The Gö 622, Gö 623, Gö 624 and Gö 625 airfoils with thickness/chord ratios of respectively 8 %, 12 %, 16 % and 20 % for use in windmill rotor blades

ing. A. Kragten

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Engineering office Kragten Design
Populierenlaan 51
5492 SG Sint-Oedenrode
The Netherlands
telephone: +31 413 475770
e-mail: info@kdwindturbines.nl
website: www.kdwindturbines.nl

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1 Introduction

In report R443D (ref. 1) of the former Wind Energy Group of the University of Technology Eindhoven, about eighty airfoils are assembled which have been measured for low Reynolds numbers. A problem with this report is that it is no longer available and that most of the given airfoil graphs are too small for use in rotor blade calculations. Fortunately the original measuring points are given for almost all airfoils and using these points, new accurate graphs can be made.

Airfoils with a flat lower side are of interest for windmill rotor blades, especially if they are manufactured from massive wood. The Gö 623 airfoil which has a maximum thickness of 12 % of the chord is used in all my present VIRYA-windmills with wooden blades. The characteristics and geometry of this airfoil are given in my report KD 35 (ref. 2). Sometimes thinner or thicker airfoils are needed. The four airfoils Gö 622, Gö 623, Gö 624 and Gö 625 with thickness/chord ratios of respectively 8 %, 12 %, 16 % and 20 % have originally been measured by Riegels (ref. 3). These measurements are also given in R445D (ref. 1). The C_l-α and C_l-C_d curves are generally given for three Reynolds values at page 3-70 of R445D. Tables with measuring points are given at page 3-72 of R445D but only for two values of Reynolds $1.1 * 10^5$ (or $1.2 * 10^5$) and $4.2 * 10^5$ (exept for the Gö 625 which has been measured for many Reynolds values). The curves for $2.2 * 10^5$ (or $2.3 * 10^5$) are estimated by copying the given curves as accurate as possible.

2 The Gö 622, Gö 623, Gö 624 and Gö 625 airfoil geometry

The airfoil geometry for all four airfoils is copied from the tables given on page 3-76 of R445D. The distance x is the value from the airfoil nose for an airfoil with a chord of 100 mm. The distance y_u is the corresponding value for the upper part of the airfoil. The distance y_l is the corresponding value for the lower part of the airfoil.

	Gö 622		Gö 623		Gö 624		Gö 625	
x (mm)	y_u (mm)	y_l (mm)	y_u (mm)	y_l (mm)	y_u (mm)	y_l (mm)	y_u (mm)	y_l (mm)
0	2.40	2.40	3.25	3.25	4.00	4.00	5.50	5.50
1.25	3.75	1.45	5.45	1.95	7.15	2.25	9.00	3.30
2.5	4.50	1.05	6.45	1.50	8.50	1.65	10.80	2.35
5.0	5.45	0.60	7.90	0.90	10.40	0.95	13.30	1.25
7.5	6.15	0.35	9.05	0.35	11.75	0.60	14.95	0.75
10	6.60	0.25	9.90	0.20	12.85	0.40	16.35	0.40
15	7.30	0.15	10.95	0.10	14.35	0.15	18.25	0.15
20	7.70	0.05	11.55	0.05	15.30	0.05	19.30	0.10
30	8.00	0	12.00	0	16.00	0	20.00	0
40	7.80	0	11.70	0	15.40	0	19.05	0
50	7.10	0	10.65	0	14.05	0	17.35	0
60	6.15	0	9.15	0	12.00	0	15.05	0
70	5.00	0	7.35	0	9.50	0	12.10	0
80	3.55	0	5.15	0	6.60	0	8.60	0
90	1.95	0	2.80	0	3.55	0	4.75	0
95	1.15	0	1.60	0	2.00	0	2.75	0
100	0.20	0	0.30	0	0.50	0	0.65	0

table 1 Geometry of the Gö 622, Gö 623, Gö 624 and Gö 625 airfoils for a chord c = 100 mm

The y_u-values for x =100 are very small. Sometimes the airfoil tailing edge is taken somewhat thicker to prevent damage of the blades during transport. It is expected that the effect of this increase of the thickness on the airfoil characteristics, can be neglected.

It may be expected that the airfoil thickness of a thicker airfoil is found by multiplying the y-factors of a thinner airfoil by a certain value but this is not true as the y-value of the Gö 624 for x = 0 is not exactly double the y-value of the Gö 622. The four airfoils are given on scale in figure 1, 2, 3 and 4.

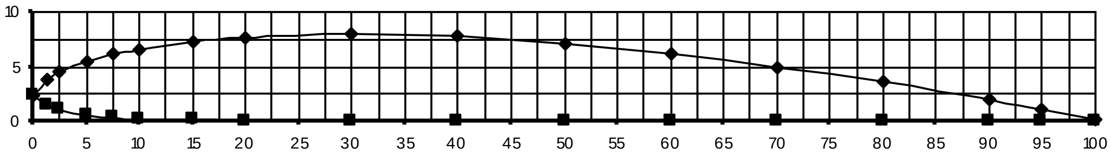

fig. 1 Gö 622 airfoil for c = 100 mm

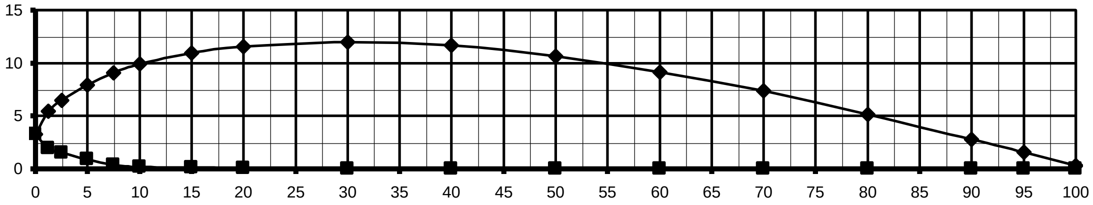

fig. 2 Gö 623 airfoil for c = 100 mm

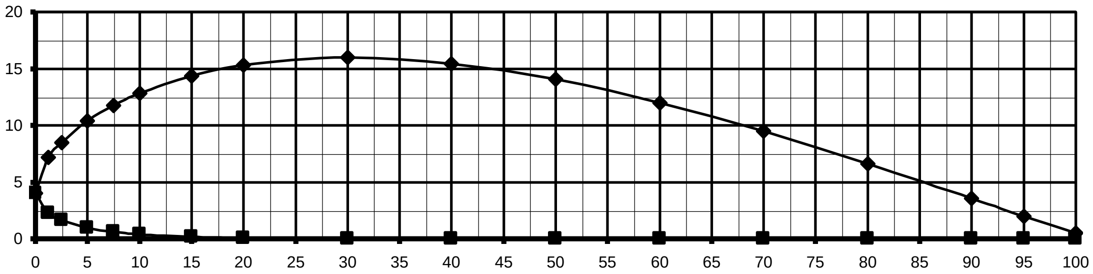

fig. 3 Gö 624 airfoil for c = 100 mm

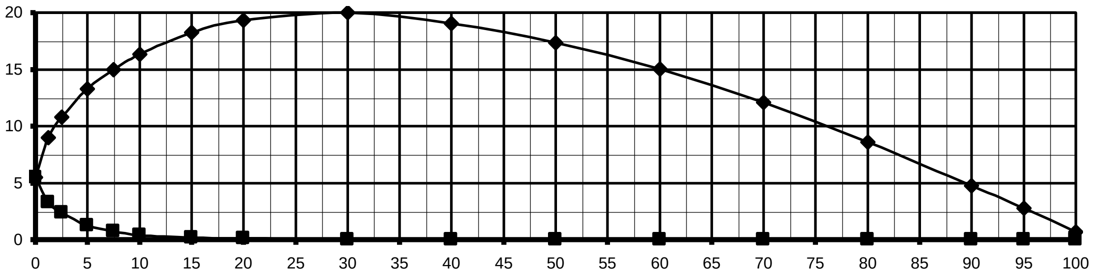

fig. 4 Gö 625 airfoil for c = 100 mm

If the real blade chord c is a factor i larger than 100 mm, all the x-values and y-values of figure 4 have to be multiplied by the same factor i. The strength of a beam is determined by the moment of resistance W which is proportional to the thickness h^2. The ratio of the strength of the four airfoils with the same chord is therefore 1 : 2.25 : 4 : 6.25. The thicker airfoils are normally only used near the blade root because the bending moment is largest there.

3 The Gö 622, Gö 623, Gö 624 and Gö 625 airfoil characteristics

The C_l-α curves are given in figure 5, 7, 9 and 11. The C_l-C_d curves are given in figure 6, 8, 10 and 12. α is the angle of attack. C_l is the lift coefficient. C_d is the drag coefficient. The moment coefficient C_m around the quart chord point was also measured by Riegels for some airfoils and for some Reynolds values. But for the calculation of the geometry of a windmill rotor, only the C_l-α and the C_l-C_d curves are of interest and so the C_m-α curves are not given.

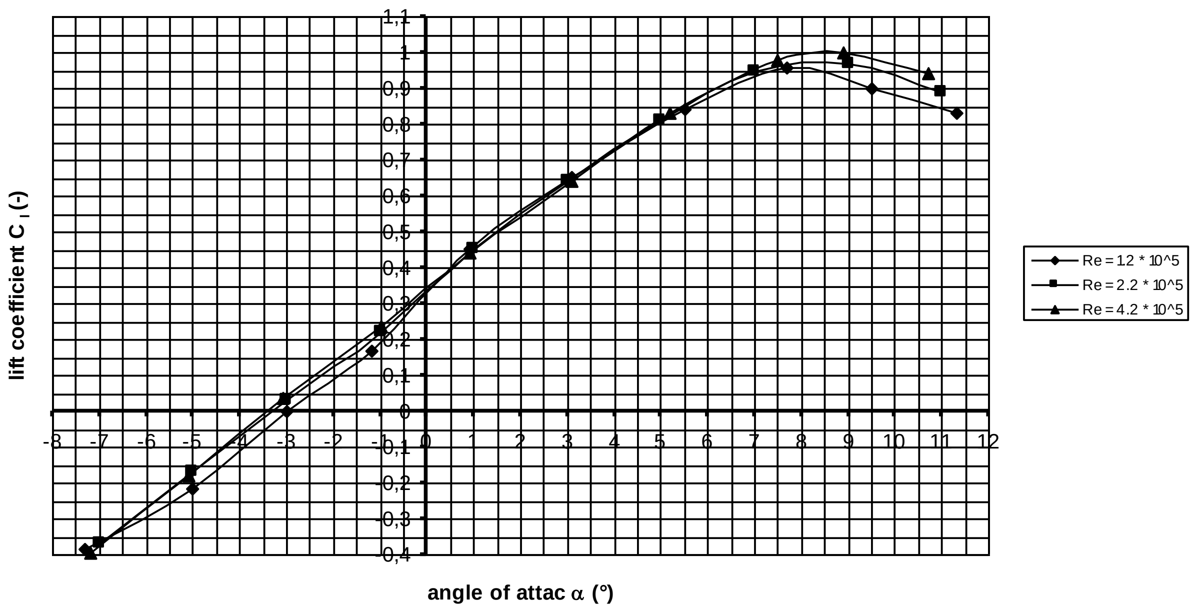

fig. 5 C_l-α curves for the Gö 622 for Re = 1.2 * 10^5, 2.2 * 10^5 and 4.2 * 10^5

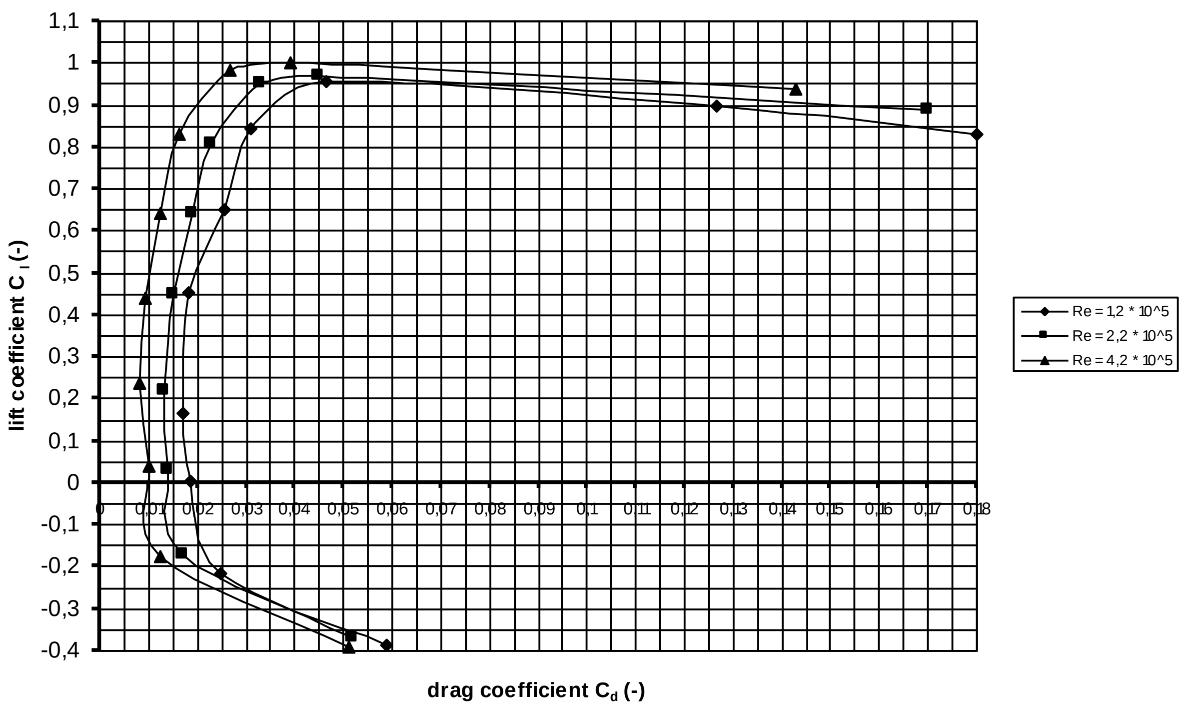

fig. 6 C_l-C_d curves for the Gö 622 for Re = 1.2 * 10^5, 2.2 * 10^5 and 4.2 * 10^5

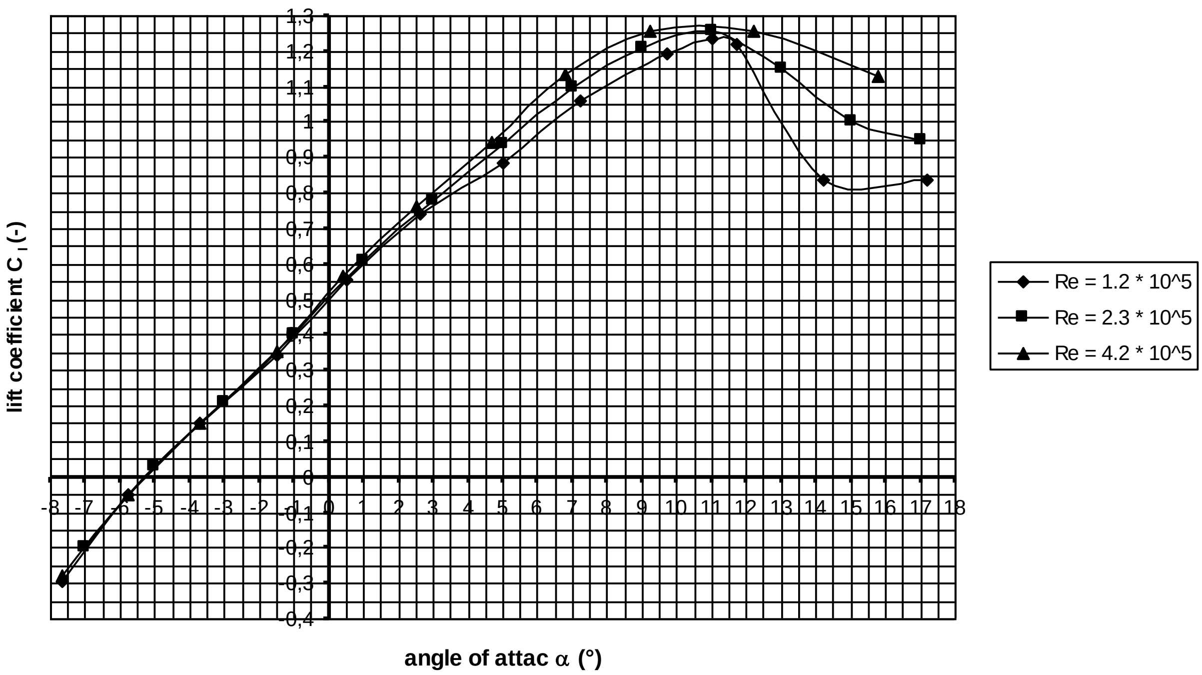

fig. 7 C_l-α curves for the Gö 623 for Re = 1.2 * 10^5, 2.3 * 10^5 and 4.2 * 10^5

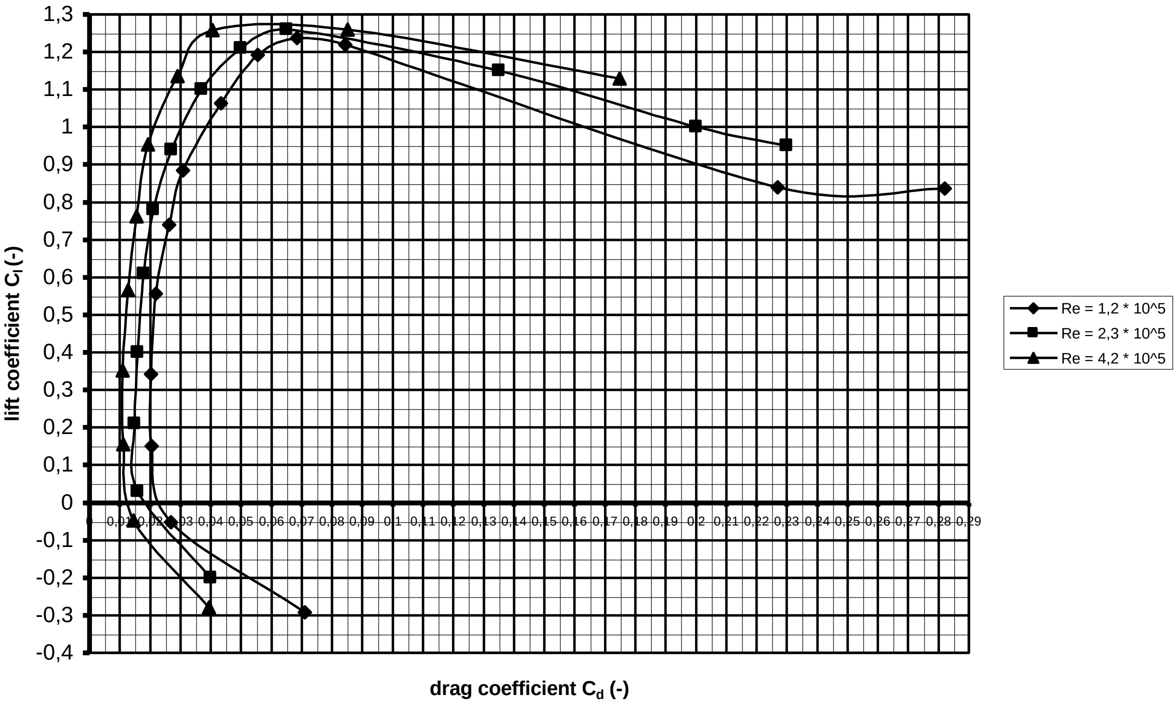

fig. 8 C_l-C_d curves for the Gö 623 for Re = 1.2 * 10^5, 2.3 * 10^5 and 4.2 * 10^5

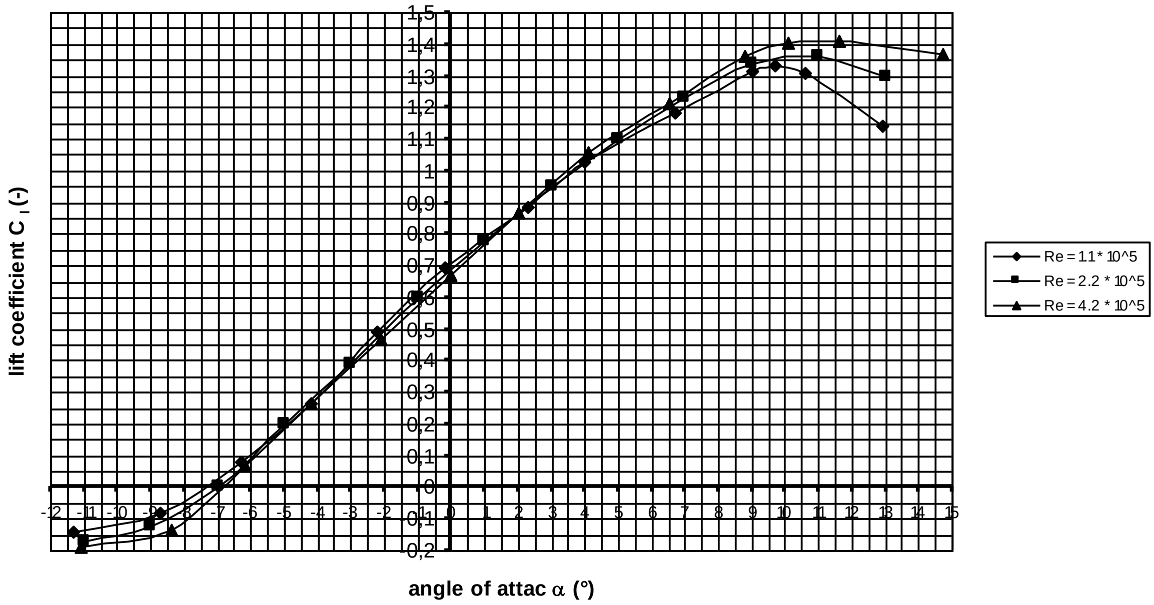

fig. 9 C_l-α curves for the Gö 624 for Re = 1.1 * 10^5, 2.2 * 10^5 and 4.2 * 10^5

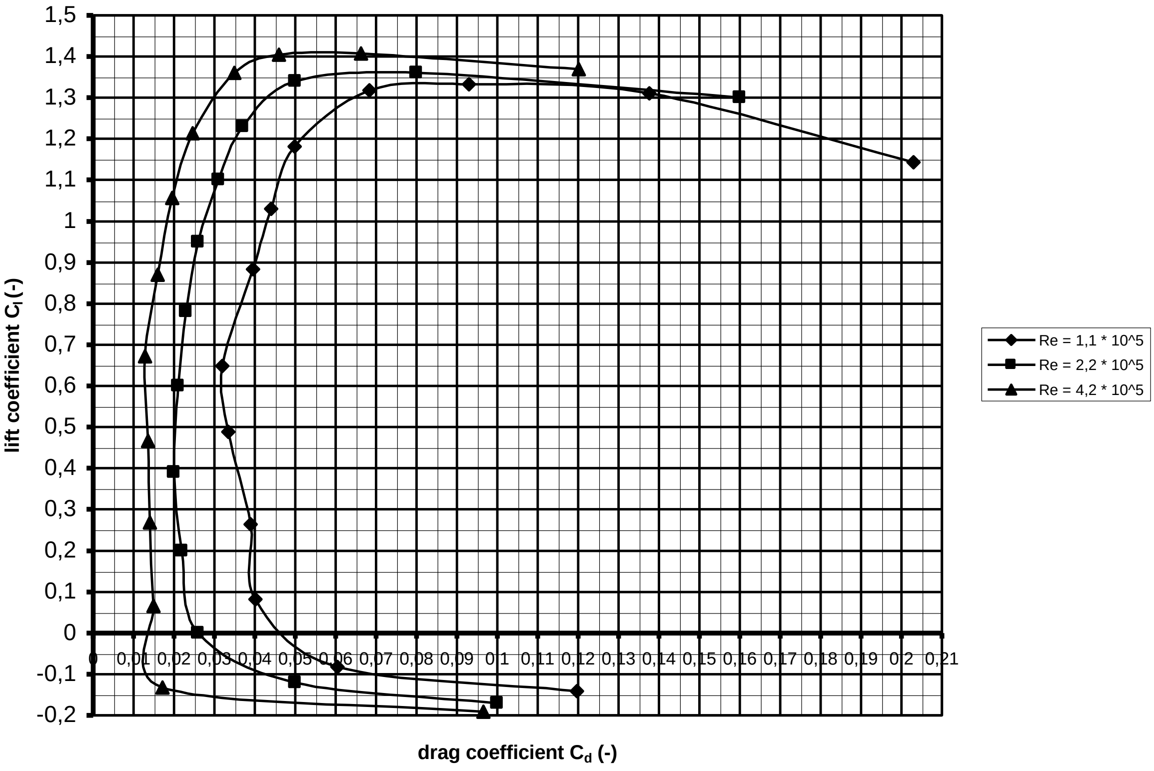

fig. 10 C_l-C_d curves for the Gö 624 for Re = 1.1 * 10^5, 2.2 * 10^5 and 4.2 * 10^5

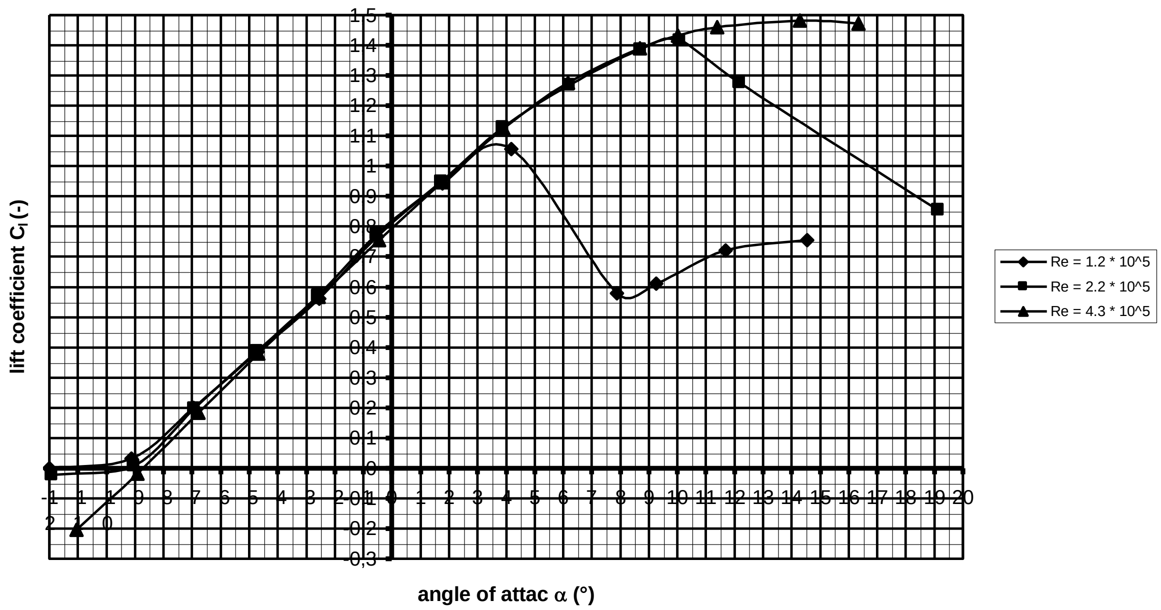

fig. 11 C_l-α curves for the Gö 625 for Re = 1.2 * 10^5, 2.2 * 10^5 and 4.3 * 10^5

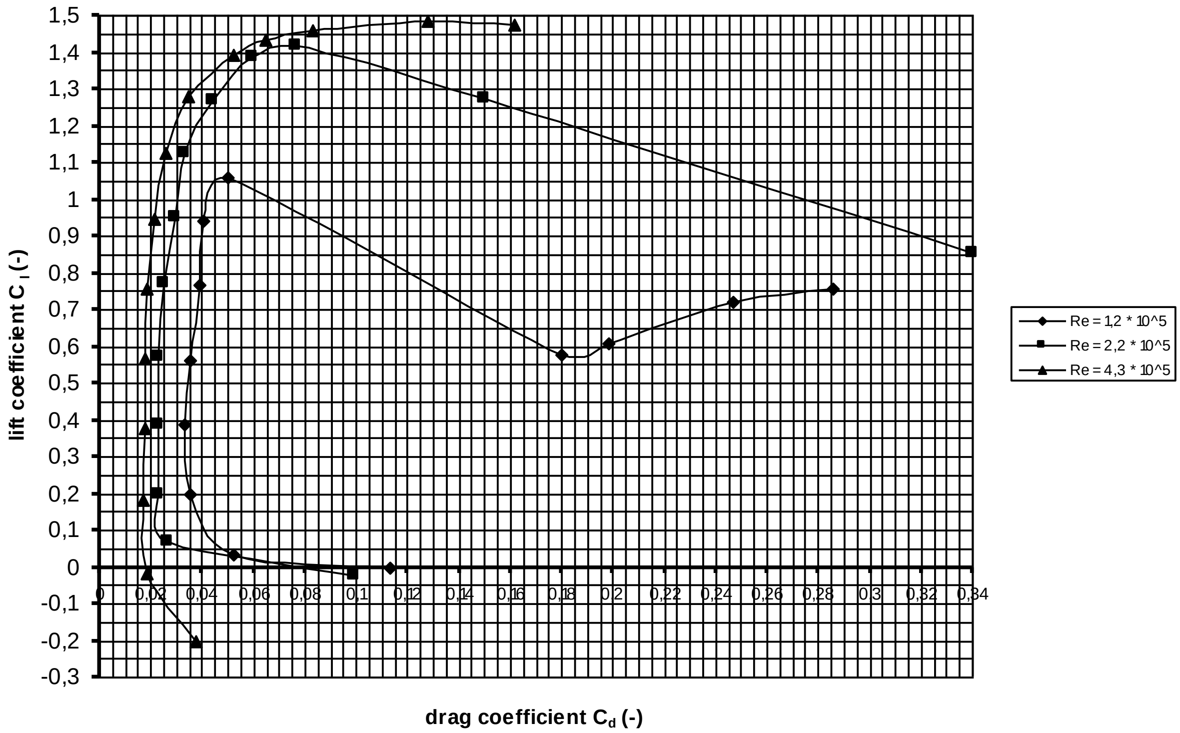

fig. 12 C_l-C_d curves for the Gö 625 for Re = 1.2 * 10^5, 2.2 * 10^5 and 4.3 * 10^5

In figure 11 it can be seen that the Gö 625 airfoil is stalling at rather low values of α for low Reynolds values. Stalling means that the C_l-value drops suddenly at increasing α.

The minimum C_d/C_l ratio for a certain Reynolds value is found by drawing a line through the origin which touches the C_d-C_l curve for that Reynolds value. This line intersects with the line C_l = 1. The C_d value for the point of intersection is the minimum C_d/C_l ratio. The thinnest airfoils have the lowest minimum C_d/C_l ratio for the highest Reynolds value.

4 Determination of C_{mh} for the Gö 623 for Re = 4.2 * 10^5

Sometimes a pitch control system is used with which the blade angle (or blade pitch) β is varied to influence the starting behaviour or to limit the rotor speed at high wind speeds. Variation of β means variation of the angle of attack α. The pitch control system is influenced by the aerodynamic moment M and the aerodynamic moment depends on the position of the axis of rotation around which the blade angel β is varied. The aerodynamic moment coefficient is normally given around the quart-chord point which lies on the zero line of the airfoil at a distance of 0.25 c from the airfoil nose. But it is possible to determine the moment coefficient around another point H. The moment coefficient arount point H is called C_{mh}.

The Gö 623 is the most commonly used airfoil of the given series of airfoils for so C_{mh} is determined for the Gö 623 airfoil. The moment coefficient around the quart chord point is called $C_{m0.25}$. The moment coefficient is given in report R443D (ref. 1) only for a Reynolds value of Re = 4.2 * 10^5. The original measuring points as given on page 3-72 of R443D are copied in table 1.

α (°)	C_l (-)	C_d (-)	$C_{m0.25}$ (-)	($C_l \cos\alpha + C_d \sin\alpha$)	$C_{m0.3}$ (-)	$C_{m0.35}$ (-)	$C_{m0.4}$ (-)
-7.7	-0.279	0.0394	-0.0155	-0.2818	-0.0296	-0.0437	-0.0578
-5.8	-0.047	0.0146	-0.0762	-0.0482	-0.0786	-0.0810	-0.0834
-3.7	0.155	0.0111	-0.0724	0.1540	-0.0647	-0.0570	-0.0493
-1.5	0.352	0.0110	-0.0700	0.3516	-0.0524	-0.0348	-0.0173
0.4	0.566	0.0128	-0.0681	0.5661	-0.0398	-0.0115	0.0168
2.5	0.763	0.0156	-0.0678	0.7630	-0.0297	0.0085	0.0467
4.7	0.954	0.0192	-0.0665	0.9524	-0.0189	0.0287	0.0764
6.8	1.135	0.0290	-0.0644	1.1304	-0.0079	0.0486	0.1052
9.2	1.258	0.0406	-0.0572	1.2483	0.0052	0.0676	0.1300
12.2	1.260	0.0852	-0.0644	1.2495	-0.0019	0.0606	0.1230
15.8	1.129	0.1749	-0.0902	1.1340	-0.0335	0.0232	0.0799

table 1 C_l, C_d, $C_{m0.25}$, $C_{m0.3}$, $C_{m0.35}$ and $C_{m0.4}$ as a function of α. Re = 4.2 * 10^5

The Gö 623 airfoil and the lift L, the drag D and $M_{0.25}$ acting on it, is given in figure 13. M_h is given around point H. Point H is lying at a distance p from the quart chord point and the line through the quard chord point and point H makes a clock wise angel γ with the flat lower side of the air foil (see figure 13).

The right hand direction of $M_{0.25}$, M_h, $C_{m0.25}$ and C_{mh} is defined positive. As $C_{m0.25}$ is negative for the whole range of α values, it means that the real direction of $M_{0.25}$ is such that it has a tendency to decrease the angel of attack α. The real direction of M_h depends on the value of p/c.

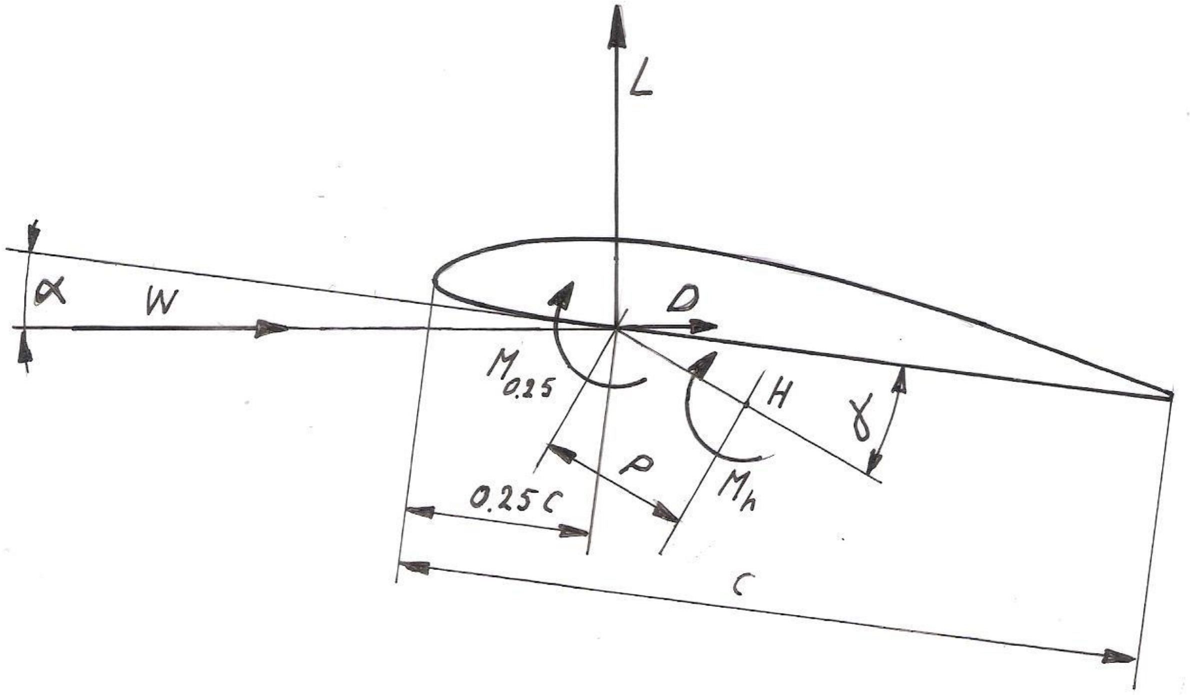

fig. 13 L, D, $M_{0.25}$ and M_h around point H at a distance p from the quard chord point

It can be proven that the moment coefficient around point H, C_{mh} is given by:

$C_{mh} = C_{m0.25} + p/c \{C_l \cos(\alpha + \gamma) + C_d \sin(\alpha + \gamma)\}$ (-) (1)

If point H is lying on the zero line of the airfoil, $\gamma = 0°$. So for this condition formula 1 changes into:

$C_{mh} = C_{m0.25} + p/c (C_l \cos\alpha + C_d \sin\alpha)$ (-) (for $\gamma = 0°$) (2)

If point H is lying on the zero line at a distance 0.3 c from the nose, p = 0.3 c – 0.25 c = 0.05 c. So p/c = 0.05. This value of C_{mh} is called $C_{m0.3}$. So $C_{m0.3}$ is given by:

$C_{m0.3} = C_{m0.25} + 0.05 (C_l \cos\alpha + C_d \sin\alpha)$ (-) (3)

If point H is lying on the zero line at a distance 0.35 c from the nose, p = 0.35 c – 0.25 c = 0.1 c. This value of C_{mh} is called $C_{m0.35}$. So $C_{m0.35}$ is given by:

$C_{m0.35} = C_{m0.25} + 0.1 (C_l \cos\alpha + C_d \sin\alpha)$ (-) (4)

If point H is lying on the zero line at a distance 0.4 c from the nose, p = 0.4 c – 0.25 c = 0.15 c. This value of C_{mh} is called $C_{m0.4}$. So $C_{m0.4}$ is given by:

$C_{m0.4} = C_{m0.25} + 0.15 (C_l \cos\alpha + C_d \sin\alpha)$ (-) (5)

The calculated values for ($C_l \cos\alpha + C_d \sin\alpha$) and for $C_{m0.3}$, $C_{m0.35}$ and $C_{m0.4}$ are also given in table 1. The $C_{m0.25}$-α curve, the $C_{m0.3}$-α curve, the $C_{m0.35}$-α curve and the $C_{m0.4}$-α curve can be derived from table 1. These curves are given in figure 14.

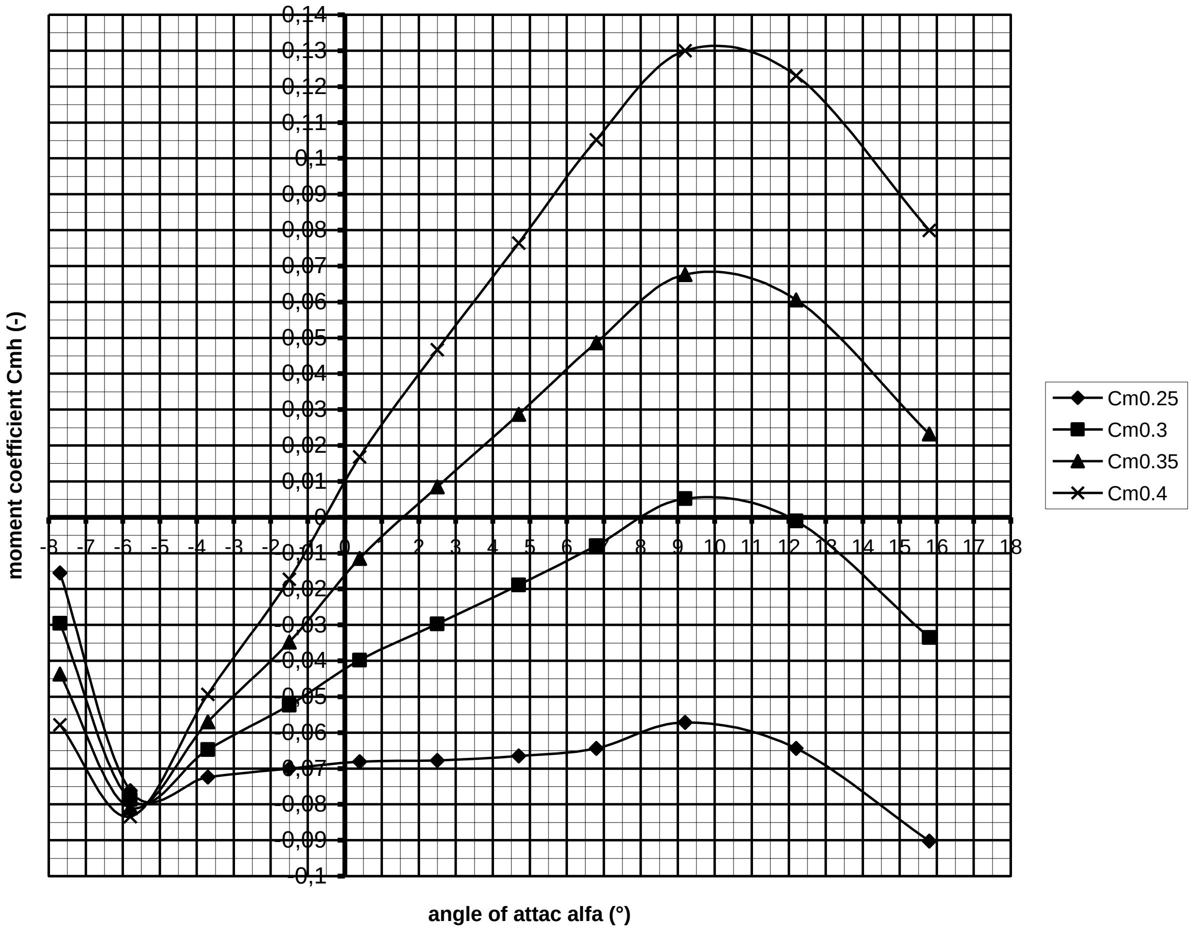

fig. 14 $C_{m0.25}$-α curve, $C_{m0.3}$-α curve, $C_{m0.35}$-α curve and $C_{m0.4}$-α curve for the Gö 623 airfoil for Re = $4.2 * 10^5$

5 References

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3 Riegels F. W. Aerodynamische Profile (in German), Oldenbourg, R. München, 1958.