Curiosity Number 2. The curious behavior of a drag polar presented by Schlichting

W. H. Mason, Dec. 10, 2014, rev. June 1, 2018

Schlichting has presented a figure in numerous papers to illustrate the importance of sweepback. Figure 1 is from *Aerodynamics of the Airplane*. The book has a better explanation of the figure than the explanation in some of his survey papers. For the drawbacks to wing sweep read Schlichting's "Third Lancaster Memorial Lecture," cited below. Figure 1 is in our *Applied Computational Aerodynamics* book, Fig. 4.45, pg 232.

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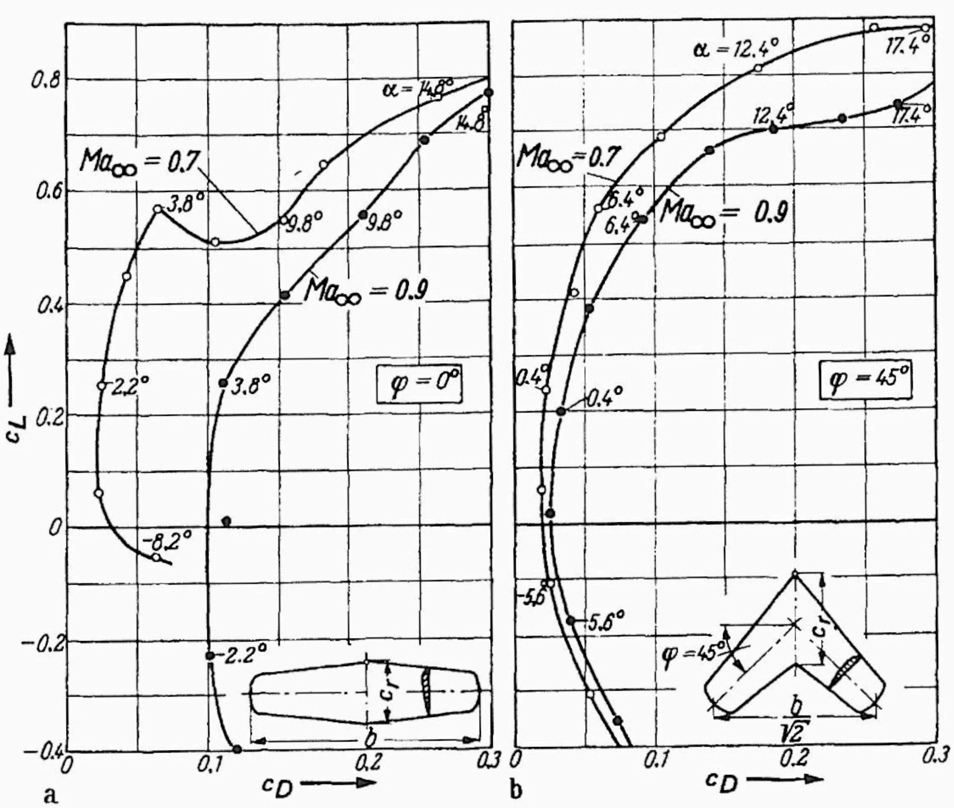

Figure 4-55 Polars, lift coefficient c_L, and drag coefficient c_D at high subsonic incident flow; Mach number $Ma_\infty = 0.7$ and 0.9, for a straight and a swept-back wing of profile Gö 623, from Ludwieg. (*a*) Straight wing, $b = 80$ mm, $c_r = 22.5$ mm; $Re = U_\infty c_r/\nu = 3.0 \cdot 10^5$ at $Ma_\infty = 0.7$, $= 3.5 \cdot 10^5$ at $Ma_\infty = 0.9$. (*b*) Swept-back wing, $\varphi = 45°$, $b' = 57$ mm, $c_r = 32$ mm; $Re = U_\infty c_r/\nu = 4.2 \cdot 10^5$ at $Ma_\infty = 0.7$, $= 5.0 \cdot 10^5$ at $Ma_\infty = 0.9$.

Figure 1. Drag Polar illustrating the importance of wing sweep (from Schlichting)

The point of Fig. 1 is zero lift drag. The unswept wing shows a large drag rise in zero lift drag from $M = 0.7$ to $M = 0.9$, while the swept wing doesn't. The figure works well to make this point.

The feature that interests me here, and the one I overlooked over the years, is the peculiar behavior of the unswept wing at $M = 0.7$ at a C_L from 0.5 to 0.6. What is the basis for the dip in the curve? The C_L reaches a maximum, decreasing and then starting to increase again.

I found the low speed coordinates for this airfoil and the low speed lift curve and drag polar in Riegels. It is the 12% thick Gö 623. The profile is shown in Fig. 2. It's highly cambered and has a flat bottom over the last 70% of the lower surface. The lift curve is shown in Figure 3. The drag polar is given in Figure 4. At low speed everything looks fine.

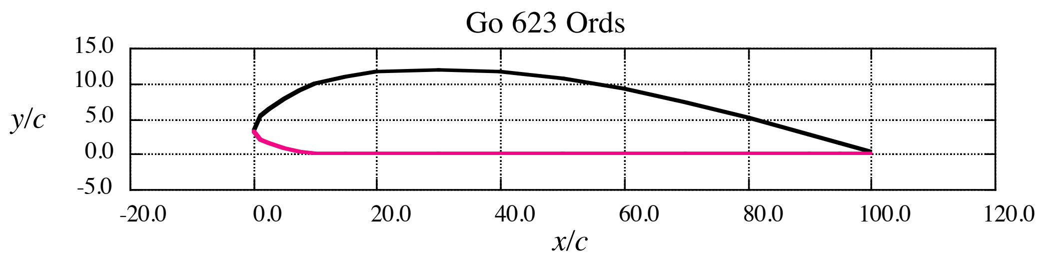

Figure 2. Gö 623 airfoil (from Riegels)

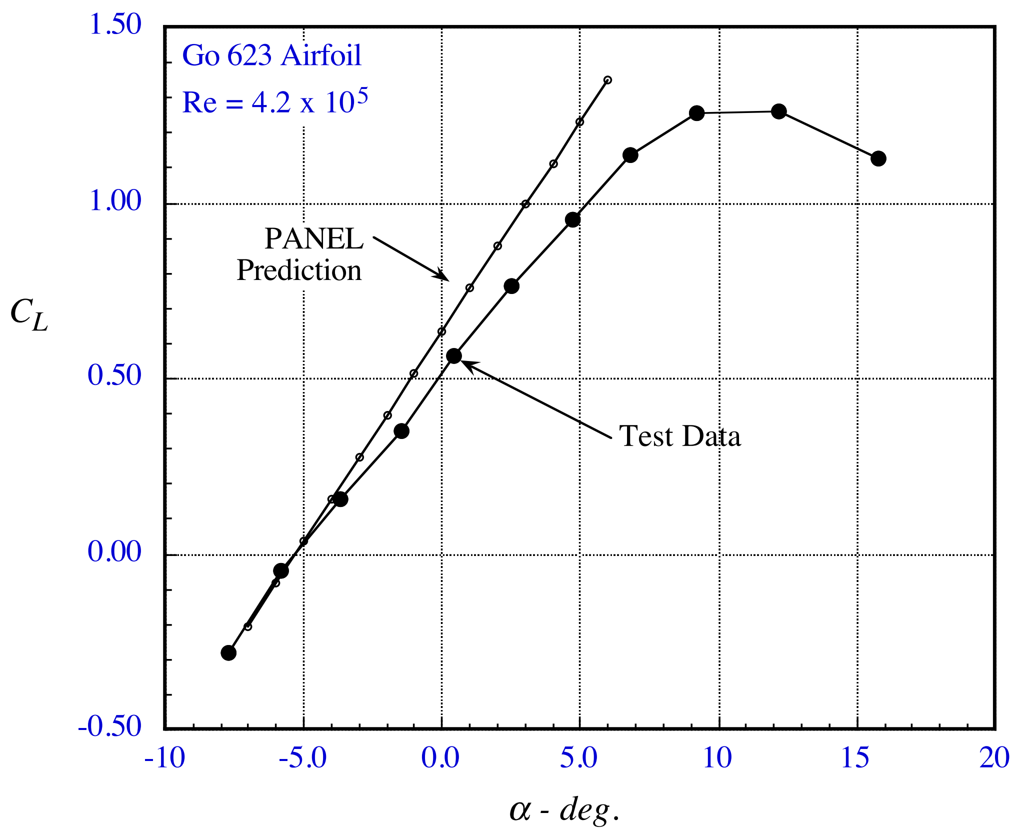

Figure 3. Low speed lift for the Gö 623airfoil (from Riegels)

The agreement with the inviscid prediction from **PANEL** is about what we would expect. The zero lift angle of attack is at between -6 and -5 degrees because of the camber.

Figure 4 shows the low speed airfoil drag polar. Although we didn't compute the drag polar, we see no unusual behavior. Certainly there is nothing to suggest the "dip" seen in Figure 1.

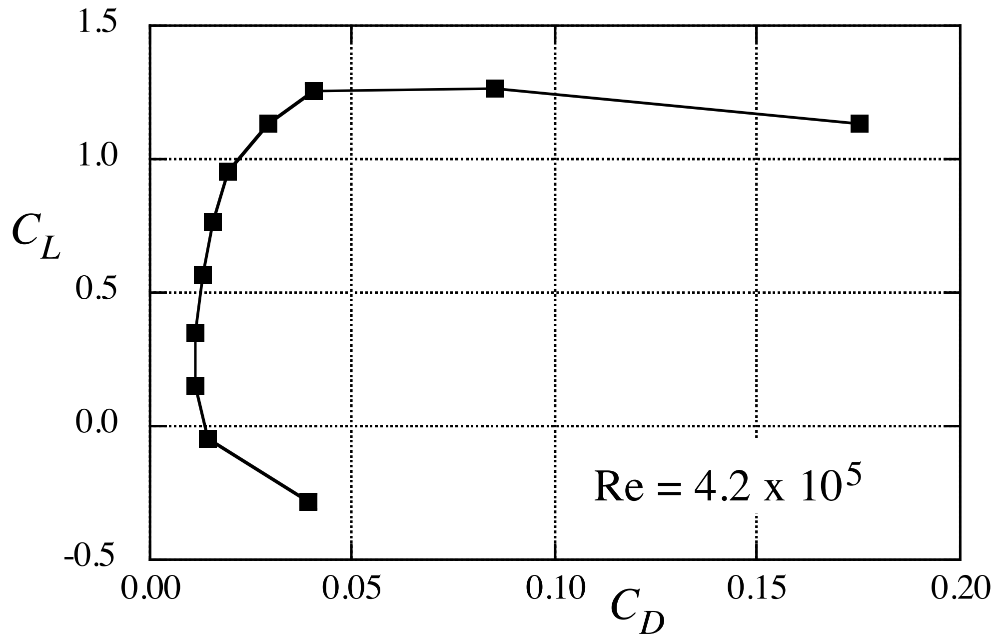

Figure 4. Low speed drag polar for the Gö 623 airfoil (from Riegels).

Since the low speed airfoil drag polar doesn't show anything unusual, we need to examine the compressibility effects. This leads to several interesting observations. We need to make transonic calculations over this airfoil. Although many of the students wouldn't think a Mach number of 0.7 is transonic, it is, especially for a 12% thick airfoil. The Grumman airfoil work that ended up on the X-29 was all done at a Mach number of 0.683. That was a much thinner airfoil than the 12% thick Gö 623.

Although not obvious in Figure 1, the Gö 623 is only defined by 17 upper and 17 lower coordinates, using only a few significant figures. My experience is that before making calculations I need some better coordinates. Although there are airfoil smoothing algorithms available, we really need to enhance the airfoil definition. A few simple trials didn't work out. Meanwhile, this year some design students started using the Clark Y airfoil. I got curiouas. The Clark Y airfoil (designed in 1922) also has a flat bottom over the last 70% chord. I decided to compare the Clark Y and the Gö 623 airfoils. Figure 5 shows the comparison. They are *very* similar. The Clark Y is 11.7% thick.

The next step was to scale the Clark Y to a 12% max *t/c*. Figure 6 shows the comparison between the scaled up Clark Y and the Gö 623 airfoil. Because I found an "enhanced" set of Clark Y coordinates on the web, I decided to use the scaled up Clark Y airfoil as a surrogate for the Go airfoil. There was however a slight complication. The enhanced Clark Y coordinate were rotated such that the leading edge was at (0.0,0.0) rather than corresponding to the "standard" coordinates given in many, many NACA reports among other sources. The rotation also obscures the fact that the lower surface is mostly flat. This required a rotation of the ordinates and rant about this "new standard" for airfoils given in Curiosity 23. Nevertheless, the smoothed ordinates rotated back to the proper orientation were used to make calculations. I used the inviscid code FLO36 because it has no problem at low Mach numbers.

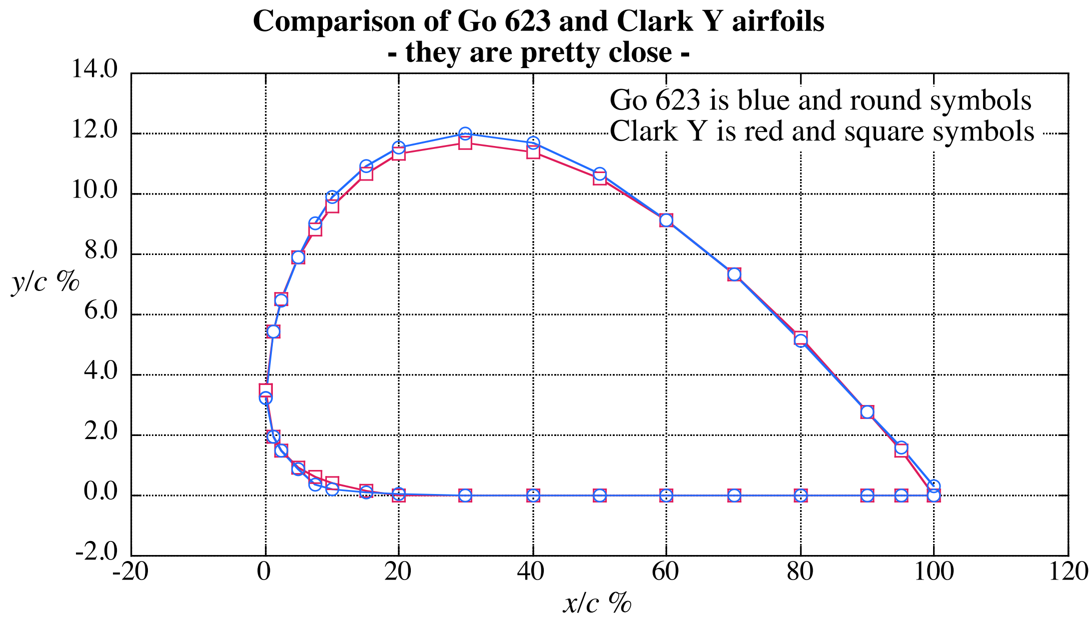

Figure 5. Comparison of Gö 623 and Clark Y airfoils (not to scale).

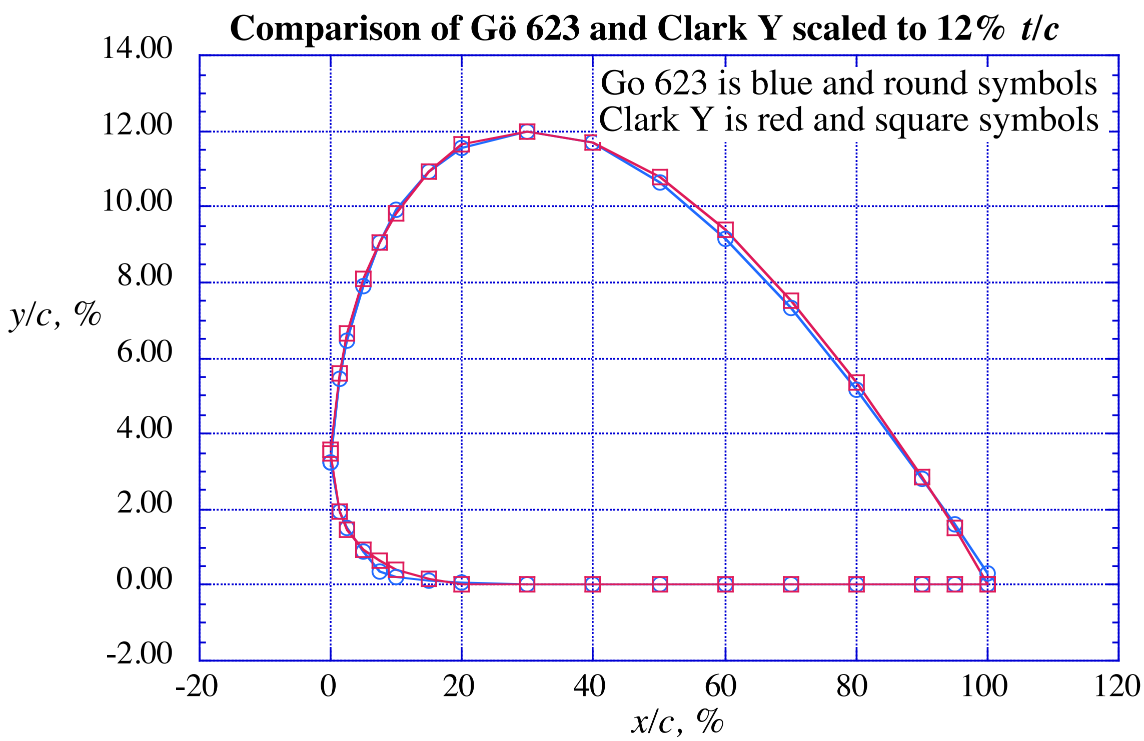

Figure 6. Comparison of Gö 623 and Clark Y airfoil scaled to 12% thick (not to scale).

I would say the main difference, and the reason I had trouble enhancing the Gö 623 is the slight "bump" of the lower surface ordinates near the leading edge. Next, we check the effect of scaling up the Clark Y with a low speed analysis of each airfoil The comparison is shown in Figure 7. At best the difference is slight.

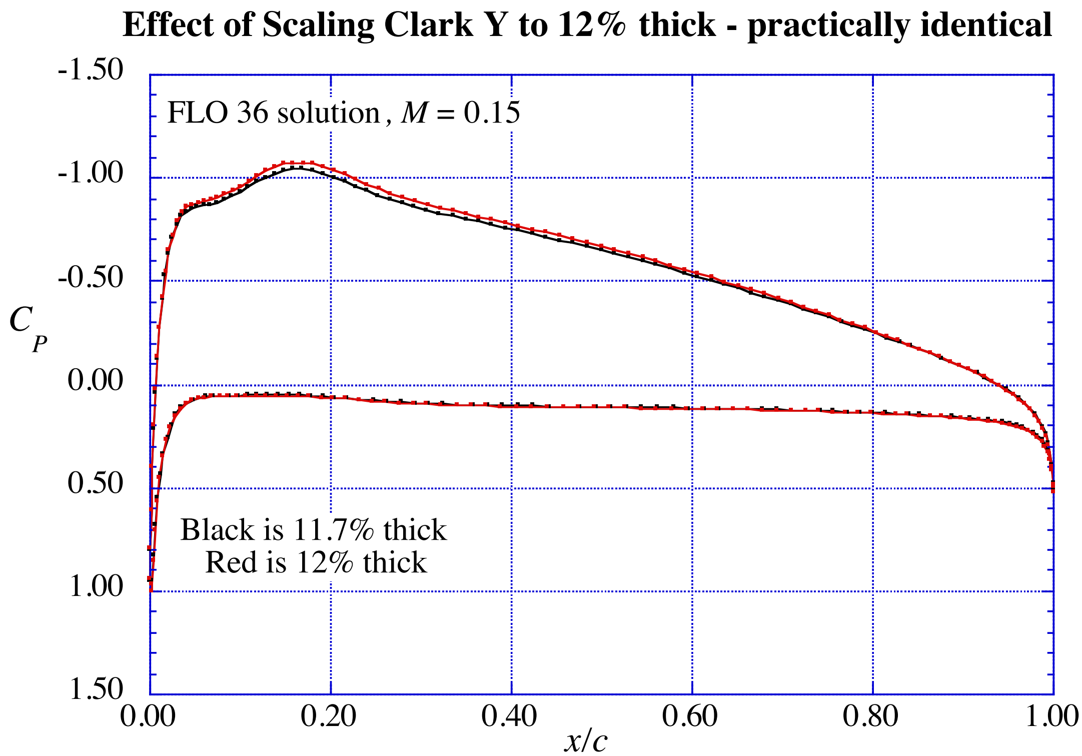

Figure 7. Low speed comparison of Clark Y and Clark Y scaled to 12% thick ($\alpha = 0$, $M = 0.15$)

With this establishing the validity of using the scaled Clark Y as a surrogate for the Gö 623 lets look at the inviscid transonic pressure distributions predicted using FLO36. The best insight into the flow development is to use three angles of attack, -2.5, -2.0, and -1.5 degrees. This is in fact a terrible transonic airfoil, as would be expected! At $\alpha = -1.5$, FLO36 was at the limit of its capability and barely converging.

We have marked the shock Mach number on the plot. Typically a shock Mach number of 1.3 is the limit before the flow separates. Thus the unswept wing is probably experiencing a shock stall as shown with the dip in Figure 1. As far as the Mach 0.9 case in the figure, the wing is likely stalled over the entire range of lift coefficients.

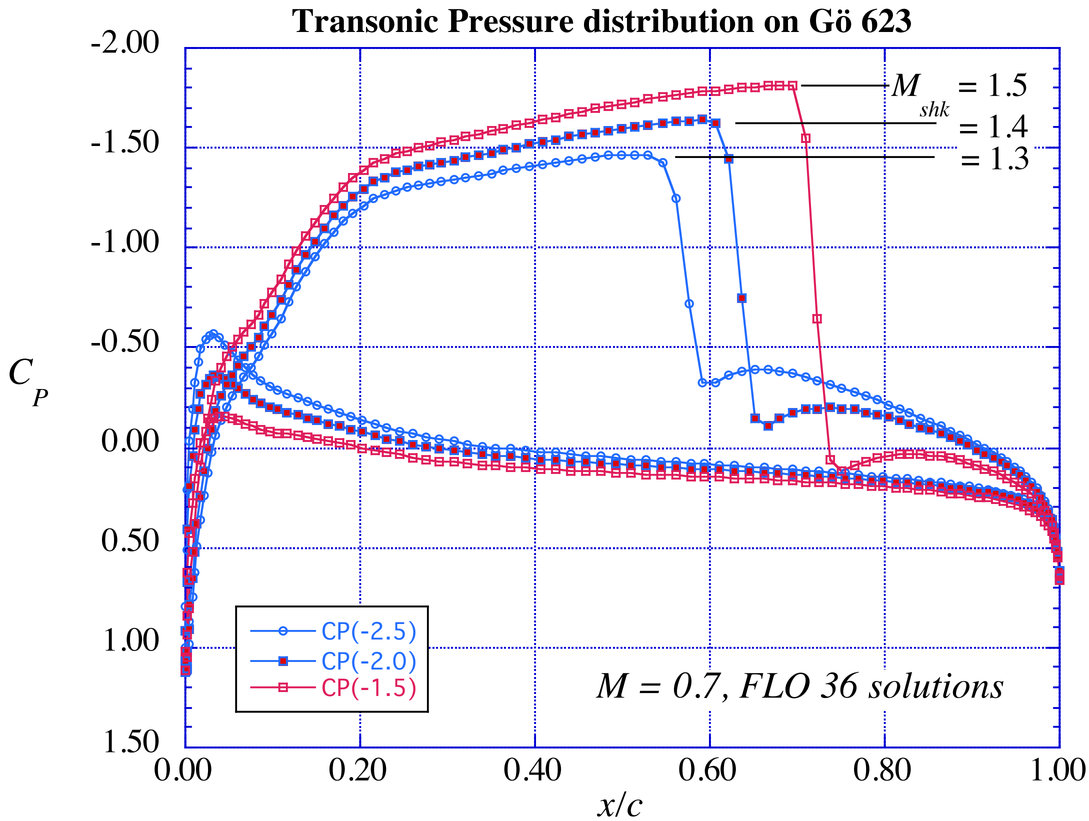

Figure 8. Transonic pressure distribution at $M = 0.7$

As I was looking for more information about Figure 1 I stumbled on a figure in Shapiro that is very similar, apparently from the same facility and makes the same point. It is at $M = 0.8$ and the results don't raise any questions. I've included it here as Figure 9.

Observations/Further work

- The Gö 623 and Clark Y are essentially identical airfoils.
- The dip is likely due to shock stall.
- I have good enough definition of the geometry that would allow a complete 3D RANS calculation to be made.
- Any future revision of our book should use the figure in Shapiro in place of the figure from Schlichting. That figure doesn't raise any questions.
- Russ Cummings sent me the original report for the Schlichting figure. The wingspan is 8 cm, so the model is small. I couldn't tell anything about the test since I don't read German.

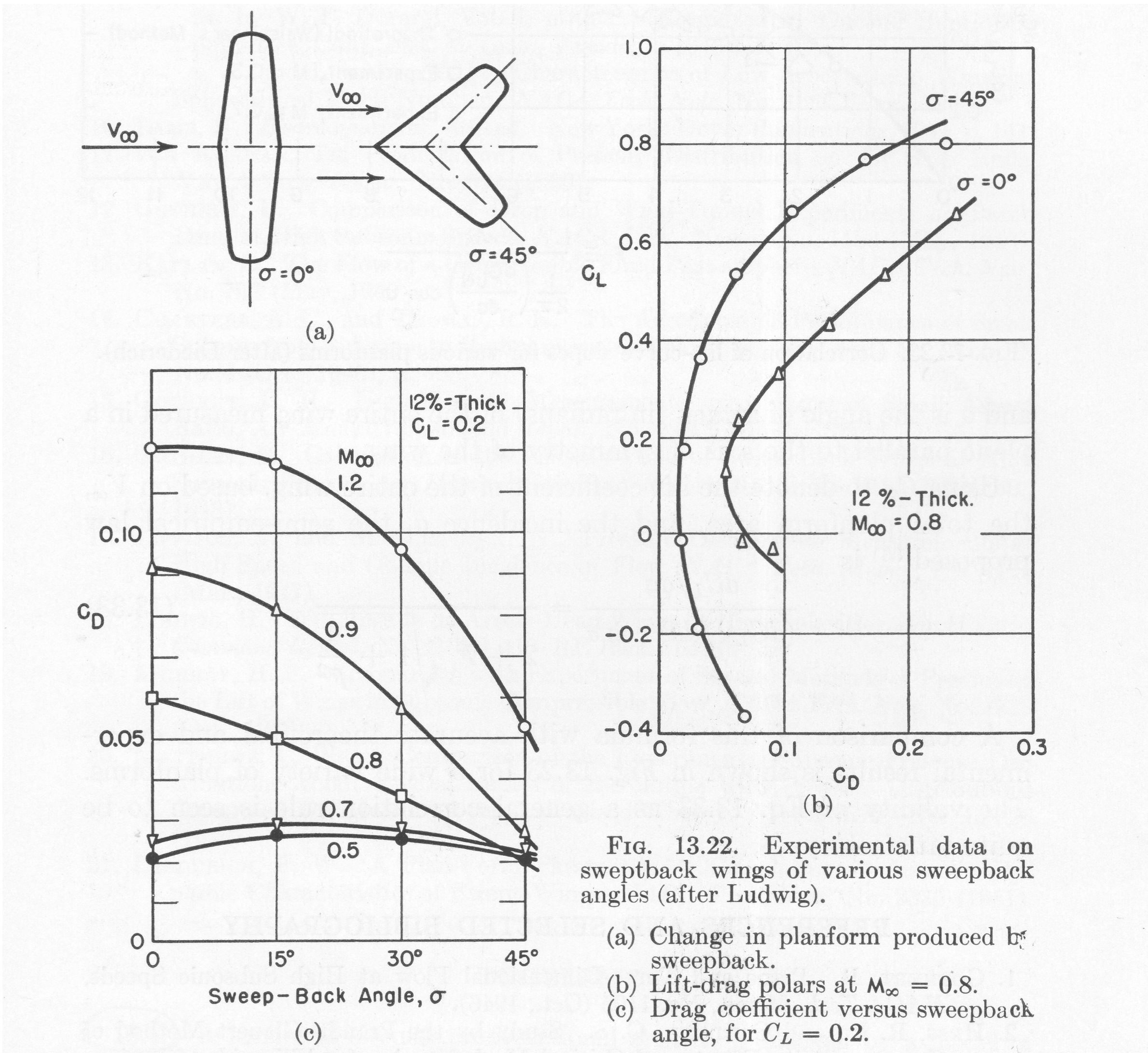

FIG. 13.22. Experimental data on sweptback wings of various sweepback angles (after Ludwig).
(a) Change in planform produced by sweepback.
(b) Lift-drag polars at $M_\infty = 0.8$.
(c) Drag coefficient versus sweepback angle, for $C_L = 0.2$.

Figure 9. A similar figure to Fig 1, contained in Shapiro.

Sources

Herman Schlichting and Erich Truckenbolt, *Aerodynamics of the Airplane*, translated by Heinrich J. Ramm, McGraw Hill, New York, 1979.

H. Schlichting, Third Lancaster Memorial Lecture, “Some Developments in Boundary Layer Research in the Past Thirty Years,” *Journal of the Royal Aeronautical Society*, Feb. 1960.

Friedrich Wilhelm Riegels, *Airfoil Sections*, translated by D. G. Randall, Butterworths, London, 1961.

Ludwig, H., “Test Results on Arrow-Head Wings at High Velocities,” *Air Material Command Report*, No. F-TS-417-RE, Oct. 1946 (undoubtedly a translation from the German), cited in Ascher H. Shapiro, *The Dynamics and Thermodynamics of Compressible Flow*, The Ronald Press, New York, 1953, Vol 1, Fig. 13.22, pg 421.