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Article

DESIGN OF AN AIRFOIL BY MATHEMATICAL MODELLING USING DATABASE

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Abstract.

The aerodynamic performance of low Reynolds airfoils with a Reynolds number less than 105 is critical for a variety of applications, including unmanned aerial vehicles, micro air vehicles, and low-speed/high-altitude aircraft. In general, most airfoils with Reynolds numbers less than 106 cannot be considered to have constant lift and drag characteristics. XFLR5 is a tool for analysing low-Reynolds-number airfoils, wings, and aircraft. It comes with XFOIL's direct and inverse analysis tools. The lifting line theory, the vortex lattice approach, and the 3-D panel method are all used to build and analyse wings. The lift coefficient (Cl) and pressure distribution (Cp) are fundamental metrics that determine the behaviour of airfoils. The cornerstone of aerodynamic analysis during aircraft development is pressure distribution data.

Keywords: laminar, Reynolds Number, Turbulence, xfoil , polar ,constant pressure, Aerodynamics.

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1. Introduction

On both the top and bottom surfaces of the airfoil, there is curvature. The difference in curvatures between the top and bottom surfaces of the wing is called camber. The camber of the top surface is more noticeable than the flatness of the bottom surface. The appearance of the airfoil varies as well. The leading edge is rounded and faces forward in flight, while the following edge is narrow and tapering and faces back. The chord line is a reference line that runs from the centre of the leading edge all the way to the end edge of the wing. The mid-cavity line, drawn across the two sides of the propeller, is the other reference line depicted in the illustration. On the midline, all places are evenly spaced [1].

An airship's form is intended to take advantage of AERS responsiveness to specified physical laws. The act of raising the positive or high pressure of the air mass below the wing, and the act of raising the negative pressure of the pressure above the wing, are both produced when the air flows over the wing. Various types of airfoils have different properties. Thousands of propellers have been tested in the wind tunnel and in flight, but none have yet met all of the flight requirements. The form of the wing is determined by the weight, speed, and function of each aircraft [2].

The most effective vane design for optimum lift is a fixed shape with a concave or hollow bottom surface. This sort of wing is inappropriate for high-speed flights because it compromises too much speed while creating lift. Modern planes, on the other hand, feature off-road design attack propellers because a perfectly streamlined propeller with negligible wind resistance may not have enough lift to lift the aircraft off the ground. The form of the intermediate varies according on the demands of the aircraft for which it is intended [3]. During flight, a wing is simply a streamlined object that is positioned in a stream of air.

The change in speed and pressure of air passing through the upper and lower sections of a teardrop wing will be the same on both sides; however, if the teardrop wing is cut in half, a familiar wing section will result; if the wing is tilted upwards, the air blades attacking at an angle from the top surface will be forced to move faster than the air moving along the lower part of the wing [4]. The wing's speed is increased as the pressure above it is reduced.

The pressure principle of Bernoulli is used. As the air velocity at the propeller's tip rises, the pressure drops. The pressure differential between the top and lower surfaces of the wing does not account for the overall lift generated [5]. Airflow from the upper surface of the wing moves down and behind it, creating downflow. This downstream meet flow from the wing's bottom at the posterior border. When Newton's third law is applied, the downward reverse flow provides upward force on the wing. Lift is also produced by pressure circumstances below the vacuum blade [6].

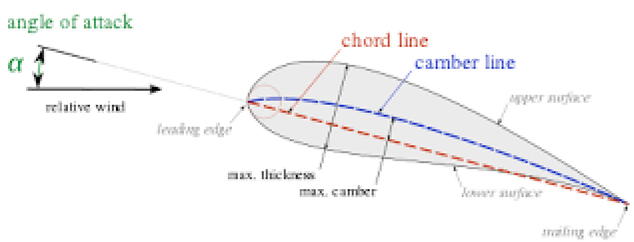


Figure 1: Airfoil[7]

2. UIUC Database:

The coordinates for about 1,600 airfoils are listed here (Version 2.0). The Airfoil Data Site at UIUC provides some background information about the database. The airfoils are alphabetized by the filename (which is generally similar to the airfoil name)[8].

XFOIL is a tool that allows you to construct and analyse subsonic isolated airfoils interactively. It consists of a set of menu-driven routines that execute a variety of helpful tasks, including:

Viscous (or inviscid) analysis of an existing airfoil, allowing

- lift and drag predictions just beyond CLmax
- Is it a forced or unforced transition?
- fixed or varied Reynolds and/or Mach numbers
- fixed or variable Reynolds and/or Mach numbers
- transitional separation bubbles
- restricted trailing edge separation
- Karman-Tsien compressibility correction

Airfoil design and redesign using two approaches involving interactive adjustment of surface speed distributions:

- XFOIL's Mixed-Inverse approach, which is an expansion of the fundamental panel method.
- A complex-mapping formulation is used in the Full-Inverse technique.

Geometry definition allows for interactive customization of geometric characteristics such as

- highpoint position,
- LE radius, and TE thickness camber line.
- maximum thickness and camber,
- Writing and reading of airfoil coordinates and polar save files
- flap deflection explicit contour geometry
- Blending of airfoils
- Plotting of geometry, pressure distributions, and multiple polars
- camber line through loading change specification (via screen cursor)

Xfoil allows the user to analyse existing airfoils for viscous and inviscid properties. With AOA[9], the user may define where a laminar boundary layer changes into a turbulent one, or have the software anticipate the transition point's movement.

XFOIL is a program that allows you to build and analyze subsonic isolation profiles in real time. With the coordinates describing the geometry of the Reynolds and Mach values, XFOIL is able to calculate the pressure distribution on a 2D flight and thus the lift and drag characteristics [10].

The pressure coefficient Cp was calculated using KarmanTsien's compression correction. The drag coefficient CD is calculated using SquireYoung's formula at the trailing edge, not at the end of the wake [11]. SquireYoung's formula, in fact, extrapolates momentum thickness to infinity [12].

```python
def Cp(self, foil):
    remove_str(foil)
    with open(foil, 'r+', errors='ignore') as infile:
        dummy = numpy.genfromtxt(infile, delimiter=[10, 9, 9], dtype=None).tolist()
        cp = [x[2] for x in dummy]
    return cp
```

The drag curve, also known as the drag pole, is the connection between the drag of an aircraft and other elements like as lift, coefficient of lift, angle of attack, and speed. It can be expressed as a graph or as an equation (sometimes called a "polar graph").

For symmetrical propellers, the moment factor around the leading edge is one-quarter of the lift factor and is expressed as cm, le = cl/4 or Momentum factor around the leading edge = Lift factor for the component. In the theoretical formula, the 2D/4 fuselage has a narrow spoiler.

```python
def polar(self, foil):
    remove_str(foil)
    with open(foil, 'r+', errors='ignore') as infile:
        dummy = numpy.genfromtxt(infile, delimiter=[9, 9, 9, 10, 9], dtype=float).tolist()
        alpha = [x[0] for x in dummy]
        cl = [x[1] for x in dummy]
        cd = [x[2] for x in dummy]
        cdp = [x[3] for x in dummy]
        cm = [x[4] for x in dummy]
    alpha = alpha[6:]
    cl = cl[6:]
    cd = cd[6:]
    cdp = cdp[6:]
    cm = cm[6:]
    return alpha, cl, cd, cdp, cm
```

```python
def plotting(x, y, title):
    plt.figure()
    plt.gca().set_aspect('equal', adjustable='box')
    plt.title(title)
    plt.plot(x, y, color='black')
    plt.show
```

2.1. Determining the points on airfoil surface:

The x,y,cp points from the xfoil may first be retrieved by importing the xfoil library. The xfoil database contains data on various shapes of airfoils derived through CFD research. These are influenced by a number of factors, including Reynolds Number, Mach Number, Critical Path, and Critical Angle. The number of points to plot is supplied in the code as 100. The airfoil form is depicted by merging the 100 points from the data.

```python
from Super_xfoil import xfoil
from zipfile import ZipFile
import os
import random

database_path = os.getcwd() + r'\\Super_xfoil\Database.zip'
code_path = os.getcwd()

z = ZipFile(database_path, 'r')
data_base = z.namelist()

foil = random.choice(data_base)
z.extract(foil, code_path)
foil = foil[0:-4]

pts = 100  # number of points to create
Re = 200000  # Reynolds number
Mach = 0  # Mach number
N_crit = 8.1  # Critical boundary layer interaction
Alpha = 0  # Angle of attack


(new_x, new_y) = xfoil.new_foil(foil, pts)

cp = xfoil.cp_fun(foil, Re, Mach, N_crit, Alpha)

print(cp)
xfoil.plotting(new_x, new_y, foil)
```

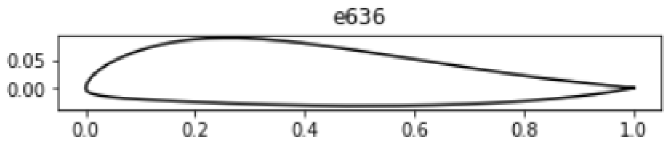
3. Results and Discussions:

The Cp values are derived using the x and y values as well as the Reynolds number, and the critical angle is calculated for the e379 file. The polar cl, cd, cdp, cm, and alpha are likewise derived from the xfoil data by obtaining the polar cl, cd, cdp, cm, and alpha. The zipped database is made up of '.dat' files that are organised alphabetically and include the values for each category.

e379		
Alfa = 0.00000		
Re = 200000.000		
Xflap,Yflap = 0.000000 0.000		
# x	y	cp
1.00000	-0.00000	-0.07218
0.99975	0.00002	*********
0.99899	0.00007	-0.58943
0.99774	0.00015	0.99974
0.99598	0.00028	-1.98762
0.99372	0.00044	0.72721
0.99097	0.00064	-1.19631
0.98772	0.00089	-0.03231
0.98398	0.00120	0.27943
0.97975	0.00157	-0.21147
0.97504	0.00201	0.19133
0.96985	0.00251	-0.13814
0.96419	0.00309	0.13036
0.95806	0.00376	0.18880
0.95147	0.00450	-0.15861
0.94442	0.00533	0.11308
0.93693	0.00625	-0.06062
0.92900	0.00726	0.15517
0.92063	0.00834	-0.24697
0.91185	0.00951	-0.05716
0.90264	0.01076	-0.03172
0.89304	0.01209	-0.02988
0.88303	0.01349	0.06964
0.87264	0.01498	-0.00381
0.86188	0.01653	-0.02050
0.85075	0.01817	0.08951
0.83927	0.01987	-0.13790
0.82745	0.02165	0.16012
0.81529	0.02348	-0.09030
0.80282	0.02539	-0.13562
0.79005	0.02735	0.01023
0.77698	0.02936	0.01203
0.76363	0.03144	-0.10825
0.75002	0.03356	-0.01043
0.73616	0.03573	-0.23016
0.72206	0.03795	-0.40626
0.70773	0.04020	0.01333
0.69320	0.04249	-0.28840
0.67847	0.04480	0.06396
0.66356	0.04715	-0.45053
0.64849	0.04950	-0.12349
0.63327	0.05187	-0.56329
0.61791	0.05425	-0.15289
0.60244	0.05662	-0.53379
0.58686	0.05899	-0.40087
0.57120	0.06135	-0.15973
0.55546	0.06368	-0.22763
0.53967	0.06597	-0.55135
0.52383	0.06823	-0.31337
0.50798	0.07043	-0.64342
0.49211	0.07254	-0.95606
0.47626	0.07460	-0.07282
0.46042	0.07655	-0.62633
0.44463	0.07833	-0.28693
0.42889	0.08000	-0.55913
0.41323	0.08150	-0.65967
0.39765	0.08278	-0.49525
0.38218	0.08393	-1.04927
0.36682	0.08486	-0.51357
0.35160	0.08549	-0.84213
0.33653	0.08602	-0.77843
0.32162	0.08630	-0.66225
0.30689	0.08626	-0.64337
0.29236	0.08605	-1.59481
0.27803	0.08573	-0.44159
0.26393	0.08518	-1.15297
0.25007	0.08426	-0.48158
0.23646	0.08319	-1.04216
0.22311	0.08193	-1.01993
0.21004	0.08042	-0.38903
0.19727	0.07879	-1.38614
0.18480	0.07698	-0.24013
0.17264	0.07500	-1.36003
0.16082	0.07291	-0.58817
0.14934	0.07068	-0.33409
0.13821	0.06832	-1.02766
0.12745	0.06587	-0.29491
0.11706	0.06329	-1.87023
0.10705	0.06062	0.19939
0.09745	0.05786	-0.96659
0.08824	0.05501	-0.15707
0.07946	0.05210	-0.27815
0.07109	0.04912	-1.15611
0.06316	0.04607	-0.15145
0.05567	0.04298	-0.48323
0.04862	0.03986	-0.31711
0.04203	0.03670	0.14750
0.03590	0.03353	-0.90428
0.03024	0.03036	0.23073
0.02505	0.02718	0.25363
0.02034	0.02402	-0.13372
0.01611	0.02089	0.40326
0.01237	0.01780	0.82900
0.00912	0.01484	-0.70219
0.00637	0.01191	0.99998
0.00411	0.00915	0.20368
0.00235	0.00658	-0.96010
0.00110	0.00383	-0.50161
0.00034	0.00060	-2.57153
0.00009	-0.00066	-33.32159
0.00034	-0.00073	0.38389
0.00110	-0.00089	-4.16378
0.00235	-0.00105	-7.67183
0.00411	-0.00092	-2.87592
0.00637	-0.00017	-3.54959
0.00912	0.00121	-0.42136
0.01237	0.00270	-0.19482
0.01611	0.00424	-0.72070
0.02034	0.00600	0.08918
0.02505	0.00798	-1.84014
0.03024	0.01016	-0.22417
0.03590	0.01255	0.60320
0.04203	0.01513	-0.60022
0.04862	0.01786	0.44339
0.05567	0.02071	-0.30751
0.06316	0.02371	-0.18735
0.07109	0.02681	0.64229
0.07946	0.02997	-0.04009
0.08824	0.03320	0.02078
0.09745	0.03650	0.42301
0.10705	0.03981	-0.09787
0.11706	0.04306	0.80034
0.12745	0.04624	0.09421
0.13821	0.04943	0.62424
0.14934	0.05252	0.34971
0.16082	0.05541	0.20761
0.17264	0.05805	0.53403
0.18480	0.06060	0.42195
0.19727	0.06299	0.35962
0.21004	0.06513	0.48886
0.22311	0.06692	0.35179
0.23646	0.06856	0.77449
0.25007	0.07005	0.15133
0.26393	0.07131	0.42518
0.27803	0.07221	0.34074
0.29236	0.07285	0.47549
0.30689	0.07343	0.64540
0.32162	0.07387	0.20786
0.33653	0.07410	0.58862
0.35160	0.07406	0.40861
0.36682	0.07378	0.13977
0.38218	0.07333	0.59148
0.39765	0.07278	0.29766
0.41323	0.07215	0.45459
0.42889	0.07124	0.47201
0.44463	0.07011	0.53810
0.46042	0.06885	0.43798
0.47626	0.06753	0.56633
0.49211	0.06603	-0.01142
0.50798	0.06439	0.37415
0.52383	0.06266	0.23003
0.53967	0.06088	0.30946
0.55546	0.05900	0.40539
0.57120	0.05704	0.43432
0.58686	0.05503	0.26379
0.60244	0.05298	0.03872
0.61791	0.05087	0.51462
0.63327	0.04872	0.01796
0.64849	0.04657	0.28884
0.66356	0.04440	0.02463
0.67847	0.04221	0.48816
0.69320	0.04003	0.20243
0.70773	0.03786	0.30009
0.72206	0.03568	0.05960
0.73616	0.03352	0.07843
0.75002	0.03140	0.24212
0.76363	0.02931	0.24450
0.77698	0.02725	0.27949
0.79005	0.02524	0.22504
0.80282	0.02327	0.02221
0.81529	0.02135	0.20518
0.82745	0.01949	0.16935
0.83927	0.01770	0.11703
0.85075	0.01597	0.22259
0.86188	0.01432	0.11114
0.87264	0.01275	0.03640
0.88303	0.01125	0.14380
0.89304	0.00983	0.02448
0.90264	0.00850	-0.01663
0.91185	0.00727	-0.09097
0.92063	0.00612	-0.19497
0.92900	0.00507	0.05418
0.93693	0.00414	-0.02900
0.94442	0.00328	-0.09765
0.95147	0.00254	-0.01252
0.95806	0.00195	-0.23291
0.96419	0.00146	0.15055
0.96985	0.00107	-0.09756
0.97504	0.00079	0.07314
0.97975	0.00060	-0.29105
0.98398	0.00046	0.23622
0.98772	0.00035	-0.10100
0.99097	0.00027	-0.74270
0.99372	0.00021	0.56711
0.99598	0.00015	-1.40394
0.99774	0.00010	0.98585
0.99899	0.00005	-0.49597
0.99975	0.00001	*********
1.00000	0.00000	-0.07218

The parameters of Reynolds Number, Mach Number, Critical Path, and Critical Angle are used to produce the points for charting the xfoil in Python. Plotting 100 points on the surface of the airfoil is derived from the data. The airfoil shape was formed by joining the points. EPPLER 636 AIRFOIL is the name of an airfoil with a chord length of 100mm and a radius of 0mm, as well as a thickness of 100%, an origin of 0%, and a pitch of 0°.

[0.48573, 0.42457, 0.35307, 0.36111, 0.34448, 0.33349, 0.30631, 0.29045, 0.27779, 0.25582, 0.23003, 0.20241, 0.19218, 0.17922, 0.16483, 0.15444, 0.14803, 0.1474, 0.14586, 0.13917, 0.13377, 0.13244, 0.12718, 0.12124, 0.11434, 0.10651, 0.10049, 0.09235, 0.08452, 0.07576, 0.06678, 0.05714, 0.04487, 0.03623, 0.02483, 0.0132, 0.00194, -0.01206, -0.02805, -0.04388, -0.05745, -0.07457, -0.09092, -0.1075, -0.12558, -0.14649, -0.16568, -0.18445, -0.20935, -0.23118, -0.2522, -0.2781, -0.30441, -0.32515, -0.35631, -0.38707, -0.40806, -0.44171, -0.47725, -0.49242, -0.53466, -0.5868, -0.57825, -0.63421, -0.70323, -0.67913, -0.67051, -0.69578, -0.79432, -0.7649, -0.72243, -0.77546, -0.73463, -0.7105, -0.7161, -0.66493, -0.63198, -0.61093, -0.55028, -0.51358, -0.47106, -0.41587, -0.36271, -0.30403, -0.2298, -0.1616, -0.07993, 0.01037, 0.10398, 0.20285, 0.32268, 0.4373, 0.5581, 0.70261, 0.80203, 0.91004, 0.99209, 0.96199, 0.77824, -0.2695, 0.34822, 0.17304, -1.07019, -1.20072, -1.03273, -0.77689, -0.53941, -0.55828, -0.50741, -0.44809, -0.40401, -0.37729, -0.34208, -0.31344, -0.28859, -0.26395, -0.24626, -0.22489, -0.20751, -0.19162, -0.17693, -0.16852, -0.16003, -0.14912, -0.14701, -0.14037, -0.13835, -0.13348, -0.13601, -0.13236, -0.1347, -0.13227, -0.1349, -0.13584, -0.13787, -0.13659, -0.14015, -0.14067, -0.1403, -0.14378, -0.14425, -0.14277, -0.14468, -0.14786, -0.14523, -0.14568, -0.15128, -0.15516, -0.14513, -0.14321, -0.14483, -0.15776, -0.15744, -0.15143, -0.15151, -0.15676, -0.15351, -0.15382, -0.15761, -0.15566, -0.15437, -0.15756, -0.15554, -0.15272, -0.15355, -0.15102, -0.14621, -0.14203, -0.14101, -0.13457, -0.12853, -0.12453, -0.11806, -0.10882, -0.10329, -0.09693, -0.08601, -0.07657, -0.0698, -0.05989, -0.04783, -0.03748, -0.02948, -0.0186, -0.00582, 0.01817, 0.04812, 0.06846, 0.09993, 0.15448, 0.19118, 0.21855, 0.24463, 0.28205, 0.31119, 0.33606, 0.3428, 0.39343, 0.48573]



4. Conclusion:

This study's XFOIL wing design approach has shown to be a potent and effective tool for developing subcritical airfoils, particularly those with low Reynolds numbers. Several visually oriented processes execute different analysis, inversion, and geometry change operations on a common airfoil model and may be readily accessed by the designer with total flexibility from a single menu structure. Any adjustments to the blade are made since the entire computational model incorporates all of the fundamental physical phenomena that determine propeller performance (stability, decoupling, bubble loss, and so on). Fans that have the ability to influence performance may be counted on for tangible impacts rather than digital ones. As a consequence, technology enables designers to swiftly test new design methodologies and develop a design philosophy for each project.

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