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MATLAB-Enhanced Wing Design and Aerodynamic Modeling

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ABSTRACT

This study presents an innovative MATLAB-based methodology for the efficient design of intricate wing surfaces, offering an alternative to the cumbersome and skill-intensive CAD software process. This method facilitates the automated creation of '.stl' files for wings of various dimensions, allowing for the integration of customizable geometrical features such as twist, sweep, and taper. It uniquely enables the incorporation of both geometric and aerodynamic twists, with the capability to produce varied root and tip airfoils internally. The software is compatible with multiple NACA airfoil series, extending support to CST and PARSEC parameterization methods. Furthermore, this research integrates the horizontal wind model (HWM) routine from the U.S. Naval Research Laboratory into MATLAB, enabling the computation of wind speed using location- and time-specific data. This data is utilized in conjunction with the XFOIL solver embedded in our framework to conduct initial aerodynamic assessments employing the vortex panel method. The generated wing surface '.stl' files are designed to be seamlessly integrated with CFD software for comprehensive aerodynamic analysis.

Keywords: Wing Design, MATLAB, Aerodynamics, Wing Generation

NOMENCLATURE

AR	Aspect ratio	--
C_L	Lift coefficient	--
C_D	Drag coefficient	--
α	Angle of attack	(^{o})
Λ	Sweep Angle	(^{o})
λ	Taper ratio	--
a	Speed of Sound	[m/s]
ρ	Density of air	[kg/m^3]
μ	Dynamic Viscosity	[Ns/m^2]
Re	Reynold Number	--
x	Normalized Chord Position	--
t	Maximum Thickness as a fraction of airfoil chord	--
m	Maximum Camber	--
p	Location of maximum airfoil camber	--
y_c	Mean camber line	--

1. INTRODUCTION

From Mini UAVs to commercial aircraft manufacture, wing selection and optimization processes require accurate predictions of the aerodynamic characteristics of the wing structure. Analytical and experimental methods are rarely implemented due to issues with complexity and cost, respectively [1]. Subsequently, numerical methods or high-fidelity computations are usually carried out to accurately model the wing surface's aerodynamic behavior. Aerodynamic analysis through numerical CFD methods requires the surface geometry of the structure to be analyzed to be imported via a '.stl' file. The generation of this '.stl' file involves the use of CAD software in a skill-intensive and time-consuming process. This process is further complicated by the implementation of standard wing geometric features such as twist, sweep, and taper into the wing model. This work concentrates on streamlining the wing surface model generation process using MATLAB scripts, hence avoiding the use of CAD software.

Prior efforts have addressed the use of MATLAB for the wing and other 3D structure generation processes. Eshagi and Nooraeefar [2] developed 'WingMesh', a MATLAB-based application designed to rapidly automatically model complicated insect wing structures. Qian [3] developed a MATLAB-based environment that can generate a finite-element model of an aircraft Wingbox, incorporating all the major wing structural components such as spars, ribs, stringers, and skin. Bibal and Siddique [4] developed an automated MATLAB application to determine the approximate bending and torsional stresses on internal wing load-carrying structural members.

2. OBJECTIVES

Our work presents a MATLAB environment that can generate a '.stl' file of complicated wing surface models incorporating features such as aerodynamic and geometric twist, sweep, and taper into the wing geometry. This process, which would normally take hours in standard computational modeling software, can be completed in mere seconds. One novel approach presented in our work is the implementation of an embedded XFOIL vortex panel method solver within the

MATLAB script, which automatically performs a preliminary aerodynamic analysis of the generated wing surface to obtain the forces and moments along with their respective coefficients. While the final aerodynamic parameters for the later stages of the design process need to be obtained from accurate CFD results, this initial analysis drastically streamlines the wing selection process as there is no need to run high-fidelity resource-intensive computational simulations for each generated wing configuration to obtain approximate initial aerodynamic parameters.

3. MATERIALS AND METHODOLOGY

1. Standardizing Airfoil Input

As the dataset of airfoils available today is extremely large, it was decided to standardize the airfoil input format for the wing generation framework. To make it as comprehensive as possible, along with the NACA 4,5 and 6-digit series airfoils, the provision to input CST and PARSEC airfoil coefficients has also been incorporated.

3.1. NACA Series Airfoils

NACA airfoils, as developed by the National Advisory Committee for Aeronautics, are some of the most common and widely used airfoils across aerospace applications. The shape of these airfoils can be described using a series of digits following the word 'NACA'. These digits can be entered into the following equations to determine either the half-thickness or the camber line of the airfoil.

3.1.1. Symmetric NACA 4 Digit Series

These airfoils have the format 'NACA 00xx', where xx represents the maximum thickness to chord ratio in percentage form. The half-thickness of the airfoil is only a function of the normalized position along the chord and the maximum thickness fraction. The equation of these airfoils is given by Eq. 1.

$$y_t = 5t\left[0.2969\sqrt{x} - 0.1260x - 0.3516x^2 + 0.2843x^3 - 0.1015x^4\right] \quad (1)$$

3.1.2. Cambered NACA 4 Digit Series

These airfoils are given in the form 'NACA mpxx' where m represents the maximum camber and p is the position of the maximum camber in terms of the normalized airfoil chord distance. The equation of these airfoils is given by Eq. 2.

$$y_c = \begin{cases} \frac{m}{p^2}(2px - x^2), & 0 \le x \le p \\ \frac{m}{(1-p)^2}((1-2p) + 2px - x^2), & p \le x \le 1 \end{cases}$$

$$X_U = x - y_t sin(\theta) \qquad X_L = x + y_t sin(\theta)$$

$$Y_U = y_c + y_t cos(\theta) \qquad Y_L = y_c - y_t cos(\theta) \quad (2)$$

$$\theta = arctan\left(\frac{dy_c}{dx}\right)$$

$$\frac{dy_c}{dx} = \begin{cases} \frac{2m}{p^2}(p - x), & 0 \le x \le p \\ \frac{2m}{(1-p)^2}(p - x), & p \le x \le 1 \end{cases}$$

3.1.3. Cambered NACA 4 Digit Series

Cited The NACA 5-digit series airfoils are given in the format 'NACA LPSTT'.

- L: Representing the theoretical optimal lift coefficient at the ideal angle of attack
- P: X-coordinate of point of maximum camber of airfoil
- S: Indicates whether the airfoil is simple or reflex
- TT: Represents maximum thickness in the percentage of chord

The equation of these airfoils is given by Eq. 3.

$$y_c = \begin{cases} \frac{k_1}{6}(x^3 - 3rx^2 + r^2(3-r)x), & 0 \le x \le r \\ \frac{k_1 r^3}{6}(1 - x), & r \le x \le 1 \end{cases} \quad (3)$$

The constants 'r' and 'k_1' in Eq. 3 are determined based on the maximum camber location and the desired lift coefficient, respectively.

3.1.4. PARSEC

The PARSEC method uses 11 basic parameters to represent the geometry of any airfoil completely. These parameters include the trailing edge thickness (ΔZ_{TE}), wedge angle (β_{TE}), coordinate (Z_{TE}) and direction (α_{TE}); the upper and lower crest locations ((X_{UP}, Z_{UP}) and (X_{LO}, Z_{LO})), the upper and lower curvatures (Z_{xxLO}, Z_{xxUP}) and the leading-edge radius (r_{LE}). The PARSEC parameters are defined in Fig. 1 [5].

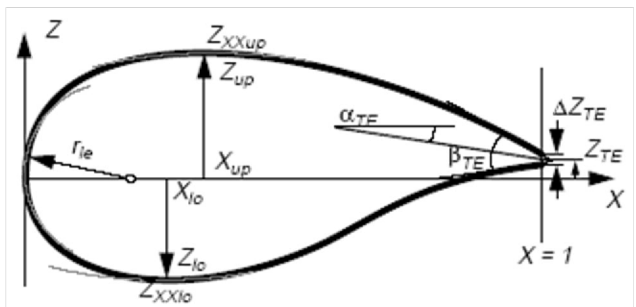

Figure 1: PARSEC Parameters

While the PARSEC method of parametrization can effectively control the curvature of the airfoil's upper and lower surfaces, it does not provide sufficient authority over the airfoil's trailing edge [6]. This is because this method joins the upper and lower maximum points with the trailing edge through smooth curves. As this might compromise the highly critical regions around the trailing edge where important flow phenomena take place, other parametrization techniques, such as the Sobieczky method, have been developed, improving the PARSEC method by giving the trailing edge a concave shape. The PARSEC method uses a linear combination of shape functions to describe the airfoil, as shown in Eq. 4.

$$Z_k = \sum_{n=1}^{6} a_{n,k} X_k^{\frac{n-1}{2}} \quad (4)$$

3.1.5. Class Shape Transformation (CST)

The Class Shape Transformation (CST) parametrization technique was developed by Brenda Kulfan [7], an aerodynamic engineer at Boeing, in 2008. It incorporates a geometric class/shape function transformation technique. A typical airfoil can be represented mathematically using Eq. 5.

$$\zeta(\psi) = \sqrt{\psi}(1-\psi)\sum_{i=0}^{N} A_i \psi^i + \psi\zeta_T \qquad (5)$$

where $\psi = x/c$, $\zeta = z/c$ and $\zeta_T = \Delta Z_{TE}/c$. The term $\sqrt{\psi}$ corresponds to the round nose of the aerofoil. While the term $(1 - \psi)$ corresponds to the sharp trailing edge of the aerofoil. The term $\psi\zeta_T$ is responsible for controlling the thickness of the trailing edge, and the summation term is a general function that defines the airfoil shape between the round nose and the sharp trailing edge. The CST parameters are defined in Fig. 2 [8].

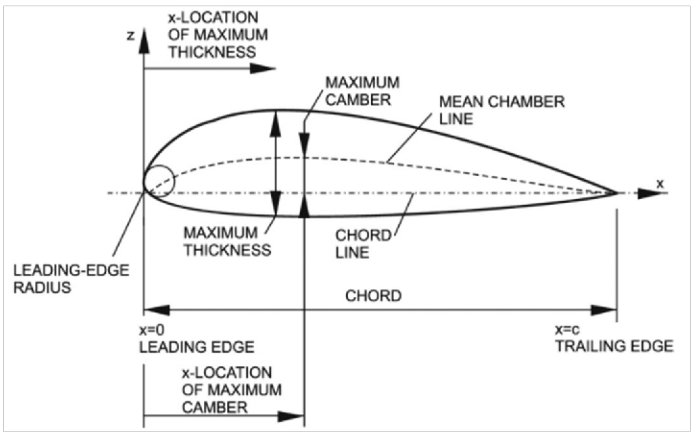

Figure 2: CST Parameters

The airfoil is defined using the n^{th}-order Bernstein polynomials, given by Eq. 6.

$$S_{r,n}(x) = K_{r,n}x^r(1-x)^{n-r} \qquad (6)$$

The binomial coefficients are given by Eq. 7.

$$K_{r,n} = \binom{n}{r} = \frac{n!}{r!(n-r)!} \qquad (7)$$

The first term of the polynomial defines the leading-edge radius, while the last term is the boat-tail angle. The other terms in between are the "shaping terms". We use a Bernstein polynomial of the 8^{th} order, leading to 8 weighted coefficients being obtained at the end of the CST process. The drag predictions and pressure distribution for a Bernstein's polynomial of the 8^{th} order agree exactly with experimental data. Hence, a Bernstein polynomial of the 8^{th} order was chosen.

3.2. Wing Generation

The wing geometry is generated based on Bezier surfaces as formulated by Sóbester and Forrester [9]. This formulation was implemented in MATLAB. This relies on the definition of the spanwise vector $\boldsymbol{\varepsilon}$. The vector $\boldsymbol{\varepsilon}$ defines the spanwise distance along which the airfoils are to be generated. The curvilinear axis is fixed to the leading edge (LE) of the wing, and its scale is normalized. This implies that the values of $\boldsymbol{\varepsilon}$ would be 0 and 1 at the root and tip sections of the wing. Using the vector $\boldsymbol{\varepsilon}$, the spanwise distribution of various properties like twist, chord (taper), and sweep can be controlled. This particular method lends the ability to generate complex wing geometries with ease.

3.3. Aerodynamic Analysis

3.3.1. XFOIL

XFOIL [10] is a program that uses a linear-vorticity panel method to design and analyze subsonic isolated airfoils. It was designed to accurately predict aerodynamic coefficients of low Reynolds Number flows while being computationally inexpensive. While XFOIL is capable of performing aerodynamic analyses more efficiently than alternative methods due to its usage of idealized computational models, this also leads to some inaccuracies. XFOIL tends to overestimate lift coefficients while underestimating drag coefficients. However, it is deemed suitable for a preliminary aerodynamic analysis.

3.3.2. Wing Analysis

To estimate the aerodynamic properties of the wing, the approach formulated by Matthew Brown [11] was used. This method uses the cranked-wing approach to define the arbitrary shapes of the platform. Each panel is defined by four distinct characteristics, namely, the airfoil section used, the length of the panel root and tip chords, the quarter chord location of the root and tip panel airfoil sections, and the incidence angle. Moreover, each point in the panel has a characteristic sweep angle. The incidence angles of various airfoil sections and the airfoil properties are assumed to vary linearly along the panel.

XFOIL was used to resolve the chordwise loading of the airfoil sections of the wing. Once the chordwise loading is obtained, the spanwise loading can be estimated using an appropriate lifting line theory. In our study, the determination of distributed aerodynamic loads in compressible subsonic flow [11], the method of lifting line theory described by W.F. Phillips [12], is utilized. This method is primarily based on the 3D vortex lifting law. In essence, a series of discrete horseshoe vortices are placed along the wing surface. This is followed by the calculation of lift using the 3D vortex lifting law. A significant advantage of this method is its ability to accurately predict the lift for complex geometries with twist and sweep angles. Further, the aerodynamic effects of airfoil thickness, control surfaces, and flaps can also be modeled effectively using the chordwise airfoil data.

3.3.3. Operating Conditions

The atmospheric data is obtained using the International Standard Atmosphere (ISA) model [13]. For a given altitude (user input), the output consists of pressure, temperature, density, and the speed of sound, as listed in the ISA table. Wind speed data is obtained via the implementation of the U.S. Naval Research Laboratory Horizontal Wind Model (HWM) [14,15] routine using built-in MATLAB functions. The output consists of the meridional and zonal wind components for a specific position in space (latitude and longitude) for a given day and time of the year.

4. RESULTS

4.1. Wings with Different Geometric Properties

This section is a compilation of different wings generated using the framework developed. Fig. 3 represents a wing with a NACA 2412 constant airfoil section.

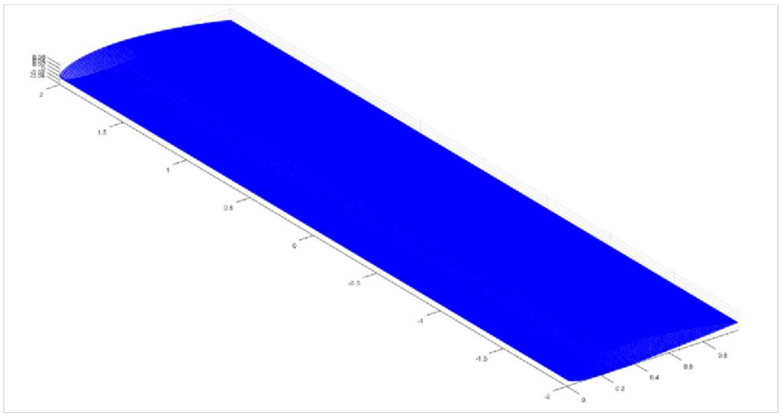

Figure 3: Constant Airfoil Section NACA 2412 Wing

Fig. 4 represents a wing with an aerodynamic twist with varying root and tip airfoils. One interesting observation here is that the wing is twisted with respect to the leading edge (the LE is fixed). This is due to the curvilinear axis being fixed to the LE of the wing.

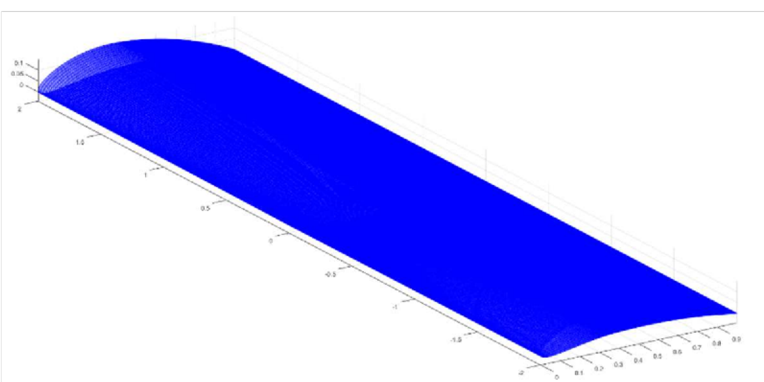

Figure 4: Wing with Aerodynamic Twist Root - NACA 2412, Tip – NACA 9412

Fig. 5(a) and Fig 5(b) represent wings with washin and washout geometric twists. Fig. 6 represents a wing with a taper ratio (λ) of 0.5. The taper ratio of a wing is given by the ratio of the tip chord to the root chord of the wing. Similarly, Fig. 7 depicts a wing with a leading edge sweep angle (Λ) of 10°.

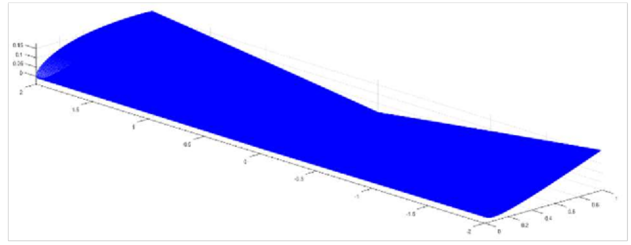

Figure 5(a): Wing with Washin Angle of 10°

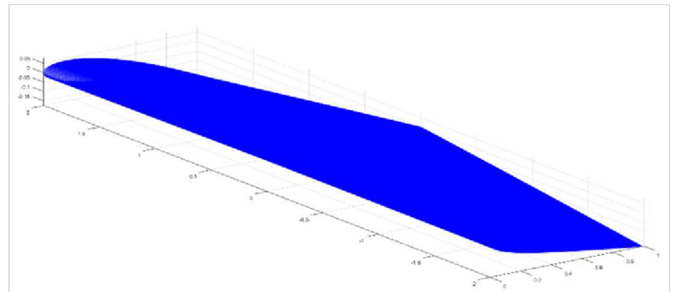

Figure 5(b): Wing with Washout Angle of 10°

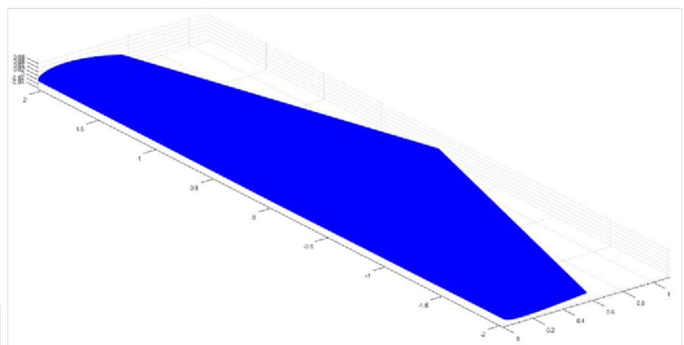

Figure 6: Wing with Taper Ratio of λ = 0.5

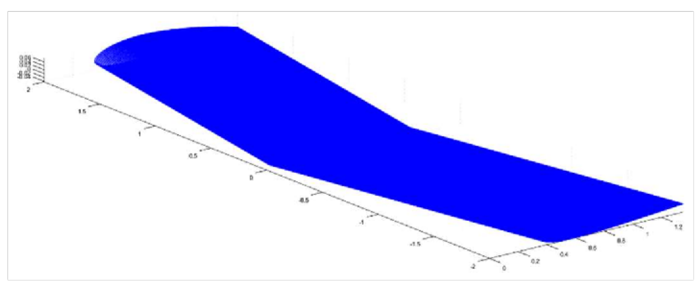

Figure 7: Wing with Leading Edge Sweep Angle (Λ) = 10°

4.2. Preliminary Aerodynamic Analysis

The analysis of the constant airfoil section (NACA 2412) wing (Fig. 3) was conducted using the XFOIL-based aerodynamic analyses routine for the following operating conditions (Table 1) generated using the International Standard Atmosphere

(ISA) model and the US Naval Research Laboratory Horizontal Wind model.

Table 1 Operating Conditions

Parameter	**Value**
Altitude	11 km
Latitude	19.07°
Longitude	72.88°
Wind Speed	4.68 m/s
Dynamic Viscosity (μ)	1.42×10^{-5}
Density (ρ)	0.3639 kg/m³
Reynolds Number (Re)	1.19×10^5
Speed of Sound	295.06 m/s
Angle of Attack (α)	4°

The results of the analysis in the fps unit system are elucidated in Fig. 8. The input parameters are suitably converted to their imperial counterparts.

```
****************** Aerodynamic Analyses **************
Loading...        Wing Data.xlsx
Calculating...    Section Aerodynamics
Calculating...    Planform Variables
Solving...        Type 1 Analysis
Converged...      Type 1 Analysis.
                  CPU Time: 0.224 s
==================== Wing Geometry ====================
Span, b                 = 13.1201 ft
Reference Area, Sref    = 43.0306 sq ft
Panel Area, S           = 43.0306 sq ft
Aspect Ratio, AR        = 4.0003 ~
Mean Aero Chord, mac    = 3.2797 ft
=============== Type 1 Analysis Summary ===============
Forces:
 Fx   = -0.033 lbs,  Fy = 0.000 lbs,  Fz = -1.558 lbs
 Lift = 1.556 lbs,  Drag = 0.076 lbs,  L/D = 20.490

Moments:
 Roll  = -0.000 ft-lbs
 Pitch = -60.391 ft-lbs
 Yaw   = 0.000 ft-lbs

Coefficients:
 CL = 0.4338,  CD = 0.0212,  CDi = 0.0153  CY = 0.0000
 Cl = -0.0000, Cm = -5.1339, Cn  = 0.0000
=======================================================
```

Figure 8: Results of the Aerodynamic Analysis

4.3. A Sample Conceptual Design Study

This section elucidates a sample conceptual design study from wing generation to its preliminary aerodynamic analysis. Assuming a wing with a constant NACA 4412 airfoil section, with a LE sweep angle of 15° and a taper ratio of 0.75. The wing generated is depicted in Fig. 9.

Assuming the wing to be flying over Mumbai at an altitude of 12 km and the associated operating conditions as given in Table 2, the results of the preliminary aerodynamic analyses conducted using the framework developed in MATLAB coupled with the vortex panel method implemented in XFOIL are given in Fig. 10.

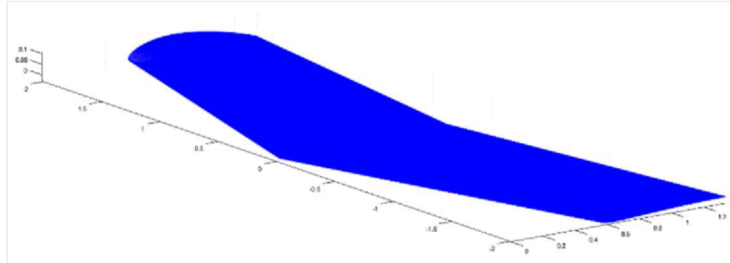

Figure 9: Wing with constant NACA 4412 Airfoil Section with LE Sweep Angle of 15° and a Taper Ratio of 0.75

Table 2 Operating Conditions

Parameter	**Value**
Altitude	12 km
Latitude	19.07°
Longitude	72.88°
Wind Speed	4.68 m/s
Dynamic Viscosity (μ)	1.42×10^{-5}
Density (ρ)	0.3639 kg/m³
Reynolds Number (Re)	1.19×10^5
Speed of Sound	295.06 m/s
Angle of Attack (α)	4°

```
****************** Aerodynamic Analyses ***************
Loading...        Wing Data.xlsx
Calculating...    Section Aerodynamics
Calculating...    Planform Variables
Solving...        Type 1 Analysis
Converged...      Type 1 Analysis.
                  CPU Time: 0.337 s
==================== Wing Geometry =====================
Span, b                 = 13.2016 ft
Reference Area, Sref    = 40.1354 sq ft
Panel Area, S           = 40.1354 sq ft
Aspect Ratio, AR        = 4.3424 ~
Mean Aero Chord, mac    = 3.061 ft
=============== Type 1 Analysis Summary ================
Forces:
 Fx   = -0.076 lbs,  Fy = -0.000 lbs,  Fz = -3.408 lbs
 Lift = 3.405 lbs,  Drag = 0.162 lbs,  L/D = 21.016

Moments:
 Roll  = 0.000 ft-lbs
 Pitch = -129.862 ft-lbs
 Yaw   = 0.000 ft-lbs

Coefficients:
 CL = 0.4385,  CD = 0.0209,  CDi = 0.0150  CY = -0.0000
 Cl = 0.0000,  Cm = -5.4632, Cn  = 0.0000
========================================================
```

Figure 10: Results of the Aerodynamic Analysis

5. CONCLUSIONS

An innovative MATLAB-based method was developed to efficiently design intricate wing surfaces, thus offering an alternative to cumbersome and skill-intensive CAD software. Further, this method facilitates the automated creation of '.stl' files for wings of various dimensions, integrating customizable geometrical features such as twist, taper, and sweep. The method developed due to its computationally inexpensive nature is best suited for the initial stages of conceptual wing design.

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