

AIRCRAFT WING AND AERODYNAMIC ANALYSIS USING INTEGRATED MATLAB AND STAAD.Pro MODELS

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Aircrafts have evolved significantly throughout the years, in large part to advances in technology. As technology grows, analytical models of aircrafts have become more accurate and play a more significant role in the design process. This paper examines the integration of commonly used mathematical and structural analysis software to conduct parametric studies on aircraft wing design and aerodynamics analysis for class project. The forces considered in this study include engine, fuel, and lift forces. The engine force is evaluated as a constant point load throughout the duration of the flight, while the fuel weight is changed throughout the flight as it was being consumed. The MATLAB algorithm is set up to analyze one fuel force input which is distributed evenly over the wing at a location where the fuel tank is located. The lift force is estimated using the Schrenk Approximation and the MATLAB algorithm is then used to distribute the lift force over the area of the wing. The methodology developed in this paper shows the potential of efficiently use non-aerodynamic software to produce a detailed preliminary structural analysis of an aircraft wing and the ability to use it in classroom projects.

Keywords: Aerodynamics, Lift, Drag, Wing design, Structural analysis.

1 WING DESIGN AND AERODYNAMIC ANALYSIS

When designing an aircraft in the early stages, the initial design factors are dependent on the flight phases of the aircraft. This study analyzes a tanker aircraft with the purpose of refueling other military aircrafts in mid-flight with a cruise altitude of 35,000ft, cruise Mach of 0.85 and a range of 4,400 nm. These mission requirements are similar to current tankers, but will allow for higher subsonic Mach number and larger range (Corke 2003). These upgrades are needed because the jets they refuel are constantly being upgraded and drive the entire design process. Because the mission of the aircraft requires a large range, the main objective of the design process is to make the cruise phase as efficient as possible. Since the flight at the cruise phase is level and the velocity is constant, all the forces are in balance. This can be described by the equilibrium equations given in Eq. (1) and Eq. (2), where T is thrust, D is drag, L is lift, and W is weight.

$$\sum F_x = T - D = 0 \rightarrow \text{Thrust} = \text{Drag} \quad (1)$$

$$\sum F_Y = L - W = 0 \rightarrow Lift = Weight \quad (2)$$

Weight estimation is the first step in the preliminary design process of an aircraft. The three factors to be considered are structural weight, payload, and fuel. The structural weight is referred to as empty weight and includes structural materials, propulsion system, avionics, and other instruments within the aircraft. Empty weight is greatly influenced by the structural material used to construct the aircraft. Since the structural material has not been determined at this stage, it is estimated to be 36% of the total weight of the aircraft for the type of mission selected based on available data. The payload weight of an aircraft consists of expendable and non-expendable. Non-expendable payload is the load added to the flight that does not change throughout the flight. This may include crew, passengers, baggage, etc. The expendable payloads are loads that are changed or used during the flight. This would for example include weapons that may be dropped from military aircrafts. In the application of this analysis, the expendable payload is very large because the mission of the aircraft is to refuel fighter jets. The fuel carried on the aircraft to fuel other jets (separate from its own fuel) is considered to be expendable payload. The calculation of the fuel weight for this aircraft is an iterative process. Because fuel is always being consumed throughout the duration of the flight, the weight estimation is determined by the amount of fuel used during each flight phase. The formulae for predicting fuel usage for each phase are based on historical data of aircrafts. Eq. (3) represents the calculation of fuel weight, taking the initial weight (W_i) and the final weight (W_f) of the flight phases to determine the percentage of fuel consumed during that phase. The amount of fuel used during the entire flight can be calculated.

$$\left(\frac{W_f}{W_i} \right)_{fuel} = \frac{W_2}{W_1} \cdot \frac{W_3}{W_2} \cdots \frac{W_n}{W_{n-1}} \quad (3)$$

The structural weight and non-expendable payload weight remains unchanged throughout the flight mission. An Excel spreadsheet is used to determine the weight of the aircraft during each flight phase. This weight is a function of the fuel consumed and the timing of the expendable payload deployment. An initial take-off weight is assumed and the spreadsheet uses an iterative process to calculate the fuel, structure, and payload weights. The next step in the design process is to calculate the wing loading. Since the proposed aircraft is a transport aircraft, the design is driven to obtain an efficient cruise. The wing loading at cruise is used to calculate the required surface area of the wing. Eq. (4) shows the wing loading at the cruise condition as a function of the dynamic pressure and lift coefficient, where W is the weight of the aircraft at cruise, S is the surface area of the wing, q is the dynamic pressure, and C_L is the lift coefficient (Raymer 1992).

$$\left(\frac{W}{S} \right)_{cruise} = q C_L \quad (4)$$

The surface area is used to determine the platform dimensions of the wing, along with historical data for taper ratios and sweep angles for aircrafts with similar missions.

Since this aircraft is designed for long range flights, there are two main design considerations: minimizing drag and maximizing thickness of the wing. These factors are dictated by the wing's airfoil. A NACA 4415 airfoil is used with a maximum thickness of 15% of the chord length occurring at 40% of the chord. The load factor is the ratio of the lift to the weight and is used to calculate the load on the wing. This ratio can be used to calculate the lift force of the total aircraft using the weight estimation discussed previously. At level flight, the load factor is 1 (i.e., $L = W$). However, during other flight phases, the load factor will vary greatly. The maximum load factor that the aircraft will experience is referred to as the limit load factor. For commercial and personal aircraft, the FAR-23 and FAR-25 documents govern the limiting load factors (Nicolai *et al.* 2010). The load factor for a total duration of the flight can be described by a V-n diagram as a function of the aircraft's velocity. Any load that the aircraft will experience falls within the boundaries of the diagram and is restricted by the limiting load factors. Other critical velocities can be determined from the V-n diagram, such as the cruise velocity, dive velocity, and stall velocity. The dive velocity is the highest velocity within the V-n diagram and the stall velocity is the minimal velocity at level flight ($n = 1$). Figure 1 shows the V-n diagram for this analysis. The calculations are heavily based on historical data from past aircrafts.

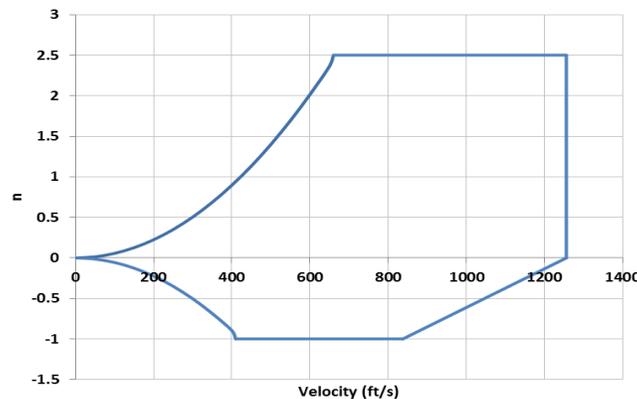


Figure 1. V-n Diagram (1ft/s = 0.305 m/s).

2 STRUCTURAL MODELING AND AERODYNAMIC FORCES

A MATLAB algorithm (Urgessa 2014) is utilized for the structural modeling of the aircraft wing and for determining the distribution of lift forces. The algorithm processes input parameters and output results in the form of STAAD.Pro editor for ease of conducting a parametric study. The structural components of the model include the nodes of the model, the aircraft skin, and spars. The algorithm also distributes the lift, fuel, and engine loads on the wing. The calculation of the nodes and force distribution is driven by the dimensions of the wing from the aerodynamic analysis spreadsheet. The dimensions, such as the wing length, surface area, taper ratio, and airfoil, are converted into a three dimensional model in MATLAB. The first step in creating the 3-

D model is creating a 2-D model (x and z components) and adding the airfoil curve in the y-component. The parameters for the 2-D model (Figure 2) are n_i and n_j , defining the mesh size of the model, and are defined as the number of divisions in the x and z directions of the wing. In theory, as these numbers approach infinity, the model will become more accurate. As the mesh size increases, the number of elements in the finite element model would increase, resulting in a longer computation time (Maji *et al.* 2008). The mesh size used here is 100 divisions in the x-direction and 40 divisions in the z-direction that takes into account accuracy of results and computing time. Eq. (5) and Eq. (6) are developed to determine the x- and z-coordinates.

$$x_{ij} = l \left(\frac{i-1}{n_i} \right) \quad (5)$$

$$z_{ij} = \left[\frac{l_{tip} \left(\frac{j-1}{n_j} \right) - \left(l_{offset} + l_{root} \left(\frac{j-1}{n_j} \right) \right)}{l} \right] x_{ij} + \left(l_{offset} + l_{root} \left(\frac{j-1}{n_j} \right) \right) \quad (6)$$

The offset length (l_{offset}) is calculated based on the length of the wing tip (l_{tip}) and the wing root (l_{root}). It is the difference in the z-direction of the first node of the root and tip of the wing. The y-coordinates are calculated using the NACA 4415 equations for a 4-series airfoil. Eq. (7) is used to calculate the airfoil's thickness in the chord-wise direction with no camber. The maximum thickness is 15% of the chord length, which occurs at 40% of the chord. The percentage of the chord is designated in the equations as z/c , where z is the location along the chord and c is the chord length. The leading and trailing edges are $z = 0$ and $z = c$ respectively.

$$y_{tij} = \frac{t}{0.2} c_j \left[0.269 \sqrt{\frac{z_{ij}}{c_j}} - 0.126 \left(\frac{z_{ij}}{c_j} \right) - 0.352 \left(\frac{z_{ij}}{c_j} \right)^2 + 0.284 \left(\frac{z_{ij}}{c_j} \right)^3 - 0.102 \left(\frac{z_{ij}}{c_j} \right)^4 \right] \quad (7)$$

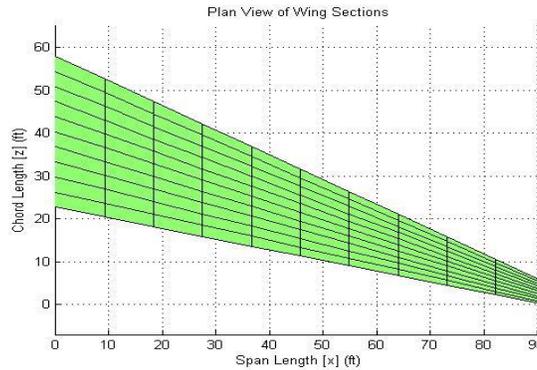


Figure 2. 2-D Aircraft Wing Model.

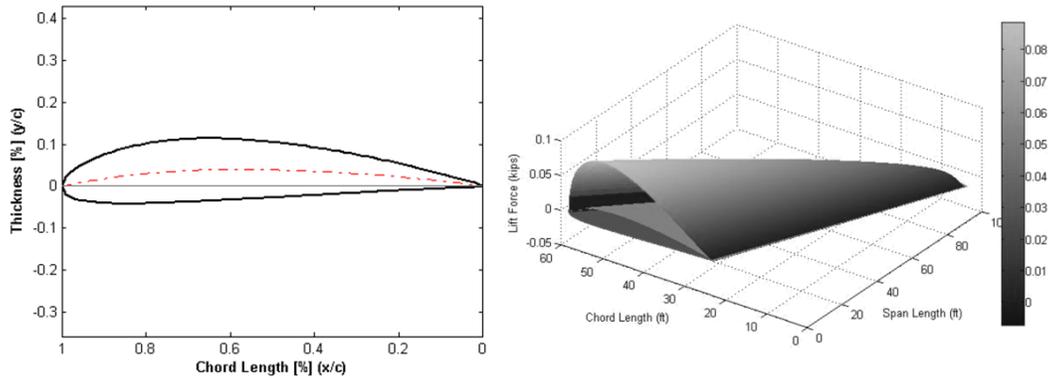


Figure 3. NACA 4415 Airfoil Model, and 3D lift distribution.

A series of calculations were conducted to accommodate the maximum camber of 4% of the chord for the selected airfoil. Figure 3 shows the plot of the selected airfoil.

The engine force is evaluated as a constant point load throughout the duration of the flight, while the fuel weight changes throughout the flight as it was being consumed. The MATLAB algorithm is set up to take one fuel force input that is distributed evenly over the wing where the fuel tank is located. This force changes for each flight phase based on the calculations done in the aerodynamics study. The MATLAB algorithm is then used to distribute the lift force from the V-n diagram over the area of the wing. The lift force calculated using the load factor is considered to be the resultant lift force applied at the center of gravity of the wing (Urgessa 2011). This was necessary because the lift force is distributed unevenly over the wing. The lift force (L) is estimated using the Schrenk Approximation taking the average of the lift force on a trapezoidal wing (L^T) and an elliptical wing (L^E) to calculate the span-wise lift distribution on the wing (Schrenk 1940).

3 PRELIMINARY RESULTS FROM STRUCTURAL ANALYSIS

STAAD.Pro is used for structural analysis and for conducting a parametric study on the aircraft wing (STAAD.Pro V8i 2012). A fixed condition is assumed at the root the wing to simulate the wing's connection the fuselage. The skin of the aircraft is modeled as a quadrilateral (4-noded) plate. The parameters of interest include loading conditions for different flight phases and the variation of wing skin thickness. The loading on the wing is investigated for different flight phases. The loads consisted of the engine, the fuel, and lift forces. Combing these three forces, the loading of any part of the flight duration could be analyzed. Figure 4 shows analysis results for the loading condition at the different phases. The deflection (bright green model) and the skin stresses are shown. This is useful to understand how stresses get redistributed during the flight, and the most extreme deflections that the wing will experience. The analysis is capable of taking a minimum and maximum thickness so the skin thickness, which varies as a

function of the loading, can be varied within the specified range. It can be seen that the loading pattern significantly affects stresses.

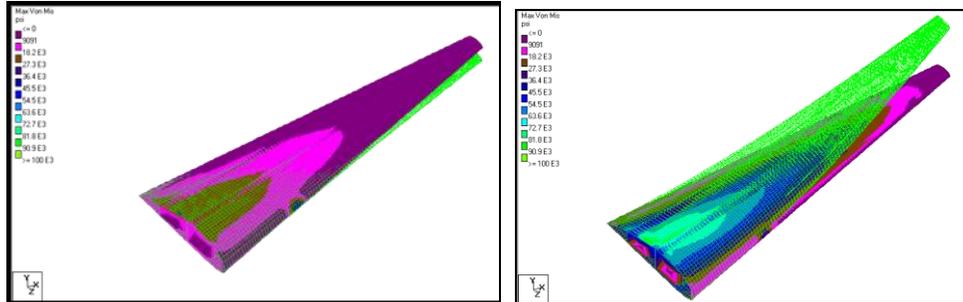


Figure 4. Stresses at the start of the flight and the climb phase.

4 CONCLUSION

This paper introduces a software integrated methodology to perform the preliminary analysis of an aircraft wing. The programs used (Microsoft Excel, MATLAB, and STAAD.Pro) are very widely used and easily available. The MATLAB algorithm produced was able to output the wing's structural components and loading information in a format that could then be analyzed in STAAD.Pro. The methodology was used to successfully evaluate the loading conditions for different flight phases and their associated deflections. It was also used to study the effects of changing the skin thickness of the aircraft wing. The methodology shows the potential to efficiently use a non-aerodynamic software to produce a detailed preliminary analysis.

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