Design and Finite Element Analysis of a Piper Warrior II Wing

ROSARIO DONNARUMMA[1]

[1]University of Naples Federico II, Department of Industrial Engineering

Correspondence: rosar.donnarumma@studenti.unina.it

ABSTRACT In this paper a modelling and subsequent finite element analysis of a Piper Warrior II aircraft wing is presented. The structure consists of seven ribs, four pairs of spars with different cross-sections and the skin. For this report, the manufacturer's choice to use aluminium *7075 - T6* as the material for the skin was respected. The finite element analysis aims (FEM) to analyse the response of the wing to an aerodynamic pressure stress. In order to optimize the airfoil design, it was decided to compare the results in the case of a symmetrical airfoil and manufacturer's ones, which instead foresees the assembly of airfoils. The analysis and modelling of the wing profiles was carried out using a Python script which, taking into account the regulations in force, generates the profile curve given the reference chord. The *Airfoil* library is also able to predict not only the behaviour of an isolated profile but also of an infinite wing. It would then be possible not only to study the induced resistance of the wing but also to construct its polar and analyse the distribution of lift along the wingspan.

KEYWORDS *Airfoil, FEMAP, Structural Analysis, Python, Piper Warrior, Winged Box.*

I. INTRODUCTION - WING BOX

The wing box is the main structural element of an aeronautical wing. It is essentially made up of: *spar, ribs, stringer* and *skin*. In addition to bending stress, the wing is also subjected to torsion due to aerodynamic forces and the wing box reproduces the theoretical geometry to resist these stresses.

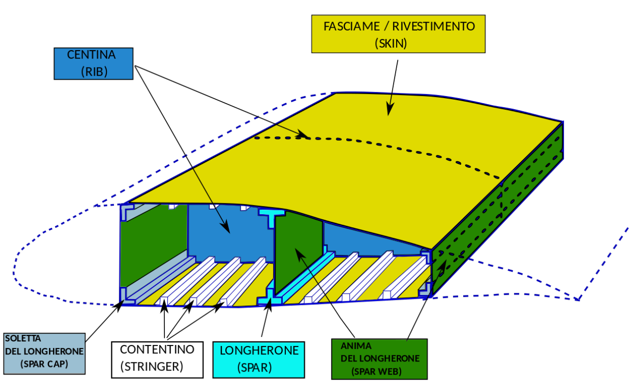

Figure 1: Wing box.

A. SPAR

The *spar* is the main structural element of the wing: it consists of a core called the spar web and a slab called the spar cap. From a static point of view, it behaves like an *shelf beam* embedded in the fuselage. The core responds to shear stresses while the slabs respond to bending moment.

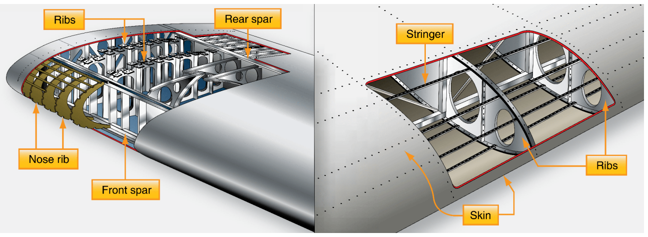

Figure 2: Representations of some spars inside a wing

B. RIBS

The essential function of the *ribs* is to give shape to the wing, support the skin, transfer stresses from the skin to the spars and reduce the size of the skin panels, thus also reducing the phenomena of wing instability. In general, the ribs are made of wood, metal or composite material. In wooden constructions, the ribs are generally double, i.e. with two plywood cores that enclose the slabs and the uprights, which in turn can be laminated or solid depending on the type of constitution. Usually the ribs extends from the front edge of the wing to the rear spar or trailing edge of the wing. Ribs give the typical curved shape and help transmit the air load from the skin to the spars. They are essential when the structural configuration of the aircraft includes fuel tanks in the wing, thus bearing the weight of the fuel tank. When it is necessary to house fuel tanks or allow control rods to pass through, in this case (fig. 4-b) in the figure, an open rib is used in the lower part suitably stiffened with box elements.

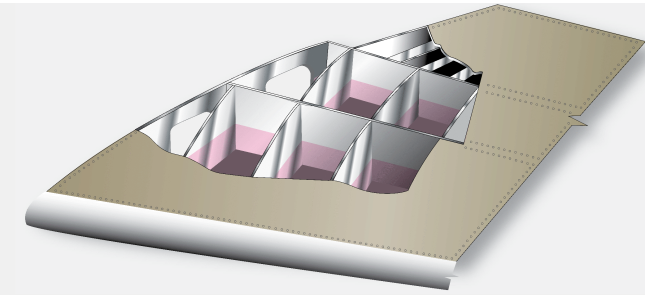

Figure 3: Representation of the ribs inside the *wing box*.

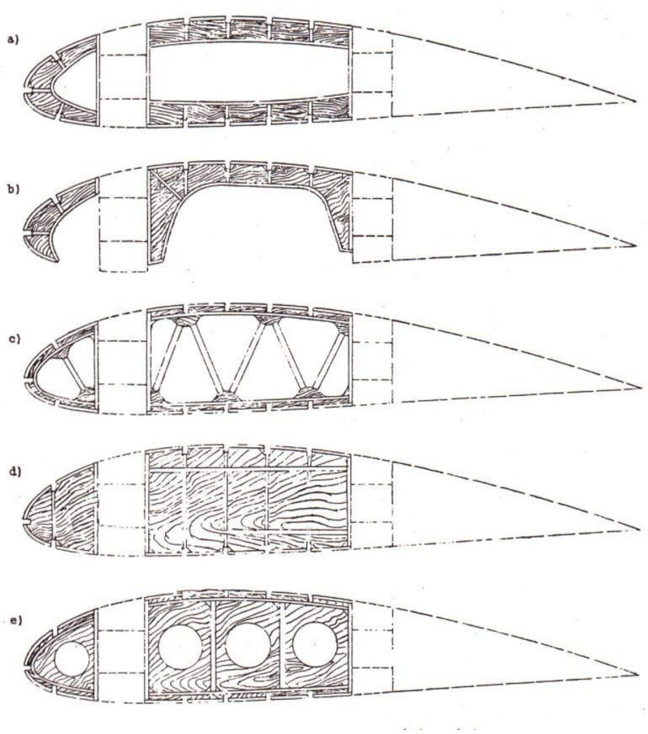

Figure 4: Different types of wing boxes

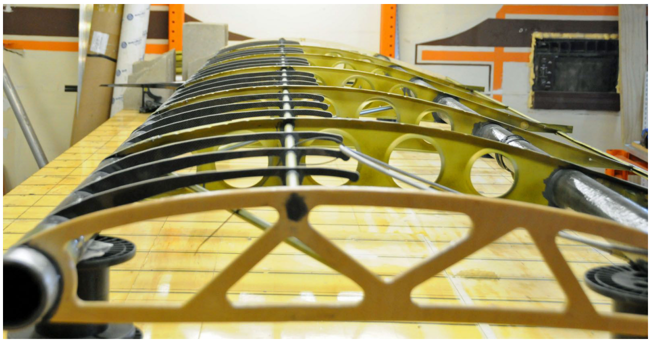

Figure 5: An example of a carbon fibre wing with aluminium ribs.

C. SKIN

The *skin*, in addition to its aesthetic function, is an important component from both a static and aerodynamic prospective. It receives the aerodynamic thrust, which is transmitted to the spars by means of the ribs, resists torsion and contributes to the bending to which the wing is subjected. Thanks to the ribs, it also prevents deformation (bulging or sinking) to which the panels could be subjected.

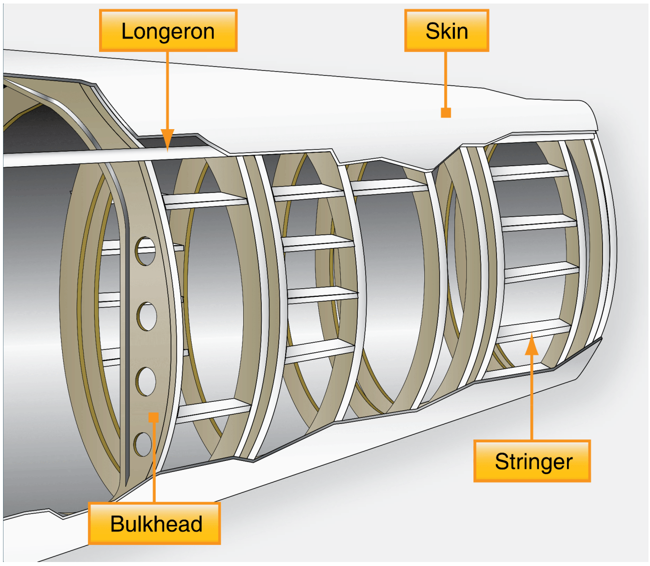

Figure 6: Cross-section of a half-wing showing the spars. [1]

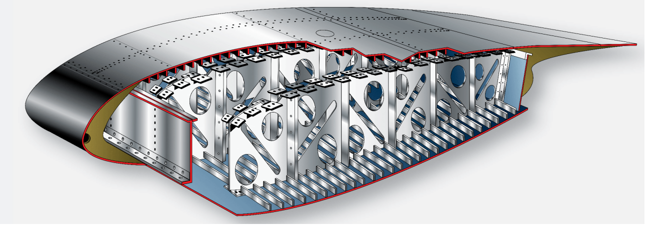

Figure 7: Another cross-section of a wing box. [1]

II. WING DESIGN

In this section, it will be used FEMAP embedded CAD tools to design the wing. For this purpose, the following procedure will be followed:

A. Design of the airfoil;
B. Overall design;
C. Choosing the materials for the different parts of the wing;
D. Evaluation of the different loads;
E. Analysis of the results.

A. AIRFOIL

The choice of the airfoils that will define the surface of the wing is of crucial importance for the complete characterisation of the aircraft aerodynamic properties. The pressure distribution on the back and on the bottom skin, generated by the geometric curvature of the profile is in fact responsible for the lift force that supports the aircraft. The choice of airfoil must take into account not only aerodynamic considerations, but also more practical factors such as the often imperative need for a tank in the wing to hold fuel. At small angles of attack, a symmetrical profile works better than the corresponding curved profile. On the other hand, at higher angles of attack, a curved airfoil works better than its symmetrical counterpart. At each normal angle of attack (up to about 12 degrees), the two profiles generate equal lift. From there the curved profile has a great advantage because it does not stall up to a much higher angle of attack. As a consequence, its maximum lift coefficient is greater. It is decided here to use a symmetrical NACA 0018 profile, with the aim of ensuring greater static stability. [8]

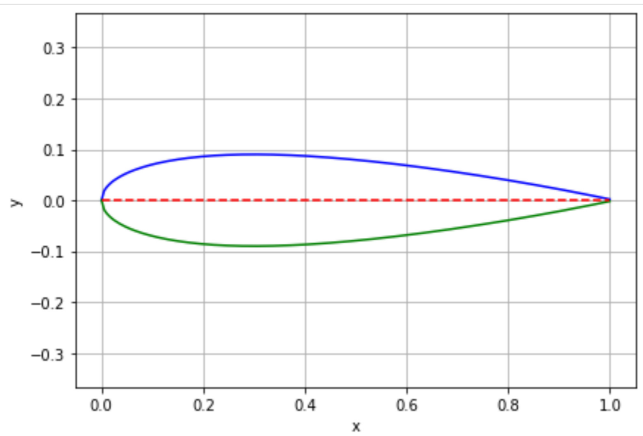

Figure 8: Representation of a NACA 0018 profile resulting from the Python script (Appendix) on a 200-point mesh

The Python script is used in order to automate the process of applying the following NACA formula:

$$y_t = 5tc\left[0{,}2969\sqrt{\frac{x}{c}} + (-0{,}1260)\left(\frac{x}{c}\right) + (-0{,}3516)\left(\frac{x}{c}\right)^2 + \right.$$
$$\left. +0{,}2843\left(\frac{x}{c}\right)^3 + (-0{,}1015)\left(\frac{x}{c}\right)^4\right]$$

where:

- c is the chord length ;
- x is the position from 0 to c;
- y_t is half of the thickness at a given value of x ;
- t is the maximum thickness expressed as a fraction of the chord, so that 100 t is equal to the last two digits of the NACA serial code.

B. OVERALL DESIGN

As previously stated, the FEMAP embedded CAD software has been used. The resulted model has the same specifications of the original wing. It can be seen that the aircraft wing is indeed tapered. The choice of tapering ratio is a compromise between structural and aerodynamic objectives. In fact, a low taper ratio leads to triangular wings with a larger section at the root, which from a structural point of view would be optimal but this leads to stall displacement in the area of the aircraft's control devices and the possible consequent loss of controls. An arrangement with a higher taper ratio leads us to completely opposite considerations. A good compromise is the midpoint between the two solutions: the best solution is therefore to choose a taper λ ratio of around 0.5.

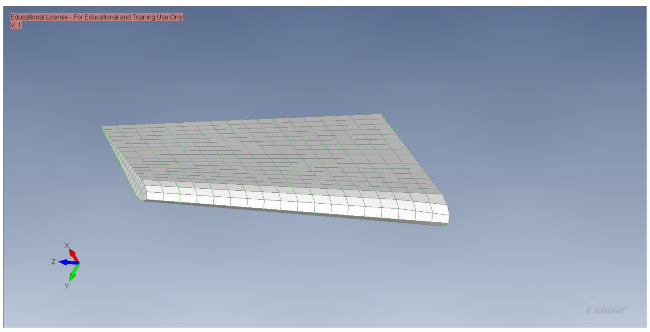

Figure 9: Resulted mesh of the wing.

C. MATERIAL

Building a geometry can involve choices that are far from simple, such as the material to be used, assembly techniques and manufacturing. In the aerospace industry, two aluminium alloys are the most successful: *2024 - T3* and *7075 - T6*. The *7075* represents the group of aluminium-based alloys in which the main alloying elements are magnesium and zinc.

The outstanding characteristics of aluminium and its alloys are strength-to-weight ratio, corrosion resistance and high thermal and electrical conductivity.

The density of aluminium is about 2770 kg/m^3 while its tensile strength and elastic modulus are about *90 MPa* and *71.7 GPa* respectively. Taking into account the cost and strength of aluminium and its alloys, they are among the most versatile materials from a manufacturing point of view. More than 70% of the structural weight of modern civil aircraft, such as the Airbus A330/A340, or the Boeing 777, is attributable to high-strength aluminium alloys. [3]

Aluminium 7075 is an alloy mainly used in the aeronautical field, particularly in the structural parts of aircraft. It is also the aluminium alloy least prone to thermodilation and therefore most suitable for outdoor use as it can withstand temperature changes better. Originally intended exclusively as an alloy for use in the aeronautical field for the manufacture of aircraft, rocket and aerospace components, it has since been used for a variety of applications. The only advantage of aluminium 7075 compared to the 6000 series alloys is the cost of processing, production and purchase, which is much higher (about three times as much). [4]

D. Loads

For the chosen analysis, I impose a load of *pressure* on the *bottom skin*, in the area corresponding to the lower part of the hypothetical tank. In order to validate the model, a similar analysis is chosen as a comparison term. If at a stress of the same type and magnitude we have a comparable result, we are satisfied with the analysis. We then impose a series of constraints on the beams by attaching them to the ends, and we attach the wing to the hypothetical fuselage.

III. Results

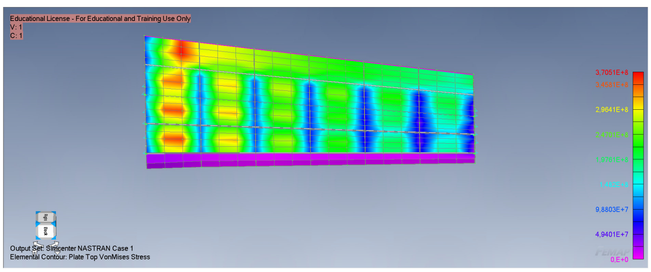

Figure 10: Finale result - *Bottom View*.

Observing the static analysis of the wing, the stress values are comparable with those found in literature. Comparing the reported case with a case in which the composite material [6] is used, it can be seen that the use of *GLARE* (Glass - Reinforced Aluminum Laminate) leads to a significant lowering of the stresses developed on the surface of the wing. This type of composite material is part of a new class of metal-fibre laminates used in high-end engineering where high performance but low weight is required. [7] In order to improve the behaviour of the wing, also from an aerodynamic point of view, it is necessary to impose a certain pitch. Furthermore, the analysis shows that the most stressed area is clearly the one closest to the root. This experimental data is well supported from a theoretical point of view: since the wing is tapered, we expect a trapezoidal load distribution since the aerodynamic lift is proportional to the length of the wing chords.

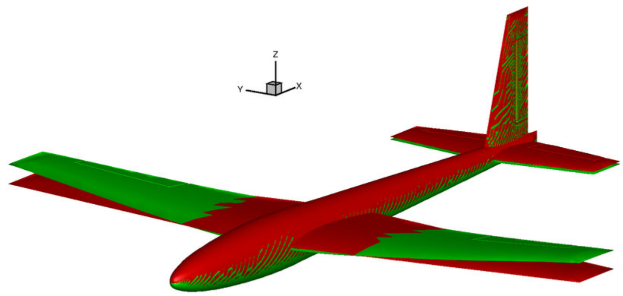

Figure 11: Undeformed structure and subsequent deformation of an aircraft with a GLARE composite wing. [9]

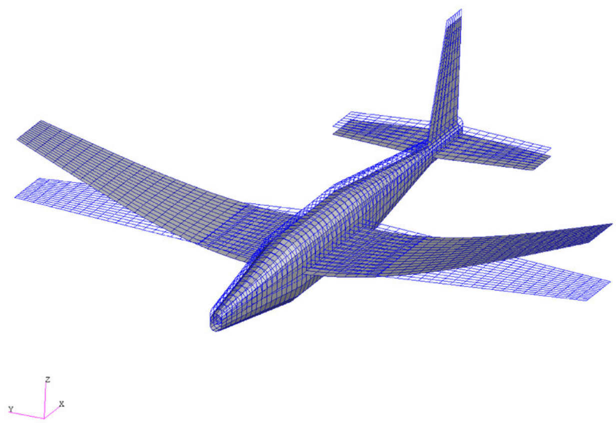

Figure 12: Finite element model of a generic aircraft and an embedded wing subjected to aerodynamic pressure. [9]

APPENDIX. PYTHON LISTING

Listing 1: Python code for the generation of NACA profile 0018

```
import numpy as np
import pandas as pd
from airfoils import Airfoil

# The aim is to create a graph for the NACA 0018 profile.

foil = Airfoil.NACA4("0018", n_points=200)
foil.plot(show=True, save=True, settings={'points': False, 'camber': True})

# When using CAD, far fewer points will be used.
# of points. In particular,
# 10 points will be needed to determine the rear
# back and one for the ellipse.

foil = Airfoil.NACA4("0018", n_points=11)
foil.plot(show=True, save=True, settings={'points': False, 'camber': True})

# Check to see if the profile is symmetrical.

x = foil.camber_line_angle(x=0.5);
y = foil.camber_line(x=0.5)

print("The chamber line is" + str(x))
print("The chamber line angle is" + str(y))

cr = 192;                                            # Root Chord
ct = 129;                                            # Tip Chord

maxthick = 0.18*cr;                                  # Evaluation of the maximum thickness
                                                     # remembering that the last digits in a
                                                     # 4-digit NACA profile represents the maximum
                                                     # the maximum percentage thickness


maxthick2 = 0.18*ct;

rastr = ct/cr;


Vector_radius1 = 0.2*cr                              # Semi-minor axis

Vector_radius2 = 0.2*ct                              # Semi-major axis
```

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