AERODAS.xls

This document explains how to use AERODAS.xls to obtain the lift and drag (C_L and C_D) for a wide variety of angle of attacks. The AERODAS.xls is developed by Prof. Ercan ERTURK from BRZ Wind Energy company, TURKEY. Prof. ERTURK is also a faculty member of Bahcesehir University. You can freely distribute AERODAS.xls.

MOTIVATION

In wind turbine blade analysis, the Blade Element Momentum (BEM) calculations require lift and drag coefficients (C_L and C_D) for a wide variety of angles of attack. For very few airfoil profiles, in the literature it is possible to find limited experimental datas for the lift and drag coefficients at high angles of attack.

Many researchers use the XFOIL simulation code (developed by Prof. Mark Drela, MIT) to obtain polar coefficients. XFOIL is a very useful simulation tool and it can provide lift and drag coefficients (C_L and C_D) for a limited range of angle of attack (until stall).

In the literature there are some extrapolation techniques (Viterna and Janetzke [1], Montgomerie [2], Lindenburg [3] and Spera [4]) to estimate the lift and drag coefficients (C_L and C_D) for a wide variety of angles of attack using the polar data of limited angle range.

The AERODAS algorithm developed by Spera [4] uses geometrical tools . The AERODAS algorithm is straight forward and easy to follow. In order to implement this algorithm, some parameters have to be calculated. If this model is used extensively, a visual tool is necessary and this motivated the development of this excel document.

MODIFICATIONS ON AERODAS EQUATIONS

In AERODAS.xls document two modifications are done on the original AERODAS equations. They are as follows:

1) In Spera [4] the equation (7a) is given as the following

$$CD1 = CD0 + \left(CD1\max - CD0\right)\left(\frac{\alpha - A0}{ACD1 - A0}\right)^{M}$$

When α is negative, this equation produces negative results for odd powers (for example M=3). In order to have positive values for all α values and for all powers (M), we use absolute value inside the parenthesis such that we modify the equation as the following

$$CD1 = CD0 + \left(CD1\max - CD0\right)\left(\left|\frac{\alpha - A0}{ACD1 - A0}\right|\right)^{M}$$

2) In Spera [4] the equation (10b) is given as the following

$$G1 = 2.300\ \exp\left\{-\left[0.65\left(t/c\right)\right]^{0.90}\right\}$$

In the manuscript Spera [4] gives two numerical examples; one for S809 airfoil and one for CLARK-Y airfoil. The “thickness to chord ratio” of S809 airfoil is 0.21. Using equation (10b) in order to calculate G1, we obtain

$$G1 = 2.300\ \exp\left\{-\left[0.65\left(0.21\right)\right]^{0.90}\right\} = 1.947$$

However in Table [5] in Spera [4], G1 is given as 1.922 for this particular airfoil. Similarly the “thickness to chord ratio” of CLARK-Y airfoil is 0.184. Again using equation (10b) in order to calculate G1, we obtain

$$G1 = 2.300\ \exp\left\{-\left[0.65\left(0.184\right)\right]^{0.90}\right\} = 1.984$$

However in the Table [6] in Spera [4], G1 is given as 1.958 for this particular airfoil.

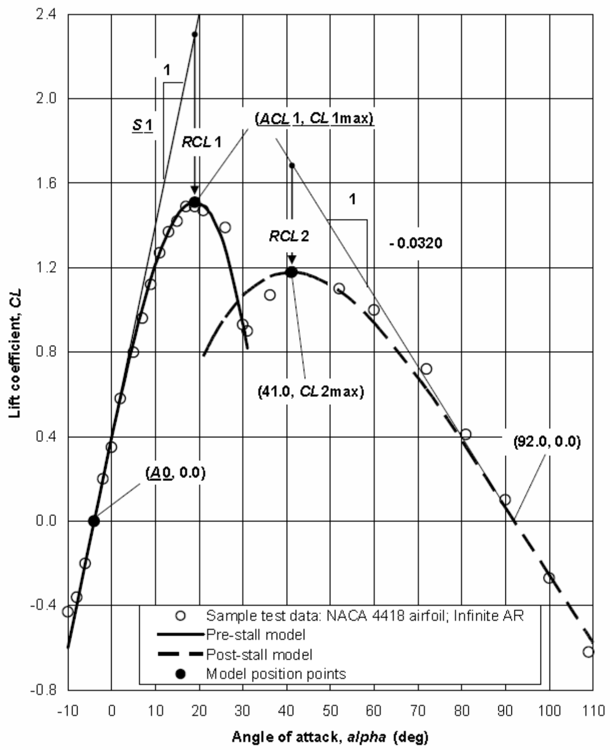
Equation (10b) is a very simple equation with only one variable (t/c). The difference between the calculated G1 values and the G1 values listed in Tables [5] and [6] are big and affects the CD2max value. There must be a mistake either in the equation (10b) or in the listed G1 values in Tables [5] and [6]. We assume that the G1 values listed in Tables [5] and [6] are correct and there is an error in equation (10b). When we modify the equation (10b) as follows

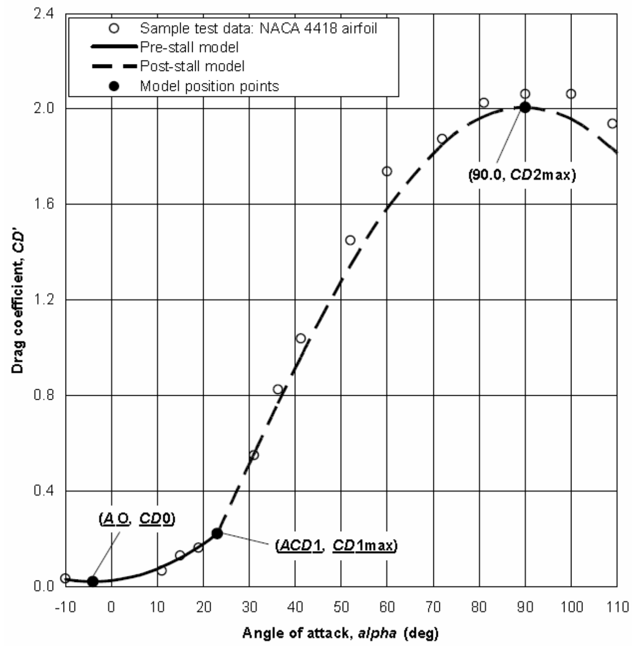
$$G1 = 2.270\ \exp\left\{-\left[0.65\left(t/c\right)\right]^{0.90}\right\}$$

for S809 airfoil the calculated G1 becomes 1.922 and for CLARK-Y airfoil the calculated G1 becomes 1.958 which agree with the listed G1 values in Tables [5] and [6]. We believe that in equation (10b) the coefficient 2.27 is rounded as 2.3 by a typing mistake. In AERODAS.xls 2.27 is used as coefficient in equation (10b)

HOW TO USE AERODAS.xls

In order to use AERODAS.xls you should first read Spera [4] and have basic knowledge on the AERODAS model. The model basically consists of two curve; one for pre-stall and one for post-stall for both lift and drag coefficients. The following two figures are from Spera [4] and shows the important parameters used in the model.





The following steps from 1 to 3 describes how to obtain the lift coefficient (C_L) and the steps from 4 to 5 describes how to obtain the drag coefficient (C_D).

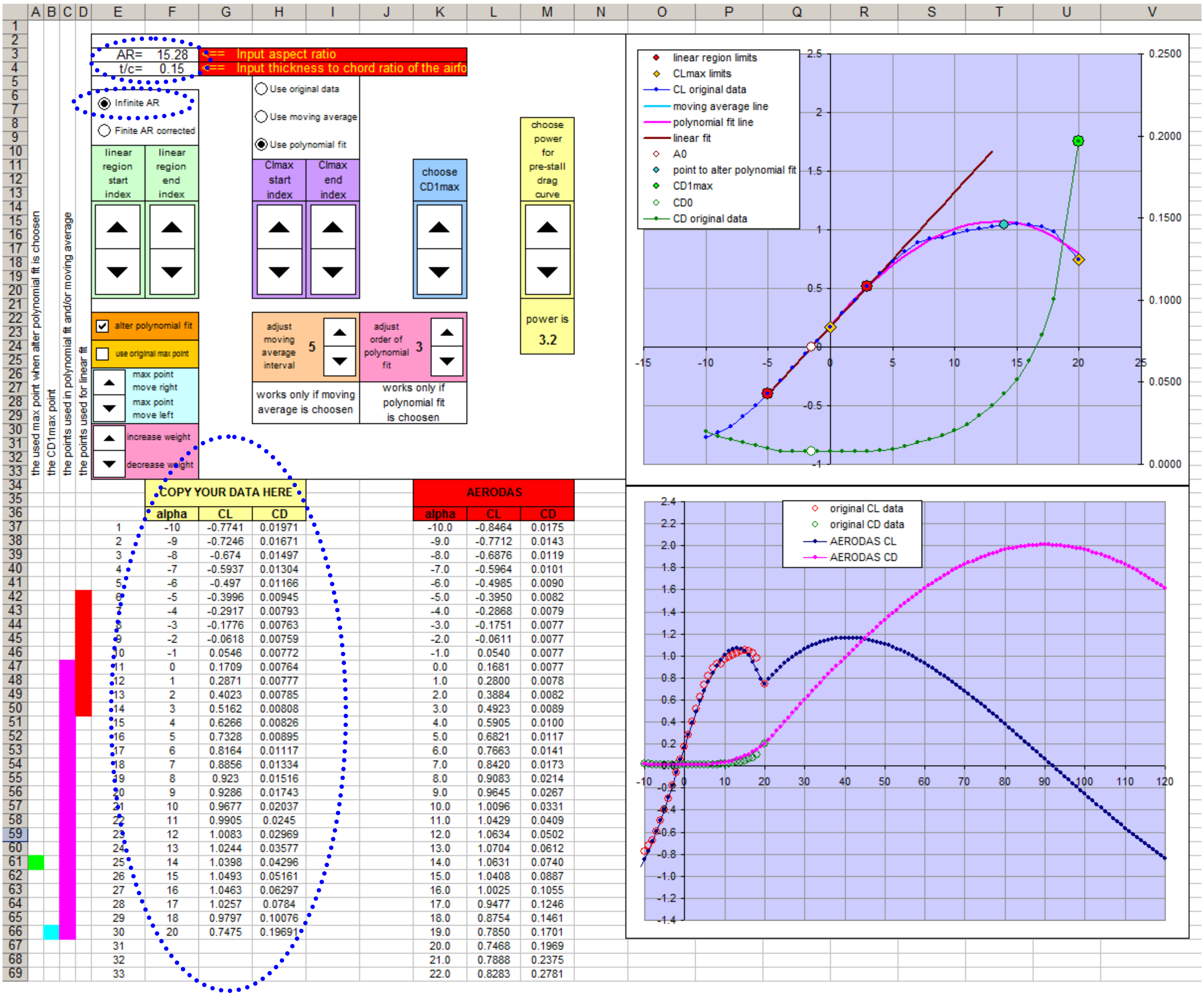
STEP 1:

(Look at Figure 1)

Open AERODAS.xls document. Note that if macros are disabled you have to allow macros. Copy and paste your data in the area shown. The first column is the angle of attack, the second column is the lift coefficient and the third column is the drag coefficient. You can enter maximum of 100 lines. Open your XFOIL polar output data file with excel, use text to column feature and have the XFOIL polar data in columns and rows. Then select only the angle, C_L and C_D values and paste it to the place shown in AERODAS.xls document.

Enter your airfoil's thickness to chord ratio, also enter the aspect ratio of your wind turbine blade.

Note that XFOIL provides polar coefficients for infinite aspect ratio. You should **always** start your analysis by selecting "Infinite AR". The calculations are done on the infinite AR polar coefficients. After you follow all the steps and finish your analysis, you can click on "Finite AR corrected" and obtain results based on the aspect ratio of your wind turbine blade you entered.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.7741	0.01971	-10.0	-0.8464	0.0175
2	-9	-0.7246	0.01671	-9.0	-0.7712	0.0143
3	-8	-0.674	0.01497	-8.0	-0.6876	0.0119
4	-7	-0.5937	0.01304	-7.0	-0.5964	0.0101
5	-6	-0.497	0.01166	-6.0	-0.4985	0.0090
6	-5	-0.3996	0.00945	-5.0	-0.3950	0.0082
7	-4	-0.2917	0.00793	-4.0	-0.2868	0.0079
8	-3	-0.1776	0.00763	-3.0	-0.1751	0.0077
9	-2	-0.0618	0.00759	-2.0	-0.0611	0.0077
10	-1	0.0546	0.00772	-1.0	0.0540	0.0077
11	0	0.1709	0.00764	0.0	0.1681	0.0077
12	1	0.2871	0.00777	1.0	0.2800	0.0078
13	2	0.4023	0.00785	2.0	0.3884	0.0082
14	3	0.5162	0.00808	3.0	0.4923	0.0089
15	4	0.6266	0.00826	4.0	0.5905	0.0100
16	5	0.7328	0.00895	5.0	0.6821	0.0117
17	6	0.8164	0.01117	6.0	0.7663	0.0141
18	7	0.8856	0.01334	7.0	0.8420	0.0173
19	8	0.923	0.01516	8.0	0.9083	0.0214
20	9	0.9286	0.01743	9.0	0.9645	0.0267
21	10	0.9677	0.02037	10.0	1.0096	0.0331
22	11	0.9905	0.0245	11.0	1.0429	0.0409
23	12	1.0083	0.02969	12.0	1.0634	0.0502
24	13	1.0244	0.03577	13.0	1.0704	0.0612
25	14	1.0398	0.04296	14.0	1.0631	0.0740
26	15	1.0493	0.05161	15.0	1.0408	0.0887
27	16	1.0463	0.06297	16.0	1.0025	0.1055
28	17	1.0257	0.0784	17.0	0.9477	0.1246
29	18	0.9797	0.10076	18.0	0.8754	0.1461
30	20	0.7475	0.19691	19.0	0.7850	0.1701
31				20.0	0.7468	0.1969
32				21.0	0.7888	0.2375
33				22.0	0.8283	0.2781

Figure 1

STEP 2:

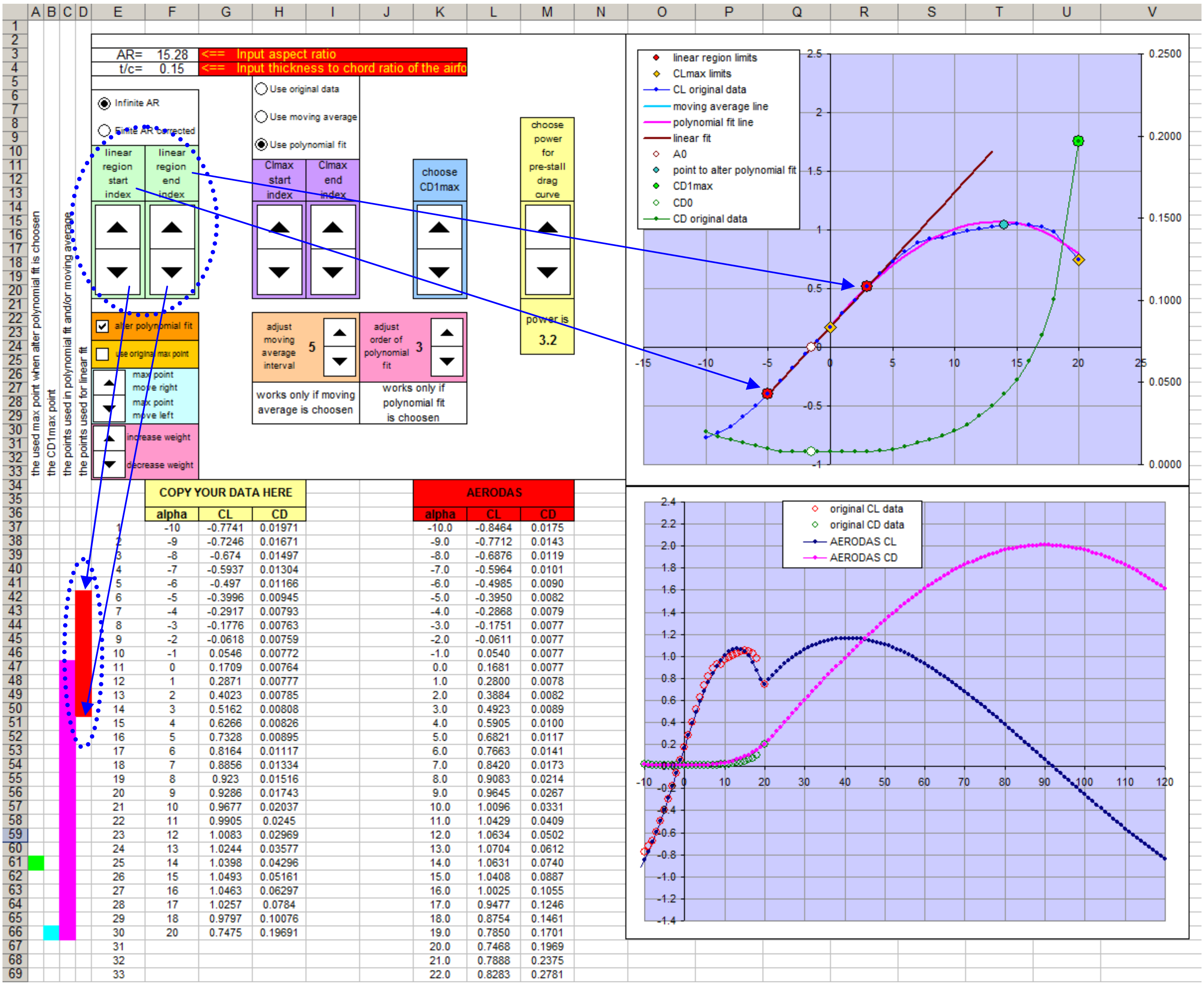
(Look at Figure 2)

In the pre-stall region the AERODAS model needs the parameter S1 which is basically the linear slope of the lift coefficient in the pre-stall ramp. In order to calculate S1, AERODAS.xls uses linear regression. The user is required to choose the region of the data points used in the linear regression. Use the arrows to set the first data point and the last data point. You can see the beginning data point and the last data point of the chosen region in the figure with two circles with red inside. You can also see the linear fitted line to the selected data points in the figure with brown line.

When the user copy and paste new data in another analysis sometimes the selected data points for the linear regression in previous analysis might be out of scope. You can easily see the selected range of data points by looking at the colored cells in column D. As you change your start and end data points by pressing up/down arrows you can see that the coloring in column D changes dynamically. All of the red colored cells in column D has to be in the range of your pasted data.

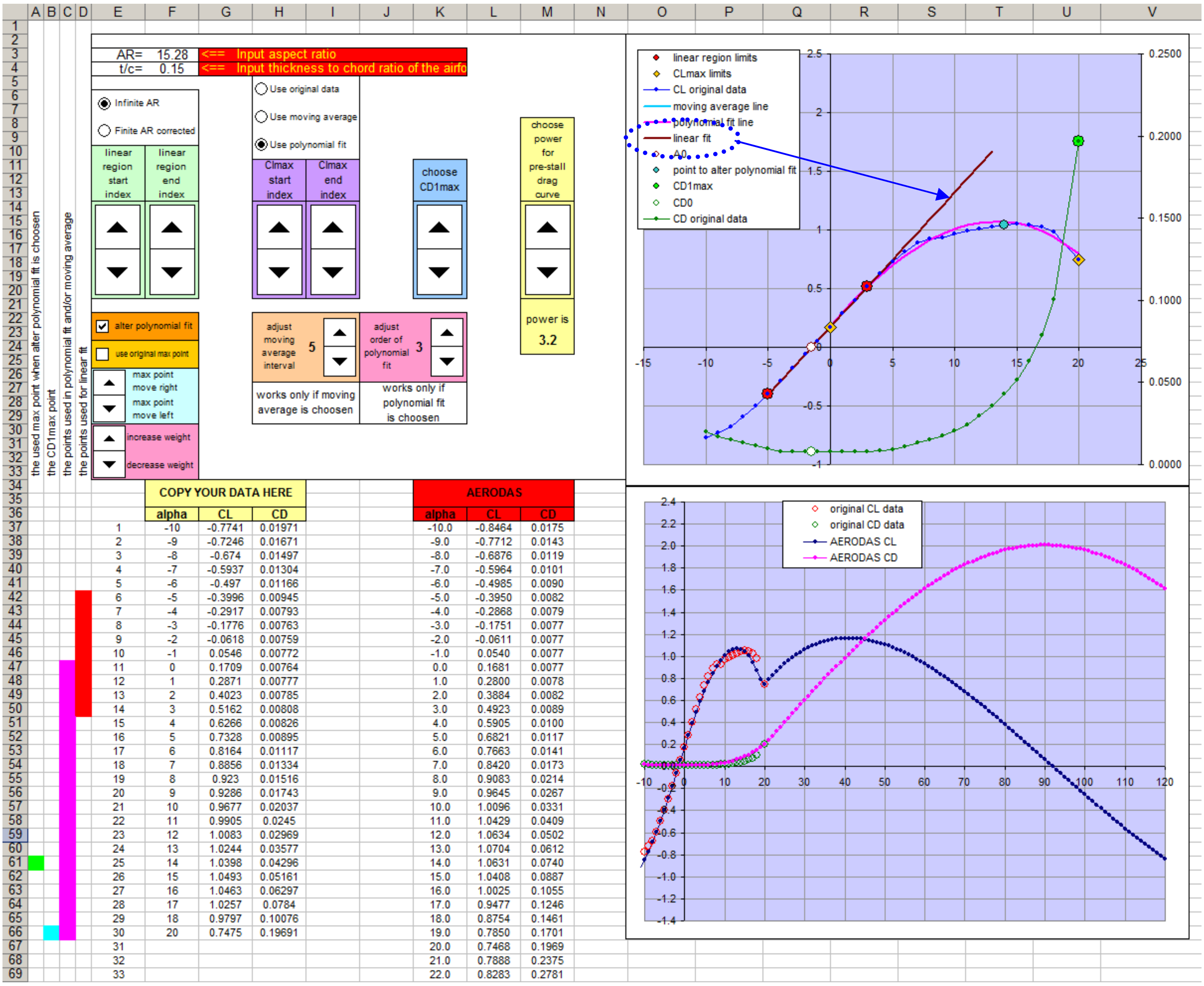
(Look at Figure 3)

Choose a region that best describes the linear fit in the lift ramp. The slope of the brown line in the figure gives S1.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.7741	0.01971	-10.0	-0.8464	0.0175
2	-9	-0.7246	0.01671	-9.0	-0.7712	0.0143
3	-8	-0.674	0.01497	-8.0	-0.6876	0.0119
4	-7	-0.5937	0.01304	-7.0	-0.5964	0.0101
5	-6	-0.497	0.01166	-6.0	-0.4985	0.0090
6	-5	-0.3996	0.00945	-5.0	-0.3950	0.0082
7	-4	-0.2917	0.00793	-4.0	-0.2868	0.0079
8	-3	-0.1776	0.00763	-3.0	-0.1751	0.0077
9	-2	-0.0618	0.00759	-2.0	-0.0611	0.0077
10	-1	0.0546	0.00772	-1.0	0.0540	0.0077
11	0	0.1709	0.00764	0.0	0.1681	0.0077
12	1	0.2871	0.00777	1.0	0.2800	0.0078
13	2	0.4023	0.00785	2.0	0.3884	0.0082
14	3	0.5162	0.00808	3.0	0.4923	0.0089
15	4	0.6266	0.00826	4.0	0.5905	0.0100
16	5	0.7328	0.00895	5.0	0.6821	0.0117
17	6	0.8164	0.01117	6.0	0.7663	0.0141
18	7	0.8856	0.01334	7.0	0.8420	0.0173
19	8	0.923	0.01516	8.0	0.9083	0.0214
20	9	0.9286	0.01743	9.0	0.9645	0.0267
21	10	0.9677	0.02037	10.0	1.0096	0.0331
22	11	0.9905	0.0245	11.0	1.0429	0.0409
23	12	1.0083	0.02969	12.0	1.0634	0.0502
24	13	1.0244	0.03577	13.0	1.0704	0.0612
25	14	1.0398	0.04296	14.0	1.0631	0.0740
26	15	1.0493	0.05161	15.0	1.0408	0.0887
27	16	1.0463	0.06297	16.0	1.0025	0.1055
28	17	1.0257	0.0784	17.0	0.9477	0.1246
29	18	0.9797	0.10076	18.0	0.8754	0.1461
30	20	0.7475	0.19691	19.0	0.7850	0.1701
31				20.0	0.7468	0.1969
32				21.0	0.7888	0.2375
33				22.0	0.8283	0.2781

Figure 2



	alpha	CL	CD	alpha	CL	CD
1	-10	-0.7741	0.01971	-10.0	-0.8464	0.0175
2	-9	-0.7246	0.01671	-9.0	-0.7712	0.0143
3	-8	-0.674	0.01497	-8.0	-0.6876	0.0119
4	-7	-0.5937	0.01304	-7.0	-0.5964	0.0101
5	-6	-0.497	0.01166	-6.0	-0.4985	0.0090
6	-5	-0.3996	0.00945	-5.0	-0.3950	0.0082
7	-4	-0.2917	0.00793	-4.0	-0.2868	0.0079
8	-3	-0.1776	0.00763	-3.0	-0.1751	0.0077
9	-2	-0.0618	0.00759	-2.0	-0.0611	0.0077
10	-1	0.0546	0.00772	-1.0	0.0540	0.0077
11	0	0.1709	0.00764	0.0	0.1681	0.0077
12	1	0.2871	0.00777	1.0	0.2800	0.0078
13	2	0.4023	0.00785	2.0	0.3884	0.0082
14	3	0.5162	0.00808	3.0	0.4923	0.0089
15	4	0.6266	0.00826	4.0	0.5905	0.0100
16	5	0.7328	0.00895	5.0	0.6821	0.0117
17	6	0.8164	0.01117	6.0	0.7663	0.0141
18	7	0.8856	0.01334	7.0	0.8420	0.0173
19	8	0.923	0.01516	8.0	0.9083	0.0214
20	9	0.9286	0.01743	9.0	0.9645	0.0267
21	10	0.9677	0.02037	10.0	1.0096	0.0331
22	11	0.9905	0.0245	11.0	1.0429	0.0409
23	12	1.0083	0.02969	12.0	1.0634	0.0502
24	13	1.0244	0.03577	13.0	1.0704	0.0612
25	14	1.0398	0.04296	14.0	1.0631	0.0740
26	15	1.0493	0.05161	15.0	1.0408	0.0887
27	16	1.0463	0.06297	16.0	1.0025	0.1055
28	17	1.0257	0.0784	17.0	0.9477	0.1246
29	18	0.9797	0.10076	18.0	0.8754	0.1461
30	20	0.7475	0.19691	19.0	0.7850	0.1701
31				20.0	0.7468	0.1969
32				21.0	0.7888	0.2375
33				22.0	0.8283	0.2781

Figure 3

STEP 3:

(Look at Figure 4)

In the pre-stall region the AERODAS model needs the parameter RCL1 which is basically the vertical distance between the peak C_L point and the slope line.

We need to find the vertical distance between the linear fit line and the peak lift value. In order to calculate RCL1 we have couple of options. If you choose “Use original data” then the document will find the maximum C_L value in the pasted data, then it will calculate the vertical distance between the brown line and this maximum C_L value. Note that the tip of the brown line will always be over this maximum C_L value in vertical direction.

According to our experiences with XFOIL, depending on the airfoil used and the Reynolds number sometimes we obtain very nice peak values but sometimes we obtain several humps or zigzag type behavior near the peak. In your calculations if the lift data has a nice and definite peak then check “Use original data”. Do not forget, in the pre-stall region the lift coefficient is defined by a curve in AERODAS model, therefore at the end of your analysis the output lift coefficient will have only one peak without any humps or zigzags. In your lift data if there is not a clear peak point, if there are humps and zigzags near the peak you may want to smooth the C_L curve in order to have a definite peak point. The AERODAS.xls allows you to smooth the C_L curve using moving averages or using higher order polynomial regression. If you want to use moving averages in order to smooth your C_L data check “Use moving average”. If you want to use higher order polynomial regression in order to smooth your C_L data check “Use polynomial fit”. If you choose “Use moving average” or “Use polynomial fit” you need to specify a region of data points in which the moving average is going to be applied or in which a best high order polynomial curve is going to be fitted. You can choose this region of data points by the up/down arrows to specify the beginning and the last points. The chosen beginning and last data points are shown in the graph as diamond with yellow inside. You can see the moving average line in the figure with light blue line and the polynomial fit line with purple line. You can also see the chosen data points from purple colored cells in column C. The purple colored cells in column C has to be in the range of your pasted data.

(Look at Figure 5)

You can change the moving average interval and also the order of the polynomial fit. In moving average if the interval is chosen as 1 then there is no smoothing, the smoothing increases as the interval increases. In polynomial fit if the order is increased then the fitted curve will behave more like the original data. You can increase the polynomial regression order until 6.

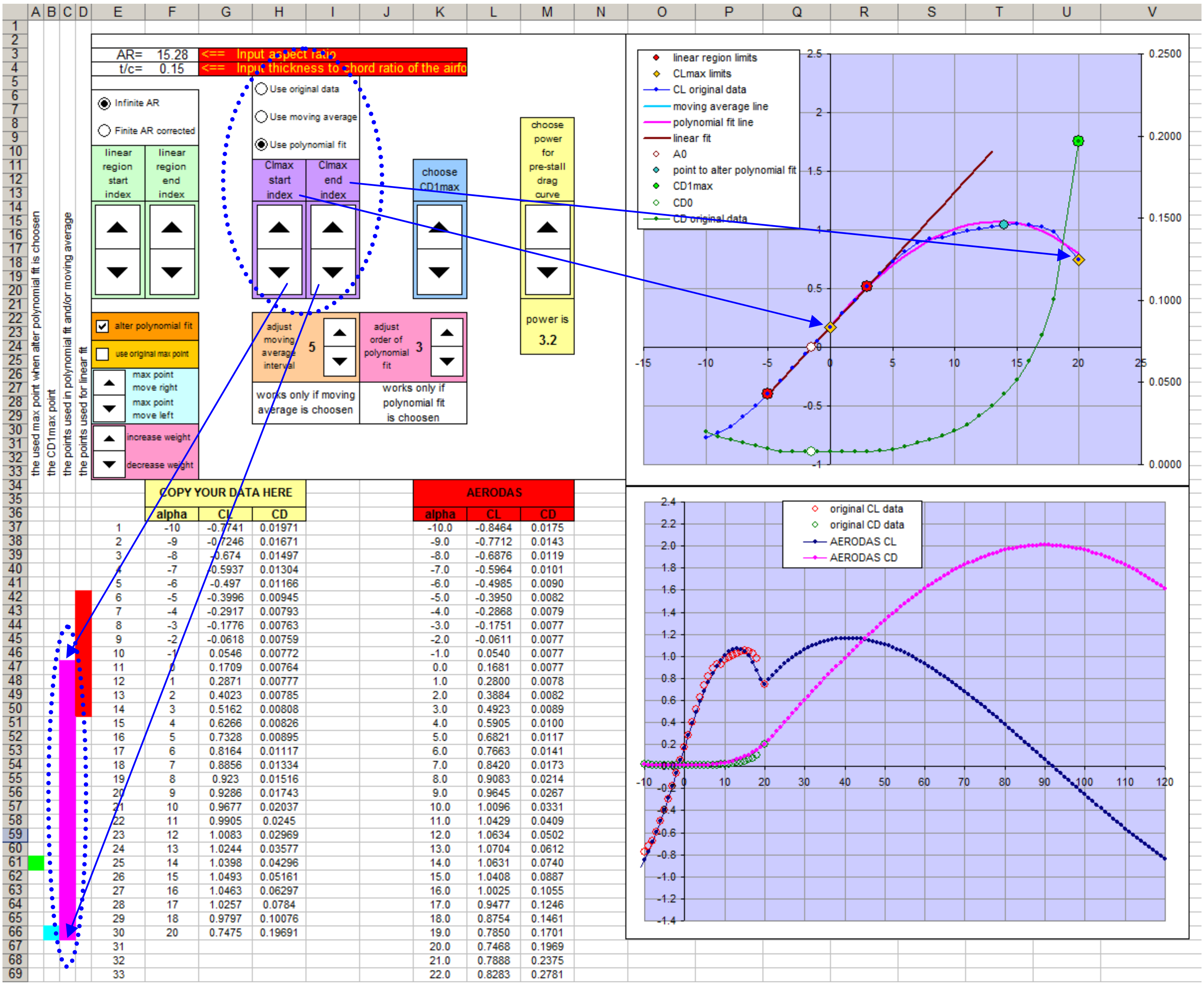
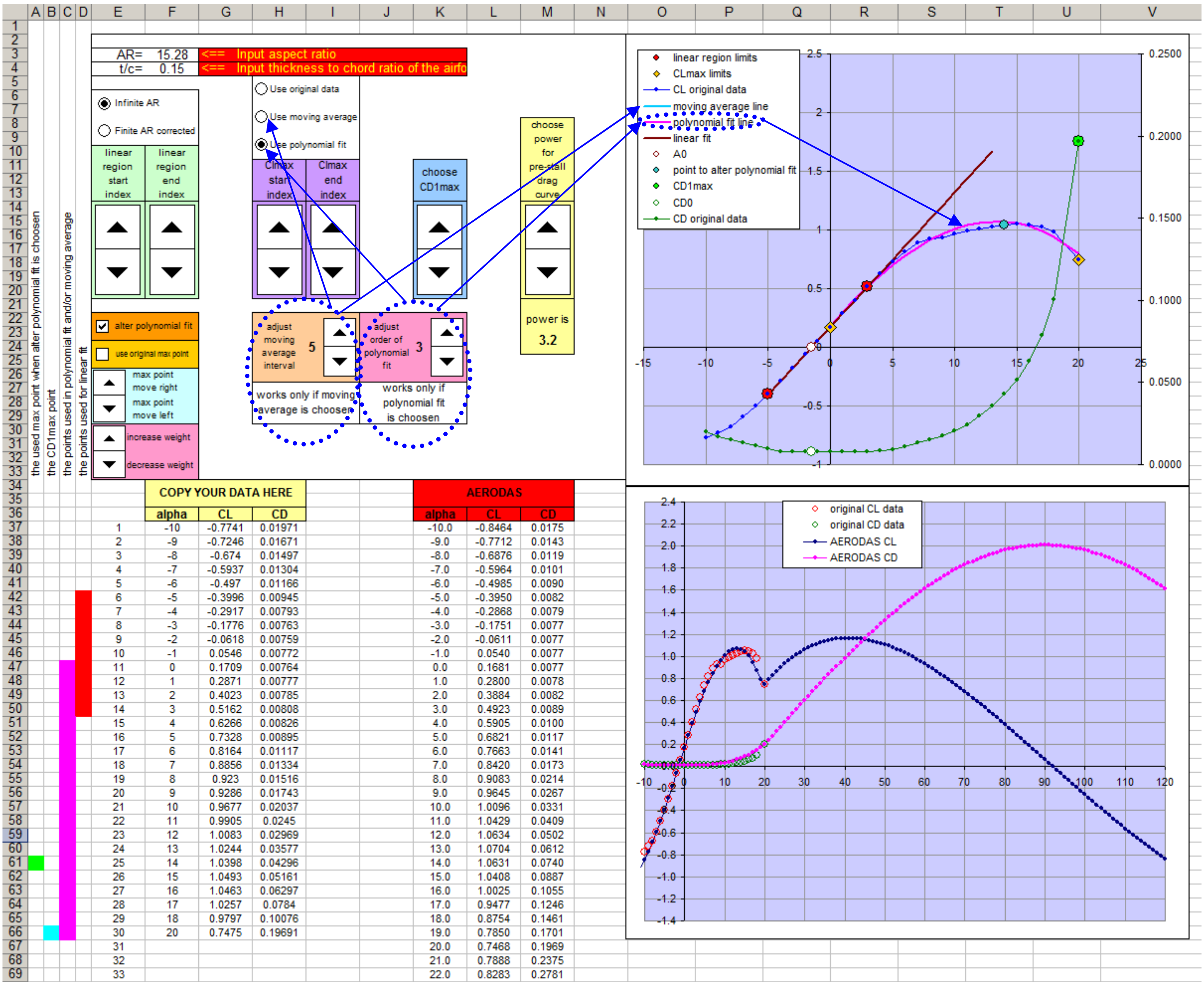

AR= 15.28
t/c= 0.15
<== Input aspect ratio
<== Input thickness to chord ratio of the airfo
Infinite AR
Finite AR corrected
linear region start index
linear region end index
Use original data
Use moving average
Use polynomial fit
Clmax start index
Clmax end index
choose CD1max
choose power for pre-stall drag curve
power is
3.2
alter polynomial fit
use original max point
max point move right
max point move left
increase weight
decrease weight
adjust moving average interval
5
works only if moving average is choosen
adjust order of polynomial fit
3
works only if polynomial fit is choosen
the used max point when alter polynomial fit is choosen
the CD1max point
the points used in polynomial fit and/or moving average
the points used for linear fit
COPY YOUR DATA HERE
alpha
CL
CD
AERODAS
linear region limits
CLmax limits
CL original data
moving average line
polynomial fit line
linear fit
A0
point to alter polynomial fit
CD1max
CD0
CD original data
original CL data
original CD data
AERODAS CL
AERODAS CD

Figure 4



AR= 15.28
<== Input aspect ratio
t/c= 0.15
<== Input thickness to chord ratio of the airfo
Use original data
Use moving average
Use polynomial fit
Infinite AR
Finite AR corrected
linear region start index
linear region end index
Clmax start index
Clmax end index
choose CD1max
choose power for pre-stall drag curve
power is 3.2
alter polynomial fit
use original max point
max point move right
max point move left
increase weight
decrease weight
adjust moving average interval 5
works only if moving average is choosen
adjust order of polynomial fit 3
works only if polynomial fit is choosen
the used max point when alter polynomial fit is choosen
the CD1max point
the points used in polynomial fit and/or moving average
the points used for linear fit
COPY YOUR DATA HERE
alpha CL CD
AERODAS
alpha CL CD
linear region limits
CLmax limits
CL original data
moving average line
polynomial fit line
linear fit
A0
point to alter polynomial fit
CD1max
CD0
CD original data
original CL data
original CD data
AERODAS CL
AERODAS CD

Figure 5

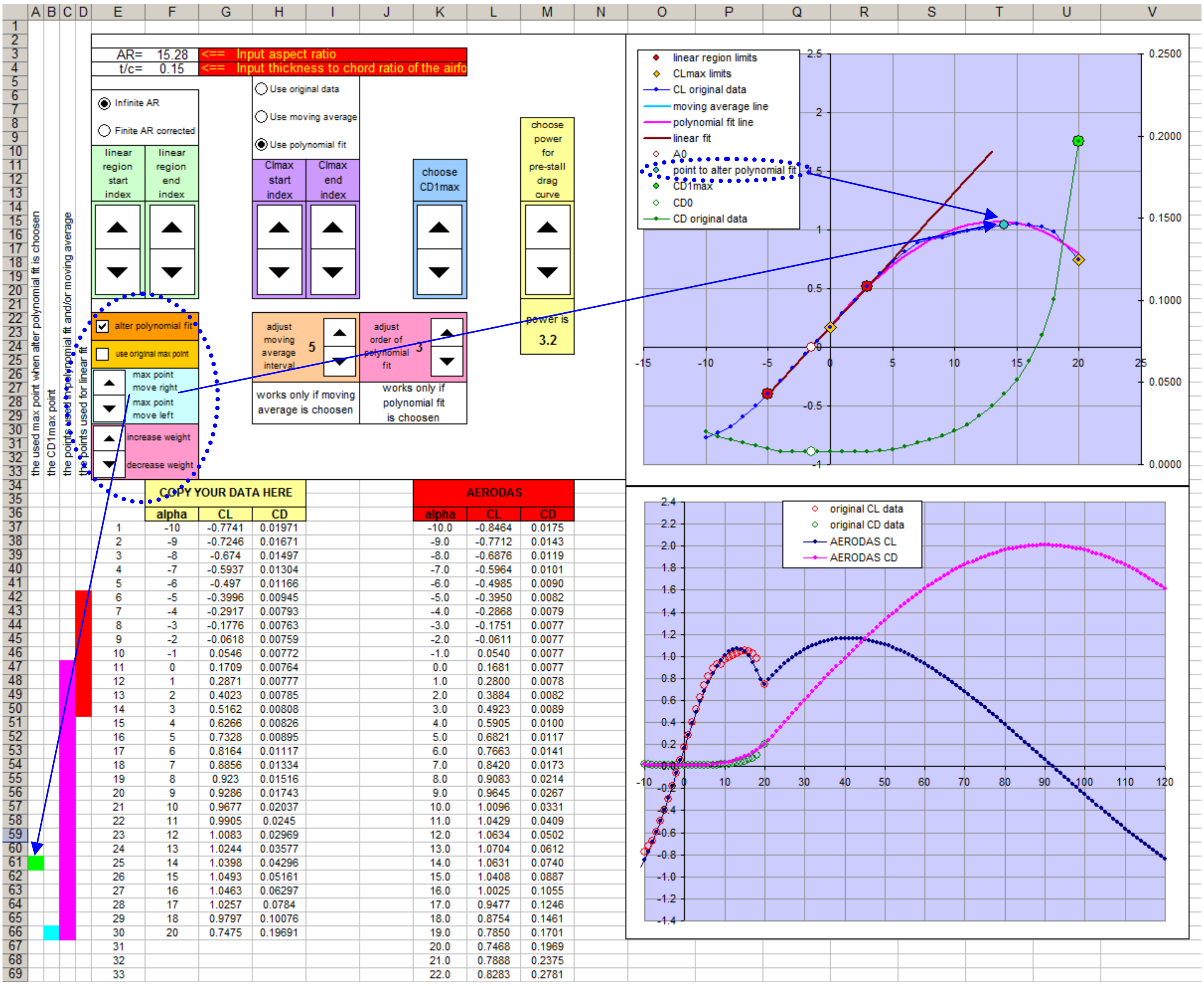
(Look at Figure 6)

If neither moving average nor polynomial regression with any order fits good to your C_L data, AERODAS.xls offers you to alter the polynomial regression fit. In order to use this feature you should choose “Use polynomial fit”. Otherwise when “Use original data” or “Use moving average” is chosen this feature is disabled. In order to alter the fitted polynomial curve you should choose “alter polynomial fit”. The AERODAS.xls alter the polynomial fit by increasing the weight of a chosen point. If you want this point to be the maximum data in your original C_L data check “use original max point”. If this is chosen you can increase the weight of this data however you can not change the point. If you would like to change the point as well then uncheck “use original max point”. By using the up and down arrows you can change the location of the point and the weight of this point. The chosen point is shown in the figure and the corresponding row of the point is colored in column A. The chosen point should always be in the range of your data.

(Look at Figure 7)

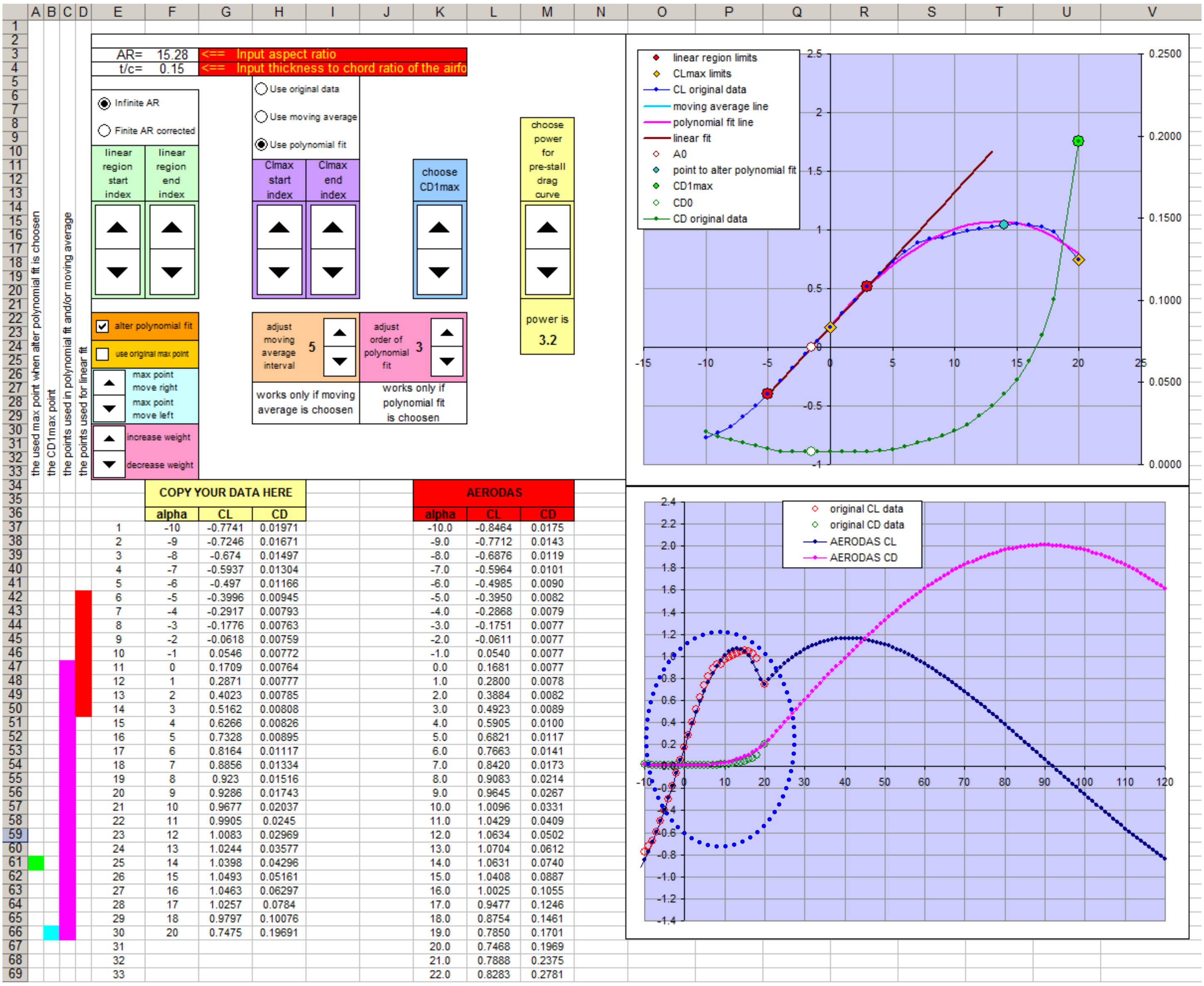
I personally use polynomial fit more often. Altering the polynomial fit is convenient in some cases. In your analysis try to have a curve that best describe your original C_L data.

With this step the analysis for the lift coefficient (C_L) is done.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.7741	0.01971	-10.0	-0.8464	0.0175
2	-9	-0.7246	0.01671	-9.0	-0.7712	0.0143
3	-8	-0.674	0.01497	-8.0	-0.6876	0.0119
4	-7	-0.5937	0.01304	-7.0	-0.5964	0.0101
5	-6	-0.497	0.01166	-6.0	-0.4985	0.0090
6	-5	-0.3996	0.00945	-5.0	-0.3950	0.0082
7	-4	-0.2917	0.00793	-4.0	-0.2868	0.0079
8	-3	-0.1776	0.00763	-3.0	-0.1751	0.0077
9	-2	-0.0618	0.00759	-2.0	-0.0611	0.0077
10	-1	0.0546	0.00772	-1.0	0.0540	0.0077
11	0	0.1709	0.00764	0.0	0.1681	0.0077
12	1	0.2871	0.00777	1.0	0.2800	0.0078
13	2	0.4023	0.00785	2.0	0.3884	0.0082
14	3	0.5162	0.00808	3.0	0.4923	0.0089
15	4	0.6266	0.00826	4.0	0.5905	0.0100
16	5	0.7328	0.00895	5.0	0.6821	0.0117
17	6	0.8164	0.01117	6.0	0.7663	0.0141
18	7	0.8856	0.01334	7.0	0.8420	0.0173
19	8	0.923	0.01516	8.0	0.9083	0.0214
20	9	0.9286	0.01743	9.0	0.9645	0.0267
21	10	0.9677	0.02037	10.0	1.0096	0.0331
22	11	0.9905	0.0245	11.0	1.0429	0.0409
23	12	1.0083	0.02969	12.0	1.0634	0.0502
24	13	1.0244	0.03577	13.0	1.0704	0.0612
25	14	1.0398	0.04296	14.0	1.0631	0.0740
26	15	1.0493	0.05161	15.0	1.0408	0.0887
27	16	1.0463	0.06297	16.0	1.0025	0.1055
28	17	1.0257	0.0784	17.0	0.9477	0.1246
29	18	0.9797	0.10076	18.0	0.8754	0.1461
30	20	0.7475	0.19691	19.0	0.7850	0.1701
31				20.0	0.7468	0.1969
32				21.0	0.7888	0.2375
33				22.0	0.8283	0.2781

Figure 6



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.7741	0.01971	-10.0	-0.8464	0.0175
2	-9	-0.7246	0.01671	-9.0	-0.7712	0.0143
3	-8	-0.674	0.01497	-8.0	-0.6876	0.0119
4	-7	-0.5937	0.01304	-7.0	-0.5964	0.0101
5	-6	-0.497	0.01166	-6.0	-0.4985	0.0090
6	-5	-0.3996	0.00945	-5.0	-0.3950	0.0082
7	-4	-0.2917	0.00793	-4.0	-0.2868	0.0079
8	-3	-0.1776	0.00763	-3.0	-0.1751	0.0077
9	-2	-0.0618	0.00759	-2.0	-0.0611	0.0077
10	-1	0.0546	0.00772	-1.0	0.0540	0.0077
11	0	0.1709	0.00764	0.0	0.1681	0.0077
12	1	0.2871	0.00777	1.0	0.2800	0.0078
13	2	0.4023	0.00785	2.0	0.3884	0.0082
14	3	0.5162	0.00808	3.0	0.4923	0.0089
15	4	0.6266	0.00826	4.0	0.5905	0.0100
16	5	0.7328	0.00895	5.0	0.6821	0.0117
17	6	0.8164	0.01117	6.0	0.7663	0.0141
18	7	0.8856	0.01334	7.0	0.8420	0.0173
19	8	0.923	0.01516	8.0	0.9083	0.0214
20	9	0.9286	0.01743	9.0	0.9645	0.0267
21	10	0.9677	0.02037	10.0	1.0096	0.0331
22	11	0.9905	0.0245	11.0	1.0429	0.0409
23	12	1.0083	0.02969	12.0	1.0634	0.0502
24	13	1.0244	0.03577	13.0	1.0704	0.0612
25	14	1.0398	0.04296	14.0	1.0631	0.0740
26	15	1.0493	0.05161	15.0	1.0408	0.0887
27	16	1.0463	0.06297	16.0	1.0025	0.1055
28	17	1.0257	0.0784	17.0	0.9477	0.1246
29	18	0.9797	0.10076	18.0	0.8754	0.1461
30	20	0.7475	0.19691	19.0	0.7850	0.1701
31				20.0	0.7468	0.1969
32				21.0	0.7888	0.2375
33				22.0	0.8283	0.2781

Figure 7

STEP 4:

(Look at Figure 8)

For the drag coefficient the AERODAS model needs CD1max and ACD1 values. ACD1 and CD1max defines the point that separates the pre-stall region and the post-stall region for the drag coefficient. ACD1 is the angle of attack (alpha) and CD1max is the C_D value at this alpha in the original data. Using the up and down arrows choose the dividing point of the pre-stall and post-stall regions. The circle with green inside shows this point in the figure and the row of this data point is colored in column B. The colored row should always be in the range of your original data.

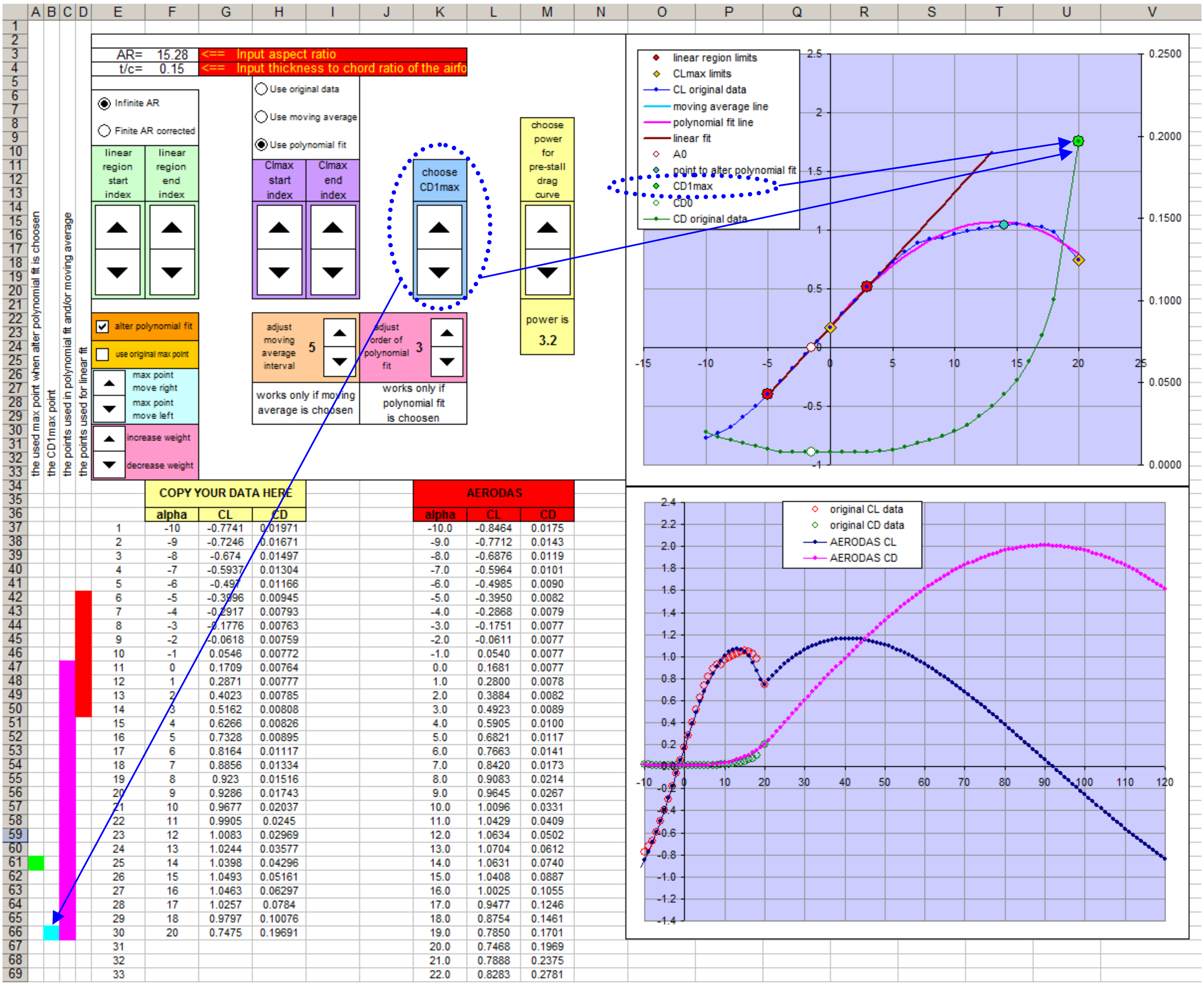
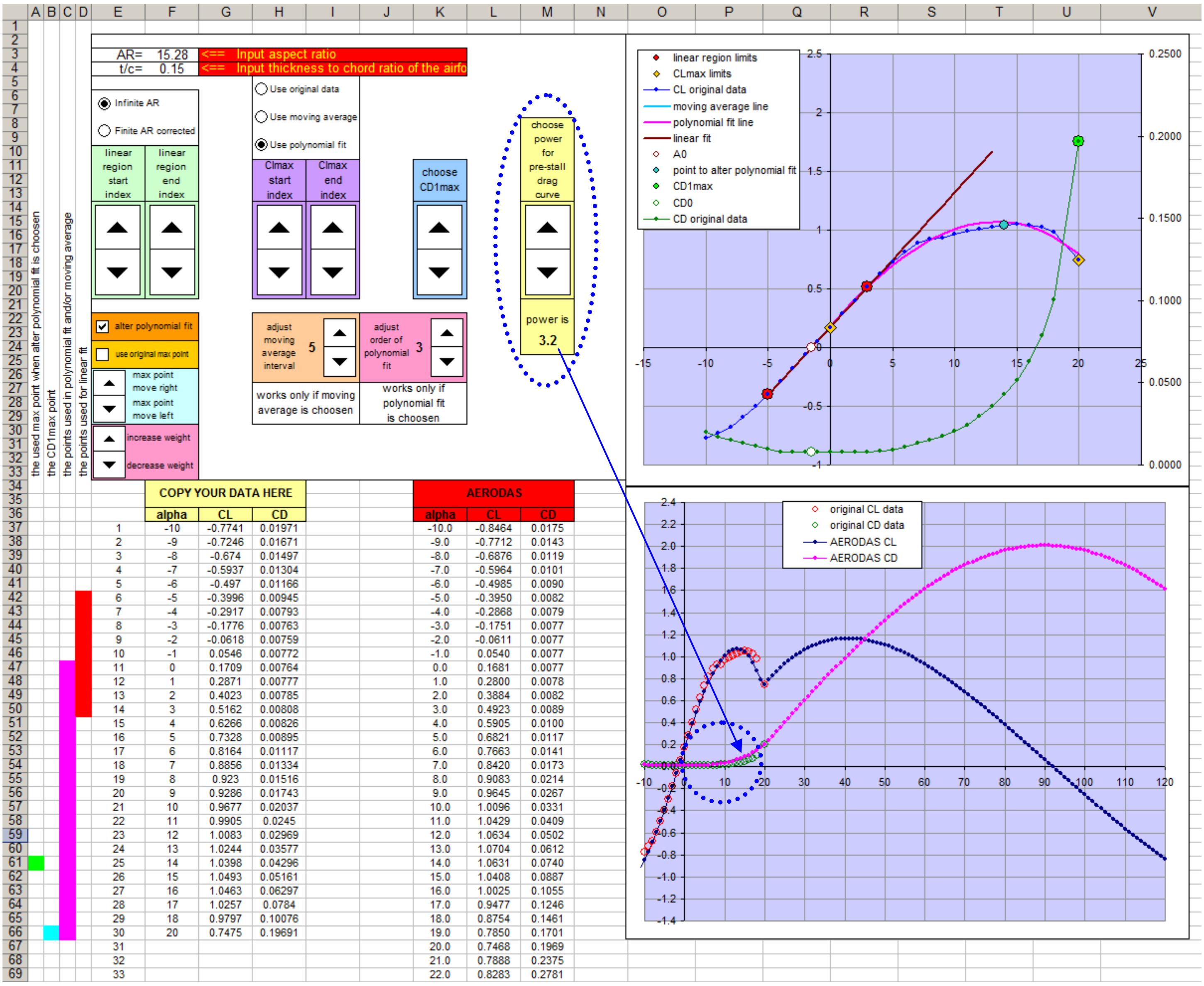


Figure 8

STEP 5:

(Look at Figure 9)

The last step is to select the power for the pre-stall drag curve. Use arrows and choose the power that best fits to your original data. As the power increases the curve gets more flat and this mainly affects the curve near the chosen CD1max point in the pre-stall region.

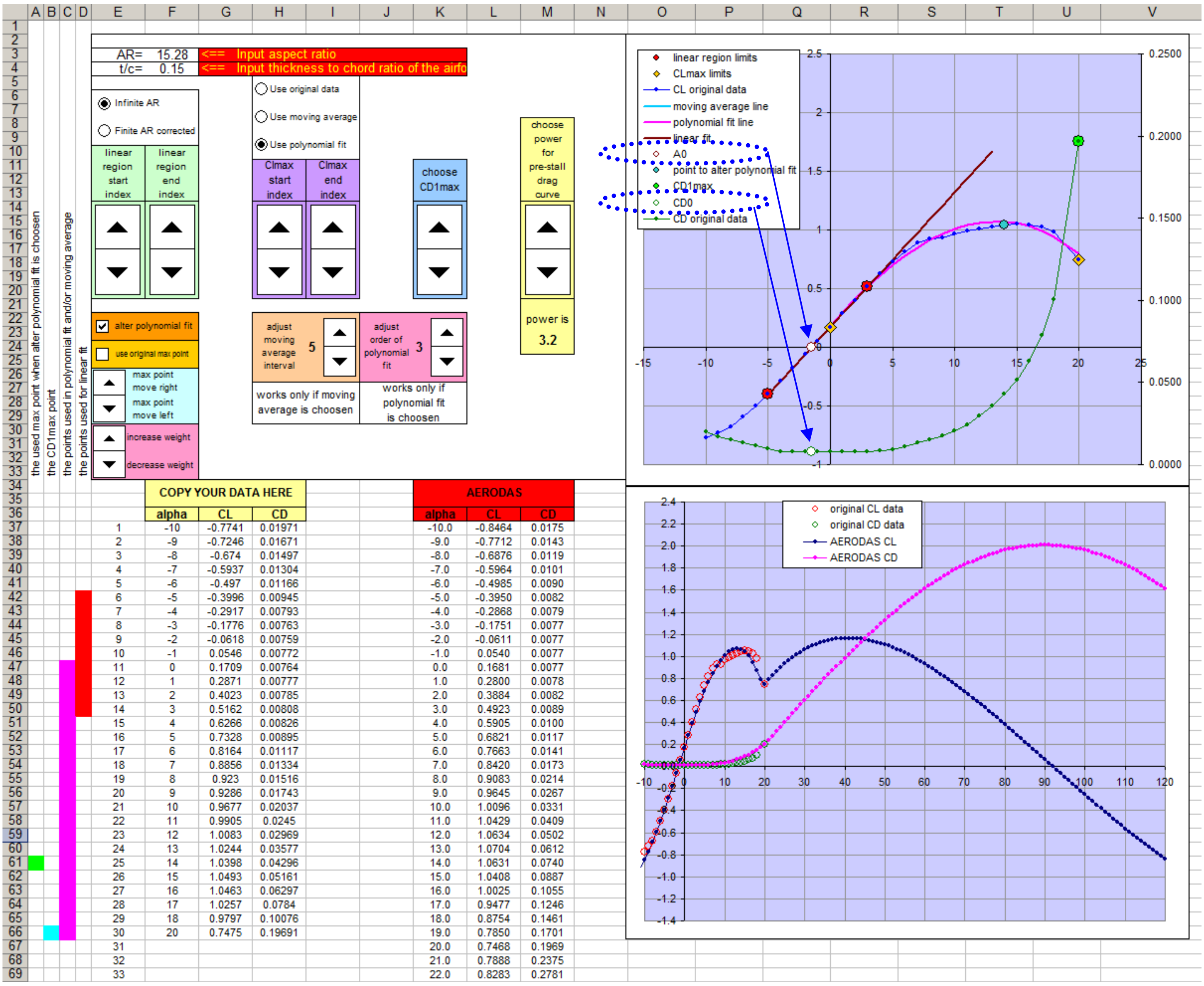


	alpha	CL	CD	alpha	CL	CD
1	-10	-0.7741	0.01971	-10.0	-0.8464	0.0175
2	-9	-0.7246	0.01671	-9.0	-0.7712	0.0143
3	-8	-0.674	0.01497	-8.0	-0.6876	0.0119
4	-7	-0.5937	0.01304	-7.0	-0.5964	0.0101
5	-6	-0.497	0.01166	-6.0	-0.4985	0.0090
6	-5	-0.3996	0.00945	-5.0	-0.3950	0.0082
7	-4	-0.2917	0.00793	-4.0	-0.2868	0.0079
8	-3	-0.1776	0.00763	-3.0	-0.1751	0.0077
9	-2	-0.0618	0.00759	-2.0	-0.0611	0.0077
10	-1	0.0546	0.00772	-1.0	0.0540	0.0077
11	0	0.1709	0.00764	0.0	0.1681	0.0077
12	1	0.2871	0.00777	1.0	0.2800	0.0078
13	2	0.4023	0.00785	2.0	0.3884	0.0082
14	3	0.5162	0.00808	3.0	0.4923	0.0089
15	4	0.6266	0.00826	4.0	0.5905	0.0100
16	5	0.7328	0.00895	5.0	0.6821	0.0117
17	6	0.8164	0.01117	6.0	0.7663	0.0141
18	7	0.8856	0.01334	7.0	0.8420	0.0173
19	8	0.923	0.01516	8.0	0.9083	0.0214
20	9	0.9286	0.01743	9.0	0.9645	0.0267
21	10	0.9677	0.02037	10.0	1.0096	0.0331
22	11	0.9905	0.0245	11.0	1.0429	0.0409
23	12	1.0083	0.02969	12.0	1.0634	0.0502
24	13	1.0244	0.03577	13.0	1.0704	0.0612
25	14	1.0398	0.04296	14.0	1.0631	0.0740
26	15	1.0493	0.05161	15.0	1.0408	0.0887
27	16	1.0463	0.06297	16.0	1.0025	0.1055
28	17	1.0257	0.0784	17.0	0.9477	0.1246
29	18	0.9797	0.10076	18.0	0.8754	0.1461
30	20	0.7475	0.19691	19.0	0.7850	0.1701
31				20.0	0.7468	0.1969
32				21.0	0.7888	0.2375
33				22.0	0.8283	0.2781

Figure 9

(Look at Figure 10)

With step 5 the analysis is finished for the **infinite aspect ratio**. The model uses the angle of attack of the zero lift point (A0) and also the corresponding drag coefficient at this angle of attack (CD0). These points are shown in the figure also.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.7741	0.01971	-10.0	-0.8464	0.0175
2	-9	-0.7246	0.01671	-9.0	-0.7712	0.0143
3	-8	-0.674	0.01497	-8.0	-0.6876	0.0119
4	-7	-0.5937	0.01304	-7.0	-0.5964	0.0101
5	-6	-0.497	0.01166	-6.0	-0.4985	0.0090
6	-5	-0.3996	0.00945	-5.0	-0.3950	0.0082
7	-4	-0.2917	0.00793	-4.0	-0.2868	0.0079
8	-3	-0.1776	0.00763	-3.0	-0.1751	0.0077
9	-2	-0.0618	0.00759	-2.0	-0.0611	0.0077
10	-1	0.0546	0.00772	-1.0	0.0540	0.0077
11	0	0.1709	0.00764	0.0	0.1681	0.0077
12	1	0.2871	0.00777	1.0	0.2800	0.0078
13	2	0.4023	0.00785	2.0	0.3884	0.0082
14	3	0.5162	0.00808	3.0	0.4923	0.0089
15	4	0.6266	0.00826	4.0	0.5905	0.0100
16	5	0.7328	0.00895	5.0	0.6821	0.0117
17	6	0.8164	0.01117	6.0	0.7663	0.0141
18	7	0.8856	0.01334	7.0	0.8420	0.0173
19	8	0.923	0.01516	8.0	0.9083	0.0214
20	9	0.9286	0.01743	9.0	0.9645	0.0267
21	10	0.9677	0.02037	10.0	1.0096	0.0331
22	11	0.9905	0.0245	11.0	1.0429	0.0409
23	12	1.0083	0.02969	12.0	1.0634	0.0502
24	13	1.0244	0.03577	13.0	1.0704	0.0612
25	14	1.0398	0.04296	14.0	1.0631	0.0740
26	15	1.0493	0.05161	15.0	1.0408	0.0887
27	16	1.0463	0.06297	16.0	1.0025	0.1055
28	17	1.0257	0.0784	17.0	0.9477	0.1246
29	18	0.9797	0.10076	18.0	0.8754	0.1461
30	20	0.7475	0.19691	19.0	0.7850	0.1701
31				20.0	0.7468	0.1969
32				21.0	0.7888	0.2375
33				22.0	0.8283	0.2781

Figure 10

(Look at Figure 11)

Finally the calculated lift and drag coefficients by the AERODAS model are tabulated and also plotted in the figure. Note that these coefficients are for **infinite aspect ratio**.

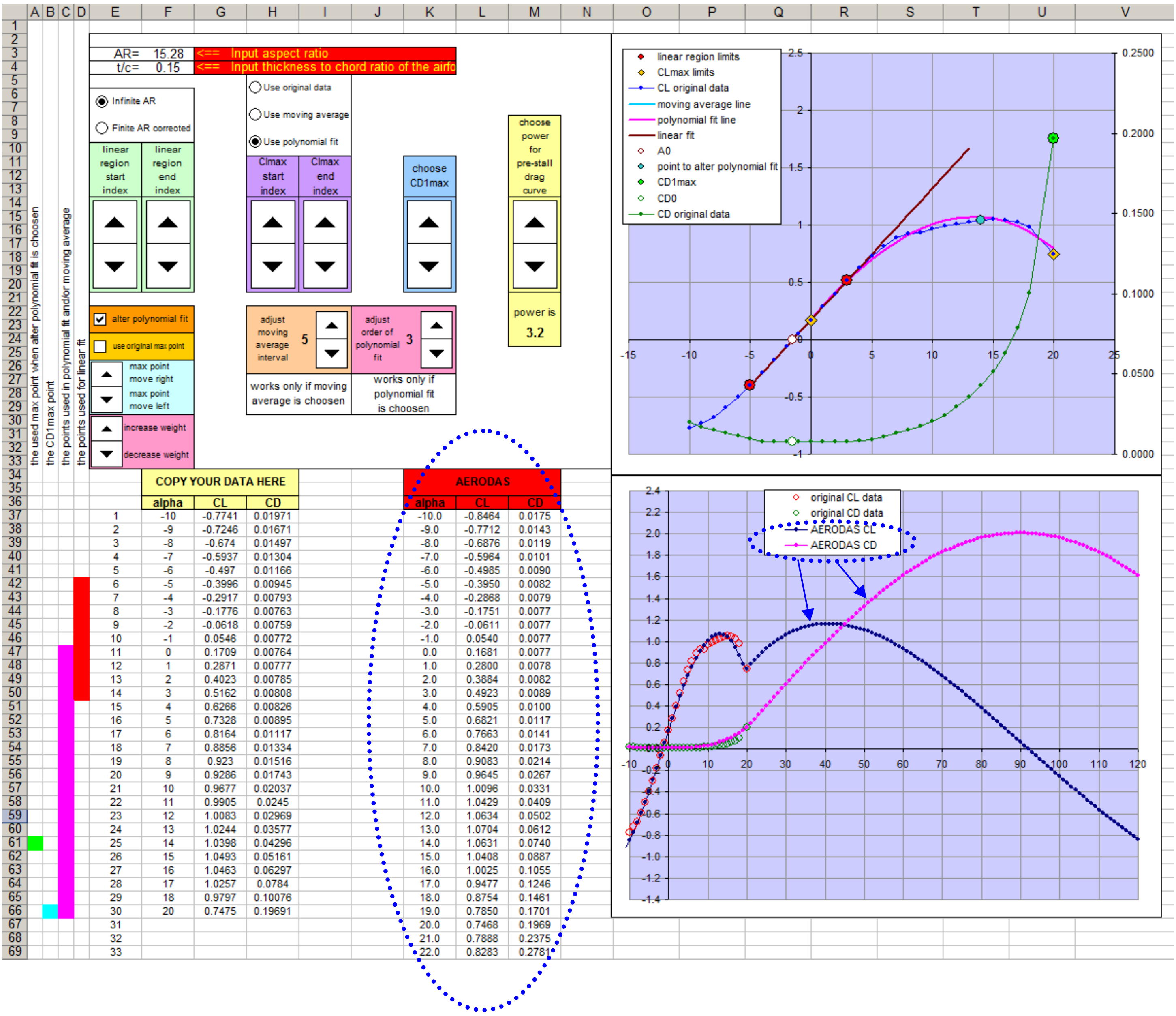
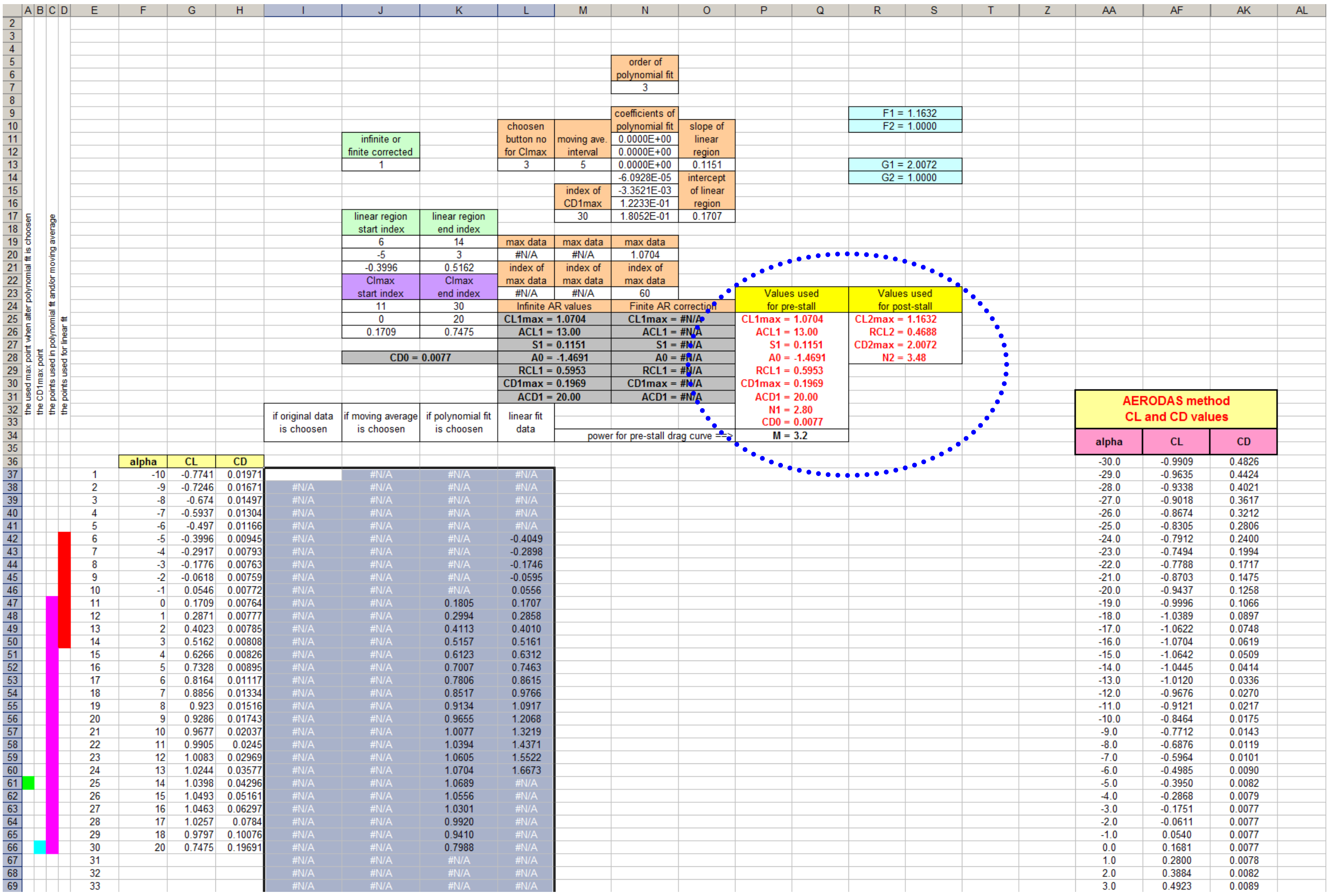


Figure 11

The AERODAS.xls has two sheets; data sheet and calculation sheet. All of the calculations are done on the calculations sheet. In calculations sheet some of the empty cells are not actually empty. The font color of some cells are conditionally becomes white such that the user sees only the used data in calculations (you can see this by selecting the cells). I am frustrated with the fact that every paper in the literature that uses Blade Element Momentum (BEM) calculations give every detail about equations and so on, but do not give any information about the polar data they used in the BEM calculations. BEM calculations are very sensitive to the polar data. It is really hard, almost impossible, to reproduce the same results in the literature if polar datas are not shared in papers. If you are going to use AERODAS model in your study and want to share your polar data, all you have to do is to share the parameters used in the model in a table as it was done in Spera [4]. These parameters are shown in the figure below. These parameters are calculated, however if you overwrite the values of these parameters in the cells, you can reproduce a shared polar data.



SOME EXAMPLE RESULTS AND COMPARISONS

<u>NACA 63-215</u>

We note that in the figures (Figures 1 to 11) the lift and drag data of airfoil NACA 63-215 obtained by XFOIL direct analysis (Re=0.55×10^{6}) was used. The XFOIL output polar file is below.

```
XFLR5 v6.09.01 beta

 Calculated polar for: NACA 63-215 AIRFOIL

 1 1 Reynolds number fixed          Mach number fixed

 xtrf =   1.000 (top)        1.000 (bottom)
 Mach =   0.000     Re =     0.550 e 6     Ncrit =   9.000

  alpha     CL        CD       CDp       Cm    Top Xtr  Bot Xtr   Cpmin    Chinge     XCp
 ------- -------- --------- --------- -------- ------- ------- -------- --------- -------
--
 -10.000  -0.7741   0.01971   0.01428  -0.0555  0.9850  0.0200  -4.4380   0.0000   0.1661
  -9.000  -0.7246   0.01671   0.01088  -0.0495  0.9024  0.0219  -4.0119   0.0000   0.1707
  -8.000  -0.6740   0.01497   0.00877  -0.0403  0.8522  0.0248  -3.5825   0.0000   0.1804
  -7.000  -0.5937   0.01304   0.00653  -0.0362  0.8244  0.0389  -3.0994   0.0000   0.1801
  -6.000  -0.4970   0.01166   0.00501  -0.0345  0.7928  0.0644  -2.5535   0.0000   0.1722
  -5.000  -0.3996   0.00945   0.00351  -0.0340  0.7644  0.2305  -2.0112   0.0000   0.1570
  -4.000  -0.2917   0.00793   0.00270  -0.0348  0.7403  0.4505  -1.4466   0.0000   0.1225
  -3.000  -0.1776   0.00763   0.00243  -0.0355  0.7108  0.5235  -0.9380   0.0000   0.0410
  -2.000  -0.0618   0.00759   0.00248  -0.0364  0.6887  0.5792  -0.5739   0.0000  -0.3552
  -1.000   0.0546   0.00772   0.00243  -0.0374  0.6613  0.6009  -0.5215   0.0000   0.9415
   0.000   0.1709   0.00764   0.00248  -0.0386  0.6409  0.6344  -0.5880   0.0000   0.4744
   1.000   0.2871   0.00777   0.00250  -0.0396  0.6148  0.6536  -0.6641   0.0000   0.3850
   2.000   0.4023   0.00785   0.00272  -0.0406  0.5919  0.6848  -0.7511   0.0000   0.3469
   3.000   0.5162   0.00808   0.00281  -0.0412  0.5489  0.7055  -0.9241   0.0000   0.3250
   4.000   0.6266   0.00826   0.00300  -0.0411  0.4878  0.7388  -1.4164   0.0000   0.3099
   5.000   0.7328   0.00895   0.00348  -0.0407  0.4024  0.7651  -1.9940   0.0000   0.2989
   6.000   0.8164   0.01117   0.00503  -0.0371  0.2054  0.7941  -2.5795   0.0000   0.2878
   7.000   0.8856   0.01334   0.00675  -0.0310  0.0828  0.8321  -3.1237   0.0000   0.2763
   8.000   0.9230   0.01516   0.00852  -0.0188  0.0374  0.8779  -3.5728   0.0000   0.2605
   9.000   0.9286   0.01743   0.01116  -0.0028  0.0218  1.0000  -3.8453   0.0000   0.2418
  10.000   0.9677   0.02037   0.01424   0.0043  0.0181  1.0000  -4.3061   0.0000   0.2329
  11.000   0.9905   0.02450   0.01855   0.0119  0.0163  1.0000  -4.6999   0.0000   0.2238
  12.000   1.0083   0.02969   0.02400   0.0182  0.0153  1.0000  -5.0595   0.0000   0.2161
  13.000   1.0244   0.03577   0.03036   0.0228  0.0147  1.0000  -5.4064   0.0000   0.2100
  14.000   1.0398   0.04296   0.03800   0.0259  0.0145  1.0000  -5.7160   0.0000   0.2056
  15.000   1.0493   0.05161   0.04700   0.0269  0.0140  1.0000  -6.0147   0.0000   0.2029
  16.000   1.0463   0.06297   0.05878   0.0254  0.0135  1.0000  -6.2286   0.0000   0.2022
  17.000   1.0257   0.07840   0.07468   0.0203  0.0132  1.0000  -6.3003   0.0000   0.2045
  18.000   0.9797   0.10076   0.09759   0.0090  0.0130  1.0000  -6.1280   0.0000   0.2130
  20.000   0.7475   0.19691   0.19459  -0.0414  0.0135  1.0000  -2.9431   0.0000   0.2777
```

The following figure is taken from Lindenburg [3]. This figure shows the aerodynamic coefficients for NACA 63-215 for Re=0.55×10^{6} for infinite aspect ratio. The aerodynamic coefficients in this figure are comparable with the calculated C_L and C_D values by AERODAS model in the above figures (Figures 1 to 11).

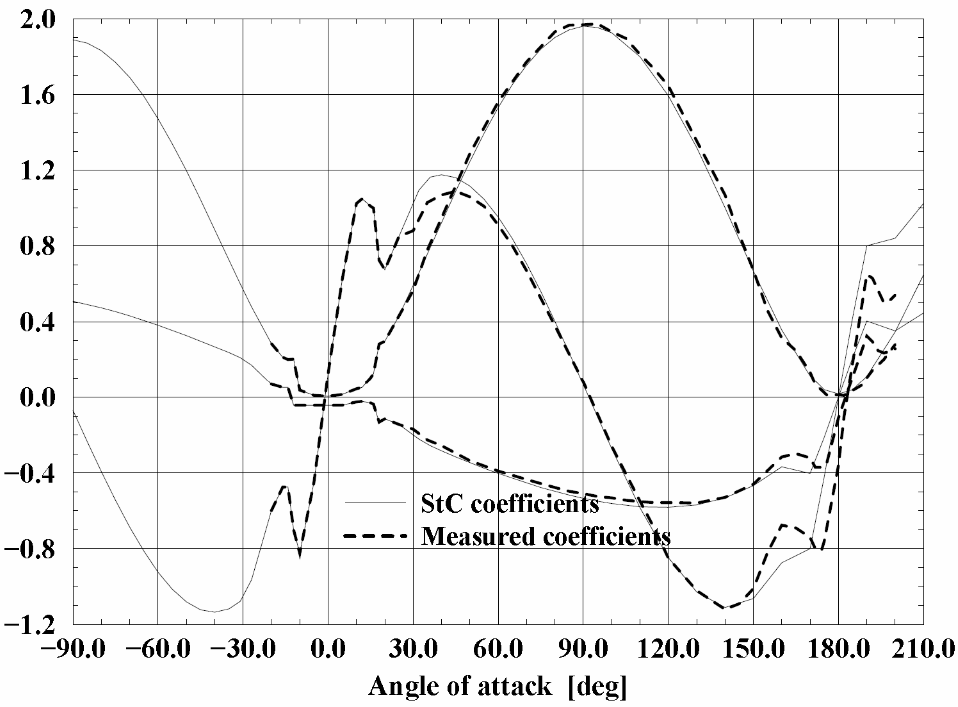


Figure 5. *Coefficients for the NACA63-215 airfoil, infinite aspect ratio.*

CLARK-Y

As an other example, the following figure is taken from Knight and Werzinger [5]. This figure shows the experimental results of Clark-Y airfoil for Re=153000.

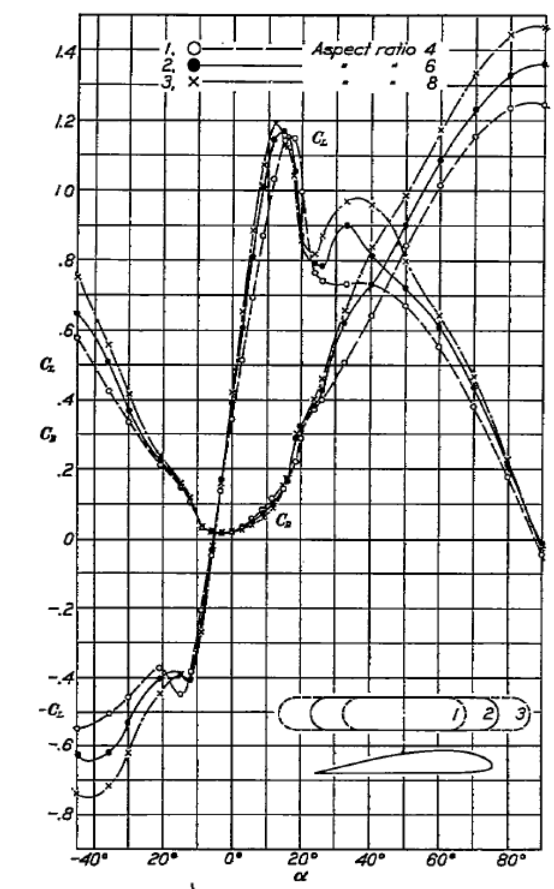
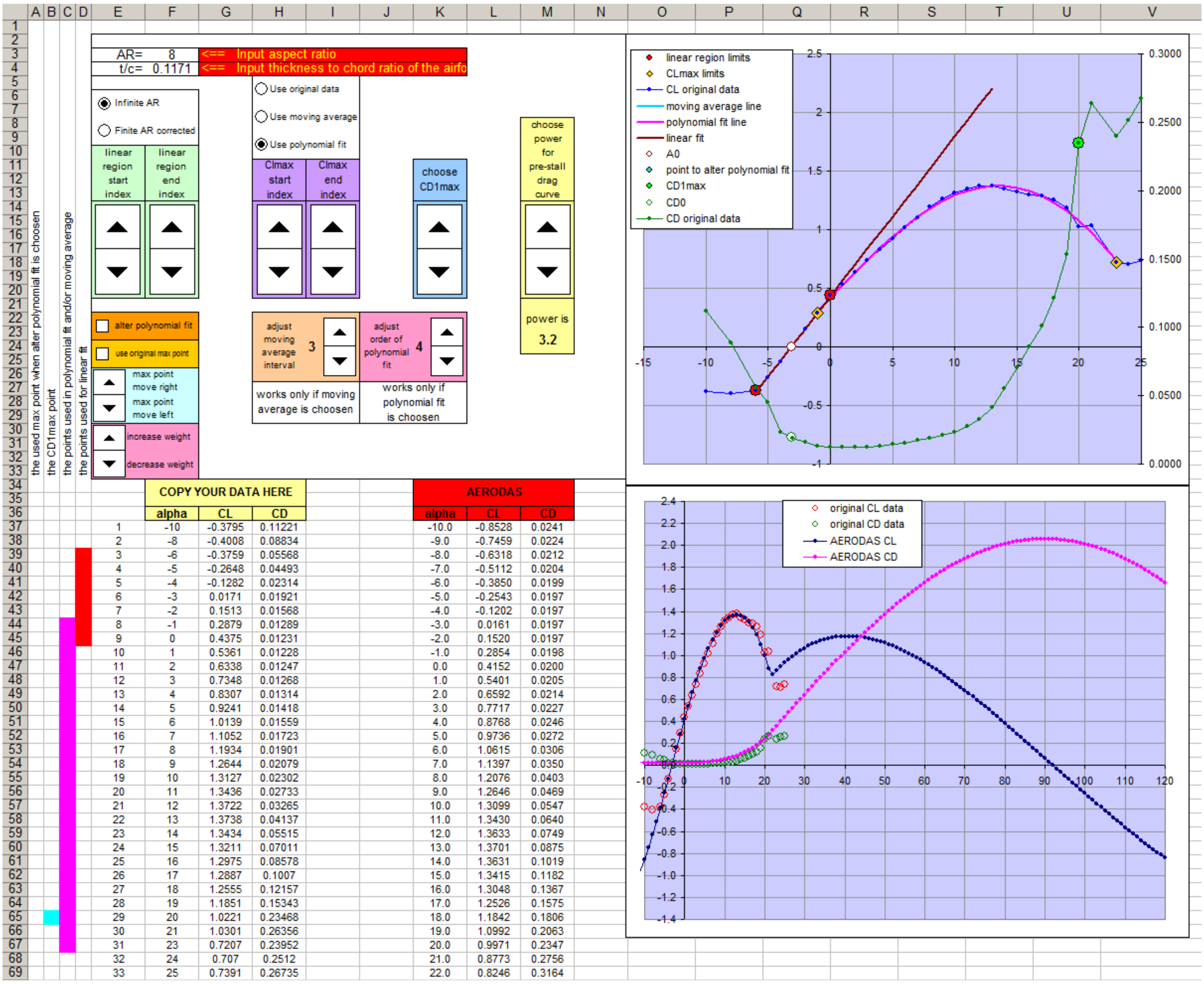


FIGURE 8.—Monoplane wings. Aspect ratio effect. Clark Y. Circular tips. 5-inch chord

The following is the XFOIL output polar file for Clark-Y airfoil for Re=153000.

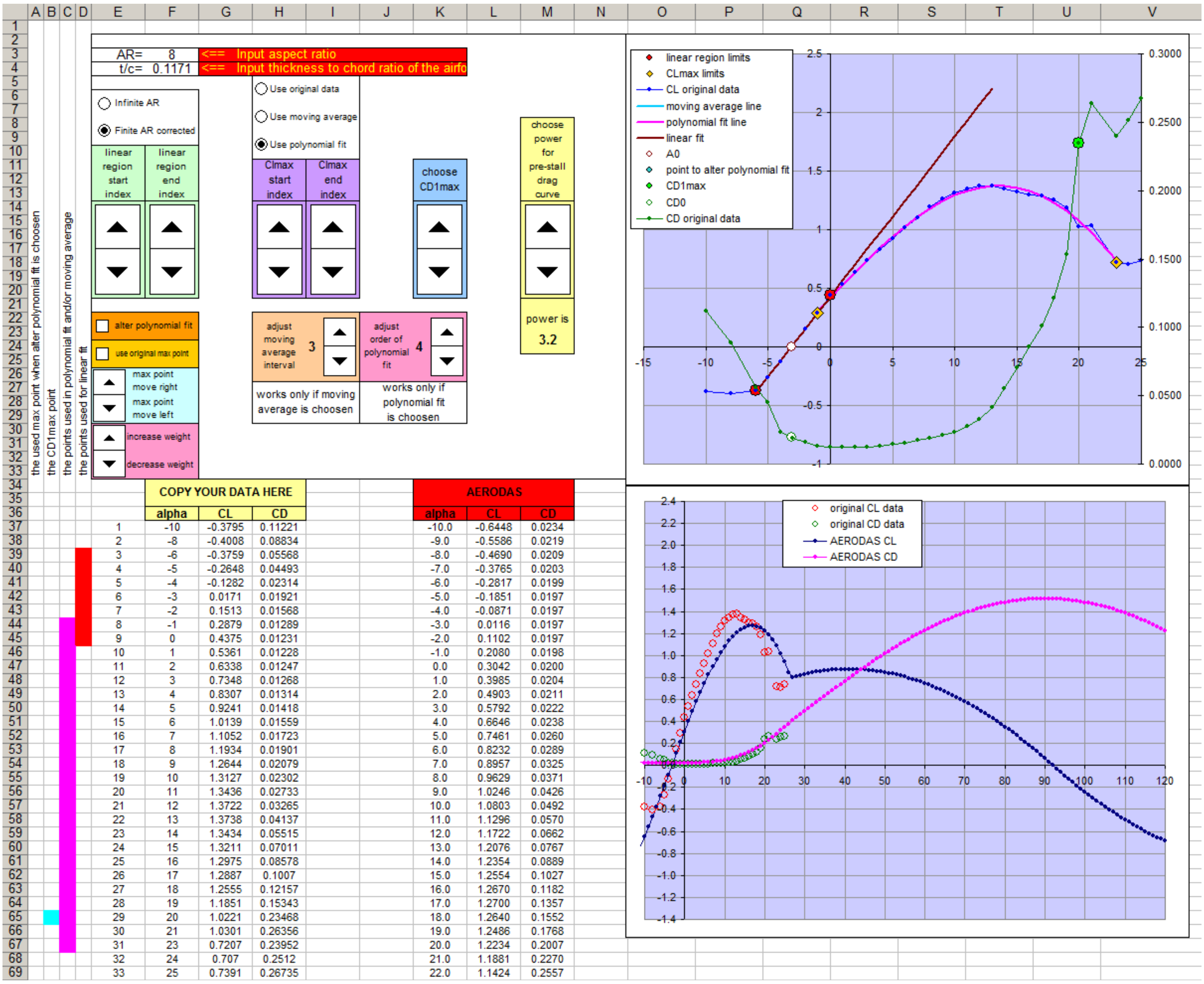
```
XFLR5 v6.09.01 beta

 Calculated polar for: CLARK-Y

 1 1 Reynolds number fixed          Mach number fixed

 xtrf =   1.000 (top)        1.000 (bottom)
 Mach =   0.000     Re =     0.153 e 6     Ncrit =   9.000

  alpha     CL        CD       CDp       Cm    Top Xtr Bot Xtr   Cpmin    Chinge     XCp
 ------- -------- --------- --------- -------- ------- ------- -------- --------- -------
--
 -10.000  -0.3795   0.11221   0.10810  -0.0441  1.0000  0.0798  -1.9117   0.0000   0.1223
  -8.000  -0.4008   0.08834   0.08462  -0.0346  1.0000  0.0966  -1.7953   0.0000   0.1555
  -6.000  -0.3759   0.05568   0.05155  -0.0607  0.9719  0.1234  -1.9493   0.0000   0.0821
  -5.000  -0.2648   0.04493   0.04020  -0.0711  0.9466  0.1569  -1.5967   0.0000  -0.0258
  -4.000  -0.1282   0.02314   0.01518  -0.0792  0.9270  0.0759  -1.8770   0.0000  -0.3833
  -3.000   0.0171   0.01921   0.01090  -0.0855  0.9073  0.0894  -1.5565   0.0000   5.3426
  -2.000   0.1513   0.01568   0.00829  -0.0893  0.8835  0.2915  -1.2126   0.0000   0.8461
  -1.000   0.2879   0.01289   0.00748  -0.0901  0.8533  0.9555  -0.7704   0.0000   0.5639
   0.000   0.4375   0.01231   0.00632  -0.0965  0.8208  1.0000  -0.7583   0.0000   0.4695
   1.000   0.5361   0.01228   0.00588  -0.0928  0.7789  1.0000  -0.8555   0.0000   0.4208
   2.000   0.6338   0.01247   0.00580  -0.0891  0.7320  1.0000  -0.9598   0.0000   0.3871
   3.000   0.7348   0.01268   0.00572  -0.0857  0.6782  1.0000  -1.0726   0.0000   0.3622
   4.000   0.8307   0.01314   0.00596  -0.0816  0.6067  1.0000  -1.2706   0.0000   0.3428
   5.000   0.9241   0.01418   0.00659  -0.0775  0.5223  1.0000  -1.5129   0.0000   0.3273
   6.000   1.0139   0.01559   0.00769  -0.0734  0.4480  1.0000  -1.7680   0.0000   0.3147
   7.000   1.1052   0.01723   0.00923  -0.0699  0.3958  1.0000  -2.0517   0.0000   0.3044
   8.000   1.1934   0.01901   0.01116  -0.0663  0.3499  1.0000  -2.6294   0.0000   0.2953
   9.000   1.2644   0.02079   0.01304  -0.0600  0.2898  1.0000  -3.3213   0.0000   0.2858
  10.000   1.3127   0.02302   0.01534  -0.0506  0.2130  1.0000  -3.9865   0.0000   0.2754
  11.000   1.3436   0.02733   0.01959  -0.0409  0.1524  1.0000  -4.5860   0.0000   0.2657
  12.000   1.3722   0.03265   0.02507  -0.0336  0.1123  1.0000  -5.2214   0.0000   0.2581
  13.000   1.3738   0.04137   0.03391  -0.0276  0.0620  1.0000  -5.6839   0.0000   0.2518
  14.000   1.3434   0.05515   0.04781  -0.0245  0.0406  1.0000  -5.9048   0.0000   0.2479
  15.000   1.3211   0.07011   0.06325  -0.0250  0.0337  1.0000  -6.1503   0.0000   0.2465
  16.000   1.2975   0.08578   0.07923  -0.0258  0.0301  1.0000  -6.3319   0.0000   0.2454
  17.000   1.2887   0.10070   0.09481  -0.0278  0.0285  1.0000  -6.5499   0.0000   0.2450
  18.000   1.2555   0.12157   0.11643  -0.0357  0.0272  1.0000  -6.5232   0.0000   0.2498
  19.000   1.1851   0.15343   0.14912  -0.0537  0.0274  1.0000  -6.0123   0.0000   0.2649
  20.000   1.0221   0.23468   0.23062  -0.0985  0.0390  1.0000  -2.9205   0.0000   0.3184
  21.000   1.0301   0.26356   0.25945  -0.1145  0.0556  1.0000  -2.2118   0.0000   0.3323
  23.000   0.7207   0.23952   0.23599  -0.0698  0.0501  1.0000  -1.4769   0.0000   0.3230
  24.000   0.7070   0.25120   0.24768  -0.0791  0.0442  1.0000  -1.1346   0.0000   0.3374
  25.000   0.7391   0.26735   0.26389  -0.0804  0.0401  1.0000  -1.3127   0.0000   0.3319
```

The following is the polar data obtained by the AERODAS model for Clark-Y airfoil for infinite aspect ratio. Note that we always start the analysis for infinite AR.



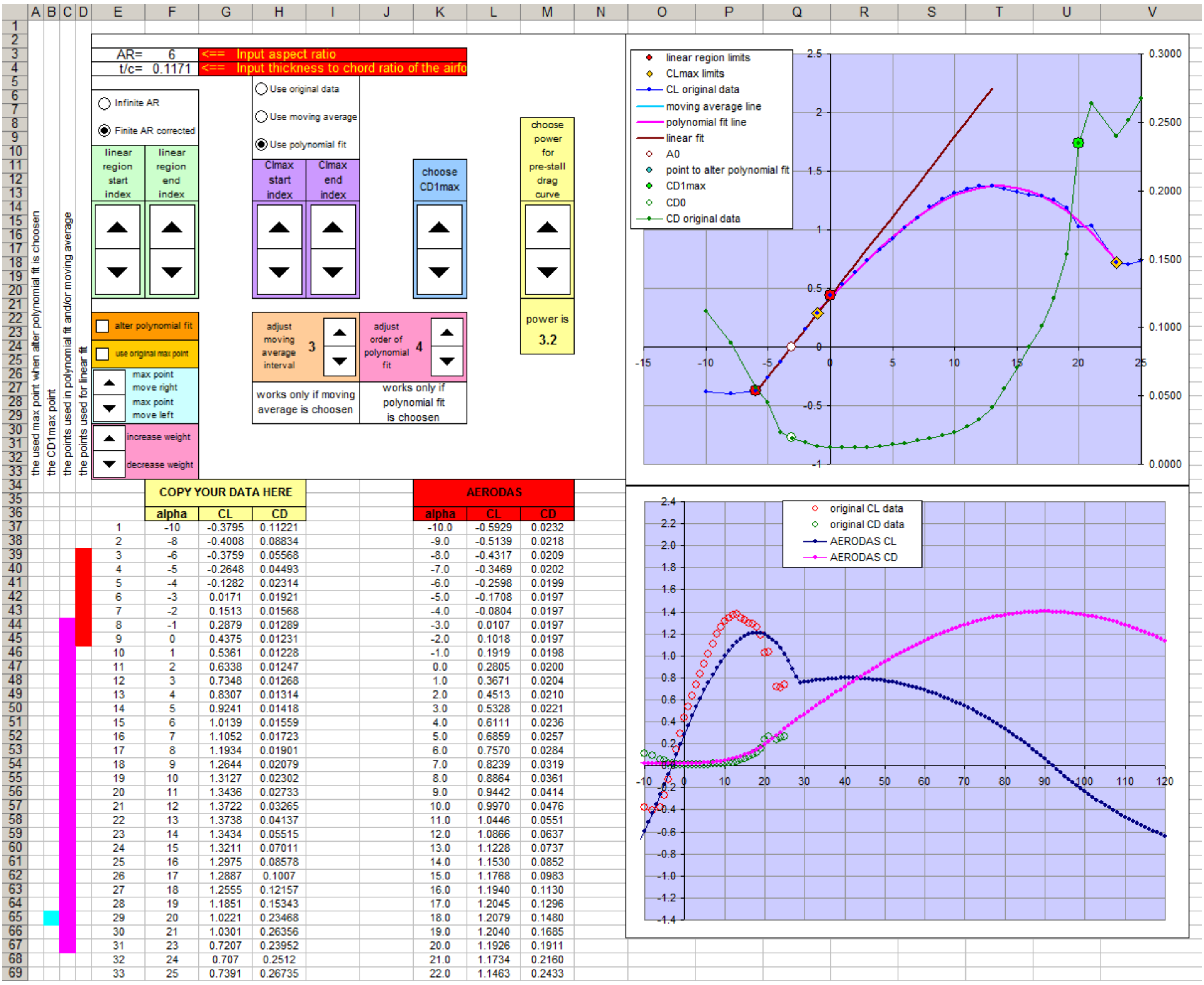
	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.3795	0.11221	-10.0	-0.8528	0.0241
2	-8	-0.4008	0.08834	-9.0	-0.7459	0.0224
3	-6	-0.3759	0.05568	-8.0	-0.6318	0.0212
4	-5	-0.2648	0.04493	-7.0	-0.5112	0.0204
5	-4	-0.1282	0.02314	-6.0	-0.3850	0.0199
6	-3	0.0171	0.01921	-5.0	-0.2543	0.0197
7	-2	0.1513	0.01568	-4.0	-0.1202	0.0197
8	-1	0.2879	0.01289	-3.0	0.0161	0.0197
9	0	0.4375	0.01231	-2.0	0.1520	0.0197
10	1	0.5361	0.01228	-1.0	0.2854	0.0198
11	2	0.6338	0.01247	0.0	0.4152	0.0200
12	3	0.7348	0.01268	1.0	0.5401	0.0205
13	4	0.8307	0.01314	2.0	0.6592	0.0214
14	5	0.9241	0.01418	3.0	0.7717	0.0227
15	6	1.0139	0.01559	4.0	0.8768	0.0246
16	7	1.1052	0.01723	5.0	0.9736	0.0272
17	8	1.1934	0.01901	6.0	1.0615	0.0306
18	9	1.2644	0.02079	7.0	1.1397	0.0350
19	10	1.3127	0.02302	8.0	1.2076	0.0403
20	11	1.3436	0.02733	9.0	1.2646	0.0469
21	12	1.3722	0.03265	10.0	1.3099	0.0547
22	13	1.3738	0.04137	11.0	1.3430	0.0640
23	14	1.3434	0.05515	12.0	1.3633	0.0749
24	15	1.3211	0.07011	13.0	1.3701	0.0875
25	16	1.2975	0.08578	14.0	1.3631	0.1019
26	17	1.2887	0.1007	15.0	1.3415	0.1182
27	18	1.2555	0.12157	16.0	1.3048	0.1367
28	19	1.1851	0.15343	17.0	1.2526	0.1575
29	20	1.0221	0.23468	18.0	1.1842	0.1806
30	21	1.0301	0.26356	19.0	1.0992	0.2063
31	23	0.7207	0.23952	20.0	0.9971	0.2347
32	24	0.707	0.2512	21.0	0.8773	0.2756
33	25	0.7391	0.26735	22.0	0.8246	0.3164

After we finish our analysis for infinite AR, then we choose "Finite AR corrected" option. Below is the polar output for AR=8.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.3795	0.11221	-10.0	-0.6448	0.0234
2	-8	-0.4008	0.08834	-9.0	-0.5586	0.0219
3	-6	-0.3759	0.05568	-8.0	-0.4690	0.0209
4	-5	-0.2648	0.04493	-7.0	-0.3765	0.0203
5	-4	-0.1282	0.02314	-6.0	-0.2817	0.0199
6	-3	0.0171	0.01921	-5.0	-0.1851	0.0197
7	-2	0.1513	0.01568	-4.0	-0.0871	0.0197
8	-1	0.2879	0.01289	-3.0	0.0116	0.0197
9	0	0.4375	0.01231	-2.0	0.1102	0.0197
10	1	0.5361	0.01228	-1.0	0.2080	0.0198
11	2	0.6338	0.01247	0.0	0.3042	0.0200
12	3	0.7348	0.01268	1.0	0.3985	0.0204
13	4	0.8307	0.01314	2.0	0.4903	0.0211
14	5	0.9241	0.01418	3.0	0.5792	0.0222
15	6	1.0139	0.01559	4.0	0.6646	0.0238
16	7	1.1052	0.01723	5.0	0.7461	0.0260
17	8	1.1934	0.01901	6.0	0.8232	0.0289
18	9	1.2644	0.02079	7.0	0.8957	0.0325
19	10	1.3127	0.02302	8.0	0.9629	0.0371
20	11	1.3436	0.02733	9.0	1.0246	0.0426
21	12	1.3722	0.03265	10.0	1.0803	0.0492
22	13	1.3738	0.04137	11.0	1.1296	0.0570
23	14	1.3434	0.05515	12.0	1.1722	0.0662
24	15	1.3211	0.07011	13.0	1.2076	0.0767
25	16	1.2975	0.08578	14.0	1.2354	0.0889
26	17	1.2887	0.1007	15.0	1.2554	0.1027
27	18	1.2555	0.12157	16.0	1.2670	0.1182
28	19	1.1851	0.15343	17.0	1.2700	0.1357
29	20	1.0221	0.23468	18.0	1.2640	0.1552
30	21	1.0301	0.26356	19.0	1.2486	0.1768
31	23	0.7207	0.23952	20.0	1.2234	0.2007
32	24	0.707	0.2512	21.0	1.1881	0.2270
33	25	0.7391	0.26735	22.0	1.1424	0.2557

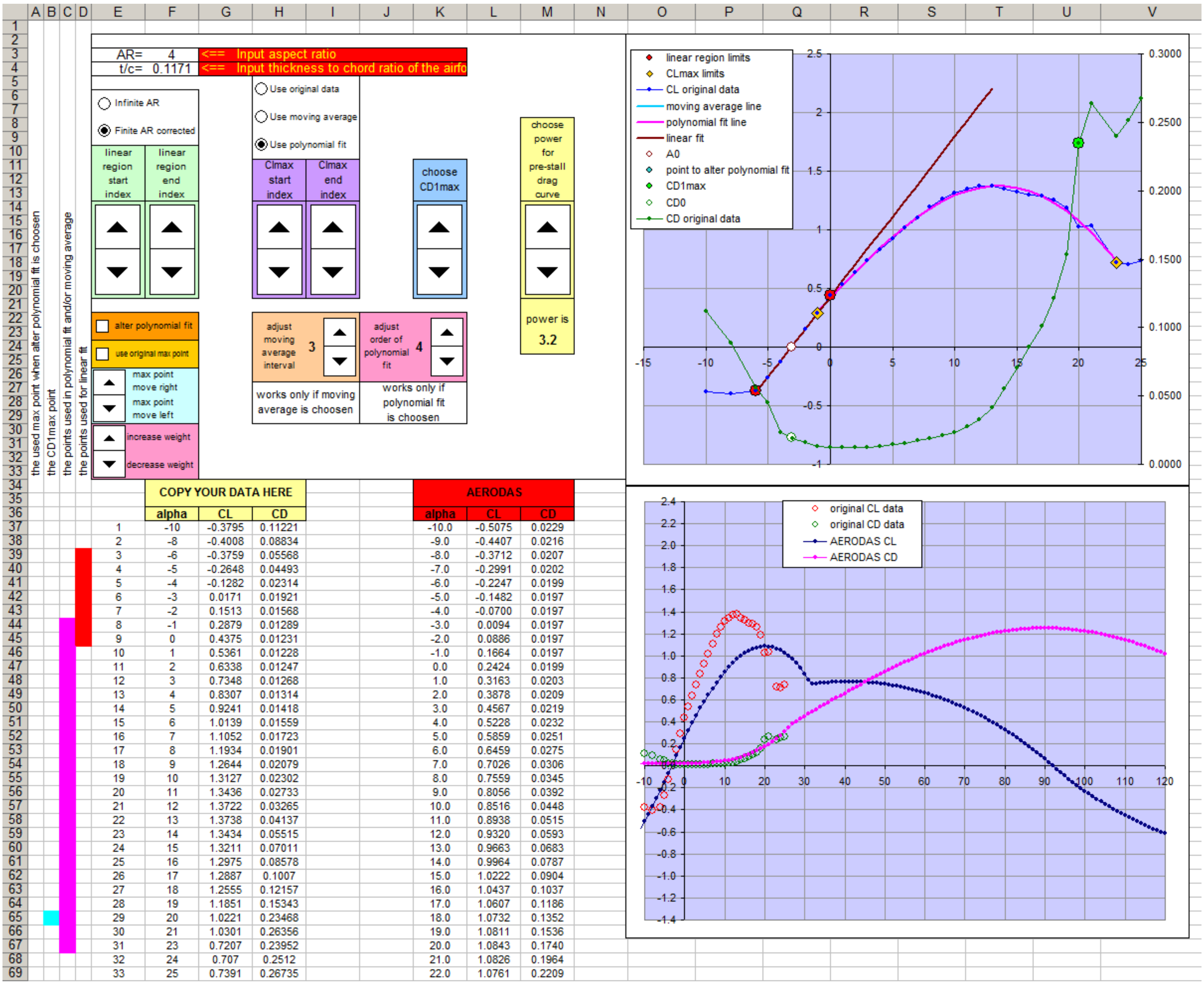
Below is the polar output for AR=6.



	COPY YOUR DATA HERE		
	alpha	CL	CD
1	-10	-0.3795	0.11221
2	-8	-0.4008	0.08834
3	-6	-0.3759	0.05568
4	-5	-0.2648	0.04493
5	-4	-0.1282	0.02314
6	-3	0.0171	0.01921
7	-2	0.1513	0.01568
8	-1	0.2879	0.01289
9	0	0.4375	0.01231
10	1	0.5361	0.01228
11	2	0.6338	0.01247
12	3	0.7348	0.01268
13	4	0.8307	0.01314
14	5	0.9241	0.01418
15	6	1.0139	0.01559
16	7	1.1052	0.01723
17	8	1.1934	0.01901
18	9	1.2644	0.02079
19	10	1.3127	0.02302
20	11	1.3436	0.02733
21	12	1.3722	0.03265
22	13	1.3738	0.04137
23	14	1.3434	0.05515
24	15	1.3211	0.07011
25	16	1.2975	0.08578
26	17	1.2887	0.1007
27	18	1.2555	0.12157
28	19	1.1851	0.15343
29	20	1.0221	0.23468
30	21	1.0301	0.26356
31	23	0.7207	0.23952
32	24	0.707	0.2512
33	25	0.7391	0.26735

AERODAS		
alpha	CL	CD
-10.0	-0.5929	0.0232
-9.0	-0.5139	0.0218
-8.0	-0.4317	0.0209
-7.0	-0.3469	0.0202
-6.0	-0.2598	0.0199
-5.0	-0.1708	0.0197
-4.0	-0.0804	0.0197
-3.0	0.0107	0.0197
-2.0	0.1018	0.0197
-1.0	0.1919	0.0198
0.0	0.2805	0.0200
1.0	0.3671	0.0204
2.0	0.4513	0.0210
3.0	0.5328	0.0221
4.0	0.6111	0.0236
5.0	0.6859	0.0257
6.0	0.7570	0.0284
7.0	0.8239	0.0319
8.0	0.8864	0.0361
9.0	0.9442	0.0414
10.0	0.9970	0.0476
11.0	1.0446	0.0551
12.0	1.0866	0.0637
13.0	1.1228	0.0737
14.0	1.1530	0.0852
15.0	1.1768	0.0983
16.0	1.1940	0.1130
17.0	1.2045	0.1296
18.0	1.2079	0.1480
19.0	1.2040	0.1685
20.0	1.1926	0.1911
21.0	1.1734	0.2160
22.0	1.1463	0.2433

Below is the polar output for AR=4.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10	-0.3795	0.11221	-10.0	-0.5075	0.0229
2	-8	-0.4008	0.08834	-9.0	-0.4407	0.0216
3	-6	-0.3759	0.05568	-8.0	-0.3712	0.0207
4	-5	-0.2648	0.04493	-7.0	-0.2991	0.0202
5	-4	-0.1282	0.02314	-6.0	-0.2247	0.0199
6	-3	0.0171	0.01921	-5.0	-0.1482	0.0197
7	-2	0.1513	0.01568	-4.0	-0.0700	0.0197
8	-1	0.2879	0.01289	-3.0	0.0094	0.0197
9	0	0.4375	0.01231	-2.0	0.0886	0.0197
10	1	0.5361	0.01228	-1.0	0.1664	0.0197
11	2	0.6338	0.01247	0.0	0.2424	0.0199
12	3	0.7348	0.01268	1.0	0.3163	0.0203
13	4	0.8307	0.01314	2.0	0.3878	0.0209
14	5	0.9241	0.01418	3.0	0.4567	0.0219
15	6	1.0139	0.01559	4.0	0.5228	0.0232
16	7	1.1052	0.01723	5.0	0.5859	0.0251
17	8	1.1934	0.01901	6.0	0.6459	0.0275
18	9	1.2644	0.02079	7.0	0.7026	0.0306
19	10	1.3127	0.02302	8.0	0.7559	0.0345
20	11	1.3436	0.02733	9.0	0.8056	0.0392
21	12	1.3722	0.03265	10.0	0.8516	0.0448
22	13	1.3738	0.04137	11.0	0.8938	0.0515
23	14	1.3434	0.05515	12.0	0.9320	0.0593
24	15	1.3211	0.07011	13.0	0.9663	0.0683
25	16	1.2975	0.08578	14.0	0.9964	0.0787
26	17	1.2887	0.1007	15.0	1.0222	0.0904
27	18	1.2555	0.12157	16.0	1.0437	0.1037
28	19	1.1851	0.15343	17.0	1.0607	0.1186
29	20	1.0221	0.23468	18.0	1.0732	0.1352
30	21	1.0301	0.26356	19.0	1.0811	0.1536
31	23	0.7207	0.23952	20.0	1.0843	0.1740
32	24	0.707	0.2512	21.0	1.0826	0.1964
33	25	0.7391	0.26735	22.0	1.0761	0.2209

These calculated C_L and C_D values by AERODAS model in the above figures are comparable with the data given by Knight and Werzinger [5].

NACA 4418

As an other example, the following figure is taken from Timmer [6]. This figure shows the polar data of NACA 4418 for Re=250000.

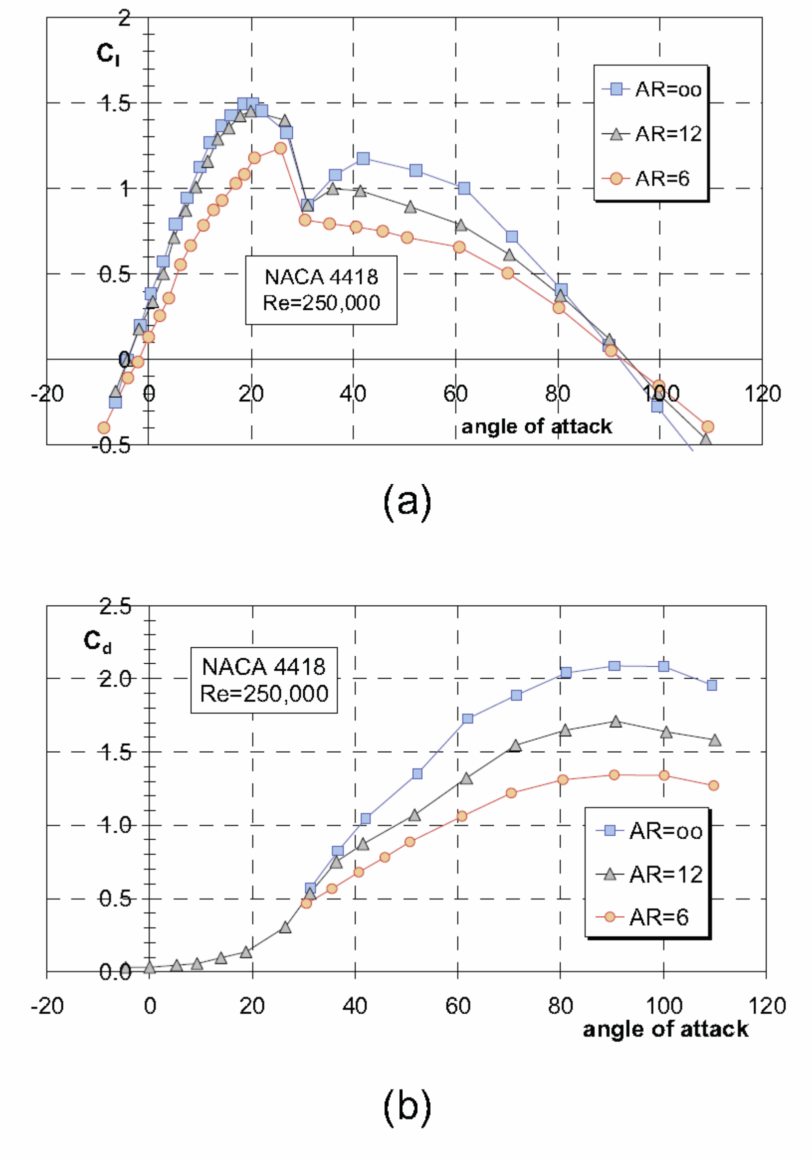
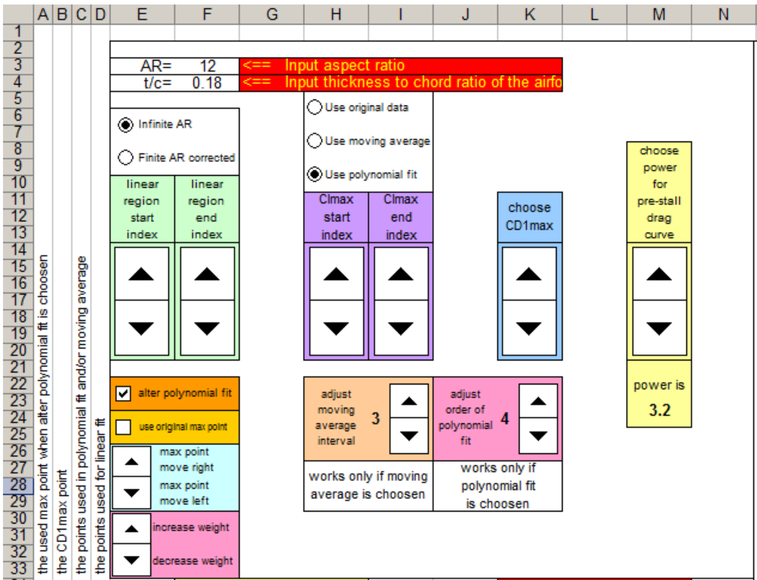


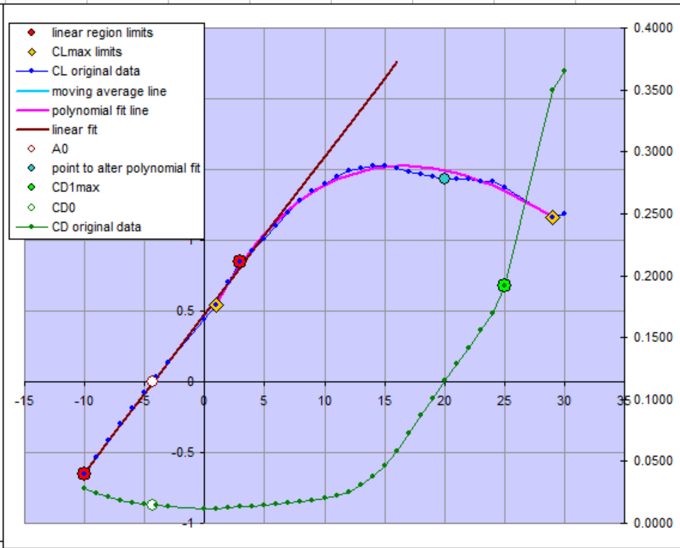
Figure 6: The lift (a) and drag (b) data of airfoil NACA 4418 [8] for various aspect ratios.

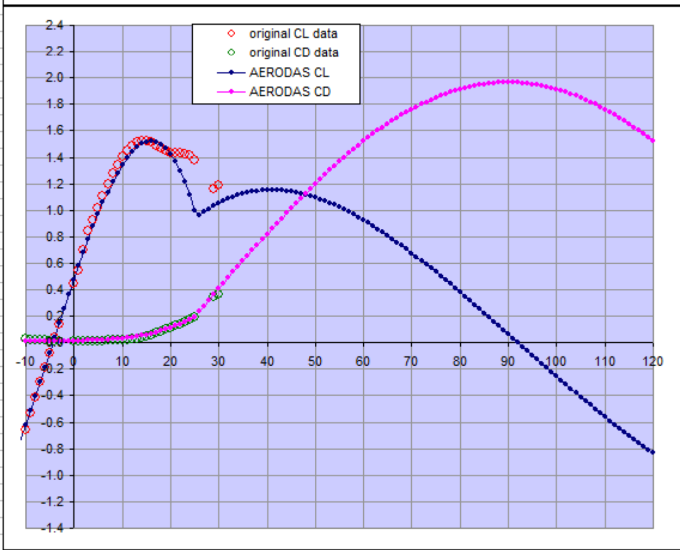
The following is the XFOIL output polar file for NACA 4418 airfoil for Re=250000.

```
XFLR5 v6.09.01 beta

 Calculated polar for: NACA 4418

 1 1 Reynolds number fixed          Mach number fixed

 xtrf =   1.000 (top)        1.000 (bottom)
 Mach =   0.000     Re =     0.250 e 6     Ncrit =   9.000

  alpha     CL        CD       CDp       Cm    Top Xtr Bot Xtr   Cpmin    Chinge     XCp
 ------- -------- --------- --------- -------- ------- ------- -------- --------- -------
--
 -10.000  -0.6563   0.02844   0.02199  -0.0986  0.9479  0.0697  -4.0826   0.0000   0.0793
  -9.000  -0.5349   0.02423   0.01739  -0.1032  0.9300  0.0790  -3.5801   0.0000   0.0358
  -8.000  -0.4117   0.02102   0.01389  -0.1064  0.9106  0.0911  -3.0861   0.0000  -0.0316
  -7.000  -0.2969   0.01866   0.01135  -0.1071  0.8884  0.1062  -2.6156   0.0000  -0.1375
  -6.000  -0.1858   0.01678   0.00943  -0.1067  0.8618  0.1272  -2.1905   0.0000  -0.3592
  -5.000  -0.0750   0.01538   0.00788  -0.1059  0.8307  0.1552  -1.8160   0.0000  -1.2299
  -4.000   0.0344   0.01418   0.00666  -0.1049  0.7960  0.1939  -1.4648   0.0000   3.4086
  -3.000   0.1390   0.01337   0.00585  -0.1030  0.7546  0.2451  -1.1738   0.0000   1.0104
   0.000   0.4446   0.01157   0.00517  -0.0959  0.6336  0.6894  -0.9756   0.0000   0.4633
   1.000   0.5437   0.01172   0.00564  -0.0913  0.6004  0.8895  -1.0676   0.0000   0.4134
   2.000   0.7026   0.01256   0.00620  -0.1001  0.5695  0.9803  -1.2053   0.0000   0.3864
   3.000   0.8481   0.01315   0.00656  -0.1080  0.5421  1.0000  -1.3468   0.0000   0.3699
   4.000   0.9266   0.01379   0.00709  -0.1021  0.5190  1.0000  -1.4536   0.0000   0.3513
   5.000   1.0120   0.01463   0.00780  -0.0972  0.4961  1.0000  -1.5786   0.0000   0.3358
   6.000   1.1024   0.01557   0.00866  -0.0934  0.4730  1.0000  -1.7270   0.0000   0.3231
   7.000   1.1937   0.01659   0.00957  -0.0899  0.4487  1.0000  -1.9062   0.0000   0.3123
   8.000   1.2782   0.01762   0.01059  -0.0854  0.4230  1.0000  -2.1228   0.0000   0.3022
   9.000   1.3437   0.01864   0.01168  -0.0775  0.3972  1.0000  -2.3610   0.0000   0.2914
  10.000   1.4014   0.02014   0.01320  -0.0690  0.3684  1.0000  -2.6236   0.0000   0.2813
  11.000   1.4487   0.02234   0.01557  -0.0603  0.3381  1.0000  -2.8999   0.0000   0.2719
  12.000   1.4878   0.02561   0.01897  -0.0522  0.3048  1.0000  -3.1853   0.0000   0.2634
  13.000   1.5125   0.03051   0.02390  -0.0445  0.2697  1.0000  -3.4582   0.0000   0.2556
  14.000   1.5260   0.03730   0.03081  -0.0381  0.2344  1.0000  -3.7055   0.0000   0.2490
  15.000   1.5215   0.04684   0.04036  -0.0333  0.2003  1.0000  -3.9304   0.0000   0.2435
  16.000   1.5101   0.05847   0.05215  -0.0307  0.1697  1.0000  -4.1337   0.0000   0.2395
  17.000   1.4871   0.07233   0.06611  -0.0300  0.1428  1.0000  -4.2934   0.0000   0.2367
  18.000   1.4643   0.08690   0.08074  -0.0307  0.1213  1.0000  -4.4397   0.0000   0.2350
  19.000   1.4496   0.10073   0.09469  -0.0324  0.1039  1.0000  -4.5988   0.0000   0.2338
  20.000   1.4364   0.11513   0.10932  -0.0359  0.0905  1.0000  -4.7639   0.0000   0.2338
  21.000   1.4289   0.12874   0.12322  -0.0401  0.0795  1.0000  -4.9589   0.0000   0.2342
  22.000   1.4281   0.14106   0.13569  -0.0446  0.0704  1.0000  -5.1692   0.0000   0.2345
  23.000   1.4192   0.15579   0.15074  -0.0518  0.0636  1.0000  -5.3247   0.0000   0.2369
  24.000   1.4121   0.16989   0.16514  -0.0594  0.0575  1.0000  -5.4587   0.0000   0.2395
  25.000   1.3751   0.19145   0.18721  -0.0732  0.0524  1.0000  -5.3885   0.0000   0.2477
  29.000   1.1592   0.34958   0.34623  -0.1642  0.0385  1.0000  -2.1992   0.0000   0.3301
  30.000   1.1879   0.36512   0.36182  -0.1689  0.0363  1.0000  -2.4044   0.0000   0.3270
```

The following is the polar data obtained by the AERODAS model for NACA 4418 airfoil for infinite aspect ratio.

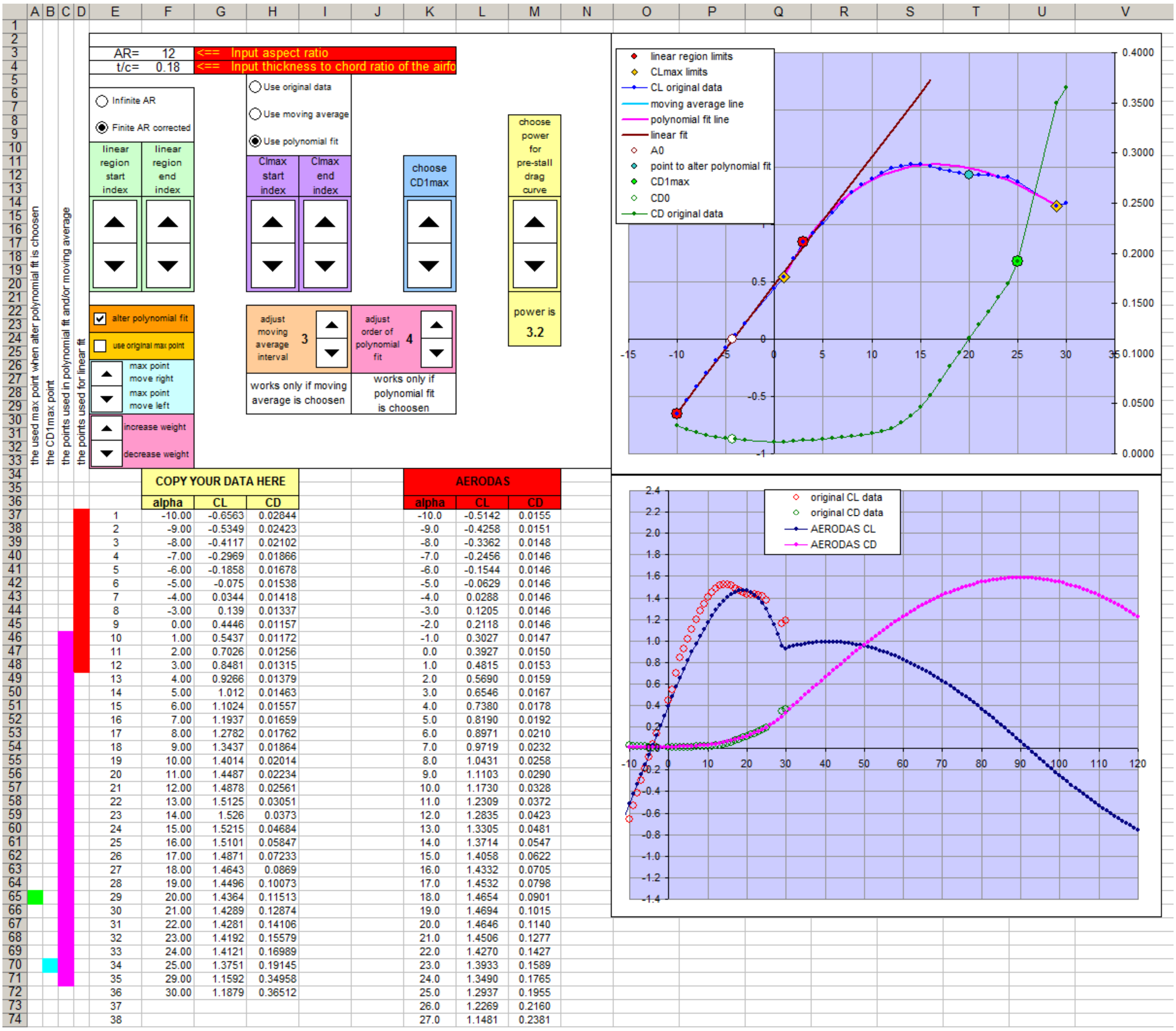






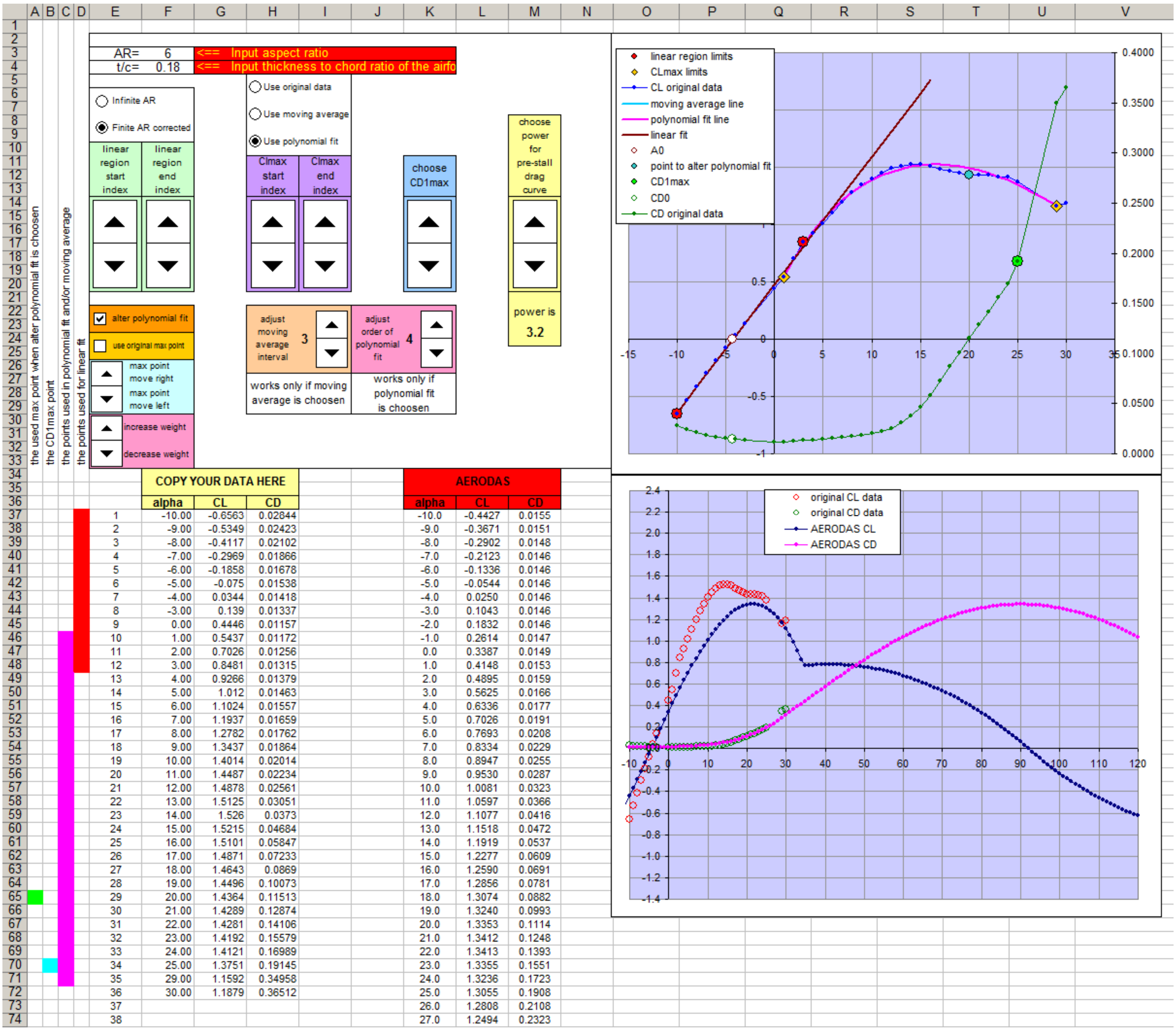
	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10.00	-0.6563	0.02844	-10.0	-0.6191	0.0155
2	-9.00	-0.5349	0.02423	-9.0	-0.5144	0.0151
3	-8.00	-0.4117	0.02102	-8.0	-0.4072	0.0148
4	-7.00	-0.2969	0.01866	-7.0	-0.2981	0.0146
5	-6.00	-0.1858	0.01678	-6.0	-0.1877	0.0146
6	-5.00	-0.075	0.01538	-5.0	-0.0765	0.0146
7	-4.00	0.0344	0.01418	-4.0	0.0351	0.0146
8	-3.00	0.139	0.01337	-3.0	0.1465	0.0146
9	0.00	0.4446	0.01157	-2.0	0.2573	0.0146
10	1.00	0.5437	0.01172	-1.0	0.3669	0.0147
11	2.00	0.7026	0.01256	0.0	0.4748	0.0149
12	3.00	0.8481	0.01315	1.0	0.5805	0.0153
13	4.00	0.9266	0.01379	2.0	0.6835	0.0159
14	5.00	1.012	0.01463	3.0	0.7831	0.0166
15	6.00	1.1024	0.01557	4.0	0.8788	0.0177
16	7.00	1.1937	0.01659	5.0	0.9701	0.0191
17	8.00	1.2782	0.01762	6.0	1.0564	0.0208
18	9.00	1.3437	0.01864	7.0	1.1372	0.0230
19	10.00	1.4014	0.02014	8.0	1.2120	0.0256
20	11.00	1.4487	0.02234	9.0	1.2800	0.0287
21	12.00	1.4878	0.02561	10.0	1.3408	0.0324
22	13.00	1.5125	0.03051	11.0	1.3939	0.0367
23	14.00	1.526	0.0373	12.0	1.4385	0.0417
24	15.00	1.5215	0.04684	13.0	1.4743	0.0474
25	16.00	1.5101	0.05847	14.0	1.5005	0.0538
26	17.00	1.4871	0.07233	15.0	1.5167	0.0611
27	18.00	1.4643	0.0869	16.0	1.5222	0.0693
28	19.00	1.4496	0.10073	17.0	1.5165	0.0784
29	20.00	1.4364	0.11513	18.0	1.4990	0.0884
30	21.00	1.4289	0.12874	19.0	1.4691	0.0996
31	22.00	1.4281	0.14106	20.0	1.4262	0.1118
32	23.00	1.4192	0.15579	21.0	1.3698	0.1252
33	24.00	1.4121	0.16989	22.0	1.2993	0.1398
34	25.00	1.3751	0.19145	23.0	1.2141	0.1556
35	29.00	1.1592	0.34958	24.0	1.1136	0.1729
36	30.00	1.1879	0.36512	25.0	0.9972	0.1915
37				26.0	0.9585	0.2343
38				27.0	0.9848	0.2771

Below is the polar output for AR=12.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10.00	-0.6563	0.02844	-10.0	-0.5142	0.0155
2	-9.00	-0.5349	0.02423	-9.0	-0.4258	0.0151
3	-8.00	-0.4117	0.02102	-8.0	-0.3362	0.0148
4	-7.00	-0.2969	0.01866	-7.0	-0.2456	0.0146
5	-6.00	-0.1858	0.01678	-6.0	-0.1544	0.0146
6	-5.00	-0.075	0.01538	-5.0	-0.0629	0.0146
7	-4.00	0.0344	0.01418	-4.0	0.0288	0.0146
8	-3.00	0.139	0.01337	-3.0	0.1205	0.0146
9	0.00	0.4446	0.01157	-2.0	0.2118	0.0146
10	1.00	0.5437	0.01172	-1.0	0.3027	0.0147
11	2.00	0.7026	0.01256	0.0	0.3927	0.0150
12	3.00	0.8481	0.01315	1.0	0.4815	0.0153
13	4.00	0.9266	0.01379	2.0	0.5690	0.0159
14	5.00	1.012	0.01463	3.0	0.6546	0.0167
15	6.00	1.1024	0.01557	4.0	0.7380	0.0178
16	7.00	1.1937	0.01659	5.0	0.8190	0.0192
17	8.00	1.2782	0.01762	6.0	0.8971	0.0210
18	9.00	1.3437	0.01864	7.0	0.9719	0.0232
19	10.00	1.4014	0.02014	8.0	1.0431	0.0258
20	11.00	1.4487	0.02234	9.0	1.1103	0.0290
21	12.00	1.4878	0.02561	10.0	1.1730	0.0328
22	13.00	1.5125	0.03051	11.0	1.2309	0.0372
23	14.00	1.526	0.0373	12.0	1.2835	0.0423
24	15.00	1.5215	0.04684	13.0	1.3305	0.0481
25	16.00	1.5101	0.05847	14.0	1.3714	0.0547
26	17.00	1.4871	0.07233	15.0	1.4058	0.0622
27	18.00	1.4643	0.0869	16.0	1.4332	0.0705
28	19.00	1.4496	0.10073	17.0	1.4532	0.0798
29	20.00	1.4364	0.11513	18.0	1.4654	0.0901
30	21.00	1.4289	0.12874	19.0	1.4694	0.1015
31	22.00	1.4281	0.14106	20.0	1.4646	0.1140
32	23.00	1.4192	0.15579	21.0	1.4506	0.1277
33	24.00	1.4121	0.16989	22.0	1.4270	0.1427
34	25.00	1.3751	0.19145	23.0	1.3933	0.1589
35	29.00	1.1592	0.34958	24.0	1.3490	0.1765
36	30.00	1.1879	0.36512	25.0	1.2937	0.1955
37				26.0	1.2269	0.2160
38				27.0	1.1481	0.2381

Below is the polar output for AR=6.



	COPY YOUR DATA HERE			AERODAS		
	alpha	CL	CD	alpha	CL	CD
1	-10.00	-0.6563	0.02844	-10.0	-0.4427	0.0155
2	-9.00	-0.5349	0.02423	-9.0	-0.3671	0.0151
3	-8.00	-0.4117	0.02102	-8.0	-0.2902	0.0148
4	-7.00	-0.2969	0.01866	-7.0	-0.2123	0.0146
5	-6.00	-0.1858	0.01678	-6.0	-0.1336	0.0146
6	-5.00	-0.075	0.01538	-5.0	-0.0544	0.0146
7	-4.00	0.0344	0.01418	-4.0	0.0250	0.0146
8	-3.00	0.139	0.01337	-3.0	0.1043	0.0146
9	0.00	0.4446	0.01157	-2.0	0.1832	0.0146
10	1.00	0.5437	0.01172	-1.0	0.2614	0.0147
11	2.00	0.7026	0.01256	0.0	0.3387	0.0149
12	3.00	0.8481	0.01315	1.0	0.4148	0.0153
13	4.00	0.9266	0.01379	2.0	0.4895	0.0159
14	5.00	1.012	0.01463	3.0	0.5625	0.0166
15	6.00	1.1024	0.01557	4.0	0.6336	0.0177
16	7.00	1.1937	0.01659	5.0	0.7026	0.0191
17	8.00	1.2782	0.01762	6.0	0.7693	0.0208
18	9.00	1.3437	0.01864	7.0	0.8334	0.0229
19	10.00	1.4014	0.02014	8.0	0.8947	0.0255
20	11.00	1.4487	0.02234	9.0	0.9530	0.0287
21	12.00	1.4878	0.02561	10.0	1.0081	0.0323
22	13.00	1.5125	0.03051	11.0	1.0597	0.0366
23	14.00	1.526	0.0373	12.0	1.1077	0.0416
24	15.00	1.5215	0.04684	13.0	1.1518	0.0472
25	16.00	1.5101	0.05847	14.0	1.1919	0.0537
26	17.00	1.4871	0.07233	15.0	1.2277	0.0609
27	18.00	1.4643	0.0869	16.0	1.2590	0.0691
28	19.00	1.4496	0.10073	17.0	1.2856	0.0781
29	20.00	1.4364	0.11513	18.0	1.3074	0.0882
30	21.00	1.4289	0.12874	19.0	1.3240	0.0993
31	22.00	1.4281	0.14106	20.0	1.3353	0.1114
32	23.00	1.4192	0.15579	21.0	1.3412	0.1248
33	24.00	1.4121	0.16989	22.0	1.3413	0.1393
34	25.00	1.3751	0.19145	23.0	1.3355	0.1551
35	29.00	1.1592	0.34958	24.0	1.3236	0.1723
36	30.00	1.1879	0.36512	25.0	1.3055	0.1908
37				26.0	1.2808	0.2108
38				27.0	1.2494	0.2323

These calculated C_L and C_D values by AERODAS model in the above figures are comparable with the data given by Timmer [6].

AERODAS model provides an easy way of estimation for C_L and C_D values over a wide range of angle of attack within small acceptable errors. With AERODAS.xls document, application of this model becomes very quick and very easy.

References:

[1] Viterna LA, Janetzke D.C. (1982), "Theoretical and Experimental Power from Large Horizontal-Axis Wind Turbines," NASA Lewis Research Center Technical Report, NASA TM-82944
http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19820025954.pdf

[2] Montgomerie B, (2004), "Methods for Root Effects, Tip Effects and Extending the Angle of Attack to ±180 with Application to Aerodynamics for Blades on Wind Turbines and Propellers", Swedish Defence Research Agency Scientific Report, FOI-R--1305--SE
http://www.foi.se/report?rNo=FOI-R--1305--SE

[3] Lindenburg C, (2003), "Investigation into Rotor Blade Aerodynamics", Energy research Centre of the Netherlands (ECN) Wind Energy publication, ECN-C--03-025
ftp://ftp.ecn.nl/pub/www/library/report/2003/c03025.pdf

[4] Spera DA, (2008), "Models of Lift and Drag Coeffi cients of Stalled and Unstalled Airfoils in Wind Turbines and Wind Tunnels", NASA Technical Report NASA/CR-2008-215434
http://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20090001311.pdf

[5] Knight M, Wenzinger CJ, (1929), "Wind Tunnel Tests on a Series of Wing Models Through a Large Angle of Attack Range, Part 1, Force Tests", NACA Report No 317
http://naca.central.cranfield.ac.uk/reports/1930/naca-report-317.pdf

[6] Timmer WA, (2010), "Aerodynamic characteristics of wind turbine blade airfoils at high angles-of-attack", Conference paper, TORQUE 2010: The Science of Making Torque from Wind, June 28-30, Crete, Greece
http://www.lr.tudelft.nl/fileadmin/Faculteit/LR/Organisatie/Afdelingen_en_Leerstoelen/Afdeling_AEWE/Wind_Energy/Research/Publications/Publications_2010/doc/Timmer.pdf