19th International Conference on
Adaptive Structures and Technologies
October 6-9, 2008
Ascona, Switzerland

Aeroelastic Tailoring of a Wing Box in Subsonic Compressible Regime

Giacomo Canale[1 *], Paul M. Weaver[2], J. Enrique Herencia[3]
[1]PhD Student, University of Bristol, Bristol, UK, giacomo.canale@bristol.ac.uk
[2]Reader in Lightweight Structures, University of Bristol, Bristol, UK
[3]Marie Curie Research Assistant, University of Bristol, Bristol, UK
*Corresponding author, Queen's Building, University walk, BS8 1 TR, Bristol, UK

Abstract

The bend-twist coupling of a wing box built with unbalanced composite laminates can be used to improve an aircraft's range. Two techniques to find the best thicknesses and fibres angle distribution are described in this work. A heuristic approach is applied inspired by physical understanding of the problem. An aircraft's range is evaluated by a quasi-static aeroelastic model, which accounts for the interaction between aerodynamic forces and elastic deformations. Effects of compressibility are also considered.

Keywords: Bend-twist coupling, composite thin walled beams, static aeroelasticity.

1. Introduction

It is well known that composite materials are useful in the construction of aeronautical structures. Typically, engineers design lightweight composite structures with orthotropic properties. However, it has also been shown that to fulfil the potential of composite materials it may be necessary to utilize their inherent anisotropy. In other words, this anisotropy can be used to drive the deformation of a structure, as a wing or a vertical fin, subjected to the aerodynamic loads, in such a way as to improve a specific aspect of performance of the whole aircraft [1].
In this paper, the beneficial effect of unbalanced, symmetric laminated composites on aircraft's range is demonstrated.

2. Model description

Range is the distance that an airplane traverses in uniform horizontal flight with a fixed amount of fuel. A simple model to evaluate such performance in subsonic compressible regime has been developed.
This model is based on two fundamental hypotheses:

- The evaluation of the aircraft's range is performed with Breguet's formulation [2].
- Loads and wing deformations, both necessary to evaluate the range, are calculated with a quasi-static iterative procedure.

The range R of an aircraft, flying at constant altitude and constant angle of attack during the cruise, can be evaluated, according to reference [2], as

$$R = \frac{2\sqrt{2}}{c\sqrt{\rho S}}\left(\sqrt{W_0} - \sqrt{W_1}\right)\frac{\sqrt{L}}{D} \tag{1}$$

where

c	is the specific fuel consumption, i.e. the quantity of fuel consumed per unit of thrust per second.
ρ	is the air density.
S	is the surface area of the wing.
W_0	is the weight of the aircraft at the beginning of the cruise.
W_1	is the weight of the aircraft at the end of the cruise.
L	is the lift of the whole aircraft.
D	is the drag of the whole aircraft.

Lift and drag are evaluated by using an iterative procedure, which is quicker and easier to implement than a coupled aerodynamic and structural system.
The wing, originally rigid, deforms when the aerodynamic loads are applied. Such a deformation induces changes in the aerodynamic field surrounding the structure, and consequently induces changes in the loads applied. Therefore, these loads are re-calculated. The new loads induce a new deformation field and so on. The process is stopped when the deformation obtained at the last step differs less than 1% with respect to the previous one.
From the structural point of view, the wing is modelled as a prismatic thin-walled composite beam as shown in Figure 1. In other words, all the stiffnesses of the wing are assumed to be coincident with those of its wing box, as proposed by Patil [3].

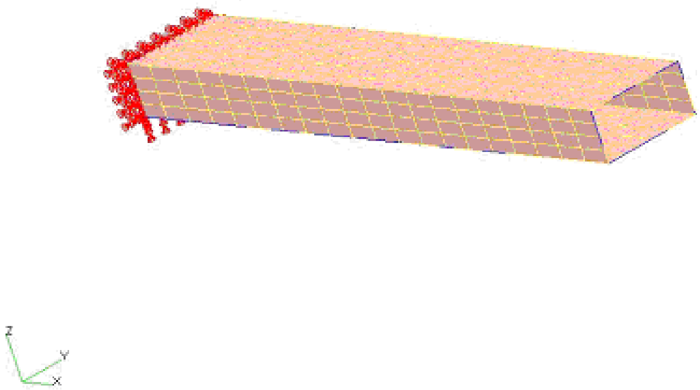

Figure 1: A wing box represented as a prismatic thin walled beam

The model used to evaluate the deformation is linear and is suitable to describe deformations of long beams. The model is formulated in matrix form as

$$\begin{bmatrix} M(x) \\ T(x) \end{bmatrix} = \begin{bmatrix} EI & -K \\ -K & GJ \end{bmatrix} \begin{bmatrix} \dfrac{d\varphi(x)}{dx} \\ \dfrac{d\psi(x)}{dx} \end{bmatrix} \tag{2}$$

where

x	is the span wise coordinate in the global reference, shown in Figure 1
$\psi(x)$	is angle of torsion at the coordinate x
$\varphi(x)$	is angle of bending at the coordinate x

M(x) is bending moment at the coordinate *x*
T(x) is twisting moment at the coordinate *x*

and

EI is the bending stiffness of the wing box cross section
GK is the torsional stiffness of the wing box cross section
K is the bend-twist coupling stiffness of the wing box cross section

The stiffnesses *EI*, *GJ*, and *K* are calculated using the model of Librescu and Song [4]. While the physical meaning of bending and torsional stiffness is very clear, a brief explanation is needed for *K* (bend-twist coupling stiffness). If *K* is different from zero and a bending moment is applied to the wing, beside the bending deformation a distribution of angle of torsion is measured and vice versa.
When the lifting force is loading the wing, two different kinds of torsional deformation are possible:

- A "nose up" rotation that tends to increase the angle of attack (Figure 2).
- A "nose down" rotation, whose effect is to decrease the angle of attack.

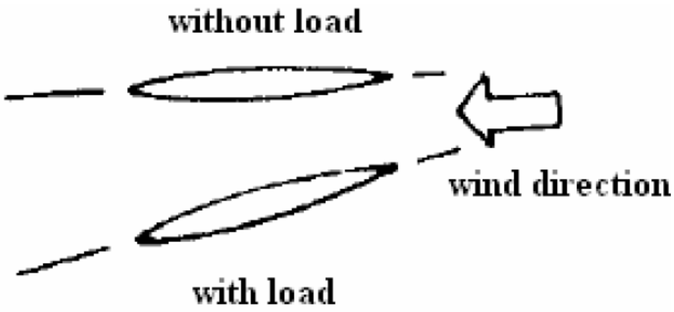

Figure 2: "nose up" rotation of a wing subjected to bending load [1]

These effects can be obtained in a symmetric wing box with the top and bottom walls built with symmetric but unbalanced composite materials. The fibre direction θ to obtain a "nose up" effect is shown in Figure 3.

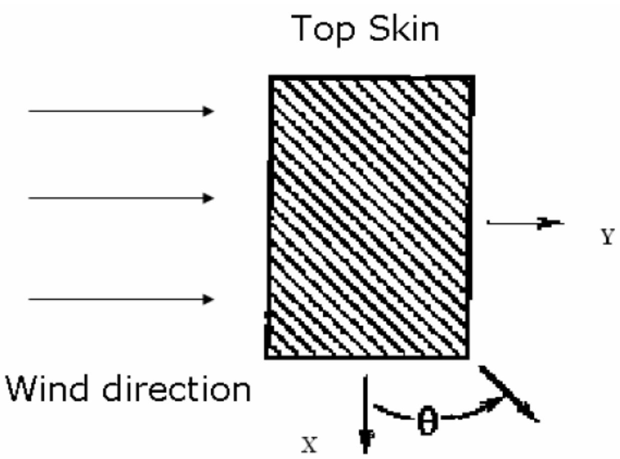

Figure 3: Fibre angle θ of the top and bottom laminates to obtain "nose up" effect. Angle is measured in the local frame *x-y* [1]

The aerodynamic model used to evaluate lift and drag is a subsonic vortex-lattex theory [5], which is based on the electric analogy of Biot-Savart's law. The Prandtl-Glauert correction is introduced to model the compressibility effects.
In the wing box considered here, vertical walls are built with orthotropic laminates having 88% of

fibres oriented with an angle of 0 degrees from the flexural axis x (shown in Figure 1) and 12% of fibres oriented with an angle of 90 degrees. Top and bottom walls of the wing box are made with symmetric laminates with 44% of fibres disposed at 0 degrees with the respect to the flexural axis 12% of fibres disposed at 90 degrees and 44% of fibres oriented with an angle θ that can be chosen in the interval [-90, 90] degrees with respect to the local axis shown in Figure 3.
The final goal is to find a distribution of fibre angles $\theta(x)$ along the span wise coordinate x of a semi-wing in such a way as to improve the range of the aircraft. Such improvement is measured with respect to the performance of an aircraft whose wing is fully orthotropic (both the vertical and horizontal walls).
The first step is to design the orthotropic wing. Once the geometry of the semi-wing and the requested load factor n are known (they are introduced as input data), the orthotropic wing can be sized. Hence, the distribution of thicknesses of the horizontal and vertical laminates along the span is sized in such a way that the Tsai-Wu static failure criterion [6] is satisfied. Then, the aircraft range is evaluated.
Different distribution of fibres angles $\theta(x)$ along the span of the wing can be investigated to find a better range performance.
Usually, this kind of problem can be formulated in terms of a constrained optimisation:

- Find the distribution of vertical thicknesses $t_v(x)$, horizontal thicknesses $t_h(x)$ and fibres angles $\theta(x)$ in such a way as to maximize the range of the aircraft under two constraints
    1. Lift of the deformed configuration in uniform horizontal flight must not vary more than 5% with respect of the lift of the rigid, i.e. non deformed, wing.
    2. The structure must be able to carry the design loads at the given load factor n

However, optimisation algorithms are often used as "black boxes", while the first aim of a low fidelity model is to have a good comprehension of the phenomena described. In this case, the formulation of the problem is quite simple. The solution in terms of $t_v(x)$ $t_h(x)$ and $\theta(x)$ can be based on its physical understanding. Moreover, an optimization technique may undergo several iterations until it reaches an optimum while having a physical understanding of the problem allows a reduction of the number of solution to investigate.
For our problem, there is no physical reason to have some part of the wing showing a "nose up" rotation and others showing a "nose down" one. Therefore, to obtain a range improvement, a global "nose up" or "nose down" rotation along the entire span is required. For this reason only two kinds of solution are investigated.
The first one consists of a constant fibres angles distribution along the span. In other words, all the fibres (44% inside each laminate, as mentioned before) along the span are oriented with the same angle within the interval [-90, 90] deg. This kind of solution has been suggested by a previous optimisation study using a standard genetic algorithm (GA) [6]. An example of such a distribution is shown in Figure 4.

0	0	0	-15	-15	-15	-15	-15	-15	-15

root→→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→→→tip

Figure 4: Example of constant angle distribution on the top and bottom laminates of the wing box

The root of the wing, remains orthotropic, to guarantee a high level of bending stiffness in that part subjected to the maximum bending load. Consequently, anisotropy is investigated for lengths greater than 30% of the span. Analyses performed have shown that if anisotropy was also allowed in the root of the wing, the "anisotropic" wing would result more than 10% heavier than the orthotropic wing.
In addiction, vertical walls of the wing box are always considered as orthotropic.
The second kind of solution investigated is a progressive distribution of "nose up" or "nose down" effect along the span of the structure. The first step is to calculate the fibre angle θ_0 giving the maximum bend-twist compliance. In other words, θ_0 is the angle that gives the maximum "nose up" or "nose down" rotation when a bending load is applied. The wing is initially orthotropic and it is

divided into several sections to simplify the analysis. Starting from the section placed at 30% of the span (also in this kind of solution the root of the wing remains orthotropic) an amount of bend-twist coupling is given. For the “nose-up” case, fibre angles varying in 5 degrees increments are investigated (or -5 degrees for “nose down” case). An example of the first step of this kind of solution is given in Figure 5.

0	0	0	5	0	0	0	0	0	0

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→tip

Figure 5: Fibre angle distribution at the top and bottom laminates of the wing box: Step 1 of “progressive nose up distribution”

Lift and range are evaluated after the first step. Then, the fibre angle of the same wing segment is increased (or decreased) by 5 degrees and the range is evaluated again. This process is stopped when the value of fibre angle becomes equal to θ_0 and consequently the maximum bend-twist compliance is obtained in that particular segment of the wing box. Then, the process is repeated in the neighbouring section and so on. The process terminates when the value θ_0 is reached in each segment of the wing box. All the steps are shown in Figure 6.

Step 2 of progressive nose up distribution

0	0	0	θ_0	0	0	0	0	0	0

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→tip

Step 3 of progressive nose up distribution

0	0	0	θ_0	5	0	0	0	0	0

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→tip

Last step of progressive nose up distribution

0	0	0	θ_0	θ_0	θ_0	θ_0	θ_0	θ_0	θ_0

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→tip

Figure 6: Fibre angle distribution at the top and bottom laminates of the wing box: Description of progressive fibres angle distribution

For each distribution investigated, a failure analysis with Tsai-Wu criterion is performed to verify if the structure is able to carry the design load. In case of failure in a particular cross section, vertical and horizontal thicknesses are locally increased until the constraint is satisfied. The wing with the best range resulting from these analyses is compared with the orthotropic wing in terms of performance and weight.

Instead of a direct measure of the aircraft’s range, for the sake of simplicity, an index *I* with the same meaning can be used to compare different solutions:

$$I = \frac{\sqrt{C_l}}{C_d} = \frac{\sqrt{C_l}}{C_{d0} + \frac{C_l^{\,2}}{\lambda_{eq}}} \tag{3}$$

where

C_l is the lift coefficient, depending linearly on the angle of attack
C_{d0} is the zero lift drag coefficient
λ_{eq} is the wing's aerodynamic aspect ratio.

3. Numerical results

A prismatic wing with the technical data reported in the following tables has been studied.

Aerodynamic chord	1.5 m
Wing length	10 m
Angle of attack of cruise	5 deg
Cruise speed	220 m/s
Design load factor	4
Distance between aerodynamic centre of pressure and shear centre at each cross section	0.1 m
Height of the prismatic wing box	0.3 m
Chord of the prismatic wing box	0.6 m
C_{d0} of the airfoil	0.028

Table 1: Geometric characteristics of airfoil and wing box

E_1	181 GPa
E_2	10.3 GPa
G_{12}	4.55 GPa
v_{12}	0.28
density	1600 Kg/m^3
ply thickness	0.002 m

Table 2: Elastic properties of the composite material

The wing has been divided in to 10 segments for the analysis. The orthotropic wing obtained has thicknesses distribution as shown in Figure 8.

Thickness of horizontal walls of the orthotropic wing box [mm]

9	7	7	6	6	5	4	4	3	3

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→tip

Thickness of vertical walls of the orthotropic wing box [mm]

5	5	4	4	4	4	3	3	3	3

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→tip

Figure 8: Thickness distribution of horizontal and vertical walls for the orthotropic wing.

The best solution, detailing thickness and fibre angle distributions, to improve the aircraft's range is shown in Figure 9.

Thickness of horizontal walls of the wing box [mm] “optimised” for range

9	7	7	9	7	5	4	4	3	3

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→tip

Thickness of vertical walls of the wing box [mm] “optimised” for range

5	5	4	7	5	4	3	3	3	3

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→tip

Fibres angle distribution [deg] of the top and bottom laminates of the wing box “optimised” for range

0	0	0	-25	-25	-25	-25	-25	-25	0

root→→→→→→→→→→→→→span wise direction→→→→→→→→→→→→→→tip

Figure 9: Results of the anisotropic wing giving the best range performance.

The range improvement obtained is 0.8%, but the weight of the “unbalanced composite” structure is 6% greater than the orthotropic one.
After a few analyses the following is observed:

1. Laminates with bend-twist coupling can be used to improve the range of the aircraft. The order of magnitude of this improvement, for the cases analyzed, is less than 1%. Even if it appears a relatively poor result, it can imply huge money savings due to the reduction of fuel consumption.
2. To carry the same design load (i.e. to be designed with the same load factor n), the structure with bend-twist coupling could be heavier than the orthotropic wing. This fact implies an increase in cost as part of the lift available must be used to carry a heavier structure instead of passengers or goods. Of course further investigations should be performed to establish if the saving induced by range improvements are greater then the losses due to a heavier structure. This issue is addressed in future work.
3. A global “nose up” or “nose down” effect is needed to improve an aircraft’s range as a function of the initial angle of attack of the wing, in uniform horizontal flight. This effect is shown in Figure 7. The best range is obtained for a particular value of C_l (same physical meaning as angle of attack) such that the range index I is maximized. This value is denoted with C_{lmax}. If the design C_l lies on the right side of C_{lmax}, a reduction of the angle of attack is required to improve the range. Of course, a deformation in the opposite direction is necessary if the initial value of C_l stays on the left of C_{lmax}.

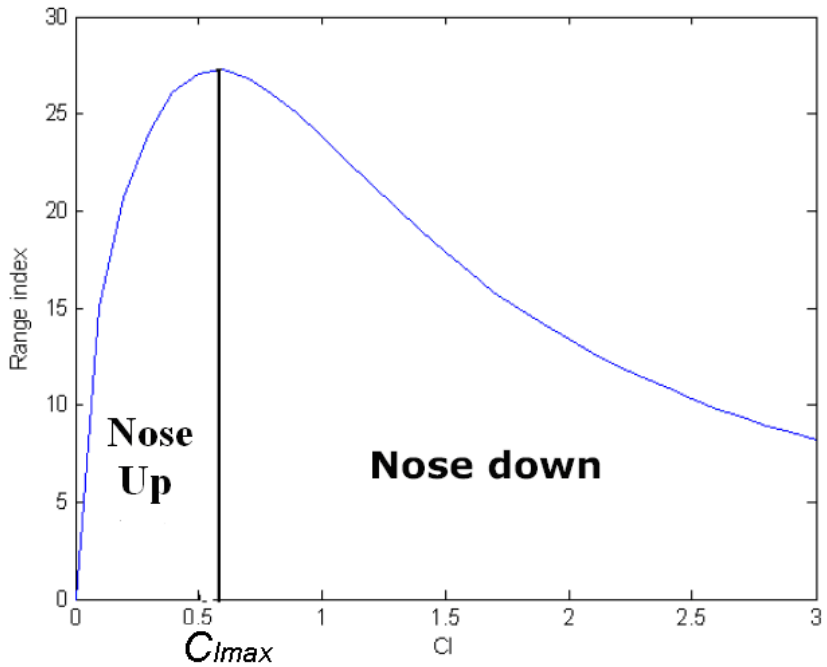

Figure 7: Qualitative shape of the function Index range $I = I(C_l)$. Areas of "nose up" and "nose down" rotation are shown.

4. Conclusions

Bend-twist structural coupling has been used to tailor wing-box deformations to improve an aircraft's range with respect to an orthotropic structure.
A quasi-static aeroelastic iterative approach has been used to evaluate the aerodynamic forces arising in the structure. Such forces have been used to evaluate range by using Breguet's formulation.
The use of unbalanced composite materials has shown gains in terms of aircraft performance (an increase of aircraft range), however, those gains had an associated structural weight penalty.

REFERENCES

1. M. Shirk, T.J. Hertz, T.A. Weisshaar, Aeroelastic Tailoring - Theory, Practice and Promises, Journal of Aircraft, 23, nr. 1 (1986)
2. Lausetti, "Elementi di meccanica del volo", Levrotto & Bella, Torino (1984).
3. M. Patil, Aeroelastic Tailoring of Composite Box Beams, Aerospace Sciences Meeting and Exhibit, 35th, Reno, NV, Jan. 6-9, 1997
4. L. Librescu, O. Song, On the static aeroelastic tailoring of composite aircraft swept wings modelled as thin-walled beam structures, Composite Engineering, 2, nrs 5-7 (1992)
5. J. J. Bertin, Aerodynamics for Engineers, Prentice Hall (2003)
6. Z. Gurdal, R. T. Haftka, P. Hajela, Design and Optimization of Laminated Composite Materials, Wiley Interscience, New York (1999)