

User Guide for the Diederich Method Excel-File

1 Introduction

The following instructions are for a practical understanding of the Diederich Method Excel Workbook. This teaching tool can be used for straight tapered swept and unswept wings

1. ... to calculate **lift distribution** along the wing span,
2. ... to calculate the spanwise **distribution of the lift coefficient**,
3. ... to calculate the **maximum lift coefficient of the wing** (based on the given maximum lift coefficients of the root and tip airfoils, which are blended over the span).

In addition, several other parameters useful for wing design are determined.

Please read the **project report** of Suna Arslan (2013) in English, if you want more background of the method. You may want to read the project report of Max Schnoor (2021), if you prefer to read in German. Please note, the updated project version is the one in English.

Raymond F. Anderson establishes in **1937** already much, what was later taken up by other like Diederich. Anderson already distinguishes between "basic lift distribution" (from twist) and "additional lift distribution" due to change of angle of attack. He also includes twist into the calculation of induced drag. His document is "Determination of the Characteristics of Tapered Wings", NACA-TR-572.

Franklin W. Diederich described the method in **1952** in NACA Technical Note 2751 "A Simple Approximate Method for Calculating Spanwise Lift Distributions and Aerodynamic Influence Coefficients at Subsonic Speeds". This text is not so easy to read and ends with the statement: "The integration indicated in equation (11) may be performed graphically or numerically".

Egbert Torenbeek (1988) takes this a step further with the method expressed in simple equation and no more need for numerical integration. Torenbeek adds the influence coefficient C_4 . Please see Torenbeek's book *Synthesis of Subsonic Airplane Design*, pages 467-476.

Please find the Excel Workbook of the Diederich Method, this User Guide, and all links to mentioned reports and books on

<http://Diederich.ProfScholz.de>

It is assumed, the reader has a general understanding of wing design and aerodynamics. If a refresher is needed, please refer to Section 7 "Wing Design" on <http://LectureNotes.AircraftDesign.org> in the Aircraft Design Lecture Notes of **Dieter Scholz** (2015).

These are the tabs of the workbook:

- Atmosphere Parameters (temperature, density, dynamic viscosity, kinematic viscosity, speed of sound)
- Lift Curve Slope ($c_{L,\alpha}$ of the tip and root airfoil)
- Factors of Diederich's Method (F, C1, C2, C3, C4)
- Sweep Correction Function (f)
- Mason's Curve (optimum lift distribution)
- Diederich's Method

The main page of the workbook is the **tab** called "**Diederich's Method**". It covers everything needed. Other tabs provide supporting information. Please see the project report (Arslan 2013) for comments on the other tabs.

2 Diederich's Method in Excel

Figure 1 shows the most important top left corner of the tab "Diederich's Method".

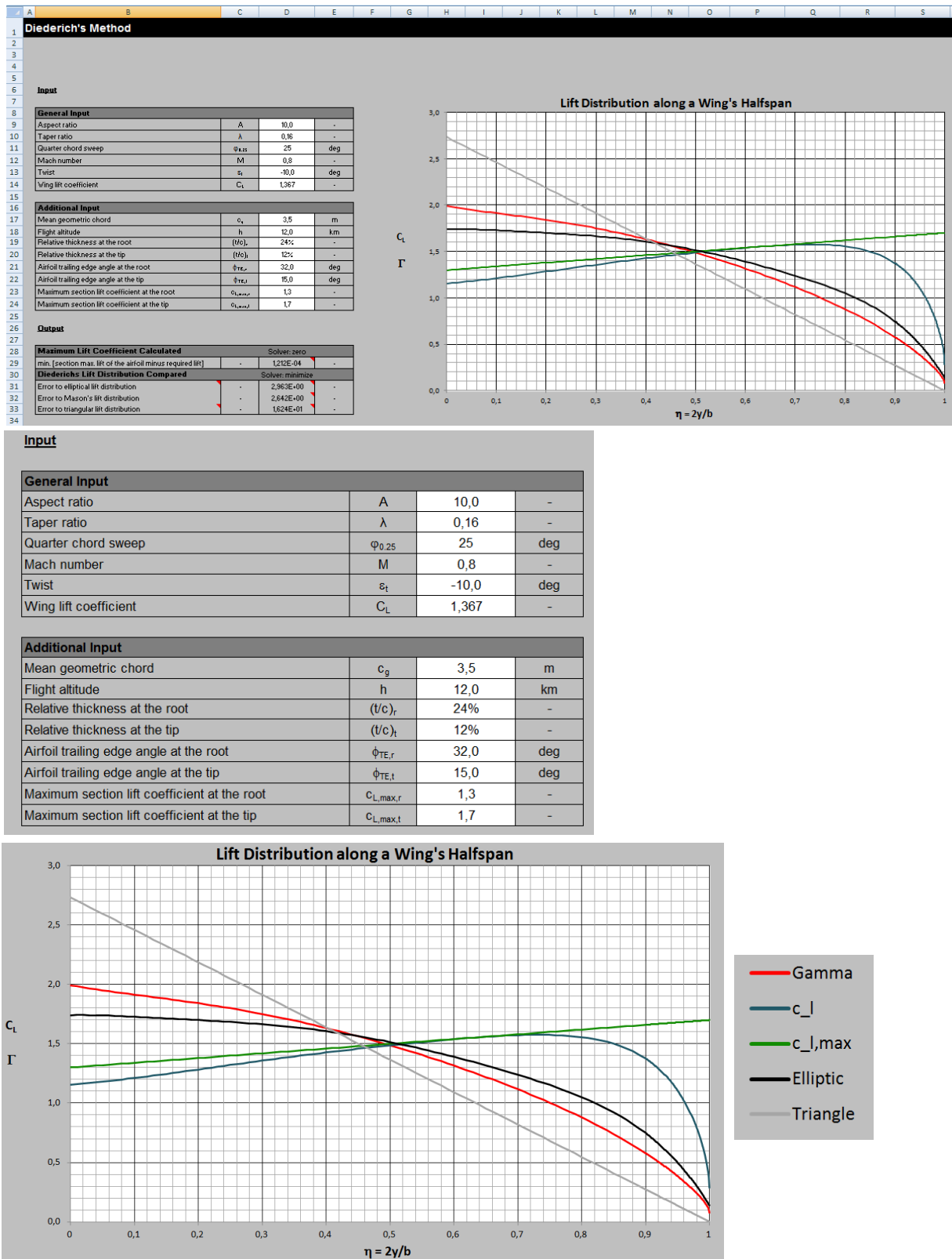



Figure 1: Screen shot of the major view of Diederich's Excel file.

The **graph to the right** shows (depending on the curve) circulation, Γ or local lift coefficient, C_L , plotted against relative half span (from root, 0 to tip, 1). Various curves exist.

Curves can be switched easily on and off in the display by clicking first into the graph and making subsequently a selecting with the "funnel tool": . The "index a" curves come from angle of attack. The "index b" curves enter only once twist has a value different from zero. These special curves may be ignored, if the tool is used as "black box". Important are the results. Standard setting could be as shown in Figure 1. These are the standard and most important curves:

The **lift distribution** or circulation, Γ (Gamma) should be between the elliptic lift distribution (black) and the triangle lift distribution (light gray). Mason's lift distribution (gray) is a lift distribution proposed to aim for. Please find details about the lift distributions in the report (Arslan 2013).

The **distribution of the lift coefficient** (blue) may at no point along the span be above the maximum lift coefficient of the airfoil (green). If this condition is violated, a smaller wing lift coefficient, C_L may be selected under "General Input". A lift coefficient (blue) above the lift coefficient of the airfoil (green) indicates that the wing will be stalled (flow is separated from the surface) in this region of the wing. Details are given in Section 5.

Input cells have a white background. Input is divided into "General Input" and "Additional Input".

"**General Input**" consists of

- Aspect ratio, A (around 10),
- Taper ratio, λ (0.45 to 1.0 for an unswept wing; smaller λ for larger sweep; λ should not be selected much smaller than 0.2),
- Quarter chord sweep (larger for higher Mach number)
- Mach number,
- Twist (0° if wing design also works fine without twist; otherwise -3° to -10° ; this is "wash out", at the wing tip more negative than at the root),
- Wing lift coefficient (select what you desire and see, if it works).

To get started, you may want to leave parameters under "**Additional Input**" initially the way they are set in the table fresh from the download. To get accurate results, however, please make sure you also set the most appropriate values under "Additional Input". The parameters are

- Mean geometric chord (set this value to get roughly the wing area you want or expect),
- Flight altitude (12 km is a typical value for passenger aircraft),
- Relative thickness at the root (the value depends on the chosen airfoil, 15% gets you started for passenger aircraft),
- Relative thickness at the tip (the value depends on the chosen airfoil, 10% gets you started for passenger aircraft),
- Airfoil trailing edge angle at the root and at the tip (also this depends on the chosen airfoil. Generally, the larger t/c , the larger also the trailing edge angle; 15° is a good value to start with; further down, the table offers values for the trailing edge angle as a function of t/c),
- Maximum section lift coefficient at the root and at the tip (also this depends on the chosen airfoil; you find these values in an airfoil catalog like Abbot 1959; links are provided in the References below).

The method is only valid, if four parameters stay in a given range. These are the parameters aspect ratio, A , quarter chord sweep, φ_{25} , altitude, h , and Mach number, M . The **valid parameter range** for the Diederich Method is given in Figure 2. Below the parameter check, information to the users is given as shown in Figure 3.

Beside choosing rational values for the input, the following guidelines should be taken into account

- Only change values in cells with white background
- | | | |
|-----|-----|-------|
| A | $>$ | 4,414 |
|-----|-----|-------|

correct data
- | | | | | |
|-------------|-----|------------------|-----|------------|
| -35° | $<$ | $\varphi_{0.25}$ | $<$ | 35° |
|-------------|-----|------------------|-----|------------|

correct data
- | | | |
|-----|-----|-------|
| h | $<$ | 20 km |
|-----|-----|-------|

correct data
- | | | |
|-----|-----|-----|
| M | $<$ | 0,8 |
|-----|-----|-----|

correct data

Figure 2: The tab "Diederich's Method" checks for the correct range of the input parameters. This panel is located on top of the tab to the right of graph with the lift distribution.

For further use the following statements might be helpful


- Single curves can be hidden by **first clicking the graph**, then the following button: 
- The general input is used for major changes in lift distributions while the additional input refines these results
- The additional output values are only representative for a correct additional input
- For detailed values scroll down to the plot tables after finishing the input data or use your cursor and point on a selected curve section

Figure 3: Hints for user on how to operate the Excel table.

The **output** is presented on gray background. Little red triangles make known that further information is available to this cell. The note appears, if the mouse hovers over the cell. The most important output parameters are shown in Figure 4.

Output

Maximum Lift Coefficient Calculated		Solver: zero	
min. [section max. lift of the airfoil minus required lift]	-	1,212E-04	-
Diederichs Lift Distribution Compared		Solver: minimize	
Error to elliptical lift distribution	-	2,963E+00	-
Error to Mason's lift distribution	-	2,642E+00	-
Error to triangular lift distribution	-	1,624E+01	-
Additional Output			
Root chord length	c_r	6,03	m
Tip chord length	c_t	0,97	m
Maximum lift coefficient of the wing	$C_{L,max}$	1,367	-
Lift curve slope of the wing (DATCOM)	$C_{L,\alpha}$	6,62	1/rad
Lift curve slope of the airfoils (mean value)	$c_{L,\alpha}$	9,49	1/rad
Reynolds number	Re	1,81E+07	-
Leading edge sweep (0%)	$\varphi_{0.0}$	28,31	deg
Trailing edge sweep (100%)	$\varphi_{1.0}$	13,99	deg
Mid chort sweep (50%)	$\varphi_{0.5}$	21,50	deg
Wing span	b	35,00	m
Wing surface area	S	122,50	m ²
True airspeed	V_{TAS}	236,06	m/s
True airspeed	V_{TAS}	849,80	km/h

Figure 4: Next to the graph shown in Figure 1, the most important output from the tab "Diederich's Method" is shown here.

Again, **the Excel table is mainly used ...**

1. ... to calculate lift distribution,
2. ... to calculate spanwise distribution of the lift coefficient,
3. ... to calculate the maximum lift coefficient of the wing.

The distributions 1. and 2. are given in the graph. The maximum lift coefficient of the wing (3.) is given in Figure 4 (third line under "Additional Output").

The **maximum lift coefficient of the wing** is calculated from an evaluation as briefly explained in Arslan 2013 (Section 3.4, Equation (3.35)). This is done with data collected along the span further down in the tab "Diederich's Method" where most of the calculations are done. The maximum lift coefficient of the wing is generated without any action by the user. To see the graph based on $C_{L,max}$, the user needs to read the value from "Additional Output" and must type exactly this value under "General Input" for "Wing lift coefficient", C_L .

The first four output lines are used to **optimize the graphs with the Excel Solver**. The Solver is Excel's built-in optimizer.

The first line "min. [section max. lift of the airfoil minus required lift]" is an alternative way to calculate the maximum lift coefficient of the wing. How this works is explained in Section 4.

The next three lines under "Diederich's Lift Distribution Compared" are used to optimize the lift distribution. It is possible to ask Excel to match the lift distribution as closely as possible either

1. ... to the elliptical lift distribution,
2. ... to the triangular lift distribution,
3. ... to Mason's lift distribution.

1. leads to minimum induced drag,
2. leads to reduced wing root bending moment and hence to a lighter wing, which in turn also leads to reduced aircraft mass and induced drag,
3. is a compromise between 1. and 2.

How to work with the Solver to optimize the lift distribution is explained in Section 3 and 5. See Arslan 2013 for details on the lift distributions.

The output parameters under "**Additional Output**" (Figure 5) show additional wing characteristics and geometry parameters:

- root and tip chord length,
- **maximum lift coefficient of the wing**,
- **lift curve slope of the wing** (calculated according to DATCOM)
- lift curve slope of the airfoils (mean value; as such the lift curve slope of the wing),
- Reynolds number,
- the sweep of the wing, measured at the leading edge (0% chord), mid chord (50% chord), and at the trailing edge (100% chord),
- the wing span,
- the wing surface area,
- the true airspeed (calculated from Mach number and altitude).

Other values are calculated and displayed under "**Calculated Values**". Of interest is maybe the speed of sound in flight altitude. The other parameters are used only internally in the method. As long as the Diederich Method is seen as a "black box" method, these values are not of importance.

Calculated Values			
$2\pi A / (c_{L,\alpha} \cos \varphi_{0,25})$	F	7,009	-
Diederich's Factor	C_1	0,3476	-
Diederich's Factor	C_2	0,3666	-
Diederich's Factor	C_3	0,2861	-
Diederich's Factor	C_4	0,4642	-
Jone's edge velocity factor	E	1,028	-
Factor	$\alpha_{0,1}$	-0,409	-
Prandtl's compressibility correction	β	0,600	-
Effective sweep angle	φ_β	33,28	deg
Theoretical lift curve slope of the airfoil at the root	$c_{L,\alpha,r,theory}$	7,547	1/rad
Theoretical lift curve slope of the airfoil at the tip	$c_{L,\alpha,t,theory}$	6,879	1/rad
Lift curve slope of the airfoil at the root	$c_{L,\alpha,r}$	9,033	1/rad
Lift curve slope of the airfoil at the tip	$c_{L,\alpha,t}$	10,237	1/rad
Kinematic viscosity in flight altitude	ν	4,574	E-5*m ² /s
Speed of sound in flight altitude	a	295,1	m/s

Figure 5: Further output parameters used internally as part of the Diederich Method.

A nice visualization is **the plan view** of the wing. It can be found a little further down and to the right on the tab "Diederich's Method" (Figure 6).

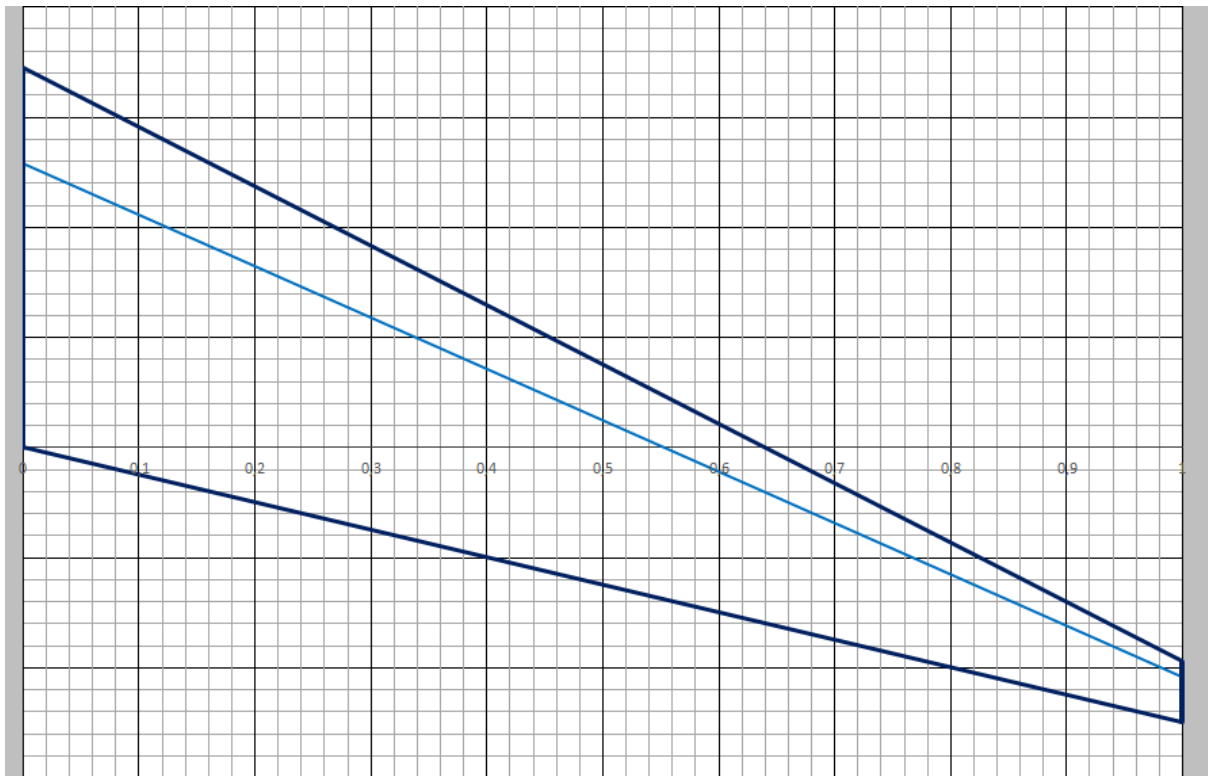


Figure 6: Plan view of the wing according to the input parameters chosen by the user.

Figure 7 shows (a little further down on the tab) **example input parameters** the user may select to get started with the Diederich Method.

Only under "Example" (Figure 7), the **name of the airfoil** at the root and the tip of the wing are given. The Diederich Method does not care what airfoil it is. Of interest to the method are only some characterizing values. The airfoil names are not an input.

It is quite easy here to work with NACA airfoils. All 4-digit, 5-digit and 6-digit NACA airfoils show the **relative thickness** with their last two digits (in percent). Accordingly, $(t/c)_r$ and $(t/c)_t$ can be selected under "Additional Input".

The **trailing edge angle** is defined in DATCOM 1978 as shown in Figure 8.

Example (no input, do not edit)

General Input

Aspect ratio	A	10,0	-
Taper ratio	λ	0,16	-
Quarter chord sweep	$\varphi_{0.25}$	25	deg
Mach number	M	0,8	-
Twist	ε_t	-10,0	deg
Wing lift coefficient	C_L	1,366	-

Additional Input

Mean geometric chord	c_g	3,5	m
Flight altitude	h	12,0	km
Relative thickness at the root	$(t/c)_r$	24%	-
Relative thickness at the tip	$(t/c)_t$	12%	-
Airfoil trailing edge angle at the root	$\phi_{TE,r}$	32,0	deg
Airfoil trailing edge angle at the tip	$\phi_{TE,t}$	15,0	deg
Maximum section lift coefficient at the root	$C_{L,max,r}$	1,3	-
Maximum section lift coefficient at the tip	$C_{L,max,t}$	1,7	-

Airfoil at root		NACA 2424	-
Airfoil at tip		NACA 2412	-

Figure 7: Input parameters the user may select to get started with the Diederich Method.

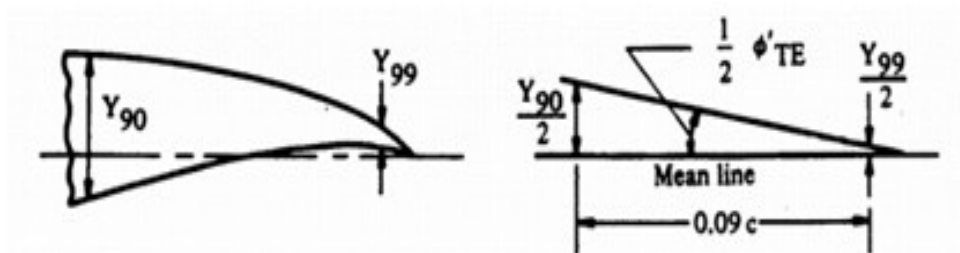


Figure 8: Definition of the trailing edge angle of the airfoil ϕ'_{TE} or Φ_{TE} . (DATCOM 1978)

Mason 2017 investigated the trailing edge angle of airfoils (Figure 9).

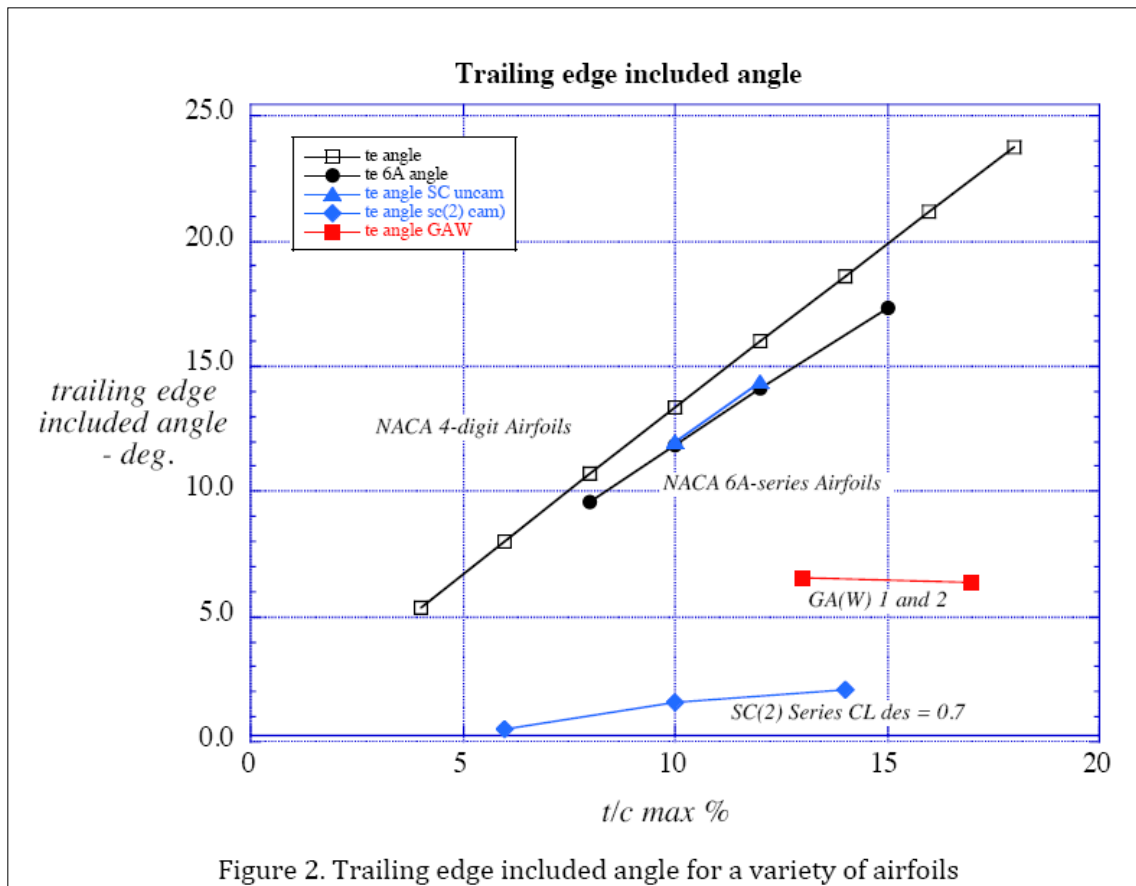


Figure 2. Trailing edge included angle for a variety of airfoils

Figure 9: Trailing edge angle as function of airfoil relative thickness. From Mason 2017.

NACA 4-digit airfoils show a clear relationship between relative thickness and trailing edge angle.

$$\phi_{TE} = 135.71^\circ t/c - 0.4286^\circ$$

with t/c dimensionless and ϕ_{TE} in degree

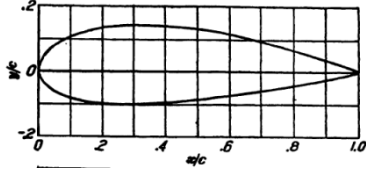
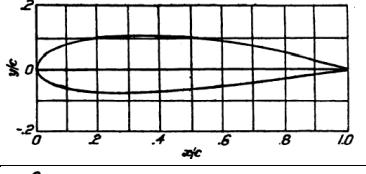
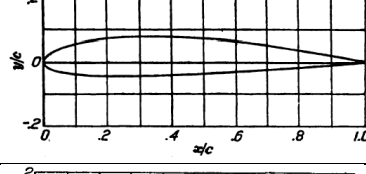
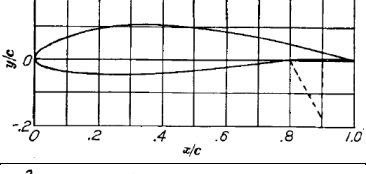
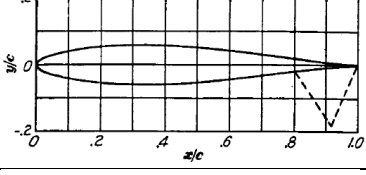
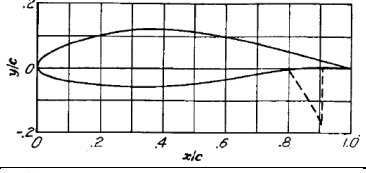
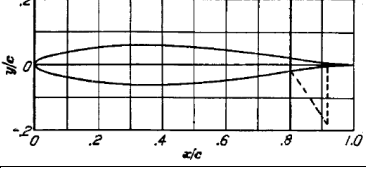
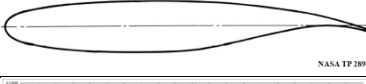
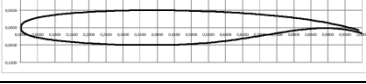
This equation is used in the Excel tab under "Estimating (for NACA 4-digit airfoils, based on Mason 2017) ... the airfoil trailing edge angle at the root ... and the tip" (Figure 7). NACA 4-digit airfoils are used for low speed aircraft (single engine piston props, trainers) and for tail surfaces. Airfoil data is in Abbot 1959.

NACA 6-digit airfoils seem to have by 1° to 2° smaller trailing edge angles compared to NACA 4-digit airfoils. NACA airfoils are documented well in Abbot 1959 and a summary of NACA 6-digit airfoils is given also in Scholz 2002. For early passenger jets the 63- and 64- sections were popular.

Transonic airfoils – like SC(2) – can have very small trailing edge angles of only a few degrees. Transonic airfoils were not systematically analyzed in public. A selection of public transonic airfoils can be found in Schiktanz 2011.

Table 1 summarizes three proposals for a root/tip airfoil selection.

Table 1: Proposed airfoil selection: NACA 4-digit, 6-digit, and transonic airfoils

airfoil name	position	t/c	ϕ_{TE}	$c_{L,max}$	α_0	airfoil	source	page
NACA 2424	root	24%	32°	1.3	-2°		Abbot 1959	495ff
NACA 2418	root	18%	24°	1.5	-2°		Abbot 1959	491ff
NACA 2412	tip	12%	15°	1.7	-2°		Abbot 1959	487ff
NACA 63 ₂ -615	root	15%	17°	1.7	-4°		Scholz 2002	PDF 15
NACA 63 ₁ -012	tip	12%	14°	1.4	0°		Scholz 2002	PDF 11
NACA 64 ₃ -618	root	18%	20°	1.6	-4°		Scholz 2002	PDF 26
NACA 64 ₁ -012	tip	12%	14°	1.5	0°		Scholz 2002	PDF 20
NASA SC(2)-0714 ¹	root	14%	0°	2.2	-3°		Schiktanz 2011	PDF 196
NASA SC(2)-0710 ²	tip	10%	0°	2.2(?)	-3°		Schiktanz 2011	PDF 177

¹ Year 1975, $c_{L,design} = 0.7$, $M_{design} = 0.74$

² Year 1975, $c_{L,design} = 0.7$, $M_{design} = 0.78$

3 Excel Solver

The Solver is the optimizer in Excel. Together with the Diederich Method, the Excel Solver is used

- ... to find the maximum lift coefficient of the wing (using the alternative method),
- ... to find the parameters for the desired lift distribution (elliptic, triangle, Mason).

The Solver is an Add-In to Excel by Frontline Systems. Much information is available online from

- [Frontline Systems](#)
- [Load the Solver Add-in in Excel](#)
- [Microsoft: Define and Solve a Problem](#)
- Many other sites

The Solver is part of the Excel installation, but needs to be activated once. This is done with

- Datei – Menue top left,
- Optionen – down-left
- Verwalten – last line
- Excel-Add-Ins – in the selection box
- "Los ..."
- This opens a window. The box left to "Solver" needs to be selected (ticked). See Figure 10.

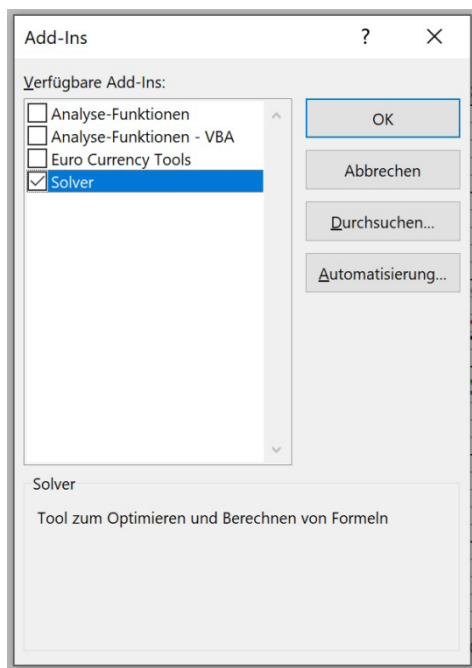


Figure 10:

Activate the Solver in Excel with this dialog box deep in the Options.

Figure 11 shows the dialog box of the Solver. This is where the optimization task is defined:

- Line 1 (Ziel festlegen): Specify the cell in which a parameter is waiting to get optimized (example: \$D\$32).
- Line 2 (Bis): Define how should be optimized: maximized, minimized, or to reach a certain value, which has to be specified at the far right end of Line 2 (example: Min.).
- Line 3 (Durch Ändern ...): Select at least one variable that may be changed to improve the result. Select more variables to increase the chance to reach your goal better (example: Eps_t).
- Line 4 (Unterliegt den Nebenbedingungen): Boundary conditions can be specified.
- Line 5 (Lösungsmethode): GRG-Nichtlinear is the default solver for linear and non-linear problems.
- Last line (Lösen): Select to start the calculation. When done, a dialog opens and asks, if the solution is accepted. In this case data from the optimization is stored in the table and the old values in the table are overwritten.

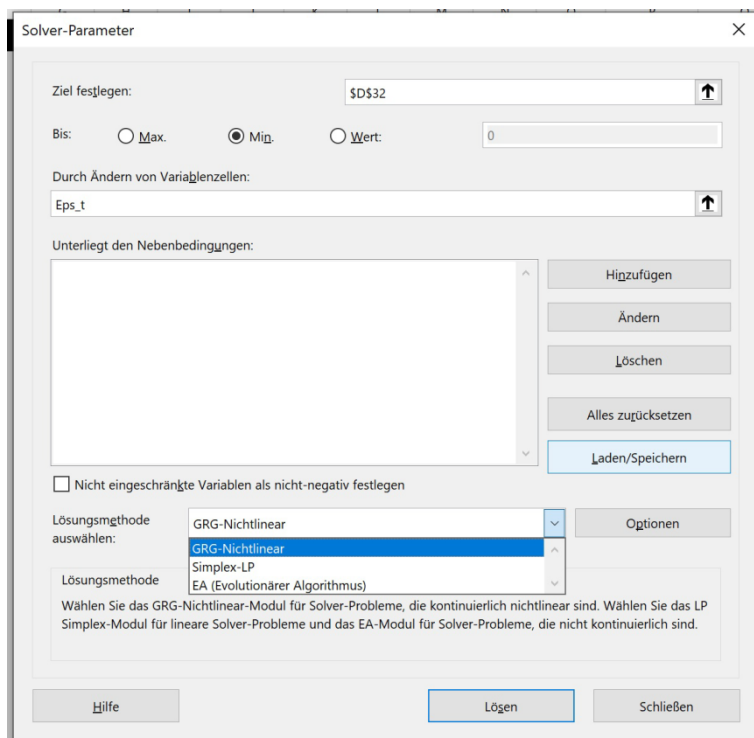


Figure 11: Dialog box of the Solver in Excel.

4 Maximum Lift Coefficient of the Wing Obtained with the Solver

The maximum lift coefficient of the wing is obtained directly and the result is included under "Additional Output". An alternative solution uses the solver. **The solution with the Solver is easier to understand and confirms the other result.**

Background: The Excel table allows the wing to be described with more than one airfoil. To limit the complexity, one airfoil can be selected at the wing root and one at the wing tip. If the wing applies the same airfoil throughout, the same airfoil is specified at the wing root and at the wing tip. The airfoils are not specified with their coordinates. It is sufficient to input the relative thickness, t/c , the airfoil trailing edge angle, Φ_{TE} , and the maximum section lift coefficient, $c_{l,max}$ at the wing tip and the wing root. It is assumed that the airfoil at the wing root blends linearly into the airfoil at the wing tip.

The Excel table works such that the lift coefficient of the wing, C_L is entered and the lift distribution, $\Gamma(\eta)$ is calculated such that its integration over the span yields the lift coefficient of the wing. The distribution of the local lift coefficient, $c_L(\eta)$ is calculated from the lift distribution, $\Gamma(\eta)$.

Figure 3.4 from the report (Arslan 2013) is given here as Figure 12. It shows the situation of the maximum lift coefficient of the wing. The curve $\Gamma(\eta)$ is shown in blue and $c_L(\eta)$ is shown in green.

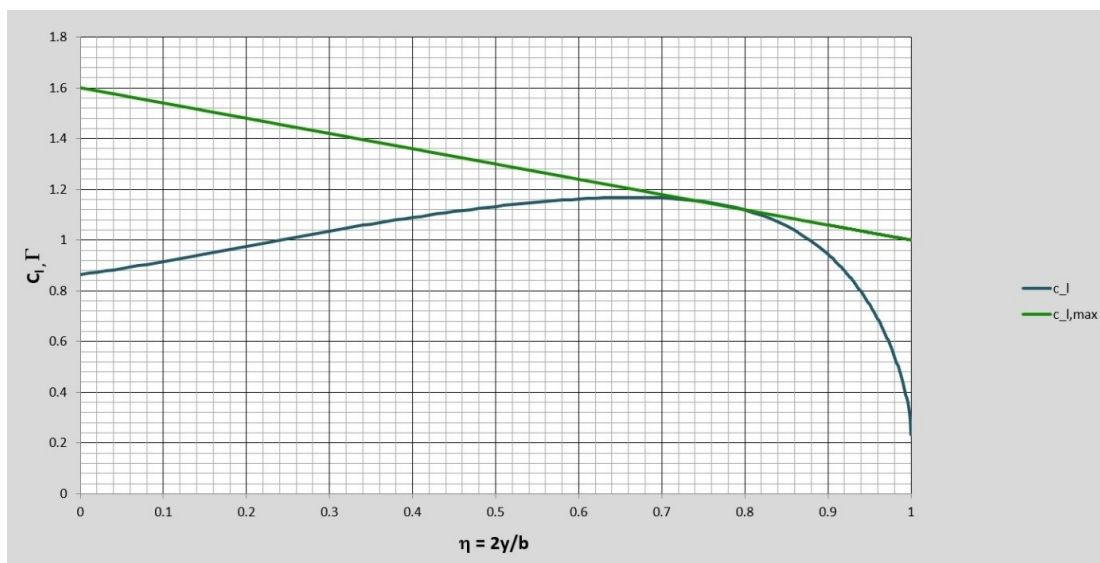


Figure 12: The maximum lift coefficient of the wing is reached when (at least) one spanwise position on the wing reaches the local maximum lift coefficient.

The maximum lift coefficient of the wing is reached when (at least) one spanwise position on the wing the local lift coefficient (blue line) reaches the maximum lift coefficient of the airfoil (green line) at this position.

The situation of the **maximum lift coefficient of the wing** can be translated **with the Excel Solver** such that

- the difference between the maximum lift coefficient of the airfoil (green line) and the local lift coefficient (blue line) is calculated for all span positions and the smallest difference is selected. This is the minimum of all these values. It is given in D29 (under output),
- first line: D29 is selected,
- second line: "Wert" / "Value" is selected and 0.0 (zero) is entered,
- third line: C_L (wing lift coefficient, D14) is selected,
- last line: the Solver is started,
- the "maximum lift coefficient of the wing" can be read in the cell for C_L (wing lift coefficient, D14).

5 Lift Distribution Optimized with the Solver

Three lift distributions are calculated and can be made visible in the graph

1. the elliptical lift distribution,
2. the triangular lift distribution,
3. Mason's lift distribution.

1. leads to minimum induced drag,
2. leads to reduced wing root bending moment and hence to a very light wing, which in turn also leads to reduced aircraft mass and induced drag,
3. is a compromise between 1. and 2.

1. and 2. are both extreme in their own right and will seldom be taken. A compromise lift distribution is thought in the middle between them. In the Excel table **basically four parameters can be modified** with a strong influence on the lift distribution. In the order of plausibility these parameters are

1. taper ratio, λ ,
2. twist, ϵ_t ,
3. quarter chord sweep, φ_{25} ,
4. aspect ratio.

Wing twist can come from

- geometric twist, $\epsilon_{t,geo}$: a typical negative geometric twist is obtained, if the tip chord line is not parallel to the root chord line, but nose down,
- aerodynamic twist, $\epsilon_{t,aero}$ (different airfoils chosen): a typical negative aerodynamic twist is obtained, if the root airfoil has more camber and hence a more negative zero lift angle of attack, α_0 , then the tip airfoil; see Table 1 for values of aerodynamic twist of the airfoils and find the difference between tip and root,
- total twist, ϵ_t : the sum of geometric twist and aerodynamic twist.

$$\epsilon_{t,geo} = \epsilon_{t,geo,t} - \epsilon_{t,geo,r}$$

$$\epsilon_{t,aero} = -(\alpha_{0,t} - \alpha_{0,r})$$

$$\epsilon_t = \epsilon_{t,geo} + \epsilon_{t,aero}$$

Example:

$$\varepsilon_{t,geo} = -6^\circ - (+2^\circ) = -8^\circ$$

$$\varepsilon_{t,aero} = -(0^\circ - (-5^\circ)) = -5^\circ$$

$$\varepsilon_t = -8^\circ + (-5^\circ) = -13^\circ$$

Critical acclaim:

4. aspect ratio, A , follows from other design considerations and has been selected already, when the lift distribution is optimized,
3. at a high cruise Mach number, M , quarter chord sweep follows from Mach number; at low cruise Mach number, a sweep larger than necessary is usually not a good compromise (e.g. it leads to a reduction of maximum lift coefficient),
2. twist, ε_t , can be used, however, it optimizes the lift distribution only at the design lift coefficient and may have negative consequences e.g. at lift coefficients below the design lift coefficient; if you have decided on a value for twist, check with various wing lift coefficients,
1. it is best to work with taper ratio, λ , however, large sweep angles require low λ , but λ may not be much smaller than 0.2.

Finding the best compromise can be seen more as an expert judgment than a pure mathematical operation. Nevertheless, some numerical help is certainly appreciated. The Solver offers this help.

To optimize the lift distribution with the Solver, its dialog box (Figure 11) has to be filled out:

- first line: it needs to be selected, close to which of the three given lift distributions the present lift distribution should be modeled:
 - select D31 for the elliptical lift distribution,
 - select D32 for Mason's lift distribution,
 - select D33 for the triangular lift distribution,
- second line: select "Min."
- third line: select the cell (or variable name) or several cells (or variable names; semicolon-separated) of the parameters that should be considered in the optimization, variable names are:
 - aspect ratio: A_w ,
 - taper ratio: λ ,
 - quarter chord sweep: ϕ_{25} ,
 - twist: ε_t

- fourth line: boundary conditions can be specified as necessary, if the solution shows unrealistic results (negative taper ratio, ...),
- last line: the Solver is started,
- the new lift distribution (red line) can now be seen and evaluated on the graph, this will lead to new ideas of other input to the Solver.

Advise:

- Since the parameters also depend on the selected wing lift coefficient, select a usual one: 0.7.
- It is easier to come close to the elliptical lift distribution than to the triangular lift distribution. For this reason choose the triangular lift distribution (D33) in line one and see how close you get with realistic parameters.
- Optimize initially with one parameter at a time. Follow the sequence advised above. Take the parameters in this sequence: taper ratio, $\lambda \Rightarrow$ twist, $\epsilon_t \Rightarrow$ quarter chord sweep, ϕ_{25} .

6 Summary and Recommendations

The Diederich Method Excel-File helps to answer the **classical questions from aerodynamics**:

- What is the lift curve slope?
- What is the lift distribution? How can I optimize the lift distribution for minimum induced drag?
- What is the maximum lift coefficient of the wing?

Advantages: Since the method is based on analytical equations, the solution is instant. There is no need to set a new geometric model followed by meshing and tuning of the numerical results to experimental results.

The method is around since at least 1952. Enhancements in 1988 improved easy handling of the method. Nevertheless, the method is nothing, if it is available only in a report or book. It only come to life once set up e.g. in a spreadsheet. This has been done.

Long established rules can undergo a check:

- How does the maximum lift coefficient change with sweep? Is the proportionality with $\cos(\varphi_{25})$ justified? See Scholz 2015 (7.37).
- At what relative span, η , do we find the maximum local lift coefficient? See Scholz 2015 (7.38).
- How does optimum taper ratio changes with sweep? See Scholz 2015 (7.39).
- What is optimum twist? See Scholz 2015, Section "Wing Design", Paragraph "Twist".

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