

Optimization of an Unmanned Aerial System' Wing Using a Flexible Skin Morphing Wing

Oliviu Sugar Gabor, Andreea Koreanschi, and Ruxandra Mihaela Botez
Ecole de Technologie Superieure

ABSTRACT

In this paper, we describe a practically efficient methodology of improving the aerodynamic characteristics of an UAS's wing using a morphing approach. We have replaced a part of the original wings' upper and lower surfaces with a flexible, composite material skin whose shape can be modified, according to the variable airflow conditions, using internally placed actuators. The optimal displacements of the actuators, as functions of the external flow characteristics, are determined using a genetic algorithm based optimizer, coupled with a three - dimensional numerical extension of the classical lifting line model for estimating the modified wing aerodynamic coefficients. We have used the optimization tool to decrease the overall drag coefficient of a military grade UAS' wing equipped with the flexible skin. We have obtained good quality solutions for only a fraction of the computational cost needed when performing viscous flow field calculations.

CITATION: Sugar Gabor, O., Koreanschi, A., and Botez, R., "Optimization of an Unmanned Aerial System' Wing Using a Flexible Skin Morphing Wing," *SAE Int. J. Aerosp.* 6(1):2013, doi:10.4271/2013-01-2095.

INTRODUCTION

In recent years, Unmanned Aerial Systems (UAS) have become increasingly used in both military and civil aviation. Their main goal has been performing long time surveillance flights, at various altitudes and flight speeds, and sometimes in rapidly changing weather conditions. Because of the increasing demand of UAS's, engineers and designers have searched for methods to improve their flight performances, in order to make them more adaptable for various flight missions, to improve their aerodynamic efficiency and to increase their effective range and payload.

One answer to all these aircraft design challenges is to use a morphing technique, to provide the aircraft with the capacity of detecting the changes occurring in the airflow around it and to adapt to them by modifying its geometry, usually the wing, during flight. Sofla et al [1] conducted a comprehensive study of the various aircraft morphing solutions proposed by different authors.

A remarkable project into developing a functional morphing aircraft was Lockheed Martin's unmanned combat air vehicle (UCAV), which had wing folding capabilities. [2] [3] The UCAV would be capable of long range cruise by minimizing fuel burn using an extended wing span, as well as transitioning into the attack mode, with higher maximum speed and increased manoeuvrability by decreasing its wing

span. A functional model of the UCAV was designed, fabricated and tested in the wind tunnel, at speeds ranging from the subsonic incompressible regime up to transonic conditions.[4]

Gamboa et al [5] presented another concept of a morphing wing developed for a small Unmanned Aerial Vehicle. The morphing wing created had both span and chord expansion capabilities, its purpose was to obtain the overall drag reduction with respect to the original, fixed wing, for low speeds ranging from 15 m/s up to 50 m/s. Drag reductions between 14.7% at a speed of 20 m/s and 34.5% at a speed of 50 m/s have been obtained.

NEXTGEN Aeronautics proposed an UAV design with a wing structure capable of being transformed from a high span configuration for slow speed flights, to a configuration with a reduced wing span, adapted to high speed flight. [6] In this proposed solution, the wing was based on a moveable truss structure that could be controlled using electro - mechanical actuators, in order to adjust the wing span, area and shape according to variable flight conditions. In August 2006, NEXTGEN performed a successful flight test of the prototype.

The CRIAQ 7.1 project took place between 2006 and 2009 and was realized following a collaboration approach between teams from École de Technologie Supérieure (ÉTS), École Polytechnique de Montréal, Bombardier Aerospace,

Thales Canada and the Institute for Aerospace Research - National Research Canada (IAR - NRC). The objective of the project was to improve and control the laminarity of the flow past a morphing wing, in order to obtain important drag reductions.

In this project, the active structure of the morphing wing combined three main subsystems: a flexible, composite material upper surface, stretching between 3% and 70% of the airfoil chord; a rigid inner surface; a Shape Memory Alloy (SMA) actuator group located inside the wing box, which could morph the flexible skin at two points, located at 25.3% and 47.6% of the chord. [7] The reference airfoil chosen was the WTEA laminar airfoil. The morphing airfoil was designed for low subsonic flow conditions. A theoretical study of the morphing wing system was performed, [8] and very promising results were obtained; the morphing system was able to delay the transition location downstream by up to 30% of the chord, and to reduce the airfoil drag by up to 22%. The wind tunnel tests were performed in the 2 m by 3 m atmospheric closed circuit subsonic wind tunnel at IAR - NRC and validated the numerical simulations. [9]

FINITE SPAN WING MODEL

Prandtl's classical lifting line theory, first published in 1918 [10], represented the first analytical model capable of accurately predicting the lift and induced drag of a finite span lifting surface. The aerodynamic characteristics predicted by the theory were repeatedly proven to be in close agreement with experimental results, for straight wings with moderate to high aspect ratio.

The solution of Prandtl's classical equation is in the form of an infinite sine series for the bound vorticity distribution. Traditionally, the series is truncated to a finite number of terms, and collocation methods are used to determine the sine series coefficients, method that was firstly presented by Glauert [11]. Popular methods of determining the bound vorticity distribution included those developed by Tani [12] and Multhopp [13]. Other authors have proposed modified versions of the original lifting line theory, modifications that increase the quality of the obtained results [14], increase the applicability or the model [15] or make use of information regarding the wing's airfoil sections. [16] More modern solutions include that of Rasmussen and Smith [17], who presented a more rigorous method based on a Fourier series expansion.

With the development of more efficient and powerful computers, several authors have also proposed purely numerical methods for solving Prandtl's lifting line equation. Among these, we can mention McCormick [18], Anderson et al [19] or Katz and Plotkin [20].

However, all the above mentioned numerical methods were based on Prandtl's assumptions of a straight distribution of bound vorticity, and therefore are subjected to all the limitations of the classical lifting line model: a single lifting surface of moderate to high aspect ratio, with no sweep angle

and no means of considering the effects of the various wing sections airfoils.

The method used in this paper is based on the original work of Phillips and Snyder [21]. Whereas the classical lifting line theory is based on the assumption of a straight lifting surface and the application of the two - dimensional Kutta - Joukowski vortex lifting law for a three dimensional flow, the modern adaptation uses a general horseshoe vortex distribution and uses a fully three - dimensional vortex lifting law. Because of these characteristics, the method has a much wider applicability range compared to the original theory, including multiple lifting surfaces and wings arbitrary camber, sweep and dihedral angle.

WING CALCULATION METHOD

In numerical lifting line models, the continuous distribution of bound vorticity over the lifting surface and the continuous distribution of trailing vorticity are approximated using a finite number of horseshoe vortices. The bound portion of the vortices can be aligned with the wing's quarter chord line, thus taking into consideration the local values of the sweep and dihedral angles, while the trailing portions remain aligned with the free stream velocity, as shown in Figure 1.

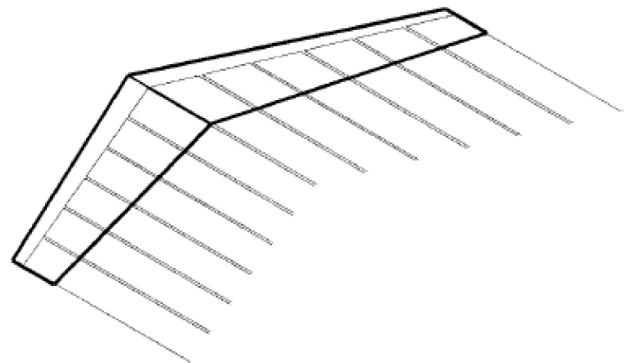


Figure 1. Horseshoe vortices distributed along the wingspan [21]

Each horseshoe vortex is made up of three straight vortex segments. To calculate the velocity induced, at an arbitrary point in space, by a vortex segment, we apply the Biot - Savart law [20]:

$$\mathbf{V} = \frac{\Gamma}{4\pi} \frac{\mathbf{r}_1 \times \mathbf{r}_2}{|\mathbf{r}_1 \times \mathbf{r}_2|^2} \mathbf{r}_0 \left(\frac{\mathbf{r}_1}{r_2} - \frac{\mathbf{r}_2}{r_1} \right) \quad (1)$$

In the above equation Γ is the vortex segment intensity, $\mathbf{r}_1$ and $\mathbf{r}_2$ are the spatial position vectors from the starting and ending points of the vortex segment to an arbitrary point in space and $\mathbf{r}_0$ is the vector along the straight vortex segment.

Phillips [21] proposes a form more suitable from a numerical point of view:

$$\mathbf{V} = \frac{\Gamma}{4\pi} \frac{(r_1 + r_2)(r_1 \times r_2)}{r_1 r_2 (r_1 r_2 + r_1 r_2)} \quad (2)$$

Figure 2 presents the typical geometry of one horseshoe vortex.

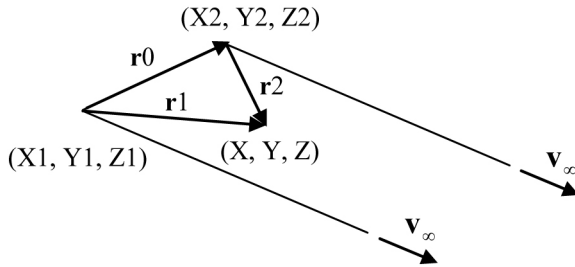


Figure 2. Geometry and position vectors for one horseshoe vortex

When we apply equation (2) for all segments of the horseshoe vortex, we can determine the velocity induced at an arbitrary point in space:

$$\begin{aligned} \mathbf{V} = & \frac{\Gamma}{4\pi} \frac{\mathbf{v}_\infty \times \mathbf{r}_2}{r_2 (r_2 - \mathbf{v}_\infty \mathbf{r}_2)} + \\ & + \frac{\Gamma}{4\pi} \frac{(r_1 + r_2)(r_1 \times r_2)}{r_1 r_2 (r_1 r_2 + r_1 r_2)} - \\ & - \frac{\Gamma}{4\pi} \frac{\mathbf{v}_\infty \times \mathbf{r}_1}{r_1 (r_1 - \mathbf{v}_\infty \mathbf{r}_1)} \end{aligned} \quad (3)$$

When considering the distribution of horseshoe vortices over the lifting surface, equation (3) can be used to compute the velocity induced at any point on the lifting surface, by any of the horseshoe vortices, provided that the intensities Γ are known.

If we approximate the continuous distribution of bound vorticity with a finite number of N distinct horseshoe vortices, each one having its own intensity Γ_i , we will need a mathematical system of N equations relating these intensities to some known properties of the wing. In order to find such a relation, we turn to the three dimensional vortex lifting law [22]. Using the same approach as that suggested by Saffman [22] or by Phillips [21], the non - viscous force acting on a horseshoe element of the lifting surface is equal to:

$$\mathbf{F}_i = \rho \Gamma_i \mathbf{V}_i \times \mathbf{I}_i \quad (4)$$

In the above equation, ρ is the air density, Γ_i is the unknown intensity of the horseshoe vortex, $\mathbf{V}_i$ is the local

airflow velocity and $\mathbf{I}_i$ is a spatial vector along the bound segment of the horseshoe vortex, aligned in the direction of the local circulation.

In order to calculate the local velocity, with each horseshoe vortex we have to associate a control point at which the induced velocities will be determined. We consider the control points to be situated on the wing surface, at equal distance between the two semi- infinite trailing legs of the horseshoe vortex, at the three quarter chord point, as measured from the local section leading edge. The importance of choosing the three quarter chord point has been revealed by various authors [13], [20], [15]. Thus, the local velocity at each control point becomes:

$$\mathbf{V}_i = \mathbf{V}_\infty + \sum_{j=1}^N \Gamma_j \mathbf{v}_{ij} \quad (5)$$

We denote by $\mathbf{V}_\infty$ the free stream velocity, while $\mathbf{v}_{ij}$ represents the velocity induced by the horseshoe vortex j , considered to be of strength equal to unity, at the control point i , and is given by equation(3).

From wing theory, we know that the magnitude of the lifting force acting on an A_i area section of the wing located at a control point i can be written as:

$$\|\mathbf{F}\|_i = \frac{1}{2} \rho V_\infty^2 A_i C_{L_i} \quad (6)$$

The lift coefficient C_{L_i} is that of the local airfoil situated at the wingspan section corresponding to control point i and it depends on the local effective angle of attack. If the lift coefficient can be determined using other means, such as experimentally determined lift curves or using a two - dimensional airfoil calculation solver, then, by replacing the local velocity given by equation(5) into equation (4), and then equating the modulus of the three - dimensional vortex lifting force with the expression give in equation (6), we obtain the following relation:

$$\left\| \rho \Gamma_i \left(\mathbf{V}_\infty + \sum_{j=1}^N \Gamma_j \mathbf{v}_{ij} \right) \times \mathbf{I}_i \right\| = \frac{1}{2} \rho V_\infty^2 A_i C_{L_i} \quad (7)$$

By writing equation (7) for all control points along the wing span, we obtain a nonlinear system of N equations for the calculation of the unknown horseshoe vortex intensities. The system can be solved in an iterative fashion, using Newton's method. However, in order to avoid the necessity of analytically calculating all the entries in the system's Jacobian matrix, we use Broyden's quasi - Newton method [23], in which the Jacobian is replaced by an approximation that is updated at each iteration.

Once the intensities of the horseshoe vortices have been determined, the total non - viscous aerodynamic force can be