Wind Tunnel Testing of full-scale Yardstik aircraft : Part 1: Pitch axis

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1 List of Symbols

N :Normal force

A :Axial force

M :Pitching moment

α :Angle of attack of aircraft

L :Lift force

D :Drag force

C_L :Lift coefficient

ρ :Density of air

V :Air speed

S :Lifting surface of aircraft

b :Wing span

c :Wing chord length

C_D :Drag Coefficient

$M_{1/4}$:Moment at wing quarter chord point

$C_{M_{1/4}}$:Moment coefficient at wing quarter chord point

P_0 :Static pressure

T :Temperature

i :Incidence angle of the aircraft

C_L :Lift coefficient of the aircraft

$C_{L_{\alpha_{\delta e}}}$:Coefficient for change of lift due to elevator deflection

$C_{M_{cg}}$:Coefficient for change of moment about cg

C_{M_0} :Coefficient for moment at zero AOA

C_{M_α} :Coefficient for change of moment w.r.t AOA

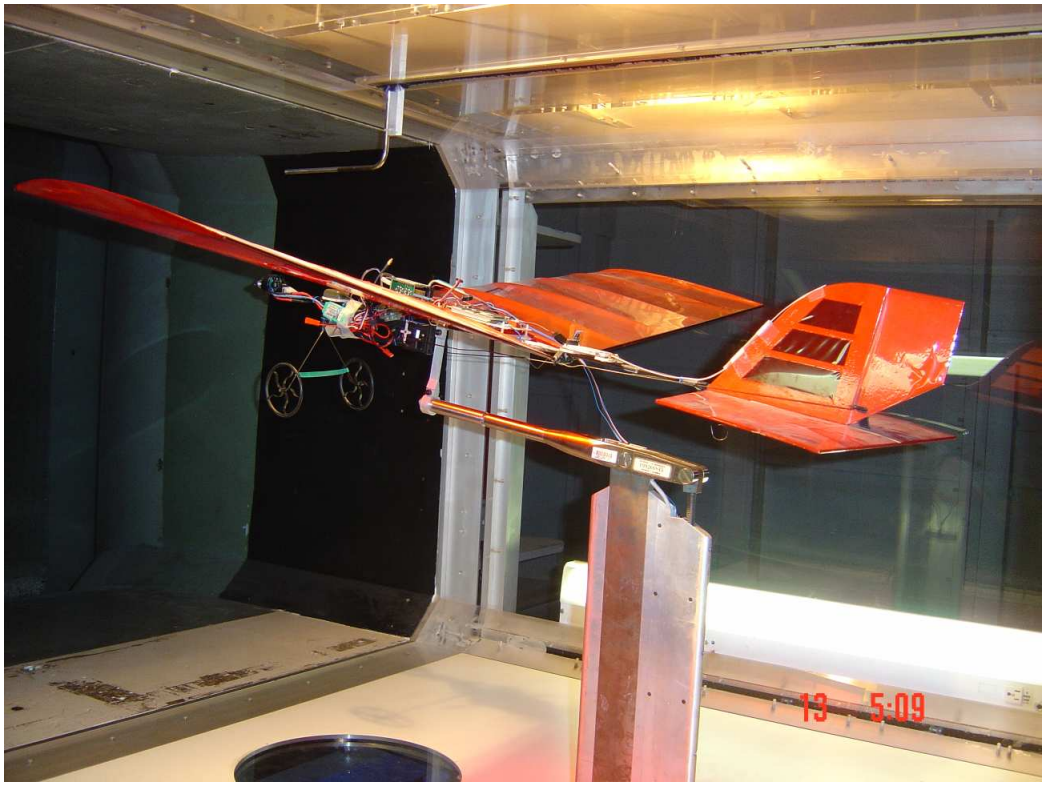
C_{L_α} :Coefficient for change of lift w.r.t AOA

X_{cg} :Center of gravity of airplane

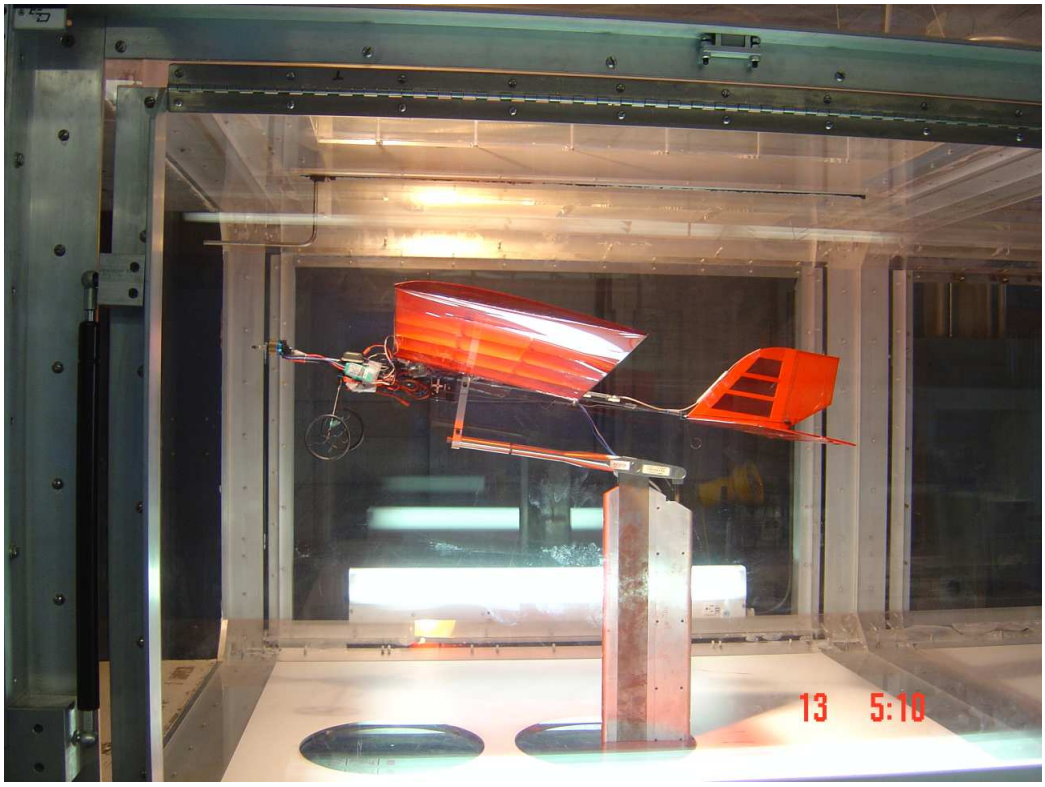
2 Wind tunnel setup

The wind tunnel setup for testing the full-scale Yardstik aircraft is shown in Figure 1. Using this setup, we determine the aerodynamics coefficients through analysis of the pitch angle of the aircraft. The force moment balance is used to measure the forces and moments generated by the aircraft. All measurements were made with reference to the measurement point on the plane and a calibration process was done to determine this reference location from the force moment balance measurement point. The forces and moments that are measured by the balance are as follows:

1. Normal, Transverse and Axial forces
2. Normal, Transverse and Axial moments



(a)



(b)

Figure 1: Yardstik aircraft mounted on sting

3 Lift, drag and moment calculation

3.1 Lift and drag forces

From the measured normal and axial forces, the lift and drag forces can be calculated by resolving these 2 forces along the body axis at various pitch angle α that the aircraft holds. Figure 2 shows the relationship between the lift and drag forces with the measured normal and axial forces.

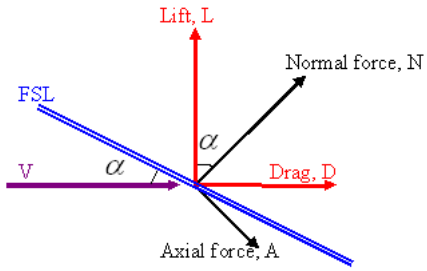


Figure 2: Relationship between the lift and drag forces with the measured normal and axial forces

The lift force is given by:

$$L = -Asin\alpha + Ncos\alpha \tag{1}$$

With the lift calculated, the lift coefficient is given by:

$$C_L = \frac{L}{\frac{1}{2}\rho V^2 S} \tag{2}$$

The drag force is given by:

$$D = Acos\alpha + Nsin\alpha \tag{3}$$

With the drag calculated, the drag coefficient is given by:

$$C_D = \frac{D}{\frac{1}{2}\rho V^2 S} \tag{4}$$

3.2 Pitching moment calculation

To calculate the moment at the quarter chord point of the wing ($M_{1/4}$), we need to determine the moment arms from the measurement point to the quarter chord point of the wing. A calibration test using known load was done to determine this moment arm. From the calibration, the moment about the quarter chord is given by:

$$M_{1/4} = \frac{M}{12} - 0.256561A - 0.25656133N \tag{5}$$

With $M_{1/4}$ calculated, the moment coefficient at quarter chord is given by:

$$C_{M_{1/4}} = \frac{M_{1/4}}{\frac{1}{2}\rho V^2 Sc} \tag{6}$$

4 Experimental data and result

4.1 Lift characteristics

Figure 3 shows the plot of C_L vs α for α between -8 deg to 16 deg. From the graph, we can deduce the following data:

1. $C_{L_{max}} = 1.1$ @ $\alpha = 13$ degree.
2. $C_{L_0} = 0.31$

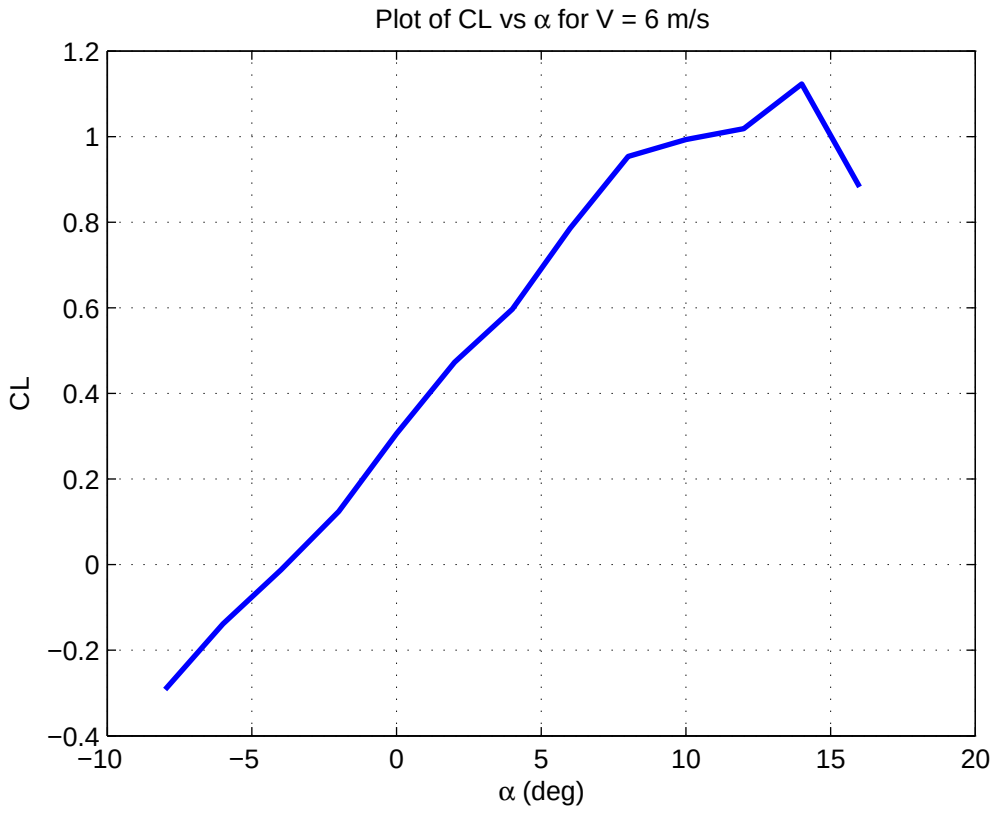


Figure 3: Plot of C_L vs α for V = 6 m/s

Figure 4 shows the plot of C_L vs α for α between -8 deg to 8 deg where the data has a linear relationship. We did a linear fitting to the graph to obtain the C_{L_α}, which is 0.0777 per degree.

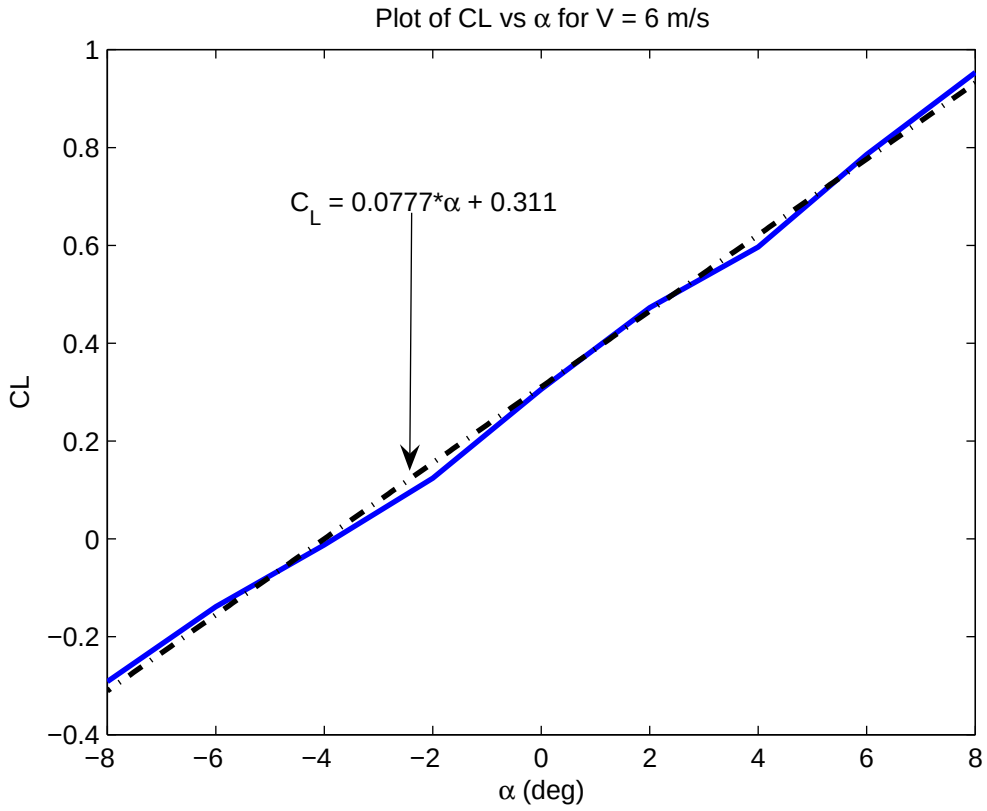


Figure 4: Plot of C_L vs α for V = 6 m/s

4.2 Drag characteristics

Figure 5 shows the plot of C_D vs α for α between -8 deg to 16 deg. From the graph, we can deduce the following data:

1. $C_{D_{min}}$ = 0.0392 @ α = -2.5 degree.
2. C_L @ $C_{D_{min}}$= 0.11675.

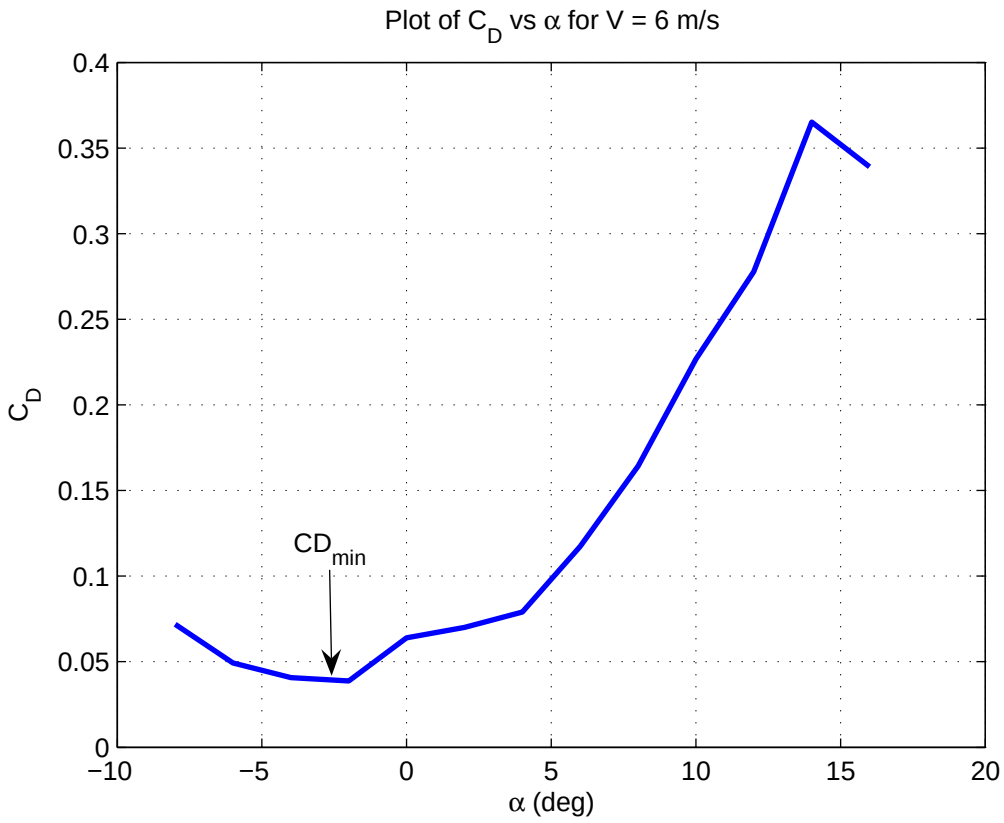


Figure 5: Plot of C_D vs α for V = 6 m/s

4.3 Pitch moment characteristics

Figure 6 shows the plot of C_M vs α for α between -8 deg to 8 deg. From the graph, we can deduce the following data:

1. C_{M_0} = -0.054
2. C_{M_α} = -0.012 per degree

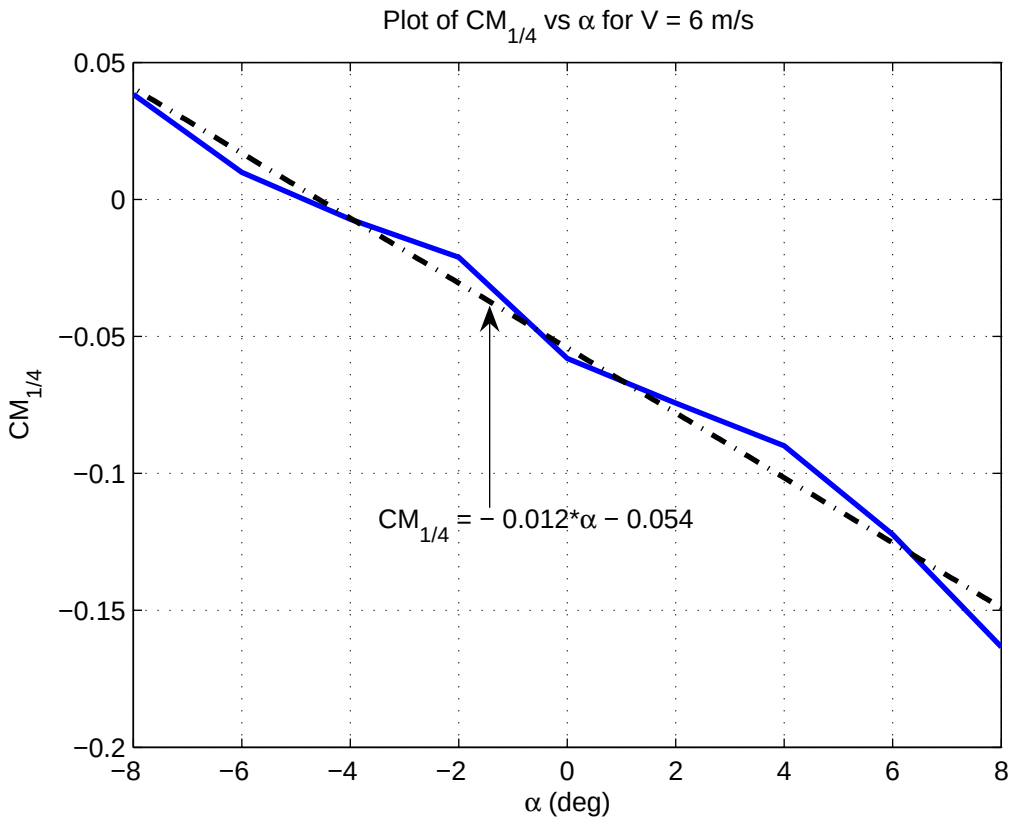


Figure 6: Plot of C_M vs α for V = 6 m/s

5 Elevator control characteristics

5.1 Lift

Figure 7 shows the plot of C_L vs α for α between -8 deg to 8 deg for various elevator deflections. From the graph, we can deduce that $CL_{\delta e}$ is 0.005 per degree.

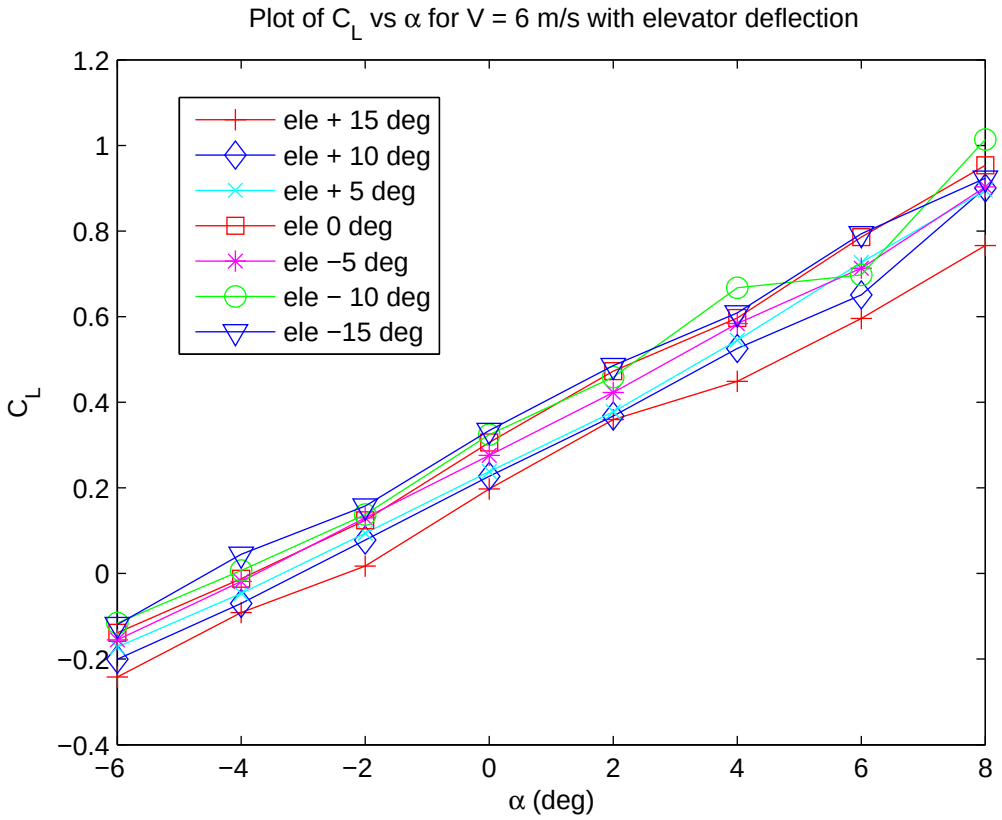


Figure 7: Plot of C_L vs α for V = 6 m/s with various elevator deflection

5.2 Drag

Figure 8 shows the plot of C_D vs α for α between -8 deg to 8 deg for various elevator deflections. From the graph, we can deduce that $CD_{\delta e}$ is 0.000667 per degree.

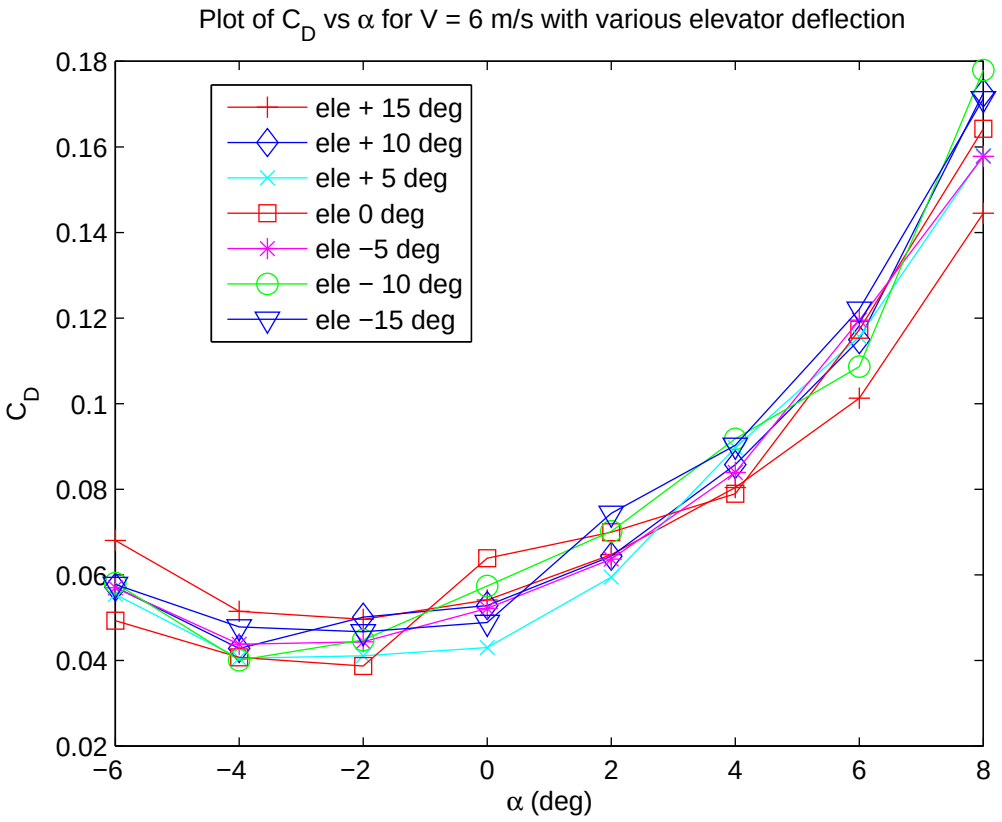


Figure 8: Plot of C_D vs α for V = 6 m/s with various elevator deflection

5.3 Pitch moment

Figure 9 shows the plot of C_M vs α for α between -8 deg to 8 deg for various elevator deflections. From the graph, we can deduce that $C_{M_{\delta e}}$ is -0.008 per degree.

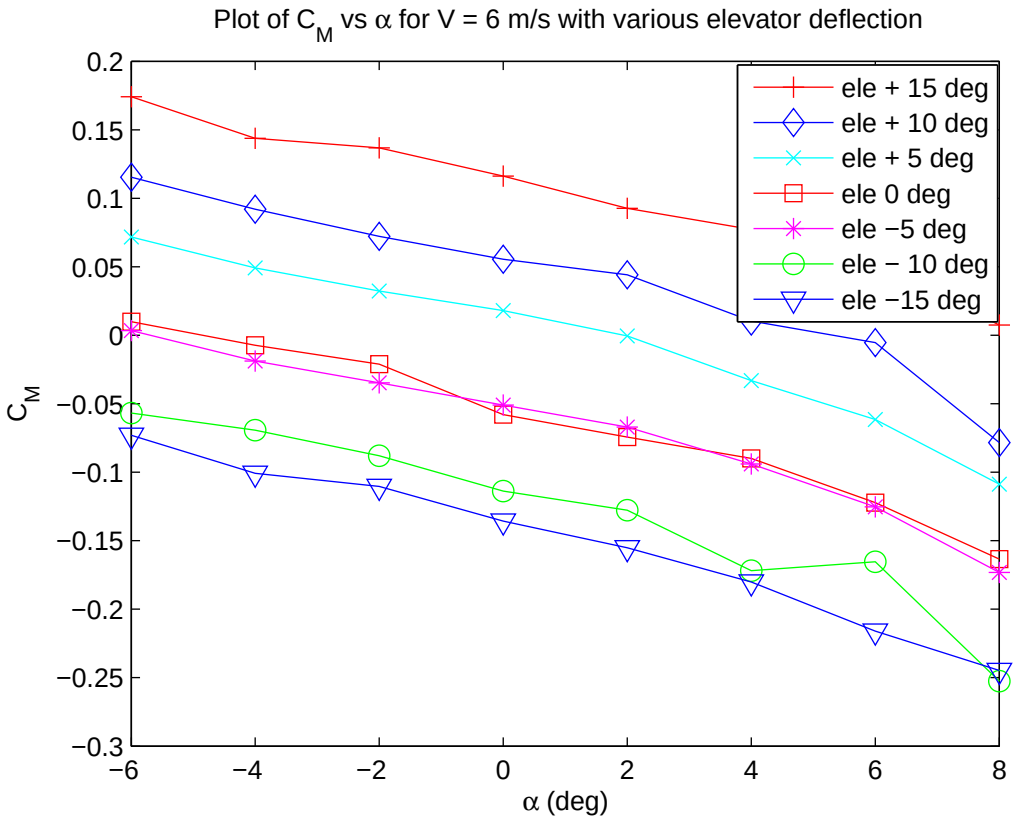


Figure 9: Plot of C_M vs α for V = 6 m/s with various elevator deflection

Date: March 13 / 14, 2007

All force measurement are in lbf and all moments measurement are in in.lb

Run 1: Determine CL max, 6 m/s, 0 Elevator, 0 Rud

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD
-8	0	0	0	0	0	0	-0.458	0.007	0.047	-0.163	-0.559	-0.393	-0.458	0.047	-0.559	-0.44700945	0.11025224	0.058863389	0.038389942	-0.291534	0.071905
-6	0	0	0	0	0	0	-0.219	-0.018	0.053	-0.11	-0.329	-0.421	-0.219	0.053	-0.329	-0.2122643	0.07559013	0.015172532	0.009895329	-0.138436	0.049299
-4	0	0	0	0	0	0	-0.023	0.019	0.061	-0.064	-0.017	-0.387	-0.023	0.061	-0.017	-0.01869104	0.06245514	-0.01116598	-0.007282306	-0.01219	0.040732
-2	0	0	0	0	0	0	0.188	0.021	0.066	-0.039	0.393	-0.438	0.188	0.066	0.393	0.190187791	0.05940205	-0.03241656	-0.021141659	0.124038	0.038741
0	-0.001	0.002	-0.002	0.004	0	-0.006	0.468	0.014	0.096	0.008	0.678	-0.611	0.469	0.098	0.678	0.469	0.098	-0.08897024	-0.058025242	0.305876	0.063914
2	0	0	0	0	0	0	0.728	0	0.082	0.01	1.126	-0.639	0.728	0.082	1.126	0.724696663	0.10734406	-0.11398132	-0.074337142	0.472638	0.070008
4	0	0	0	0	0	0	0.921	0.024	0.057	0.007	1.357	-0.727	0.921	0.057	1.357	0.914784657	0.12107449	-0.13783363	-0.089893312	0.596611	0.078963
6	0	0	0	0	0	0	1.217	0.04	0.053	0.027	1.66	-0.752	1.217	0.053	1.66	1.204802688	0.17985684	-0.18749954	-0.122284776	0.785757	0.1173
8	0	0	0	0	0	0	1.483	0.086	0.046	0.139	1.701	-0.992	1.483	0.046	1.701	1.462183414	0.25184254	-0.25053226	-0.163393902	0.953617	0.164248
10	0	0	0	0	0	0	1.56	-0.101	0.078	0.286	1.26	-0.795	1.56	0.078	1.26	1.522786296	0.34757143	-0.31524743	-0.205600303	0.993142	0.226682
12	0.019	-0.008	0.065	-0.014	0.234	-0.018	1.635	0.093	0.157	0.338	0.956	-0.599	1.616	0.092	0.722	1.561603867	0.42580907	-0.37804005	-0.246552839	1.018458	0.277707
14	0.029	-0.006	0.122	-0.023	0.486	-0.022	1.835	0.124	0.249	0.226	-0.276	0.511	1.806	0.127	-0.762	1.721699374	0.55992523	-0.55943301	-0.364854981	1.12287	0.365176
16	0.032	-0.008	0.172	-0.026	0.735	-0.015	1.476	0.189	0.299	0.455	-1.098	-0.438	1.444	0.127	-1.833	1.353129561	0.51990902	-0.55580781	-0.362490671	0.882494	0.339078
14	0	0	0	0	0	0	1.673	-0.141	0.25	0.419	-0.551	0.339	1.673	0.25	-0.551	1.562904448	0.64711567	-0.53928402	-0.351714072	1.019306	0.42204
12	0	0	0	0	0	0	1.759	0.223	0.176	0.377	0.787	-0.61	1.759	0.176	0.787	1.684026272	0.53769184	-0.43086278	-0.28100314	1.0983	0.350676
10	0	0	0	0	0	0	1.548	0.192	0.107	0.145	1.436	-0.81	1.548	0.107	1.436	1.505935149	0.37404856	-0.3049423	-0.198879428	0.982151	0.24395

Run 2: Determine ele derivative, 6 m/s, +15 Elevator, PPM = 42260

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD
-6							-0.38	-0.01	0.065	-0.217	2.233	-0.408	-0.38	0.065	2.233	-0.37112951	0.10434504	0.266900174	0.174068845	-0.242046	0.068052
-4							-0.145	0	0.069	-0.136	2.411	-0.369	-0.145	0.069	2.411	-0.13983638	0.07894166	0.220415351	0.143752044	-0.091199	0.051485
-2							0.024	0.011	0.077	-0.155	2.829	-0.428	0.024	0.077	2.829	0.026671294	0.07611598	0.209837331	0.136853196	0.017395	0.049642
0							0.303	0.048	0.083	-0.144	3.325	-0.484	0.303	0.083	3.325	0.303	0.083	0.178050687	0.116122358	0.197613	0.054132
2							0.554	-0.039	0.08	-0.129	3.658	-0.569	0.554	0.08	3.658	0.550872315	0.09927584	0.142173477	0.092723705	0.359272	0.064746
4							0.695	0.056	0.075	-0.097	3.793	-0.725	0.695	0.075	3.793	0.688079643	0.1232737	0.118531134	0.077304474	0.448757	0.080398
6							0.925	0.085	0.059	-0.11	3.991	-0.844	0.925	0.059	3.991	0.913773821	0.15531711	0.080127004	0.052257797	0.595951	0.101296
8							1.194	-0.029	0.056	-0.102	3.986	-0.872	1.194	0.056	3.986	1.174602065	0.22154455	0.011465023	0.00747734	0.76606	0.144488
10							1.286	0.058	0.082	0.018	3.765	-0.832	1.286	0.082	3.765	1.252250519	0.30395499	-0.03722587	-0.024278233	0.816702	0.198236

Run 3: Determine ele derivative, 6 m/s, +10 Elevator, PPM = 37360

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD
-6							-0.315	0.015	0.055	-0.202	1.323	-0.374	-0.315	0.055	1.323	-0.30752998	0.08760884	0.176955964	0.115408393	-0.200567	0.057137
-4							-0.112	-0.015	0.058	-0.151	1.526	-0.401	-0.112	0.058	1.526	-0.10768362	0.06566763	0.141020998	0.091972073	-0.07023	0.042828
-2							0.117	0.001	0.081	-0.132	1.938	-0.417	0.117	0.081	1.938	0.119754226	0.07686954	0.110700883	0.072197686	0.078102	0.050133
0							0.348	0.022	0.081	-0.101	2.342	-0.476	0.348	0.081	2.342	0.348	0.081	0.085101883	0.055502348	0.226961	0.052827
2							0.567	0.049	0.079	-0.131	2.8	-0.591	0.567	0.079	2.8	0.563899286	0.09872991	0.06759474	0.044084416	0.367768	0.06439
4							0.813	0.033	0.075	-0.07	2.926	-0.692	0.813	0.075	2.926	0.805792492	0.1315008	0.016006897	0.01043491	0.525527	0.085763
6							1.011	-0.049	0.071	-0.037	3.231	-0.85	1.011	0.071	3.231	0.998049473	0.17623635	-0.00834934	-0.005445329	0.650915	0.114939
8							1.404	-0.01	0.07	-0.012	3.095	-0.968	1.404	0.07	3.095	1.380612986	0.26462007	-0.12025471	-0.078428569	0.900418	0.172582
10							1.57	0.03	0.094	0.082	2.977	-1.071	1.57	0.094	2.977	1.529857551	0.36506421	-0.17883469	-0.116633674	0.997753	0.23809

Run 4: Determine ele derivative, 6 m/s, +5 Elevator, PPM = 33470

Airspeed = 6.04 Pressure = 20.65 Temp = 27.48 Rho = 1.13 P0 = 97512

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD
-6							-0.271	0.009	0.057	-0.199	0.662	-0.414	-0.271	0.057	0.662	-0.26356182	0.08500097	0.11007081	0.07178676	-0.171892	0.055437
-4							-0.076	0.011	0.057	-0.132	0.847	-0.394	-0.076	0.057	0.847	-0.07184095	0.0621601	0.075458017	0.049212744	-0.046854	0.04054
-2							0.142	-0.013	0.068	-0.101	1.241	-0.446	0.142	0.068	1.241	0.144285548	0.0630054	0.04953881	0.032308572	0.094101	0.041091
0							0.365	0.009	0.066	-0.106	1.659	-0.533	0.365	0.066	1.659	0.365	0.066	0.027672089	0.018047379	0.238048	0.043044
2							0.582	0.024	0.071	-0.089	2	-0.607	0.582	0.071	2	0.579169212	0.09125801	-0.00086786	-0.000566006	0.377727	0.059517
4							0.843	0.027	0.079	-0.041	2.23	-0.801	0.843	0.079	2.23	0.835440603	0.1375827	-0.05071619	-0.033076442	0.544864	0.08973
6							1.127	0.108	0.06	-0.045	2.523	-0.931	1.127	0.06	2.523	1.114563889	0.17741572	-0.09428828	-0.061493597	0.726904	0.115708
8							1.393	0.084	0.049	0.06	2.44	-0.989	1.393	0.049	2.44	1.372641092	0.2422941	-0.16662809	-0.108672686	0.895219	0.158021
10							1.298	0.04	0.075	0.154	2.335	-1.031	1.298	0.075	2.335	1.265283324	0.29914396	-0.15767535	-0.102833825	0.825201	0.195098

Run 5: Determine ele derivative, 6 m/s -5 Elevator, PPM = 26810

Airspeed = 6.04 Pressure = 20.65 Temp = 27.48 Rho = 1.13 P0 = 97512

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD
-6							-0.246	0.014	0.062	-0.145	-0.504	-0.401	-0.246	0.062	-0.504	-0.23817626	0.08736172	0.005207305	0.003396137	-0.155335	0.056976
-4							-0.033	0.012	0.065	-0.106	-0.247	-0.399	-0.033	0.065	-0.247	-0.02838782	0.06714262	-0.02879327	-0.018778602	-0.018514	0.04379
-2							0.197	0.003	0.075	-0.07	0.197	-0.477	0.197	0.075	0.197	0.19949625	0.06808264	-0.05336799	-0.034805914	0.130109	0.044403
0							0.423	0.046	0.08	-0.101	0.612	-0.545	0.423	0.08	0.612	0.423	0.08	-0.07805032	-0.050903412	0.275875	0.052175
2							0.651	0.039	0.075	-0.085	1.002	-0.636	0.651	0.075	1.002	0.647987694	0.09766242	-0.1027635	-0.067021028	0.422609	0.063694
4							0.902	0.085	0.066	0.022	1.252	-0.775	0.902	0.066	1.252	0.895203403	0.12872788	-0.14401801	-0.093926687	0.58384	0.083955
6							1.107	0.132	0.068	0.01	1.313	-0.916	1.107	0.068	1.313	1.093837534	0.18328243	-0.19204287	-0.125247881	0.713387	0.119534
8							1.405	0.303	0.047	0.013	1.285	-1.152	1.405	0.047	1.285	1.384802633	0.24198278	-0.2654437	-0.173118953	0.90315	0.157818
10							1.751	-0.245	0.071	0.124	0.97	-0.912	1.751	0.071	0.97	1.712102438	0.37382782	-0.38662139	-0.252149473	1.116611	0.243806

Run 6: Determine ele derivative, 6 m/s, -10 Elevator, PPM = 22050

Airspeed = 6.04 Pressure = 20.65 Temp = 27.48 Rho = 1.13 P0 = 97512

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD

-6							-0.186	0.01	0.07	-0.086	-1.402	-0.464	-0.186	0.07	-1.402	-0.17766881	0.08904939	-0.0870722	-0.056787361	-0.115873	0.058077
-4							0.006	0.025	0.062	-0.071	-1.067	-0.481	0.006	0.062	-1.067	0.010308112	0.0614308	-0.10636282	-0.069368455	0.006723	0.040064
-2							0.209	0.022	0.076	-0.038	-0.74	-0.43	0.209	0.076	-0.74	0.21152383	0.06866345	-0.13478662	-0.087906093	0.137953	0.044781
0							0.497	0.001	0.088	-0.033	-0.291	-0.5	0.497	0.088	-0.291	0.497	0.088	-0.17433835	-0.113701219	0.324137	0.057392
2							0.707	0.008	0.083	0.015	0.083	-0.626	0.707	0.083	0.083	0.703674561	0.10761093	-0.19576676	-0.127676549	0.458927	0.070182
4							1.031	0	0.069	0.051	0.223	-0.743	1.031	0.069	0.223	1.02368032	0.14071461	-0.26363411	-0.171938758	0.667631	0.091772
6							1.081	0.12	0.054	0.018	0.452	-0.931	1.081	0.054	0.452	1.06944248	0.16664268	-0.25353043	-0.165349268	0.697477	0.108682
8							1.577	0.013	0.054	0.151	0.375	-0.948	1.577	0.054	0.375	1.554156714	0.27284045	-0.38720151	-0.252527823	1.013601	0.177943
10							1.566	0.214	0.066	0.291	0.299	-1.144	1.566	0.066	0.299	1.530777967	0.33679491	-0.3937914	-0.256825665	0.998354	0.219653

Run 7: Determine ele derivative, 6 m/s, -15 Elevator, PPM = 20170 Airspeed = 6.04 Pressure = 20.65 Temp = 27.48 Rho = 1.13 P0 = 97512

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD
-6							-0.191	0.009	0.069	-0.08	-1.719	-0.466	-0.191	0.069	-1.719	-0.18274592	0.08857725	-0.11194949	-0.073012014	-0.119185	0.057769
-4							0.063	0.008	0.078	-0.046	-1.421	-0.456	0.063	0.078	-1.421	0.068284942	0.07341775	-0.15459179	-0.100822767	0.044535	0.047882
-2							0.239	-0.033	0.08	-0.02	-1.05	-0.435	0.239	0.08	-1.05	0.2416451	0.07161456	-0.16934304	-0.110443341	0.157598	0.046706
0							0.512	0.029	0.075	-0.033	-0.689	-0.545	0.512	0.075	-0.689	0.512	0.075	-0.20801814	-0.135666745	0.33392	0.048914
2							0.748	0.03	0.088	-0.029	-0.28	-0.665	0.748	0.088	-0.28	0.744475201	0.11403804	-0.23781858	-0.155102203	0.485537	0.074374
4							0.941	0.066	0.073	0.018	-0.192	-0.759	0.941	0.073	-0.192	0.933620449	0.13842997	-0.27615316	-0.180103526	0.608895	0.090282
6							1.23	0.022	0.059	0.058	-0.006	-0.99	1.23	0.059	-0.006	1.217104691	0.18718219	-0.33120753	-0.216009275	0.79378	0.122078
8							1.439	0.163	0.063	0.201	0.121	-1.095	1.439	0.063	0.121	1.416246434	0.26255673	-0.37527176	-0.244747396	0.923658	0.171236
10							1.6	0.206	0.082	-0.01	-0.153	-0.947	1.6	0.082	-0.153	1.561484977	0.35845316	-0.44428613	-0.289757674	1.01838	0.233779

Run 8: Determine CL at 8 m/s, 0 rudder, 0 elevator Airspeed = 8 Pressure = 36.1 Temp = 27.98 Rho = 1.13 P0 = 97519

	Wind off						Wind On						Corrected			Aero Data					
incidence angle (deg)	Normal	Transverse	Axial	Normal M	Transverse M	Axial M	Normal	Transverse	Axial	Normal M	Transverse M	Axial Moment	Normal Force	Axial Force	Transverse M	Lift	Drag	M1/4	CM1/4	CL	CD
-6							-0.418	0.014	0.131	-0.29	-0.461	-0.76	-0.418	0.131	-0.461	-0.40202616	0.17395392	0.035216478	0.022967732	-0.262196	0.113451
-4							-0.049	0.027	0.11	-0.169	0.119	-0.695	-0.049	0.11	0.119	-0.04121143	0.11314865	-0.00573354	-0.0037393	-0.026878	0.073794
-2							0.327	0.025	0.113	-0.153	0.89	-0.742	0.327	0.113	0.89	0.330742647	0.10152488	-0.03872028	-0.025252867	0.215706	0.066213
0							0.778	0.023	0.123	-0.187	1.916	-1.088	0.778	0.123	1.916	0.778	0.123	-0.07149505	-0.046628149	0.507402	0.080219
2							1.157	0.108	0.135	-0.118	2.742	-1.107	1.157	0.135	2.742	1.151586857	0.1752761	-0.10297719	-0.067160395	0.75105	0.114313
4							1.559	0.043	0.114	-0.027	3.234	-1.499	1.559	0.114	3.234	1.547257989	0.22241788	-0.15972707	-0.10417193	1.009102	0.145058
6							2.073	0.138	0.094	-0.073	3.411	-1.811	2.073	0.094	3.411	2.051834677	0.31006363	-0.27171837	-0.177211211	1.33818	0.202219