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Airfoil Project

NACA 2412

Table 1 Lift and Moment Coefficients for NACA 2412 Calculated by VORPAN

alpha	C_L	C_m_le	C_m_qtr
-12	-1.18434	0.249132	-0.04048
-11	-1.06515	0.219833	-0.04156
-10	-0.94564	0.190142	-0.04268
-9	-0.82583	0.160095	-0.04382
-8	-0.70578	0.129729	-0.045
-7	-0.5855	0.09908	-0.04621
-6	-0.46505	0.068186	-0.04744
-5	-0.34446	0.037084	-0.0487
-4	-0.22377	0.005813	-0.04999
-3	-0.103	-0.02559	-0.05131
-2	0.017792	-0.05709	-0.05264
-1	0.138582	-0.08864	-0.054
0	0.25933	-0.1202	-0.05537
1	0.379999	-0.15175	-0.05676
2	0.500553	-0.18323	-0.05817
3	0.620953	-0.21462	-0.05959
4	0.741165	-0.24587	-0.06103
5	0.861151	-0.27694	-0.06247
6	0.980874	-0.3078	-0.06393
7	1.100299	-0.33841	-0.06539
8	1.219389	-0.36874	-0.06686
9	1.338107	-0.39873	-0.06833
10	1.456417	-0.42837	-0.0698
11	1.574284	-0.45761	-0.07127
12	1.691671	-0.48641	-0.07274
13	1.808543	-0.51475	-0.0742
14	1.924865	-0.54259	-0.07566
15	2.040599	-0.56988	-0.07712
16	2.155713	-0.59661	-0.07856

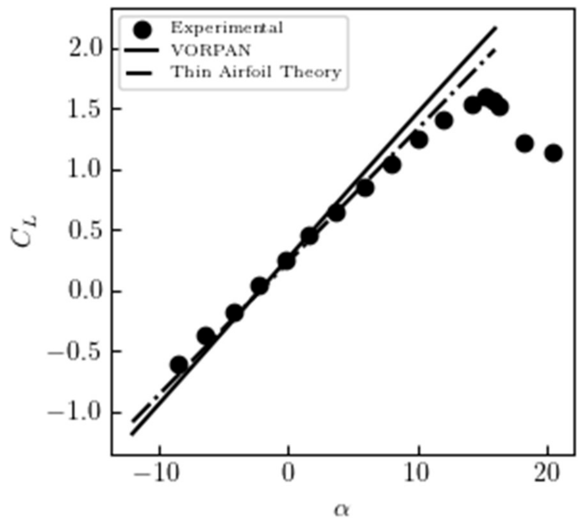

Figure 1 Lift Coefficient for NACA 2412

Close to zero angle of attack, VORPAN does well at predicting lift compared to experimental results. Thin airfoil theory does well at higher angles of attack. Neither method predicts nonlinearities and stall.

NACA 2421

Table 2 Lift and Moment Coefficients for NACA 2421 Calculated By VORPAN

alpha	C_L	C_m_le	C_m_qtr
-12	-1.25946	0.284108	-0.02388
-11	-1.13194	0.251477	-0.02631
-10	-1.00407	0.218402	-0.0288
-9	-0.87589	0.184921	-0.03136
-8	-0.74745	0.151076	-0.03397
-7	-0.61877	0.116909	-0.03663
-6	-0.48992	0.08246	-0.03935
-5	-0.36091	0.047771	-0.04211
-4	-0.23179	0.012886	-0.04492
-3	-0.1026	-0.02215	-0.04777
-2	0.026621	-0.05731	-0.05065
-1	0.155833	-0.09253	-0.05357
0	0.284997	-0.12777	-0.05652
1	0.414075	-0.163	-0.0595
2	0.543027	-0.19817	-0.0625
3	0.671814	-0.23324	-0.06552
4	0.800395	-0.26817	-0.06855

5	0.928733	-0.3029	-0.0716
6	1.056788	-0.3374	-0.07466
7	1.184521	-0.37164	-0.07771
8	1.311894	-0.40556	-0.08077
9	1.438866	-0.43912	-0.08383
10	1.565401	-0.47229	-0.08688
11	1.691458	-0.50502	-0.08992
12	1.817	-0.53727	-0.09295
13	1.941989	-0.56901	-0.09596
14	2.066387	-0.60019	-0.09894
15	2.190154	-0.63079	-0.1019
16	2.313255	-0.66075	-0.10484

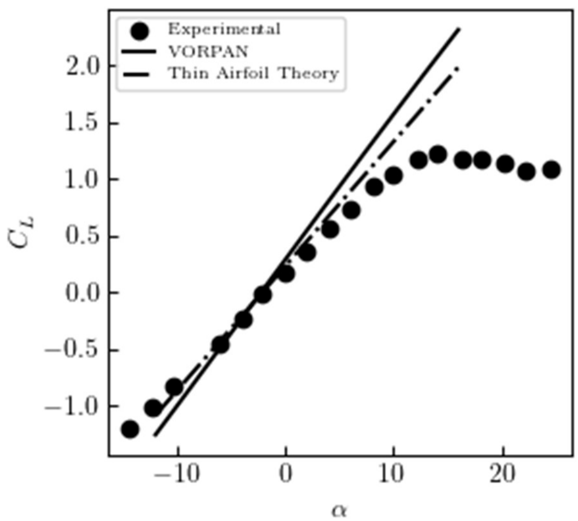

Figure 2 Lift Coefficient for NACA 2421

Experimental results show that the NACA 2421 airfoil stalls sooner than the 2412. This is most likely due to the thicker airfoil. As with the NACA 2412, VORPAN performs well close to zero and thin airfoil does well at higher angles of attack.

NACA 0015

Table 3 Lift and Moment Coefficients for NACA 0015 Calculated by VORPAN

alpha	C_L	C_m_le	C_m_qtr
-12	-1.47119	0.380935	0.021176
-11	-1.35017	0.350843	0.019503
-10	-1.22874	0.320324	0.017807
-9	-1.10693	0.289414	0.016088
-8	-0.98479	0.258152	0.01435

-7	-0.86235	0.226576	0.012595
-6	-0.73965	0.194723	0.010824
-5	-0.61672	0.162633	0.009041
-4	-0.4936	0.130345	0.007246
-3	-0.37033	0.097898	0.005442
-2	-2.47E-01	6.53E-02	3.63E-03
-1	-0.12349	0.032686	0.001817
0	6.08E-15	1.68E-15	3.20E-15
1	0.123494	-0.03269	-0.00182
2	0.246949	-0.06533	-0.00363
3	0.37033	-0.0979	-0.00544
4	0.493598	-0.13034	-0.00725
5	0.616715	-0.16263	-0.00904
6	0.739645	-0.19472	-0.01082
7	0.862349	-0.22658	-0.0126
8	0.984791	-0.25815	-0.01435
9	1.106933	-0.28941	-0.01609
10	1.228737	-0.32032	-0.01781
11	1.350168	-0.35084	-0.0195
12	1.471186	-0.38094	-0.02118
13	1.591757	-0.41056	-0.02282
14	1.711843	-0.43969	-0.02444
15	1.831408	-0.46828	-0.02603
16	1.950414	-0.4963	-0.02759

Table 4 NACA 0015 Section forces computed by XFOIL

alpha	C_L	C_m_le	C_m_qtr	C_D
-12	-1.3138	0.325932	0.0039	0.01462
-11		0		
-10	-1.102	0.268091	-0.0037	0.01098
-9	-0.9793	0.235498	-0.0067	0.00989
-8	-0.8738	0.211037	-0.0056	0.009
-7	-0.7711	0.187988	-0.0036	0.00821
-6	-0.6649	0.163111	-0.0024	0.00751
-5	-0.5567	0.137296	-0.0015	0.00693
-4	-0.4474	0.11089	-0.0008	0.00645
-3	-0.337	0.083814	-0.0004	0.00609
-2	-0.2253	0.056242	-0.0001	0.00584
-1	-0.1127	0.028096	-0.0001	0.0057
0	0.0001	-2.5E-05	0	0.00566

1	0.1127	-0.0281	0.0001	0.0057
2	0.2253	-0.05624	0.0001	0.00584
3	0.337	-0.08381	0.0004	0.00609
4	0.4474	-0.11089	0.0008	0.00645
5	0.5567	-0.1373	0.0015	0.00693
6	0.6649	-0.16311	0.0024	0.00751
7	0.7711	-0.18799	0.0036	0.00821
8	0.8738	-0.21104	0.0056	0.009
9	0.9738	-0.23414	0.0067	0.00989
10	1.102	-0.26809	0.0037	0.01098
11		0		
12	1.3138	-0.31807	0.0039	0.01334
13	1.3885	-0.32895	0.0101	0.01462
14	1.4595	-0.33882	0.0162	0.0162
15	1.5217	-0.34615	0.0225	0.01828
16	1.5797	-0.35327	0.0278	0.02097

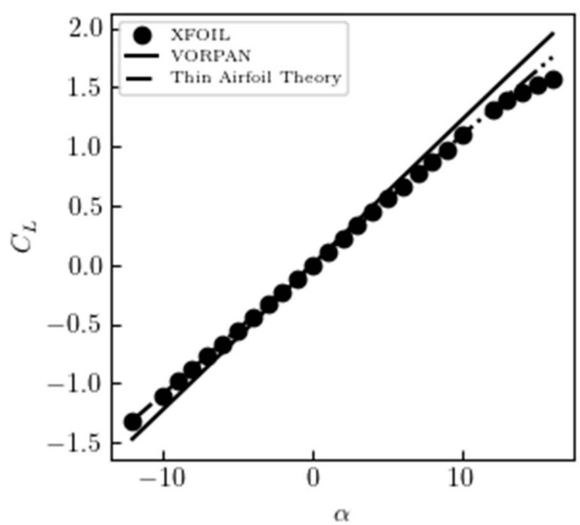

Figure 3 Lift Coefficient for NACA 0015

For the NACA 0015, XFOIL is followed fairly well by thin airfoil theory despite XFOIL being a vortex panel code. This difference between XFOIL and VORPAN is most likely due to the skin friction negatively impacting lift as the angle of attack increases.