DESIGN OF LOW REYNOLDS NUMBER AIRFOIL FOR MICRO AERIAL VEHICLE

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Abstract. The aerodynamic characteristics of low Reynolds number flows make the dilemma of new airfoil design. It is the fact that the boundary layer is a great deal less capable of managing an adverse pressure gradient without separation. Hence, very low Reynolds number designs do no longer have astringent pressure gradients and the maximum lift functionality is restrained. In many commercial applications, the evaluation of the impact of laminar separation can be essential for the presage of specific ecumenical and local aerodynamic performances. In this paper, the low Reynolds number airfoil coordinates from the UIUC airfoil database is extracted. Utilizing preliminary analysis implement i.e. *Xfoil* panel code method analyzed the aerodynamic characteristics over 200 plus airfoil. Based on the outcome of the *Xfoil* results, the best three airfoils i.e. FX63137sm, S1223, and e423 are chosen for understanding the aerodynamic characteristics. The Reynolds number ranges from 3.42 X 10^5 to 10.28 X 10^5. Using XFLR5 software and adopted Foil Direct Design method, the new airfoil is generated based on the maximum lift coefficient by modifying the airfoil parameters i.e. maximum thickness, max camber, location of thickness and camber. The aerodynamic characteristics of new airfoil are analysed and validated against the reference airfoil such as FX63137sm, S1223, and e423 airfoil.

Key Words: *Low Reynolds Number airfoil design, XFOIL, XFLR5, Polar Curves.*

1. INTRODUCTION

Airfoil Design Methods

The design of airfoil proceeds from a knowledge of the relationship between geometry and pressure distribution and also the boundary layer properties. Generally, airfoil are design to maximise the lift and reduce the drag with the constraints are off-design performance, or thickness, or pitching moment, or other unusual constraints. The airfoil design methods are classified into two categories: [1]

1. Direct method for airfoil design
2. Inverse method for airfoil design

Direct Methods for Airfoil Design

Direct method for airfoil design involves the geometry specification of a section and the calculation of pressure and performance. In this method the given shape is evaluated initially and then the shape is modified to improve the performance. The two main sub-problems in this type of method are. [1]

1. The identification of the measure of performance
2. The approach to changing the shape so that the performance is improved

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Inverse Method for Airfoil Design

The inverse method of airfoil design also involves changing the airfoil shape to improve the aerodynamic performance. This may be done in two ways: 1. Changing the geometry 2. Numerical optimization: with the help of shape functions changing the airfoil geometry and compute the sequence of geometry modifications to improve the design. [1]

The design of low Reynolds number airfoil is complicated due to the formation of the separation bubble. It increases the drag and decreases the lift. [2]

In this paper, a attempt has been made to design the low Reynolds number airfoil with the help of direct design method, particularly for micro aerial vehicle applications to have better aerodynamic performance.

2. METHODOLOGY

The focus of the proposed study is to design the new low Reynolds number airfoil to improve the aerodynamic performance in micro aerial vehicles, for the Reynolds number ranges from 3.42 X 10^5 to 10.28 X 10^5 at different angle of attack (-5 to 20 degree) and to achieve at an optimum result against the numerical/experimental and existing literature results. The rigorous work involves the design and analysis of low Reynolds number 2D airfoil using *Xfoil* and FXLR5. Aerodynamic characteristics such as pressure distribution, lift coefficient, and drag coefficient over the 2D airfoil at different flight conditions are studied and an optimum configuration is suggested for safe operations. The flow chart in Fig.1 represents the step by step working procedure of this work. Utilizing preliminary analysis implement i.e. *Xfoil* panel code method analyzed the aerodynamic characteristics over 200 plus airfoil. Based on the outcome of the *Xfoil* results, the best three airfoils i.e. FX63137sm, S1223, and e423 are chosen for understanding the aerodynamic characteristics. The Reynolds number ranges from 3.42 X 10^5 to 10.28 X 10^5. The S1223 airfoil gives maximum lift coefficient within the given range of Reynolds number and chosen for further process design a new low Reynolds number airfoil, when compared with the FX63137sm, and e423 airfoil. XFLR5 software (Foil Direct Design method). With this method modifying the airfoil parameters i.e. maximum thickness, max camber, location of thickness and camber the incipient airfoil generated. The incipient airfoil is examined for the aerodynamic performance and it's validated against to the reference airfoil i.e. S1223.

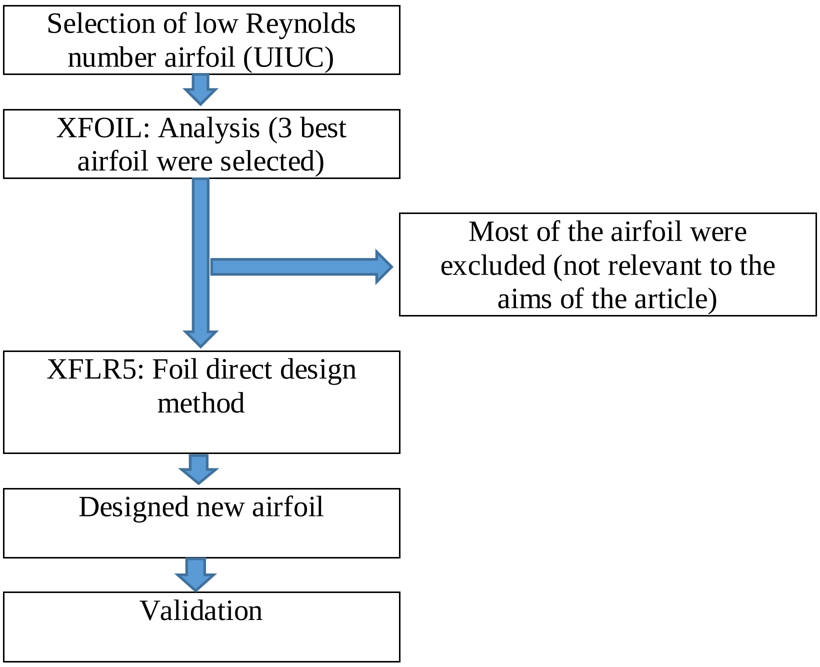

Figure 1 Methodology

3. FOIL ANALYSIS AND DESIGN MODES

3.1 Xfoil:

Panel code method, fairly straightforward and freely available subsonic airfoil development interactive program. The aerodynamic characteristics for selected 2D airfoil are obtained from *Xfoil*. In this program, airfoil coordinates are entered to get the geometry. Later entered the Reynolds number and AOA sequence from -5^0 to $+20^0$, the aerodynamic characteristics have been generated such as lift, drag, moment coefficient, and pressure coefficient:

$$Re = \frac{\rho V c}{\mu}$$

Where ρ = 1.22 kg/m^3, V = 5, 10, 15 m/s, chord (c) = 1 m, μ = 1.78 X 10^{-5} N-s/m^2

Case – 1: V = 5 m/s; Re = 342697

Case – 2: V = 10 m/s; Re = 685393

Case – 3: V= 15 m/s; Re = 1028090

The selected airfoil parameters are as follows.

Table 1. Airfoil Parameters

Airfoil Parameters	FX 63-137	E423	S1223
Thickness (%)	13.67	12.52	12.13
Max. Thickness Position (%)	30.3	24.24	20.21
Max Camber (%)	5.95	10.03	8.67
Max. Camber Position (%)	50.51	44.45	49.50
Number of Panels	69	72	81

3.2 XFLR5:

With the help of Foil Direct Design Method, modifying the airfoil parameters i.e. maximum thickness, max camber, location of thickness and camber, the incipient airfoil generated. The following airfoil parameters are modified and analysed for the following conditions.

Condition – 1: 1% Increment in camber

Condition – 2: 2% Increment in camber

Condition – 3: 1% Increment in thickness

Condition – 4: 2% Increment in thickness

Condition – 5: 3% Increment in thickness

Condition – 6: 1% Decrement in camber

Condition – 7: 1% Decrement in thickness

Condition – 8: 1% Increment camber & thickness

Condition – 9: 1% Increment thickness & increment 10% location

Condition – 10: 1% Increment thickness & decrement 10% location

Condition – 11: 1% Increment thickness & increment 5% location

$$Re = \frac{\rho V c}{\mu}$$

Where ρ = 1.22 kg/m^3, V = 5, 10, 15 m/s, chord (c) = 1 m, μ = 1.78 X 10^{-5} N-s/m^2

Case – 1: V = 5 m/s; Re = 342697

Case – 2: V = 10 m/s; Re = 685393

Case – 3: V= 15 m/s; Re = 1028090

4. RESULT AND DISCUSSIONS

4.1. *Xfoil*:

In this section, the non-dimensional aerodynamic force coefficients are computed for the FX63137sm, S1223, and e423 for different angle of attack at various velocity i.e. 5, 10 and 15 m/s using *Xfoil* panel code method. Figure 2 and 3 show the variation of the coefficient of lift versus angle of attack and the variation of the lift-by-drag ratio versus angle of attack for all the cases. The S1223 airfoil gives maximum lift coefficient within the given range of Reynolds number and chosen for further process design a new low Reynolds number airfoil, when compared with the FX63137sm, and e423 airfoil. Table 2 shows the maximum lift coefficient for all cases.

Table 2. *Xfoil* Results

Airfoil	Re = 342697		Re = 685393		Re = 1028090	
	$\propto_{stall}$	C_{lmax}	$\propto_{stall}$	C_{lmax}	$\propto_{stall}$	C_{lmax}
FX63137sm	17^0	1.7751	17^0	1.8549	17^0	1.9471
S1223	11^0	2.2733	13^0	2.2767	13^0	2.2849
e423	13^0	2.0065	13^0	2.0481	13^0	2.0604

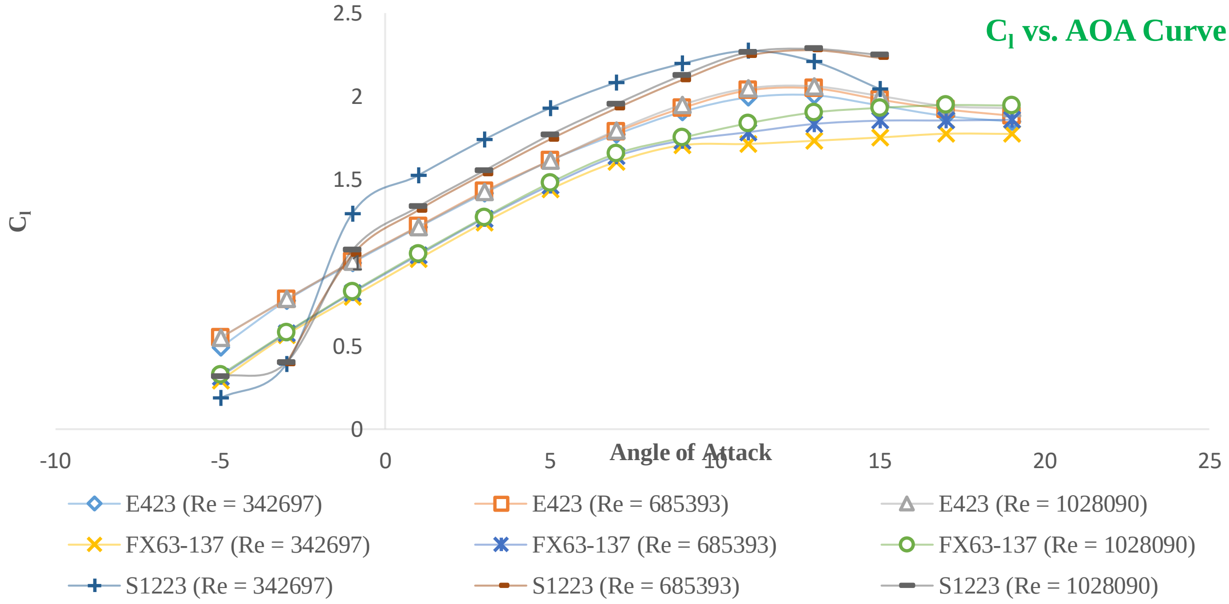

Figure 2 Lift coefficient versus angle of attack

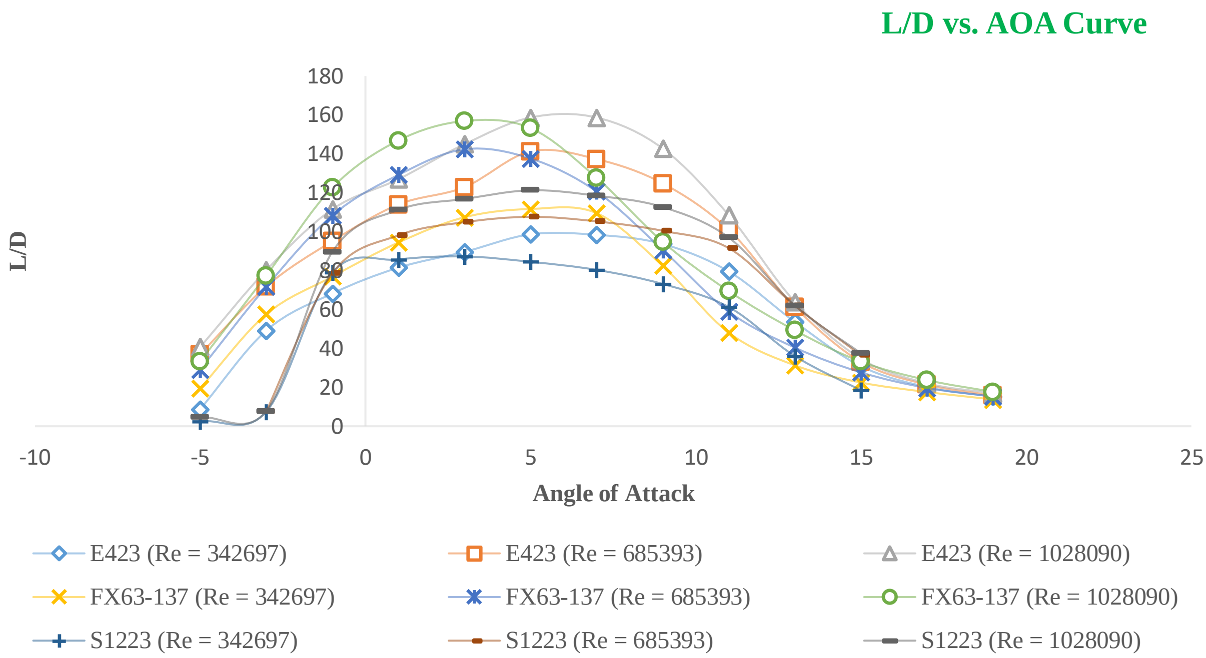

Figure 3. Lift/Drag ratio versus angle of attack

4.2. XFLR5:

In this section, the non-dimensional aerodynamic force coefficients are computed for the various conditions for different angle of attack at various velocity i.e. 5, 10 and 15 m/s using XFLR5. The following results found through the analysis as follows.

Table 3: Summary of XFLR5 Results

Airfoil Parameters		Re = 342697 C_{lmax}	Re = 685393 C_{lmax}	Re = 1028090 C_{lmax}
Camber	1% Increment	2.3796	2.4308	2.3414
	2 % Increment	2.4693	2.5162	2.5308
	1% Decrement	2.1909	2.2457	2.2895
Thickness	1% Increment	2.3112	2.3785	2.4132
	2 % Increment	2.334	2.4141	2.4481
	3% Increment	2.3445	2.4298	2.4776
	1% Decrement	2.1293	2.3074	2.3379
1% Increases camber & thickness		2.4064	2.4789	2.5012
1% Increased thickness & increased 10% location		2.1779	2.1929	2.1499
1% Increased thickness & decreased 10% location		2.2508	2.2858	2.3578
1% Increased thickness & increased 5% location		2.2749	2.2521	2.2779
S1223 (Reference Airfoil)		2.2733	2.2767	2.2849

Table 3 shows the maximum lift coefficient for various conditions and the second condition (2% increment in camber) gives the max lift coefficient compare to the other conditions. Based on the maximum lift coefficient, the new airfoil coordinates and geometry were developed and named as SS007. The newly generated airfoil parameter and it's geometry as follows. Table 4, Table 5 and Figure 4 show the newly developed airfoil parameters, airfoil coordinates and geometry respectively.

Table 4. SS007 Airfoil Parameters

Airfoil Parameters	SS007
Thickness (%)	12.18
Max. Thickness Position (%)	20.20
Max Camber (%)	10.56
Max. Camber Position (%)	49.50
Number of Panels	200

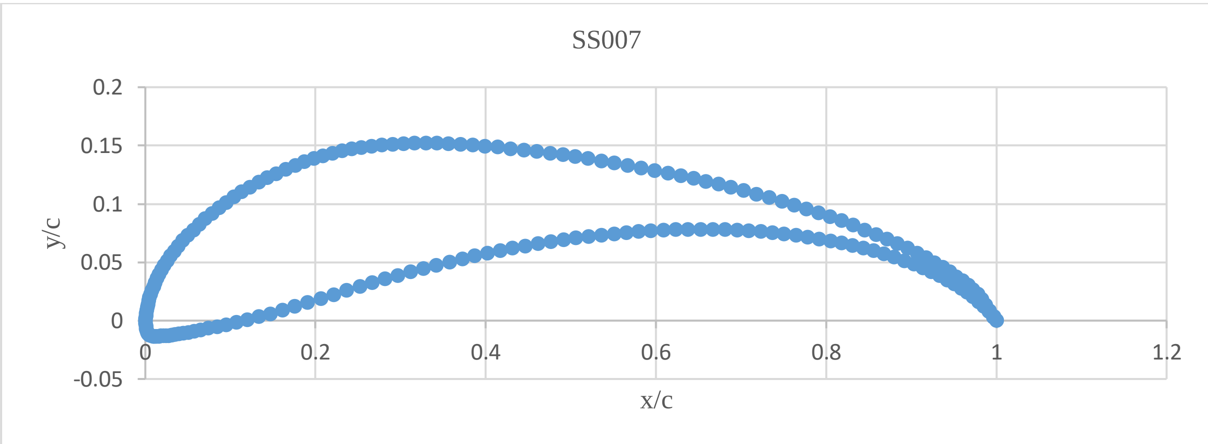

Figure 4 SS007Airfoil Geometry

Table 5. SS007 Airfoil Coordinates

Upper Surface						Lower Surface					
x/c	y/c	x/c	y/c	x/c	y/c	x/c	y/c	x/c	y/c	x/c	y/c
1	0.00008	0.58213	0.1308	0.10367	0.10585	0.00065	0.00638	0.20574	0.01881	0.75034	0.07439
0.99669	0.00345	0.56645	0.13295	0.09469	0.10136	0.0003	0.00398	0.22103	0.02234	0.76405	0.07314
0.99196	0.00861	0.55094	0.13499	0.08609	0.09675	0.0001	0.00167	0.23639	0.02588	0.77775	0.07172
0.98735	0.01363	0.53552	0.13694	0.07788	0.09205	0.00003	-0.00053	0.25161	0.02935	0.79136	0.07013
0.98268	0.01825	0.52021	0.13879	0.07009	0.0873	0.0001	-0.00266	0.26656	0.03267	0.80479	0.06838
0.97777	0.02245	0.50509	0.14052	0.06278	0.08254	0.0003	-0.00473	0.28132	0.03582	0.81796	0.06646
0.97241	0.02638	0.49012	0.1421	0.05597	0.07781	0.00065	-0.00675	0.29616	0.03884	0.83083	0.06439
0.96652	0.03018	0.47508	0.14355	0.04968	0.07314	0.00117	-0.00863	0.31123	0.04181	0.84339	0.06215
0.96	0.03396	0.45976	0.14492	0.04387	0.06853	0.00197	-0.01025	0.32652	0.04473	0.85568	0.05973
0.95285	0.03778	0.44425	0.14623	0.03854	0.06403	0.00313	-0.0115	0.34183	0.04758	0.86779	0.0571
0.94507	0.04164	0.42878	0.14747	0.03369	0.05967	0.00467	-0.01238	0.35702	0.05033	0.87973	0.05427
0.93656	0.04553	0.41357	0.14862	0.02933	0.05547	0.00652	-0.01296	0.37205	0.05294	0.8914	0.05129
0.92719	0.04954	0.39868	0.14964	0.02542	0.05139	0.00855	-0.01331	0.38694	0.05543	0.90259	0.04821
0.91702	0.05369	0.38415	0.15052	0.02192	0.04744	0.01069	-0.01347	0.40174	0.05777	0.91321	0.04504
0.90628	0.05786	0.36997	0.15122	0.01879	0.04359	0.01294	-0.01347	0.4165	0.05998	0.92326	0.04178
0.89514	0.06195	0.35614	0.15173	0.01599	0.03985	0.01533	-0.01336	0.43129	0.06206	0.93278	0.03841
0.88358	0.06594	0.34257	0.15204	0.0135	0.03622	0.0179	-0.0132	0.44612	0.06403	0.9418	0.03494
0.87141	0.06986	0.32922	0.15214	0.01129	0.03271	0.02068	-0.01304	0.46102	0.06588	0.95026	0.0314
0.85848	0.07382	0.31602	0.15204	0.00933	0.02931	0.02371	-0.01287	0.47597	0.06761	0.95814	0.0278
0.84502	0.07782	0.30299	0.15173	0.0076	0.02604	0.02701	-0.01268	0.49094	0.06924	0.96544	0.02412
0.8314	0.08174	0.29018	0.15122	0.00607	0.02288	0.03062	-0.01241	0.5059	0.07075	0.97226	0.0203

0.81776	0.08551	0.27766	0.1505	0.00474	0.01985	0.03459	-0.01201	0.52082	0.07214	0.97868	0.01632
0.8041	0.08912	0.26552	0.14958	0.00359	0.01695	0.03905	-0.01148	0.53565	0.0734	0.98477	0.01212
0.79038	0.09256	0.25374	0.14843	0.00261	0.01416	0.04415	-0.01078	0.55038	0.07452	0.99063	0.00768
0.77652	0.09585	0.24221	0.14702	0.0018	0.01148	0.05006	-0.00995	0.56498	0.07549	0.99627	0.00308
0.76232	0.09903	0.23084	0.14535	0.00115	0.00888	0.05695	-0.00898	0.57947	0.07632	1	0.00008
0.74761	0.10219	0.21961	0.14342			0.0649	-0.00787	0.59389	0.07698		
0.73252	0.10537	0.20856	0.14123			0.0739	-0.00662	0.60831	0.07749		
0.71743	0.10847	0.19764	0.13876			0.08391	-0.0052	0.62281	0.07785		
0.70255	0.11145	0.1867	0.13599			0.09486	-0.00352	0.63739	0.07807		
0.68784	0.11427	0.17565	0.1329			0.10683	-0.00151	0.65199	0.07815		
0.6732	0.11694	0.16458	0.12957			0.11972	0.00076	0.66651	0.07809		
0.65859	0.11947	0.15369	0.12604			0.13313	0.00324	0.68088	0.07789		
0.64394	0.12185	0.14308	0.12235			0.14681	0.00593	0.69507	0.07753		
0.62906	0.12413	0.13277	0.11847			0.16092	0.00887	0.70907	0.07701		
0.61374	0.12636	0.12274	0.11442			0.17556	0.01204	0.72291	0.07633		
0.59798	0.12858	0.11303	0.11021			0.19056	0.01537	0.73664	0.07545		

5. VALIDATION OF SS007 AIRFOIL

In this section, the non-dimensional aerodynamic force coefficients are computed the newly developed SS007 airfoil for different angle of attack at various velocity i.e. 5, 10 and 15 m/s using *XFLR5*. Table 6 shows the aerodynamic performance for various conditions.

Table 6. XFLR5 Results

SS007 Airfoil									
AOA	Re = 342697			Re = 685393			Re = 1028090		
	C_l	C_d	C_l/C_d	C_l	C_d	C_l/C_d	C_l	C_d	C_l/C_d
-5	0.2987	0.07956	3.754399	0.4302	0.06759	6.364847	0.401	0.07018	5.7138786
-3	0.4378	0.05875	7.451915	0.4005	0.06029	6.642893	0.4005	0.06029	6.6428927
-1	0.6403	0.04045	15.82942	1.163	0.01719	67.65561	1.2201	0.015	81.34
1	1.4661	0.01942	75.49434	1.4738	0.01689	87.25873	1.4994	0.01535	97.680782
3	1.6843	0.02194	76.76846	1.6948	0.01895	89.43536	1.7215	0.0171	100.67251
5	1.8946	0.02438	77.71124	1.9009	0.02153	88.29076	1.9237	0.01946	98.85406
7	2.0885	0.02776	75.23415	2.0747	0.02473	83.89406	2.1041	0.0225	93.515556
9	2.2548	0.03183	70.83883	2.2348	0.02886	77.4359	2.2678	0.02615	86.722753
11	2.3762	0.03747	63.41607	2.3707	0.03432	69.07634	2.4114	0.03102	77.736944
13	2.4445	0.04659	52.46834	2.4812	0.04199	59.09026	2.5308	0.03784	66.881607
15	2.4693	0.06226	39.6611	2.5162	0.05815	43.27085	2.5252	0.05807	43.485449
17	2.3731	0.09907	23.95377	2.406	0.09948	24.185766	2.406	0.09948	24.185766
19	2.2132	0.15167	14.59221	2.2369	0.1528	14.6394	2.2517	0.15233	14.781724

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Figure 5 and 6 give the overall view of the study carried out in this work for two dimensional case; three cases are considered for various angles of attack (α), that is, -5^0 through 20^0 with interval of 2^0 of angle of attack.

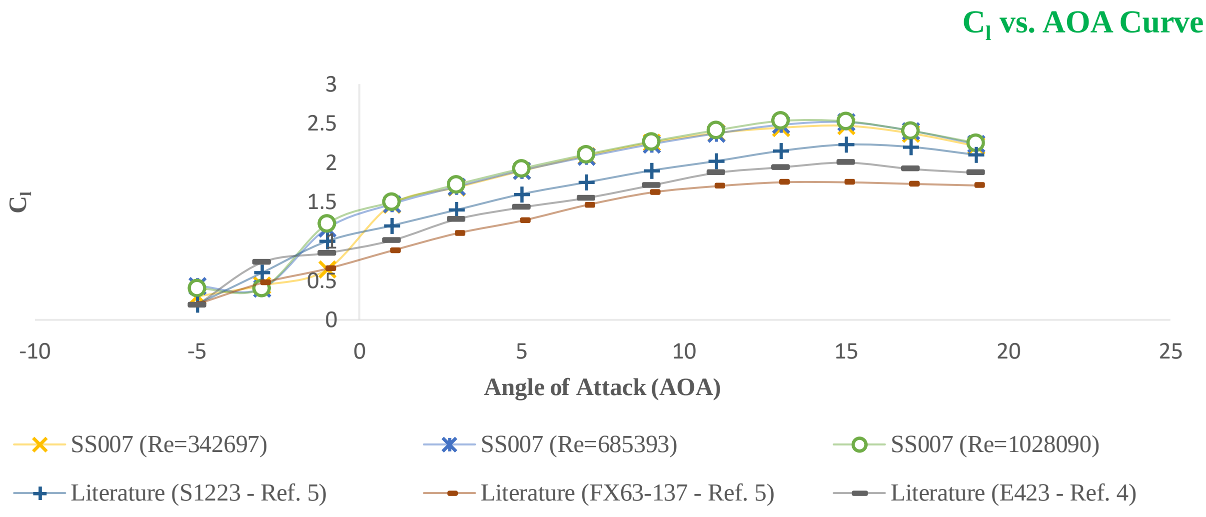

Figure 5 Lift coefficient versus angle of attack

Figure 5 shows the lift coefficient (C_l) versus angle of attack for the newly designed SS 007 2D airfoil and the plot shows very good correlation between the three cases i.e. the Reynolds number ranges from 3.42 X 10^5 to 10.28 X 10^5 as well as the experimental/numerical data obtained from the literature. The plot shows that the solution obtained from the three different cases the maximum variation of lift coefficient 2.4%. The above obtained results differences as compared to the experimental/numerical data was with maximum variation of lift coefficient as 42%. Maximum lift coefficient for the above reference values occurs at angle of attack of 15^0 and for the present work, maximum lift coefficient occurs at angle of attack of 15^0 for case-1 and case-2 for case-3 maximum lift coefficient occurs at angle of attack of 13^0, the corresponding maximum lift coefficient is $C_{lmax} = 2.4693$, $C_{lmax} = 2.5162$ and $C_{lmax} = 2.5308$ respectively.

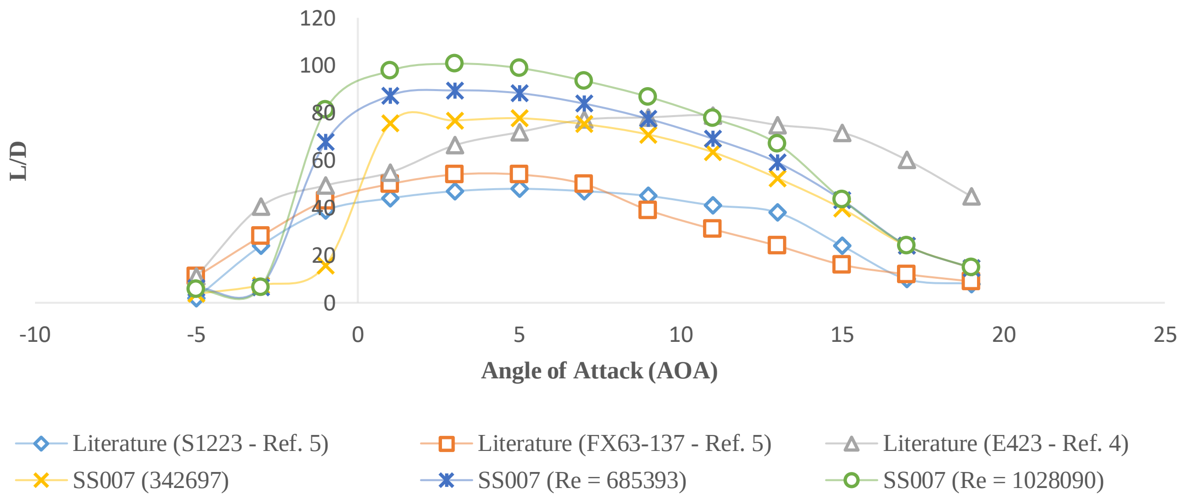

Figure 6 Lift/Drag versus angle of attack

Figure 6 shows the Lift/Drag versus angle of attack for the newly designed SS007 2D airfoil and the plot shows very good correlation between the three cases i.e. the Reynolds number ranges from 3.42 X 10^5 to 10.28 X 10^5 as well as the experimental/numerical data obtained from the literature. The plot shows that the solution obtained from the three different cases the maximum variation of lift/drag ratio as 68.63%. The above obtained results differences as compared to the experimental/numerical data was with maximum variation of lift/drag ratio as 3.45%.

6. Conclusions

As a result of this work, it is clear that low Reynolds number airfoil can be designed to achieve the maximum lift coefficient much higher than the reference airfoil. Such a high lift performance can be achieved through the use of a direct design method with the help of XFLR5 open source software. Applications of this philosophy was demonstrated through the successful design of a high lift low Reynolds number airfoil that achieved a maximum lift coefficient $C_{lmax} = 2.53$ at Reynolds number ranges from 3.42 X 10^5 to 10.28 X 10^5. From the above obtained results, the maximum variation of lift coefficient is 10.96% when compared to the S1223 reference airfoil data. So it's proved that the newly developed SS007 2D airfoil give better aerodynamic performance than the reference airfoil. This airfoil is strongly recommended for design and development of micro aerial vehicles (MAV).

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