Effects on the Polar Due To Changes or Disturbances To The Contour of the Wing Profile

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Many different factors control the precision demands on the profile contour of airfoils. Changes or disturbances to the surface of wind tunnel models were deliberately brought about, and the composition of airfoil surfaces was changed for test purposes. The results of the wind tunnel measurements shall be reported in this text.

To begin with, something must be said about modifications (intentional or not) to the profile contour which are limited to a special part of the profile. The most important area of the profile is actually the *airfoil nose*. Some people are of the opinion, because of computer calculations, that contour mistakes in the magnitude from 0.004 to 0.008 inches with wing chords of 39 inches influence the width of the laminar bucket. It is impossible to build wind tunnel models with such precision at a justifiable expenditure, not to mention the construction of real airplanes. The models used for measurements in the laminar wind tunnel at the Institute of Aero- and Gasdynamics in Stuttgart are in general built with profile accuracies from 0.020 to 0.027 inches. The profile templates are made by hand. It was determined by comparative measurements of different wind tunnel models of the same profile, but with different profile accuracies, that the polars of all models were the same, and hence the trueness of the contour which was achieved by our level of fabrication accuracy is sufficient. Contour differences in the magnitude of 0.008 inches are smoothed out because of the boundary layer. The smallest fluctuations greatly influence the numerical calculation techniques. In this case, the so-called "computer wind tunnel" model supplies increasingly doubtful results.

In **Figure 1** the wind tunnel measurements have been represented for a 14.3% thick flapped airfoil. The airfoil nose of the wind tunnel model was changed through enlargement up to 0.14 inches in the way the diagram shows (profile depth = 27.6 inches). The hashed polar has been measured on the model with the changed airfoil nose. The width of the laminar bucket of both models is the same; by the changed nose

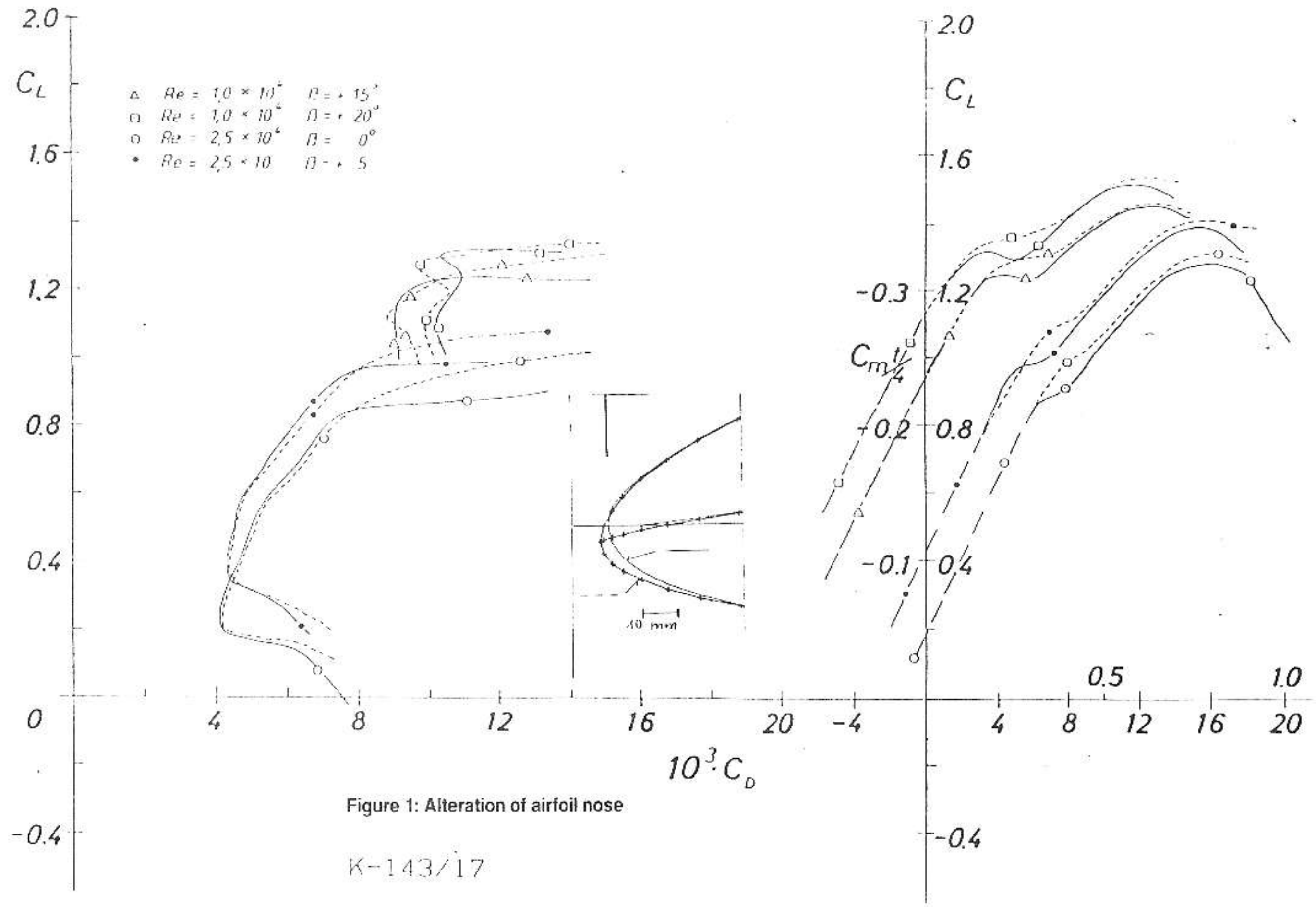

Figure 1: Alteration of airfoil nose

the increase of drag on the upper range of the laminar bucket is less. This is to be expressed in the $c_a(\alpha)$-curve. The stall characteristics are somewhat more favorable. **Figure 2** shows comparative measurements of another airfoil with flaps. Here, above all, the maximum lift and the stall characteristics are improved.

The subsequent enlargement of the nose leads to smooth *contour aberrations* in the transition region of the original surface. The polars show, nevertheless, that the changes don't negatively effect the performance, if one takes into consideration the order of the contour changes on the whole. Of course, no *waviness* or apparent low points can be tolerated. On the other hand, local limited thickening or thinning of the profile in the spar area could have a greater effect on the polar because of the control of the transition region.

Many profiles are equipped with a relatively strong reflexed *trailing edge*. In the manufacturing of wings the reflex often is taken out for simplicity. That means the reflexed aft portion of the profile is replaced by a straight piece. **Figure 3** shows the effects of such changes on the polar of the profile. The straightening of the trailing edge works like a negatively declined flap (flap depth of 6%, $\beta = -10$). **Figure 4** shows a flapped airfoil where a 5% deep *auxiliary flap* was attached, which was declined at angles of $\beta = \pm 10°$. The displacement of the laminar bucket because of this small flap is substantial.

On another wind tunnel model a "*trim vane*" made out of hard aluminum foil (Mylar foil), which hung out at 2% of the profile chord over the trailing edge and was bent 15° downward, was fastened on the trailing edge of the upper side. Because of this, the laminar bucket was pushed upwards to a difference $\Delta c_a = 0.15$. Through such a change, a wing with a rigid profile could be adapted to special weather conditions, or flight weights. In the same way an aerodynamic twist of the wings could be attained. The probabilities of success by each profile should, nevertheless, be examined in wind tunnel experiments.

An often-asked question concerns the tolerable *trailing edge thickness* of a wing. It usually is not practical to build the trailing edge of airplane wings extremely sharp. **Figure 5** shows polars of a 17% thick profile, of which the trailing edge thickness was varied. The mean increase in drag Δc_w, in relation to the profile with a trailing edge thickness of 0.2% of the profile chord, is provided. Measurements with a Reynolds number of Re = 3 million, as well as similar investigations of other profiles, yielded about the same drag increases.

Another possibility for the modification of the profile contour is the *thickening* or *cambering* of the whole contour. On a wing, the same profile is seldom used over the entire length of the span. Profiles of different thickness and camber are used at the wing root, in the aileron and at the wing tip. The other profiles lying between these special profiles are achieved by *sheering*. Through this we get profiles that have different contours than the original ones, and which could have other polars than the original profiles. **Figure 6** shows the influence of thickness *variation*. As an example the profile FX 61-168 was thinned to a thickness of 14%. The thin profile (hashed polar) has a smaller laminar bucket, that is attributed to the sharp airfoil nose. Its minimum drag is

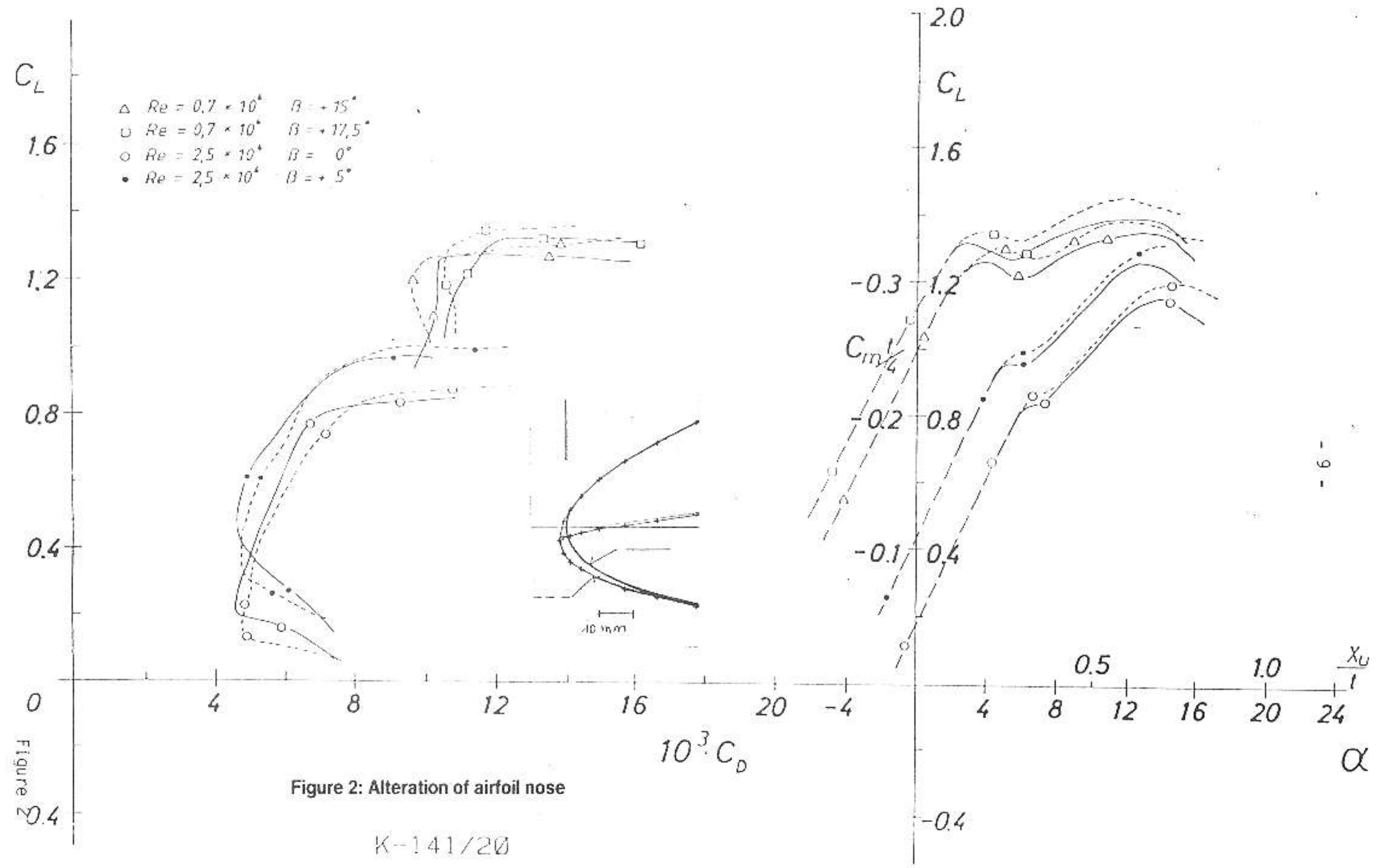

Figure 2: Alteration of airfoil nose

smaller, because on the profile's upper side the transition of the boundary layer occurs somewhat later, and the pressure rise in the turbulent boundary layer is smaller with a thinner profile.

Figure 7 shows the influence of *changes in camber*. The camber of the profile, which in Fig. 6 was thinned from 16.8% to 14%, was here raised from 3% to 3.6% of the profile chord. The lift coefficient for an $\alpha = 0°$ was increased from 0.5 to 0.6. The lower border of the laminar bucket was displaced far above, since transition on the lower side occurs substantially earlier.

Regarding the influence of the *composition* of the surfaces: In general the motto is, a profile surface should be as smooth as possible. In some cases, i.e., for profiles with thick laminar separation bubbles, the bubbles can be reduced or entirely eliminated through disturbances on the profile surface. By this means the initial conditions become more favorable for the turbulent boundary layer. A decrease in drag is what follows. Mylar tapes with bumps are sometimes used to achieve the disturbances. The bumps have a height of about 0.040 inches and a distance apart of about 0.12 inches (Ref. 1). Disturbances on the profile surface don't always have to be disadvantageous!

At least in the airfoil nose area, the laminar boundary layer places relatively high demands on the surface roughness. If the critical roughness is overstepped, then we see slightly behind this a transition to turbulent boundary layer. R. Johnson (Ref. 2) claimed that he received better measured results on a not entirely smooth PIK-20 wing, because the laminar separation bubble had been reduced through the surface roughness. On a wind tunnel model, which had a relatively big laminar separation bubble on the upper side, experiments with different surface roughness were made. The point of instability lay at 35% of the chord for $\alpha = 0°$. The profile was sanded and polished originally with wet sandpaper of a granulation of 1200. Afterwards it was roughened with sandpaper of granulations of 400, 320 and 240 in the range between 35 and 60% of the profile chord. After each treatment a polar measurement was made and no change of the polar could be detected. NASA measurements (Ref. 3) proved that practical metal construction profiles covered with a plastic film had the best drag coefficients. According to this knowledge the instability of the laminar boundary layer will not be effected by the surface roughness, as long as it lies under the critical roughness. Since the permissible roughness for turbulent boundary layers is smaller than the critical roughness for laminar boundary layers, we have to pay special attention to the smoothness of the surfaces in the area of the turbulent boundary layer.

Further disturbances on the surface occur through gaps between the wing and the flaps or the aileron and around the airbrakes. **Figure 8** shows measurements on a profile; one without a flap, one with a 17% chord flap and one with a 20% chord flap. Up to a lift coefficient of $c_a = 0.8$ there is no difference in drag coefficient to be seen. The profile with a 20% chord flap nevertheless shows a somewhat greater drag at higher lift coefficients. This is predominantly due to a flow of air through the gap between the wing and the flap which was not sealed. The gaps are located in the area of the turbulent boundary layer on both sides of the profile.

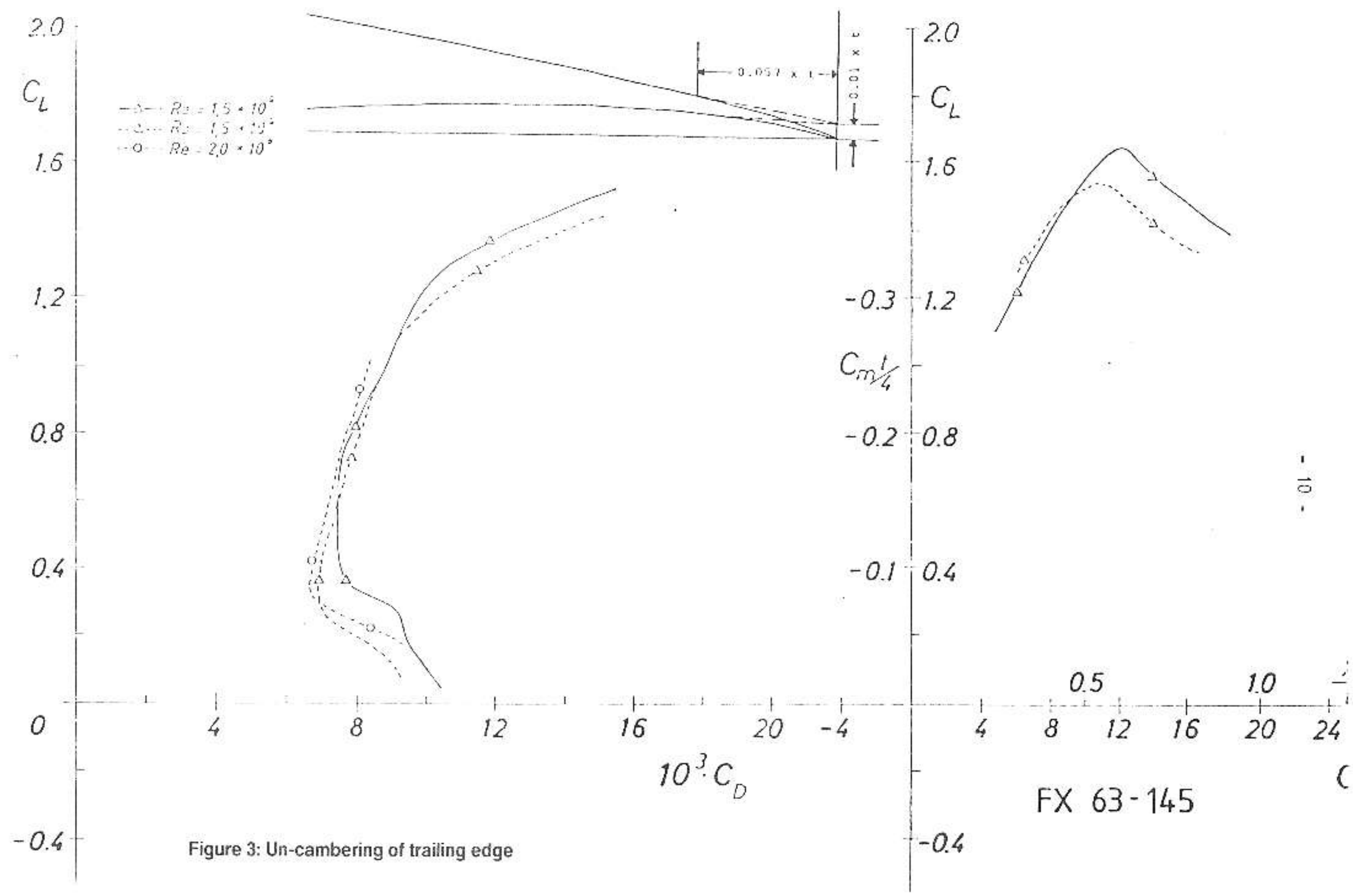

Figure 3: Un-cambering of trailing edge

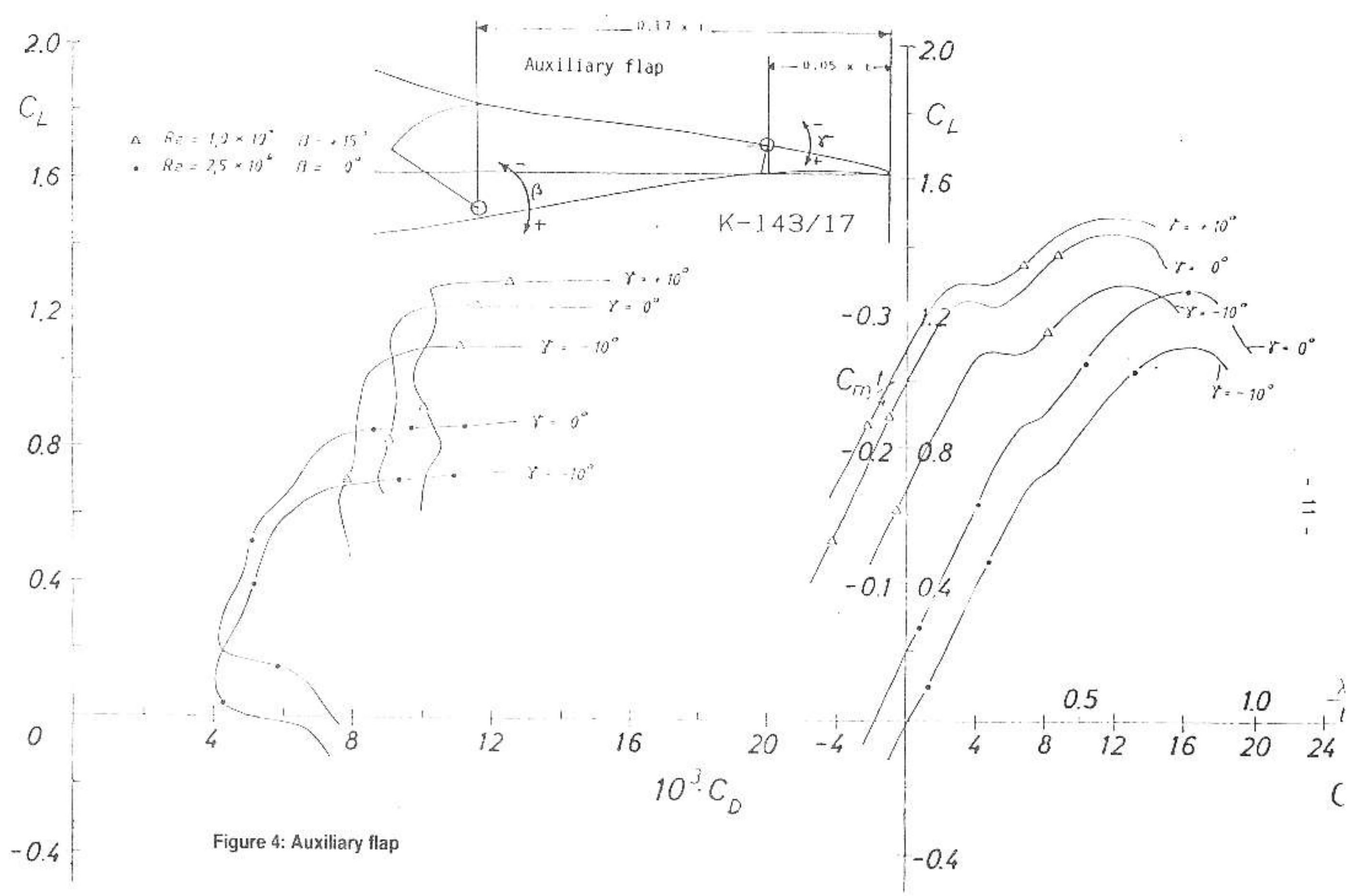

Figure 4: Auxiliary flap

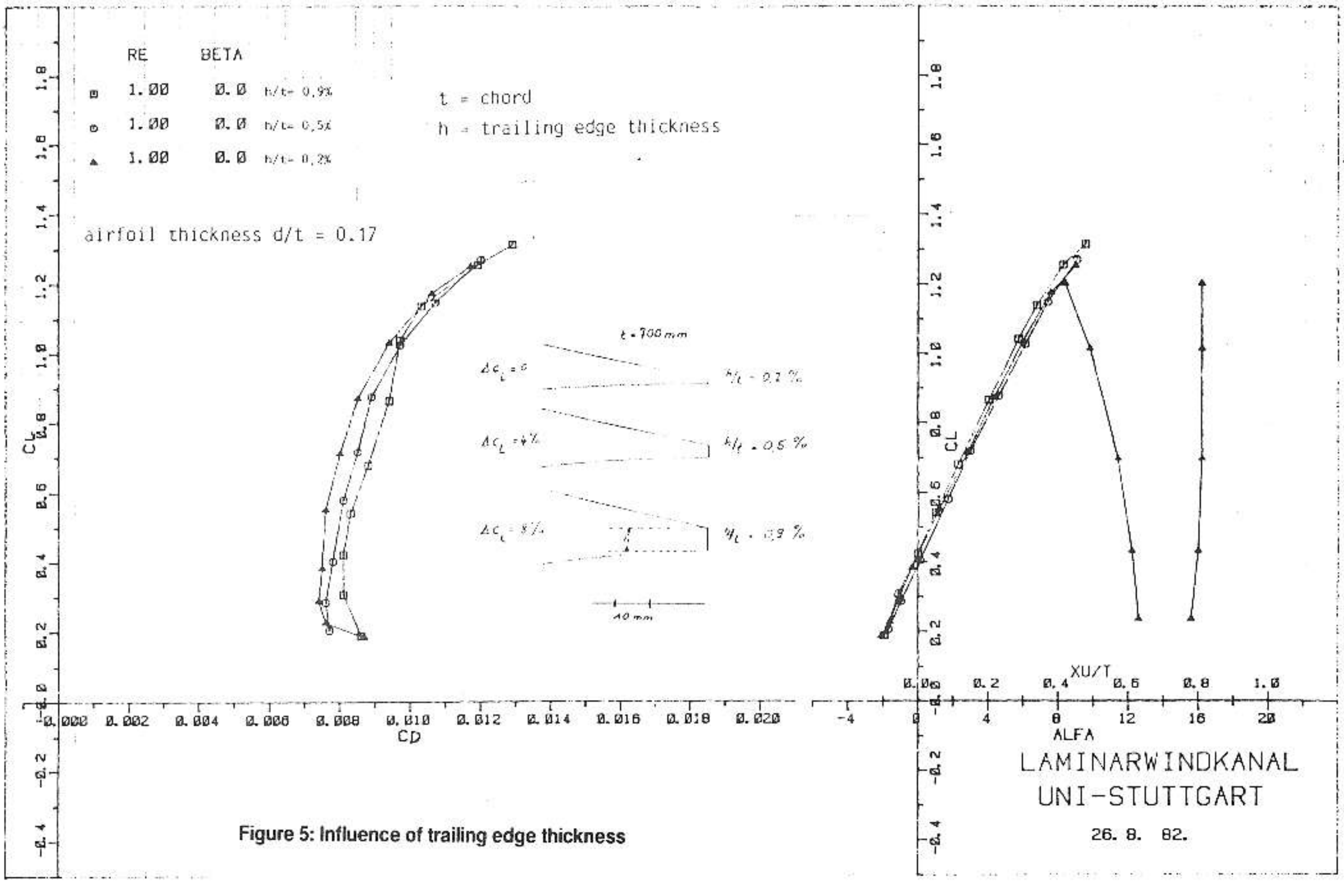

Figure 5: Influence of trailing edge thickness

Figure 9 represents the polars for another profile; one without a flap and one with a 25% chord flap. The profile has a nearly flat underside and, as the measurements of the transition regions show, a completely laminar flow. In this case the transition on the lower side is tripped by the flap space at 75% of the profile chord. This means, like the polars show, an increase in drag. In **Figure 10** the measurements are represented for the same profile with a Reynolds number of Re = 1 million. Here another measurement is illustrated by which the gap was sealed on the upper and lower sides, and furthermore taped over with Scotch tape so that the space between the wing and the flap had been spanned. It is well known how important the sealing of the flap space is on a declined flap or aileron. On tapered wings sealing on the surface is advantageous. The influence of flap gaps on profiles with undeclined flaps can be further investigated in the "Stuttgart Profilkatalog I," in which the measurements of profiles with different flap depths are contained.

R. Eppler (Ref. 4) tried to prove theoretically the influence of such disturbances on profile surfaces. He proposed a so-called "disturbance model" based on the logarithmical law of the wall for turbulent boundary layers. He assumed that at the point of a single imperfection, the momentum loss of the boundary layer increases at the amount of $\Delta\delta_2 = k\pi\iota^{r}\#\nabla u_h \cdot h$, and that the increase of the energy thickness is $\Delta\delta_3 = \Delta\delta_2$. The value k is a proportionality constant, h is the height of the roughness, and u_h is the speed at a distance h from the surface. As long as the disturbance is located in the laminar boundary layer, transition is assumed to take place at the obstruction. As an example, polar calculations of profile E 603 will be listed with the roughness height h/t = 0.001 (a trip wire) on the upper side of the profile at x/t = 0.4 and 0.6 and 0.81.

Figure 11 shows the comparison between wind tunnel measurements and the theory. Listed are the percentage increases in drag corresponding to the drag coefficients without trip wire for Reynolds numbers RE = 1 million and Re = 3 million. The differences between theory and the experiment are considerable, especially for the Re = 3 million case. A doubling of the proportionality constant from 0.15 to 0.3 brings almost a doubling of the respective increase in drag. Likewise we cannot obtain agreements with the experiment.

The mentioned examples show how complex the influence of contour changes or disturbances on the surfaces of wings actually is. However, they could be at least a hint about the impact of some parameters.

References

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3. Somers, Dan: An Exploratory Investigation of the Effect of a Plastic Coating on the Profile Drag of a Practical Metal Construction Sailplane Airfoil. NASA CP 2085, Part I, 1979.
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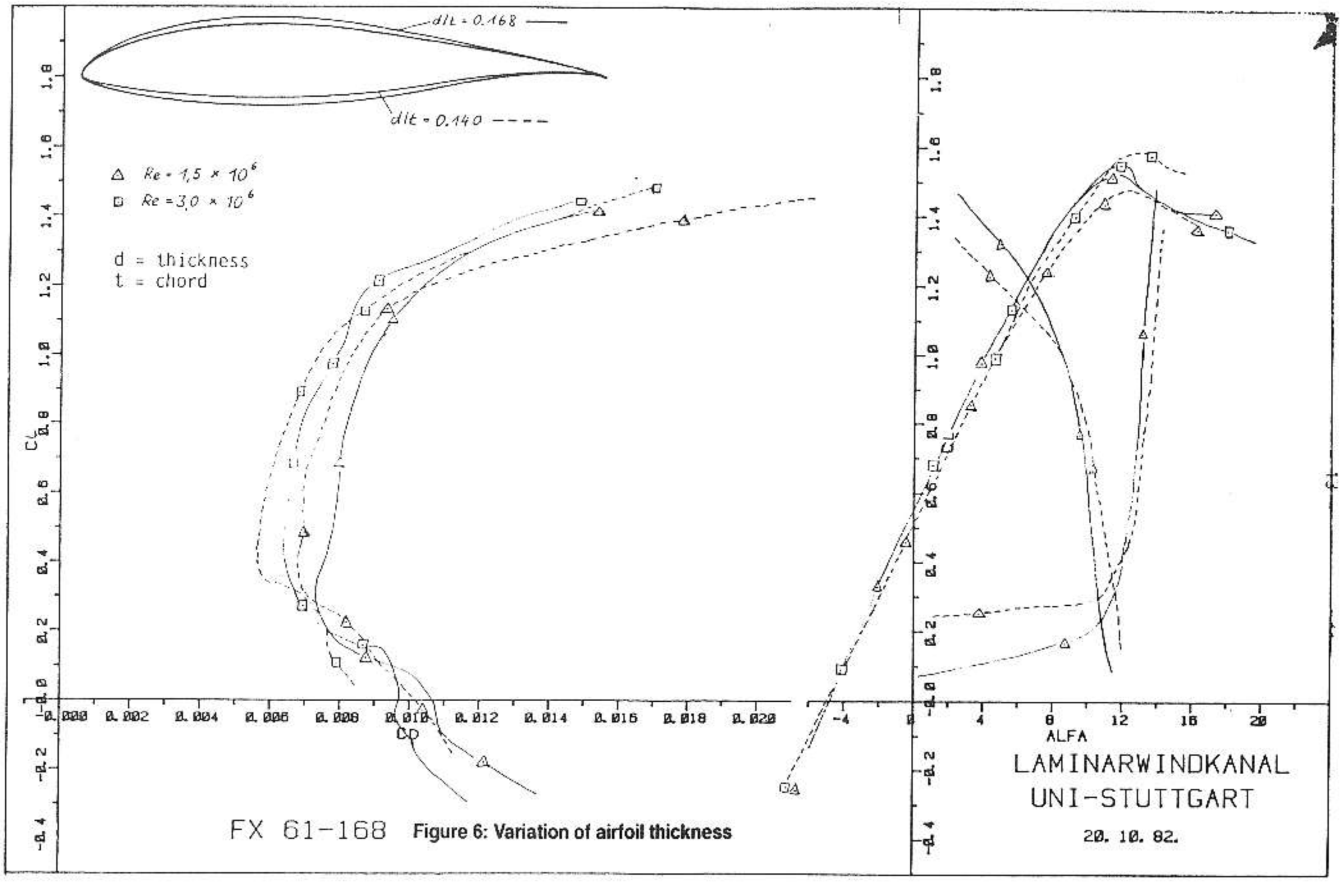

Figure 6: Variation of airfoil thickness

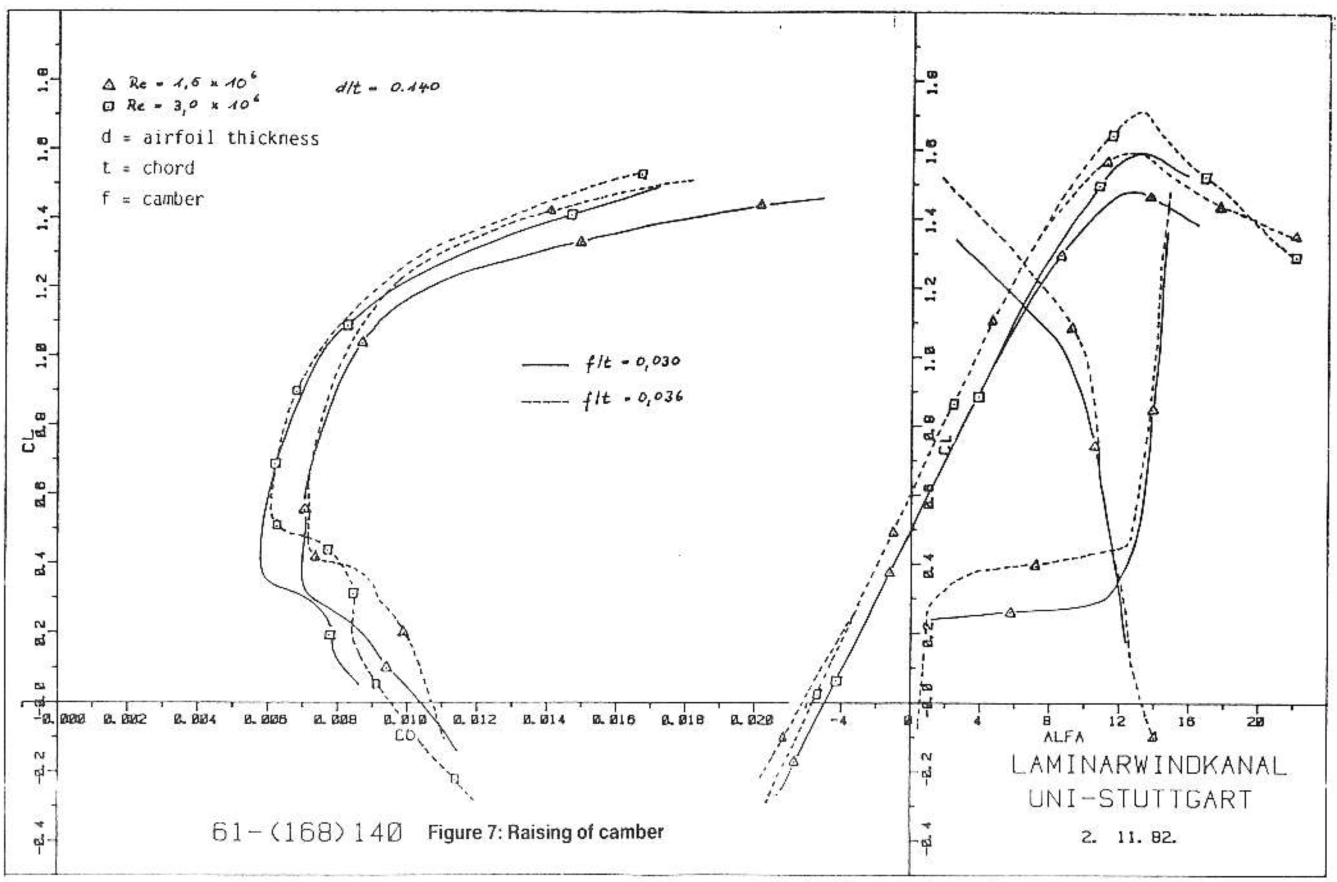

Figure 7: Raising of camber

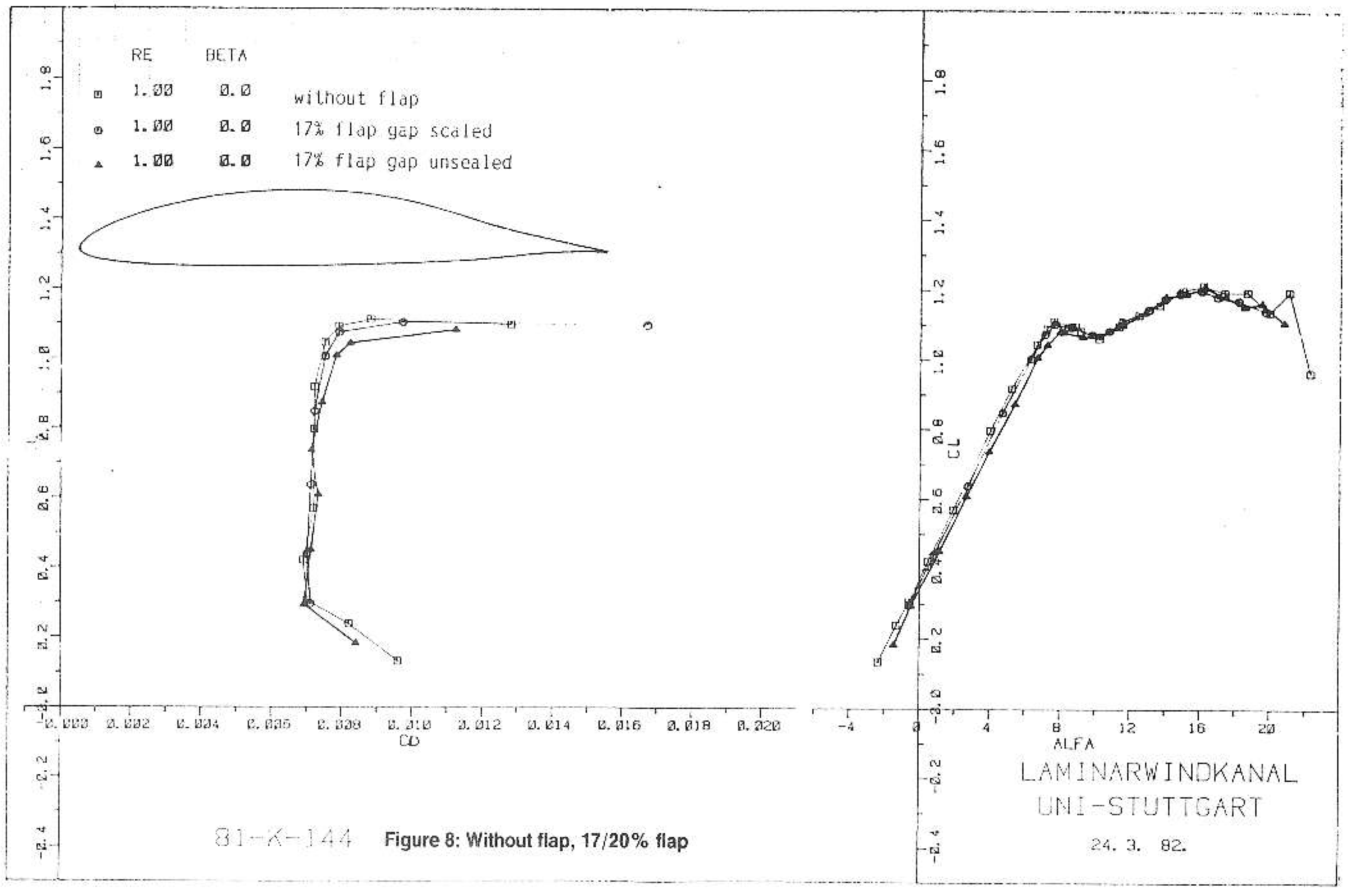

Figure 8: Without flap, 17/20% flap

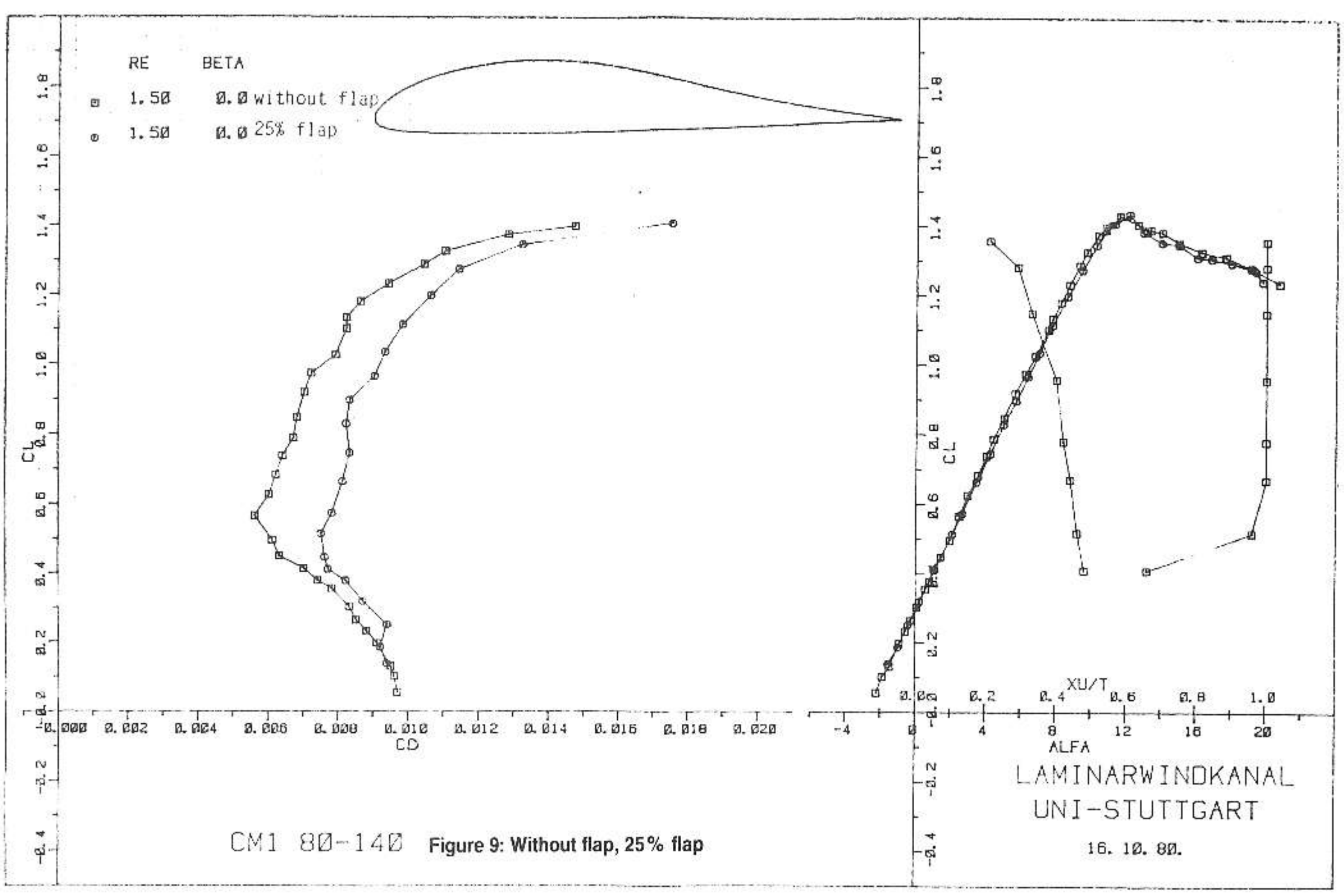

Figure 9: Without flap, 25% flap

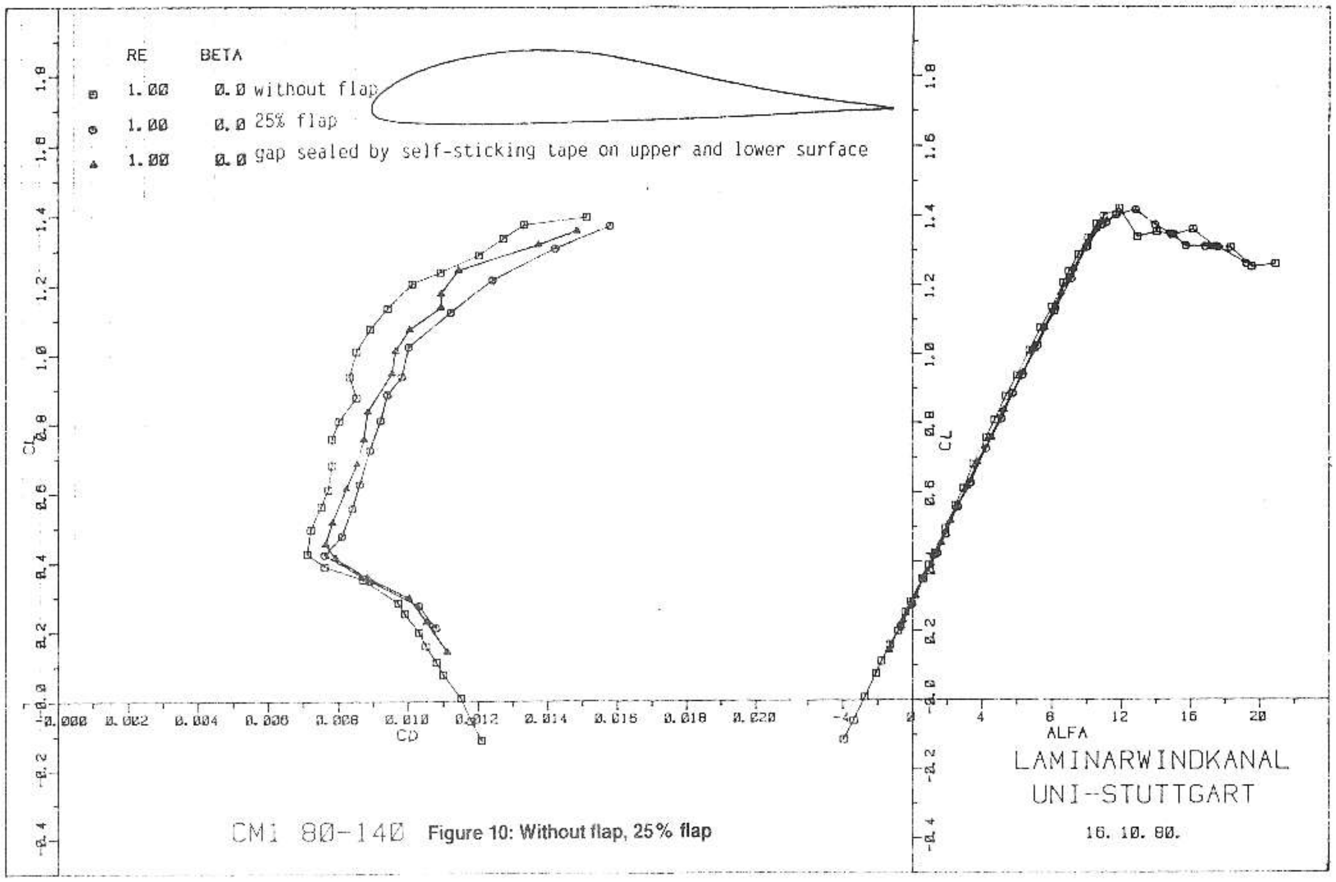

Figure 10: Without flap, 25% flap

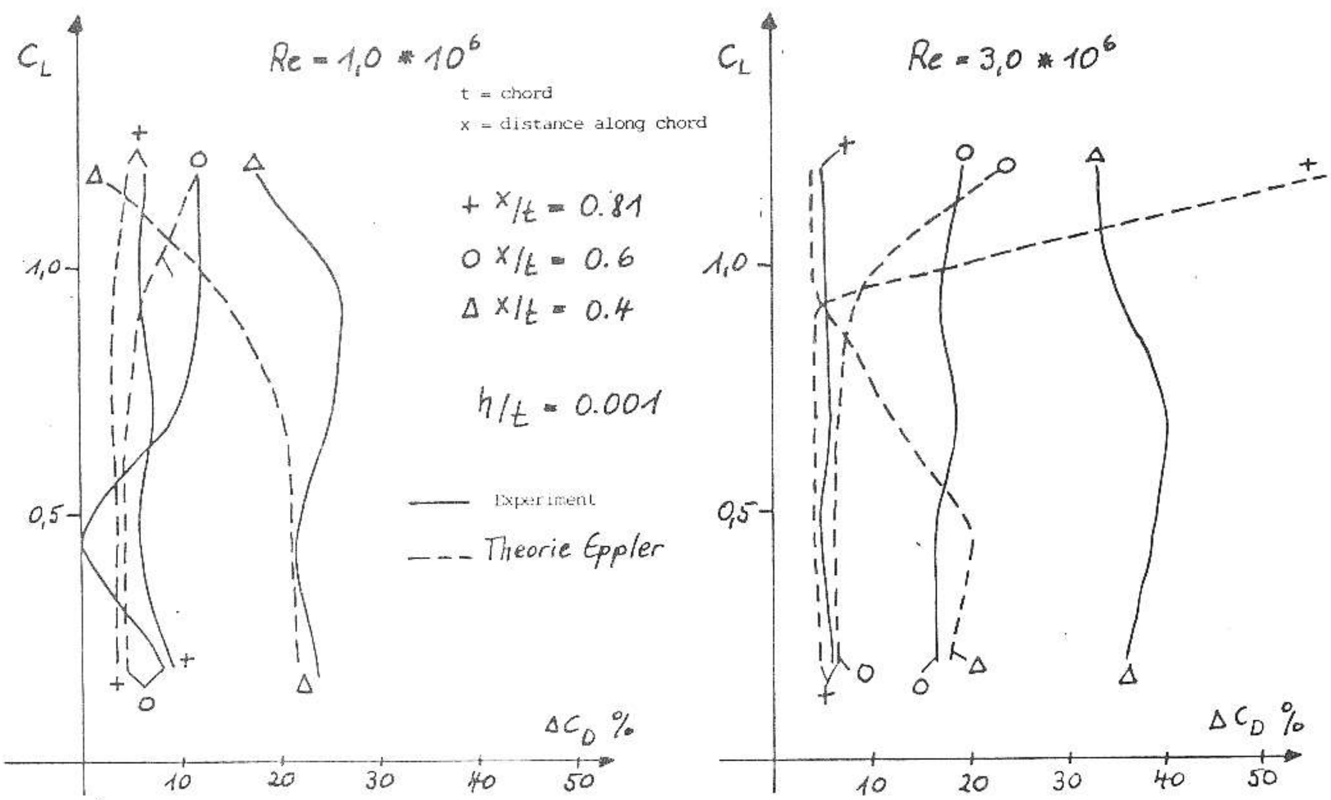

Figure 11: Alteration of drag by a trip wire on the upper airfoil surface