoil were hinged in the middle, the flap nose was rounded and the gaps between airfoil and flap were rather small. The deflection of the flap is defined to be positive in downward direction. (In the Figures c_a holds for c_L and c_W for c_D.)

Owing to a normal airfoil the drag polar for $\beta = 0°$ shows a rather steadily increasing drag with growing angle of attack or lift. The transition points marked by the dashed lines are moving steadily forward and backward with varying c_L. They are measured simply by a stethoscope. For measurements in flight a special method was developed (Ref. 2). At positive angles of flap deflection β the lower branches of the drag polar are shifted upwards in direction of c_L but at the upper branches there is scarcely any effect up to $\beta = +10°$. As can be seen at the $c_L(\alpha)$ polars the angle of attack α corresponding to a tolerable drag is shifted to more and more smaller values as the turbulent boundary layer is separating. At negative angles of flap deflection up to $\beta = -15°$ the increase in drag is rather small, for $\beta = -20°$ the boundary layer has separated. In the diagram one polar for $Re = 0.5 \times 10^6$ at flap deflection angle $\beta = 0°$ is included. The minimum of the drag is nearly the same as

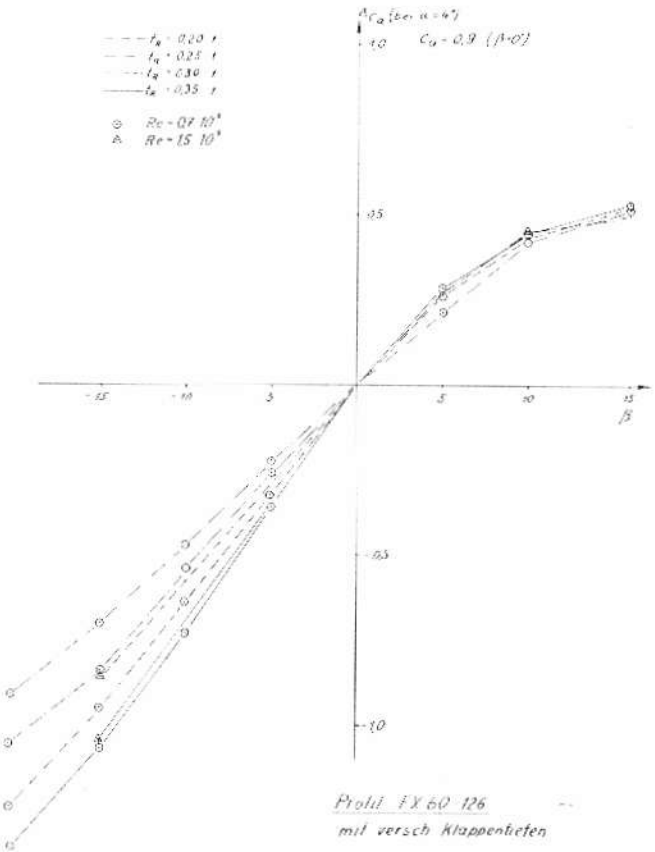

Figure 2

at Re = 0.7×10^6, yet the drag increases more rapidly with growing c_L. The $c_a(\alpha)$ polar shows that turbulent separation occurs now at even lower angles of attack.

The measurements on the same airfoil with other flap chords showed very similar results.

(Notice: On all Figures there are only drawn some few points, but many points were actually measured.)

If a flap is used as an aileron the changes in lift coefficient c_L gained by a deflection of the flap should be as big as possible at low flight speeds as the dynamic pressure is low. At the same time the increase in drag should be small, at least at positive flap deflections. At an angle of attack of $\alpha = +4^o$ no essential increase in lift is gained deflecting the flap from $\beta = +10^o$ to $\beta = +15^o$. But the drag grows considerably as the turbulent boundary layer has separated as the drag polar for $\beta = +15^o$ shows. At higher angles of attack this would be still more pronounced. At negative angles of flap deflection up to $\beta = -15^o$ the increase in drag is rather small.

Figure 2 shows the increase of lift, c_L, gained by various deflections, β, of the flap for the airfoil FX 60-126 with flaps of various chords at an angle of attack of $\alpha = +4^o$ ($c_L = 0.9$ of the airfoil with undeflected flap). At positive flap deflections all flaps have nearly the same efficiency. Beyond $\beta = +10^o$ there is only a weak increase in lift but a high increase in drag as has been shown before. At negative deflections of the flap the efficiency for constant β is growing with increasing flap chord, even at high negative values of β. The measurements were carried out with the gap between airfoil and flap sealed. Some measurements were repeated with open gap; as the gap was very small no difference could be measured.

The other type of airfoil mentioned above are the well-known laminar airfoils: At laminar airfoils the velocity distributions at the angles of attack corresponding to the upper and the lower corner of the laminar bucket are constant on the upper and lower side of the airfoil respectively up to 60 - 80% of the airfoil chord, followed by more or less step pressure rises, depending on the thickness of the airfoil. Within the thus-defined range of angles of attack the transition points are rather fixed to the beginning of the pressure rises. Outside the laminar bucket they very soon move forward to the nose of the airfoil resulting in an increase of drag.

By fixing a flap to a laminar airfoil the flap knee may come to lie before the transition point, thus, because of its gap — without deflection — or by the suction peak caused by deflection, transition or even laminar separation occurs now at the knee of the flap even at low angles of attack. Since the velocity distribution of laminar airfoils is flat at the corners of the laminar bucket a small deflection of the flap will produce a suction peak at the nose on the opposite side bringing the transition point forward. At laminar airfoils the flap chord and the deflection of the flap are therefore expected to have a greater influence than on a normal airfoil.

Figure 3 shows polar diagrams of a laminar airfoil with a thickness of 12.4% at Re = 1.0 and 1.5×10^6 and with two various flaps of 36% and 20% chord. The airfoil has a small laminar bucket. The velocity distri-

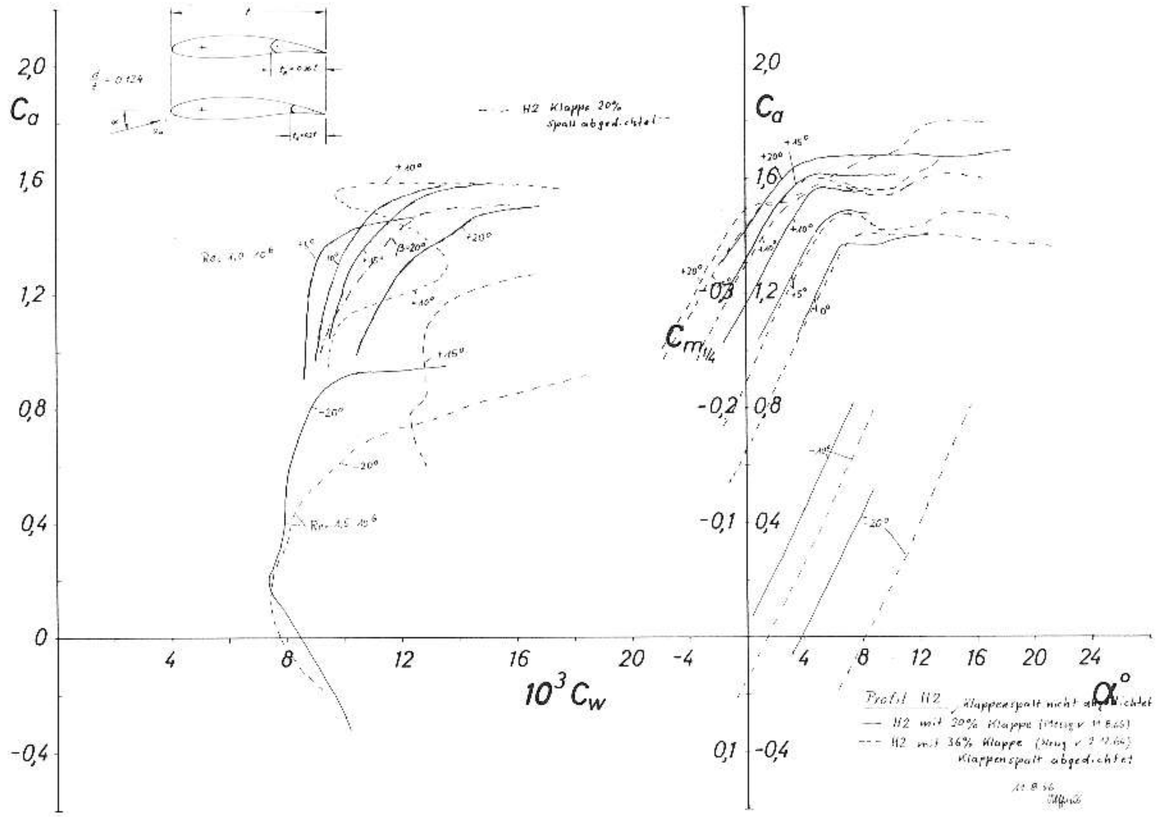

Figure 3

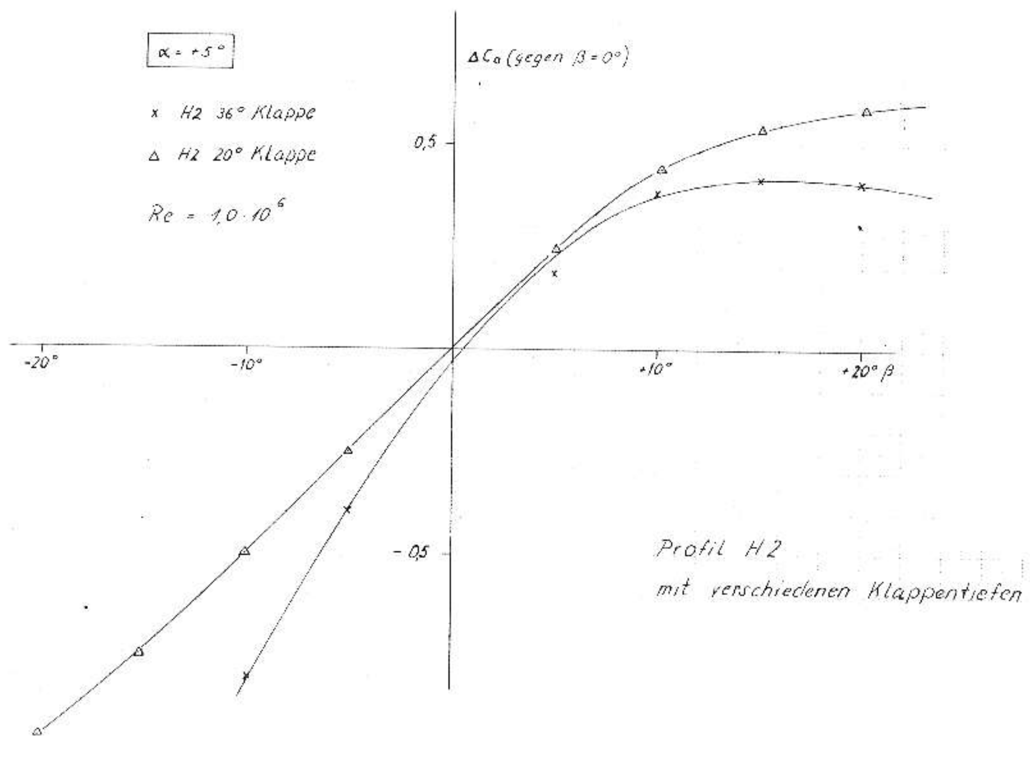

Figure 4

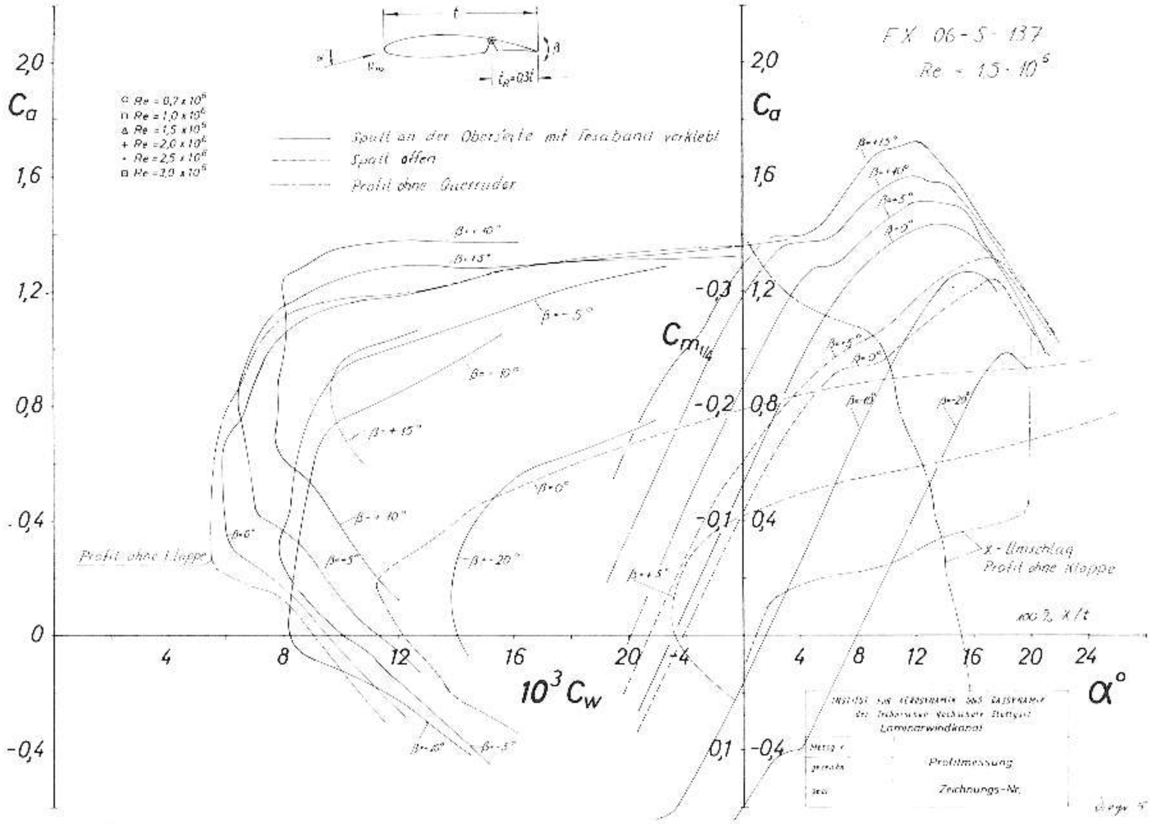

Figure 5

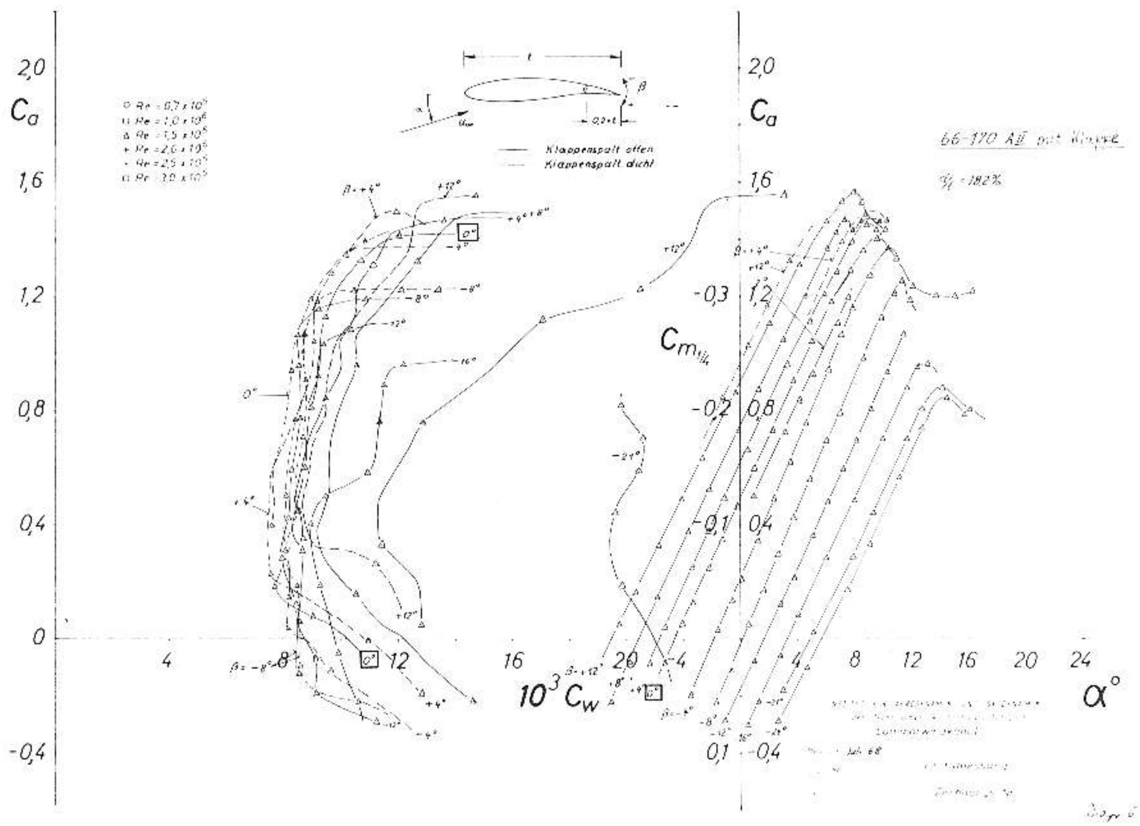

Figure 6

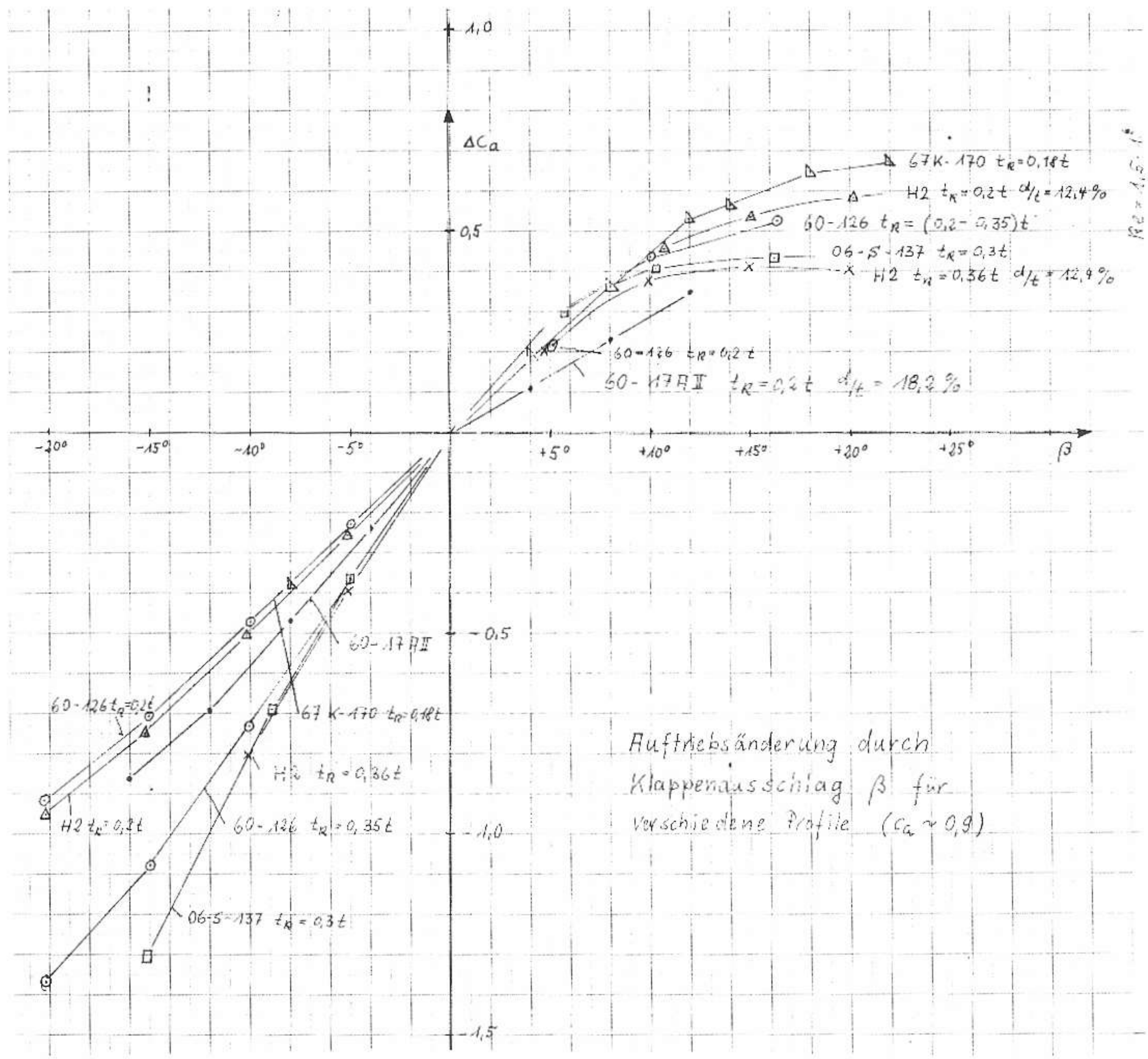

Figure 7

bution on the upper side is flat up to 45% of chord, then it decreases linearly. On the lower side it is rather flat up to the trailing edge. In the Figure only the upper corners of the drag polars for positive flap deflections are shown.

The drag polars for the various flaps show an appreciable influence of the flap chord. At the 36% chord flap (dashed lines) the drag is magnified and separation occurs much earlier as the polars for $\beta = +10^o$, $+15^o$, -20^o show. The knee of the flap is situated too far ahead. As at small angles of attack boundary layer transition normally would take place behind it - laminar separation will occur producing a high drag.

With the flap of 20% chord (full lines) the suction peak at the flap knee is smaller owing to the minor chord of the flap. As the knee is situated 16% further aft, transition takes place ahead of the knee, the turbulent boundary layer can overcome the pressure rise over the flap. The $c_L(\alpha)$ polars show little difference at positive flap deflections but a more pronounced effect at the negative ones.

At the measurements on the 36% chord flap the gap was sealed, on the 20% flap it was open. There is one drag polar for the 20% chord flap with sealed gap marked by a dash-pointed line. The drag is about 15% smaller than with the opened gap.

In Figure 4 the effectiveness of the two various flaps is shown for $\alpha = +5^o$. At positive angles of flap deflections there is a considerable difference in c_L. Above $\beta = +10^o$ there is nearly no further gain in lift with the 36% chord flap. This might be an example that at laminar airfoils the flap chord may be not too large.

In Figure 5 there are shown polars of a laminar airfoil with a flap of 30% chord at $Re = 1.5 \times 10^6$. Here the flap is hinged at the upper side of the airfoil contour, at the lower side there is a wedge-shaped gap between flap and the airfoil (as on the sailplane Ka-6). The transition point of the airfoil without flap would be at 100% chord at the lower side. The full lines correspond to the airfoil with the sealed gap. By comparing the dash pointed drag polar for the airfoil without flap with the corresponding polar for flap deflection $\beta = 0^o$, it can be seen that the drag is increased by 10% owing to the flap. At positive flap deflections the polars are shifted upwards in the direction of c_L. At $\beta = +15^o$ separation occurs. At negative deflections, β, the drag is increased rather soon owing to the gap and the sharp corner of the flap.

The influence of sealing the gap between airfoil and flap at the upper side is especially remarkable. There are two measurements for flap deflections of $\beta = 0^o$ and $\beta = +5^o$ with the gap open (dashed lines). As the drag polars and the $c_L(\alpha)$ polars show, the boundary layer has separated from the flap at zero angle of attack either on its upper side or its lower side. At this flap arrangement the flap has to be sealed.

Figure 6 shows another example of the influence of the gap seal. The polars correspond to the airfoil FX 66-170 AII with 20% chord with the flap hinged in the middle. The drag polar for $\beta = +12^o$ with sealed gap (dashed line) shows an appreciable gain of drag against the polar with open gap(full line). The same holds for the lift. As this airfoil has a thickness of 18.2% of chord the efficiency of flap deflection on lift c_L is not so big as on the other airfoils showed before.

Figure 7 shows a summary of the variation of lift c_L gained by various flap deflections β at $c_L \approx 0.9$ for all airfoils mentioned. At positive flap deflections, β, the normal airfoil FX 60-126 with flap chords from 20 to 35% and the laminar airfoil H 2 with 20% flap chord show the best efficiency. The laminar profile FX 06-S-137 with 30% flap chord and H 2 with 36% flap chord show no increase in lift above $\beta = +10^o$ owing to early separation. The efficiency of the airfoil 60-17 AII with 20% flap chord is less, owing to its thickness of 18.2%.

At negative flap deflections, the efficiency of the flaps becomes higher with growing flap chords. In this case the effectiveness on the thick airfoil 60-17 AII is larger than on the other airfoils with the same flap chord.

For comparison, measurements on the airfoil FX 67-K-170 (17% thickness) with 18% flap chord are shown, too. This airfoil is especially designed for operation with flap. Though it is a rather thick airfoil it has better efficiency at positive flap deflections than all airfoils shown and the same efficiency as the airfoils with 20% flap chord on negative angles of flap deflection. On sailplanes with flaps along the whole span the ailerons are partly deflected with the inner flaps, they therefore must have a good efficiency even at high angles of flap deflection.

In summarizing all these measurements on cambered airfoils the following statement can be made:

The chord of the flap could be up to 30% of airfoil chord on a normal profile with a thickness of about 14%. At a laminar airfoil the flap chord might not exceed 20% of airfoil chord.

The tolerable deflection of the flap might not be more than $\beta = +10^o$ at positive direction as the drag is increasing causing a high pitch and yaw moment. At negative flap deflections the pitch and yaw moment is diminished by the additional airfoil drag, therefore deflections up to $\beta = -15^o$ or -20^o may be allowed.

The flap can be constructed with the the hinge in the middle and its nose rounded. A flap with the hinge in the lower contour of the airfoil shape and a rounded nose at its upper side has the advantage that at negative flap deflections the pitch and yaw moment is diminished by separation at the knee of the flap on the lower side.

At all flaps on an airplane the gaps between airfoil and flap should be sealed as the drag is increased appreciably by open gaps, especially at low Reynolds numbers.

2. Symmetrical Airfoils with Flaps

Symmetrical airfoils with flaps are used on vertical and horizontal tailplanes. They are working in the range of Reynolds-numbers from 0.5 to 0.7×10^6. As there is not much known about airfoils with flaps at these low Reynolds-numbers, windtunnel tests were made on some airfoils with flaps of various chords.

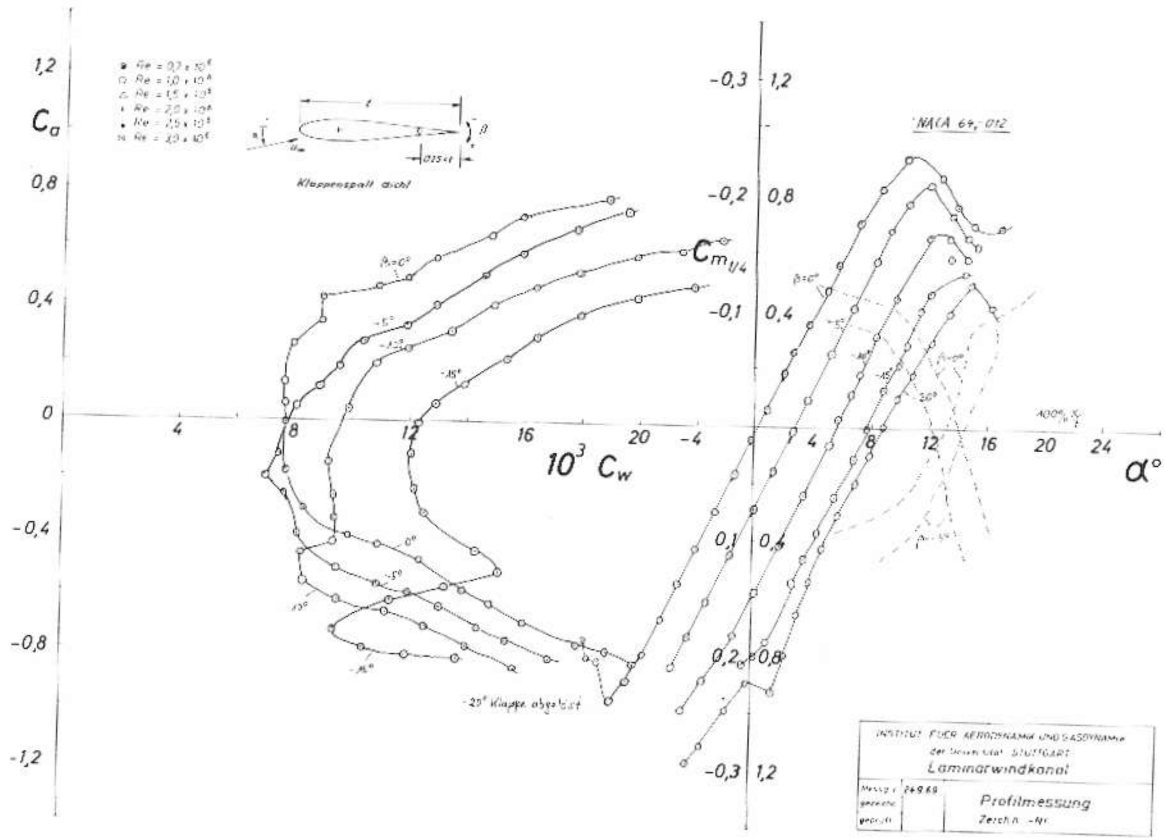

Figure 8

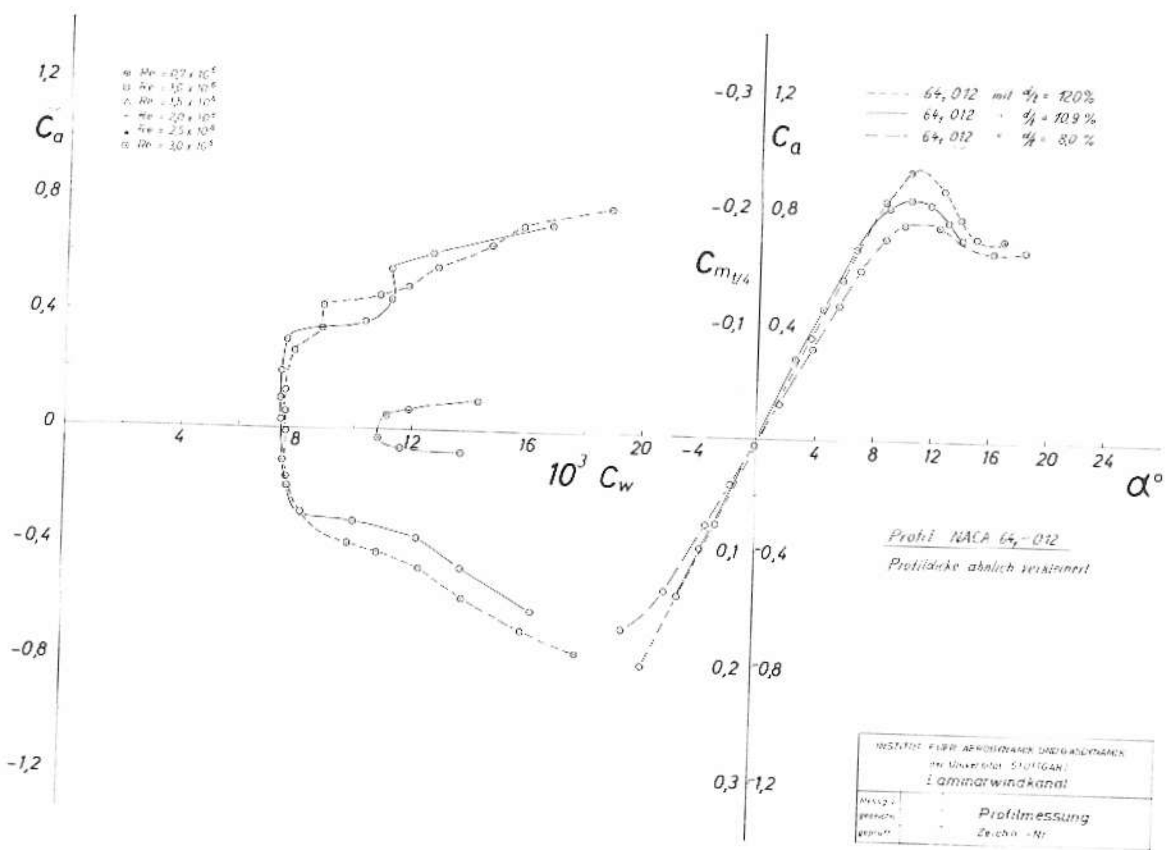

Figure 9

Figure 8 shows polar diagrams at Re = 0.7×10^6 of the NACA 64_1 012 airfoil with a flap of 25% chord. Already at low flap deflections the drag is increased by a laminar separation bubble at the knee of the flap. The $c_L(\alpha)$ polar shows that the variation in lift is small for $\beta = -20^o$ against $\beta = -15^o$. On many horizontal tailplanes the airfoils are thinned in the outer part of the tailplane by geometrical transformation. Diagram 9 shows polars measured on parts of original tailplanes with the airfoil NACA 64_1 012 at Re = 0.7×10^6. Polars of the original and of the same by geometrically transformed profiles with thicknesses of 10.9% and 8% are shown.

With decreasing thickness the laminar bucket grows smaller, the maximum lift $c_{L\ max}$ and the lift curve slope decrease too. This is an example for that it is not advisable to vary the thickness of an airfoil by merely geometrical transformation. The variation of thickness has to be achieved by suitable changes on the velocity distribution of the original airfoil.

On the airfoil gained by geometrical transformation and commonly on all airfoils with thicknesses less than 10% of chord already at small angles of attack a laminar separation bubble is formed in the vicinity of the nose at these low Reynolds numbers. Behind the laminar separation bubble a thick turbulent boundary layer forms causing a relatively high drag.

On Figure 10 the transition line on an original tailplane is to be seen. At this tailplane the thickness of the NACA 64_1 012 airfoil is varied down to 8% of chord in the lower part of the picture. The picture shows the upper side of the tailplane at an angle of attack of $\alpha = +3.5^o$ with $\beta = 0^o$ and Re $\sim 0.7 \times 10^6$. The transition line is visualized by a mixture of oil and lampblack. In the turbulent region where the skin friction is high the black mixture is transported away, making the white paint of the model visible. The boundary between black and white marks the transition of boundary layer which is at 45% of chord in the upper part and at only 5% of chord in the lower and thin part of the

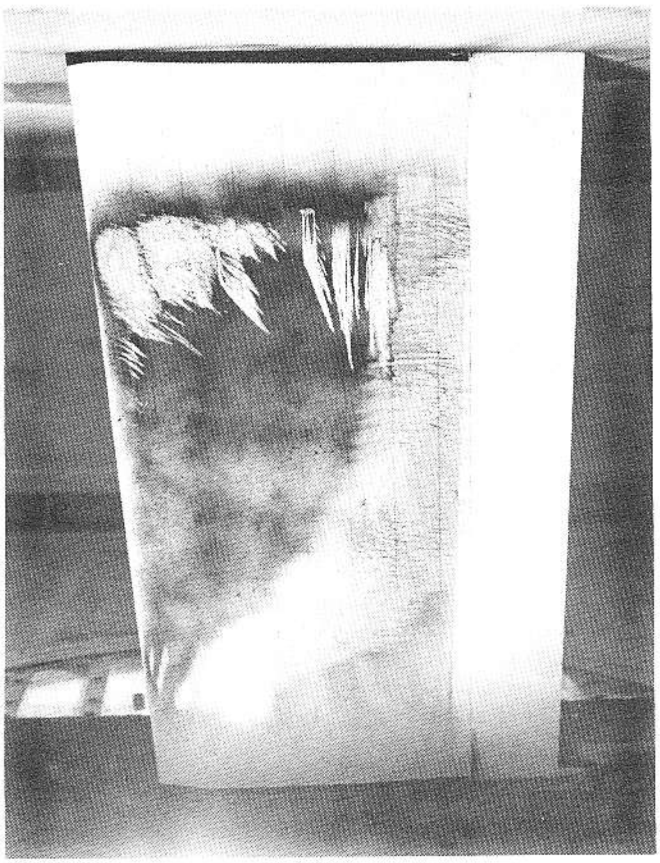

Figure 10. Transition indication on an original tailplane: upper side, $\alpha = +3.5^o$, $\beta = 0^o$ (Flow direction from left to right.)

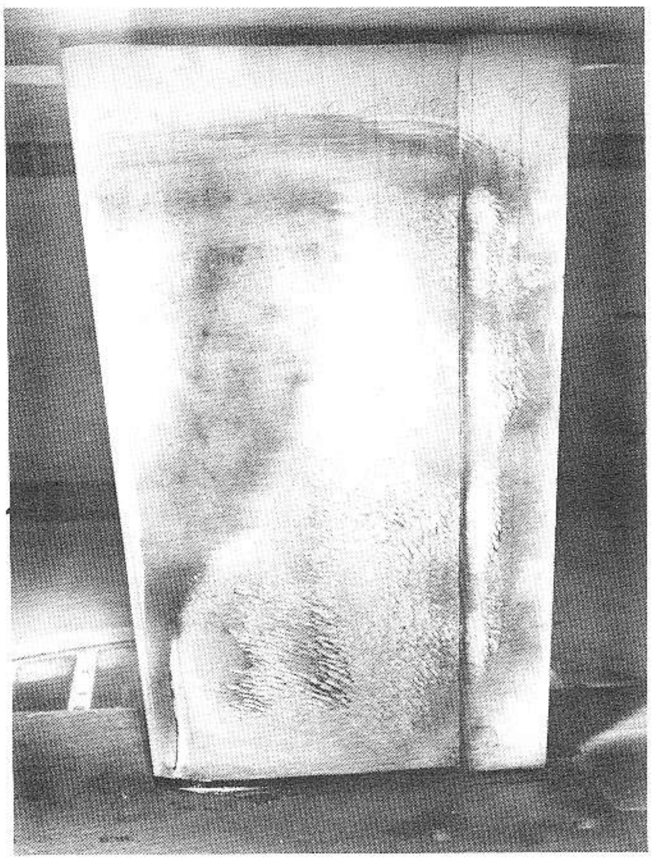

Figure 11. Transition indication on an original tailplane: upper side, $\alpha = +5^o$, $\beta = -5^o$ (Flow direction from left to right.)

tailplane.

Figure 11 shows another transition indication on the upper side of the same tailplane at $\alpha = +5^o$ and $\beta = -5^o$. In the lower part of the tailplane a thick line of the black mixture at 5% of chord is formed which marks a laminar separation bubble. The laminar boundary layer separates just before that line; after getting turbulent it reattaches behind the line so forming a bubble beneath it.

Figure 12 shows a transition indication on the upper side too at $\alpha = -5^o$, and $\beta = +5^o$. Although the angle of attack is negative, a laminar separation bubble is caused as the flap is deflected to the other side of the airfoil causing a suction peak at the knee on this side.

These measurements show that it is not advisable to alter the thickness of airfoils by geometrical transformation and that airfoils with and without flaps, at least at low Reynolds-numbers, ought to be thicker than 10% chord.

In Figure 13 the variation of lift by

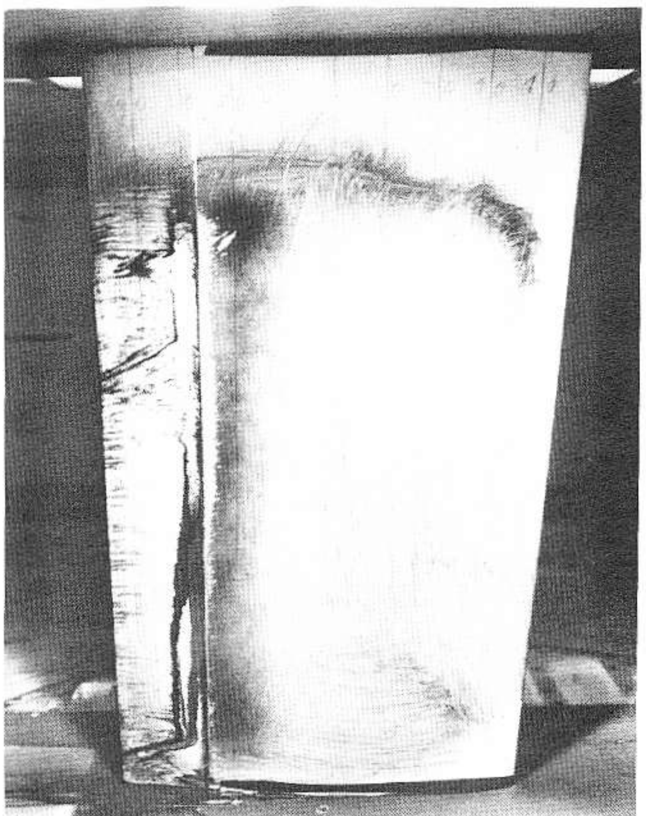

Figure 12. Transition indication on an original tailplane: upper side, $\alpha = -5^o$, $\beta = +5^o$ (Flow direction from right to left.)

deflection of the flap is shown for the symmetrical airfoil NACA 64_1 012 at an angle of attack $\alpha = -4^o$, as vertical and horizontal tailplanes commonly work with negative angle of attack and positive flap deflection or vice versa. The difference in efficiency on lift is not big between the 12% and the 8% thick airfoils. But the 12% thick airfoil has an appreciably lower drag.

There are also shown measurements on the same airfoil having a flap of 40% chord as it is used in vertical tailplanes working at Re $\sim$ 1.0×10^6. The measurements are made for a Reynolds number of 1.0 and 1.5×10^6. The Reynolds-number has a rather big influence and at Re = 0.7×10^6 the effectiveness of the 40% chord flap would probably be the same as that of the 25% chord flap.

Figure 14 shows polar diagrams of the symmetrical profile L 111-142 K with a 25% chord flap. The minimum drag is a little higher as on the NACA 64_1 012 owing to the thickness of 14.2% but the laminar bucket is wider and the drag does merely not increase by deflection of the flap.

There is one drag polar for $\beta = -15^o$ with open gap between airfoil and flap. By sealing the flap the drag can be reduced appreciably. At the lower Reynolds-number of 0.5×10^6 the drag is about 10% higher.

Figure 13

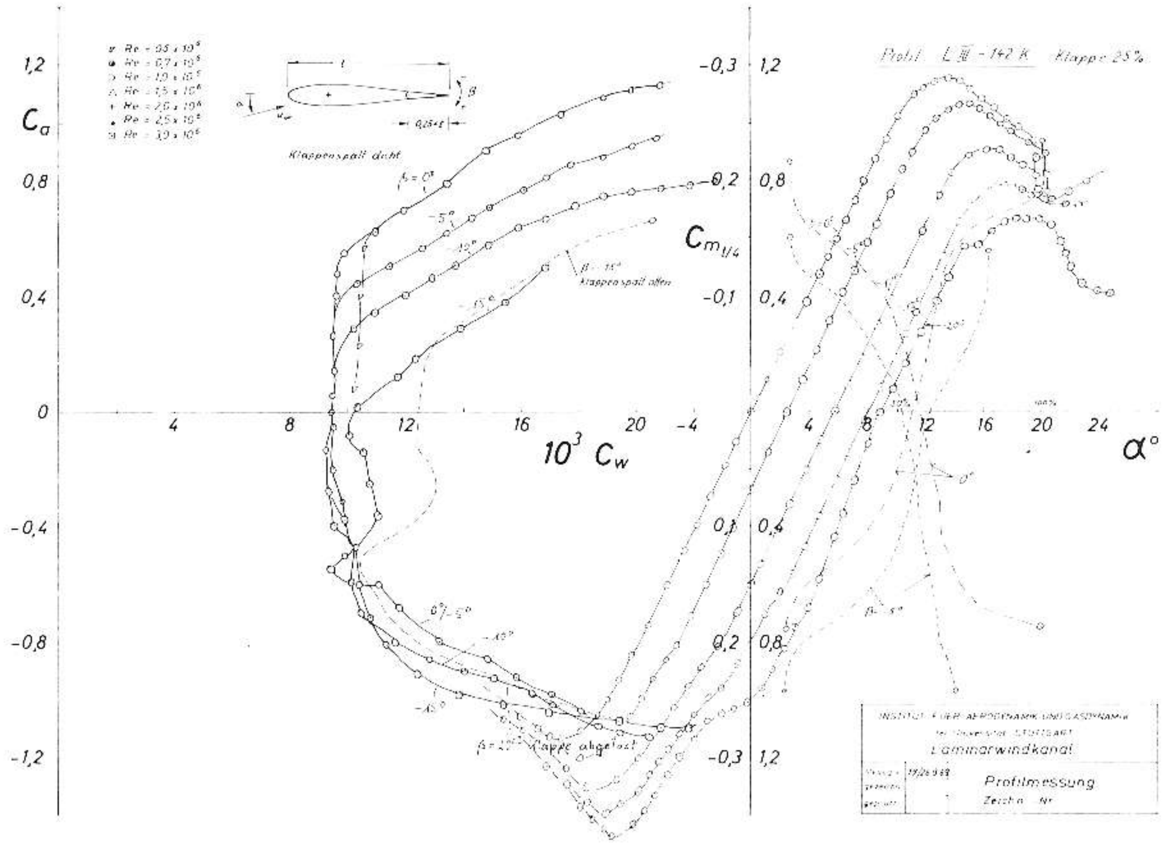

Figure 14

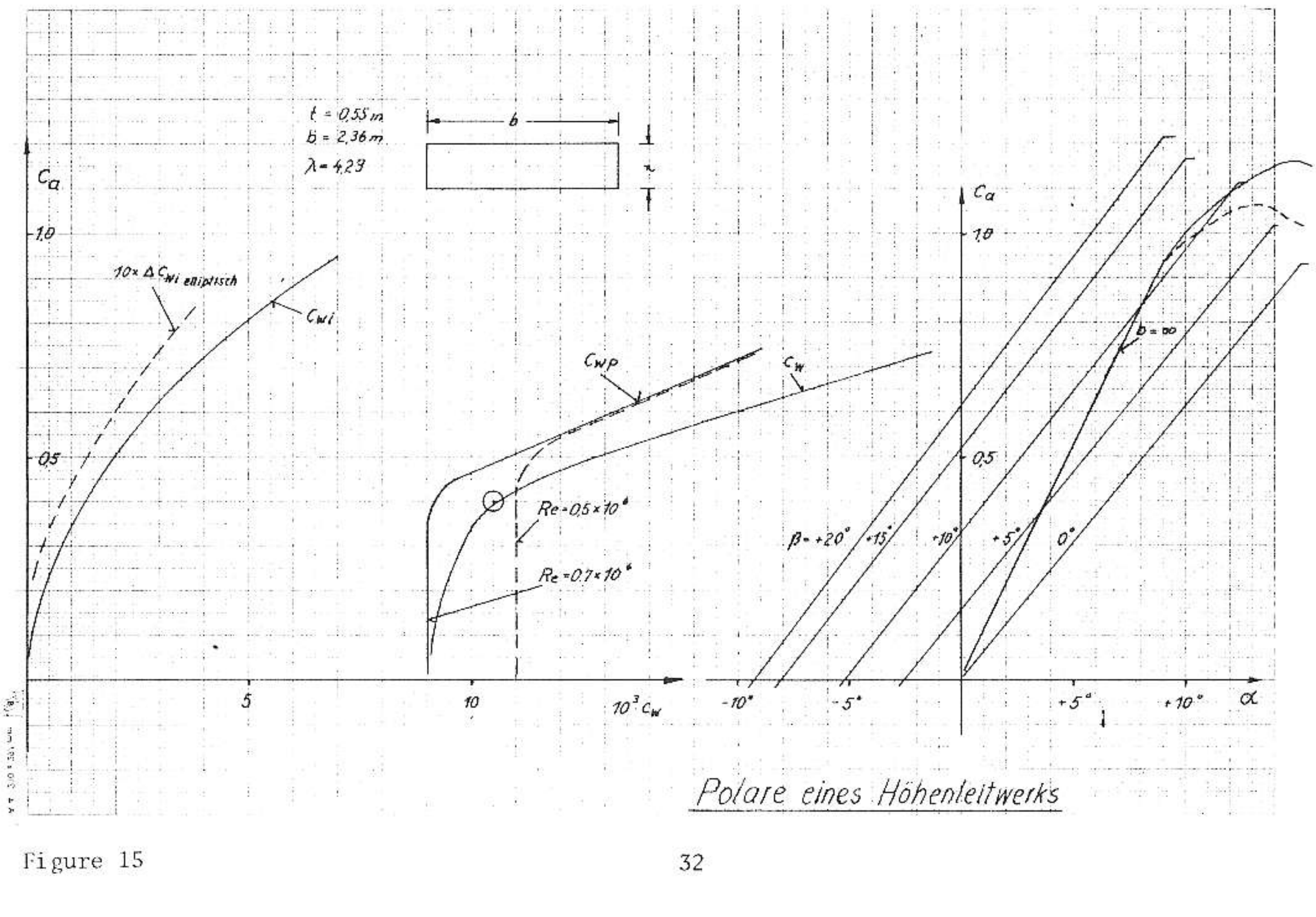

Figure 15

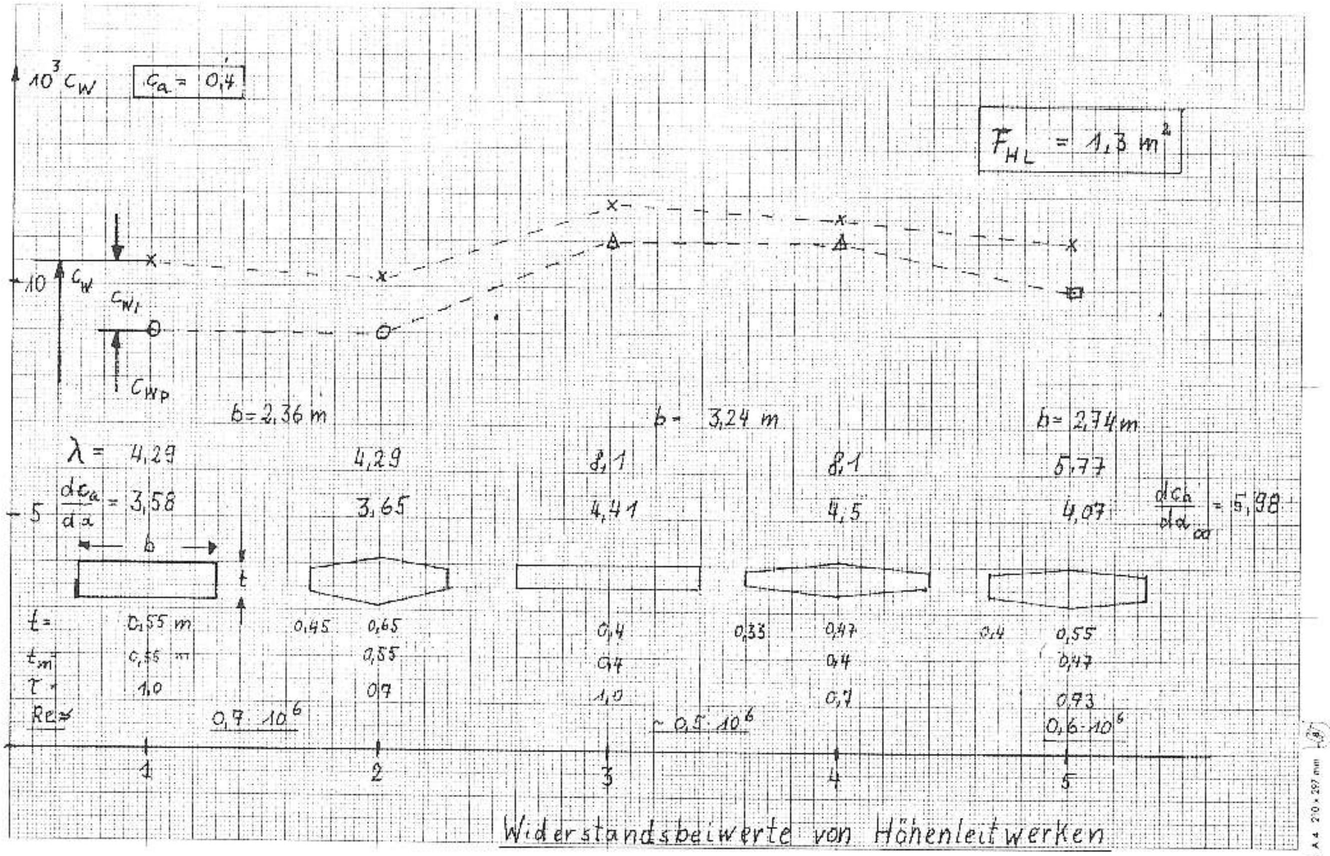

Figure 16

3. The Influence of the Airfoil Polar and The Aspect Ratio on the Drag of Horizontal Tailplanes

Depending on the position of the center of gravity horizontal tailplanes are working at lift coefficients of up to +0.2 at low speed and down to -0.2 at high speed. At a normal tailplane with stabilizer and elevator the angle of attack is positive ($\alpha \sim +5^o$) at low speed. The lift coefficient must be reduced to $c_L \sim 0.2$ by a negative flap deflection. At high speed the angle of attack is negative ($\alpha \sim -5^o$) and the flap deflection therefore must be positive. An airfoil suitable for a horizontal stabilizer therefore should have low drag in the range of at least $c_L = \pm 0.3$ with flap deflections up to $\beta = \pm 10^o$.

Figure 15 shows the drag polar and the $c_L(\alpha)$ polar for a tailplane with rectangular shape. The drag c_D (= c_W) is composed of the drag of the two-dimensional airfoil c_{Dp} (= c_{Wp}) and the induced drag c_{Di} (= c_{Wi}). The induced drag and the lift of the tailplane are calculated by the method of lifting lines according to Weissinger. The $c_L(\alpha)$ polar of the two-dimensional airfoil is shown too. Δc_{Wi} means the difference of the induced drag to the induced drag of an elliptical lift distribution. Airfoil drag polars for Re = 0.7 and 0.5 x 10^6 are listed too.

In order to show the influence of the aspect ratio and the combined variation in Reynolds-number causing variation of drag, calculations for 5 tailplanes of various shape were carried out. The results are summarized in Figure 16.

All tailplanes have the same area F = 1.3 m^2. The chord of tailplane 1 is selected that at a flight speed of about 80 km/h the Reynolds number is 0.7 x 10^6. At the same flight speed tailplane 3 with a smaller chord flies with Re = 0.5 x 10^6. Tailplanes 2 and 4 have the same mean chord as tailplanes 1 and 3 but a taper of τ = 0.7. Tailplane 5 has at its root and tip the chords of tailplanes 1 and 4 respectively. Its Reynolds-number is 0.6 x 10^6. For each tailplane configuration the span, aspect ratio and lift curve slope are listed too.

For a lift coefficient $c_L = 0.4$ the airfoil drag $c_{DP} (= c_{WP})$ for the corresponding Reynolds-number and the induced drag c_{Di} $(= c_W)$ giving together the total drag c_D $(= c_W)$ are shown for each configuration. The induced drag is low with respect to the airfoil drag. By an increase of the aspect ratio from 4.3 to 8.1 it is scarcely reduced. But the profile drag is enlarged as the Reynolds-number gets smaller. Even if the influence of the Reynolds-number on airfoil drag would be smaller the tailplanes 1 or 2 with small aspect ratio would be superior.

The lift curve slope of a tailplane in any case should not be too high as the lift produced by changes in angle of attack must be brought back on standard used for stability by an opposite deflection of the flap.

As the efficiency of a flap is better at higher Reynolds-numbers it is advisable to build tailplanes with a long chord and small aspect ratio as tailplane 1. The taper is more of optical than of aerodynamical use. These tailplanes might have even a smaller weight, too. As long as the drag of an airfoil with flap is not increased by the deflection of the flap, an all-flying tailplane has no advantage. In order to get a suitable mass balance all-flying tailplanes are often tapered and swept. Both will have no good effect on the boundary layer at these low Reynolds-numbers.

Wind tunnel measurements on this subject not published in this paper and all other wind tunnel measurements on FX-airfoils carried out in the "Laminarwindkanal des Instituts fur Aerodynamik der Universitat Stuttgart" are prepared to be published by the Institute.

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